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660-22

Orbiter Description Document
For
Jupiter Orbiter Probe 1981/1982 Mission

Jet Propulsion Laboratory
California Institute of Technology
Pasadena, California 91103

Ames Research Center
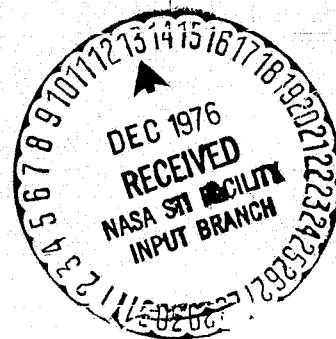Moffett Field, California 94035

September 1976

National Aeronautics and Space Administration

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FIGURES

SECTION I

INTRODUCTION

This document describes the Jupiter Orbiter as it is presently understood.

Because the Jupiter Orbiter Probe 1981/1982 (JOP 81/82) Project will be severely cost constrained, it must be understood that the Orbiter design and capabilities described herein will not be subject to major change or modification as a result of instrument selection. The JOP science payload should conform to this defined capability and the specified interfaces.

This document specifically excludes a description and discussion of any science instrument complement, except the Imaging Science Subsystem and the Radio Science capability which are provided by the Project as Orbiter facilities. These Orbiter facilities are described in Appendices A and B of this document. Appendix C describes an infared capability that may be provided by the Project. The decision on this matter has not been made.

Additional appendices are Appendix D, "The Environmental Design Requirements", and Appendix E, "The Mission Operations Description".

This document has been prepared specifically for the JOP Announcement of Opportunity (Ref. 1-1) proposal preparation package and describes the Jupiter Orbiter in sufficient detail to allow the science community to propose scientific instrumentation consistent with its capabilities and limitations.

REFERENCES

1-1. Announcement of Opportunity, AO#OSS-3-76, Outer Planets Orbiter Probe (Jupiter), National Aeronautics and Space Administration, Washington, D. C. , 1976.

SECTION II

SYSTEM REQUIREMENTS

The JOP 81/82 dual-spin Orbiter design is intended to meet the following requirements:

A. SCIENCE

1. Probe Delivery and Probe Data Relay Link

The Orbiter will deliver an instrumented Probe to the vicinity of Jupiter, separate the Probe at a spin rate of (TBD)[1] rev/min for entry into the jovian atmosphere, receive Probe data transmitted for a period of 30 to 40 min at a rate of 88 sym/s, simultaneously store the data on board, and then retransmit it to Earth.

2. Satellite Remote Sensing

The Orbiter will acquire imaging data in the visible as well as in the infrared portions of the spectrum from Jupiter and two Galilean satellites, preferably a rocky and an icy satellite, and transmit these data to Earth.

3. Fields and Particles

The Orbiter will acquire scientific data relating to the topology and behavior of the magnetic field and the energetic particle fluxes of the jovian system while orbiting around Jupiter and will transmit these data to Earth.

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[1] To be determined.

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B. PROBE REQUIREMENTS

1. Physical

For the baseline Probe configuration, a conical cavity with a maximum diameter of 96.5 cm (38 in.) and a maximum height of 68.5 cm (27 in.) will be provided.

The total mass of the Probe in its separated flight configuration is 150 kg (330 lb).

The Orbiter will impart a spin rate of (TBD) rpm to the Probe for stabilizing the Probe during entry into the jovian atmosphere. In the physical separation of the Probe from the Orbiter, the following constraints must not be exceeded:

    (a) Tipoff angle, less than 2.0 deg.
    (b) Tipoff rate, less than 1.6 deg/s.
    (c) Separation velocity, 0.1 ±0.01 m/s.

2. Electrical

The Orbiter will have an umbilical connector for routing signals between the Orbiter and Probe prior to Probe separation. At Probe separation, the umbilical cable will disconnect on the Orbiter side of the interface.

One component of the umbilical cable is an RF coaxial cable which carries the RF signal of the Probe transmitter for test purposes, bypassing the Probe power amplifier and the Probe and Orbiter antennas of the normal Probe/Orbiter RF link.

The Orbiter will supply a maximum of 30 W of electrical power to the Probe for Probe checkout. For normal cruise operations, the Orbiter

2-2

will supply 5 W of power. In addition, the Orbiter may be required to supply power to charge the Probe battery at a suitable voltage and current

3. Data

a. Preseparation. The Orbiter will provide telemetry words within the Orbiter engineering telemetry formats for Probe status information. The number of words or bits allocated for this purpose is (TBD).

b. Separation. The Orbiter will provide a relay telemetry system for receiving data from the Probe during the Probe entry phase. The relay radio must be capable of operating over a maximum Orbiter-to-Probe communication distance of 6 R_J, at a 88 sym/s information rate for a period of time from entry to about 40 min past entry.

Probe data may be routed to the Orbiter by the umbilical lines for on-board action by the Orbiter; e. g. , for the Probe battery voltage monitoring.

C. MISSION REQUIREMENTS

1. Launch

A single launch is planned for the end of the 1981 and the beginning of the 1982 opportunity, with the 1983 launch opportunity as backup. The Shuttle/ Interim Upper Stage (IUS) will be used as the launch vehicle. The injection energy requirement is $C_3 = 80$ km^2/s^2.

2. Mission Duration

The maximum overall mission duration will consist of a 3-year cruise-to-Jupiter period followed by a maximum orbital period of 2 years including any extended mission.

3. Major Mission Events and Velocity Increments

The following are the major mission events and associated velocity increments:

(a) Lift-off.

(b) Separation of cargo (spacecraft and IUS) from Shuttle Orbiter.

(c) IUS burn (first through third stages).

(d) IUS fourth stage burn (spin stabilized).

(e) Spacecraft separation (provided by IUS fourth stage).

(f) Sun and Canopus acquisition.

(g) Earth-Jupiter cruise trajectory corrections, 25 m/s.

(h) Probe separation [Jupiter Orbit Insertion (JOI) -57 d].

(i) Orbiter deflection maneuver (JOI-55 d), 200 m/s.

(j) Pre-entry trajectory correction, 15 m/s.

(k) Probe entry and Probe data acqu ition.

(l) JOI, 1010 m/s.

(m) Perijove raise/plane change maneuver, 500 m/s.

(n) Satellite encounters over a number of orbits; navigation velocity increments total, 250 m/s.

The total additive velocity increment is 2000 m/s.

4. Closest Approach

The initial Orbiter closest approach to Jupiter will be 6 R_J at perijove. Satellite closest approaches will be ≥1000 km.

D. RELIABILITY REQUIREMENTS

1. Primary Mission Objective

No single failure will prevent Probe delivery and the relaying of Probe data to Earth.

2. Secondary Mission Objective

No single failure will cause the permanent loss of science data from more than one scientific experiment.

E. ENVIRONMENTAL REQUIREMENTS

The Orbiter will be compatible with the natural and induced environments encountered during the pre-launch, launch, cruise, and orbit mission phases. These requirements are described in Appendix D.

F. DESIGN REQUIREMENTS

1. Mass

Spacecraft mass allocations are based on preliminary Shuttle/IUS performance. For a 10-day launch period and a $C_3 = 80$ km^2/s^2 requirement, a total injected mass of 1500 kg is available for the Orbiter, Probe, and adapters.

2. Standard Elements

NASA standard hardware will be used in all Orbiter subsystems unless substantial cost, mass, or performance penalties are incurred by such use.

3. Available Hardware

When NASA standard hardware can not be provided, existing or off-the-shelf will be used to the greatest extent possible.

4. Telecommunications

The Orbiter will use an X-band carrier as the science-data telemetry downlink and will be capable of transmitting data at information data rates up to 128 kb/s over a range up to 6 AU. Provisions for ranging will be included. The Orbiter will also transmit engineering data continuously, using an S-band carrier. The use of the 26-m Deep Space Network (DSN) will be maximized during all mission phases. The number of different telemetry formats and data rates will be minimized.

5. Power Source

The Orbiter power will be obtained from radioisotope thermoelectric generators (RTGs).

6. Instrument Pointing

The Orbiter will be designed so that it can point remote-sensing instruments at the specified target within 0.1 deg, with a pointing knowledge of 0.05 deg.

G. SPACE TRANSPORTATION SYSTEM REQUIREMENTS

1. Launch Abort

The Orbiter will meet Space Transportation System (STS) requirements applicable for the case of an abort at launch and return to Earth including crew safety requirements in the case of a crash landing.

2. Safety

The Orbiter design will meet the applicable STS safety requirements. Further, the Orbiter will be designed in such a manner that caution, warning signals, and safety commands will not be required.

SECTION III

SPACECRAFT SYSTEM DESCRIPTION

A. SYSTEM SUMMARY

The baseline dual-spin Orbiter consists of a spinning section and a despun section connected to each other by means of a despin bearing assembly. Each section contains electronics compartments. The spinning section supports the Probe, the deployable appendages, and the propulsion module. Science instruments are mounted on the scan platform and on the despun section, or they are body-fixed to the spinning section.

The Orbiter comprises the following subsystems: Structure (STRU), Radio Frequency (RFS), Telemetry Modulation (TMS), Power (PWR), Data Handling and Control (DHC), Attitude and Articulation Control (AACS), Pyrotechnics (PYRO), Cabling (CABL), Propulsion (PROP), Temperature Control (TEMP), Mechanical Devices (DEV), Data Storage (DSS), S/X-band Antenna (SXA), Relay Radio Telemetry (RRT), and the individual science subsystems. The signal flow between subsystems is illustrated in the simplified functional block diagrams, Figs. 3-1(a). and (b). Subsystem functions and design are described in Section IV of this document.

B. CONFIGURATION

1. Major Elements

Figure 3-2 depicts the launch configuration, and Fig. 3-3, the flight configuration (prior to Probe release) of the dual-spin, Probe-carrying, Jupiter Orbiter. This configuration is not necessarily an optimized design but is intended to serve as a basis for comparison of alternative designs. The major elements of the design which determine the configuration are:
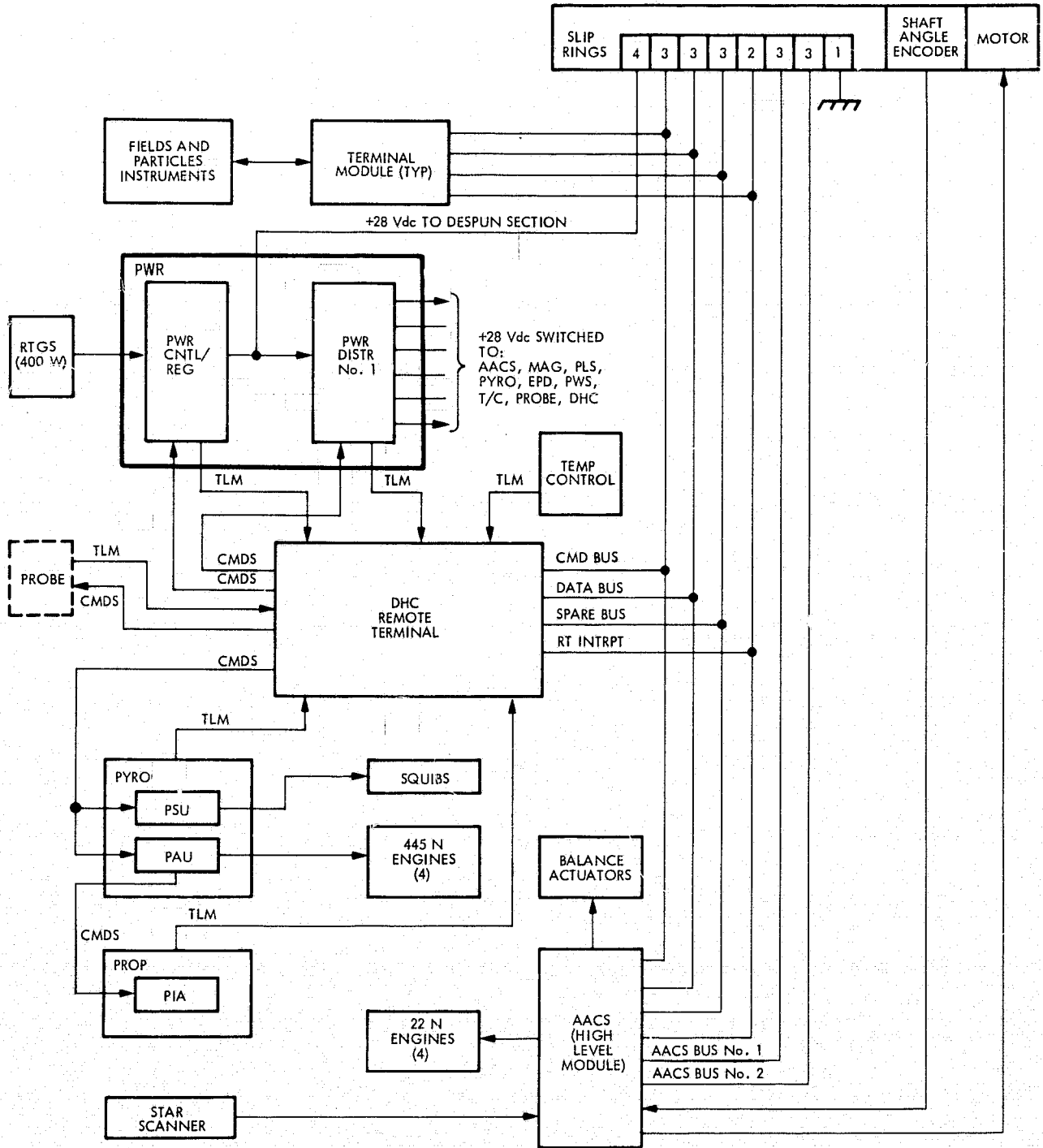
Fig. 3-1(a). JOP spin section
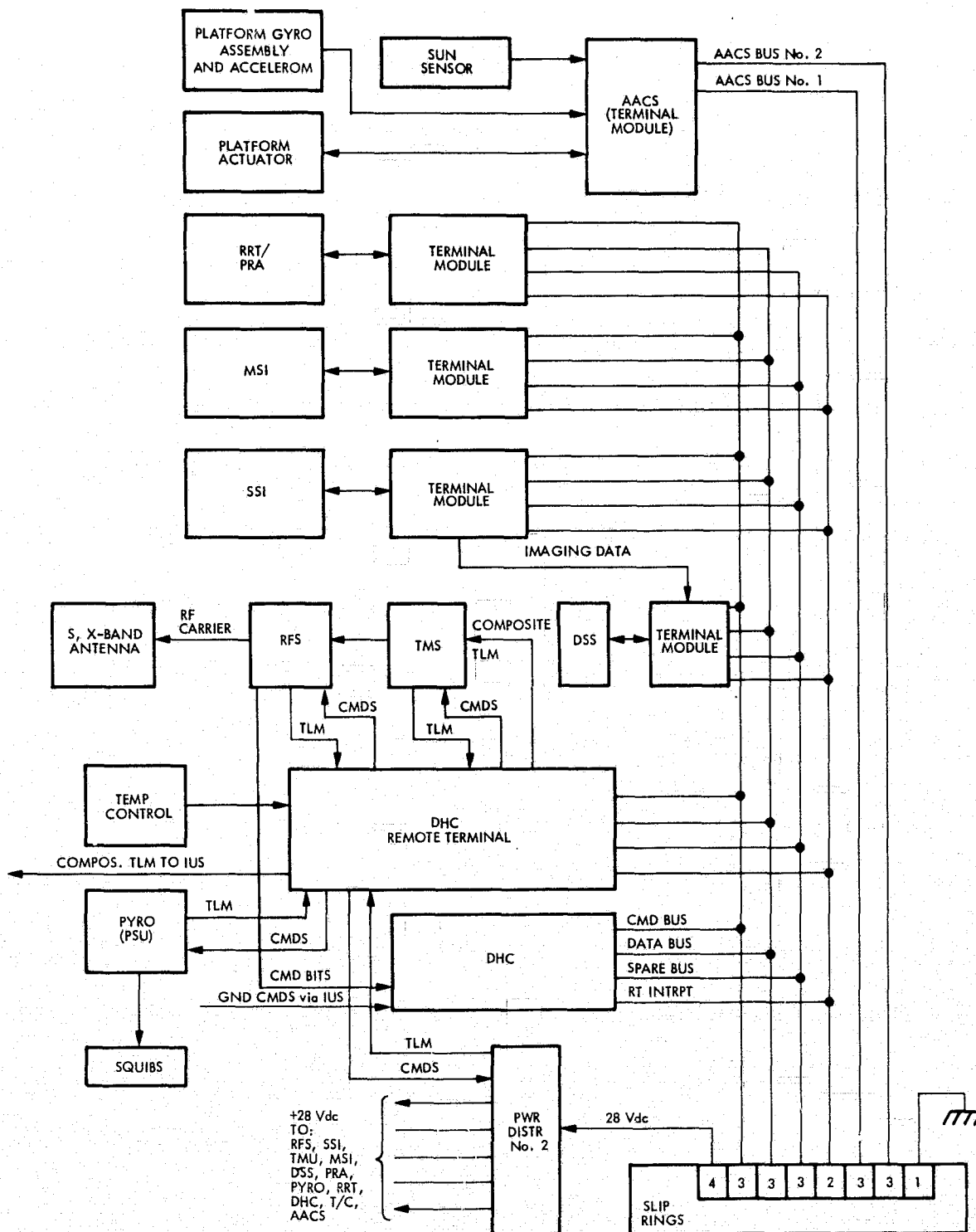
Fig. 3-1(b). JOP despun section
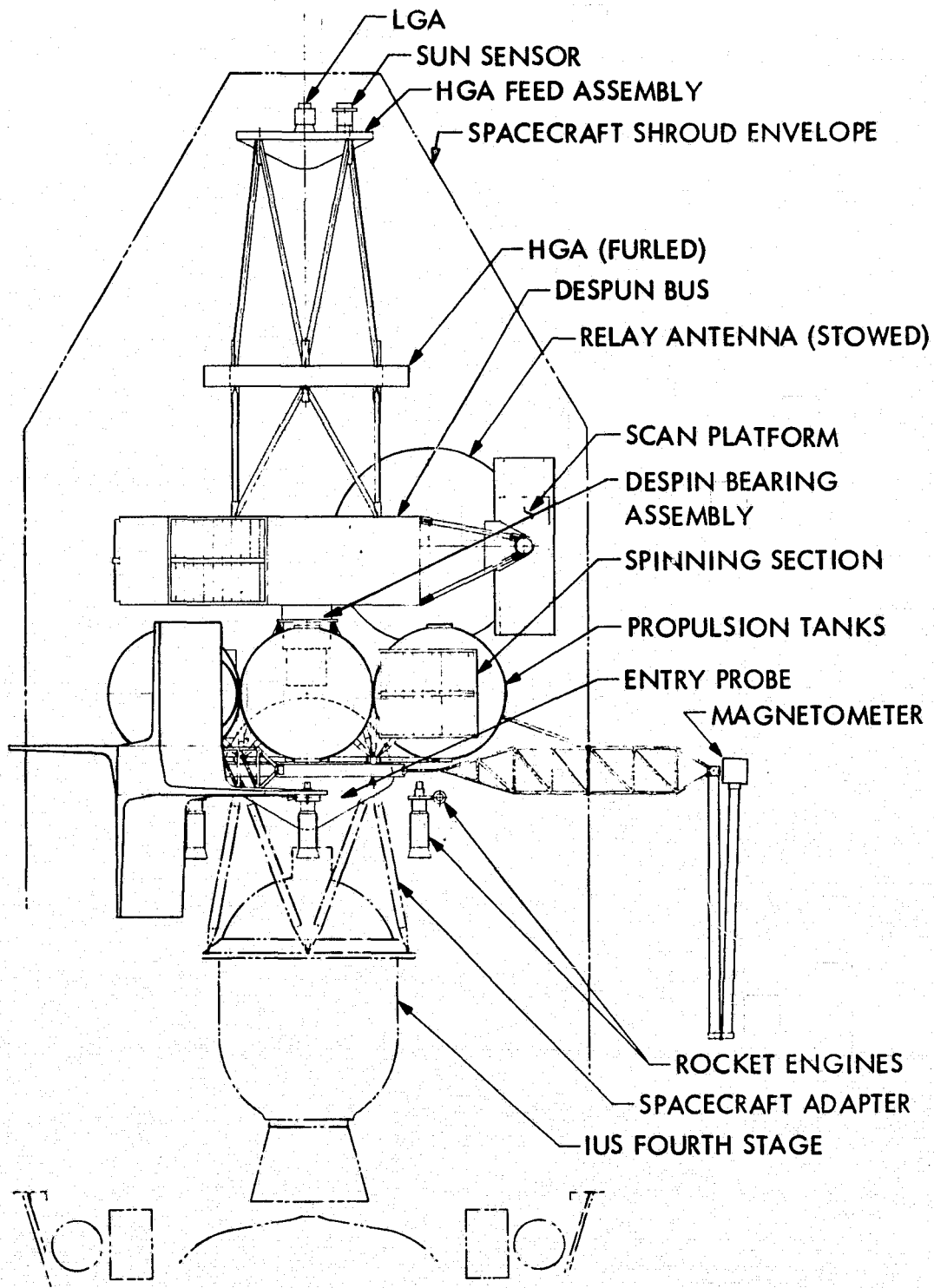
Fig. 3-2. Launch configuration of JOP spacecraft

HIGH-GAIN ANTENNA

PROBE RELAY
ANTENNA

MAGNE-
TOMETER

CCD IMAGER

INFRARED
SPECTROMETER

DESPUN SECTION

PROPULSION

DESPIN BEARING

SELENIDE RTGs
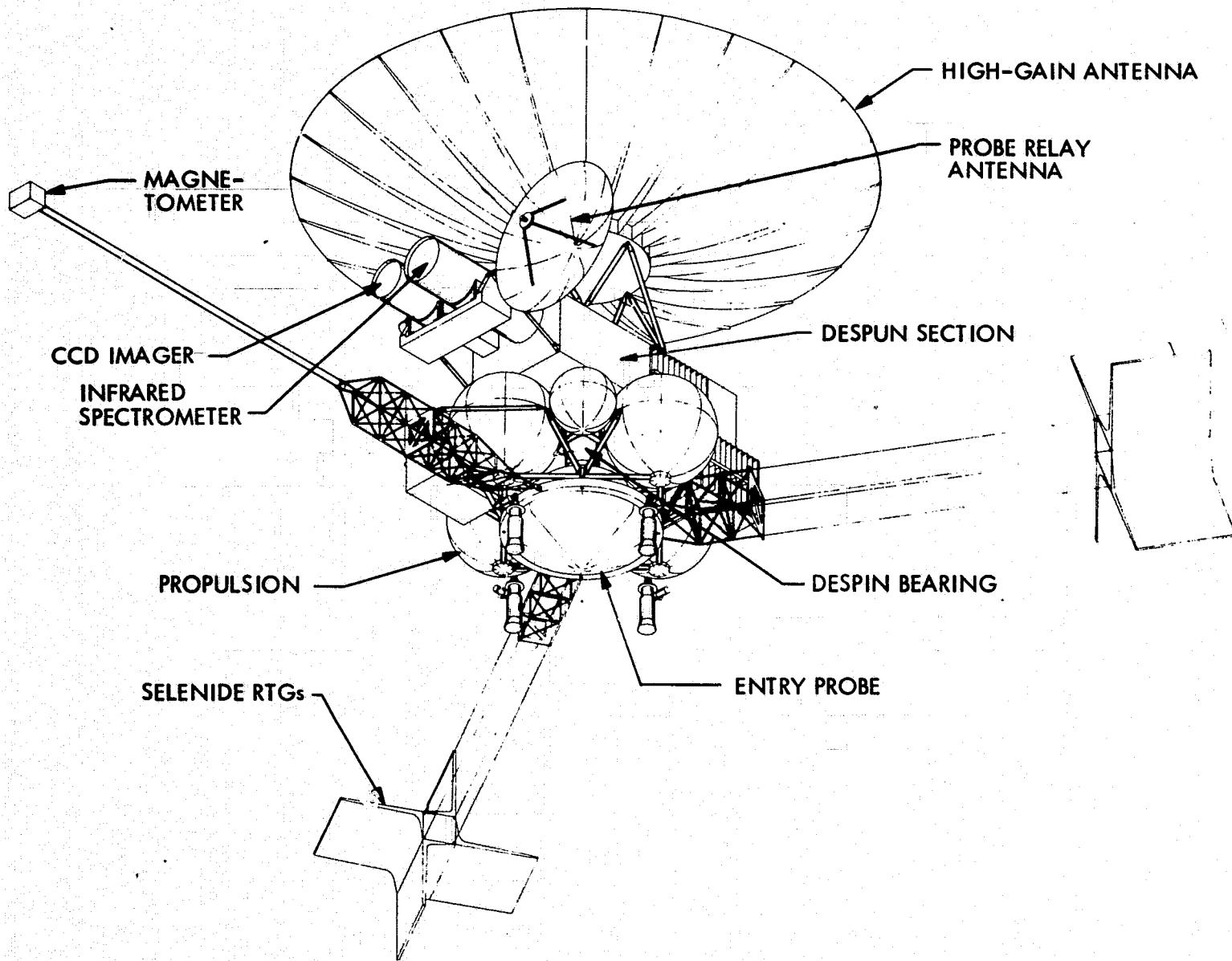
ENTRY PROBE

Fig. 3-3. Flight configuration of JOP 81/82 spacecraft

(a) The Orbiter science payload, consisting of:

    (1) Remote sensing instruments.

    (2) Fields and particle instruments.

(b) The entry Probe.

(c) One-meter diameter Probe relay antenna, steerable because it is mounted on the scan platform.

(d) Deployable, one-degree-of-freedom scan platform.

(e) Deployable selenide RTG power sources.

(f) Electronic equipment compartments for housing Mariner Jupiter/ Saturn 1977 (MJS'77) or NASA standard electronics assemblies.

(g) Four-stage IUS.

(h) Bipropellant propulsion module used for maneuver velocity increments, for attitude control, and for precession control.

(i) Five-meter furlable, parabolic high-gain antenna (HGA).

(j) Mass property distribution compatible with the dual-spin Orbiter attitude-stability design requirements.

The most difficult of the configuration design problems is the integration of the above Orbiter elements so that the center of mass does not shift in an adverse manner during staging, propulsion burns, Probe separation, or appendage deployment. Satisfaction of this requirement tends to increase stack height as elements become strung along the thrust axis (Z axis). Spin stability requirements call for a short stack-height.

2. Science

Those science instruments which require a fixed stabilized pointing direction, such as imaging and the Probe relay antenna, are mounted on a one-degree-of-freedom scan platform on the despun section of the Orbiter. Selective orientation of the despun section relative to the Orbiter roll axis (obtainable by angular-speed control in the despin bearing assembly) provides the

capability of pointing the platform at any desired clock orientation, thus providing two degrees-of-freedom for platform-mounted instruments. The platform gimbal axis positions the instruments (or antenna) at cone angles ranging from 45 to 95 deg. The despin bearing assembly provides unobscured viewing for the instruments for all clock angles. The somewhat larger relay antenna has an unobstructed region ranging from 55 to 85 deg cone at all clock angles. Cone angles outside the values mentioned above suffer obscuration by the HGA or the RTGs.

Fields and particles instruments which have broader viewing requirements will be mounted on the spinning section of the Orbiter and may be body-fixed or boom-mounted, as required.

3. Appendages and other Deployable Elements

All Orbiter appendages (RTG and instrument booms), the HGA dish, the scan platform, the Probe relay antenna, and the despun section of the Orbiter will be stowed and latched during launch. Following injection and separation from the spent fourth stage of the IUS, these items will be sequentially unlatched and deployed. The RTGs and instrument booms will be deployed simultaneously at a controlled rate prior to unlatching and despin of the upper portion of the Orbiter. Once the despun section is stabilized, the HGA, relay antenna, and scan platform may be unlatched and deployed.

4. Probe

The Probe is attached to the Probe adapter by means of three ball-lock pin devices spaced 120 deg apart, with the Probe center-of-mass on the Orbiter Z axis. The attached fittings position the Probe accurately on the Probe adapter, carry all launch loads of the Probe, control the conductive heat flow between Probe and Probe adapter during the interplanetary cruise phase, and effect the separation of the Probe from the Probe adapter. The Probe adapter, in turn, is fastened to the Orbiter.

Probe separation from the Probe adapter is accomplished by simultaneous release of the ball-lock bolt at each of the three Probe support points. The release of the ball-lock bolts is initiated by an Orbiter command which fires six pyrotechnic cartridges, two for each bolt. After the Probe is released, the Orbiter will experience a center-of-mass shift along the Z axis. This Z-axis CG migration results in no change to the thrust-vector control moment arm length.

5. Propulsion Module

Several elements combine to provide the Orbiter with the required attitude control and velocity increments during the various mission phases:

(a) Four large 455-N (100-lbf) bipropellant engines provide trajectory-correction and orbit-insertion maneuver velocity increments.

(b) Four small 22-N (5-lbf) bipropellant thrusters provide roll and precession control during the mission.

(c) Four propellant tanks and two pressurant tanks are required. Two monomethylhydrazine (MMH) fuel tanks and the two oxidizer (N_2O_4) tanks are located to facilitate a mass balance about the roll axis; the pressurant (helium) tanks are also located symmetrically about the roll axis.

6. Celestial-Reference Sensors

Redundant sun sensors and a star scanner provide the attitude-control references necessary for orientation of the spin axis and monitoring of the spin rate. The sun sensor is mounted on the HGA feed with its boresight coincident with that of the HGA and along the roll axis. The sensor will be electronically biased to keep the roll axis pointed at Earth, e.g., during high data rate telemetry transmission periods, or it may be deactivated during cruise for attitude-control fuel conservation. The star scanner, which is mounted on

the spinning section of the spacecraft, provides spin rate control and angular position information to the attitude-control electronics. A signal is generated each time a given star enters the sensor's field of view.

7. Antennas

    a. High Gain Antenna (HGA). Communication with Earth is through a 5-m deployable parabolic dish whose boresight is along the roll axis. For launch, the antenna dish is stowed in a central hub approximately 1 m in diameter. Following Orbiter/IUS separation, dish deployment is initiated pyrotechnically.

    b. Low Gain Antenna (LGA). The low-gain antenna is located on top of the HGA feed assembly near the Orbiter centerline.

    c. Relay Antenna. The Probe relay antenna is a parabolic reflector 1-m in diameter which is mounted on the scan platform. The antenna can be pointed to track the Probe from Probe release through the entry phase of the mission. During launch, the antenna is stowed against the side of the platform.

8. Spacecraft Bays

    Spacecraft electronics are housed in five standard Mariner-size bays. Two bays, mounted individually on the spinning section of the Orbiter, constitute one compartment; and three bays, on the despun portion, constitute another. The bays utilize the MJS'77 dual-shear-plate design which is compatible with MJS'77, Viking Orbiter, and NASA-standard subsystems. The compartment design is flexible in that the volume available may be tailored to accommodate additional electronics.

C. LAUNCH VEHICLE INTERFACES

1. Mechanical Interfaces

The launch vehicle adapter serves to support the Orbiter from the upper stage of the IUS. The adapter consists of four tabular bipods which attach to four points on the IUS fourth stage interface ring and to four points on the Orbiter. Each bipod is equipped with an explosive bolt and a pushoff spring for release and separation of the Orbiter from the IUS following the IUS fourth-stage burn.

2. Electrical Interfaces

All electrical interfaces between the Orbiter and the Space Transportation System (STS) will be between the Orbiter and the IUS. Data and command interfaces will be between the Orbiter and the third stage of the IUS. Interfaces related to Orbiter separation will be with the fourth stage of the IUS.

a. **Telemetry Data.** Orbiter engineering telemetry data, at a rate of 2 kb/s and in a fixed (non-programmable) format, will be carried from the Orbiter DHC via hardline to the IUS telemetry system, where the data are time-multiplexed with IUS telemetry data. The data will be used for general Orbiter-performance monitoring by the ground crew as well as for indicating results of on-orbit checkout.

b. **Commands.** Any ground commands required for on-orbit checkout, and for restoring Orbiter subsystem states to their launch mode after such a checkout, will be sent from the ground to the STS Orbiter where they are interfaced with the IUS command decoder via hardline. The IUS will, in turn, strip out any Orbiter-addressed commands and forward those, via hardline, to the Orbiter DHC in the form of command data.

c. **Orbiter Separation.** Energy to activate the Orbiter separation devices at the appropriate time will be provided by the IUS fourth stage.

This stage will also provide a signal to the Orbiter, at a known time prior to separation, which will be used to initiate Orbiter subsystem state changes such as arming pyrotechnics.

3. RTG Cooling

It is assumed that, to maintain satisfactory RTG temperatures during all phases of the mission up to payload deployment, active RTG cooling is required. In addition, if the RTG is to survive a post-launch abort in reusable condition, active cooling will be mandatory.

The current Orbiter configuration employs two sets of RTGs located about 120 deg apart on the spinning section of the Orbiter. A coolant passage at the root of each cooling fin is assumed, with manifolding at the in-board end of the RTG assembly and coolant supply and return fittings at the out-board end. This concept was chosen to facilitate the connection and disconnection of the RTG cooling system.

Figure 3-4 shows a schematic of the representative cooling system. For inflight operation, coolant is circulated through the RTGs and then through an evaporative heat exchanger which accepts heat from the coolant loop and rejects it to space by means of boiling water at regulated pressure. For ground operations (on-pad before launch or post-landing in case of abort), chilled water from a ground supply passes through the RTGs and goes overboard. Electrically activated valves control the flow path between ground or on-board cooling.

A heat exchanger coupled to the Shuttle Orbiter Active Thermal Control System may be used to reject heat through the Shuttle radiators during on-orbit operations to reduce coolant requirements or to extend the time in orbit.

Each RTG set is expected to deliver 200 W (electrical) at 10% efficiency. This results in a maximum heat removal load of 2000 W (thermal) per RTG set or 4000 W (thermal) total. The representative cooling system
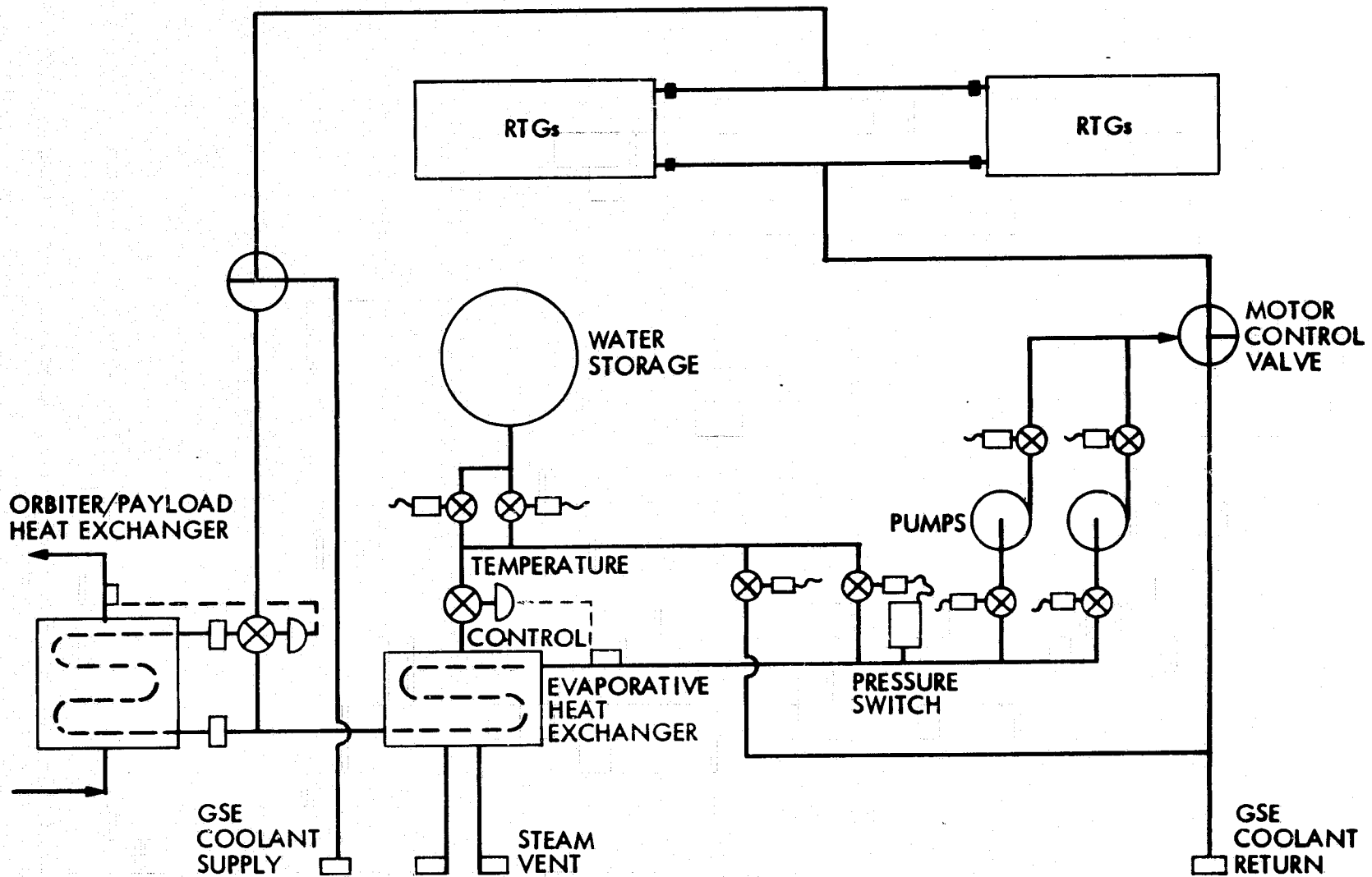
Fig. 3-4. RTG thermal control system

has adequate capacity relative to the above requirement to account for growth, losses in efficiency, and heat inputs from the hot Shuttle structure during entry and soak-back.

It should be noted that the cooling system is returned with the Shuttle Orbiter and can be reused for future missions with minor refurbishment.

4. Spacecraft Shroud

The need to enshroud the spacecraft is a result of concern about contamination. Although the Shuttle cargo bay will be cleaned prior to cargo installation, the possibility of particulate contamination will still exist. In addition, after cargo (spacecraft plus IUS) deployment, the Shuttle fires thrusters to move away to a safe distance. The Earth-storable propellants used by these thrusters produce some residual partially reacted material which may condense on spacecraft optics or other sensitive cold areas. These considerations make some type of protection necessary.

Use of a shroud represents an adequately conservative approach and is chosen for the present baseline; however, this decision could be subject to review as the Shuttle environment becomes better defined.

The shroud design used is a right circular cylinder with a truncated conical top. The shroud is cantilevered from the IUS and has no structural interface with the spacecraft or the Shuttle. It is fabricated from aluminum honeycomb material. It separates longitudinally into three parts with two split lines aligned with the RTGs and the third equally spaced between them and aligned with the magnetometer boom. Since the RTGs must be outside the shroud for thermal-control reasons, cutouts are provided in the structure at these points. Penetration is also required in the third split line for the magnetometer boom. These openings are closed by flexible rip panels of fiberglass or other type of fabric to allow the shroud segments to separate. Springs at the top and base of each shroud segment provide the separating force.

Because the shroud is not subject to aerodynamic forces or heating, elaborate latching systems or Super Zip joints are not required. Nylon bands at the top and bottom of the cylindrical section hold the segments together. These bands are released by pyrotechnic bolt cutters to allow separation. The structural attachment at the base involves a hinge arrangement that allows the hinge joint to separate after rotating some distance (perhaps 30 deg) to ensure clearing the spacecraft. The bottom of the shroud is closed by a flexible fiberglass diaphragm which remains with the IUS.

The shroud as described above is suitable for protection during launch and orbital flight. In the event of a post-launch abort, some additional capability would be required, if a controlled environment is desired. During entry, the Shuttle bay repressurizes with the ambient air which is filtered but not controlled for temperature, humidity, or condensible material content. To protect the spacecraft from this environment, it will be necessary to purge the shroud with dry nitrogen during entry. The nitrogen flow would be regulated to maintain 1 to 2 in. H_2O pressure above ambient in the shroud. To protect the spacecraft from excessive heating during entry and after landing, but prior to connecting ground cooling, the shroud will probably have to be insulated. Insulation would increase the shroud mass, as would the purge system. The insulation and purge options have not been designed but are considered as options to be investigated.

D. MASS AND INERTIAL PROPERTIES

The baseline spacecraft mass is summarized in Table 3-1. A subsystem mass breakdown is shown in Table 3-2. The Orbiter mass does not include any special allocations for radiation shielding. Figure 3-5 is a schematic for Tables 3-3(a) and (b), which list the Orbiter inertial properties.

Table 3-1. JOP 81/82 spacecraft mass summary

Subsystem	Mass (kg)		
	Total	Spin	Despun
01 STRU	93.2	29.5	63.7
02 RFS	26.8	-	26.8
03 TMS	2.7	-	2.7
04 PWR	78.9	75.1	2.8
06 DHC	15.0	5.5	9.5
07 AACS	46.0	33.6	12.4
08 PYRO	5.0	3.0	2.0
09 CABL	26.3	10.5	15.8
10 PROP (see below)	-	-	-
11 TEMP	17.9	13.8	4.1
12 DEV	5.5	5.2	0.3
16 DSS	6.3	-	6.3
17 SXA	4.8	-	4.8
52 RRT	10.3	-	10.3
Subtotal	338.7	177.2	161.5
Subtotal (science)	60.2	29.2	31.0
Total (Orbiter with science)	398.9	206.4	192.5
10 PROP	741.0	741.0	-
Total (Orbiter with science and Propulsion module)	1139.9	947.4	192.5
Probe (total)	150.0	150.0	-
Total (injected mass)	1289.9	1097.4	192.5

Table 3-2. JOP spacecraft, subsystem, and equipment characteristics

Subsystem/assembly	Mass, kg	Power, W	Orbiter sections		Design base remarks
			Spin	Despun	
01 Structure					
Spun compartment assembly (including shear plates)	9.4		X		2 bays
Despun compartment assembly (including shear plates)	20.1			X	3 bays
HGA (dish, parabolic)	28.6			X	ATS (5-m furlable)
LGA	0.4			X	Mariner
Antenna support structure	1.7			X	
Feed support structure	5.5			X	
RTG booms (2)	3.7		X		3.35 m long
Propulsion support structure	9.1		X		
Platform support structure	1.9			X	
Scan-platform assembly	3.3			X	
Probe support	0.5		X		
Miscellaneous mounting brackets	4.0		X	X	
MAG boom, fixed portion	3.9		X		1.6 m long
Mag boom, deployable portion	1.1		X		2.8 m long
Total	93.2		(29.5)	(63.7)	
02 Radio frequency					
S/X band transponder A	3.6	10.5		X	NASA standard
S/X band transponder B	3.6	10.5		X	NASA standard
Control unit/X-band hybrid	2.5	0.5		X	MJS
S-band switch/hybrid	2.0	0		X	MJS
Output filters (2)	0.2	0		X	MJS
S-band power amplifier A	1.35	25.0		X	NASA standard, 5 W
S-band power amplifier B	1.35	25.0		X	NASA standard, 5 W
Dual X-band power amplifier	10.9	67.0		X	MJS
Ultra-stable oscillator	1.35	2.5		X	MJS
Total	26.85			(26.85)	

Table 3-2 (contd)

Subsystem/assembly	Mass, kg	Power, W	Orbiter sections		Design base remarks
			Spin	Despun	
03 Telemetry modulation					
Telemetry modulation unit A	1.35	3		X	MJS
Telemetry modulation unit B	1.35	3		X	MJS
Total	2.7			(2.7)	
04 Power					
RTGs	60.6		X		Selenide, 400 W
DC (shunt) regulator	3.6	9.0	X		MJS, modified
Discharge controller	4.7		X		MJS, modified
Power control	4.5		X		MJS, modified
Power distribution 1	2.75		X		MJS, modified
Power distribution 2	2.75			X	MJS, modified
Total	78.90		(76.15)	(2.75)	
06 Data handling and control					New
Power converter A	1.5	2		X	
Power converter B	1.5	2		X	
Control processor	0.5	1		X	
Data handler	0.5	1		X	
Spare high level module (HLM)	0.5	1		X	
Test HLM	0.5	0		X	SE-powered only
Central timers 1 and 2	0.5	1		X	
Discrete interface unit (remote terminal)	2.0	4		X	
DSS terminal module	0.5	1		X	
SSI terminal module	0.5	1		X	
MSI terminal module	0.5	1		X	
RRT terminal module	0.5	1		X	
Power converter C	1.25	1.5	X		
Power converter D	1.25	1.5	X		
Discrete interface unit (remote terminal)	1.0	2	X		

Table 3-2 (contd)

Subsystem/assembly	Mass, kg	Power, W	Orbiter sections		Design base remarks
			Spin	Despun	
06 (contd)					
F and P Terminal modules	2.0	4	X		
Total	15.0		(5.5)	(9.5)	
07 Attitude and articulation control					
Sun-sensor assembly	1.8	0.6		X	MJS type
Star-scanner assembly	2.7	2.0	X		Pioneer-Venus
Despin bearing assembly	16.8	7.0	X		OSO-8 design base
Gyro assembly (includes integrating ΔV accelerometer	7.0	14.0		X	MJS, modified
Controller (high-level modules)	9.1	6.5	X		MJS HYPACE design base
Scan-platform actuator	0.9	2.5		X	MJS design base
Balance actuators	3.6	–	X		OSO-8 derivative
Nutation (hoop) damper	1.4	0	X		Passive device, GMS
Terminal modules (including actuator drive)	2.7	4.0		X	New (similar to DHC)
Total	46.0		(33.6)	(12.4)	
08 Pyrotechnics					
PSU 1/PAU	2.7	0.8[a]	X		VO'75
PSU 2	1.8	0.8[a]		X	MJS
Squibs	0.5		X	X	
Total	5.0		(3.0)	(2.0)	
09 Cabling					Mariner
Electronic assembly harnesses:					
spin section	2.8		X		
despun section	4.2			X	

[a]Steady-state power requirement.

Table 3-2 (contd)

Subsystem/assembly	Mass, kg	Power, W	Orbiter sections		Design base remarks
			Spin	Despun	
09 (contd)					
System harnesses:					
spin section	7.7		X		
despun section	11.6			X	
Total	26.3		(10.5)	(15.8)	
10 Propulsion (see end of table)					
11 Temperature control					
Despun compartment blanket	1.0			X	Mariner
Spinning compartment blanket	0.4		X		Mariner
Louvers	2.4		X	X	Mariner
Science blankets	1.0		X	X	MJS
RHUs	0.9		X	X	MJS
Micrometeorite shields	0.6		X	X	MJS
Propulsion module blanket	8.3		X		VO'75
Propulsion blanket support	1.4		X		VO'75
Propulsion module thermal paint	1.9		X		VO'75
Total	17.9		(13.8)	(4.1)	
12 Mechanical devices					
Scan-platform deployment mechanism	0.3			X	Mariner
RTG-deployment mechanisms	2.5		X		MJS
MAG boom-pin pullers	0.2		X		MJS
MAG boom-deployment mechanism	0.5		X		MJS
MAG sensor supports	0.6		X		MJS
RTG pin pullers	0.5		X		MJS
Orbiter IUS separation devices	0.9		X		MJS
Total	5.5		(5.2)	(0.3)	

Table 3-2 (contd)

Subsystem/assembly	Mass, kg	Power, W	Orbiter sections Spin	Orbiter sections Despun	Design base remarks
16 Data storage					
Tape recorder	6.3	2.5 to 14		X	NASA standard
Total	6.3			(6.3)	
17 S/X-band antenna					
HGA feed, X-band	0.8			X	New
HGA feed, S-band	0.4			X	New
HGA and LGA S-band coaxial cable	1.5			X	MJS
HGA waveguide assemblies	1.3			X	MJS
HGA subreflector, probes	0.8			X	MJS
Total	4.8			(4.8)	
52 Relay radio telemetry					
Relay receiver	3.5	3.0		X	New
Relay telemetry unit	1.4	3.0		X	New
Antenna	5.4	—		X	New
Total	10.3			(10.3)	
36 Solid state imaging					New
Camera and electronics Total	18.0	10.5		(18.0)	
39 Multispectral imaging					New
Near-IR multispectral imaging instrument Total	7.0	4.5		(7.0)	
10 Propulsion					
Bipropellant tank assemblies (4)	35.6		X		New
Helium pressurant tank assembly	36.1		X		VO75
445-N (100 lbf) rocket engines (4)	8.9	26 (each)	X		Marquart, Apollo
22-N (5 lbf) rocket engines (4)	5.2	35 (each)	X		Designs in work
Pressurant control assembly	12.7		X		VO'75
Propellant isolation assembly (2)	11.2		X		VO'75

Table 3-2 (contd)

Subsystem/assembly	Mass, kg	Power, W	Orbiter sections		Design base remarks
			Spin	Despun	
Temperature transducers (2)	0.1		X		
Pressure transducers (8)	1.8		X		
Tubing and fittings	14.3		X		
Subtotal (inerts)	125.9				
Propellant holdup:					
Oxidizer (N_2O_4)	4.0		X		
Fuel (MMH)	2.0		X		
Subtotal (holdup)	6.0				
Usable propellants:					
Usable oxidizer	396.8		X		(including 6.7 kg for AACS)
Statistical-variation oxidizer	7.9		X		
Usable fuel	198.4		X		(including 3.3 kg for AACS)
Statistical-variation fuel	4.0				
Subtotal (usable bipropellant)	607.1				For total ΔV = 2000 m/s
Pressurant (helium)	2.0		X		
Propulsion Total	741.0		(741)		
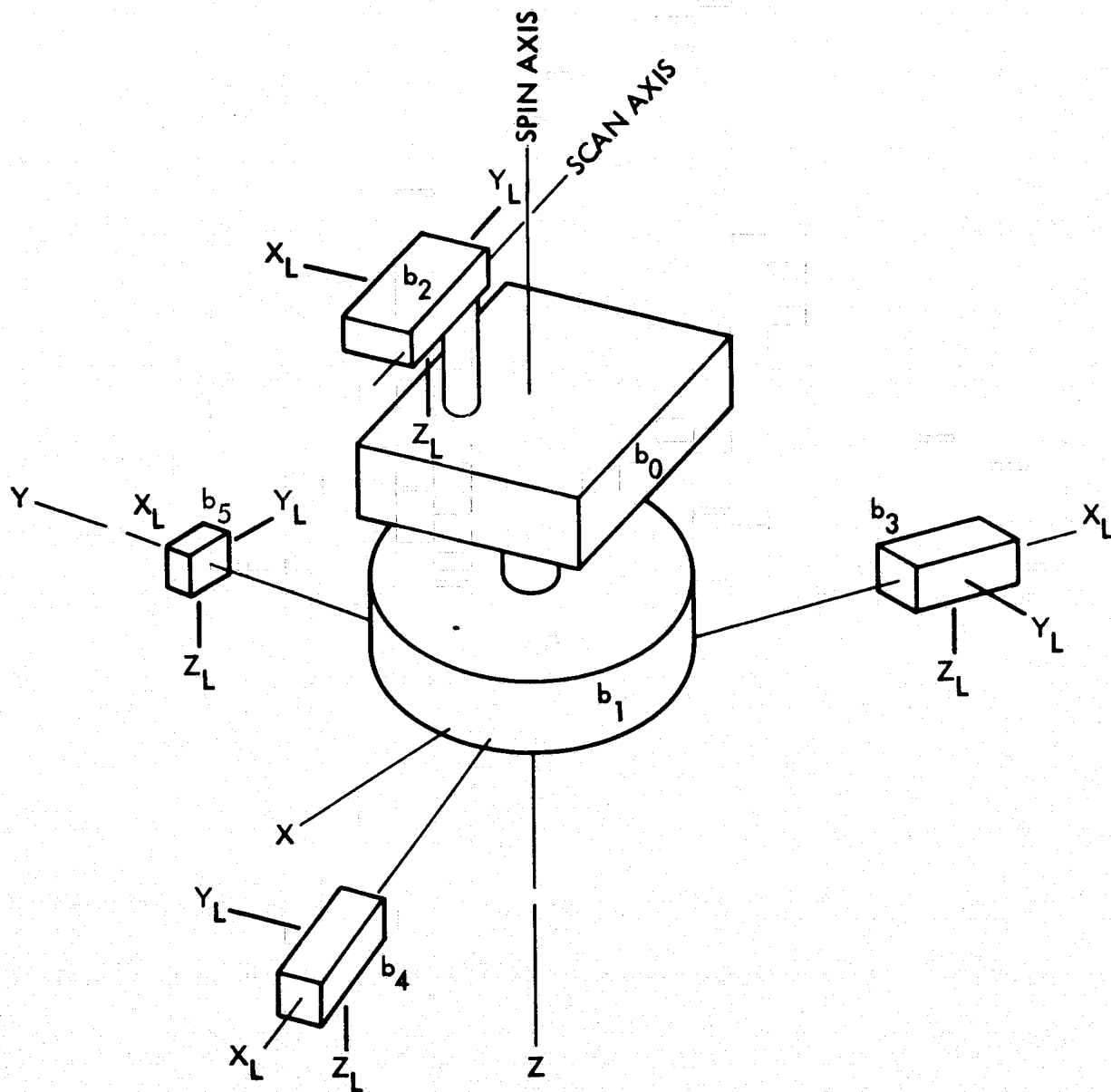
Fig. 3-5. Schematic for mass inertial properties

Item	Description	Configuration	Mass		$\overline{X}$		$\overline{Y}$		$\overline{Z}$	
			kg	(lb)	cm	(in.)	cm	(in.)	cm	(in.)
b_0	Despun section	Stowed	192	(423)	3.1	(1.24)	-9.04	(-3.56)	-135.9	(-53.5)
		Deployed	142	(423)	4.2	(1.65)	-7.75	(-3.05)	-141.2	(-55.6)
b_1	Spinning section	Launch (stowed)	1098	(2420)	-0.66	(-0.26)	0.23	(0.11)	-26.7	(-10.5)
		Launch (deployed)	1098	(2420)	-0.66	(-0.26)	-4.4	(-1.74)	-26.7	(-10.5)
		EOM (no Probe)	488	(1077)	-1.47	(-0.58)	-9.93	(-3.91)	-9.60	(-3.7)
$b_0 + b_1$	Total Orbiter	Launch (stowed)	1290	(2843)	-0.09	(-0.035)	-1.11	(-0.438)	-42.9	(-16.9)
		Launch (deployed)	1290	(2843)	0.07	(0.027)	-4.90	(-1.93)	-43.9	(-17.3)
		EOM (no Probe)	680	(1500)	0.13	(0.050)	-9.30	(-3.66)	-46.7	-18.4

Item	Description	Configuration	Mass		$\overline{X}$		$\overline{Y}$	
			kg	(lb)	cm	(in.)	cm	(in.)
b_2	Scan	Deployed	32.2	(71.1)	0	(0)	113	(44.5)
b_3	RTG and boom	Deployed	33.2	(73.1)	-302	(-119)	-175	(-69)
b_4	RTG and boom	Deployed	33.2	(73.1)	302	(119)	-175	(-69)
b_5	Magnetometer and boom	Deployed	11.6	(25.5)	0	(0)	355	(139.7)

OUT FRAME

Table 3-3. JOP spacecraft inertial properties
(a) composite (b) local

(a)

$\overline{Z}$ (in.)	$\overline{Z}$ cm	(in.)	I_{XX}		I_{YY}		I_{ZZ}		I_{XY}	I_{XZ}	I_{YZ}
						kg-m^2 (slug-ft^2)					
(-3.56)	-135.9	(-53.5)	347	(256)	262	(193)	309	(228)	209 (154)	206 (152)	197 (145)
(-3.05)	-141.2	(-55.6)	369	(272)	282	(208)	317	(234)	213 (157)	206 (152)	197 (145)
(0.11)	-26.7	(-10.5)	580	(428)	469	(346)	777	(573)	152 (112)	150 (111)	159 (117)
(-1.74)	-26.7	(-10.5)	884	(652)	994	(733)	1608	(1186)	150 (111)	150 (111)	141 (104)
(-3.91)	-9.60	(-3.78)	579	(427)	786	(580)	1234	(910)	64 (47)	66 (49)	64 (47)
(-0.438)	-42.9	(-16.9)	1124	(829)	926	(683)	1089	(803)	359 (265)	350 (258)	371 (274)
(-1.93)	-43.9	(-17.3)	1467	(1082)	1490	(1099)	1927	(1421)	363 (268)	348 (257)	344 (254)
(-3.66)	-46.7	-18.4	1186	(875)	1307	(964)	1551	(1144)	276 (204)	262 (193)	256 (189)

(b)

$\overline{X}$ (in.)	$\overline{Y}$ cm	(in.)	$\overline{Z}$ cm	(in.)	I_{XL}		I_{YL}		I_{ZL}	
					kg-m^2 (lb-in.^{2})					
(0)	113	(44.5)	-117	(-46.0)	1.61	(5500)	2.48	(8,500)	3.59	(12,300)
(-119)	-175	(-69)	0	(0)	0.52	(1800)	14.02	(48,000)	14.02	(48,000)
(119)	-175	(-69)	0	(0)	0.52	(1800)	14.02	(48,000)	14.02	(48,000)
(0)	355	(139.7)	0	(0)	0.08	(270)	32.12	(110,000)	32.12	(110,000)

E. POWER

The Orbiter power source is a 400-W selenide RTG array designed to provide 28 Vdc to all users, using the equipment described in Section III of this document. The Orbiter power requirements are shown in the power profile, Table 3-4.

F. STABILIZATION AND POINTING

The Orbiter is spin stabilized, obtaining attitude information from celestial sensors and maintaining the required attitude by means of a set of four 22-N bipropellant thrusters which provide roll and precession control. Four 445-N thrusters are used for ΔV maneuvers and for pitch- and yaw-axis thrust-vector control (TVC). The programmable guidance electronics also control the two-degree-of-freedom articulation of the scan platform in a closed-loop mode.

The equipment used for attitude and articulation control provides the following stabilization and pointing performance:

1. Orbiter Pointing Accuracy

    (a) Orbiter pointing during all cruise phases: 5 mrad (total), 3 σ.
        Orbiter pointing knowledge: 1 mrad (total), 3 σ.
    (b) Probe separation control: 4 mrad.

2. Scan Platform Accuracy

    (a) Pointing control: 2 mrad (total), 3 σ.
    (b) Pointing knowledge: 1 mrad (total), 3 σ.
    (c) Pointing stability (jitter): 0.35 mrad.
    (d) Settling time: 8 s.

Table 3-4. Power profile

Subsystem	Element/remarks	Launch (on STS)	TCM-JOI	Probe checkout	Orbital cruise	Satellite encounter
RFS	All but X-TWTA	41.5	41.5	41.5	41.5	41.5
	X-TWTA	67.0[a]	67.0	67.0	67.0	67.0
TMS		3.0	3.0	3.0	3.0	3.0
PWR	Housekeeping and regulator losses	9.0	9.0	9.0	9.0	9.0
DHC	Includes terminal modules and remote terminals	14.5	14.5	15.5	18.5	20.5
AACS	Sensors and electronics	11.1	11.1	11.1	11.1	13.1
	Scan actuator/replacement heater	--	2.5	2.5	2.5	2.5
	Platform gyro assembly	14.0	--	--	--	14.0
	Despin bearing assembly	7.0	7.0	7.0	7.0	7.0
	22-N thrusters	--	70.0	--	--	--
PYRO	Electronics	--	1.6	1.6	--	--
	445-N thrusters	--	52.0	--	--	--
TEMP	Electrical heaters	20.0	10.0	80.0	80.0	80.0
DSS		2.5	2.5	2.5	2.5	14.0
RRT		--	--	6.0	--	--
Probe		30.0[a]	--	30.0	--	--
Science	Fields and particles	30.3[a]	--	--	30.3	30.3
	Visual and IR imaging	15.0[a]	--	--	--	15.0
Subtotal		264.9[a]	291.7	276.7	272.4	316.9
Contingency		30.0	30.0	30.0	30.0	30.0
Total		294.9[a]	321.2	306.7	302.4	346.9
RTG capability		400.0	400.0	400.0	400.0	400.0

NOTES:

[a] Only during on orbit checkout; otherwise, 5 W for the Probe and 0 W for science and X-Band TWTA total 127.6 W.

G. MANEUVER CAPABILITY

Maneuver velocity increments are provided by the bipropellant propulsion system using four 445-N (100-lbf) thrusters each having a specific impulse of 280 s. Propellant allocations were established as shown in Table 3-2. The minimum ΔV achievable, after the maneuvers required for the initial orbit, is 0.1 m/s, if valve operation is assumed to be the only limiting factor; or 1.5 m/s, if a 1-s minimum time between "on" and "off" commands is assumed as an additional limiting factor.

H. DATA HANDLING

1. General

Data handling comprises those elements of the functional block diagram [Fig. 3-1(a) and (b)] identified as the DHC, DSS, TMS, RFS, and the HGA and LGA. This configuration provides a two-channel downlink. One channel is used for the continuous transmission of fixed-format and low-rate (62.5 b/s), real-time, uncoded engineering data via S-band. The other channel is used for real-time or playback data at data rates between 2 kb/s and 128 kb/s via X-band, as described below in further detail. The data flow in the Orbiter is illustrated in Fig. 3-6. The individual telecommunications and data handling and storage elements are described in Section IV of this document. The following types of data will be transmitted from the Orbiter:

(a) Engineering data consisting of digitized analog measurements as well as digital status and discrete event indicators necessary to monitor the operation of the Orbiter including science subsystems and the Probe while it is attached to the Orbiter.

(b) Probe relay data, consisting of science and engineering data transmitted to the Orbiter during Probe entry via the relay radio telemetry link. These data are transmitted at a rate of 88 sym/s (44 b/s, convolutionally coded) for a total period of about 40 min.
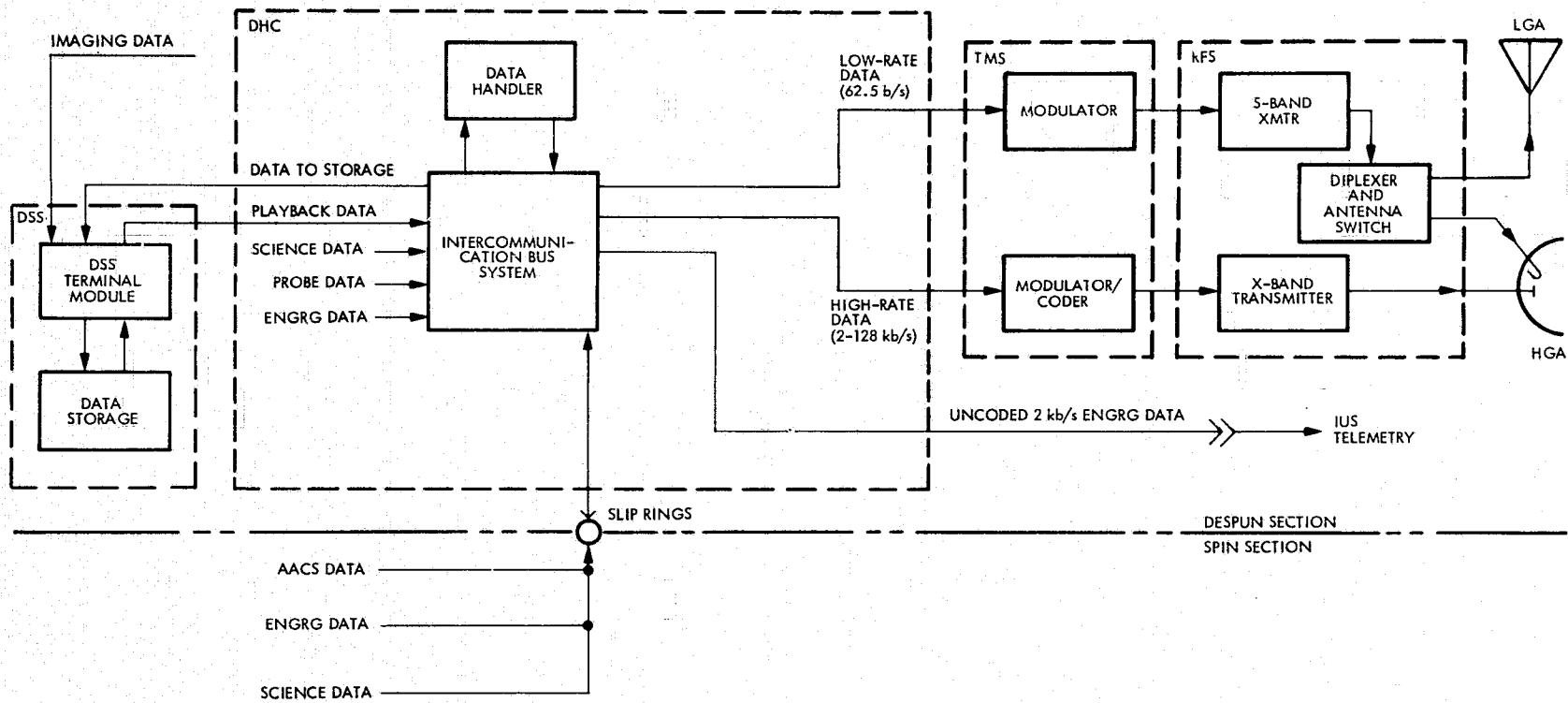
Fig. 3-6. Data flow (simplified diagram)

(c) Orbiter science data, consisting of digital data from each of the science subsystems including the needed housekeeping and calibration-reference data.

2. Data Storage

A NASA standard 4.5×10^8 bit digital tape recorder is used for data storage. The primary purpose of the tape recorder is to store the Probe data during Probe entry. In addition to this function, the tape recorder is used to store imaging data that exceeds the real-time communication rate (e. g. , pictures taken on centers of less than 60 s). The data storage subsystem can also be used to store engineering and fields and particle data for playback at a later time.

3. Data Rates and Formats

To reduce cost, a minimal number of data rates, formats, and telemetry modes were established. The format used for telemetry during STS and IUS operations will be influenced by the STS requirements.

The relatively long orbital periods and the tape-recorder capacity permit a low-rate-record/high-rate data dump mode of operation which tends to reduce mission operations costs by reducing high activity operation time.

The type of hardware used in the DHC (microprocessors) permits the implementation of conventional multi-interleaved formats or block telemetry formats. Whichever format is decided to be the most cost effective can be implemented by software. The data rates and formats are summarized in Tables 3-5(a) and (b). It should be noted that all formats are intended to be fixed (nonreprogrammable) prior to Orbiter system testing.

Table 3-5(a).   Telemetry formats and data rates

Format	Transmitted data rate	Function	Format contents
LRE	62.5 b/s	Cruise engineering	Optimized for overall subsystem performance monitoring and verification
HRE	2 kb/s	Launch, TCM and encounter engineering and real-time probe relay data	Optimized for propulsion and attitude/articulation control parameters; also provides for the 88 sym/s probe relay data
MRO	2 kb/s	Memory readout	Readout of DHC memories
FPS	2 kb/s	Fields and particles science	Fields and particles instrument data
IRS	64 kb/s	Infrared science	IR instrument data, fields and particles, instrument data, HRE, and filler bits as required
IMG	128 kb/s	Imaging science	CCD imaging data, fields and particles instrument data, HRE, and filler bits as required

Table 3-5(b).   Playback formats

Format	Transmitted data rate	Function	Format contents
HRE-PB	32 kb/s	Maneuver data and probe relay data when so recorded)	Recorded HRE data
FPS-PB	32 kb/s or 128 kb/s	Fields and particles data	Recorded FPS and LRE data (playback rate 30 kb/s) or data dumps, (playback rates) 126 kb/s) and real-time FPS data
IRS-PB	128 kb/s	IR data	Recorded IRS data (playback rate 126 kb/s) and real-time FPS data
IMG-PB	128 kb/s	Imaging data	Recorded IMG data (playback rate 126 kb/s) and real-time FPS data

4. <u>Telemetry Modes</u>

a. <u>Real-Time and Playback Data</u>. Telemetry modes were established consistent with STS compatibility and mission/science requirements. The telemetry modes are summarized in Table 3-6. The first four modes differ only in the interface across which the high-rate engineering (HRE) data are transmitted and in possible Orbiter state changed during an on-orbit checkout.

For redundancy, transmission of real-time data for Mode 10 (Probe entry) as well as storing these data for later playback is mandatory. Transmission of real-time fields and particles data during orbital cruise is a mode which can be replaced, when desired, by the data-dump mode (Mode 13). This allows data recorded over a period of 62.5 h (time to fill the tape recorder at 2 kb/s) to be played back in 1 h at a rate of 126 kb/s. Real-time transmission of IR and imaging science data is planned but is not necessary for collecting remote sensing data because of the data storage capability.

b. <u>On-orbit Checkout Data</u>. Telemetry during Mode 2 (on-orbit checkout) will be limited to the HRE format (2 kb/s, uncoded). If science instruments are turned on, only their engineering and status data contained in the HRE format will be used for performance verification. It is planned that science and DSS engineering and status data will be included in HRE format in sufficient quality and quantity to permit an adequate performance verification.

5. <u>Data Coding</u>

The reduction of bit-error rates (BER) by means of on-board data coding and ground decoding was considered but was not specifically addressed. A candidate coding scheme is one using Golay/Convolutional coding.

Table 3-6. JOP telemetry modes

Mode	Function	S-band status				X-band status					Tape recorder status
		Format	Data rate, b/s	Exciter Power Amplifier	Antenna	Format	Data rate, kb/s	Exciter Power Amplifier	TWTA	Format transmitted via	
1	Launch	LRE	62.5	On	LGA	HRE	2	Off	Off	IUS/Orbiter	Record 2 kb/s
2	On-orbit checkout	LRE	62.5	On	LGA	HRE	2	(As required)		IUS/Orbiter	As required[a]
3	On-orbit, detached	LRE	62.5	On	LGA	HRE	2	Off	Off	IUS	Record 2 kb/s
4	Kick-stage burn	LRE	62.5	On	LGA	HRE	2	Off	Off	None	Record 2 kb/s
5	Preacquisition	LRE	62.5	On	LGA	None	None	Off	Off	None	Off
6	Interplanetary cruise	LRE	62.5	On	HGA	None	None	Off	Off	None	Off
7	TCM	LRE	62.5	On	HGA	HRE	2	On	Off	None	Record 2 kb/s
8	Post-TCM	LRE	62.5	On	HGA	HRE-PB	32	On	On	HGA	Playback 32 kb/s
9	Memory readout	LRE	62.5	On	HGA	MRO	2	On	On	HGA	Off
10	Probe entry	LRE	62.5	On	HGA	HRE	2	On	On	HGA	Record 2 kb/s
11	Post-entry	LRE	62.5	On	HGA	HRE-PB	32	On	On	HGA	Playback 30 kb/s
12	Orbital cruise	LRE	62.5	On	HGA	FPS	2	On	On	HGA	Record 2 kb/s
13	Fields and particles data dump	LRE	62.5	On	HGA	FPS-PB	128	On	On	HGA	Playback 126 kb/s
14	IR science RT	LRE	62.5	On	HGA	IRS	64	On	On	HGA	Record 64 kb/s
15	IR playback	LRE	62.5	On	HGA	IRS-PB	128	On	On	HGA	Playback 126 kb/s
16	Imaging science RT	LRE	62.5	On	HGA	IMG	128	On	On	HGA	Record 128 kb/s
17	Imaging playback	LRE	62.5	On	HGA	IMG-PB	128	On	On	HGA	Playback 126 kb/s

NOTES:

[a] Playback data, if any, not to be transmitted (data rate into IUS limited to 2 kb/s).

6. <u>Image Recording and Playback</u>

A cursory analysis was made using predicted imaging data character-istics to make a preliminary assessment of the number of pictures storable, the smallest interval between pictures being recorded, and the time to fill the tape recorder at the highest record rate (2.56 Mb/s). The calculations were based on 8-bit encoding of each of the picture elements (pixels) on a 800 X 800 pixel CCD raster.

A picture line was arbitrarily determined to contain 88 bits of over-head on the following basis:

Frame synchronization	32
Format (and block) ID	8
Spacecraft Time	24
Picture and line count	16
SSI Status and Engineering	8
	88 bits

Because one line contains 800 pixels, the number of pixel plus over-head bits-per-line is 6488, and the number of bits-per-picture (800 lines) is 5,190,400 bits. Additionally, engineering data (at 2 kb/s) and fields and particles data (at 2 kb/s) are being recorded. On this basis, the following were established:

    (a) Number of pictures which can be stored in the DSS = 86.

    (b) Minimum picture interval (at max. record rate) = 2 s.

I. TELECOMMUNICATIONS

The Jupiter Orbiter Telecommunication System provides the general functions of carrier transponding, ranging, command detection, telemetering of multiple rate data, and relaying of data from the Jupiter Probe to Earth. Performance of these functions is dependent on the overall system design of

the telecommunications links described in paragraph J of this section. A block diagram is shown in Fig. 3-7. The performance of the overall links is documented using design control tables for specific points in the mission and plots of performance versus time. The predicted performance is contingent on the systems and subsystems meeting their required performance parameters.

1. Carrier Transponding

The Orbiter will accept a modulated or unmodulated S-Band (2115 MHz) uplink carrier from the DSN and have the capability to use that carrier to generate phase and frequency coherent downlink carriers at S-band (2297 MHz) and X-band (8422 MHz). The capability will also exist to generate a downlink S-and X-band carrier; when the uplink carrier is not received, the source of the basic frequency reference can be either an auxiliary oscillator or an ultra stable oscillator.

2. Ranging

Telecommunications will receive and demodulate the ranging data from the uplink carrier and will modulate (a) the S-band carrier with ranging data upon command, (b) the X-band carrier with ranging data upon command, and (c) both carriers simultaneously with the ranging data upon command.

3. Commanding

The Orbiter will receive and demodulate the command signal from the uplink carrier. This command signal will be a 31.25-b/s pulse-code modulation signal biphase modulated on a 16-kHz subcarrier. This signal will conform to the NASA command standard. The command signal will be demodulated from the carrier by the standard command detector, which is a portion of the transponder. This detector will acquire and track subcarrier and bit synchronization, detect the data bits, and supply them to the DHC. In addition, detector status will be provided to the DHC.

RELAY LINK

S/X BAND RADIO LINK

```
┌─────────────┐                              ┌─────────────┐   TRACKING AND RANGING   ┌─────────────┐
│             │                              │             │◄─────────────────────────►│             │
│   JUPITER   │          TELEMETRY           │   JUPITER   │    S-BAND TELEMETRY       │     DSN     │
│    PROBE    │─────────────────────────────►│   ORBITER   │──────────────────────────►│  TRACKING   │
│             │                              │             │        COMMAND            │   STATION   │
│             │                              │             │◄──────────────────────────│             │
│             │                              │             │    X-BAND TELEMETRY       │             │
│             │                              │             │──────────────────────────►│             │
└─────────────┘                              └─────────────┘                          └─────────────┘
```
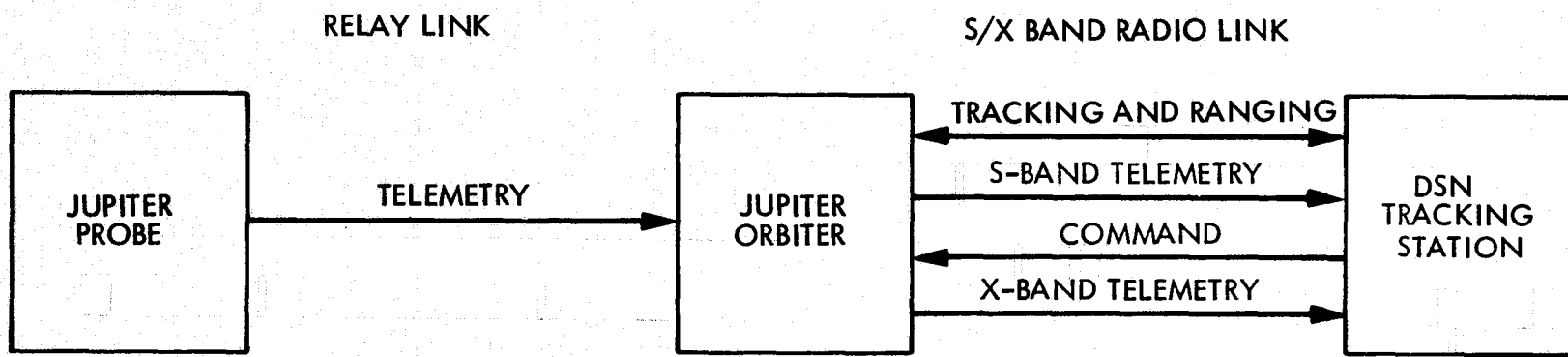
Fig. 3-7. Telecommunications

4. Telemetering

The Orbiter will be able to transmit low-rate engineering data continuously at S-Band, minimizing the use of the 64-m stations, and will also transmit high-rate data at 2 kb/s, 32 kb/s, 64 kb/s, and 128 kb/s. High-rate data may be encoded in the TMS with a relatively short constraint-length code having a symbol rate up to twice the bit rate. The low-rate data will not be encoded, and the capability will exist to transmit uncoded data at 2 kb/s.

5. Relay Telemetry

The Orbiter will receive data transmitted from the Jupiter Probe and then transmit this data uncoded in real time to Earth as well as store the data on the Orbiter for later playback to Earth.

J. TELECOMMUNICATION LINKS

1. General

The telecommunication links are defined to include the Orbiter telecommunication flight hardware (S-and X-band), the communication media, and the Deep Space Station (DSS) hardware from the antenna through the bit-demodulation or modulation. The performance of the links are documented and controlled by the use of design control tables. Subsystem parameters are specified and controlled by a telecommunication control document.

2. Requirements

a. Radiometric. Radiometric requirements are listed as follows:

(1) Carrier Transponding: the JOP design will provide carrier transponding with the S-band downlink ($\approx$2295 MHz) to be 240/221 times the S-Band uplink ($\approx$2115 MHz) frequency. The

performance of this link will allow resolution of radial Orbiter velocity of less than 1 mm/s during specified periods. In addition, an X-band ($\simeq$8422 MHz) downlink carrier 880/221 times the uplink S-band ($\simeq$2115 MHz) carrier uplink frequency will be provided when the HGA is pointed (<0.14°) at Earth. The performance of this link allows resolution of radial spacecraft velocity to 1 mm/s.

(2) One-way carrier: continuous downlink carrier at S-band is provided when no uplink is sent to the Orbiter. The frequency stability of the Orbiter will be less than (TBD).

b. Command. Earth-based command performance will be via:

(1) The LGA from a 26-m, 20-kW DDS station for the first 140 days of the mission with the baseline trajectory.

(2) The HGA from a 26-m 10-kW DSS from HGA acquisition to end of mission.

(3) The LGA from a 64-m 100-kW DSS from launch to end of mission with the LGA boresight pointed within 30 deg of Earth.

c. Radio Science. The general radio-science requirements will be the study of the atmospheres of Jupiter and the satellites, IO and Ganymede. In addition, the radiometric data will be used to determine the masses of the satellites. Detailed-radio science requirements will be generated.

d. Relay Link. These requirements will be generated by the Mission Design Team and will be documented in the Orbiter-to-Probe interface requirements.

3. Baseline Trajectory

Figures 3-8 and 3-9 indicate the range and Earth/Orbiter/Sun angles used for calculating the link performance. The data was obtained from the JOP design trajectory with a January 9, 1982 launch date and an
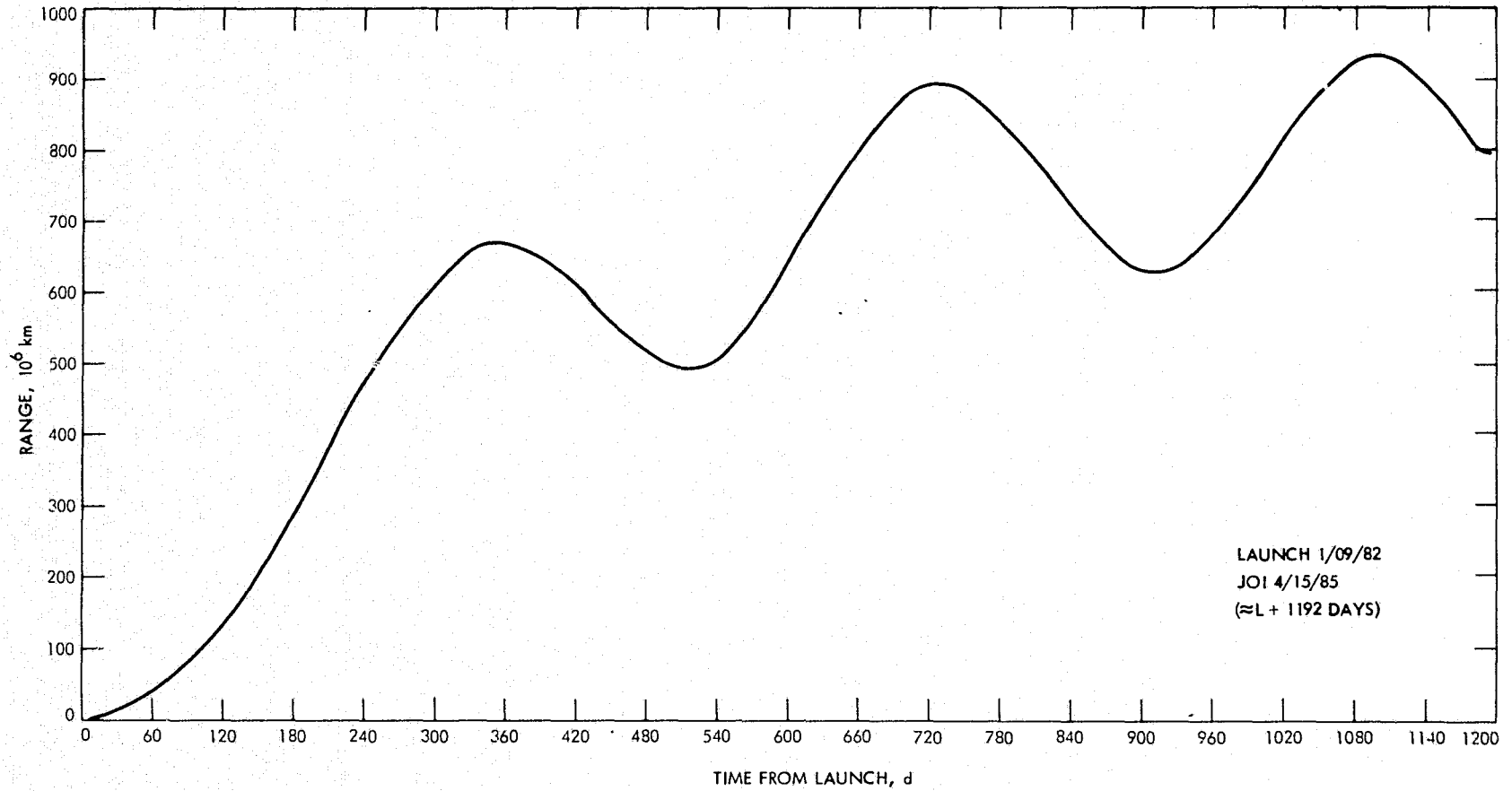
Fig. 3-8. Range versus time (baseline trajectory)
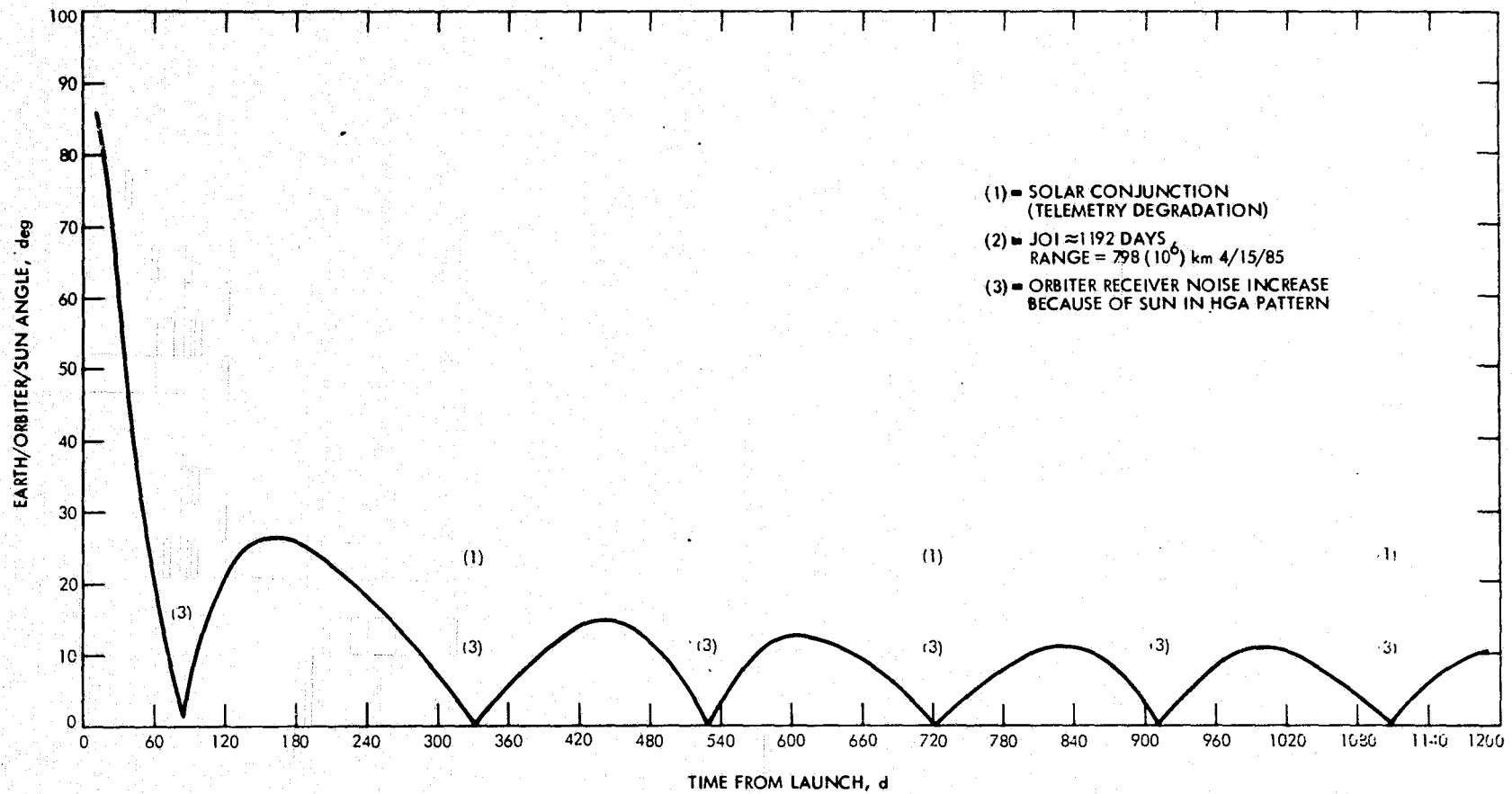
Fig. 3-9. Earth/spacecraft/Sun angle versus time (baseline trajectory)

April 15, 1985 Jupiter arrival date[1]. Further interaction will be required with the Mission design personnel to avoid heavy activities during solar conjunction periods.

4. Cruise Telemetry Performance

Table 3-7 is a computer printout of the engineering telemetry performance for day 250 with the Orbiter LGA 26-m DSS and the baseline trajectory. The -24 db performance margin resulted in the following actions:

(a) Bias the Orbiter off the sun toward Earth to keep the low-gain axis within 60 deg of Earth early in the mission.

(b) Use the Orbiter bias to allow the HGA to be pointed at Earth by day 60. Figure 3-10 plots the resulting performance for the 62.5 b/s cruise telemetry. There is a gap in 26-m coverage from day 20 to day 60 during which 64-m coverage will be required, or the Orbiter will have to be biased further than the present 25 deg off the Sun.

5. Command Performance

Table 3-8 is a printout of the command link performance for day 250 over the LGA using the baseline trajectory (no Orbiter bias is assumed). Figure 3-11 illustrates the performance during the total mission. Command coverage is available over the LGA with the 26-m DSS at 20 kW for approximately 135 days. After that time, commands must be over the Orbiter HGA or by using the 64-m antenna at 100 kW. It will be possible to command the Orbiter over the LGA using the 64-m DSS with 100 kW for the entire mission with cone angles of less than 40 deg. Use of the 400-kW transmitter will allow cone angles of approximately 70 deg.

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[1]Interoffice memorandum, IOM 393.1-1478, R.J. Boain to J.R. Kolden, "JOP Trajectory Data Relevant to Spacecraft Communications," April 5, 1976.

Table 3-7. Downlink carrier design-control table (S-band telemetry,
5 W, ranging on, LGA, 26-m DSN subnet)

	DESIGN	FAV TOL	ADV TOL	MEAN	VARIANCE
TRANSMITTER PARAMETERS					
				36.27	.169
1) RF POWER TO ANTENNA, DBM					
TRANSMITTER POWER, DBM	37.00	1.00	-1.00	37.00	.167
TRANSMIT CIRCUIT LOSS, DB	-.69	.05	-.13	-.73	.003
2) ANTENNA CIRCUIT LOSS, DB	.00	.00	.00	.00	.000
3) ANTENNA GAIN, DBI	6.63	.36	-.56		
4) POINTING ERROR, DB	.00	.00	.00	6.56	.035
LIMIT CYCLE, DEG	.05	-.05	.00		
ANGULAR ERRORS, DEG	.00	-.09	.09		
PATH PARAMETERS					
5) SPACE LOSS, DB	-273.68			-273.68	.000
FREQ = 2295.00 MHZ					
RANGE = 5.019+08 KM					
= 3.35 AU					
6) ATMOSPHERIC ATTENUATION,DB	.00	.00	.00	.00	.000
RECEIVER PARAMETERS					
7) POLARIZATION LOSS, DB	-.03	.01	-.02		
8) ANTENNA GAIN, DBI	53.30	.60	-.60		
9) POINTING LOSS, DB	-.04	.04	-.08	53.23	.121
10) NOISE SPEC DENS, DBM/HZ	-181.70	-.27	.57	-181.55	.020
TOP (ZENITH), K	41.00	-3.00	6.88		
DELTA TOP (EL ANGLE), K	8.00	.00	.00		
810-5 / SIN(EL), K	.00	.00	.00		
HOT BODY NOISE, K	.00	.00	.00		
ELEV ANGLE = 25.00 DEG					
11) CARR THR NOISE BW, DB-HZ	10.79	-1.00	.00	10.46	.056
TOTAL POWER SUMMARY					
12) RCVD POWER,PT,DBM				-177.62	.326
(1+2+3+4+5+6+7+8+9)					
13) RCVD PT/NO, DB-HZ, (12-10)				3.93	.346
14) RANGING SUPPRESSION, DB	-.54	.20	-.40	-.61	.016
15) TELEMETRY SUPPRESSION, DB	-3.84	.10	-.12	-3.85	.002
16) CARR PWR/TOT PWR,DB(14+15)				-4.45	.018
17) RCVD CARR PWR,DBM(12+16)				-182.07	.344
18) CARR SNR IN 2BLO,DB(17-10-11)				-10.98	.419
				2S=	1.17
				3σ=	1.76

Table 3-7 (contd)

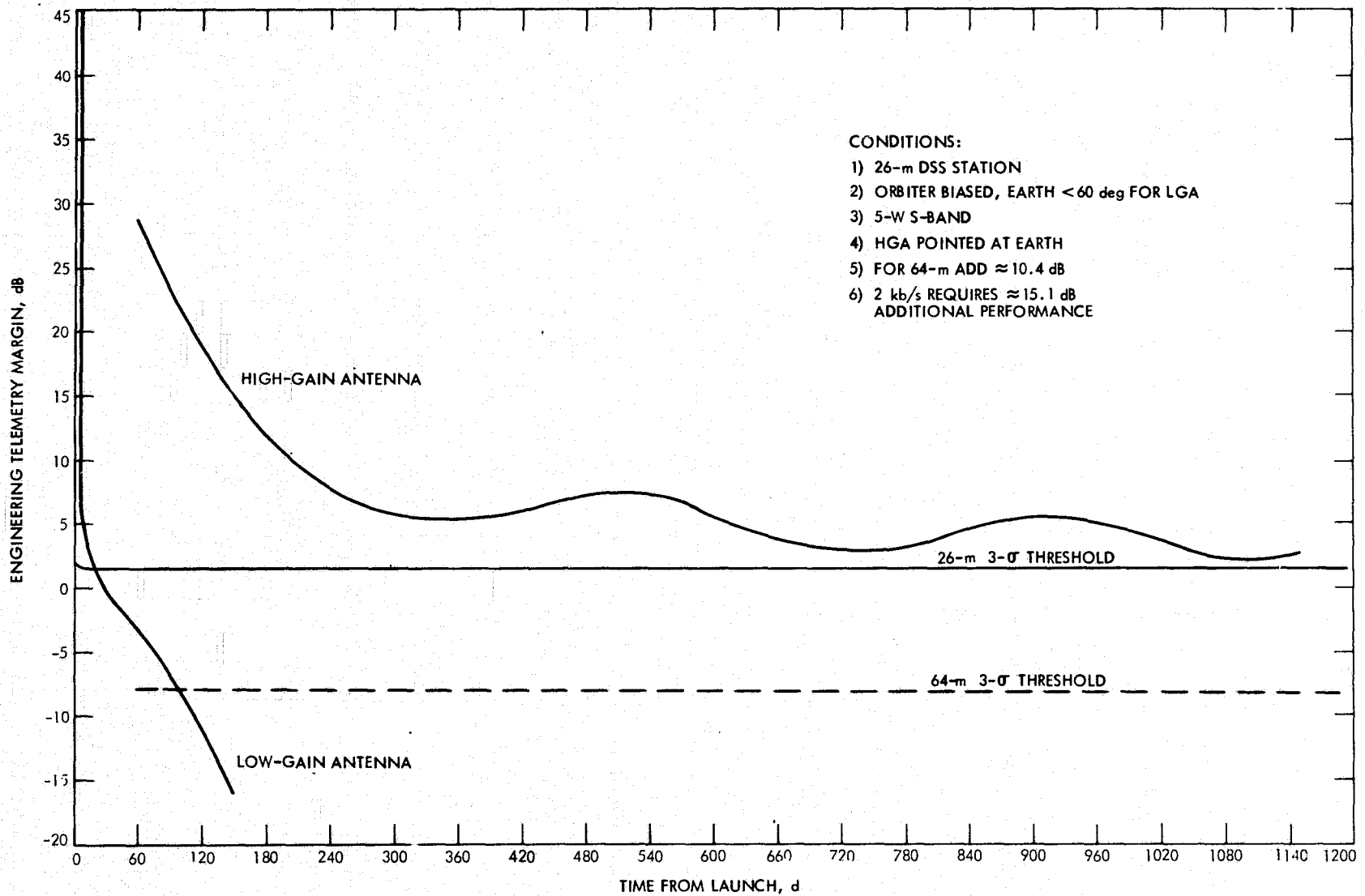|	DESIGN	FAV TOL	ADV TOL	MEAN	VARIANCE
DATA CHANNEL PERFORMANCE					
19) DATA BIT RATE, DB	17.96	.00	.00	17.96	.000
BIT RATE = 62.5 BPS					
20) DATA PWR/TOTAL PWR, DB	-2.31	.00	.00	-2.31	.000
TLM MOD INDEX = 50.0 DEG					
21) DATA PWR TO RCVR,DBM(12+14+20)				-180.54	.342
22) ST/NO TO RCVR,DB(21-19-10)				-16.94	.362
23) SYSTEM LOSSES, DB	-2.30	.54	-.73	-2.36	.068
RADIO LOSS, DB	-1.20	.50	-.70		
DEMOD, DETECT LOSS, DB	-1.00	.20	-.20		
WAVEFORM DIST LOSS, DB	-.10	.05	-.05		
24) ST/NO OUTPUT, DB (22+23)				-19.31	.429
25) THRESHOLD ST/NO, DB	5.30	.00	.00	5.30	.000
THRESHOLD BIT ERROR RATE	5.00-03				
26) PERFORMANCE MARGIN,DB(24-25)				-24.61	.429
				2S= 1.31	
				3σ= 1.97	
TIME = 250.00					

Fig. 3-10. Engineering telemetry (62.5 b/s) performance versus time

Table 3-8.   Uplink carrier design-control table (S-band command link, 20 kW, ranging off, LGA, 26-m DSN subnet)

	DESIGN	FAV TOL	ADV TOL	MEAN	VARIANCE
TRANSMITTER PARAMETERS					
1) RF POWER, DBM	73.01	.50	-.50	73.01	.042
POWER OUTPUT = 20.0 KW					
2) CIRCUIT LOSS, DB	.00	.00	.00	.00	.000
3) ANTENNA GAIN, DBI	51.80	.90	-.90		
ELEV ANGLE = 25.00 DEG					
4) POINTING LOSS, DB	-.40	.00	.00	51.40	.270
PATH PARAMETERS					
5) SPACE LOSS, DB	-272.97			-272.97	.000
FREQ = 2115.00 MHZ					
RANGE = 5.019+08 KM					
= 3.35 AU					
6) ATMOSPHERIC ATTENUATION,DB	.00	.00	.00	.00	.000
RECEIVER PARAMETERS					
7) POLARIZATION LOSS, DB	-.03	.01	-.02		
8) ANTENNA GAIN, DBI	6.63	.36	-.56		
9) POINTING ERROR, DB	.00	.00	.00	6.50	.070
LIMIT CYCLE, DEG	.05	-.05	.00		
ANGULAR ERRORS, DEG	.00	-.09	.09		
10) REC CIRCUIT LOSS, DB	-.69	.05	-.13	-.73	.003
11) NOISE SPEC DENS, DBM/HZ	-167.08	-.78	.82	-167.06	.071
OPERATING TEMP, K	1419.00	-234.00	293.00		
HOT BODY NOISE, K	.00	.00	.00		
12) CARR THR NOISE BW, DB-HZ	12.55	-.51	.46	12.53	.039
TOTAL POWER SUMMARY					
13) RCVD POWER,PT,DBM				-142.79	.384
(1+2+3+4+5+6+7+8+9+10)					
14) RCVD PT/NO, DB-HZ (13-11)				24.27	.455
15) RANGING SUPPRESSION, DB	.00	.00	.00	.00	.000
16) COMMAND SUPPRESSION, DB	-5.66	.50	-.50	-5.66	.042
17) CARR PWR/TOT PWR,DB(15+16)				-5.66	.042
18) RCVD CARR PWR, DBM (13+17)				-148.45	.426
19) CARR SNR IN 2BLO,DB(18-11-12)				6.08	.536

2S = 1.31
3σ = 1.97

Table 3-8 (contd)

	DESIGN	FAV TOL	ADV TOL	MEAN	VARIANCE
DATA CHANNEL PERFORMANCE					
20) DATA BIT RATE, DB BIT RATE = 31.3 BPS	14.95	.00	.00	14.95	.000
21) DATA PWR/TOTAL PWR, DB	-1.38	.18	-.20	-1.39	.006
22) DATA PWR TO RCVR,DBM(13+15+21)				-144.18	.390
23) ST/NO TO RCVR,DB(22-20-11)				7.94	.461
24) SYSTEM LOSSES, DB RADIO LOSS, DB DEMOD, DETECT LOSS, DB WAVEFORM DIST LOSS, DB	-2.61 .00 .00 .00	1.14 .00 .00 .00	-1.15 .00 .00 .00	-2.61	.219
25) ST/NO OUTPUT, DB (23+24)				5.32	.680
26) THRESHOLD ST/NO, DB THRESHOLD BIT ERROR RATE	10.60 1.00+06	-.50	.50	10.60	.042
27) PERFORMANCE MARGIN,DB(25-26)				-5.28 2S= 1.70 3σ= 2.55	.721

TIME = 250.00

CONDITIONS:

(1) LOW-GAIN ANTENNA, 31.25 b/s

(2) BASELINE TRAJECTORY

(3) 26-m 20-kW DSS

(4) FOR 64-m 100-kW ADD 15.9 dB

(5) FOR HIGH-GAIN ANTENNA, AFTER 60 d, ADD APPROXIMATELY 31 dB TO MARGIN

3-σ TOLERANCE, 26 m

20 kW

3-σ TOLERANCE, 64 m

100 kW

COMMAND MARGIN, dB

TIME FROM LAUNCH, d
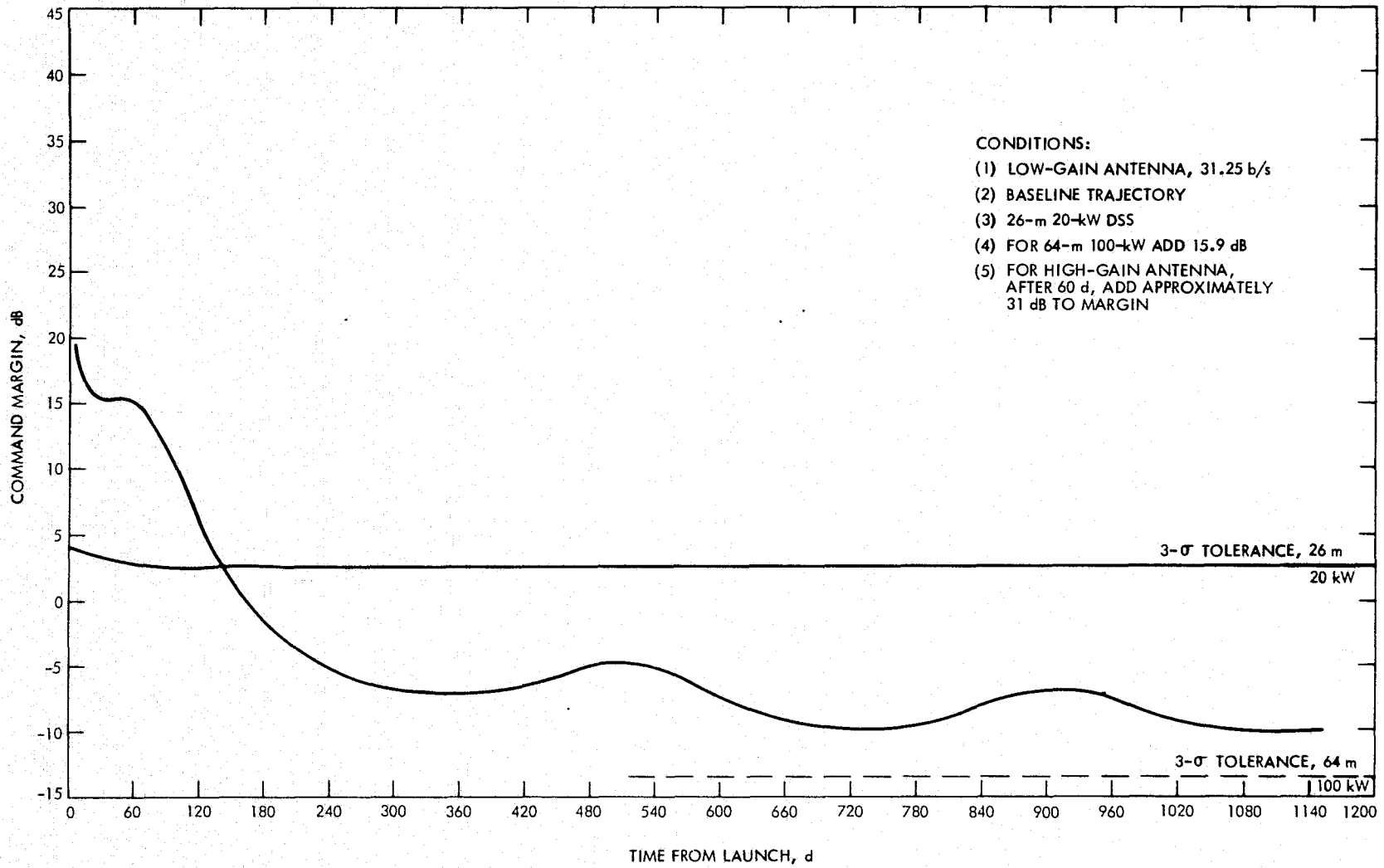
Fig. 3-11. Command performance versus time

6.   High Rate Telemetry

Table 3-9 is a computer printout of the high-rate telemetry (128-kb/s) performance with the 5-m Orbiter antenna transmitting 20 W power at X-band, and with the 64-m DSS receiving.   Figure 3-12 indicates that there is adequate performance margin throughout the mission.   The indicated performance is based on expected weather, no additional sun-induced noise within 5 deg of the ground antenna boresight, and no increase in noise caused by Jupiter in the antenna beam.

Table 3-9.   Downlink carrier design-control table (X-band telemetry, high power, ranging on, HGA, 64-m DSN subnet)

	DESIGN	FAV TOL	ADV TOL	MEAN	VARIANCE
TRANSMITTER PARAMETERS					
1) RF POWER TO ANTENNA, DBM				43.80	.064
TRANSMITTER POWER, DBM	43.79	.64	-.60	43.80	.064
TRANSMIT CIRCUIT LOSS, DB	.00	.00	.00	.00	.000
2) ANTENNA CIRCUIT LOSS, DB	-.18	.10	-.10	-.18	.003
3) ANTENNA GAIN, DBI	50.20	.50	-1.00	50.03	.097
4) POINTING ERROR, DB	-.43	.43	-.61	-.43	.050
LIMIT CYCLE, DEG	.05	-.05	.00		
ANGULAR ERRORS, DEG	.00	-.09	.09		
PATH PARAMETERS					
5) SPACE LOSS, DB	-284.97			-284.97	.000
FREQ = 8422.00 MHZ					
RANGE = 5.019+08 KM					
= 3.35 AU					
6) ATMOSPHERIC ATTENUATION,DB	-.29	.05	-.09	-.31	.002
RECEIVER PARAMETERS					
7) POLARIZATION LOSS, DB	-.06	.04	-.04		
8) ANTENNA GAIN, DBI	72.30	.70	-.30		
9) POINTING LOSS, DB	-.20	.20	-.60	72.07	.164
10) NOISE SPEC DENS, DBM/HZ	-182.65	-.41	.69	-182.51	.033
TOP (ZENITH), K	21.40	-2.00	2.00		
DELTA TOP (EL ANGLE), K	7.10	-.30	.10		
810-5 / SIN(EL), K	10.88	-2.89	6.48		
HOT BODY NOISE, K	.00	.00	.00		
ELEV ANGLE = 25.00 DEG					
11) CARR THR NOISE BW, DB-HZ	10.00	-1.00	1.00	10.00	.167
TOTAL POWER SUMMARY					
12) RCVD POWER,PT,DBM				-119.98	.380
(1+2+3+4+5+6+7+8+9)					
13) RCVD PT/NO, DB-HZ, (12-10)				62.52	.413
14) RANGING SUPPRESSION, DB	-.21	.05	-.11	-.23	.001
15) TELEMETRY SUPPRESSION, DB	-15.21	.58	-.62	-15.22	.060
16) CARR PWR/TOT PWR,DB(14+15)				-15.45	.061
17) RCVD CARR PWR,DBM(12+16)				-135.44	.441
18) CARR SNR IN 2BLO,DB(17-10-11)				37.07	.641

$$2S = 1.33$$
$$3\sigma = 2.00$$

Table 3-9 (contd)

|	DESIGN	FAV TOL	ADV TOL	MEAN	VARIANCE
DATA CHANNEL PERFORMANCE					
19) DATA BIT RATE, DB	51.07	.00	.00	51.07	.000
BIT RATE = 128000.0 BPS					
20) DATA PWR/TOTAL PWR, DB	-.13	.01	-.01	-.13	.000
TLM MOD INDEX = 80.0 DEG					
21) DATA PWR TO RCVR,DBM(12+14+20)				-120.34	.381
22) ST/NO TO RCVR,DB(21-19-10)				11.09	.415
23) SYSTEM LOSSES, DB	-.65	.05	-.34	-.75	.008
RADIO LOSS, DB	-.30	.00	.00		
DEMOD, DETECT LOSS, DB	-.17	.03	-.34		
WAVEFORM DIST LOSS, DB	-.18	.04	-.04		
24) ST/NO OUTPUT, DB (22+23)				10.34	.422
25) THRESHOLD ST/NO, DB	2.46	.00	.00	2.46	.000
THRESHOLD BIT ERROR RATE	5.00-03				
26) PERFORMANCE MARGIN,DB(24-25)				7.88	.422
				2S= 1.30	
				3σ= 1.95	

TIME = 250.00
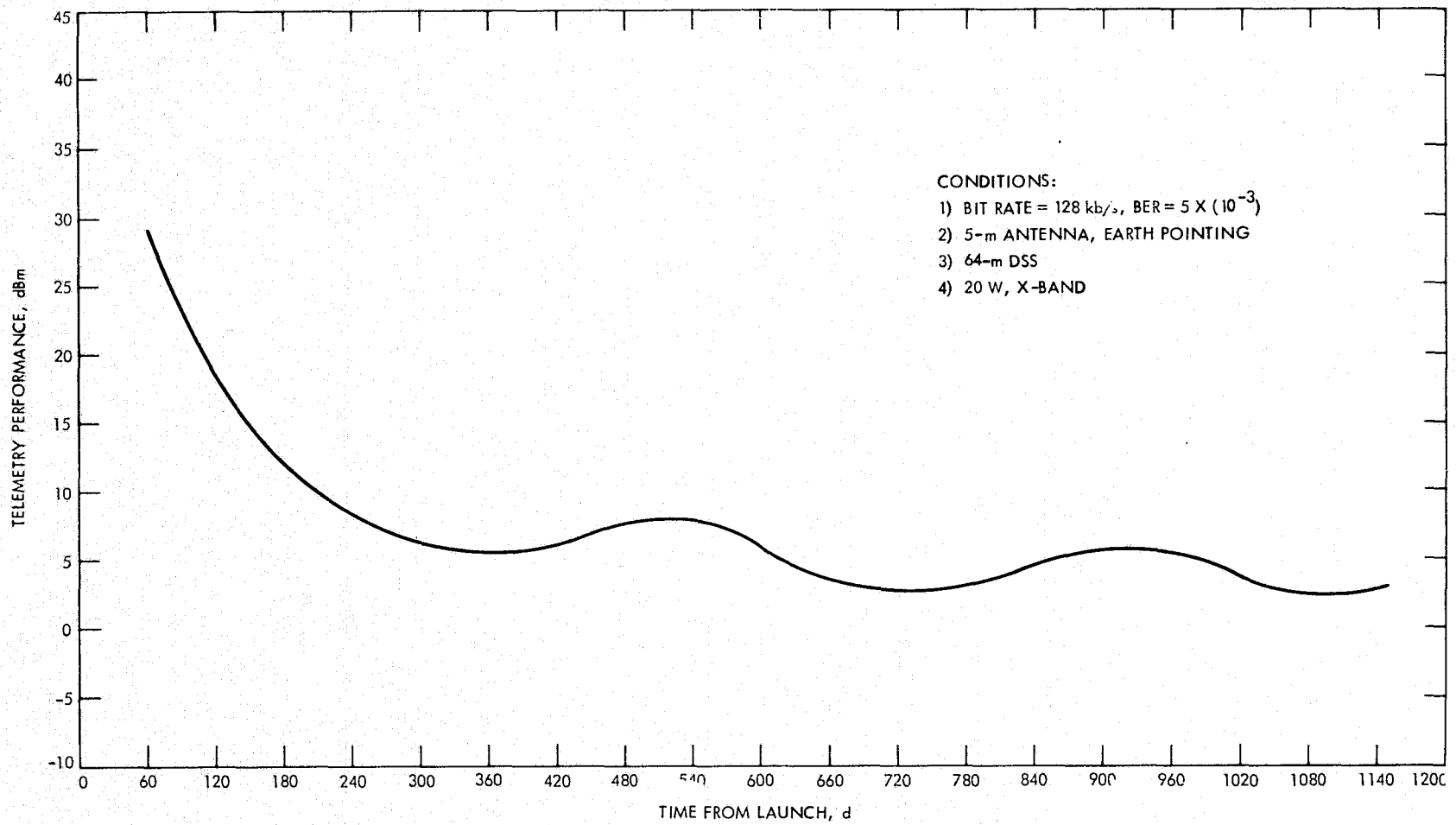
Fig. 3-12. High-rate telemetry performance, 128 kb/s

SECTION IV

ORBITER SUBSYSTEM DESCRIPTIONS

A. MECHANICAL SUBSYSTEMS

1. Structure

The primary function of the structure subsystem is to integrate, support, align, and protect the individual elements of flight equipment which comprise the Probe and Orbiter. As such, the structure provides:

(a) Structural load paths and support for all components during assembly, testing, ground handling, launch, and flight.

(b) Isolation from, or attenuation of, shock- and vibration-induced environments prior to and during flight.

(c) Mounting surfaces and/or bracketry which locate, position, and/or align all critical elements to the degree required for proper function.

(d) Thermal conductance, thermal isolation, and view factors necessary for normal operation of the individual components.

(e) Radiation shielding and/or separation of electronics and sensors for protection from Orbiter- and mission-induced environments.

The major structural and mechanical design features of the Orbiter are described in Section III B.

2. Electronic Packaging

Electronic assemblies are housed in both the spinning and despun electronic bays. Bay design is such that MJS, Viking, NASA-standard, and other packaging configurations can be accommodated.

3. Temperature Control

The temperature control subsystem will provide the active and passive means of controlling component temperatures to within required limits. The JOP temperature control mechanization will be roughly equivalent to that utilized for the MJS spacecraft. Solar-thermal inputs will decrease with time from one solar constant at Earth to essentially a no-Sun condition at Jupiter, which necessitates a thermal design geared to rejecting heat at Earth and retaining it at Jupiter. Standard Mariner techniques and hardware (as utilized in the MJS'77 design) should suffice. Louvers, bay shields, blankets, commandable replacement heaters, and RHUs will be used wherever appropriate.

B. RADIO FREQUENCY SUBSYSTEM

The RFS, along with the TMS and S- and X-band antenna subsystems, make up the Orbiter Telecommunications System. A block diagram of this system is shown in Fig. 4-1. The performance capabilities of this system are discussed in Section III-I.

1. Elements

The RFS consists of the following elements:

(a) Two NASA standard deep space transponders, shown in Fig. 4-2, each of which includes the S-band exciter, the S-band receiver, the X-band exciter, and the command-detector unit.

(b) One control unit which includes the X-band filter/hybrid.

(c) Two S-band solid-state power amplifiers.

(d) Two X-band traveling-wave-tube amplifiers (TWTA).

(e) One S-band microwave switching unit which includes a filter/hybrid and an S-band diplexer.

(f) Two output filters.

(g) One ultra stable oscillator.

Figure 4-3 is a block diagram of the RFS.
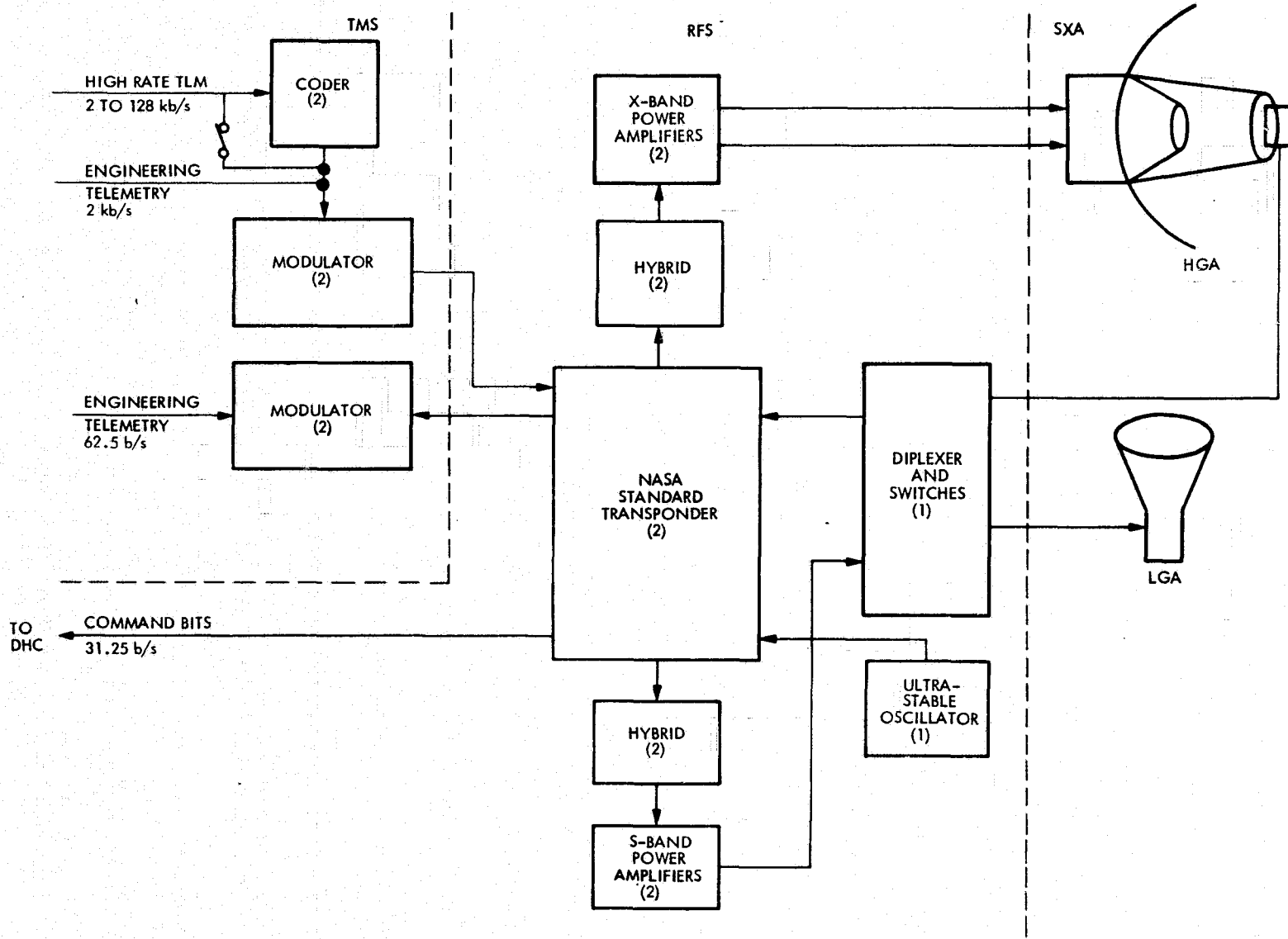
Fig. 4-1. Telecommunications system

Fig. 4-2. NASA standard deep-space transponder

Fig. 4-3. Radio Frequency Subsystem

2. Operation

a. S-Band Receivers. The S-band signal from the DSN is received through either the LGA or HGA; this signal is switched through the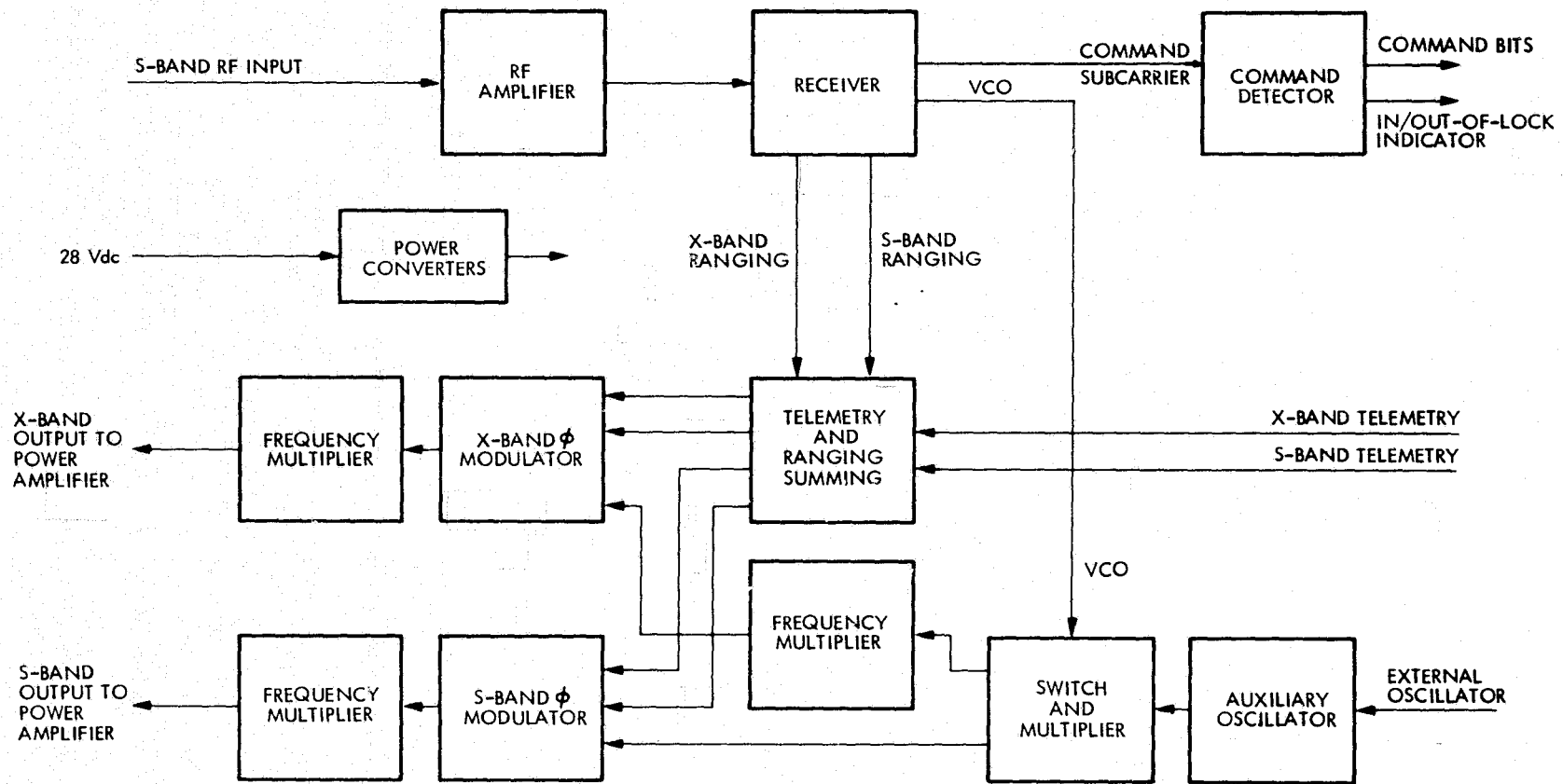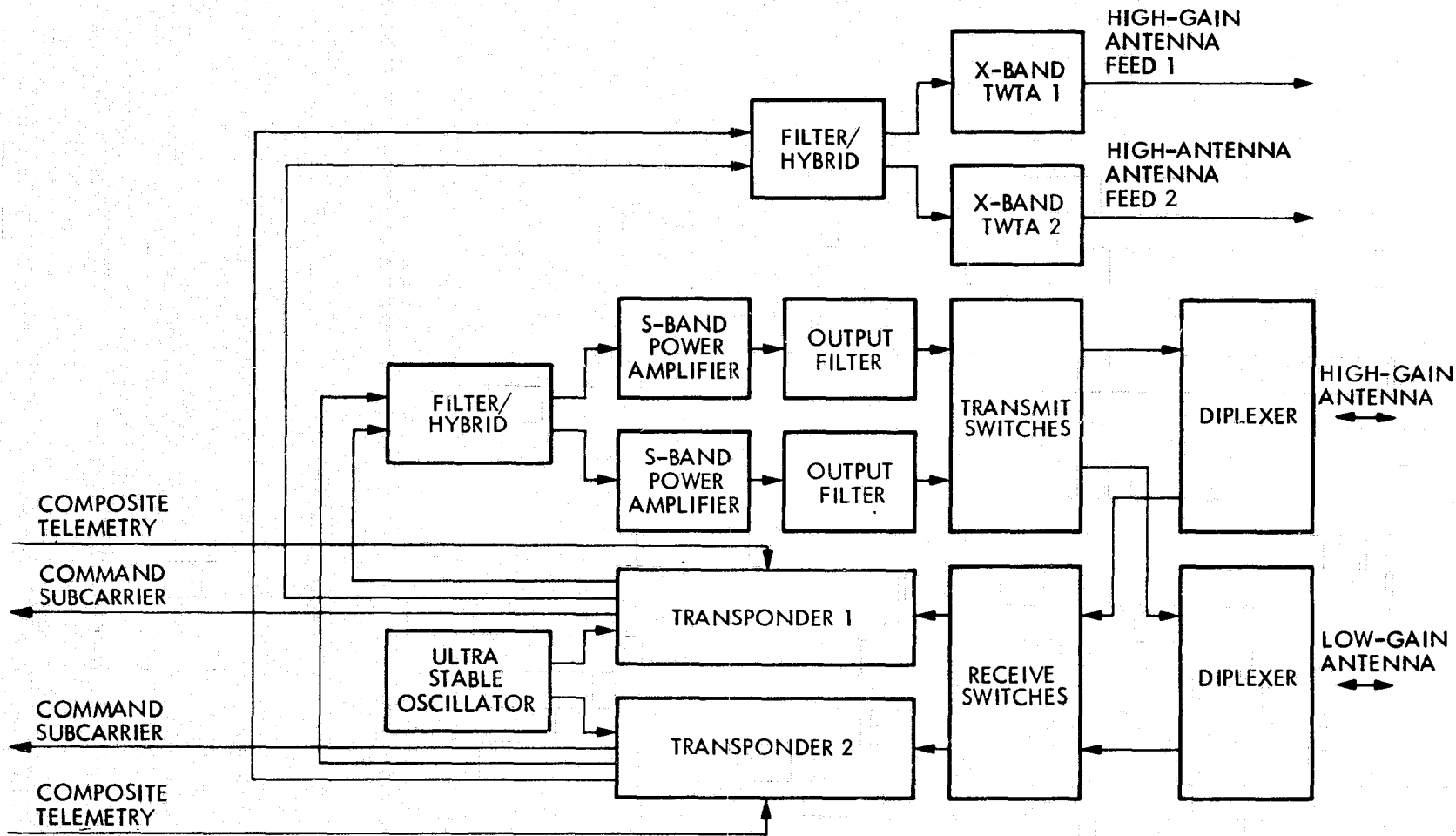 diplexer to a receiver. The receiver operates continuously and is a narrow-band, automatic-phase tracking double-conversion superheterodyne, which acquires and tracks the phase of the incoming carrier.

The receiver provides demodulation of the composite command subcarrier and detects and routes the command bits to the command processor within the DHC. It also provides ranging signals to modulate the downlink carrier. The receiver provides an in-lock/out-of-lock indication to be used by the exciter for selection of the receiver voltage-controlled oscillator (VCO) signal or the auxiliary oscillator signal for generation of the downlink carrier. Command detector in/out-of-lock status is supplied to the command processor, as well as input signal signal-to-noise ratio and detector status indications for telemetry. A functional block diagram of the command detector is shown in Fig. 4-4.

b. S-Band Exciters. The S-band exciters receive a reference signal from the receiver (when in lock) or from the auxiliary oscillator or ultra-stable oscillator (when out of lock). This reference signal is multiplied in frequency and phase to be 240/221 times the uplink frequency when the operating mode is two-way (in lock). The S-band exciter also contains a phase modulator to modulate the carrier with the Orbiter telemetry information and turn-around ranging information.

c. S-Band Power Amplifier. The exciter provides a signal of relatively constant amplitude to drive the filter hybrid which allows either exciter to drive either of the S-band power amplifiers. The power amplifiers are solid-state amplifiers in which the composite signal is amplified to a level of approximately 6.3 W. This signal is switched to one diplexer; the output of the diplexer, approximately 5 W, is passed to either the LGA or HGA.

d. X-Band Exciters. The X-band exciters operate similarly to the S-band except that the reference frequency is derived from the S-band exciters.
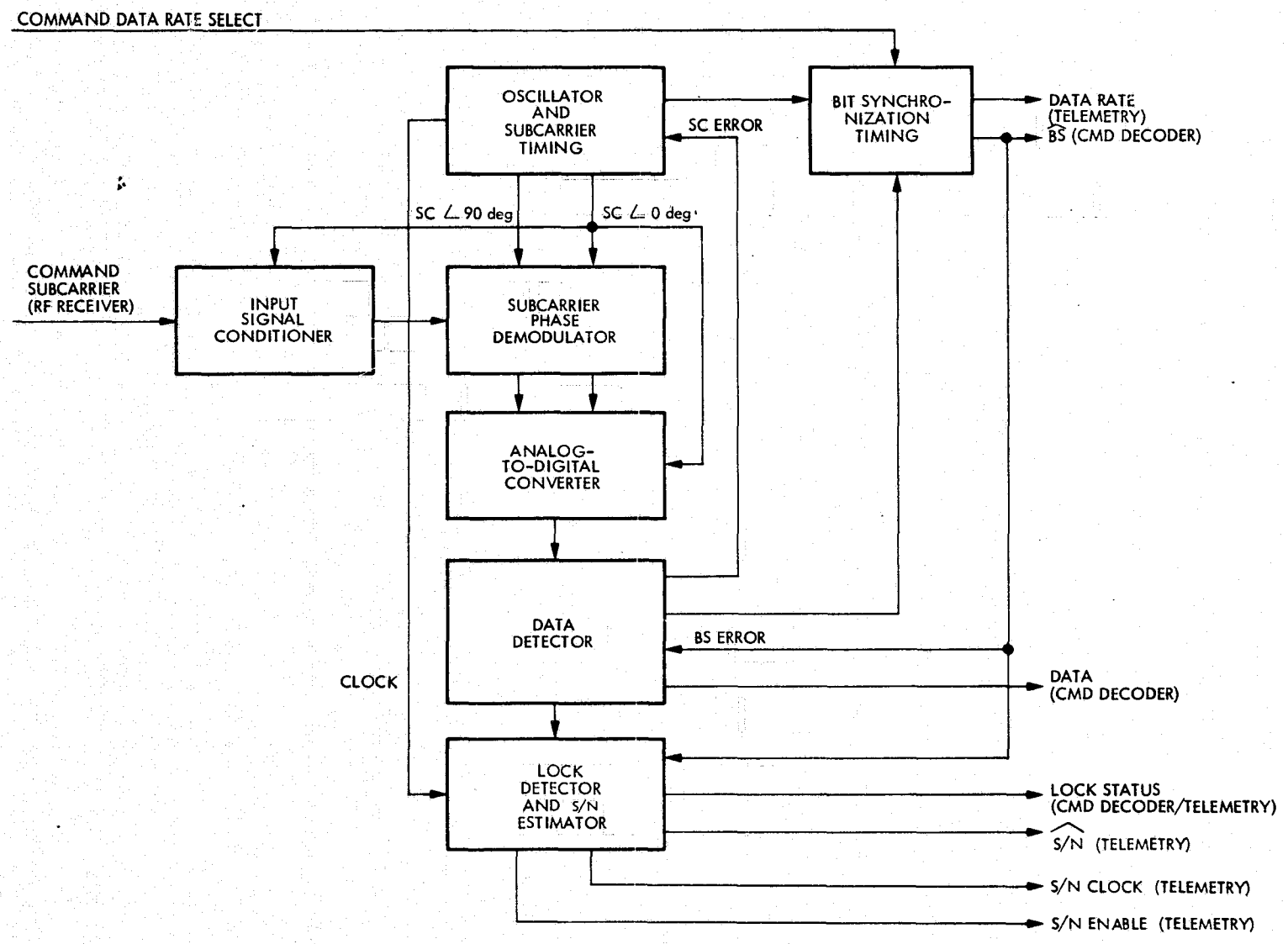
Fig. 4-4. NASA standard command detector unit

The reference frequency is multiplied so that the frequency and phase is 11/3 times the S-band transmitted frequency. The X-band exciters provide adequate output signal level to drive the filter hybrid, which allows the powered exciter to drive the powered X-band TWTA. The X-band exciter phase modulates the carrier with the composite high-rate telemetry information and ranging information.

   e.   X-Band Power Amplifier. The TWTA provides one of two power levels (12 W and 21 W) through its waveguide to the HGA feed. The X-band portion of the HGA feed consists of two feeds (one fed by each TWTA), one with right-hand circular polarization and one with left-hand circular polarization; this eliminates switching of the X-band output signal. The X-band signal goes only to the HGA.

C.   TELEMETRY MODULATION SUBSYSTEM

   1.   Elements

      The TMS consists of the following elements:

      (a)   Two MJS'77 telemetry modulators.
      (b)   A data coder.

   2.   Operation

      a.   Modulator. The TMS receives two data streams from the data handling and control remote terminal (see Section IV-F). One data stream with low-rate data biphase modulates a 22.5-kHz square-wave subcarrier. This composite signal is routed to modulate the S-band exciter only. The second data stream (which may be coded) biphase modulates either a 22.5-kHz or a 360-kHz subcarrier. This composite telemetry signal is fed to the RFS where the X-band carrier is phase-modulated.

      b.   Coder. No specific coding scheme was selected for JOP. The telemetry link performance, however, was based on use of a Golay coder, in combination with a convolutional coder, for the high-rate channel.

D. S- AND X-BAND ANTENNA SUBSYSTEM

1. Elements

The S- and X-band Antenna Subsystem (SXA) consists of:

(a) LGA.

(b) HGA with S-band and X-band feeds.

(c) Associated waveguides and coaxial cables.

It should be noted that the HGA reflector and LGA mass is included as part of the Structure Subsystem.

2. Operation

a. LGA. The low-gain antenna consists of an RF probe in an open-ended piece of circular waveguide. The radio signal is routed through the circular waveguide linearly polarized. Phase shifters at the output end of the circular waveguide convert the linearly polarized radio signal from the probe into a right-hand circular polarized signal for transmission to Earth. An external beam shaper provides the required hemispherical pattern.

b. HGA. The HGA consists of a 5-m furlable parabolic reflector and a feed structure that allows transmission and reception of S-band right-hand circularly polarized radiation. The antenna uses a cassegrain dual-polarization feed with a dichroic-hyperbolic reflector at X-band and a focal-point feed at S-band. The hyperbolic reflector is a reflector at X-band, but it is RF-transparent at S-band. The feed also allows for transmission of both left- and right-hand circular polarized X-band signals from two sources.

E. POWER SUBSYSTEM

The Power Subsystem provides a basic voltage of 28 Vdc to all subsystems. Each subsystem can then employ any dc-ac-dc converters/regulators required. A block diagram of the Power Subsystem is shown in Fig. 4-5.
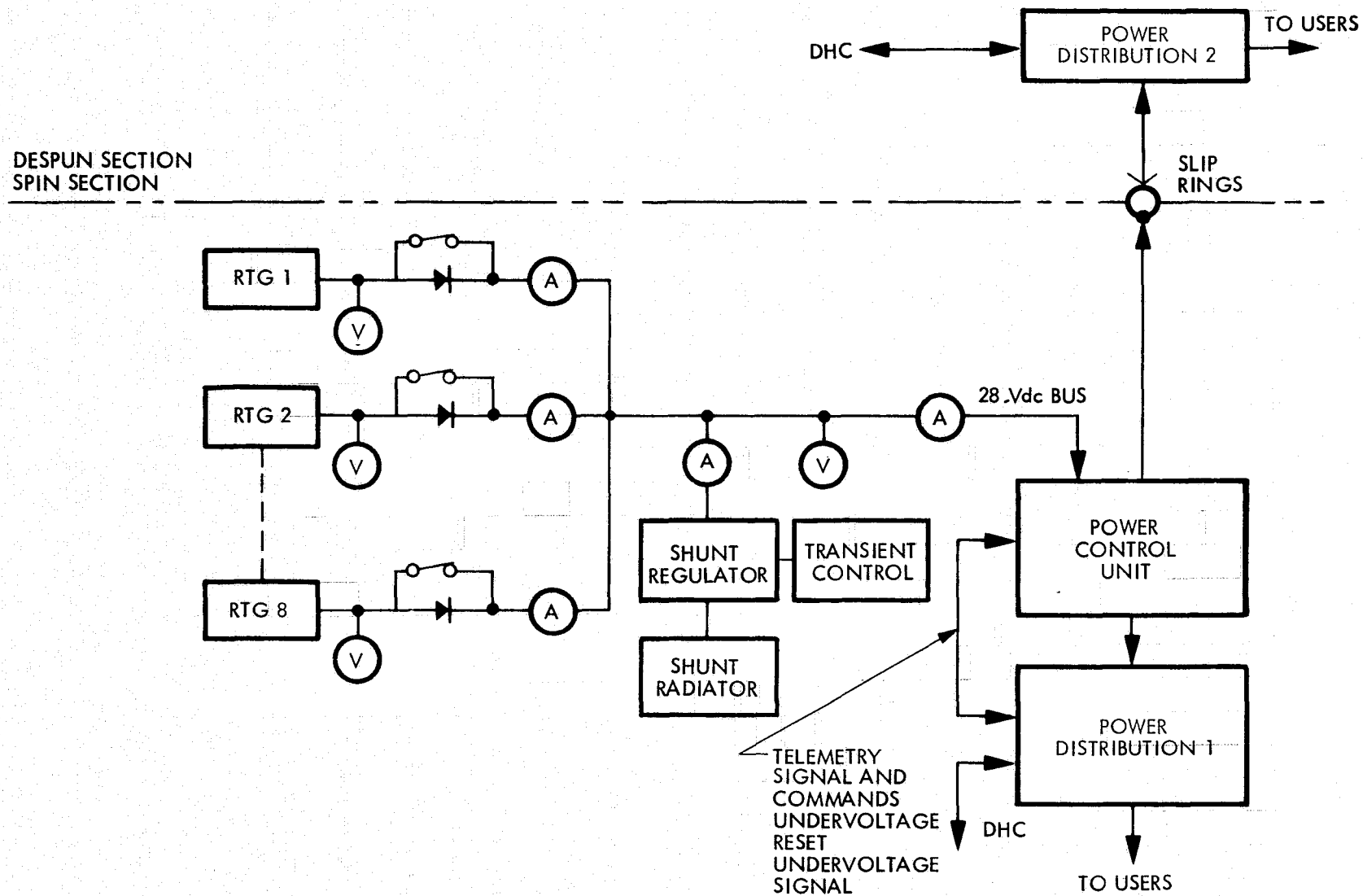
Fig. 4-5. Power Subsystem (simplified diagram)

1. Elements

   The Power Subsystem consists of the following elements:

   (a) RTGs.
   (b) Power control unit (PCU).
   (c) Direct current bus regulator.
   (d) Power distribution units.
   (e) Shunt radiator.

2. Operation

   a. RTG. The RTG is a modular selenide thermoelectric converter system being developed by the Nuclear Research and Application Division of the Energy Research and Development Administration (ERDA). The total power required from the RTG set is 400 W. The specific power density goal is 6.6 W/kg, and the efficiency goal, 10% (or better). The voltage output will be 30 Vdc at full load. The assumed number of RTG modules in parallel to provide a total of 400 W is eight, four each in two assemblies on two booms.

   b. Power Control Unit. The PCU receives power from the RTGs (or the power support equipment during ground testing) via isolation diodes. The PCU includes switching for short circuiting the RTGs during ground testing. The RTGs shorting function is accomplished via a relay quad and is controllable only from the power support equipment. The PCU functions also include under-voltage sensing, the switching for bypassing the RTG isolation diodes, and conditioning circuits for telemetry measurements of power system voltages and currents. The undervoltage sensing circuit detects low-voltage conditions on the 28-Vdc regulated bus and disconnects a selected set of loads, if an out-of-tolerance condition exists. Resetting of the disconnect can occur in response to a DHC command.

c. <u>DC Bus Regulator</u>. The dc bus regulator provides regulation of the 28-Vdc bus. Shunt regulation is employed to dissipate the majority of excess RTG power in a shunt radiator. The design is based on the MJS'77 approach.

d. <u>Shunt Radiator</u>. The shunt radiator is composed of etched Inconel resistive elements on a Kapton blanket which is in turn mounted on an aluminum honeycomb substrate. The design is based on the MJS'77 approach.

e. <u>Power Distribution Unit</u>. The power distribution units receive regulated power from the RTGs via the PCU and distribute power to the engineering and science subsystems on the spinning and despun sections. Two power distribution units are provided, one for the spinning and one for the despun sections. Four slip rings are assigned to carry power to the PCU on the despun section. Control of the power distribution relays is provided by the DHC remote terminals.

f. <u>Transient Control</u>. The power subsystem employs a capacitor energy storage capability which is used to augment RTG power during transient overload conditions of up to a 4-Ws demand. The design is based on MJS'77.

F. DATA HANDLING AND CONTROL SUBSYSTEM

The Orbiter functions of command and telemetry acquisition and formatting are provided by the DHC. The DHC employs distributed computing/processing architecture whereby telemetry, command, and data processing functions are distributed among "standard" microprocessor modules which are located both within the DHC and at the user locations on both the spin and despun sections of the Orbiter. The microprocessor modules are connected via an intercommunications bus system. Figure 3-1 (a) and (b) shows a functional block diagram of the DHC.

1. Elements

The DHC is made up of the following elements:

(a) Terminal modules which in turn contain a selectable set of stand-
     ard input/output (I/O) devices. Representative I/O devices are:

    (1)   Pulse output.
    (2)   Level output.
    (3)   Serial data-word I/O.
    (4)   Parallel data word I/O.
    (5)   Pulse and bilevel input.
    (6)   Frequency counter.
    (7)   Adjustable frequency generator.
    (8)   Local analog multiplexer.
    (9)   High-rate buffered DMA input.
    (10) High-rate buffered DMA output.
    (11) Analog-to-digital converter.

(b) High-Level Modules
(c) Intercommunications bus system.
(d) Control processor.
(e) Data handler.
(f) Remote terminal.
(g) Direct access adapter.
(h) Central timing unit.

2. Operations

Both inter- and intra-DHC communications are handled on the inter-
communications bus system (IBS). With reference to Fig. 3-1, high-level
modules are the control processor, data handler, their associated spare, and
the direct access adapter. External to the DHC are remote terminals and ter-
minal modules. Each science instrument will interface with a terminal module.
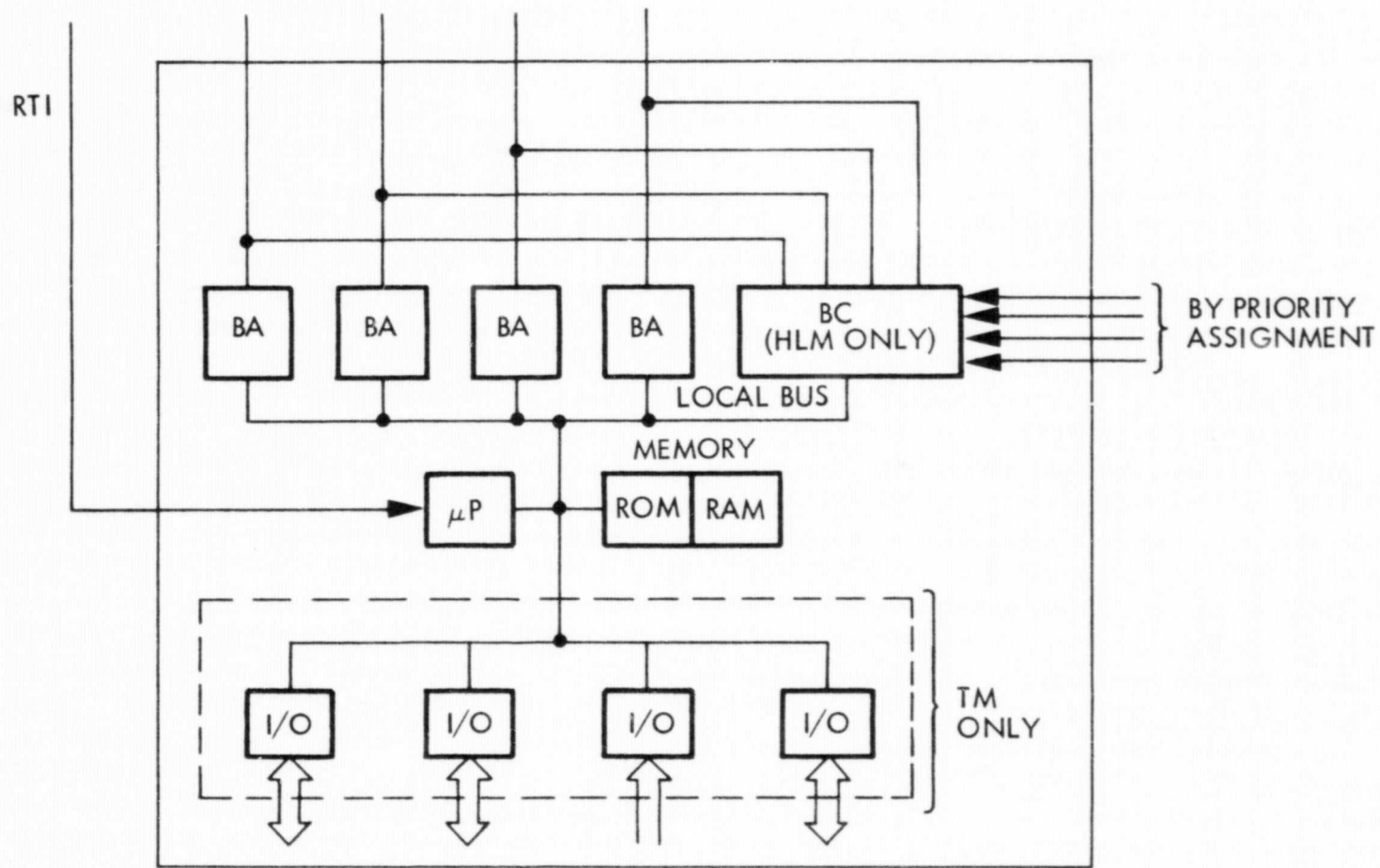
The architecture of the DHC is standardized on all levels. A small set of building-block logic elements are used for I/O, processing, memory, and intercommunications. These elements are used to construct the modules described above. A standard hardware and software interface is employed for timing, commanding, and data transfer between the modules which make up the JOP system; a common executive language will be supplied along with a flight interface test set for use in a user's bench checkout equipment. (See Fig. 4-6.)

a. <u>Terminal Modules.</u> These modules are responsible for control and data gathering within a user subsystem. A terminal module contains a microprocessor memory and a set of I/O elements [see 1(a) above] which interface with the user's I/O signals. Only those I/O elements required by the user are included in a user's terminal module. Terminal modules interface with the other modules of the DHC in the following ways:

(1) A single real-time interrupt, which is common to all modules, is used for timing and synchronization.

(2) Each terminal module contains a passive interface (bus adaptor) to each of the intercommunication buses. Data words can be entered into or extracted from the memory of the terminal module computer via the bus using direct memory access (DMA) techniques. A terminal module can not initiate intermodule communications but can support DMA transactions into and out of its memory. Commands are received and data is returned through predetermined memory locations under DMA control by the high-level modules. Structured control is thus enforced, preventing unexpected interactions between modules, and also preventing each terminal module from influencing the higher-level modules in an unexpected fashion. The terminal module can be accessed by several buses simultaneously. The module outputs are not affected for any composite DMA access rate below a specified worst-case value.

b. <u>High-Level Modules.</u> These modules are responsible for coordinating the processing which is carried out by the various users' terminal

Fig. 4-6. Typical microcomputer module

BA - BUS ADAPTER
BC - BUS CONTROLLER
μP - MICROPROCESSOR
TM - TERMINAL MODULE

RTI - REAL-TIME INTERRUPT
HLM - HIGH-LEVEL MODULE
ROM - READ-ONLY MEMORY
RAM - RANDOM-ACCESS MEMORY

modules and the DHC remote terminals, and for control of the IBS. A high-level module is like a terminal module in that it contains a microprocessor and memory, but, in addition, a high-level module contains a bus controller. The controller gives the high-level module the capability of controlling the intercommunications bus and thereby controlling all command and telemetry functions that take place on board the Orbiter. A high-level module only communicates over the intercommunications bus and does not contain I/O circuitry other than its connections to the IBS.

The unique mechanism contained within the high-level module is the bus controller circuit which can move data between memories of all modules connected to the bus system. The computer within the high-level module activates its bus controller by specifying which of several buses to use and by presenting the controller with the address of a table within memory which specifies the source, destination, and length of the requested transfer. The controller initiates and controls the specified transmission and signals the high-level module when it is completed.

Using the bus controller, the high-level module can direct the transfer of data from within any internal memory location for a specified source module, to a specified set of contiguous locations within one or more other terminal modules.

c. The Intercommunication Bus Structure. This structure is the redundant vehicle for communicating between all elements of the DHC. The IBS consists of several independent serial buses, each of which has the capability to handle data (telemetry or command) at a 1-Mb/s rate. Each bus is connected to one passive DMA interface (bus adapter) in each of the high-level, remote terminals, and terminal modules. Each bus is assigned a primary bus controller whose high-level module has complete control over that bus, but which can relinquish control over it under two conditions: (1) when it is not powered, and (2) when its processor specifically releases the bus to a lower priority controller for a designated time interval. Thus, the set of buses may be operated simultaneously with each one controlled by a different high-level module or with individual buses time-shared between several such modules.

Access to each bus by the various high-level modules is based on a fixed hardware priority assignment between controllers. A hardware structure is utilized for each bus to establish this priority. Modules of higher priority signal release of the bus via the priority hardware which then activates the hardware necessary to allow bus access within modules of lower priority.

The individual buses within the IBS are physically independent, and, therefore, no central bus system controller exists as a potential catastrophic failure mechanism. The number of buses within the IBS may be selected to meet mission redundancy and data requirements. The concept facilitates reconfiguring throughout a mission, if and when failures occur. In the extreme, a single remaining bus can support essential functions of the system.

d. <u>Control Processor.</u> The control processor is responsible for the coordination of Orbiter processes and is the central timekeeper aboard the Orbiter. Ground commands, received from the standard transponder, are decoded within the control processor. Command messages received by the processor are handled in a manner specified within each message, and are (1) dispatched immediately to a specified subsystem for execution or (2) are integrated within the processor for execution at a specified time or under specified Orbiter conditions. Ground command messages are typically specified at a high level and result in a sequence of simpler command messages being generated within the control processor and sent to subsystems.

The control processor counts time and, using the direct memory access bus system, places the next time count into a designated memory location of every user subsystem terminal module at the beginning of each time synchronization interval. Similarly command messages are sent to designated terminal modules using the bus system, by placing the message into the command buffer of the user subsystem terminal module.

e. <u>Data Handler.</u> The data handler directs the transmission of information between modules by establishing and controlling the periodic movement of data between the various modules. Buffered information within

the various modules is directed in the appropriate timed sequence to the TMS to form a telemetry stream for downlink transmission. The buffered information for data storage is directed in a similar manner to the DSS terminal module for storage on the tape recorder. The data handler directs the playback of data from the DSS for downlink transmission.

f. **Remote Terminal Units.** Remote terminal units (RTUs) provide interfaces with both sections of the Orbiter. The RTU on the despun section of the Orbiter provides interfaces with the PYRO, TEMP, PWR, TMS, and RFS. The RTU on the spin section interfaces with the Probe, PYRO, PWR, and TEMP. Two or more terminal modules tied together are required to handle all the functions of an RTU as well as to provide redundancy protection against single point failure. An RTU performs the following functions:

(1) It gathers analog and digital engineering data and buffers the collected information within the RTU memory for periodic extraction by the data handler in forming the various telemetry formats.

(2) On the basis of commands received from the control processor, it provides discrete outputs to various subsystems connected to it onboard the spacecraft.

(3) The RTU receives the downlink telemetry streams from the bus (under control of the data handler, which places this information in memory buffers), performs rate buffering and supplies this information to the TMS for downlink transmission.

g. **Direct Access Adapter.** The direct access adapter, or test high-level module, is for testing purposes only. This module is powered by support equipment and will be disabled in flight. The direct access adapter performs the following functions:

(1) It buffers the high-speed bus data for communication to and from the system test support equipment.

(2) It allows direct access communication to the subsystems connected to the bus system.

h.   Central Timing Unit.   The central timing unit provides the basic timing and synchronization signals to the various Orbiter subsystems. This unit provides two square-wave frequencies: (1) a 1 MHz clock, and (2) a 400-pulse/s real-time interrupt. The central timing unit is internally redundant and supplies two lines for each of the two timing signals.

Units which require different frequencies will contain count-down logic to generate these frequencies internally. An adjustable frequency generator chip (modulo N divider) will be available for this purpose [see 1(a)].

G.   ATTITUDE AND ARTICULATION CONTROL SUBSYSTEM

Orbiter attitude control is based on the tendency of the angular momentum of the spinning section to stabilize the spin axis in inertial space. The direction of the angular momentum changes only with the addition of momentum in another direction. The time integral of disturbance torques is a source of such additional momentum and thus causes a directional drift or precession of the spin axis. The stored momentum is large enough that the effect of integrated disturbance torque, even over many days, is small. Periodically, pairs of the 22-N (5-lbf) thrusters are pulsed to provide momentum to cancel the effects of disturbance torques. The Sun sensor, mounted on the despun section, provides the information to make these corrections. To ensure stability of the spin axis, the moment of inertia of the spinning section about the spin axis is larger than that of any transverse axis of the composite vehicle. Damping to the stable condition is provided by a passive "hoop" damper on the spinning section. During the Earth-Jupiter transit phase of the mission, the two Orbiter components spin as a single unit, which has the effect of extending bearing lifetime. Periodically during transit, the bearing assembly is activated to exercise the mechanical elements of the bearing assembly.

The attitude control thruster array consists of four 22-N (5-lbf) thrusters spaced 90 deg apart. Each thruster axis lies in the plane (see location of the small rocket engines in Fig. 3-2). Thrusters 1 and 3 provide counterclockwise spin control whereas thrusters 2 and 4 provide clockwise spin control. Any of the pairs 1-2, 2-3, 3-4, or 4-1, together with the damper, can provide precession control. Because the hoop damper has a long time constant (>100 s),

settling following a precession firing can be long. Shorter settling can be obtained by a series of properly timed precession firings.

1. Elements

The Attitude and Articulation Control Subsystem (AACS) consists of the following elements:

(a) Redundant sun sensors.

(b) Star scanner.

(c) Despun bearing assembly.

(d) Scan platform gyro assembly.

(e) Controller (located on the spin section).

(f) Terminal module (located on the despun section).

(g) Balance actuators.

(h) Passive nutation damper.

(i) Scan actuator.

A simplified functional block diagram is shown in Fig. 4-7.

2. Operation

a. Redundant Sun Sensors. The sun sensor is mounted on the despun part of the Orbiter. Its function is to sense the roll axis deviation from the desired spin direction (usually the Earth line), and to provide appropriate control signals to the AACS during cruise for cone angle control.

b. Star Scanner. The star scanner is mounted on the spinning section. A pulse is issued when a star crosses its field of view. Successive pulses for the reference star determine the spin rate of the spinning element. The sensor provides the roll error signal to the AACS during cruise for spin control.

c. Despun Bearing Assembly. The despun section of the Orbiter is coupled to the spinning section through the despun bearing assembly. This assembly contains slip rings, redundant motors, and relative position
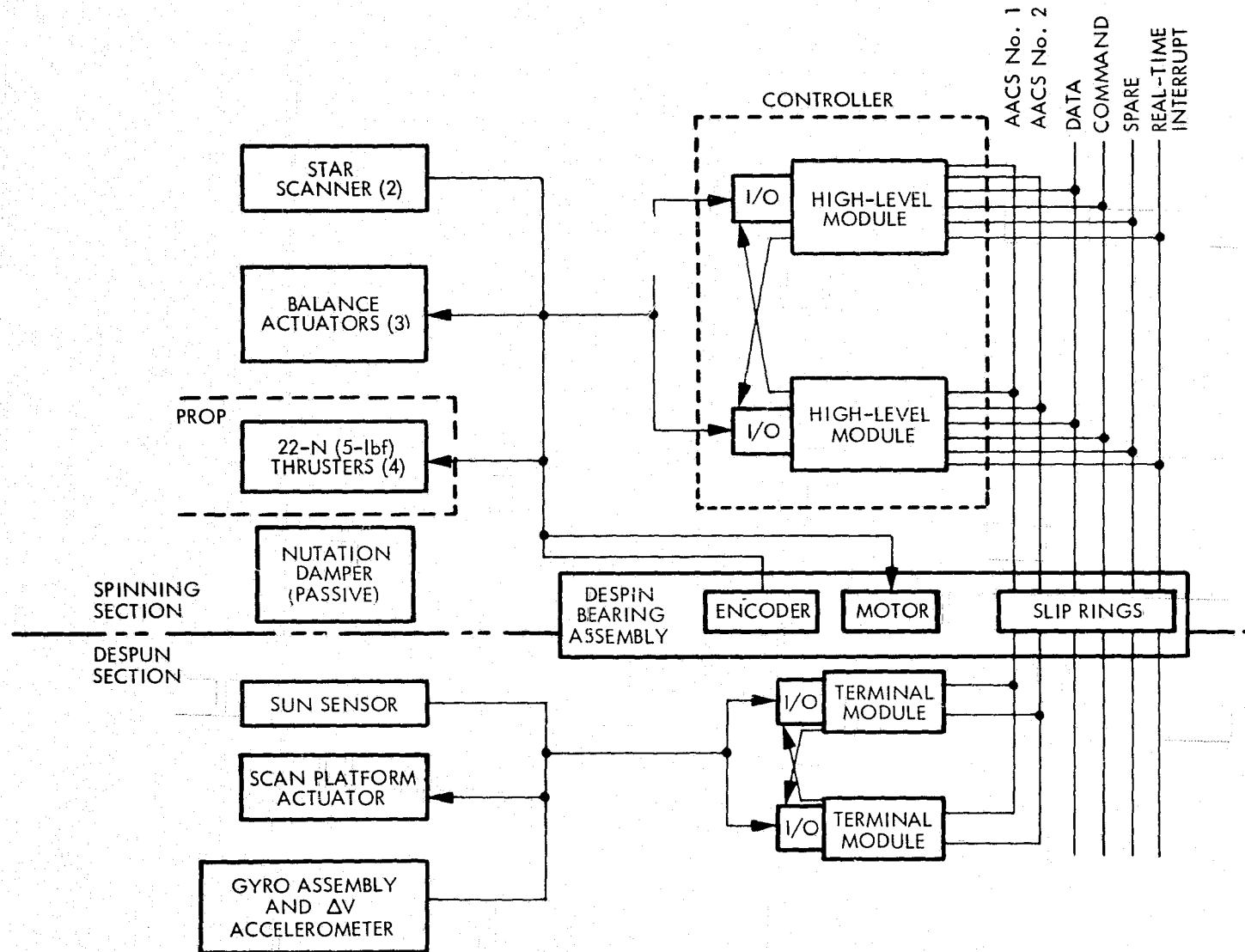
Fig. 4-7. Attitude and Articulation Control Subsystem

encoders. Relative spin between the two Orbiter sections is provided by the motors. The control electronics derives the despun position by reading the encoder when the star scanner sweeps past the reference star. Drive to the despun bearing motor is based on the difference between the measured and commanded positions of the despun part of the Orbiter.

d. <u>Scan Platform Gyro and ΔV Accelerometer Assembly</u>. The gyro package located on the despun sections contains a two-degree-of-freedom gyro. Scan platform rate and position information is provided to the controller by the gyro package. The gyro signals measure platform pointing errors induced by spin-axis wobble and despin bearing assembly jitter. These error signals together with the encoder and the feedback elements of the scan-platform actuator are used to meet the scan-platform pointing requirements. The function of the ΔV accelerometer is to sense Orbiter acceleration during ΔV maneuvers to provide a signal to shutdown the rocket engines when the required ΔV has been imparted to the Orbiter.

e. <u>Controller (High-Level Module)</u>. The controller consists of redundant microcomputers with appropriate special purpose I/O capability. The attitude and articulation control data buses allow the computers to communicate, through remote terminal modules, with elements of the subsystem on the despun section. The controller processes signals from the Sun sensors, star scanner, encoder, and gyro assembly and appropriately commands the bearing motor, scan actuators, balance actuators, and spin and precession thrusters.

f. <u>Terminal Modules</u>. The terminal modules are microcomputers, located in the despun section, which permit communication between the high-level module on the spinning unit, and the Sun sensor, scan-platform actuator, gyro assembly, and ΔV accelerometer on the despun section.

g. <u>Balance Actuators</u>. Inexact ground balancing and uneven depletion of propellant can produce wobble of the spin axis. This behavior can be reduced by vernier control of the spacecraft mass properties (e.g., CG location, moments-of-inertia). It is accomplished by small changes in the RTG and

magnetometer boom locations. The deployment mechanisms for these elements are designed to include balance actuators to accommodate these small changes in closed-loop fashion in response to wobble errors sensed by the two-degree-of-freedom gyro.

h. <u>Passive Nutation Damper</u>. This is a hoop-configured device which is partially filled with fluid. Its plane is normal to the spin axis, and its center lies slightly off the spin axis. Whenever nutation is non-zero, the dynamics of the hoop couples with the spacecraft dynamics to reduce the nutation angle.

i. <u>Scan Actuator</u> Clock orientation of the scan platform is obtained by relocating the despun section of the spacecraft relative to the spinning component via the despun bearing assembly. Cone orientation is accomplished by a single degree of articulation relative to the despun section. Cone orientation is driven by a simple actuator. Scan platform position information is provided to the controller by the gyro package, the despun bearing assembly encoder, and the feedback element of the cone actuator. Continuous rapid scan platform maneuvers will cause the spin axis to precess. This can be minimized by discontinuous, properly timed stepping of the platform.

H. PYROTECHNICS SUBSYSTEM

1. <u>Elements</u>

The Pyrotechnics Subsystem consists of the following elements:

(a) Squibs which provide for the actuation of electro-explosive release devices, pinpullers, cable cutters, and valves.

(b) Pyrotechnic switching units (PSUs) for power conditioning and energy storage for initiating squibs.

(c) Propulsion actuation unit which operates the 445-N thruster valves and the propellant isolation assembly latching solenoid valves in response to commands received from the DHC remote terminal.

A functional block diagram of the Pyrotechnic Subsystem is shown in Fig. 4-9. Pyrotechnic events and devices are required by the Orbiter, as follows:

    (a) Solid rocket motor jettison, 4.

    (b) Unlatch de-spun portion of the Orbiter, 1.

    (c) Scan platform deployment, 1.

    (d) Magnetometer boom release, 1.

    (e) RTG deployment, 2.

    (f) Unlatch and deploy relay antenna, 1.

    (g) Unlatch and deploy high-gain antenna reflector, 1.

    (h) Separate Probe from Orbiter, 3.

    (i) Separate Probe umbilical, 1.

    (j) Actuate, upon command, pyrotechnic valves in the Propulsion Subsystem, 9.

2. Operation

Since pyrotechnic events occur on both sides of the despun bearing assembly, separate PSUs are provided for the despun and the spin sections of the Orbiter. With the rocket engines located on the "spin" side, the PSU may be combined with the propulsion actuation unit as one package for the spin section. The pyrotechnics are prevented from being fired until after a pre-separation signal is sent from the IUS.

I. PROPULSION SUBSYSTEM

The propulsion subsystem is an all-bipropellant configuration which provides the required impulse to:

    (1) Perform the ΔV maneuvers, including JOI.

    (2) Adjust the spin rate of the Orbiter.

    (3) Generate the torque to precess the Orbiter about its spin axis.

PYROTECHNIC ARM
SIGNAL

(IUS)

PWR DISTRIBUTION 2 — 28 Vdc

CMD EXCITATION VOLTS

DHC
REMOTE
TERMINAL

SQUIB FIRE COMMANDS

PSU TELEMETRY

PSU

TO SQUIBS
(≈6 BRIDGEWIRES)
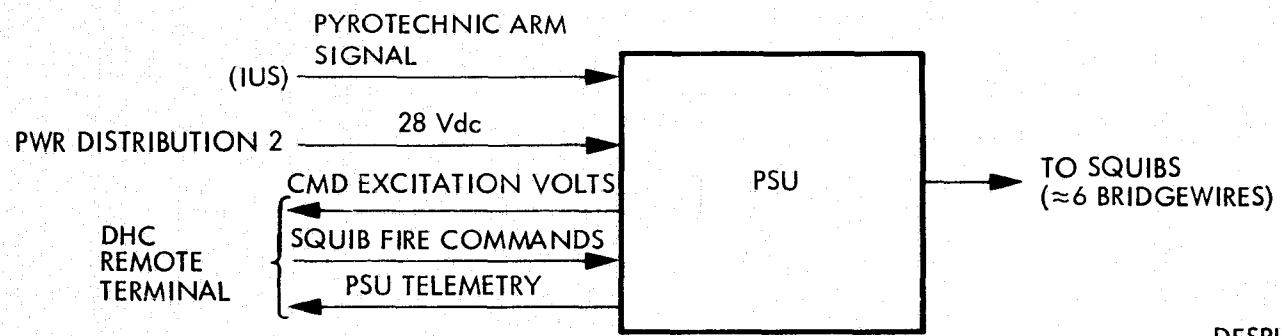
DESPUN SECTION

SPIN SECTION

PYROTECHNIC ARM SIGNAL
(IUS)

PWR DISTRIBUTION 1

28 Vdc

CMD
EXCITATION VOLTS

SQUIB FIRE
COMMANDS

PSU TELEMETRY

PSU

TO SQUIBS
(≈42 BRIDGEWIRES)

DHC
REMOTE
TERMINAL

28 Vdc

VALVE CMD EXCITATION
VOLTS

VALVE COMMANDS

VALVE SWITCH TLM

PAU

TO PROP ENGINE VALVES (4)

TO PIA SOLENOID VALVES (2)
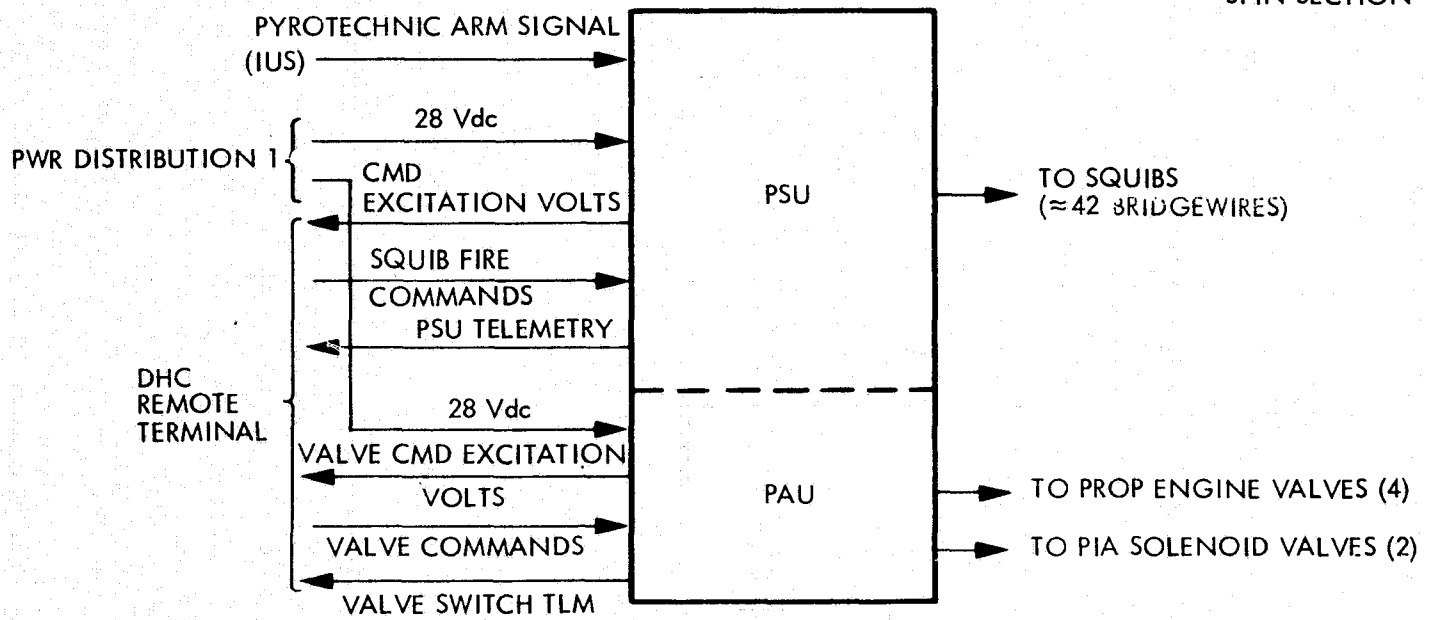
Fig. 4-8. Pyrotechnics Subsystem and interface

Impulse is delivered by four 445-N (100 lbf) and four 22-N (5 lbf) rocket
engine assemblies (REA). Orbiter ΔV maneuvers will be performed using one
pair of the 445-N REAs. The redundant pair serves as a backup in the event
of a failure in the primary pair. Attitude adjustments of the Orbiter and alter-
ations in spin rate will be accomplished with the 22-N REAs. These rocket
engines are configured in pairs to form opposing couples allowing the rate of
spin to be increased or decreased, as desired. Precession of the Orbiter spin
axis will be achieved by operating pairs in a pulse mode.

Consideration has been given to using previously developed hardware where-
ever possible to minimize costs. Although a mass penalty of 20 kg results, use
of the VO 75 pressurant tank is being considered based on cost.

1. Elements

The Propulsion Subsystem which, with its supporting structure,
comprises the propulsion module, consists of the following elements:

(a) Bipropellant and pressurant tanks.
(b) Pressurant control assembly.
(c) Propellant isolation assembly.
(d) Four 445-N (100 lbf) thrusters.
(e) Four 22-N (5 lbf) thrusters.
(f) Required plumbing, valves, filters, and pressure and temperature
transducers.

A schematic diagram is shown in Fig. 4-10.

2. Operation

Table 3-2 includes a mass breakdown of the Propulsion Subsystem for
the dual spin Orbiter. The following assumptions were used in determining
the propellant requirements.

VO75 PCA

FUEL   FUEL

OXID   OXID

R

F

NC

NO

NC

NO

NC

NC

HE

PIA   PIA

THRUSTERS: 22 N   22 N   445 N   445 N   445 N   445 N   22 N   22 N

VENT AND RELIEF

R   PRESSURE REGULATOR

F   FILTER

M   MANUAL VALVE

SOLENOID VALVE

CHECK VALVE

CAPPED PORT

P   PRESSURE TRANSDUCER

X   POSITION INDICATOR

NC   PYRO VALVE (NORMALLY CLOSED)

NC   PYRO VALVE (NORMALLY OPEN)
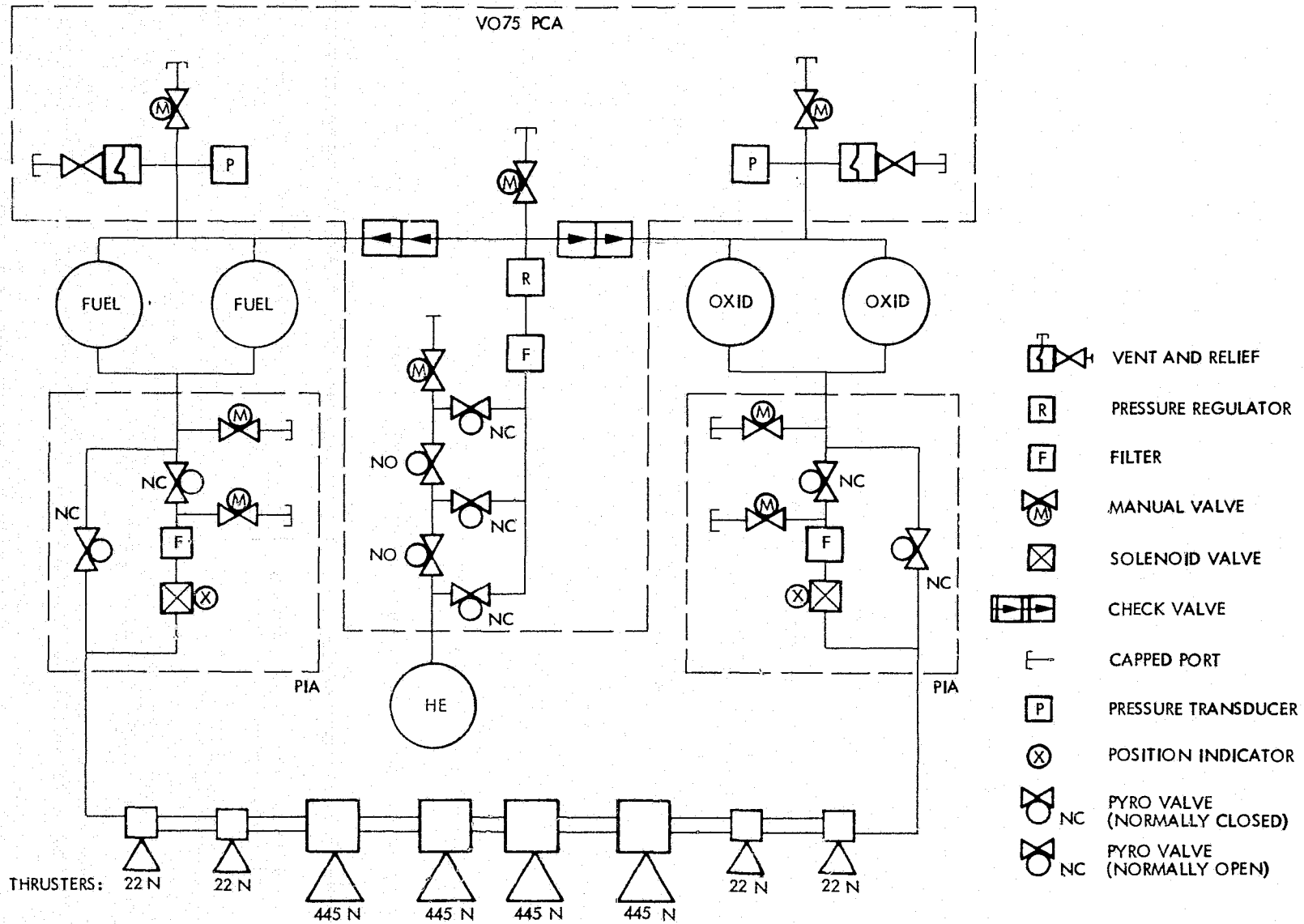
Fig. 4-9. Propulsion Subsystem

(a) A total ΔV capability of 2000 m/s of which only a 25 m/s midcourse correction maneuver would be made with the Probe attached, all other maneuvers occurring after Probe separation.

(b) Ten kilograms of propellant were allocated for attitude control functions during the mission.

(c) The specific impulse (I_s) for the 2746 Ns/kg is 280 s based on the Marquardt R-4D expected performance.

(d) Assumed total injected spacecraft mass of 1290 kg, including a 400-kg mission module and a 150-kg probe.

a. Bipropellant and Pressurant Tanks. The propellants, nitrogen tetroxide (N_2O_4) and MMH, are contained in four equal-volume tanks, each of which has an internal device which provides slosh damping control of the propellants. The tank outlets are configured so that the centrifugal spin forces position the propellants for gas-free expulsion during rocket engine firings. Weight estimates are based on a separate spherical storage tank from VO 75 for the helium pressurant.

b. Pressurant Control Assembly. Flow control, regulation, and isolation of the pressurant gas is accomplished by a VO 75 pressurant control assembly.

c. Propellant Isolation Assembly. Propellant control, filtration, and isolation is provided by two VO 75 propellant isolation assemblies.

d. Thrusters (445-N). A candidate for the 445-N REA is the Marquardt R-4D, previously used on the Apollo and Lunar Orbiter programs. The availability of R-4Ds makes them, potentially, low-cost.

e. Thruster (22-N). No 22-N thruster has been identified as being presently available. However, two possibly useable designs are currently in development.

f.   **Plumbing, Valves, Filters, and Transducers.**   Mariner-based designs will be used, employing stainless-steel tubing, welded connections between components, pyrotechnic and solenoid valves of flight-tested design, and flight-proven pressure and temperature transducers.

J.   DATA STORAGE SUBSYSTEM

The DSS consists of the NASA standard 4.5×10^8-bit tape recorder which will interface with a DHC terminal module.

K.   RELAY RADIO TELEMETRY SUBSYSTEM

The Relay Radio Telemetry Subsystem (RRT) provides the data link from the Probe to the Orbiter during Probe entry.

1.   Elements

The RRT consists of the following elements:

(a) A relay receiver.
(b) A relay telemetry unit.
(c) A relay antenna.

2.   Operation

a.   **Relay Receiver.**   The relay receiver obtains the RF signal received from the Probe transmitter through the relay antenna.   The received signal is amplified by an RF amplifier with a 4-dB noise figure.   This signal is mixed with a 370-MHz local-reference signal, and the difference signal at 30 MHz is further amplified and mixed with a second reference to a difference frequency of 3 MHz.

The output of the second intermediate frequency (IF) amplifier is passed to dual filters each with a 35-kHz bandwidth.   One filter is at the mark frequency, and the second at the space frequency.   The filter outputs are

squared, and the difference signal provides the binary data stream (plus noise) which is sent to the relay telemetry unit for bit synchronization and bit detection.

b. <u>Relay Telemetry</u>. The relay telemetry unit obtains bit synchronization and detects the data bits that are routed by the DHC to the TMS for real-time transmission and to the DSS for storage and later transmission. The relay telemetry unit may be based on the NASA standard transponder command detector design.

c. <u>Relay Antenna</u>. The relay antenna consists of a pointable parabolic dish 1-m in diameter to provide reception of radio signals transmitted by the Probe during Probe entry. The orientation will be optimized to maximize the link performance during specified periods of the entry.

APPENDIX A

IMAGING SCIENCE CAPABILITY

This description of a reference JOP 81/82 Imaging Science Subsystem is included to provide imaging science experiment proposers with information on the baseline subsystem capability provided by the Project as an Orbiter facility and planned for this mission. It is important for experiment proposers to understand that the described capability is tentative and that close interaction between the Project and the Imaging Science Team will be necessary to define the best final configuration of the instrumentation from the standpoint of both cost-versus-science return and total cost. This information indicates the range of characteristics that can be considered as a basis for final instrument definition.

The proposal of a single-camera instrument incorporating a new class of solid state, silicon detectors called charge-coupled devices (CCD) is a departure from previous Mariner-Viking camera systems. The CCD has a number of advantages over the selenium-sulfur vidicons previously used. CCDs provide greatly increased quantum efficiency and response extending into the near infrared (see Fig. A-1). Very low CCD preamplifier read-out noise allows much higher sensitivity and dynamic range. The CCD has a fixed geometry that is invariant with time. The CCD also has been proven to possess a linear-response characteristic, easily normalized by a simple slope and offset correction on an element-by-element basis.

An additional improvement over past cameras is increased high frequency response. With the CCD sensor, the system response at the sampling frequency (33 line pairs/mm) including the optics, is expected to be about 30 percent. The spatial frequency given for the CCD is based upon a format of 800 x 800 elements with 0.6 mil center-to-center spacing between photoelements, which is the present sensor-design goal.

The proposed basic framing camera will consist of a shutter and filter wheel of the type used in recent Mariner and Viking Orbiter imaging instruments. The filter wheel will hold from six to eight filters. (The filter spectral and polarization properties will be defined by the Imaging Science Team.) The proposed optics are the 1500-mm optical system first developed for the MVM'73 imaging instruments. These optics are an all-spherical, catadioptric

Cassegrain design with all elements manufactured from fused silica. The optics are radiation resistant.

The sensor will be radiatively cooled to an anticipated -40°C to reduce dark current buildup to about 50 electrons/s for each pixel. The sensor will also be radiation-shielded to allow useful imaging at closest jovian approach, although with potentially increased dark current and noise.

Sensor responsivity and system star-detection capability are shown in Figs. A-1 and A-2.

Table A-1 is a summary of the performance characteristics of the CCD array coupled with the Mariner 1.5-m focal length optics.

Table A-1. Performance characteristics

Parameter	CCD
Resolution	10 μrad/pixel
Focal length	1.5 m
Spectral range	0.35 - 1.1 μm
Noise equivalent exposure (broadband)	0.15 $\mu J/m^2$ [a]
Readout noise	~25 electrons rms/pixel [a]
Dark current	~50 electrons/s for each pixel [a]
Readout time	~2 s
Data encoding	8 bits/pixel
Filter wheel	6 to 8 filters
Format size, pixels	800 x 800 [a]
Readout data rate	2.5 Mbs
Exposure times	5-ms minimum (upper limit TBD)

[a] Estimated
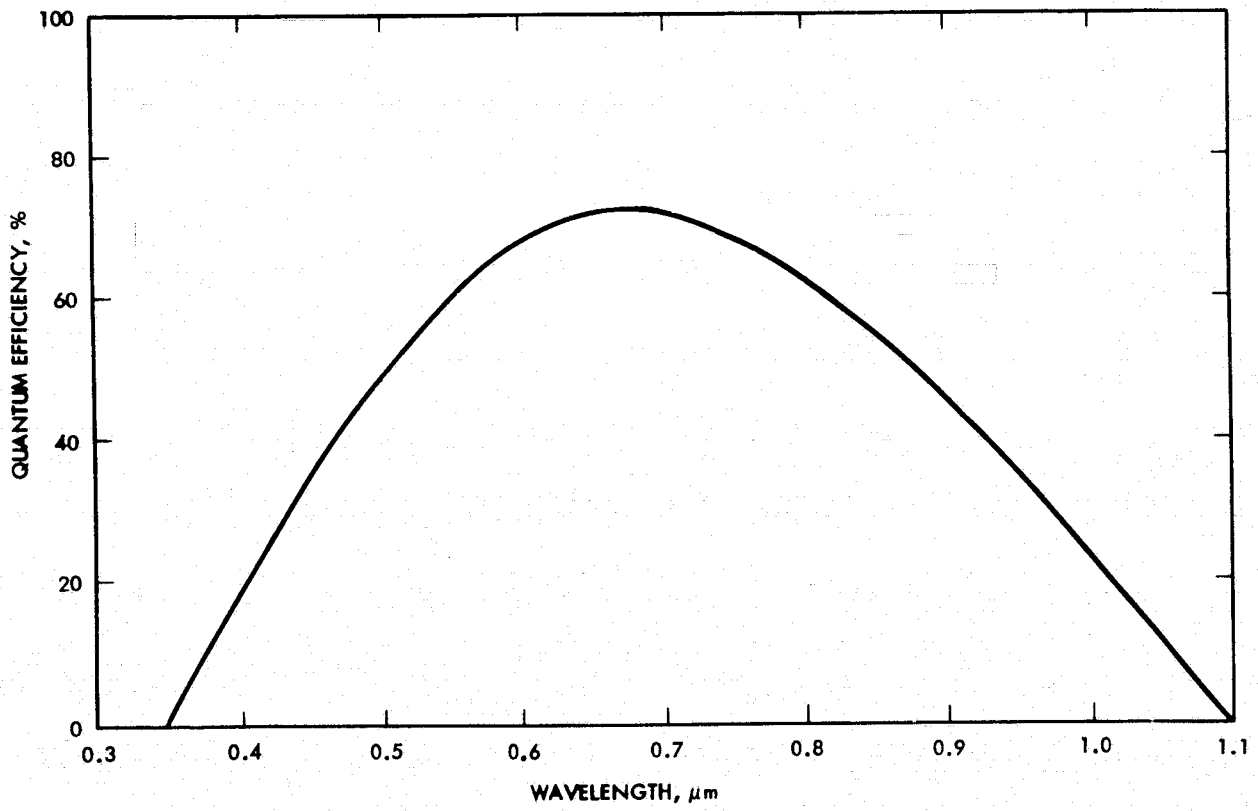
Fig. A-1. Estimated CCD spectral response
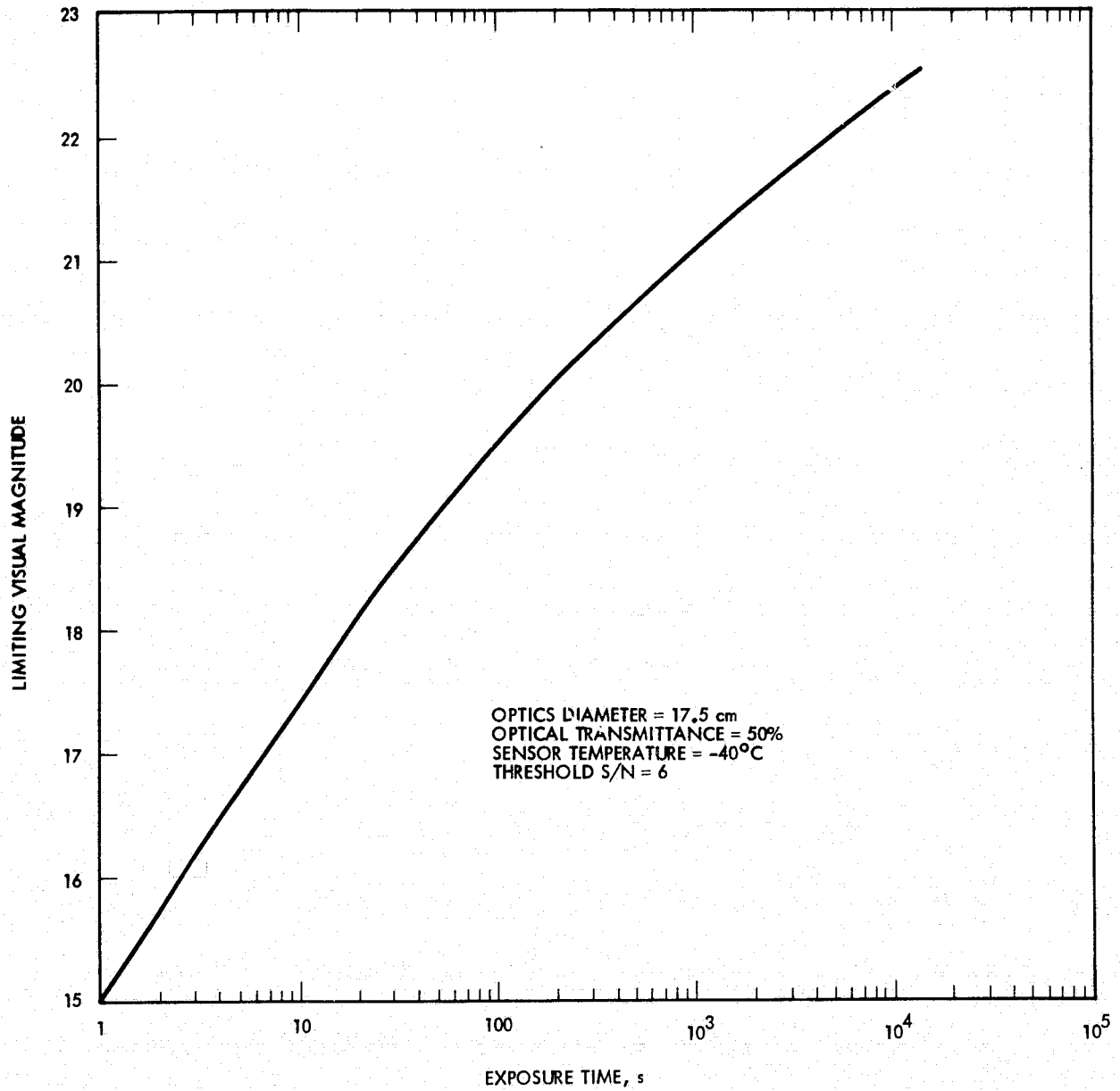
Fig. A-2. Estimated star detection capability

APPENDIX B

RADIO SCIENCE CAPABILITY

A. INTRODUCTION

A description of the Jupiter Orbiter Probe 1981/1982 (JOP 81/82) communication system is included in this appendix to provide potential radio science/celestial mechanics experiment proposers information on the equipment which will be available for use in conducting the proposed experiments. It is important that experiment proposers understand this communication system because it is provided by the Project as an Orbiter facility.

The main functions of the communication system are to: (a) provide tracking-data angle, doppler, and ranging; (b) provide a command uplink between the command bits, received from operations, to the command decoder on the Orbiter; (c) provide a telemetry downlink between the Orbiter data system and the ground telemetry decoders.

The communication system, illustrated in Fig. B-1, consists of the Orbiter equipment, the ground equipment, and the communication media. This description will emphasize the Orbiter mounted equipment with only a brief description of the ground stations.

The Deep Space Network (DSN) includes three subnets; two 26-m subnets, and one 64-m subnet. The 64-m subnet has both S- and X-band receive capability, whereas the 26-m subnets have S-band receive capability. All subnets have S-band transmit capability. The DSN tracking system includes the angle tracking, planetary ranging assembly, and doppler extractor. This system obtains radiometric data, i.e., angles and one and two-way doppler and range, and transmits the data to the Mission Control. The DSN telemetry system receives, decodes, records, and retransmits the engineering and scientific data to Mission Control. The DSN command system accepts coded signals from Mission Control and transmits them to the Orbiter to update the Orbiter state. The DSN provides both phase-coherent and open-loop receivers. The phase-coherent receivers are used in the performance of the tracking, telemetry, and command functions, and the open loop receivers, in conjunction with the recording capability, can be used for occulation and other experiments.
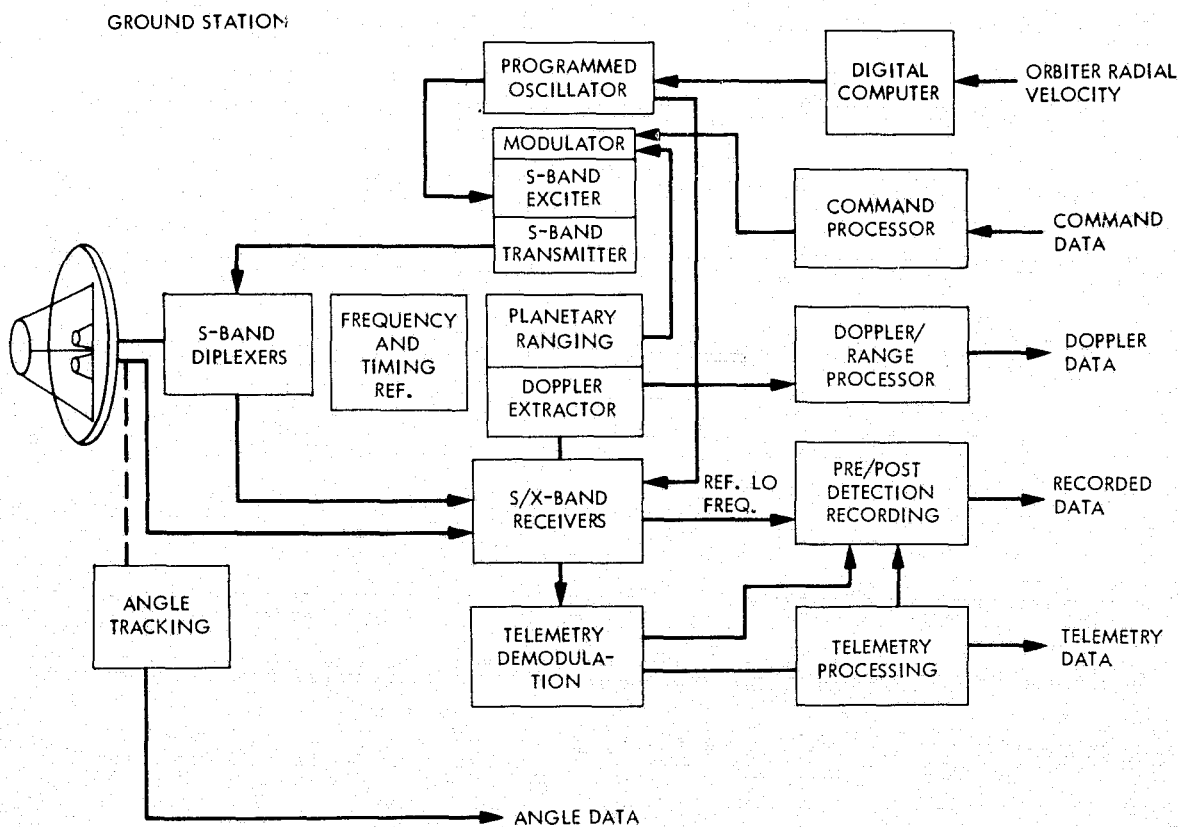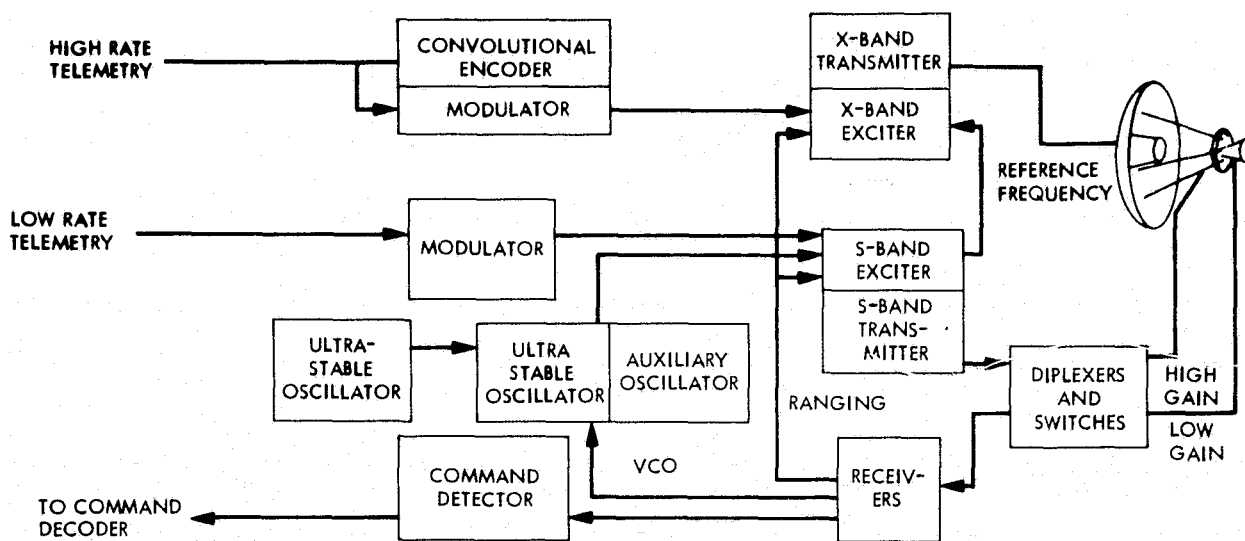
Fig. B-1. JOP 81/82 communications

B.   ORBITER RADIO

The radio frequency subsystem (RFS), illustrated in greater detail in
Fig. B-2, consists of two antennas, redundant NASA standard transponders,
S- and X-band power amplifiers, and associated μ-wave switches, diplexers,
cabling, and control circuits.

1.   Antennas

a.   High Gain Antenna.   The high gain antenna (HGA) consists of an
unfurlable paraboloid main reflector (5 m in diameter, when deployed), an
X-band cassegranian feed, and a focal-point S-band feed.   The X-band casse-
granian feed consists of a dual polarization [left- and right-hand circular (LHC
and RHC)] feed horn with a dichroic hyperboloidal subreflector.   The subreflec-
tor reflects at X-band and also allows transmission of the S-band signal from
the focal-point feed.

The antenna allows reception and transmission of right-hand cir-
cularly polarized S-band signals.

Polarization is right hand circularly when operating with X-band
transmitter one (TWTA 1) and left hand when operating with X-band transmitter
two (TWTA 2).   Table B-1 tabulates the HGA parameters.

b.   Low Gain Antenna.   The low-gain antenna is a printed circuit ver-
sion of a cavity-backed turnstile.   The antenna is mounted on the HGA subrefelc-
tor and the radiation axis coincides to that of the HGA.   The antenna receives
and transmits right-hand circularly polarized radiation at S-band.   The radiation
pattern approximates a cardioid of revolution and provides forward hemisphere
radiation.   The on-axis gain is approximately 7 dBi and the -3 dB beamwidth is
approximately 60 deg at 2295 MHz.   This antenna has very little value on the
downlink after launch plus 120 days because of its low gain.
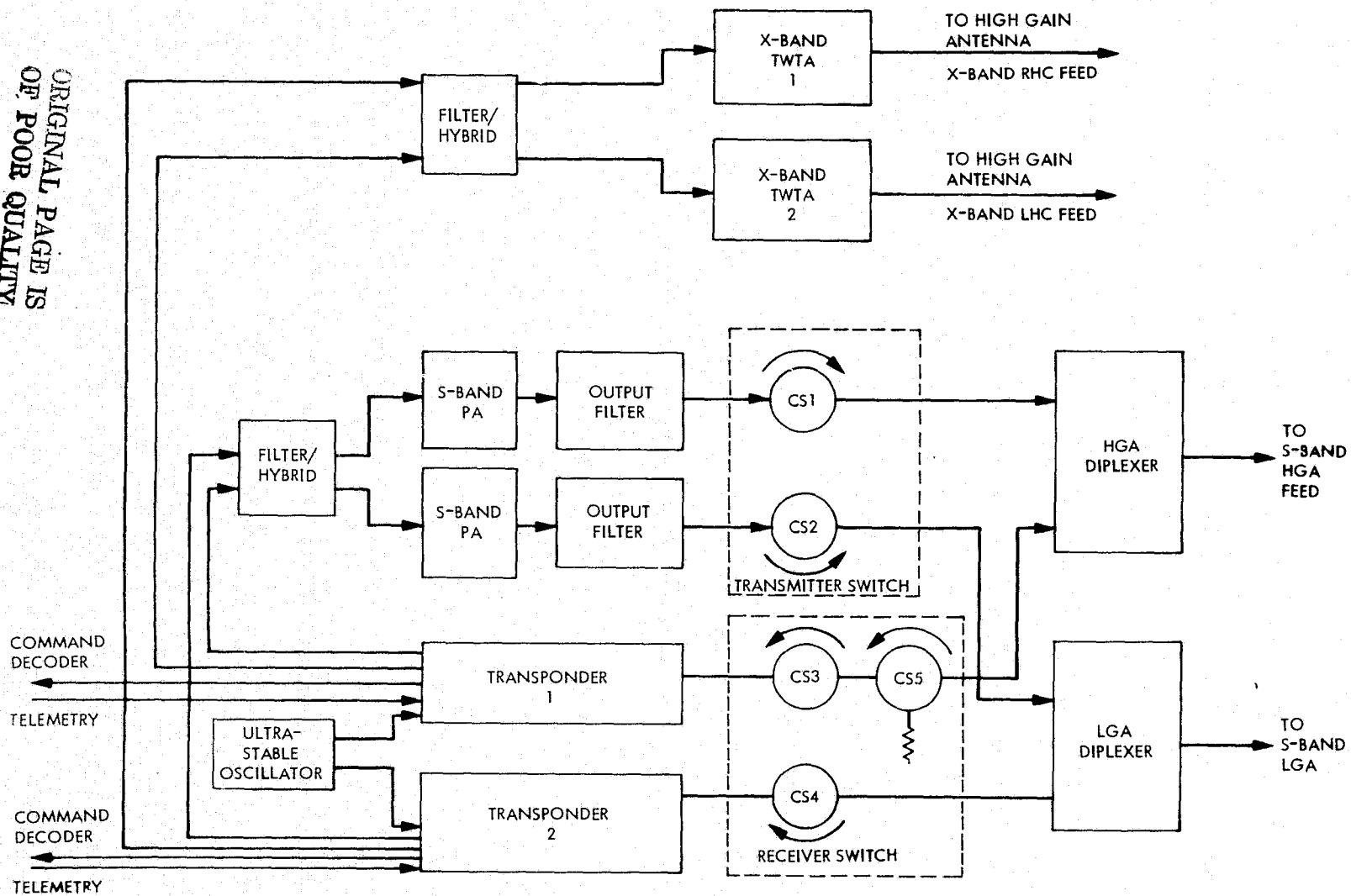
Fig. B-2.   Radio Subsystem block diagram

Table B-1. HGA parameters

Frequency	Gain, dBi	3-dB beamwidth, deg
2115 MHz (S-up)	38	2.0
2295 MHz (S-down)	38.6	1.8
8415 MHz (X-down)	50	0.48

2. Transponder

Each transponder, illustrated in Fig. B-3, consists of a phase-coherent receiver, S-band exciter, X-band exciter, and an auxiliary oscillator.

a. Receiver. Each receiver is connected to a ferrite circulator switch which connects to the HGA diplexer or the LGA diplexer. The normal circulator state has receiver one to the HGA and receiver two to the LGA with only one receiver powered. The third circulator ports are connected, thus allowing the receiver to be switched to either antenna.

The receiver is a dual-conversion superhetrodyne narrow-band (18 Hz at threshold), second-order phase-locked loop mechanization. The first intermediate frequency (IF) is about 124 MHz, the second IF is 12.25 MHz. Automatic gain control (AGC) is applied to the first and second IF amplifiers. The noise bandwidth through the second IF to the ranging channel is 8 MHz, whereas the IF bandwidth preceding the command detector is 250 kHz. A crystal filter with a 3-kHz bandwidth precedes the AGC detector and the loop-phase detector. The overall phase-lock-loop noise bandwidth is 18 Hz at threshold. The receiver performance parameters are tabulated in Table B-2.

b. S-Band Exciter. The S-band exciter can be driven by either receiver VCO, by the auxiliary oscillator, or by the ultra-stable oscillator. The receiver will automatically switch and use the VCO as a frequency source for the exciters when in lock. This switch can be overridden by ground command to prevent disturbance of the downlink frequency and phase during uplink acquisition. The auxiliary oscillator or ultra-stable oscillator (USO) is
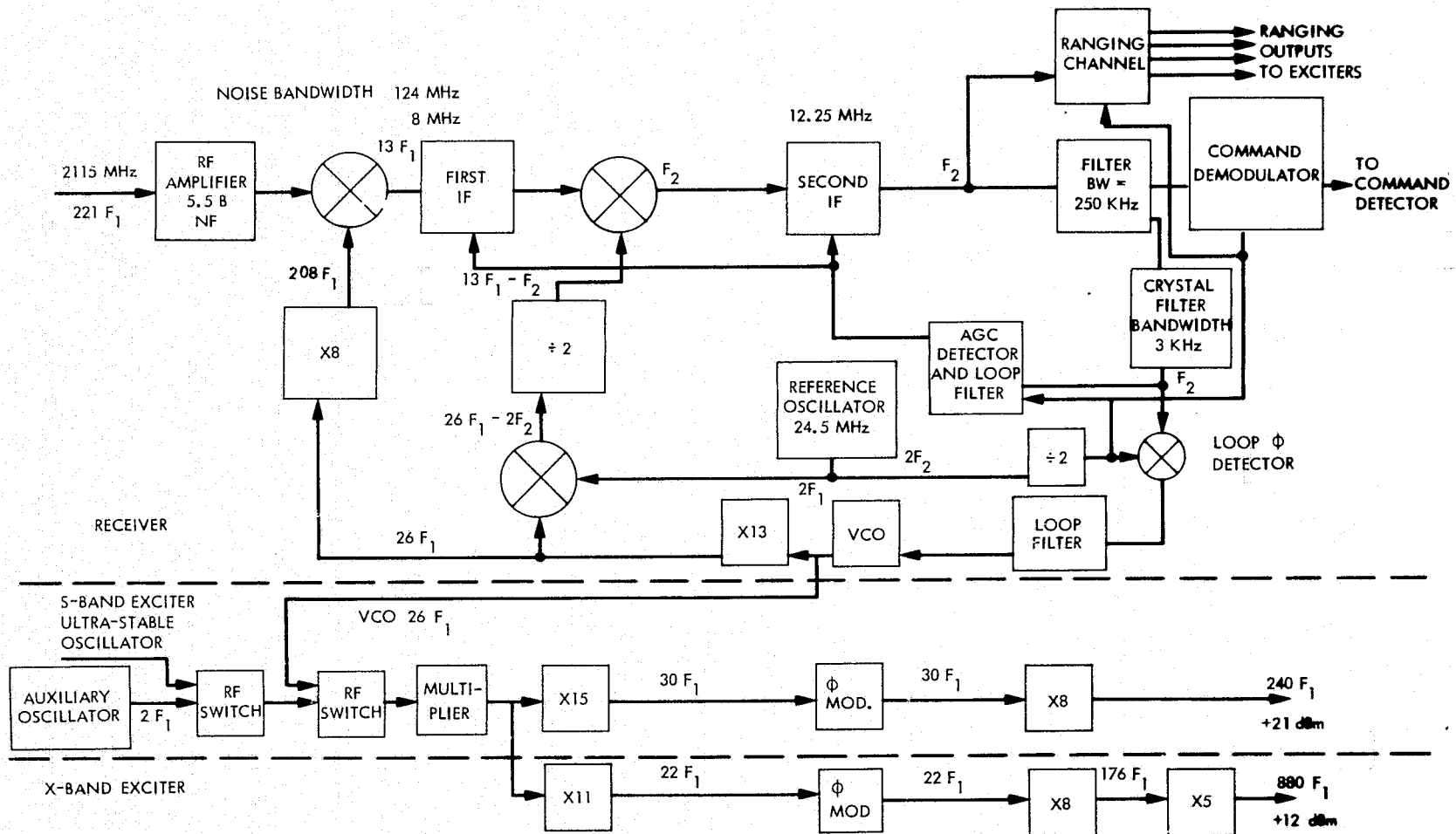
Fig. B-3.   Transponder functional diagram

Table B-2. Standard S/X-band transponder characteristics

Parameter	Value
Frequency	
Uplink (Fu)	One channel out of 29 in allocated band from 2110 to 2120 MHz
Downlink (coherent operation)	
S-band	$F_D = 240/221$ F uplink
X-band	$F_D = 880/221$ F uplink
Channel separation	
Uplink	$F_{SU} = 341,049$ Hz
Downlink	$F_{SD} = 370,370$ Hz (three major channels have 3.3-MHz spacing)
Downlink (one way)	
S-band	$F_D = 120$ F auxiliary oscillator
X-band	$F_D = 440$ F auxiliary oscillator
Noise figure (receiver)	5.5 dB
Noise temperature (receiver)	739 K (No = -169.9 dBm/Hz)
Predicted system temperature	840 K (No = -169.4 dBm/Hz)
Loop-noise bandwidth ($2\beta_{LO}$)	18 Hz ±1.8 Hz
System threshold (0 dB S/N in $2\beta_{LO}$)	-156.4 dBm
Tracking range	±150 kHz at -120 dBm
Capture range (no modulation)	±1.5 kHz at -120 dB
Tracking rate	400 Hz/s at -120 dBm (30 deg ϕ error)
Ranging-channel bandwidth	
Lower 3 dB frequency	1 kHz
Upper 3 dB frequency	3 MHz
Range delay (S and X)	≤ 20 ns peak to peak
Differential S/X range delay	≤ 1 ns
Carrier ϕ delay	≤ 2.5 ns peak to peak
Difference S-band ϕ delay versus range delay	≤ 12.5 ns
Command channel bandwidth	
Lower 3 dB frequency	18 Hz at ≈ -152 dBm / 800 Hz at ≈ -120 dBm
Upper 3 dB frequency	≈ 40 kHz
Telemetry channel	
Frequency response	20 kHz to 3 MHz ±1 dB
Linearity	1% to 2 rad
Sensitivity	1 rad/V
Stability	±5%

automatically selected when out of lock. The choice of either auxiliary oscillator or USO is by ground command. The stability of each of the frequency sources is tabulated in Table B-3. The S-band exciter provides frequency multiplication [to 30 F_1 phase modulation (telemetry and ranging, if present)] and final frequency multiplication to 240 F_1 ($\simeq$ 2297 MHz) with an output level of 21 dBm.

c. X-Band Exciter. The X-band exciter uses the X-11 component of the first S-band multiplier (X 15). This signal is phase-modulated (telemetry and ranging, if present) and multiplied in frequency to 880 F_1 ($\simeq$ 8422 MHz) with an output level of 12 dBM.

3. Transmitters

a. S-Band Transmitters. The S-band signal from each exciter is coupled to a 3-dB hybrid. The two outputs of the hybrid provide drive to two 5-W S-band solid-state power amplifiers. The two-power amplifier outputs go to the three-port ferrite circulator switch, one amplifier connected to the HGA diplexer and the second to the LGA diplexer. The third ports are connected together allowing either power amplifier to be switched to either

Table B-3. Orbiter frequency-source stability

Source	Time Period			
	1 min[a]	10 min[b]	12 hr[c]	5 years
Auxiliary Oscillator	±2 in 10^{10}		±16 in 10^6	
Ultra-Stable Oscillator		±2.5 in 10^{11}	1 in 10^{11}	±1 in 10^6

[a]Constant temperature.

[b]Over FA temperature.

[c]1-s, integration time +5°C to +25°C

diplexer. The two diplexers connect to the HGA and LGA, allowing transmission/reception at S-band on HGA or LGA. One S-band power amplifier is on throughout the mission.

b. X-Band Transmitters. The X-band signal from each exciter is coupled to a 3-dB hybrid; the two outputs from the hybrid provide drive to the X-band TWTA. The output of each TWTA (power level of 15 W or 25 W) are connected by waveguide to the two sections of the HGA X-band dual-polarization feed. One TWTA is turned on when required for high rate telemetry or when dual frequency operation is required.

C. RADIOMETRIC PERFORMANCE

The current best estimate of the radiometric data performance for the combined Orbiter/DSN Communication System is shown in Table B-4. The ranging channel will have a bandwidth of approximately 3 MHz centered at 12.25 MHz. This signal is coherently detected, filtered, amplified, and sent to the modulator for retransmission to the ground at both S- and X-band. The detector continues to operate with the receiver out of lock, but its usefulness is limited by the frequency drift of the VCO.

Table B-4. Closed loop radiometric performance

Parameter	3-σ accuracy, m
S- and X-band integrated doppler (12 h)	1.1
S- and X-band differenced integrated doppler	0.13
Absolute group delay	3.3
S- and X-band group delay stability (12 h)	1.1
S- and X-band differenced group delay	0.7
S-band differenced integrated phase/group delay	1.6

Note: The ground station has the capability to digitize and record the output signal of open-loop receivers during occultations. Right- and left-hand components of S- and X-band can be received, digitized, and recorded simultaneously.

APPENDIX C

INFRARED SPECTROMETER CAPABILITY

This description of a reference JOP 81/82 Infrared (IR) spectrometer is included to provide IR experiment proposers with information on the subsystem capability that may be available for use as a Project-provided facility in conducting their experiments. The basic configuration of the JOP IR spectrometer is expected to be very similar to the Modified Infrared Interferometer Spectrometer (MIRIS), which is presently under construction for the MJS'77 mission. Its characteristics are summarized in Table C-1. It is important that experiment proposers understand that the described capability is tentative and that close interaction between the Project and the IR Science Team will be necessary to define the best final configuration of the instrumentation from the standpoint of both cost-versus-science return and total cost.

The MJS'77/MIRIS is a combination of two Michelson interferometers and a radiometer. One of the interferometers is optimized for the far IR $50-600 \text{ cm}^{-1}$) and the other for the near $(1000-7000 \text{ cm}^{-1})$. The broadband radiometer measures visible and infrared radiation over the range 0.3 to $1.25 \mu\text{m}$. The two interferometers share the Michelson motor, a neon reference interferometer, and substantial fractions of electronic circuitry.

IR radiation is collected by a Cassegrain telescope 50 cm in diameter. The field of view (FOV) of the interferometers and the radiometer is 0.15 deg, which is determined by an aperture at the focal plane of the telescope. In addition, a small mirror (~2 mm in diameter) with the FOV pointed at 20 deg off the telescope boresight (primary FOV) is used to calibrate the instrument by observing the Sun.

An off-axis port is required for solar viewing, because the detectors and instrument optics are extremely sensitive and the telescope boresight must not be pointed closer than 20 deg from the Sun. Pointing will be determined by the scan-platform pointing for both fields of view.

The choice of detectors is based on the desire to measure the expected low-level signals from the outer planets (see Fig. C-1). The instrument employs four detectors; one in the far IR interferometer, two in the near IR interferometer, and one in the radiometer. The far IR interferometer and the

Table C-1. MJS MIRIS system parameters and specifications

	Mode 1	Modes 1 and 2 combined	Mode 2
Spectral range, cm^{-1}			
Far IR		50 - 600	
Near IR		1000 - 7000	
Radiometer		8000 - 30000	
Spectral resolution, apodized, cm^{-1}			
Far IR	6.6		1.7
Near IR	6.6		2.0
Resolved spectral intervals, apodized			
Far IR	83		323
Near IR	909		3000
Interferogram time, s	45.6		141.6
Frame time, s	48		144
Reference wavelength, μ		0.5852	
Reference frequency, Hz		160	
Telescope			
Field-of-view, deg		0.15	
Active diameter, cm		50.8	
Word rate, words s^{-1}		80	
Words per interferogram	3648		11328
Bit rate, b/s		1120	
Frequencies in data channel, Hz			
Far IR		0.24 - 2.81	
Near IR		4.68 - 32.8	
NER, Wcm^{-2} sr^{-1}/cm^{-1}			
Far IR, at 420 cm^{-1}	1×10^{-9}		2.0×10^{-9}
Near IR, at 2000 cm^{-1}	2×10^{-11}		4.0×10^{-11}
Operating temperature, K		140 ± 1.0	
Weight, kg		25.0	
Power, average, W		17.2	
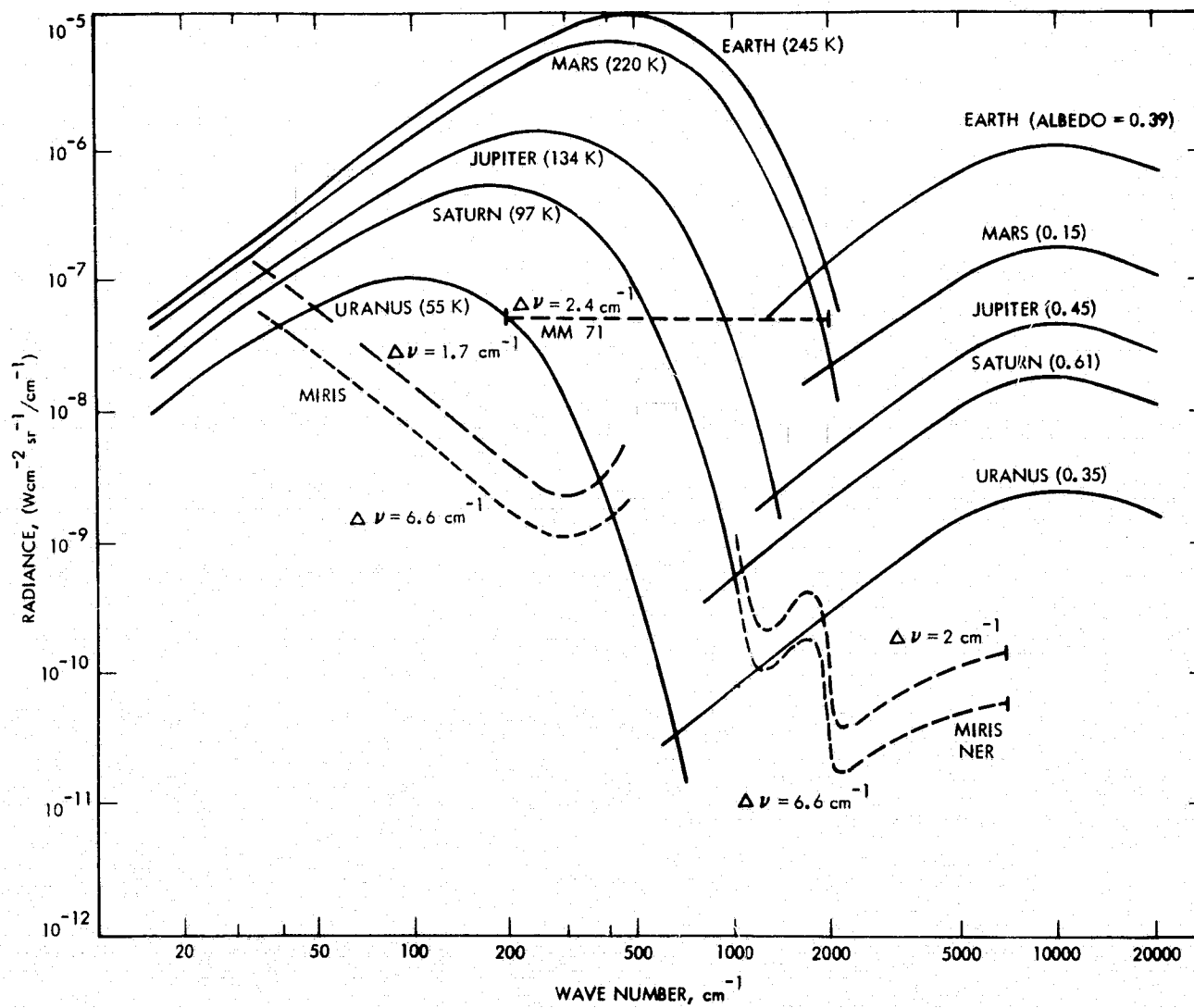
Fig. C-1. Radiance of outer planets

radiometer employ thermopile detectors. High sensitivity is needed in near IR, which is achieved by two Hg Cd Te detectors, optimized for response at 5 μm and 9 μm, respectively.

The MJS'77/MIRIS is designed to accept command timing and reference frequency signals from the Flight Data Subsystem (FDS) on the MJS'77 spacecraft. Because this capability will not be available on Jupiter Orbiter, a minicomputer command subsystem may have to be included with the MIRIS.

A neon reference interferometer is used to phase-lock the speed of pathlength variation in the two interferometers to a clock signal from the FDS. The reference interferometer will comprise a Michelson interferometer suitable for observing the 0.5852-μm line of neon. It will include a monochromatic neon source, with suitable detector, beamsplitter, and compensator.

The MIRIS interferometers may be operated in two modes to select spectral resolution; Mode 1 is a low-resolution mode (6.6 cm^{-1}, apodized) with a 48-s interferogram, and Mode 2 is high-resolution mode (1.7 and 2 cm^{-1}, apodized) with a 144-s interferogram.

A system of temperature sensors, electrical heaters, proportional heater controllers, radiators, and a thermal isolator will be used to maintain the optics module at 140 ±1.0 K with variations from the set point not to exceed ±0.01 deg per day.

A deployable cover of the metal beryllium will be provided to protect the optics surfaces of the telescope from particles and vapors during launch activity and from possible exposure to direct sunlight during "sun-acquire" or other maneuvers. The cover will be deployed not less than four days after launch.

The inflight calibration of the instrument is performed by using several techniques. The long wavelength end of the interferometer is calibrated by viewing deep space. The short wavelength end of the interferometer and the radiometer are calibrated by viewing a diffuse plate (mounted on the Orbiter) illuminated by the Sun. Observations of stable IR stars can also be used as

independent primary calibrations for the instrument, although this technique is time consuming. Calibrations by observing the Sun through the 20 deg off-axis port are of lesser quality and will be used to span the long time between the major calibrations.

The MJS'77/MIRIS design takes into consideration both the Jupiter radiation environment and the MJS'77 science objectives at Jupiter and the Galilean satellites. The instrument has, however, been optimized for observations at Saturn and Uranus. Appropriate modifications may be made to optimize the JOP/MIRIS performance at Jupiter.

APPENDIX D

ENVIRONMENTAL DESIGN REQUIREMENTS
(INTERIM)

A. INTRODUCTION

1. Document Purpose

These JOP 81/82 interim environmental design requirements are intended to assist the scientific instrument proposer(s) in the preparation of their respective instrument proposal(s) in response to the JOP 81/82 Announcement of Opportunity (Ref. 1-1). It is felt that instruments "sized" to withstand these criteria will be environmentally adequate for the JOP 81/82 Mission.

2. Requirements Rationale

The requirements constitute conservative boundaries of probable natural and induced conditions to be encountered by the scientific instruments during the pre-launch, Space Transportation System (STS) launch, interim upper stage (IUS) flight and separation, cruise, Probe separation, and Orbiter mission phases.

The conservative aspect accounts for uncertainties in environmental conditions, in effects of configuration [spacecraft mated to IUS, and IUS mated to STS, instrument mounting (locations and orientations), and mission-peculiar sequences and events].

It is anticipated that some proposed instruments will incorporate exotic sensors, precise alignments, or fragile elements that absolutely can not survive some of the interim requirements. The instrument supplier and the Project should negotiate a suitable strategy for mitigating each specific hardship. This involves such techniques as mounting on isolators, temperature-control emphasis (blankets, louvers, heaters), notching vibration inputs over specific frequency bands, and reducing margins.

The requirements are derived from:

(a) Engineering judgment based on experience with similar items or "quick analyses."

   (b)   MJS'77 environmental design criteria.

   (c)   Space Shuttle System Payload Accommodations (Ref. D-1).

   (d)   NASA Payload Requirements for IUS[1].

B.   PRE-LAUNCH

The mating of the spacecraft and the IUS and installation in the cargo bay is shown in Fig. D-1. The ground operation sequence applicable to the cargo installation is shown in Fig. D-2.

1.   Handling Shock

During handling, payload equipment may experience a 20-g terminal sawtooth shock pulse of 11-ms duration in any direction (plus or minus).

2.   Hoist Acceleration

The hoist acceleration is 2-g vertical.

3.   Ground Transportation Vibration

The sinusoidal vibration requirement discussed in paragraph C-3 covers this condition.

4.   Temperature

It is assumed that the ground temperatures will be as described in the temperature requirement in paragraph C-7.

5.   Conditioned Air

The payload-bay purge system supplies conditioned air to the payload bay during prelaunch operations until 80 min prior to Orbiter cryogenics loading. At this time, gaseous nitrogen (GN2) will be supplied to provide an inert payload-bay atmosphere through lift-off. Two supply pressure levels

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[1]Marshall Spaceflight Center, unpublished document

SERVICE AT
EXPLOSIVE
SAFE FACILITY
(ESF) PAYLOAD

TRANSPORT TO
SPACECRAFT
ASSEMBLY AND
ENCAPSULATION
FACILITY (SAEF)

INTEGRATE
PAYLOAD AND
ENCAPSULATE

MATE
PAYLOAD
TO IUS IN
SAEF

EMPLACE CARGO
IN CANISTER

TRANSPORT TO
VERTICAL ASSEMBLY
BLDG. (VAB)

ERECT CANISTER
IN VAB

TRANSPORT
TO PAD

MATE CANISTER TO
PAYLOAD CHANGE-
OUT ROOM (PCR)

EXTRACT CARGO
FROM CANISTER
IN PCR

REMOVE
CANISTER

MATE PCR
TO ORBITER

INSERT CARGO
INTO ORBITER

DEMATE PAYLOAD
HANDLING
MECHANISM
(PHM)

Fig. D-1. Cargo installation (option 1)

SPACECRAFT CHECKOUT FACILITY
RECEIVING INSPECTION
PRESSURE LEAK TEST
FINAL ASSEMBLY
INTEGRATED SYSTEM TEST
SUBSYSTEM AND INSTRUMENT
TESTS
DSN INTERFACE TESTS

EXPLOSIVE SAFE FACILITY
LOAD PROPELLANT AND PYROS
TCA FIRING TEST
WEIGHT AND CG

PAYLOAD ASSEMBLY AND ENCAPSULATION FACILITY
INTEGRATE PAYLOAD
MATE TO THIRD STAGE AND SPIN TABLE
ELECTRICAL AND RF CHECKS
ENCAPSULATE PAYLOAD
MATE TO IUS
INTEGRATED SYSTEM TEST
INSTALL CARGO IN CANISTER
CONNECT SERVICE AND PREPARE FOR TRANSPORT

VEHICLE ASSEMBLY BUILDING
ROTATE CANNISTER TO VERTICAL POSITION

LAUNCH PAD
MATE CARGO TO PCR
INSTALL CARGO IN CARGO BAY
CARGO INTERFACE VERIFICATION
PAYLOAD READINESS TEST
COUNTDOWN PREPARATION
COUNTDOWN AND LIFTOFF

Fig. D-2. Ground operations sequence, cargo installation on pad (option 1)

are provided just upstream of the ground portion of the umbilical disconnect:

(a) The 2.5-psig system supplies either air or GN2 to the payload bay during all operations not involving cryogenic payloads.

(b) The 10.0-psig system supplies only GN2 and is used for cryogenic payloads only.

The purge gas will be either air or GN2. The gas will be class 100, guaranteed class 5000 (HEPA) filtered with 15 parts/10^6 or less hydrocarbons based upon a methane equivalent.

6. Contamination Control

a. **Volatile Condensable Material.** Material will be selected for low outgassing characteristics. Selection criteria will be 1% total mass loss and 0.1% volatile condensable material (VCM) as defined in the applicable specification[2].

b. **Spacecraft Assembly, Checkout, Integration.** A facility cleanliness of class 100,000 or better is assumed.[3]

During mating with the IUS, the payload must be maintained within an environment meeting a cleanliness level of class 10,000, as defined by Federal Standard.[4] After mating and during installation of the IUS/spacecraft into the Orbiter bay, contamination control will be as specified.[5]

The payload will be provided with all special handling equipment and protective covers required prior to IUS mating, and adequate protection will be provided, if any environment better than a facility cleanliness of class 100,000 or visibly clean is required.

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[2] JSC Specification SP-R-0022.
[3] As defined in Federal Standard 209B.
[4] Ibid.
[5] Payload Contamination Control Requirements for STS Induced Environment, July 22, 1975.

7. Humidity

A humidity of 0 to 70% and a temperature range of 5 to 45°C are assumed for a controlled condition. The instrument proposer should identify any instrument-peculiar restrictions on these conditions.

8. Access to Payload

The pre-launch Shuttle operations sequence will provide spacecraft operation verification at a time before launch which will be specified. Physical access to the payload will be available in the PAEF.

9. Caution and Warning

Data will be required from the payload to meet the caution and warning monitoring requirements of the Shuttle. Such data would include safe/arm status of pyrotechnic devices, etc.

C. LAUNCH, ASCENT STS, AND IUS

The ascent time sequence is shown in Fig. D-3.

1. Pressure

The Orbiter payload bay is vented during launch. The payload bay pressure history during ascent is shown in Fig. D-4. It is assumed that the spacecraft pressure time-histories follow those shown in Fig. D-4.

2. Acoustics

The composite of the internal noise environments generated by rocket engines during lift-off and that generated by aerodynamic turbulence during the boost phase is presented in Table D-1 as one-third octave band spectra. (These levels assume 0 dB attenuation by the contamination control shroud).

△ LIFTOFF

☐ SOLID ROCKET BOOSTER (SRB) BURNOUT

☐ SPACE SHUTTLE MAIN ENGINE (SSME) BURNOUT

☐ FIRST ORBITER MANEUVERING SYSTEM (OMS) BURN

☐ OPEN PAYLOAD BAY DOORS

☐ 2nd OMS BURN

☐ 3rd OMS BURN   ORBITER COAST PERIOD

ATTACHED PHASE CHECKOUT, REMOVE
RTG COOLING JACKETS, etc.   IUS GUIDANCE UPDATES

COMMIT TO IUS DEPLOYMENT (EARLIEST) △

DEPLOY OPERATIONS

IUS PREPARATION AND
DETACHED PHASE CHECKOUT

△
COMMIT TO
IUS IGNITION
(EARLIEST)

0   20   40   60   80   100   120   140   160   180

TIME, min

Fig. D-3.   Generalized launch to IUS ignition sequence

VENTS OPEN AT T + 10 s

3 - σ DISPERSION WITH HEAD WIND

EXPECTED
VALUE

SRB STAGING

3 - σ DISPERSION
WITH TAIL WIND

PRESSURE, psia

RANGE TIME, s
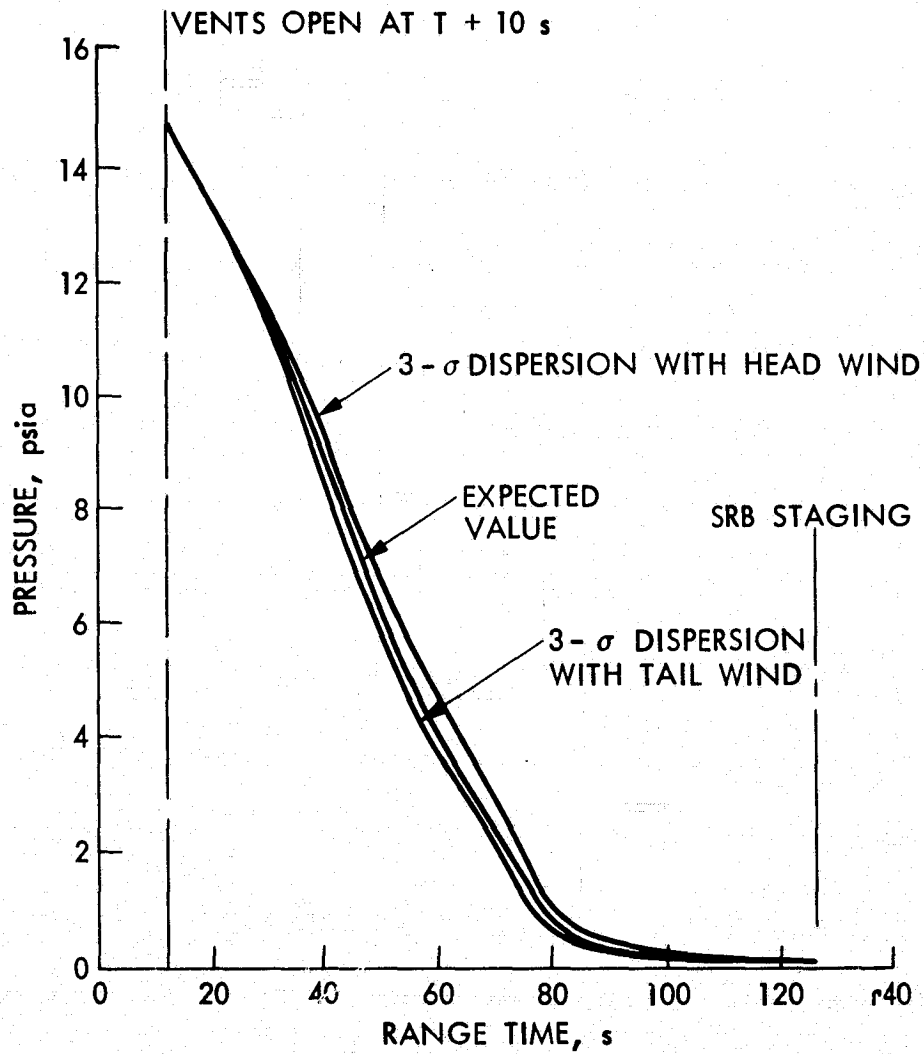
Fig. D-4. Orbiter payload bay internal pressure
histories during ascent

Table D-1. Acoustic design levels

One-third octave center frequency, Hz	Noise level, dB RE 20 N/M^2
12	118
16	120
20	122
25	124
31	126
40	128
50	130
63	132
80	133.5
100	135
125	136.5
160	137.5
200	138
250	138
312	137.5
400	137
500	136.5
630	135
800	133.5
1000	132
1250	130.5
1600	129
2000	127.5
2500	126.5
3120	125.5
4000	124.5
5000	124
6300	123.5
8000	123
10000	122.5
Overall	149

Duration: 3 min

3. Sinusoidal Vibration

Sinusoidal vibration design requirements for instruments are presented in Table D-2. These levels should be assumed at the instrument mounting points in any direction.

The sweep rate is 2 oct/min.

4. Random Vibration

Random vibration design requirements for scientific instruments are presented in Table D-3. These levels should be assumed at the instrument mounting points in any direction.

5. Static Acceleration

The design acceleration is 15 g in any direction.

Table D-2. Sinusoidal vibration design levels

Frequency, Hz	Vibration level
5 to 23	1.02 cm (0.4 in.) double amplitude
23 to 100	11.2 g peak
100 to 200	6.4 g peak

Table D-3. Random vibration design levels

Frequency, Hz	PSD level
25 to 100	-6 dB/octave
100 to 1000	0.1 g^2/Hz
1000 to 2000	-12 dB/octave
Overall level = 11.1 g rms	Duration: 3 min/axis.

6. Pyrotechnic Shock

Shock levels are as defined by the response shock spectra presented in Fig. D-5. This spectrum covers responses typically induced during the launch and deployment phases of the mission. For design purposes, the wave-form of the transient defined by the shock spectra may be assumed to consist of a summation of quasi-sinusoidal transients with a decay time of approximately 30 ms.

Additionally, instruments that incorporate pyrotechnic devices in their design will withstand detonation of such devices.

7. Temperature

The temperature requirements are categorized by possible spacecraft location and type of temperature control (active or passive). The indicated
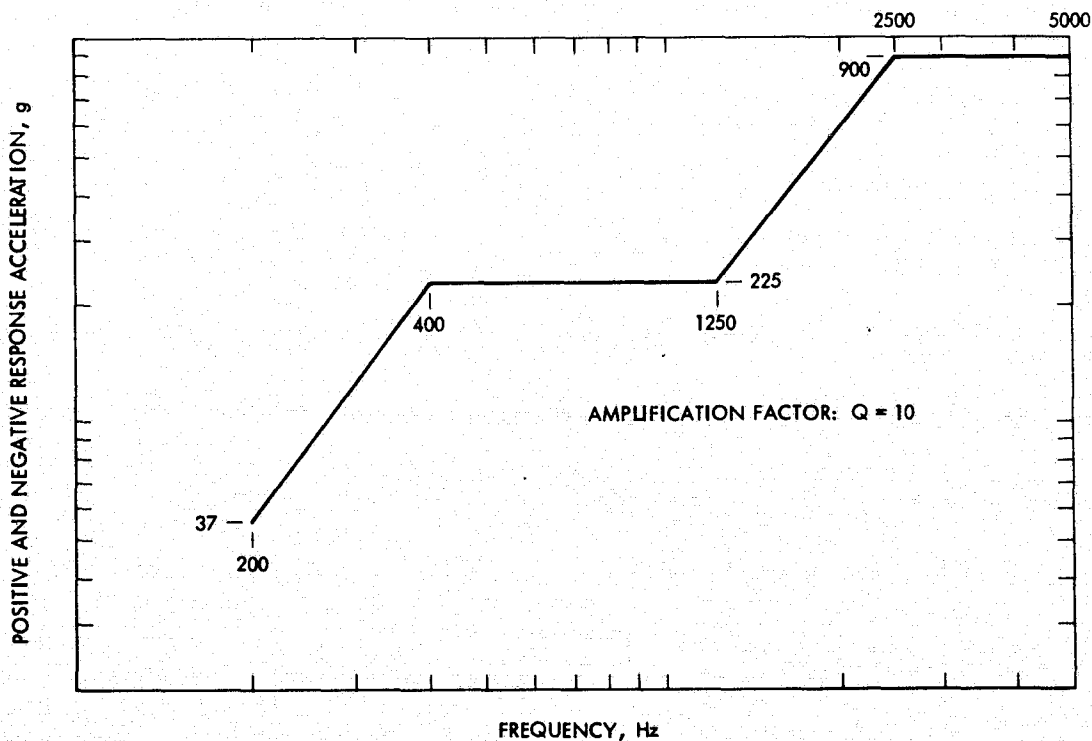

Fig. D-5. Shock design levels

temperature ranges are extremely soft and represent a composite of MJS'77 scientific instrument criteria, listed below:

(a) Instruments (or portions thereof, such as electrical packages) mounted interior to the Orbiter in a controlled environment, will be subjected to a temperature range of -20°C to +75°C.

(b) Instruments mounted on booms or at isolated locations will have a temperature range of -60°C to +80°C. Isolated instruments requiring a temperature range different from this will probably need some form of active temperature control.

(c) Unique Elements (such as sensor optics) will have a range of -140°C to +50°C.

An important factor relative to instrument temperature control is whether or not the instrument is required to operate in the near-Earth environment of 1.0 to 1.5 AU. If the instrument is not required to operate near Earth, the aforementioned temperature ranges can probably be narrowed.

8. Electromagnetic Compatibility

Experiments and engineering subsystems will be designed and constructed following standard practices to minimize the susceptibility to internal or external interference and to minimize the generation of interference which could be detrimental to other subsystems on the Orbiter or Probe. Specific requirements which follow assume a telemetry and system wiring design similar to the MJS'77 spacecraft from which these requirements are derived. Circuits which are more sensitive than the design values or which cause undesired emissions exceeding the design values should be shielded as necessary to satisfy the requirements. It is assumed that shielding attenuation will exceed 20 dB, andit should be so designed.

a.   Conducted Susceptibility.   The conducted-susceptibility for the DC power (assuming 30 V DC) is: ripple, 200 mV, 30 Hz to 150 kHz transient[6] +10 V, -10 V.   The susceptibility for the signal and control lines (digital and analog) is shown in Table D-4.

b.   Conducted Emissions.   Conducted emissions for the DC power are: 30 Hz to 50 kHz bandwidth ±2% of average steady-state load current or 10 mA whichever is more, maximum peak-to-peak ripple or transient current.

c.   Radiated Susceptibility.   The radiated susceptibility is shown in Table D-5.

Table D-4.   Conducted susceptibility

Wire Category		Unshielded	Twisted
Low/medium level, 0 to 10 V	Transient[6]	2 V	1 V
High level, 0 to 30 V	Transient[6]	6 V	3 V

Table D-5.   Radiated susceptibility

Frequencies	Orbiter, V/m	Probe, V/m
UHF probe to Orbiter link (assuming 380 MHz, 30 W)	35	40
S-band Orbiter to Earth (assuming 2295 MHz, 30 W)	60	60
Downlink X-band from Orbiter (assuming 8415 MHz, 20 W)	40	40
Jovian radiation 5-40 MHz	3	3
40 MHz - 10 GHz	1	1

[6]MIL STD 462 method, 300 Hz rate separate positive and negative tests.

d. _Radiated Emissions._ Radiated emissions (Table D-6) will be measured at one meter from the face of the subsystem. A radio science experiment is assumed to be on the Orbiter.

e. _Isolation._ The Orbiter shall be designed in such a manner that it is not possible to have multiple ground return paths for any signal or power source from one subsystem to another, except deliberate multiple adjacent return wires from one subsystem to another for the purpose of reducing IR voltage losses. Circuits providing isolation for this purpose ("isolated interfaces") shall have at least one megohm isolation from signal return to isolated circuit common. If several isolated circuits share one return, the requirement of one megohm shall be divided by the number of circuits sharing the return. If a subsystem is internally isolated from chassis, it will be isolated by at least one megohm. AC isolation will be achieved at isolated subsystem interfaces as much as possible but without placing capacitors at those interfaces.

f. _Bonding._ Adequate bonding (electrical conductivity) will be provided across contiguous assemblies, subassemblies, subchassis, covers, and mounted components. Bonding resistance will be 25 milliohms or less except for articulating or moving parts, which will use a bonding wire of less than 0.25 ohms, and thermal blankets, which will be bonded so that not more than 10 V can be present between the blankets and the electronic packages.

Table D-6. Radiated emissions

Frequencies	Orbiter, dBμV/m	Probe, dBμV/m
UHF 380 ±10 MHz	-30	-30
S-band 2110 to 2120	-19	-19
100 Hz to 10 kHz	40 decreasing to 10	40 decreasing to 10
50 MHz to 10 GHz (except as specified above)	76	76
10kHz to 50 MHz	10	10

g. Magnetics. To preclude interference to a magnetic experiment, the use of magnetic materials and devices shall be minimized. Essential magnetic devices will be selected and designed to satisfy the magnetic requirements below. The maximum radial magnetic field as measured at one meter from the center of the subsystem, after being magnetized with a peak demagnetizing field of 4 mT and also during the time it is energized in the most magnetic mode, will be not more than the value given in Table D-7. The magnitude of vector field changes will be less than 5 nT at one meter because of mode changes (including ON to OFF, if OFF is a normal flight mode) and motion of moving, articulating, or stepping devices, unless occurring infrequently and correlated by telemetry.

9. RTG and RHU Nuclear Radiation

Instruments will be designed to function within specification during exposure to the neutrons and gammas emitted from the RTGs and RHUs as specified in Table D-8. The neutron fluence and gamma dose design requirements are integration values over four years and are total spectrum values having an average energy in the ranges of $0.3 \leq E \leq 3.0$ MeV for gammas and $1.0 \leq E \leq 3.0$ MeV for neutrons. The combined RTG and RHU radiation limits specified in Table D-8 are to be interpreted as the minimum allowable of

Table D-7. Maximum radial magnetic field at 1 m energized or deenergized after 4 mT demagnetization

Major subsystem assemblies on despun platform (in relation to magnetometer sensor)	20 nT
Major subsystem assembly on spinning platform (in relation to magnetometer sensor)	10 nT
All other assemblies of subsystem	5 nT

Note:

nt (nanoTesla = 10^{-5} gauss)

mT (milleTesla)

Table D-8. Nuclear radiation

Particles	Peak flux (unshielded), cm^{-2}	Fluence (unshielded), cm^{-2}
RTG and RHU neutrons $(1.0 \leq E \leq 3.0 \ MeV)$	80	1×10^{10}
RTG and RHU gamma $(0.3 \leq E \leq 3.0 \ MeV)$	3200	1×10^{3} Rad (Si)

NOTE: All values unshielded - shielding will alter the local electronics environments significantly.

spacecraft equipment susceptibility to nuclear radiation. The radiation contribution from both RTGs and RHUs must be considered to ensure that an instrument will not be affected by the total radiation resulting from the RHUs and RTGs. Compatibility with the total nuclear radiation environment must be demonstrated by analysis. Detailed radiation spectra needed for design analysis for each of these radiation fields will be available later.

a. Radioisotope Thermoelectric Generators (RTG). The RTG radiation is currently based on the HELIPAK RTG design and fuel that is five years old at beginning of mission, pure plutonium oxide with 1.2 parts/10^6 (236 $_{Pu}$, 232 $_{U}$, and 228 $_{Th}$), and a neutron source intensity of 7,000 n/s-g 238 $_{Pu}$ at the fuel sphere level. A self multiplication factor of 1.18 was used to obtain neutron source intensity at the RTG level.

b. Radioisotope Heater Unit (RHU). The RHU radiation is currently based on a 1-W unit design with fuel that is five years old at the beginning of mission, pure plutonium oxide with 1.2 parts/10^6 (236 $_{Pu}$, 232 $_{U}$, and 228 $_{Th}$), and a neutron source intensity of 7,000 n/s-g 238 $_{Pu}$. Orbiter assemblies employing RHUs or adjacent RHUs will be designed to function within the RHU neutron and gamma radiation levels of Table D-9.

Table D-9. RHU neutron and gamma radiation

Distance from RHU, cm	Neutron		Gamma	
	Peak, flux $(cm^{-2}\text{-}s^{-1})$	Fluence, cm^{-2}	Peak, flux $cm^{-2}\text{-}s^{-1}$	Fluence, rad-Si
0	3.1×10^2	3.9×10^{10}	3.2×10^3	2.0×10^3
2	1.1×10^2	1.4×10^{10}	1.3×10^3	8.0×10^2
4	5.7×10^1	7.0×10^9	6.4×10^2	4.0×10^2
6	3.1×10^1	3.9×10^9	3.2×10^2	2.0×10^2
8	1.9×10^1	2.5×10^9	2.0×10^2	1.3×10^2
10	1.3×10^1	1.7×10^9	1.4×10^2	9.0×10^1
15	5.7×10^0	7.4×10^8	6.4×10^1	4.0×10^1
20	3.2×10^0	4.2×10^8	3.2×10^1	2.0×10^1
50	6.4×10^{-1}	8.3×10^7	6.0×10^0	4.0×10^0
100	1.8×10^{-1}	2.3×10^7	1.6×10^0	1.0×10^0

10. Electrostatic Charge Potential

Maximum charging currents equal to 2×10^{-10} A/cm^2 will be experienced during Jupiter encounter. The Orbiter will be designed so that the maximum difference of potential between any two points on the Orbiter surface does not exceed 10 V when these currents are conducted through the surface. Known exceptions to this requirement will be treated on an individual basis.

D. CRUISE

1. Near-Earth Radiation

The natural (solar) nuclear radiation environment in terrestial space consists of (a) galactic cosmic radiation, (b) geomagnetically trapped radiation, and (c) solar-flare particle events.

a. <u>Galactic Cosmic Radiation (Mainly Protons)</u>

   (1) Composition: 85% protons, 13% alpha particles, 2% heavier nuclei.

   (2) Energy Range: 10^7 to 10^{19} eV; predominant 10^9 to 10^{13}.

   (3) Flux outside Earth's magnetic field:
       0.2 to 0.4 particles/cm^2/sr/s.

   (4) Integrated yearly rate: approximately 1×10 protons/cm^2.

   (5) Integrated yearly dose: approximately 4 to 10 rad.

b. <u>Trapped Radiation (Protons, Electrons)</u>

   (1) Energy: Electrons >0.5 MeV, Protons >34 MeV.

   (2) Peak electron flux: >10 electrons/cm^2/s (omnidirectional).

   (3) Peak electron flux altitude: approximately 1000 nmi at equator.

   (4) Peak proton flux: 10^4 to 10^5 protons/cm^2/s (omnidirectional).

   (5) Peak proton flux altitude: approximately 1900 nmi at equator.

c. <u>Solar Particle Events</u>

   (1) Composition: energetic protons and alpha particles.

   (2) Occurrence: sporadically and lasting for several days.

   (3) Particle event model (free space): see Section 2.4.3 of NASA TMX 64627.

$$N_p(>T) = \begin{cases} 7.25 \times 10^{11} T^{-12} & 1 \text{ MeV} \leq T \leq 10 \text{ MeV} \\ 3.54 \times 10^{11} E^{-P(T)/67} & 10 \text{ MeV} \leq T \leq 30 \text{ MeV} \\ 2.64 \times 10^{11} E^{-P(T)/73} & T > 30 \text{ MeV} \end{cases}$$

   (4) Protons: (as above)

   Alphas: $N_d(>T) = N_p(>T)$ $T < 30$ MeV

   $7.07 \times 10^{12} T^{-2} \, 14$ $T \geq 30$ MeV

where

$$NP (>T), \ Nd \ (>T) = protons/cm^2, \ alphas/cm^2 \ with \ energy >T$$

$$P \ (T) = particle \ magnetic \ rigidity \ in \ mV$$

$$P \ (T) = \frac{1}{Ze} \left[(T \ (T + 2m \ c^2) \right] 1/2$$

$$Ze = 1 \ for \ protons, \ 2 \ for \ alphas$$

$$c^2 = 938 \ MeV \ for \ protons, \ 3728 \ MeV \ for \ alphas$$

For near-Earth orbital altitudes, the above free-space event model must be modified because the Earth's magnetic field deflects some of the low-energy particles that would enter the atmosphere at low latitudes to the poles.

2. Meteoroids

Meteoroid design fluences and particle characteristics for MJS'77 suitable for JOP 81/82 instrument proposal purposes are shown in Table D-10.

E. PROBE SEPARATION AND PLANETARY ORBIT

1. Charged Particle Radiation Guidelines

The Orbiter will be exposed to an intense flux of energetic electrons and protons in the Jupiter magnetosphere in addition to the neutrons and gamma radiation from the Orbiter RTGs and charged particles from the solar wind and possible solar flare events. Consequently, the Orbiter engineering assemblies and the associated science assemblies will be required to function within acceptable performance limits while they are exposed to the peak flux of these particles and also after they have been exposed to an accumulated fluence of these particles corresponding to the mission fluence, as specified in other parts of this document. Experience on previous programs have shown that it is necessary during the design of the assemblies to consider radiation received by the assemblies to ensure that the assemblies are sufficiently "radiation-hard". The purpose of this section is to provide some radiation design guidelines to assist the designer in arriving at a design of adequate radiation hardness.

Table D-10. Meteoroids

Particle mass, grams	Integral fluences (particles m^{-2} of mass greater than M)
10^{-10}	2.0×10^3
10^{-9}	7.9×10^2
10^{-6}	8.8
10^{-5}	7.0×10^{-1}
10^{-4}	5.4×10^{-2}
10^{-3}	2.6×10^{-3}
10^{-2}	1.9×10^{-4}
10^{-1}	1.0×10^{-5}
10^{0}	8.0×10^{-7}

Mean relative speed, km/s
Particle mass density, g/cm^3

In general, the designer should give particular attention to selecting and incorporating electronic parts which have the greatest radiation tolerances and, so far as possible, to design the electronic circuits so that the circuits can accommodate changes in parts parameters which may occur because of radiation degradation. This last guideline implies that, unless absolutely necessary, the circuits should be designed so that tight part parameter tolerances are avoided. For the MJS'77 program, a number of semiconductor devices such as bipolar transistors, junction field effects transistors (JFETs), linear ICs, complementary metal-oxide semiconductors (CMOS), and diodes, rectifiers, and zener diodes have been characterized and specific radiation design criteria documented (see Ref. D-2).

a.   Long-Term Ionization Effects.   Most of the potential problems with the Orbiter electronics are expected to be due to-long-term ionizing

radiation effects. These effects are manifestations of charges trapped in insulating layers on the surfaces of the semiconductor devices. They are most important in MOS structures in which trapped charge in the gate oxide layer produces a first-order change in the apparent gate voltage. Trapped charge in surface passivation layers is also important in junction devices where it plays the role of producing an inversion layer that spreads out in the effective surface area, thereby increasing recombination-generation currents. These currents are most important in bipolar transistors operated at low collector currents and in n-channel JFET devices. The susceptibility to charging depends on the quality of the oxide layer and is not usually consciously controlled in semiconductor device manufacturing.

In optical materials, long-term ionization effects appear primarily as the introduction of optical absorption in otherwise transparent spectral regions for the particular material. These are usually manifestations of charge trapping at a pre-existing defect, so the rate of coloration is a strong function of the initial material.

In quartz crystals used for precision oscillators or filters, the same type of long-term ionization effects can produce significant resonant-frequency shifts. In this case, selection of the quartz crystal growth method can minimize the effect.

The magnitude of long-term ionization is a function primarily of ionizing energy deposition, i.e., the dose, as measured in rads in the material in question. Since the energy deposition is almost independent of material, rads are used herein without specifying the material.

The devices and materials of concern are:

(1) MOS devices (threshold voltage shift, enhanced leakage in CMOS pairs).

(2) Bipolar transistors (h_{FE} degradation especially at low collector current I_C), and JFETs (enhanced source-drain leakage current).

(3) Analog microcircuits (offset voltage, offset current and bias-current changes, gain degradation).

(4) Digital microcircuits (enhanced transistor leakage).

(5) Quartz resonant crystals (frequency shifts).

(6) Optical materials (coloration).

(7) External polymeric surfaces (mechanical degradation).

MOS Devices. In unhardened MOS structures charge trapping in the gate oxide and interface states can produce threshold voltage shifts of 1 V per dose of 10 krad (Si). This effect produces increased leakage current through a CMOS pair and eventually makes the device incapable of switching. The effect is most serious in n-channel MOS devices because it is enhanced by positive gate bias.

For the hardened CMOS units the design must still tolerate large increases in supply currents, and leakage current for the multiplexer. Typically, supply currents, which may start in the nA range prior to irradiation, may increase into the μA range after exposure to ~100 krad (Si). Off-state leakage in the multiplexer may increase to ~100 nA. The CMOS memory units are not screened and must be shielded to <5 krad (Si), depending on the results of sample radiation tests. Some unhardened MOS based switches must also be shielded (e.g., DGM 111 analog switches).

Bipolar Transistors and JFETs. The same oxide charging that occurs in the gate oxide of MOS devices also occurs in the passivation layer of bipolar devices and JFETs. The primary effect is to produce an inversion layer near the surface of doped p-type regions, producing enhanced PN junction leakage by increasing the depletion volume (increasing recombination-generation currents). In extreme cases, the inversion layer could bridge across an NPN structure or extend up to the contact on a P layer, producing an effective device short.

In transistors, this inversion appears mostly as a decreased h_{FE}, primarily at low I_C, $[\Delta(1/h_{FE})]$ depends roughly on $I_C^{-1/2}$. Figure D-6 is a sample of data on the effect of 150 krad (Si) on various transistors. Most
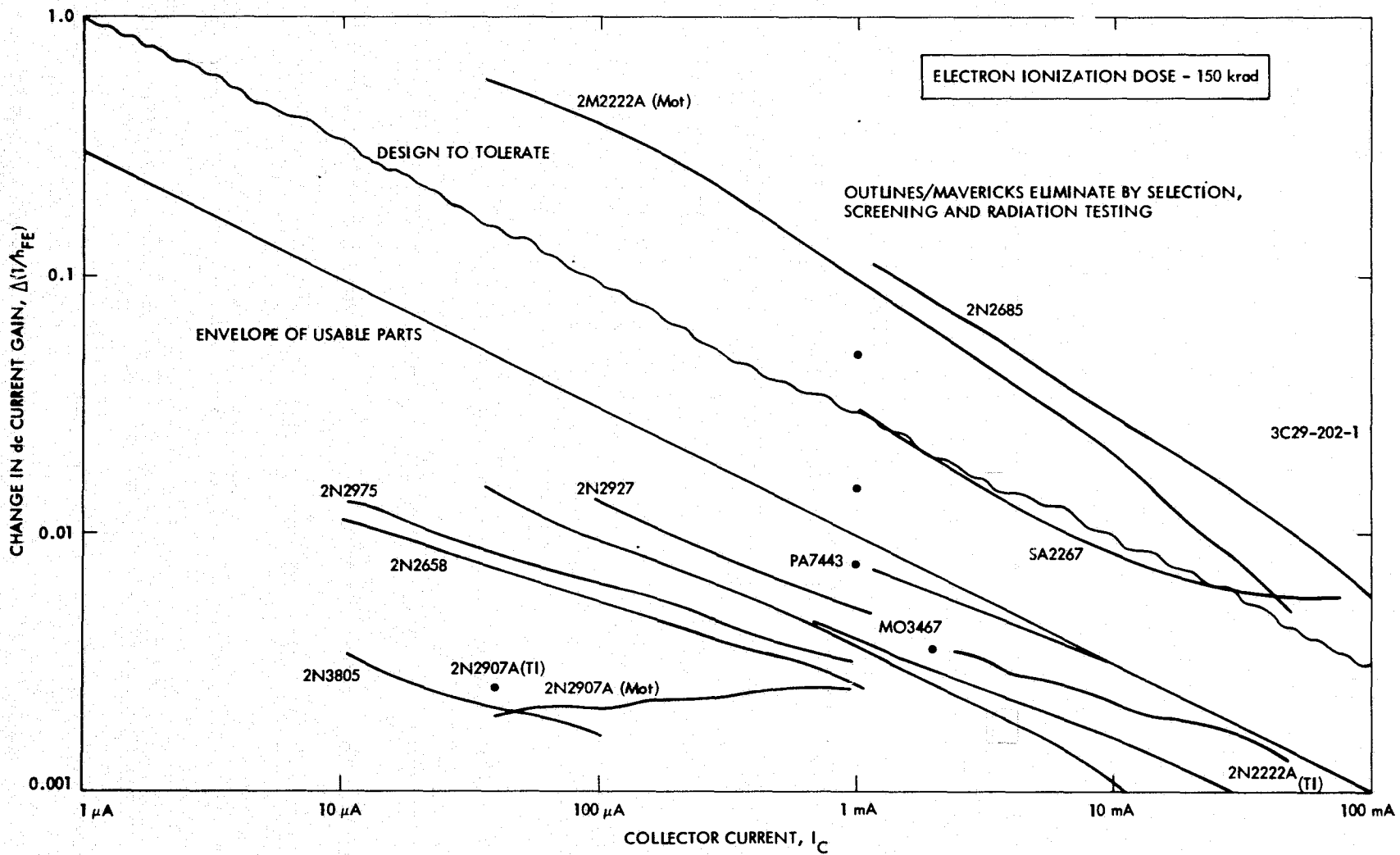
Fig. D-6. Typical long term ionization effects in bipolar transistors (worst-case response from small samples)

transistors appear to fall below a line $\Delta(1/h_{FE}) = 0.01\ I_C^{-1/2}$ where I_C is an mA. It is important to note the difference between the Motorola 2N2222A, with $\Delta(1/h_{FE})$ a factor of 50 greater than the TI 2N2222A. This illustrates that this damage is a function of the passivation process, not the device type number.

The best approach to this problem is to design the circuits to tolerate $\Delta(1/h_{FE}) \sim 0.03\ I_C^{-1/2}$, and assure by tests on samples from flight purchase lots that devices from poorer passivation processes are not incorporated in the flight hardware.

In JFETs, the primary effect is enhanced leakage through the substrate in n-channel devices. This effect is particularly important in JFET switches and can also contribute extra noise in sensitive input stages. Data on specific devices sampled from the purchased lots, preferably the same diffusion lot for critical applications, are required to quantify the response.

Analog Microcircuits. OpAmps are particularly susceptible to surface effects because the input transistors are invariably operated at low collector current to increase the input impedance. Again, the quality of the passivation layer controls the response, rather than the design of the semiconductor circuit.

Experimental data indicate that 150-krad (Si) exposures of unhardened OpAmps may produce large offset voltage changes (up to 500 mV), and offset currents ($\sim$100 nA). Hardened versions seem to be able to achieve $\Delta V_{OS} < 2$ mV and $\Delta I_{OS} < 1$ nA. Effective hardening requires diffusion lot sampling to check on oxide quality. Similar effects occur in other linear microcircuits, many of which incorporate some version of an OpAmp (e.g., voltage regulators).

Digital Microcircuits. Digital microcircuits do not appear to be a problem in the MJS'77 environment. However, since the 54L series used in MJS'77 operates at relatively low currents, consideration should be given to perform some sampling tests to ensure that the quality of the passivation is not accidentally degraded to the point exhibited by the worst transistors shown in

Fig. D-6. If the 54L transistors were degraded as much as the Motorola 2N2222A (Fig. D-6), it is unlikely the logic would function acceptably.

• Quartz Resonant Crystals. Ionization also produces frequency shifts in quartz oscillator and filter crystals. The magnitudes of such shifts depend upon the source of the quartz, as shown in Fig. D-7. For all applications in which a frequency shift up to 1 part in 10^7 can be tolerated for MJS'77, the use of swept synthetic quartz will ensure that the specifications are met. It is recommended that a sample from each of the stones used be irradiated to ensure that the material was properly swept. The use of natural quartz is not recommended unless changes as large as 10 parts/10^6 can be tolerated. For applications in which the radiation-induced changes must be less than 1 part in 10^7, a further screening process should be applied to carefully selected swept synthetic materials.

Optical Materials. Optical materials undergo changes in optical transmission as the result of exposure to ionizing radiation. Table D-11 is a summary of data on the rate of coloration of various optical glasses. At worst, exposure to 100 krad (Si), as in reasonably shielded locations in the electronics packages, can produce an absorption coefficient of the order of 1 cm^{-1} for material that darkens very effectively. Absorption coefficients a factor of 50 larger can be achieved in theory but are not generally observed in practice. The best materials are the purest fused silica, such as Suprasil 1, Corning 7940, and Suprasil W-1. After exposure to 10^5 rad (Si), the absorption coefficient will still be less than 10^{-3} cm^{-1}.

Particular concern must be paid to optical materials exposed to the external proton environment because the dose in a thin surface layer can exceed 10^8 rad (Si). If possible, ultraviolet-grade fused silica should be used for windows and coatings applied to the inside surface only. If not, specific high dose data on the material to be used must be acquired. Again, the color-ability of coating materials can depend on impurities that are not normally controlled, so that tests may be required on samples prepared at the same time as light devices.
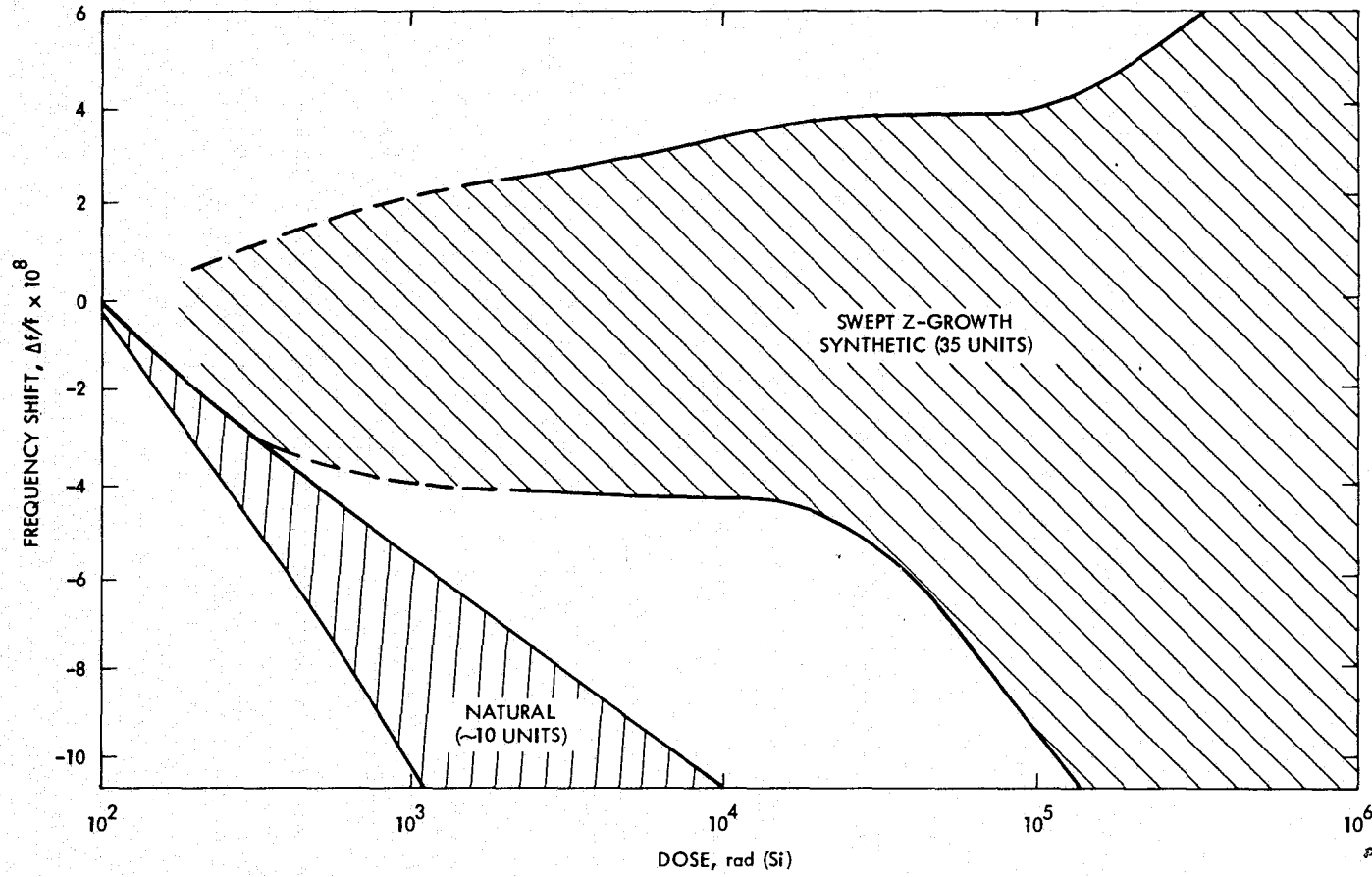
Fig. D-7. Composite steady-state frequency shift data versus dose
for quartz crystal resonators

Table D-11. Absorption in optical materials[a]

Relative rating	Material	Absorption coefficient cm^{-1} rad^{-1}
Excellent	Suprasil 1	$\sim 10^{-10}$
Very good	Corning 7940	$\sim 10^{-8}$
	Suprasil W1	$\sim 10^{-8}$
	Polystyrene	$\sim 3 \times 10^{-8}$
Good	Amersil T0-8	$\sim 10^{-7}$
	Lead silicate (ordinary glass)	$\sim 2 \times 10^{-6}$
	Corning 5010 fiber	$\sim 10^{-5}$
Worst case	From defect formation (assumes $\mu = 10^{-16} Nd$)	$\sim 5 \times 10^{-4}$

[a]See Ref. D-3.

External Polymeric Surface.   The MJS'77 dose internal to the electronics packages is insufficient to cause significant degradation of mechanical or electrical properties of even the most sensitive polymers.   The proton dose on external surfaces, however, can exceed 10^8 rad (Si).   Any polymeric materials exposed to this dose must be examined for their radiation response (mostly in the literature).

b.   Transient Ionization Effects (Interference).   Interference is defined as transient ionization effects which persist only while the electronics are being irradiated, and whose severity is generally proportional to the dose rate.   There are four types of interference in MJS'77 electronics:

(1)   Primary photocurrents in low-current sensitive input stages to the electronics.

(2)   Electron emission from cathodes of electron multiplier-type detectors.

(3) Ionization-induced conductivity in photo-sensitive materials, such as those in the VIDICON detector surface.

(4) Ionization-induced fluorescence in optical materials such as detector windows and lenses (fluorescence efficiencies vary strongly with the kind of material).

Interference effects are dependent primarily on the rate of ionization energy disposition, i.e., the dose rate measured in rad/s. At the low rates of interest to MJS, the effects are essentially proportional to dose rates.

Interference effects at the relatively low-peak dose rates (less than rad (Si)/s at internal positions) can be important only in devices operating at extremely low currents. The effect of ionizing radiation on semiconductor devices can be represented by an equivalent current generator across reverse biased junctions (primary photocurrent) whose magnitude is up to 100 pA for power semiconductor devices exposed to 10 rad (Si)/s and whose magnitude is proportional to the instantaneous dose rate. Devices whose normal operating point is at currents in the μA or mA range will not be significantly affected by interferences.

Interference effects can also occur at the cathode of electron multipliers because the quiescent currents are very low and the gain between the cathode and electronic circuitry is very high. The flux of secondary electrons emitted from a surface by passage through it of energetic electrons is about 5 to 10% of the incident electron flux.

Ionizing radiation will also affect optical materials. During excitation by ionizing radiation a small fraction of the energy deposited may be re-emitted as fluorescence. Typically, fluorescence efficiencies are much less than 1% unless the material has been carefully prepared to give high fluorescence efficiencies (e.g., nuclear particle scintillation detectors). Higher purity optical materials (e.g., ultraviolet-grade fused silica) tend to have lower fluorescence efficiencies.

When the foregoing interferences are unacceptable the following corrective techniques can be used:

(1) Use semiconductor devices with minimum junction area to minimize the primary photocurrents. Devices exhibiting less than 10 pA at 10 rad (Si)/s are available.

(2) The use of very pure fused silica, such as the ultraviolet-grade materials, will minimize fluorescence. In some materials the fluorescence yield is actually less than the Cerenkov radiation emitted by the fast electrons. An optical filter can sometimes be interposed between the glass and the detector to block the fluorescence wavelengths while passing the desired optical signal.

(3) There is no effective means of suppressing secondary electron emission from surfaces other than shielding them from the incident electrons.

c. Displacement Effects. Displacement of atoms in crystal lattices cause permanent changes of material properties. The expected proton and electron fluences during the JOP 81/82 mission are not expected to represent as severe an environment for displacement effects as for long-term ionization effects. Therefore, only the most sensitive devices will be significantly affected by displacement effects.

Displacement effects are mostly of concern for silicon devices. Therefore, all further reference to displacement effects will use silicon as a reference material.

Displacement effects can affect the following devices and properties in the MJS electronics:

(1) Bipolar transistors with low f_T (h_{FE}, $V_{CE\ SAT}$, $V_{BE\ SAT}$).

(2) PN junction diodes (V_F, V_B).

(3) Light-emitting diodes (LED) (light-emitting efficiency).

(4) Semiconductor photodetectors (sensitivity).

(5) Devices incorporating lateral p-n-p transistors.

d. Other Considerations. Two other factors should be considered. One is the radiation design margin to be incorporated in the design. Generally, a design margin of a factor of two has been considered minimal. Second is the treatment of so called "maverick" parts, i.e., parts in a population which fail catastrophically at small radiation levels, although the population as a whole is generally radiation-resistant. This concern seems to be part-type and process-dependent. Some guidance is provided in the Radiation Design Criteria Handbook (Ref. D-2).

F. ABORT

An abort situation affecting the payload can occur in the two areas; first, during ascent and, second, in orbit, if the IUS can not be separated from the Shuttle. Either of these situations can result in a safe landing at a designated field or a crash landing.

1. Conditioned Air

Within 30 min following touchdown, the purge system supplies conditioned air to the payload bay utilizing <2. 0 psig supply pressure.

2. Pressure

The payload bay pressure history for a typical reentry is shown in Fig. D-8.

3. Landing Shock

Rectangular pulses of the peak accelerations shown in Table D-12 will be experienced.
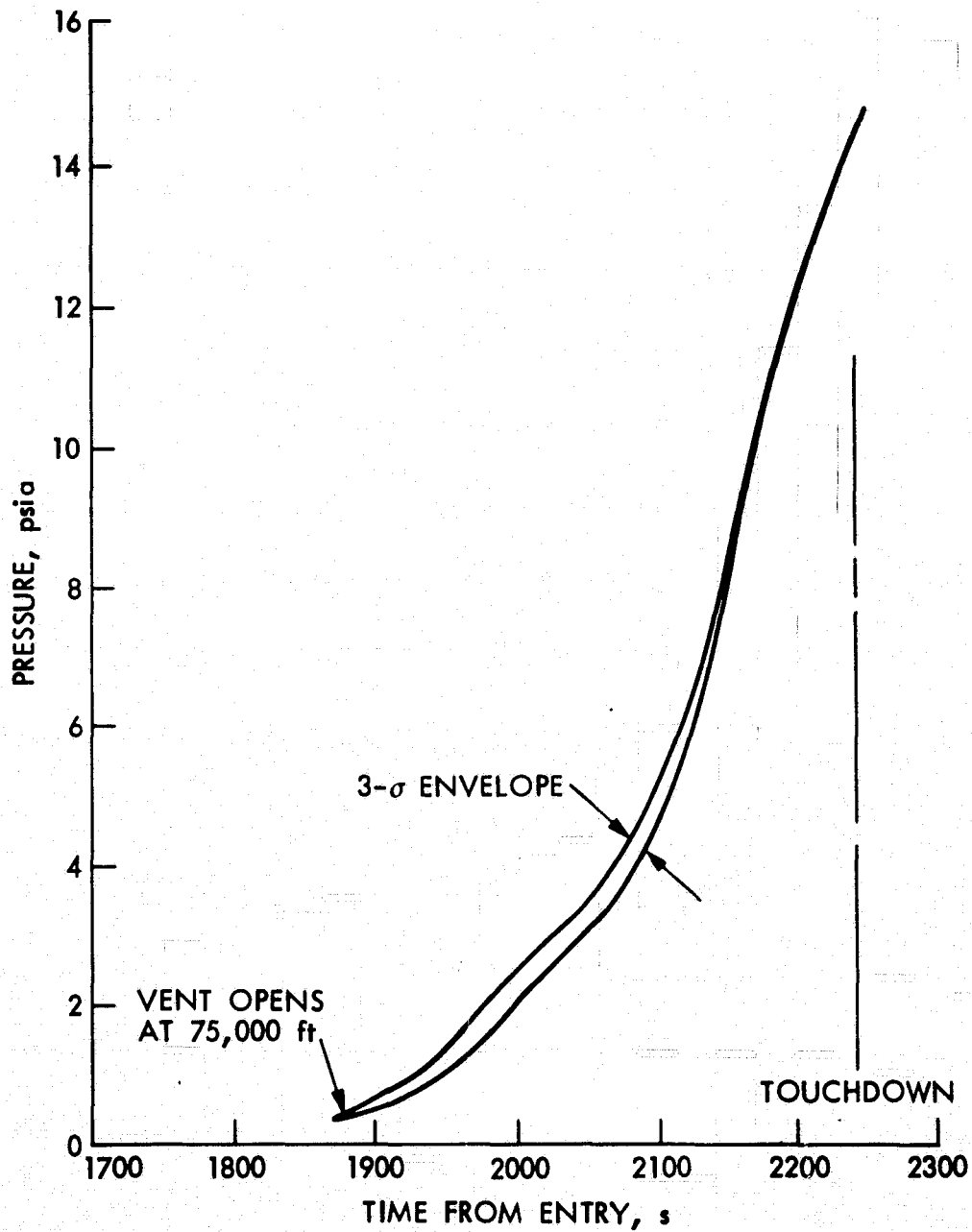
Fig. D-8. Orbiter payload bay internal pressure histories during entry

Table D-12. Landing shock probabilities

Acceleration, g peak	Time, ms	Applications, per 100 missions
0.23	170	22
0.28	280	37
0.35	330	32
0.43	360	20
0.56	350	9
0.72	320	4
1.50	260	1
		125

4. Crash Landings

The payload design shall consider provisions to maximize the probability of safe crew egress following crash landing or water ditching. To this end, the mounting structures for equipment and crew provisions in the crew compartment, for large equipment items, for pressure vessels, and for the payload attachments are designed to specified load factors generally not of interest to the instrument proposers. However, the static acceleration requirement of paragraph C-5 covers probable instrument crash load factors.

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D-1. Space Shuttle System, Payload Accommodations, JSC 07700, Vol. XIV, Johnson Spaceflight Center, Change 15, 4-27-76.

D-2. Radiation Design Criteria Handbook, Technical Memorandum 33-763, Jet Propulsion Laboratory, Pasadena, Calif.

D-3. Mattern, P. L., Watkins, L. M., Skoog, C. D., Brandon, J. R., Barsis, E. H., IEEE Transactions Nuclear Science, NS-21, 81, 1974.

APPENDIX E

MISSION OPERATIONS DESCRIPTION

A. GENERAL

Mission operations for JOP will be carried out by the JOP Mission Operations Complex (MOC). The MOC is comprised of the Mission Operations Organization, the Ground Data System, and the plans and policies and procedures which are developed jointly by the JOP Mission Operations System (MOS), Mission Computing System (MCS), and the Tracking and Data System (TDS).

B. OPERATIONS CONCEPT

The planning for JOP Mission Operations is based upon the following factors:

(1) The requirement to provide, at minimum cost, operations planning and support to deliver a probe into the jovian atmosphere, place a spacecraft in orbit around Jupiter, perform multiple encounters with various jovian satellites, and retrieve data collected by the Probe and the Orbiter.

(2) The requirement to operate, at minimum cost, scientific instruments which are not only spatially remote, but remote also in the time dimension.

(3) The requirement to handle, at minimum cost, high-data rate transmissions from the spacecraft and concurrently from the Deep Space Network (DSN) to the Mission Control and Computer Center (MCCC) at JPL.

These requirements have resulted in the identification of certain operational concepts, including the following:

(1) To reduce operational costs, which are directly related to mission duration, operational staffing must be minimized.

(2) Mission sequences will be highly structured and repetitive to allow for simple sequence design.

(3) The mission operations organization will not be responsive to short-term sequence change requests.

C. MOC ORGANIZATION

1. Management Organization and Structure

The MOC management organization is depicted schematically in Fig. E-1. The management is responsible for formulating the plans, policies, and procedures required for the MOC to assist the Project in accomplishing its objective.

Although the TDS and MCS managers report to the Project Manager for performance of their respective systems, they are also responsive to the MOS manager for the design, implementation, and operation of the MOC.

The CMO is responsible to the MOS manager in the development and implementation phase to ensure that the operations organization, Ground Data System, and their operating procedures are developed to support the mission objectives. The CMO is also responsible to the MOS manager for the training of the operations organization.

The GDSE is responsible to the MOS manager for the development of the ground hardware and software which is required to support the MOC.
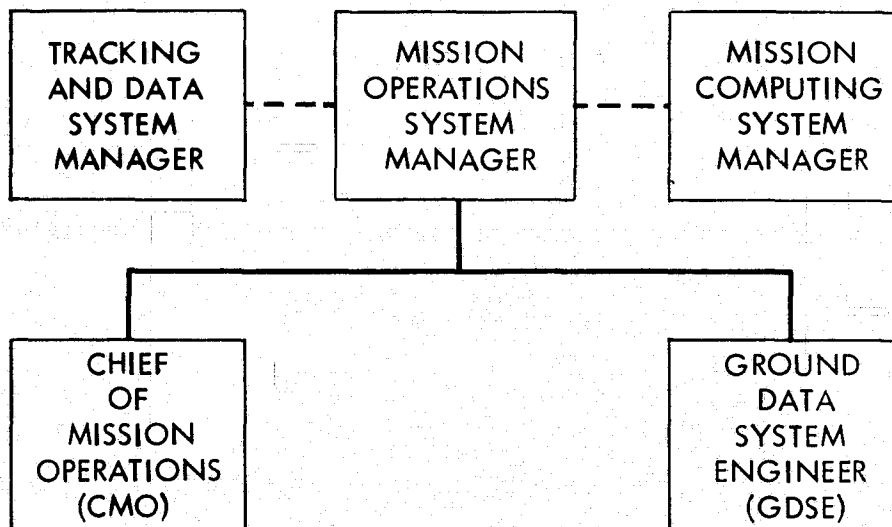
Fig. E-1. MOC management organization

2. Operations Organization

The organization described herein reflects certain underlying concepts which are intended to promote operating efficiency while holding down costs.

These are:

(a) Organization elements (teams) are defined around major functions to unify team operations and to simplify inter-team interfaces.

(b) Spacecraft monitoring is performed at the system level, insofar as possible, to minimize staffing requirements. Provision is made for occasional subsystem augmentation for critical activities.

(c) Planning and analysis functions are performed by off-line teams.

(d) The organization provides a simplified operational interface with the science arm of the project. It is assumed that:

(1) The organization will provide a mechanism for supplying science activity requests, which represent integrated requests from all experiments, in a form and to a schedule to be negotiated.

(2) First-order instrument performance analysis will be provided by the MOC. Detailed instrument performance analysis will be provided by the experimenters.

The JOP Operations Organization is depicted schematically in Fig. E-2, and the functions of the various teams are identified in Table E-1. Planned levels of support are shown in Table E-2.

Fig. E-2. JOP operations organization

Table E-1. JOP operations organization functions
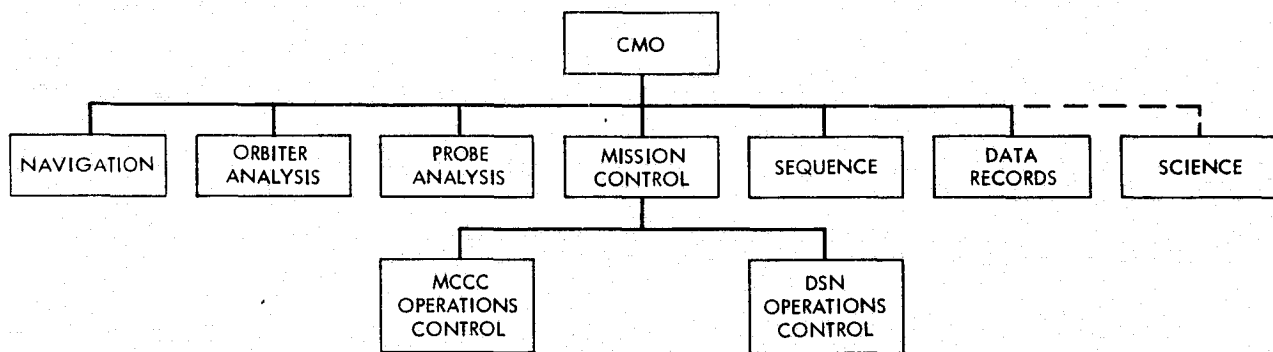
Team	Primary Functions
Chief of Mission Operations	Direct all operations activities
Navigation Team	Navigate the spacecraft to Probe delivery, JOI and subsequent encounters
	Provide information on spacecraft flight path for both sequence planning and flight-path reconstruction
	Provide tracking requests
	Provide metric data MDR
	Provide STS trajectory parameters
Orbiter Analysis Team	Predict Orbiter performance
	Analyze and evaluate Orbiter performance
	Pursue Orbiter anomalies
	Provide and maintain guidelines for Orbiter operations
Probe Analysis Team	Predict Probe performance
	Analyze Probe performance
	Provide and maintain guidelines for Probe operations
Sequence Team	Design science observation sequences
	Develop integrated spacecraft and ground sequences
	Generate commands and sequences of events
Mission Control Team	Control real-time operations of the MOC
	Direct execution of operational sequences
	Monitor Orbiter performance
	Interface with STS operations
	Conduct on-orbit Orbiter checkout
Data Records Team	Coordinate preparations of all project-generated Data Records (except matric data MDR)
Mission Control and Computing Center Team	Operate MCCC facilities
Deep Space Network Team	Operate the DSN
Science Team[a]	Analyze performance of instruments
	Provide sequence requests

[a]The Science Team is not part of the MOC, but has a strong interface with it.

Table E-2. Operations support levels

Function	Level of support
CMO	Off-line, one shift
Navigation	Off-line, cruise: one shift
	orbit: one shift, augmented for orbital operations
Orbiter analysis	Off-line, one shift
Probe analysis	Off-line, one shift, augmented for Probe operations
Sequence	Off-line, one shift, augmented for orbital operations
Mission control	
Overall direction	On-line, whenever tracking
Spacecraft monitor	On-line, whenever tracking
Command	On-line, cruise: one shift, 5 days/wk
	orbit: one shift, 7 days/wk
Data records	Off-line, cruise: one shift, 5 days/wk
	orbit: one shift, 7 days/wk
MCCC operations	On-line, whenever tracking
DSN operations	On-line, whenever tracking

D. DATA SYSTEM

1. End-to-End Data System

The End-to-End Data System is depicted in Fig. E-3. The salient features of its design are:

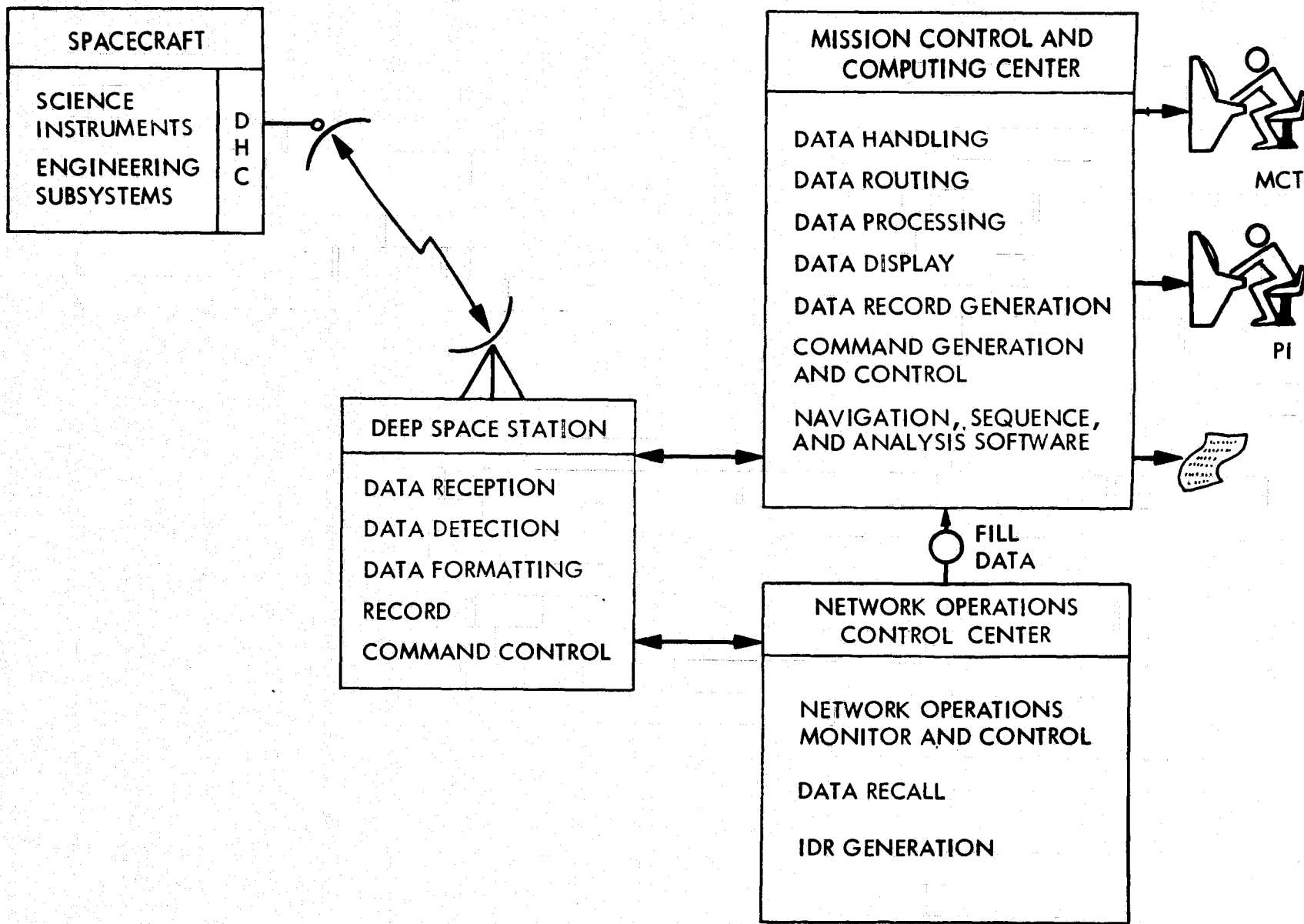(a) Orbiter data is transmitted and received on both S- and X-band carriers.

Fig. E-3. JOP end-to-end data system

(b) The maximum data rate is 128 kb/s.

(c) Probe data will be transmitted to the Orbiter for about 30 min, where it will be both relayed and stored.

(d) During non-track periods the Orbiter will record data which will be read out at high rates at infrequent intervals.

(e) Science data, after receipt at MCCC, is routed to science-funded processors for real-time, near-real-time, or non-real-time processing.

(f) Real-time telemetry and command displays will be driven in the Mission Control Team area only.

(g) The Engineering and Science MDRs are mostly generated in real time. Metric MDR is not generated in real time.

2. Ground Data System

The Ground Data System (GDS) portion of the MOC comprises all hardware and software used for ground support of flight operations and pre-flight operational testing and training. Functions performed by the GDS include simulation, acquisition, recording, routing, processing, and display of data.

The GDS comprises nine functional systems as shown in Fig. E-4. Five of the systems are real time in nature: Tracking, Command, Telemetry, GDS Information, and Simulation. The remaining four systems are non-real time in nature and comprise the Project's non-real-time software: Navigation, Mission Sequence, Spacecraft Analysis, and Data Records.

Figure E-4 also illustrates that the GDS may be represented by a matrix in which data flow takes place in systems characterized by the type of data handled. These systems are functional in nature and provide a framework for describing end-to-end data flow through the various ground-based facilities. The real-time systems are represented as using elements in the Deep Space Stations (DSS), Ground Communication Facility (GCF), Network Operations Control Center (NOCC), Mission Control and Computing Center (MCCC), Satellite Tracking and Data Network (STDN), and the NASA Communications Network (NASCOM).

TDS   MCS

DSS	GCF	NOCC	MCCC

TRACKING   NAVIGATION

COMMAND   MISSION SEQUENCE

DATA RECORDS

TELEMETRY   SPACECRAFT ANALYSIS

NON-REAL-TIME PROJECT SOFTWARE

MON, OR AND OPERATIONS CONTROL   MONITOR AND OPERATIONS CONTROL   GDS INFORMATION

SIMULATION

FOR STS	STDN STATION	NASCOM	MCCC
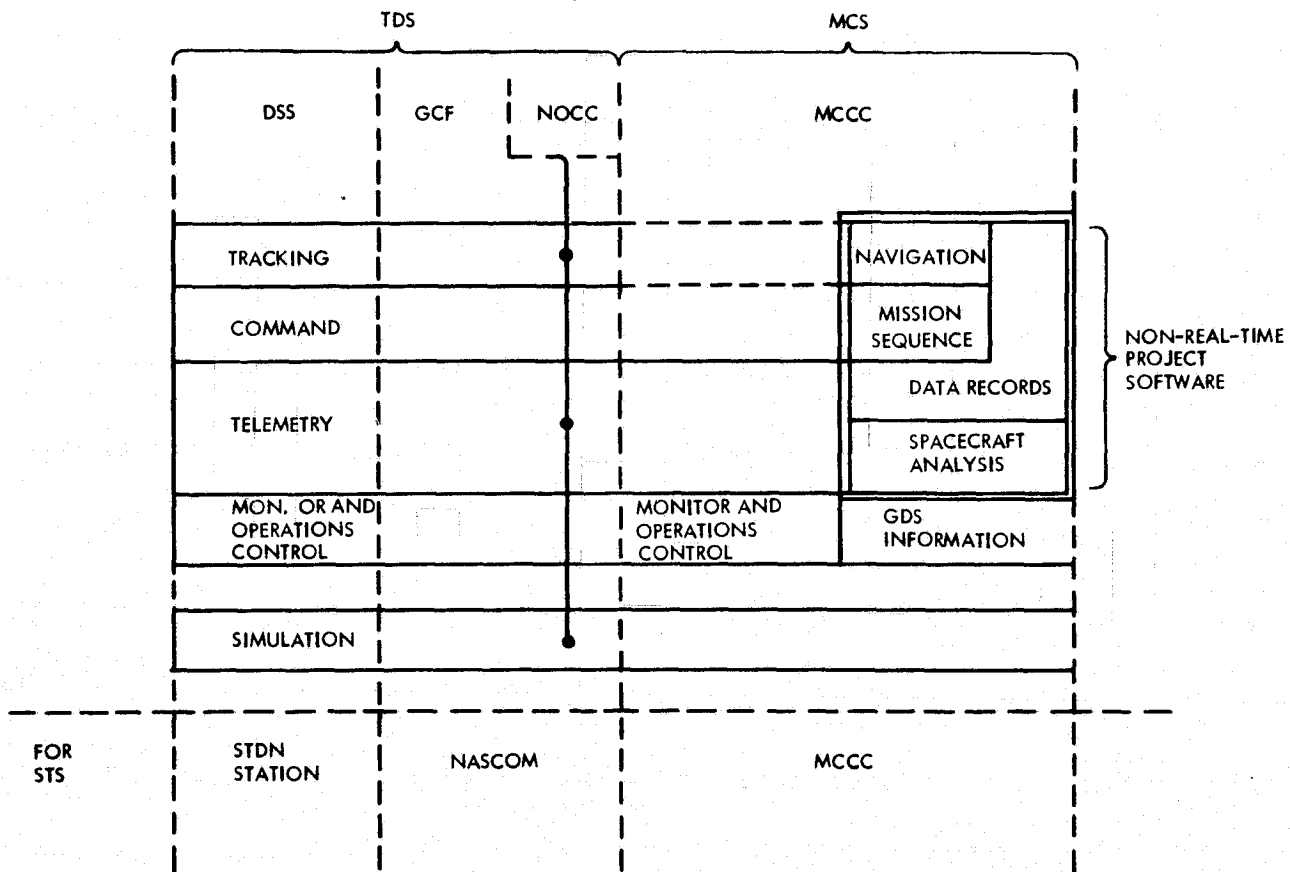
Fig. E-4. Elements of the ground data system

The non-real-time systems defined exist completely in the MCCC portion of the GDS. It should be noted that except for the GDS Information System and the Tracking System, the NOCC is not a serial element in the GDS; rather, it interfaces with the GCF in parallel with the MCCC.

A new element of the GDS for JOP is the interface with the Shuttle data system. A description of this interface is contained in Ref. E-1. Since this is a new and complicated interface, extensive testing of this element will be conducted during the GDS integration phase.

The following paragraphs identify the functions of each of the systems.

a. **Tracking System.** The Tracking System provides the data from which the spacecraft-flight path parameters are determined. Specifically, it generates, predicts, and transmits them to the tracking stations. The tracking stations use the predicts to acquire the spacecraft. When the spacecraft has been acquired, the Tracking System generates radiometric data including time-tagged antenna pointing angles, doppler, and ranging. The data is routed to the NOCC, where it is recorded and an intermediate data record (IDR) is generated for input to the Navigation System.

b. **Command System.** The Command System provides the means by which ground commands entered at either the MCCC or the DSS are transmitted to the spacecraft and that transmission is confirmed. The Command System will be designed to accept an automated command file generated by the Mission Sequence System.

The system will transmit commands at (TBD) b/s. The capability will exist to automatically initiate periodic reacquisition sequences and to repeat transmission of command blocks for reliability.

c. **Telemetry System.** The Telemetry System provides the means by which engineering and science data transmitted from the spacecraft are made available to users in the form of real-time and near-real-time displays and files. The Telemetry System also produces a number of real-time digital tape products which are used as inputs for near-real-time and non-real-time data records production.

The Telemetry System can also provide data for optical navigation, if required. The system will be capable of handling all data rates planned for JOP, and all data received will be routed to MCCC in real time.

d. **GDS Information System.** The GDS Information System provides visibility into the GDS configuration and performance. It provides the mechanism by which data files (e.g., Sequence of Events) can be automatically sent from the MCCC/NOCC as appropriate via GCF to the DSSs. The GDS Information System will be the prime tool for assisting operations personnel in GDS

problem isolation during mission training and operations. The GDS Information System uses data supplied by the MCCC and DSN Monitor and Operations Control Systems.

    e.   Simulation System. The Simulation System provides the capability for realistically testing elements of the GDS and for training all people associated with mission operations. It will be capable of providing command responsive telemetry data and tracking data inputs to the DSS or MCCC.

    f.   Mission Sequence System. The Mission Sequence System is the set of tools to be used by the Sequence Team during mission operations to design and validate mission sequences and to generate the data products required to monitor and control the execution of the sequences.

    g.   Navigation System. The Navigation System is a set of programs which enable the Navigation Team to determine the spacecraft trajectory and target ephemerides, provide statistical error information associated with the trajectory, and design trajectory correction maneuvers. The Navigation System reads and processes the Tracking IDR and produces the metric data Master Data Records (MDRs).

    h.   Spacecraft Analysis System. The Spacecraft Analysis System is a set of programs which enable the Orbiter Analysis Team and the Probe Analysis Team to predict spacecraft performance and to assess actual spacecraft performance.

    i.   Data Records System. The Data Records System is a set of programs which are used to generate MDRs, Experiment Data Records (EDRs) and Supplementary Experiment Data Records (SEDRs).

3. Data Processing

A data processing concept has been adopted for JOP which provides a simplified description of the various levels of processing of the telemetry data. This concept is shown in Fig. E-5 and is discussed below.

a. Level I Processing. Level I processing is the real-time processing of telemetry data required for spacecraft performance monitoring and mission control purposes. This processor drives real-time displays in the MCT areas. It also routes the science data to a data base and generates a system data record which may be usable as an MDR. This processor resides in a MCCC minicomputer.

b. Level II Processing. Level II processing is the near-real-time processing of telemetry data which accesses data from the data base. This processor is primarily geared to provide science data processing and display. The processor resides in a MCCC minicomputer.

c. Level III Processing. Level III processing is all JOP non-real-time data processing, at JPL, performed on a large scale MCCC computer.

E. OPERATIONS PLAN

The JOP Mission Operations organization and GDS designs will be documented in the MOS Design Book (Ref. E-2).
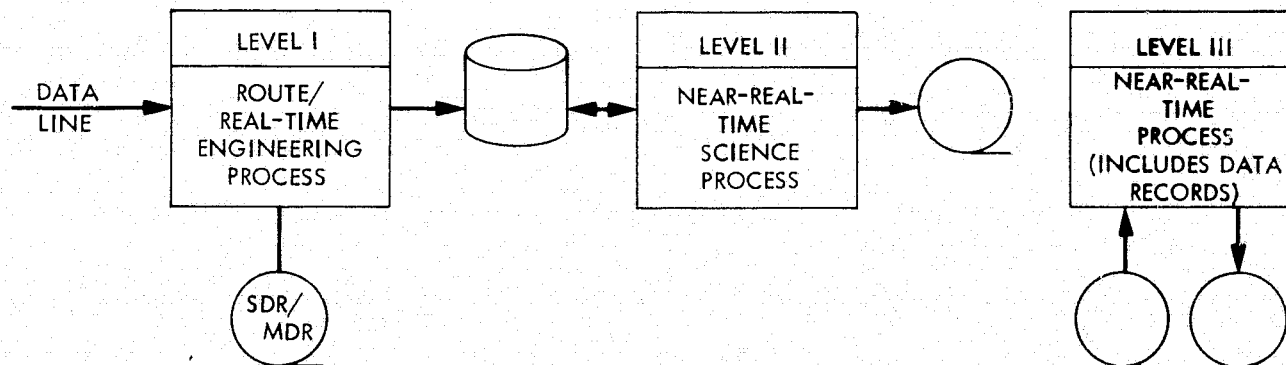
Fig. E-5. JOP data processing levels

The plans and procedures by which JOP mission operations will be conducted will be documented in the <u>Space Flight Operations Plan (SFOP)</u> (Ref. E-3). Some of the anticipated characteristics of the SFOP are described in the following paragraphs.

1. <u>Training Philosophy</u>

Personnel training for launch operations will not begin until 90 days prior to launch. Integration of MOS training exercises and the spacecraft test program will be maximized, and the use of a spacecraft computer model will be minimized.

Personnel training for Probe delivery, orbit insertion, and orbital operations will begin 90 days prior to Jupiter Orbiter Insertion (JOI) and last for 60 days.

2. <u>Sequence Design and Implementation</u>

Sequences of events for the acquisition of science data and for performing spacecraft engineering function will be designed and implemented by the Sequence Team. This process involves targeting of science instruments, design of a spacecraft sequence of events, specification of a supporting ground sequence of events, designing the load for the on-board sequencer, generation of required command sequences, and validation of the final sequence.

The following points characterize the sequence design and implementation capability planned for JOP:

(a) All critical sequences (e.g., launch and Probe delivery, JOI) will be planned far enough in advance to allow their use during training for the respective activity.

(b) Probe delivery, JOI, and encounter sequences will be planned prior to launch; however, detailed sequence development will not begin until one year prior to JOI.

(c) For the prime-mission, sequence-development resources available during orbital operations should be sufficient to accommodate encounters as close as 60 days apart.

(d) The Sequence Team will be capable of generating about one sequencer update per month during cruise and one per day during orbital operations.

(e) The Sequence Team will have the capability to accomplish limited sequence redesign to:

(1) Respond to problems identified during training.

(2) Accommodate a retargeting of the spacecraft from the planned mission.

(3) Respond to anomalous spacecraft performance.

(f) Sequences will be designed to afford limited near-real-time flexibility for certain types of changes, including:

(1) Improving instrument pointing parameters from the latest navigation update.

(2) Improving maneuver parameters from the latest available navigation data.

(g) Sequences will be designed to afford a scientifically viable backup mission in the event of:

(1) Adverse ground weather conditions.

(2) GDS problems interfering with scheduled playbacks and real-time data.

(3) Failure of the spacecraft Data Storage Subsystem.

3. Sequence Execution

Sequence execution is carried out by the Mission Control Team. This process involves the overall direction of all real-time operations including operations team and GDS control, as well as command control and spacecraft performance monitoring.

The general operations plan provides for the overall operations direction whenever a spacecraft is being tracked.

a. Command Control. The capability to transmit commands at a density of less than one command per hour will be provided whenever a spacecraft is being tracked. The capability to transmit commands at greater densities will be provided:

      (1) 8 h/day, 2 days/wk, during cruise.

      (2) 8 h/day, 7 days/wk, during orbital operations.

b. Performance Monitoring. System level performance monitoring, augmented by automatic dynamically updated telemetry alarm monitors, will be provided whenever a spacecraft is being tracked.

During normal cruise activity, augmented performance monitoring at the system and selected subsystem level for increased spacecraft activities (e.g., updates, science maneuvers) normally will be provided no more than 8 h/day, 5 days/wk.

4. Data Acquisition and Delivery

The data acquisition plan is determined largely by navigational requirements and is tempered by tracking-station availability. Anticipated coverage is given in Table E-3.

Table E-3. JOP projected coverage profile

Mission phase	26- or 34-m subnet	34- or 64-m subnet
I Launch to launch + 30 days	Continuous	1 tracking cycle/10 days
II Launch + 30 days; JOI - 150 days	1 8-h pass/wk	1 tracking cycle/month
III JOI - 150 days - JOI - 90 days	1 8-h pass/wk	1 tracking cycle/wk
IV JOI - 90 days; JOI + 30 days	None	Near-continuous
V JOI + 30 days and each subsequent encounter + 7 days to next encounter - 7 days	None	2 tracking cycles/w
VI Encounter - 7 days Encounter + 7 days	None	Near-continuous

NOTE: Near-continuous tracking cycles require TCM - 10 days to TCM + 10 days.

4. Tracking Data Acquisition

In accordance with the profile specified in Table E-3, the following types of tracking data will be acquired:

(a) Doppler (S-band at 26 m, S- and X-band at 64 m).

(b) Planetary ranging with 64-m net.

(c) Simultaneous S/X band doppler, ranging and DRVID at 64 m.

(d) Near-simultaneous radiometric S/X ranging during overlap periods of tracking cycles.

(e) Open-loop doppler data at 64-m stations for occultation measurements.

5. Telemetry Data Acquisition

In accordance with the profile specified in Table E-3, all telemetry rates and modes, subject to telecommunications performance, can be acquired. During cruise, the data rate will be 2 kb/s on the S-band carrier supplying engineering data and will be acquired by the 26-m or 34-m network. Once per week, the X-band subcarrier will be turned on, as required, in conjunction with a 64-m station track, and stored data will be read out at rates of up to 128 kb/s. For all orbital operations, the 64-m stations will be required to acquire the high-rate science data.

F. DEVELOPMENT PLANS

A general MOC schedule is provided in Fig. E-6. The following guidelines apply to MOC development:

(1) There will be no development for optical navigation.
(2) Capabilities not required until encounter will be implemented after launch.
(3) The development of the MOS real-time telemetry processor will be based on the Spacecraft System test processor, and the design of that processor will be a joint MOS/Spacecraft test effort.
(4) The design and implementation of the MOS/Spacecraft System test level I and science level II processors will be a combined effort resulting in an optimized system design.

1. MCCC Planned Capabilities

a. General. The JPL will provide a JOP MCCC which will be capable of meeting the general JOP MOS requirements listed in this document. Formal commitments and applicable exceptions and constraints will be documented in the MCCC Support Plan (MSP) (Ref. E-4) for JOP to be produced in response to the SIRD.

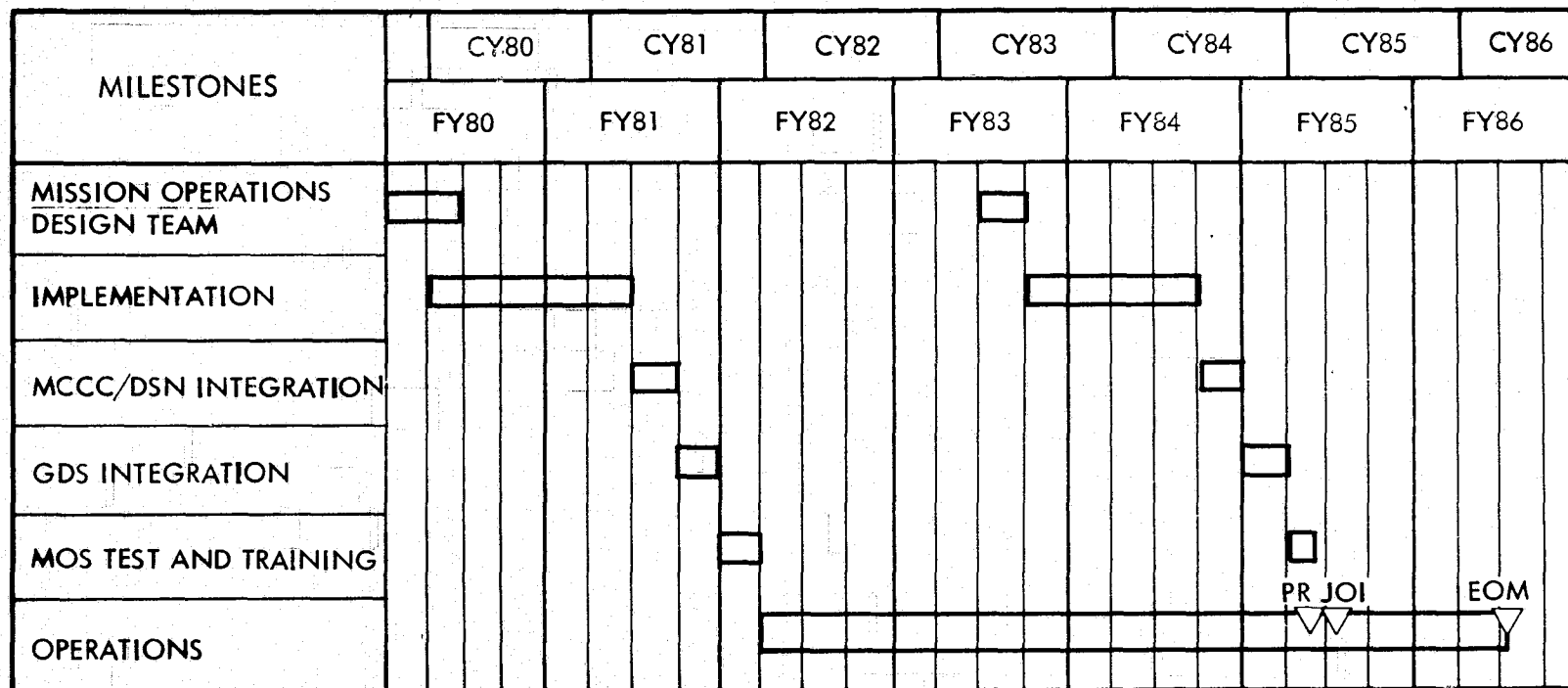| MILESTONES	CY80	CY81	CY82	CY83	CY84	CY85	CY86
	FY80	FY81	FY82	FY83	FY84	FY85	FY86
MISSION OPERATIONS DESIGN TEAM							
IMPLEMENTATION							
MCCC/DSN INTEGRATION							
GDS INTEGRATION							
MOS TEST AND TRAINING							
OPERATIONS						PR JOI	EOM

Fig. E-6. JOP MOS schedule

The JOP MCCC includes the buildings, internal communications, computing and display capabilities, personnel, and procedures through which negotiated support will be provided to the JOP MOS, including the real-and non-real-time support of science operations. Specifically, the MCCC comprises the:

(1) Physical plant: The JOP Mission Support Area (MSA) has tentatively been assigned to occupy the third floor of JPL Building 264 (approximately 12,000 sq ft). JPL Building 230 will primarily be used to house the computers and MCCC Operations Team supporting JOP flight operations.

(2) Computing capabilities: Provided by computing systems which receive/transmit, process, and display telemetry, radiometric data and monitor and command data from/to the Project, DSN, and JSC in support of STS and flight operations. Various parts of these systems perform operations in real, near-real, and non-real time. All real-time data processing will be accomplished in dedicated Modcomp IV minicomputers. Non-real-time analysis and radiometric data processing will be accomplished on the existing UNIVAC 1108 computer through the launch and early cruise phases. The 1108 will then be replaced by a compatible computer. The transition will be accomplished at no cost to the Project. Project real-time software development will be supported by a separate development system on a scheduled basis, as required. The preliminary computer delivery schedule to support MOS software development, GDS testing, and MOS personnel training/testing is shown in Fig. E-7.

(3) Operations: the MCCC will ensure that the functional capability of all data systems is available to the Project user in the operational environment. Project cost for use of the data systems will generally be based on a "shift per week" unit (except for development) and will cover all charges for dedicated computer operating personnel, supporting personnel, maintenance, and supplies. In addition, Division 91 will provide,

MILESTONES	1978	1979	1980	1981
	J F M A M J J A S O N D	J F M A M J J A S O N D	J F M A M J J A S O N D	J F M A M J J A S O N D
1 JOP REQUIREMENTS				
2 DEVELOPMENT SYSTEM AVAILABILITY		▽	S/C TEST	
3 MCCC DATA SYSTEM AVAILABILITY		▽②	▽③ ▽④ F/S	
4 DATA SYSTEM OPERATIONAL			▽ S/C ▽ GDS	
5				
6 MCCC COMPUTER DELIVERIES	▽①			
7 DEVELOPMENT SYSTEM		▽②		
8 SPACECRAFT TEST			▽③	
9 FLIGHT SUPPORT			▽④	
10				
11				
12				
13				
14				
15				
16				
17				
18				
19				

① ALL COMPUTERS MODCOMP IV.
② SIMULATION (TEST DATA GENERATOR).
③ TELEMETRY IMAGING.
④ SIMULATION (MATHEMATICAL MODEL), COMMAND, TELEMETRY, MONITOR/OPERATIONS.

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Fig. E-7. Key JOP MCCC Data System development milestones

1, 3

IMAGING SYSTEM

IMAGE GATHERING
IMAGE SUMMARIZATION
ENGINEERING DATA EXTRACTION
FORMAT EXPANSION
DATA RECORDS AND SUMMARIES
IMAGE PROCESSING
FILM PRODUCT GENERATION
DISPLAYS

SYSTEM
TEST

USERS

1, 2, 3

TEST AND TELEMETRY SYSTEM

FRAME SYNCHRONIZATION
DECODE AND CORRECT
EXTRACTION
DECOMMUTATION
NON-IMAGE SCIENCE PROCESSING
ENGINEERING PROCESSING
DATA DISPLAY

USERS

SYSTEM
TEST

3

DATA RECORDS SYSTEM

DATA COLLECTION
FILE STORAGE AND RETRIEVAL
DEMAND PROCESSING
  AND DISPLAY
DATA RECORDS PRODUCTION
COMMUNICATIONS
STATUS AND PERFORMANCE
  DATA GENERATION
IDR PROCESSING
SPECIAL PROJECT NON-REAL-TIME
  PROCESSING

USERS

1, 2, 3

OPERATIONS CONTROL SYSTEM

CONFIGURATION OF MCC RESOURCES
ROUTING AND ACCOUNTABILITY
  OF DATA/MESSAGES
GENERATION OF OPERATIONS CONTROL
  SYSTEM CONFIGURATION AND
  PERFORMANCE DATA
COLLECTION AND DISPLAY OF
  MCCC DSN AND GCF
  CONFIGURATION AND PERFORMACE
  DATA
SUPPLY OPERATIONS SUPPORT SERVICES

STS HSD
GCF HSD,
WBD AND
MWBD
INTERFACE

WBD   HSD   USERS

2, 3

COMMAND SYSTEM

CONTROL
SELF TEST
DATA INPUT VALIDATION
DATA FORMATTING
STATUS MONITORING
FILE MANAGEMENT
DATA COMMUNICATIONS
DISPLAY
DATA ACCOUNTABILITY
PERFORMANCE ANALYSIS

USERS

1, 2, 3, 4

SIMULATION SYSTEM

GENERATE SIMULATED
  STATION DATA
GENERATE SIMULATED
  TRACKING DATA
PROCESS COMMAND DATA
GENERATE SIMULATED
  PROJECT TLM STREAMS
PROVIDE INTERFACE AND HOST
  ENVIRONMENT FOR PROJECT
  TELEMETRY MATHEMATICAL MODELS
GENERATE SYSTEM STATUS AND
  PERFORMANCE DATA
PROVIDE LONG LOOP SCA
  CONTROL
PROVIDE TELEMETRY DATA
  VALIDATION

USERS

2

TRACKING SYSTEM

ACCEPT, VALIDATE AND
REFORMAT TRACKING DATA
FROM JSC

USERS

2, 3

NON-REAL-TIME PROCESSING

SCIENCE ANALYSIS
ACCEPT, VALIDATE AND
  REFORMAT TRACKING
  DATA FROM DSN
NAVIGATION
MISSION SEQUENCE
FLIGHT ANALYSIS

USERS

1. REQUIRED FOR SPACECRAFT SYSTEM TEST.
2. REQUIRED FOR LAUNCH, CRUISE PHASE.
3. REQUIRED FOR ORBITAL, PROBE PHASES.
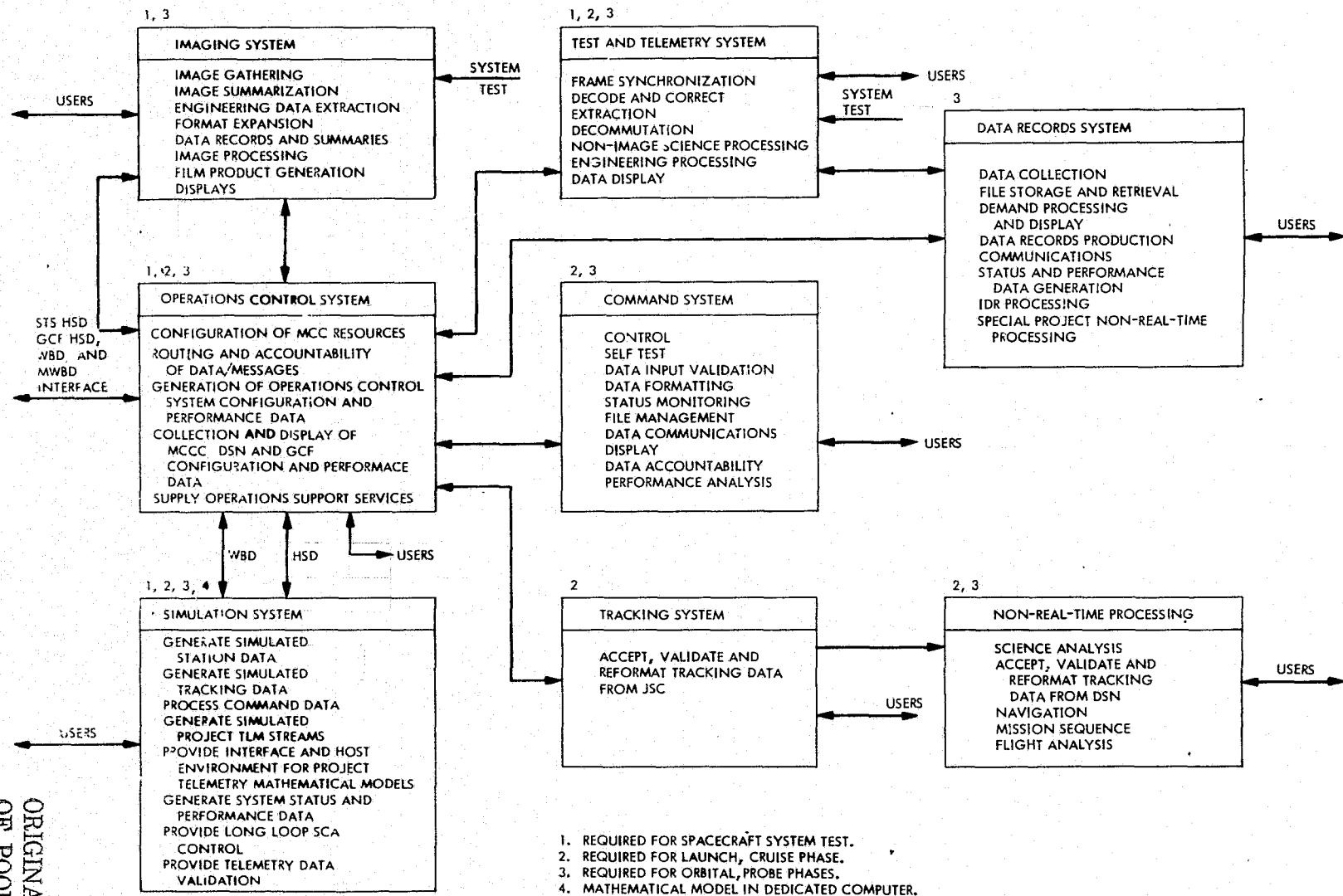4. MATHEMATICAL MODEL IN DEDICATED COMPUTER.

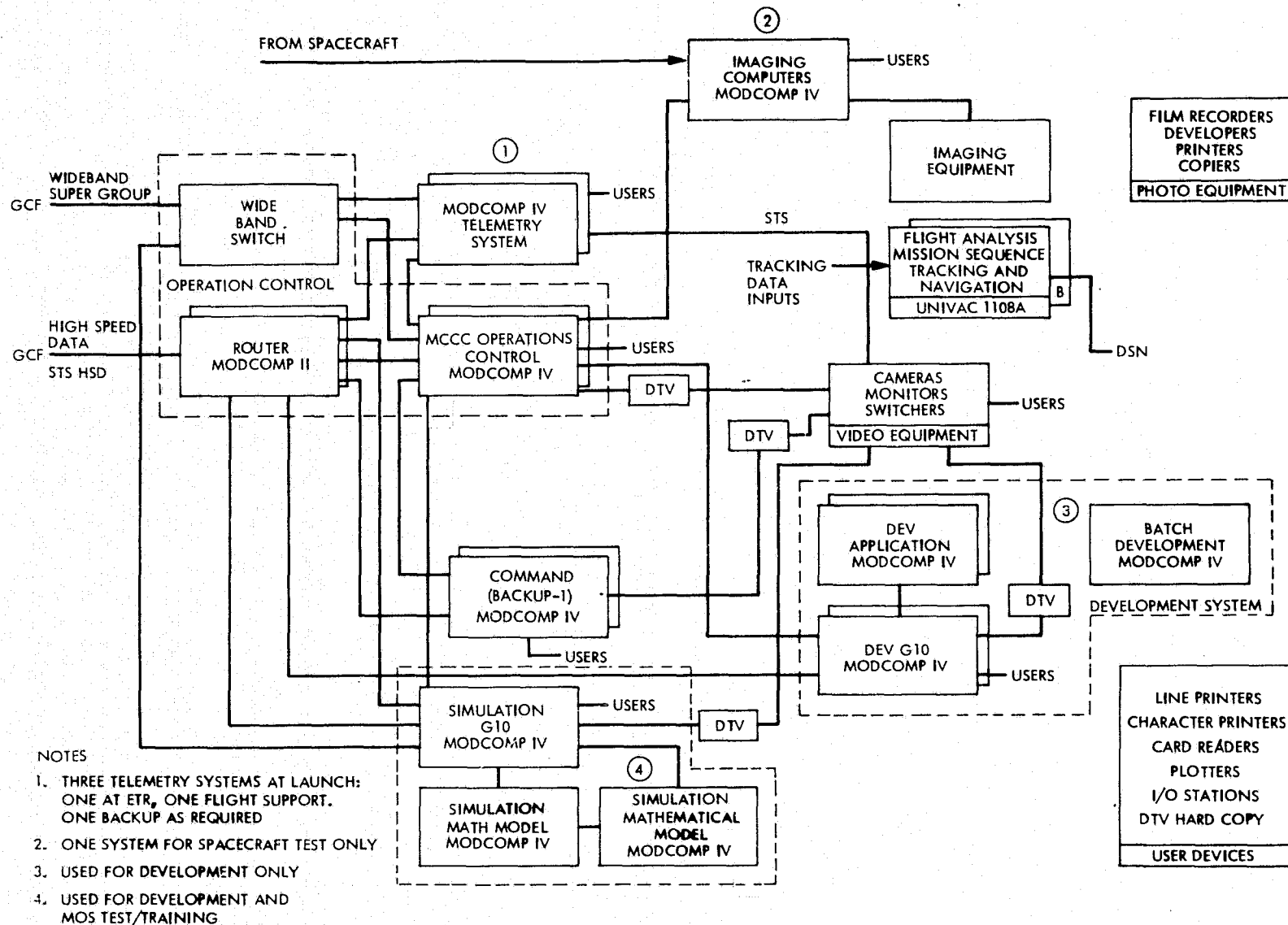Fig. E-8. JOP Data System functional configuration

as required, Project-directed and funded mission support personnel to perform certain related Project-required operational functions (e.g., MSA terminal operation).

    b.   <u>MCCC Data System</u>. The MCCC Data System for JOP will exist in three configurations: spacecraft test, launch/cruise, and orbital. The real-time flight support functions will be provided on Project and functionally dedicated minicomputers. Figure E-9 shows the relationship of these systems. Most of the non-real time support of Flight Projects, as well as non-flight activities, will be supported with large scale computers. The MCCC real-time flight support functions are performed in seven functional systems as described below. In addition to these systems, there are a number of support systems, including a separate real-time development system, which can be used for any software or hardware development related to the mission.

    (1)   The Test and Telemetry System processes all non-imaging telemetry data, both science and engineering. Data processing support is provided for Spacecraft System test and all mission operations phases.

    (2)   All spacecraft command data handling capability utilizes the Command System.

    (3)   The Tracking System provides a backup capability to JSC for first orbit determination and predicts.

    (4)   The Imaging System provides the data handling capabilities to acquire and produce picture images for Project support.

    (5)   The Operations Control System provides visibility and control of the MCCC in support of the Flight Projects, and the interface to the DSN.

    (6)   The Simulation System provides capabilities for support of software development, testing the entire MCCC data system, and mathematical models of spacecraft for test and training.

    (7)   The Data Records System (DRS) provides the capability to process master and/or intermediate data records in non-real time for use by Principal Investigators and others.

NOTES

1. THREE TELEMETRY SYSTEMS AT LAUNCH:
   ONE AT ETR, ONE FLIGHT SUPPORT.
   ONE BACKUP AS REQUIRED

2. ONE SYSTEM FOR SPACECRAFT TEST ONLY

3. USED FOR DEVELOPMENT ONLY

4. USED FOR DEVELOPMENT AND
   MOS TEST/TRAINING

Fig. E-9. JOP Data System configuration for Spacecraft System test and launch/cruise

The MCCC Data System Development Plan (DSDP) will describe the implementation of these systems.

c. <u>Spacecraft Test Data System Configuration</u>. This configuration will consist of Telemetry and Imaging Systems and a simulation subsystem as shown in Fig. E-10 and described below:

(1) The Telemetry System will process hardline inputs from the spacecraft, telemetered data at spacecraft data rates, and generate displays and records as follows:

    (a) Level 1 processing of engineering and science status data.

    (b) Level 3 processing of science data.

    (c) Generation of a separate engineering data MDR containing 99% of MCCC received data.

    (d) Generation of quick-look science EDRs from the MDR only, containing 99% of MCCC received data.

(2) The Imaging System will format images in near-real time from telemetered data including frame synchronizing, labeling, image formatting, display, and recording. The records produced will contain 99% of MCCC received data.

(3) The Simulation Subsystem will be used to generate test data for development of imaging and telemetry systems.

d. <u>Launch/Cruise Data System Configuration</u>. This configuration will consist of dedicated telemetry, command, and tracking systems, an operations control system, a simulation subsystem, and a non-real time processing capability as shown in Fig. E-10.
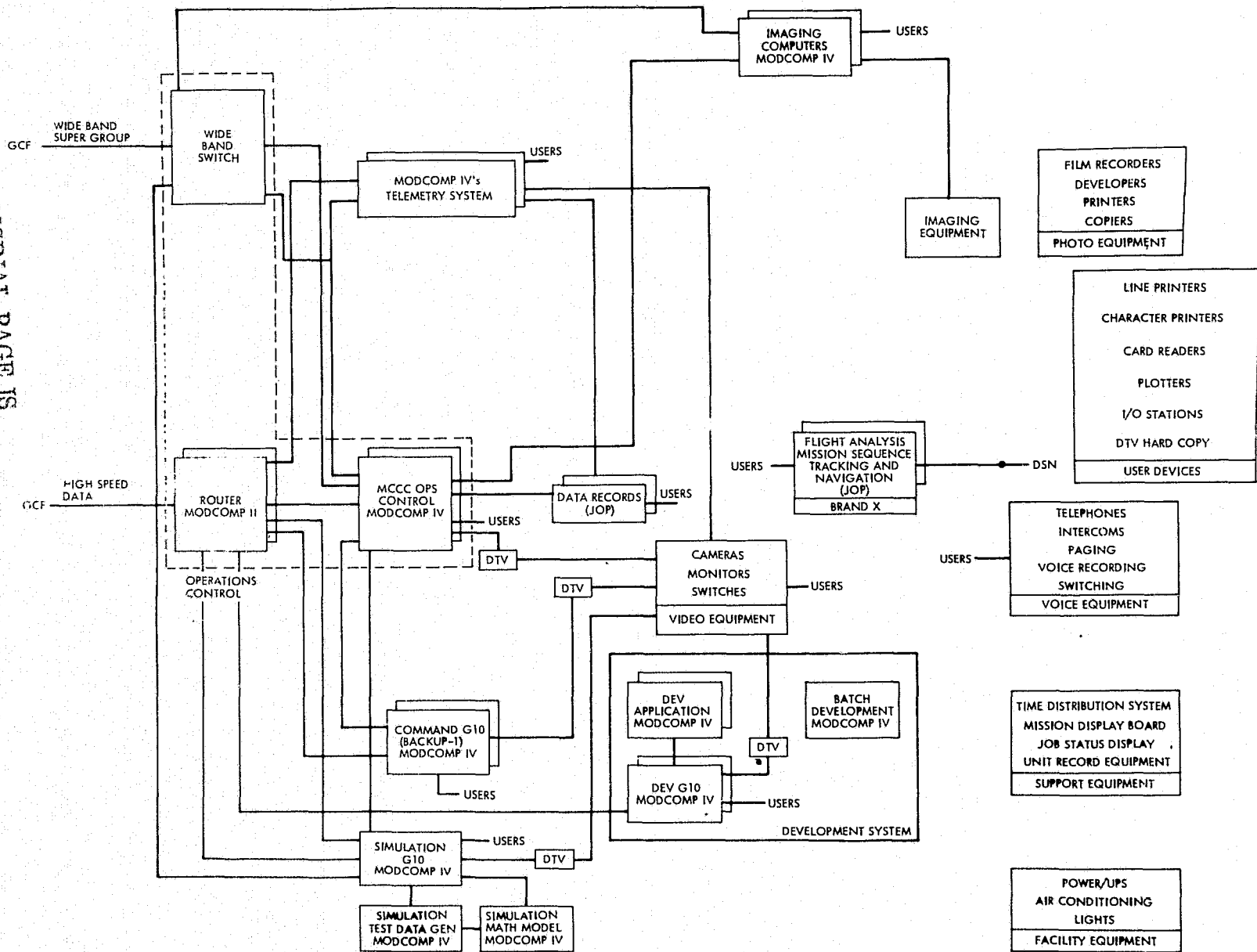
Fig. E-10. MCCC Data System JOI/orbital configuration

During the launch period while the spacecraft is mated to the Shuttle Interim Upper Stage (IUS), the interfaces for telemetry and command will be through a 7.2 kb/s high-speed data line to JSC.

> (1) The Telemetry System will process engineering telemetry data and generate displays and engineering data records as follows:
>
> > (a) Level 1 processing of engineering and science status data.
> >
> > (b) Generation of separate engineering MDRs and "quick look" science MDRs/EDRs containing 99% of MCCC received data.
>
> (2) The Command System will provide the capability to command the spacecraft either manually or from tape input, and generate displays and data records.
>
> (3) The Tracking System will process tracking data received from JSC for the first orbit calculations.
>
> (4) The Operations Control System will control and monitor the MCCC real-time system operations.
>
> (5) The Simulation Subsystem will support telemetry development and system level tests and, with the spacecraft mathematical model, support MOS test and training.
>
> (6) Non-real time processing will be accomplished by a large scale GP computer used for non-real time processing of navigation, mission sequence, and spacecraft analysis.

e. Orbital and Probe Phase Data System Configuration. This configuration will consist of the Command, Tracking, Operations Control, and Non-Real Time Systems as used in the launch/cruise phase and new or modified telemetry, imaging, data records, and simulation systems as shown in Fig. E-10.

> (1) The Telemetry System will add science data processing to the engineering data processing capability of the launch/cruise phase as follows:

(a) Level 1 processing of engineering and science status data.

(b) Level 2 processing of science instrument data.

(c) Generation of a science data MDR containing 99% of received data.

(2) The Imaging System will generate processed images in near-real time, including frame synchronizing, error correction, first-order image enhancement, image formatting and display, labeling and recording. The records produced will contain 99% of MCCC received data.

(3) The Data Records System will provide for the non-real time generation of science and imaging data records for the user as follows:

(a) Science MDRs will be filled from IDRs as necessary to achieve quality, quantity, and continuity requirements specified for critical and non-critical mission phases.

(b) EDRs will be generated from MDRs and filled from IDRs as necessary to achieve the quality, quantity, and continuity requirements specified for critical and noncritical mission phases.

(4) The Simulation Subsystem will support telemetry and imaging development and system level tests, and, with the spacecraft mathematical model, MOS test and training.

(5) The non-real time processing will be accomplished by a large-scale GP computer used for the non-real time processing of science data as well as navigation, mission sequence, and spacecraft analysis.

The Command and Operations Control Systems will be the same for this mission phase as for launch and cruise.

f.   <u>MCCC Data Operations</u>. Operational support for JOP involves operation of both the multimission and dedicated portion of the MCCC Data System in support of MOS.

The control and coordination functions of the MCCC Data System are carried out on three distinct levels. Level 1 are functions associated with the MCCC Operations Control Chief (OPSCON). Level 2 consists of controller functions for the MCCC distributive Computer System. These functions will be executed from operating positions co-located with OPSCON and will provide the subsystem technical expertise required to direct the total operating organization. Level 3 is the coordination function for each data system. This function establishes configuration, performs recovery actions, and provides the level 2 function with required status and analysis of the data system functions.

The functional responsibilities of OPSCON are:

(1) Direct, monitor, and control all activities within the MCCC in support of mission operations.

(2) Interface with DSN, Project, and MCCC personnel to ensure effective support for all projects.

(3) Coordinate real-time schedule changes, as necessary within committed resources.

(4) Maintain cognizance of overall MCCC status, significant events, anomalies, and corrective actions.

(5) Report MCCC status to Project and Management personnel as required.

(6) Maintain MCCC operations log of significant events per Failure Accountability System Office (FASO) requirements and for performance monitoring.

(7) Interface with non-real time system personnel for Project-related activities.

Although the level 1 control function consists of only the OPSCON, the level 2 system controller functions of the MCCC '76 organizations consist of:

(1) Operations monitor and command controller. The systems under the cognizance of the operations monitor and command controller include the routers, operations, and command data systems.

(2) Telemetry operations controller. The systems under the cognizance of the telemetry systems controller consist of the telemetry and imaging data systems.

(3) Facility and communications controller. The facility and communications controller (FACCOM) is cognizant of all facility and communications support subsystems.

(4) Non-real time operations controller. The non-real time operations controller is also cognizant over the simulation subsystems and is only co-located with the OPSCON as required during heavy test activities.

(5) Simulation operations controller. The non-real time and the simulation operations are not necessarily co-located with the OPSCON at all times.

The relationships between the various control and coordinating functions is given in Fig. E-11.

2. Tracking and Data System Planned Capabilities

The Tracking and Data System (TDS) plans and capabilities described herein involve primarily the DSN portion of the TDS as projected into the first half of the 1980 decade. Particular attention is given to planned changes in the DSN as seen from a facility- and data-system point of view.

For purposes of this document, the DSN is first described as consisting of three major facilities: (a) the Deep Space Stations (DSS), (b) the Ground

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                                    │  OPSCON  │
                                    └────┬─────┘
    ┌──────────┬────────────────┬────────┼──────────┬─────────────────────────┐
┌────────┐ ┌──────────┐ ┌────────────┐ ┌─────────┐              ┌────────────┐
│  TLM   │ │OPERATIONS │ │ SIMULATION │ │ FACCOM  │              │ NON-REAL-  │
│CONTROL │ │   AND     │ │  CONTROL   │ │CONTROL  │              │   TIME     │
│        │ │ MONITOR   │ │            │ │         │              │ CONTROL    │
│        │ │ CONTROL   │ │            │ │         │              │            │
└───┬────┘ └────┬──────┘ └─────┬──────┘ └────┬────┘              └─────┬──────┘
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| TLM OPERATIONS | IMAGE OPERATIONS | COMMAND OPERATIONS | OPERATIONS CONTROL AND ROUTER OPERATIONS | SIMULATION OPERATIONS | FACILITY OPERATIONS | COMMUNI-CATIONS OPERATIONS | | 1108 OPERATIONS | BRAND X OPERATIONS |

MINICOMPUTERS;
REAL-TIME OPERATIONS

LARGE, GENERAL PURPOSE
COMPUTERS;
NON-REAL-TIME OPERATIONS
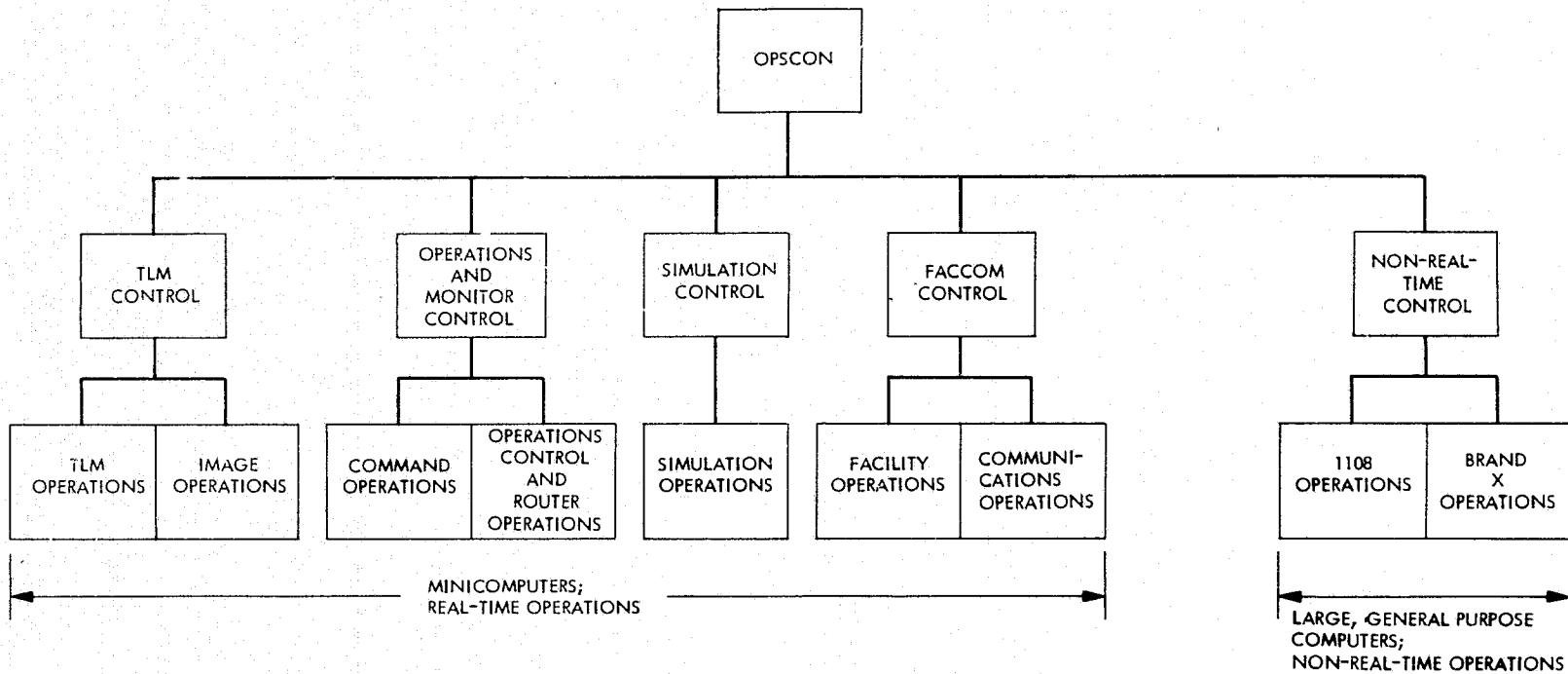
A. A SIM CONTROL ELEMENT WILL BE CO-LOCATED IN THE CENTRAL OPERATIONS CONTROL AREA
   ONLY AS REQUIRED DURING HEAVY TEST ACTIVITY.
B. SIM WILL BE MINICOMPUTER-BASED AFTER JANUARY 1979.

Fig. E-11. MCCC functional operations organization

Communications Facility (GCF), and (c) the Network Operations Control Center (NOCC).

a. Deep Space Stations (DSS). It is anticipated that the DSS facility will continue to consist of nine operational stations, with California, Australia, and Spain each having a complex of three stations. These stations are further viewed as DSN subnetworks as a function of antenna size and basic capabilities. The 64-m subnet consists of three 64-m diameter antennas, one at each of the three longitudes. This subnet will be capable of receiving both S- and X-band signals for the recovery of telemetry data and the generation of dual frequency radiometric data. For commanding spacecraft and for coherent radiometric data generation, the uplink carrier frequency will be at S-band only. Hydrogen masers will serve as the primary frequency and timing standard with Cesium standard serving as backup. Transmitter power will be variable up to 100 kW in the 64-m subnet.

The 26-m subnet consists of three 26-m diameter antennas, one at each of the three longitudes. This subnet will be capable of receiving and transmitting at S-band only for the purpose of telemetry data acquisition, command transmission, and for radiometric data generation. It is planned that Cesium standards will replace the current Rubidium standards as the primary frequency and timing reference. However, it is noted that the 26-m stations (DSS 42 and DSS 61), which are conjoint with the overseas 64-m stations, will have access to the 64-m station's hydrogen maser standard. DSS 11 represents the only stand-alone 26-m station in this subnet. Maximum transmitter uplink power will be maintained at 20 kW.

The DSN's third and final subnet will consist of three 34-m stations (DSS 12, 44, and 62), one at each of the three longitudes. The DSN plans to upgrade these existing 26-m stations during the 1978-1981 time period to provide an S-band and X-band receive capability and an extension of the antenna size to 34 m. These stations will be implemented with Cesium standards as a primary frequency and timing reference. Uplink power may be increased above the current 20 kW, subject to design constraints. The uplink carrier frequency will be at S-band only. In view of the gain and dual frequency

characteristics of this subnet, it is expected that mission operations planners will, to the maximum extent possible, design flight sequences that are supportable by this subnet rather than the 64-m subnet.

b. <u>Ground Communications Facility (GCF)</u>. The GCF, along with services provided for by the NASA Communications System (NASCOM), will continue its primary function of data and voice communications between the DSS and the Network and Mission Operations Centers at JPL. To accomplish this function, communications terminals will be maintained at the DSS and at JPL. Connector panels in the GCF Central Communications Terminal (CCT) at JPL will represent the primary physical interface between the DSN and the users for the communication of real-time data. The DSN will not be responsible for the communication of data to the MCCC-Project equipment beyond this interface or between MCCC Project processors within the Mission Operations Support Areas. Increased use of satellite and digital communications techniques are expected within the next five years. Error correction by near-real-time retransmission will be implemented in the GCF high-speed data circuits during the 1977-1978 period. It is expected that similar error correction techniques will be applied to GCF wide-band data circuits in the 1979-1981 time period. Given the improved quality of digital satellite circuits and error correction, it is expected that at least 99% of the data handled by the GCF will be delivered error-free essentially in real time.

Users should be prepared to accommodate future increases in the high-speed circuit rate from 7.2 kb/s to 9.6 kb/s. Furthermore, for standardization purposes, the GCF will be pressed to convert its 1200-bit high-speed data block to NASCOM's standard 4800-bit block. Decisions regarding such block size changes will be made well in advance of future projects' implementation periods. The 56 kb/s digital wideband circuits, which will utilize 4800-bit blocks during the MJS '77 mission, should remain essentially unchanged for the foreseeable future. Depending on the availability of satellite circuits, their cost, and circuit lease funding, the GCF, with NASCOM, will be capable of providing multiple 56 kb/s circuits to selected DSS for the real-time communications of telemetry data rates of approximately 130 kb/s. The cost of such circuits normally permit their use only for short periods during

critical planetary sequences. The standard complement of circuits serving each DSS will normally include one voice, one teletype, and one high-speed circuit. Stations in the 34-m and 64-m subnets will also include 56 kb/s wideband circuit capabilities.

The GCF will also perform data logging functions at the DSS and at the Central Communications Terminal. At the DSS, a computer-based GCF processor will produce a digital data record of all data transmitted from the DSS via the high-speed data circuits. In addition, this record will contain all command system messages received from the operations center. The DSN may access this record for the recovery of any data lost because of transmission difficulties. The GCF Central Communications Terminal will log all data received from the DSS. These records will serve as the primary source of data for the DSN's production of non-real-time deliverables, such as the telemetry intermediate data record (IDR).

c. Network Operations Control Center (NOCC). The NOCC, located at JPL, provides the DSN Operations Control Team with capabilities for centralized operational direction, control, and monitoring of the network. These tasks are performed in real time as a function of interfacing Flight Projects and their mission directions. Since the GCF provides simultaneous communication of data to both the NOCC and the Mission Operations Center, these NOCC functions are considered to be performed "off-line", and they are not in series with spacecraft data manipulation for mission operations purposes. Also, using NOCC capabilities, the Network Operations Team will generate the DSN's sequence of events and DSS predictions based on inputs from the Project in accordance with standard defined interfaces. NOCC capabilities will be employed to produce IDR deliverables in non-real time. The Network Operations Control Team will routinely access primary source records in the DSN as necessary to generate these deliverables within 72 hours. Through the routine execution of data merging procedures, the DSN's telemetry IDR is expected to contain at least 99.5% of the data available on primary record sources. In no way should the Mission Operations System use the IDR for support of real-time or near-real-time operations; the primary purpose of the IDR is to support non-real-time production of data records, as required.

d.   Facility Operations Considerations. DSN facilities will be staffed so that 24-h per day operations may be conducted when required by the Project. In view of possible Mission Operations System limited staffing plans, the following DSN positions should be considered in designing mission activities: (1) the DSN will not command the spacecraft in the Project's absence; (2) if scheduled, the DSN will track the spacecraft and acquire data, even if the MOS were not staffed; (3) in no case will the DSN attempt to observe engineering telemetry data for spacecraft performance monitoring purposes, (4) the DSN will, however, monitor normal DSN observables (signal level, polarization, carrier lock and subcarrier lock, decoding capabilities, etc.) at all times and would report any discrepancies to designated stand-by operators.

To describe the DSN in more detail from an end-to-end network data system point of view, the DSN currently consists of five data systems: (1) tracking, (2) telemetry, (3) command, (4) monitor and control, and (5) test and training. Plans are to develop a sixth system for radio science during the MJS '77 time period. These six systems relate to the primary data types involved in mission and network operations.

a.   DSN Telemetry. To support mission operations and experimenter requirements, the DSN telemetry system shall acquire and handle all telemetry data and will include the following capabilities and characteristics: (1) S- and X-band receive at 64-m and 34-m DSS, S-band receive at 26-m DSS; (2) carrier and subcarrier detection; (3) symbol synchronization at the maximum of 250 kb/s; (4) convolutional decoding; (5) formatting for GCF transmission and DSS recording; (6) digital recording at the DSS; (7) real-time transmission via high-speed data and wideband data circuits to the NOCC and Mission Operations Center or tape relay transmission, if telemetry rates exceed GCF circuit capabilities; (8) system configuration control and monitoring via the NOCC; (9) IDR production within 72 hours at the NOCC.

b.   DSN Tracking System.   To support mission navigation and experimenter requirements, the DSN tracking system will generate radiometric data and will include the following capabilities and characteristics: (1) transmit S-band uplink with ranging modulation as required; (2) receive S- and X-band at 64-m and 34-m stations, receive S-band at 26-m stations; (3) generate doppler data; (4) generate range data; (5) generate calibration data for determining transmission media effects, including differential S-X band, DRVID, ground weather data, and Faraday rotation data, subject to availability of a stationary Earth satellite to be tracked; (6) format these radiometric data for recording and transmission to the NOCC and Mission Operations Center, and (7) on-site recording and non-real-time recovery of data from recordings, as required.   Variations of these basic radiometric data types, such as near-simultaneous ranging and simultaneous interference tracking may be obtained through the application of special procedures and proper scheduling of the DSS.

During 1979-1981, the DSN plans to achieve operational status of Very Long Baseline Interferometry (VLBI) techniques involving a two-station baseline for the determination of universal time and polar motion information.   VLBI data collected at the DSS will be correlated at the NOCC to produce the standard universal time and polar motion observables for delivery to the user.   Furthermore, it is planned that VLBI technology will be employed for interstation frequency and time synchronization rather than the current moon-bounce system.   An operational variation of VLBI (sometimes called Delta VLBI), which involves tracking both the radio-star source and the spacecraft to determine the star-spacecraft angular separation, may be a built-in option within the VLBI capability.   However, in this case, the DSN would produce only the correlated parameters, and the user would be expected to solve for the desired separation angle observables.

In addition to the standard real-time transmission of radiometric data to the Mission Operations Center, the DSN plans to implement the IDR. This capability will respond to planetary tracking system requirements generated by the Engineering Coordination Team.   Preliminary specifications

call for the following: (1) the IDR will be available within 30 min of a request in critical phases and will contain 95% of the data transmitted in real time from the station; (2) the IDR will be available per an agreed upon schedule during the cruise phases (ideally, once per week) and will contain 98% of the data requested to be transmitted from the station; and (3) IDR outages of greater than 15 min will be filled, if requested prior to IDR generation. Also, the Network will provide one digital TV channel output port to the MCCC to drive slave displays in Project navigation areas. Upon failure of the DSN digital TV display, it is also planned to provide hard copies of data available for display within 30 min if requested.

c. DSN Command System. To support mission operations commanding of the spacecraft, the DSN command system will have the following capabilities and characteristics: (1) NOCC configuration control and monitoring of the system, (2) real-time high-speed data interfaces with the Project for command transmission to the DSS, (3) increased command storage at the stations to facilitate moving away from the current closed-loop real-time command operations toward the "store and forward" mode, (4) command storage at the DSS will be some eight files, each capable of storing 255 command elements of 800 bits, (5) a command transmission rate of up to 1,000 b/s may be possible, as necessary to support command activities within given time constraints, (6) DSS confirmation of accurate command transmissions including notification to the Mission Operations Center, and (7) recording of all command messages and responses within the DSN for later recall, if required.

d. DSN Radio Science System. In 1976, the DSN initiated a system approach to the development of capabilities required to support spacecraft radio science requirements. Based on a coordinated input from the radio science community, a radio science system will be implemented having general characteristics as follows: (1) implementation will involve primarily the 64-m subnet, (2) S- and X-band open-loop and phase-lock loop reception, (3) recovery of both left-circular and right-circular components, (4) replacement of current analog occultation recorders with wideband digital recorders, (5) routine recovery and recording of related station receiver parameters, and (6) delivery of a

computer-compatible digital record of occultation data. Close coordination with the radio-science community shall continue through this system's development.

e. <u>DSN Monitor and Control System</u>. By 1978, this system will essentially be an internal DSN tool for controlling Network configurations and for evaluating Network performance. Monitor data blocks transmitted from the DSS to the NOCC will not be committed for Project use. A portion of the data normally contained in DSS monitor blocks and generally required for the Project's completion of EDRs shall either be contained in the deliverables from the other data systems or will be provided via a special status block generated in the NOCC and transmitted to the Mission Operations Center. For ground data system information purposes, the MCCC may slave DTV transmission-display hardware to the DSN DTV interface for the purpose of obtaining the Network status summary, which is available to the DSN Operations Control Team.

f. <u>DSN Test and Training System</u>. This system provides internal capabilities required for Network test and training activities. Designated computers at the DSS and NOCC shall provide for the generation of all committed telemetry data types and modes for insertion into the RF and digital portions of the operational data subsystems. Plans call for the replacement of existing DSS test and training computers in the 1978-1979 period. Consequently, DSN internal capabilities should be significantly enhanced. To support Project requirements for long-loop simulation activities using realistic, dynamic data, this system will continue to provide a standard interface by which Project-generated data may be injected into the DSS subsystems.

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