**NASA CONTRACTOR REPORT** 166430

# MARS ORBITER CONCEPTUAL DESIGN STUDY
# FINAL REPORT



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# Mars Orbiter Conceptual Systems Design Study
# Final Report


TRW, Inc.
Redondo Beach, California



Prepared for
Ames Research Center
Under Contract NAS2-11223



NASA
National Aeronautics and
Space Administration

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## ACRONYMS

| | |
|---|---|
| AVG | average |
| AU | astronomical unit |
| AMO | air mass zero |
| AH | ampere hour |
| AGC | automatic gain control |
| ACS | Attitude Control System |
| ANC | active nutation control |
| AKM | apogee kick motor |
| bps | bits per second |
| BOL | beginning of life |
| BER | bit error rate |
| BASD | Ball Aerospace Development |
| CEA | Control Electronics Assembly |
| c.g. | center of gravity |
| C | centigrade |
| C&DH | Command and Data Handling |
| CNR | carrier-noise ratio |
| CCS | Command and Control System |
| CD/P | Command Decoder/Processor |
| CMP | Command Message Processor |
| CPU | Central Processor Unit |
| DSN | Deep Space Network |
| DSCS II | Defense Satellite Communications System II |
| db | decibel |
| dBi | decibels above isotropic |
| DSP | Defense Support Program |
| dBm | decibels above 1 mW |
| DHS | Data Handling System |
| DMA | Despun Mechanical Assembly |

## ACRONYMS
(Continued)

| | |
|---|---|
| EFD | Electrical Field Detector |
| EMI | Electromagnetic Interference |
| ETR | Eastern Test Range |
| EAA | earth aspect angle |
| EOL | end of life |
| EPDS | electrical power and distribution |
| EIA | Electrical Integration Assembly |
| FIS | Frost Infrared Spectrometer |
| FPI | Fabry-Perot Interferometer |
| FOV | field of view |
| FLTSATCOM | Fleet Satellite Communications |
| FSC | Fleet Satellite Communications |
| GRS | Gamma Ray Spectrometer |
| GRO | Gamma Ray Observatory |
| HGA | high gain antenna |
| Hz | hertz |
| HEAO | High Energy Astronomical Observatory |
| IUS | Inertial Upper Stage |
| $I_t$ | moment of inertia (transverse axis) |
| $I_z$ | moment of inertia (spin axis) |
| ID | identification |
| IR | infrared |
| ISEE-C | International Sun-Earth Explorer-C |
| JSC | Johnson Space Center |
| JPL | Jet Propulsion Laboratory |
| $J_2$ | second gravitational harmonic |
| kg | kilogram |
| km | kilometer |

## ACRONYMS
(Continued)

| | |
|---|---|
| LV | launch vehicle |
| lbf | pound (force) |
| LGA | low gain antenna |
| MRD | Mission Requirements Document |
| MOI | Mars Orbit Insertion |
| m/s | meters/second |
| MSM | Multispectral Mapper |
| Mb | megabits |
| MAG | Magnetometer |
| MGA | medium gain antenna |
| M | meter |
| MLI | Multilayer Insulation Blanket |
| NASA | National Aeronautics and Space Administration |
| NHB | Board |
| NMS | Neutral Mass Spectrometer |
| $N_2H_4$ | hydrazine |
| $NiH_2$ | nickel hydrogen |
| NRZ | non-return-to-zero |
| OIM | Orbit Insertion Motor |
| PPP | Planetary Protection Procedure |
| PMR | Pressure Modulated Infrared Radiometer |
| PCU | power control unit |
| PDU | power distribution unit |
| PCM | pulse code modulation |
| PROM | programmed read-only memory |
| PAM | |
| RA | Radar Altimeter |
| RPA | Retarding Potential Analyzer |
| RF | radio frequency |

## ACRONYMS
(Continued)

| | |
|---|---|
| rpm | revolutions per minute |
| REA | Regulator Error Amplifier |
| RSS | Rotating Service Structure |
| RIU | remote interface unit |
| STS | Space Transportation System |
| SOW | Statement of Work |
| SSUS | Spinning Solid Upper Stage |
| SC | spacecraft |
| SWPA | Solar Wind Plasma Analyzer |
| SAA | sun aspect angle |
| SDA | Shunt Drive Assembly |
| SEA | Shunt Element Assemblies |
| s/s | subsystem |
| SRM | solid rocket motor |
| TCM | trajectory correction maneuver |
| TIMS | Thermal Ion Mass Spectrometer |
| TWTA | Traveling Wave Tube Amplifier |
| TDRSS | Tracking and Data Relay Satellite System |
| TR | tape recorder |
| TCS | Thermal Control System |
| TLM | telemetry |
| TBD | to be determined |
| UVHP | Ultraviolet Atomic Hydrogen Photometer |
| $UVO_3$ | Ultraviolet Altimeter |
| UV | ultraviolet |
| UVS | Ultraviolet Spectrometer |
| VDC | volts, direct current |
| w | watts |
| ZAP | angle between $V_\infty$ vector and sun line |

MARS ORBITER CONCEPTUAL SYSTEMS DESIGN STUDY

FINAL REPORT

# 1. INTRODUCTION

This is the final report of the Mars Orbiter Conceptual Systems Design Study performed by TRW from May to September, 1982, for NASA/Ames Research Center under contract NAS2-11223. The purpose of the study is to develop spacecraft system and subsystem designs at the conceptual level to perform either of two Mars orbiter missions, a Climatology Mission and an Aeronomy Mission. The objectives of these missions are to obtain and return scientific data to increase our knowledge of the planet Mars.

## 1.1 MISSIONS

The Climatology Mission (using a low altitude circular orbit at Mars) concentrates on surface and atmospheric characteristics, while the Aeronomy Mission (elliptical orbit) emphasizes study of the composition of the upper atmosphere and the characteristics of the Mars environment up to 4 planetary radii.

Both missions are presumed to use the 1988 Mars launch opportunity. Major elements of the missions are as follows:

- Launch into low earth orbit by the Space Transportation System (STS)
- Injection from low earth orbit into an Earth-Mars transfer trajectory by a spin-stabilized solid stage
- Approximately 200 days transit time from Earth to Mars
- Mars orbit insertion by means of a solid motor carried by the spacecraft
- Approximately 5-10 days to achieve the desired orbit
- Orbiter operations

Orbiter operations have as a primary objective the acquisition and return of scientific data taken by the payload instruments. While many of the instruments take data continuously, there are still cycles which must be observed in the orbiter operations:

- The cycle of the orbital period, 1.9 hours (Climatology) and 6.9 hours (Aeronomy). Because of the existence of eclipses of the sun and occultations from the earth, and the fact that some instruments operate only over the daylight side of Mars, several aspects of the spacecraft operation have to be synchronized to the orbiter period. (The Aeronomy Mission also calls for data taking at higher rates when the spacecraft is near periapsis.)

- 24 hour period. This cycle reflects the operations of the Deep Space Network approximately 8 hours out of every 24. The major consequence is that data must be recorded on a spacecraft tape recorder during the non-tracking period and played back for down link transmission during the tracking period.

- Precession maneuvers. Periodic precession maneuvers are required to maintain the desired spacecraft orientation - that is, with the spin axis perpendicular to the orbit plane.

- $\Delta$V maneuvers. Occasional velocity maneuvers will be necessary to maintain orbit within prescribed limits. In the Climatology Mission a velocity maneuver to set the final orbit inclination is necessary about 100 to 150 days after arrival at Mars; this establishes the spacecraft in a sun-synchronous orbit of the desired orientation. In the Aeronomy Mission the altitude at periapsis must be maintained within close limits, and in addition velocity maneuvers will have to be performed from time to tome to compensate for atmospheric drag.

## 1.2 STUDY OBJECTIVES

From the Statement of Work these are the objectives of the study.

> The conceptual systems design study will be conducted to a depth that will provide sufficient analysis, tradeoffs of alternative designs and operational concepts and design studies to conceptually define mission parameters, requirements, constraints, and the selected spacecraft system and subsystem design for a Mars Orbiter Climatology and a Mars Orbiter Aeronomy missions. The tasks will encompass the conceptual system design of the Orbiter spacecraft(s), the accommodation of the science instruments for the two missions, mission analysis, flight operations and the project(s) cost estimates. The system design of the spacecraft should, to the greatest practical extent, be the same for each mission.

## 1.3 STUDY EMPHASES

A major emphasis of the study is to minimize overall program cost. This is used in the design of the spacecraft system and subsystems by:

- Making use of existing and previously developed hardware wherever feasible

- Using the protoflight development method for the spacecraft

- Increasing design margins (for example, an upper stage with significantly greater capability than required by the mass of the spacecraft)

- Keeping the operational procedures as simple as possible

Major topics considered in the study, in course of the conception of the total design, are:

- mission analysis

- requirements of the scientific payload

- requirements for velocity correction maneuvers, and the resulting mass history

- the spacecraft configuration approach

- system design and performance characteristics

- subsystem design - options and selection

- cost modeling and cost estimating

In addition to this final report of the study and have been several formal progress presentations by TRW to NASA/Ames Research Center. These were held on June 16 and August 10, 1982, and copies of the handouts were delivered.

## 1.4 MAJOR STUDY RESULTS

The following are the major results of the Mars Orbiter Mission Study:

1. The basic approach of Configuration A of TRW's proposal was selected for detailed study. It is considered to be applicable for both the Climatology and Aeronomy Missions. The Intelsat VI upper stage is employed for injecting the spacecraft from low-earth orbit. Major characteristics of spacecraft Configuration A are:

   - It is a dual spinning spacecraft, with the experiments and communications equipment on a platform which is despun when in Mars orbit

   - The spinning section is dominated by a solar array which is in the shape of a truncated cone

   - In orbit, the spacecraft spin axis if maintained perpendicular to the orbit plane, so that despun instruments can be oriented in relation to the nadir direction, the limb, or the ram direction, as appropriate

2. Joint missions are feasible in the time period designated (1988 launch) and with the instrument requirements as stated

3. There is a very high degree of commonality between the spacecraft for the Climatology and Aeronomy Missions

4. No advance is necessary in the state of the art of spacecraft technology

5. The spacecraft design employs many standard and existing components

6. A number of sensitive design points are observed, wherein further demands on the spacecraft system might call for more expensive solutions. These are:

   - The communications link between the spacecraft and earth up to Mars orbit insertion, while the high gain antenna is not in use

   - The maintenance of moment-of-inertia ratios which provide satisfactory spin stability

   - The overall electrical power requirements, as a significant increase would start to influence the spacecraft configuration

   - Satisfaction of field-of-view requirements: spacecraft and instrument sensors; thermal radiators; spacecraft antennas; and booms.

7. Appropriate areas are identified for further study (See Section 11), but no advance development is required.

## 1.5 REPORT ORGANIZATION

Sections 2, 3, 4 review study groundrules and constraints, mission characteristics, and instrument requirements, respectively.

Sections 5, 6, 7 treat system and subsystems as follows, for the baseline missions.

| | System | Subsystem |
|---|---|---|
| Options and Selection | 5 | 7.X.3 |
| Description | 6 | 7.X.4 |
| Performance | 6 | 7.X.5 |

(The decimal, X, is different for each subsystem)

Section 8 discusses interfaces between the spacecraft and other elements of the project.

Sections 9 and 10 address optional Climatology Missions which differ to some degree from the baseline mission. Section 9 treats the orbit option and Section 10 treats 3 missions employing optional payloads.

Section 11 discusses further study topics and research and technology necessary or appropriate to the maturing of the missions as now defined into funded programs. Section 12 identifies design drivers: aspects of the design which are sensitive to mission requirements and which push the system toward various constraints and limits.

Cost groundrules and estimates for the hardware phase of the Mars Orbiter Missions are not included in this report, but are submitted separately.

# 2. GROUND RULES AND CONSTRAINTS

This section recites the major ground rules and constraints on the missions being studied. They are summarized in Figure 2-1. The top level governing documents are (as cited in the Statement of Work):

PM-2000 Mission Requirements for Mars Orbiter Climatology Mission

PM-2001 Mission Requirements for Mars Orbiter Aeronomy Mission

Both documents are issued by Ames Research Center, and are dated March, 1982. The application recognizes amendments issued May 6 and July 6, 1982.

## 2.1 LAUNCH OPPORTUNITY; MISSION LIFETIME

Either mission is to use the 1988 Mars launch opportunity, which uses launch dates centered in July, 1988. We have determined that Type 1 trajectories are best in 1988, and typical transit times to Mars are about 200 days.

| |
|---|
| Scientific payloads and spacecraft requirements per: |
| PM-2000 Climatology Mission |
| PM-2001 Aeronomy Mission |
| 1988 Mars launch opportunity |
| STS launch, plus upper stage for injection |
| Communications via Deep Space Network |

Figure 2-1

MAJOR GROUND RULES AND CONSTRAINTS

The lifetime in orbit is to be one Martian year (687 days) for the nominal mission, and an additional Martian year for the extended mission, which should not be precluded by the spacecraft design.

Adding about 10 days for adjustment of the initial Mars orbit, after arrival at Mars, gives these mission lifetimes:

| | |
|---|---|
| Nominal Mission | 897 days |
| Extended Mission | 1584 days |

(For other launch opportunities, particularly 1990 and 1992 when Type II trajectories are superior, transit times can exceed 300 days. The total mission lifetime would have to be increased accordingly.)

## 2.2 SPACE TRANSPORTATION SYSTEM

The Space Shuttle is designated as the means of reaching low earth orbit at the start of the mission. The various requirements which must be satisfied in utilizing the Shuttle for this purpose are given in JSC 07700 Volume XIV, "Space Shuttle System Payload Accommodations," and in NHB 1700.7, "Safety Policy and Requirements for Payloads Using the Space Transportation System (STS)."

The former details interface requirements and provisions, and the latter outlines procedures and design practices which must be observed to satisfy Shuttle safety requirements.

An upper stage must also be used to boost the spacecraft from this low earth orbit on to an escape trajectory which will take it to Mars. The Mission Requirements Document suggest two possibilities for this stage, the Spinning Solid Upper Stage (SSUS-A) (baseline) and a stage based on the SRM-1 motor, the large rocket motor of the IUS (option).

While additional possibilities exist, we have gone to the SRM-1 based stage, because the SSUS-A stage has inadequate performance for the spacecraft weight.

This selected upper stage is the one under development for the Intelsat VI program. The solid motor for this stage is modified slightly from the SRM-1: the case is strengthened for use in a spin-stabilized mode; complementarily, the provisions to gimbal the nozzle for thrust vector control are removed.

In this study, we present an interface between the Mars orbiter spacecraft and the Intelsat VI upper stage which we presume will exist -- definitive details are not available now. The spacecraft design is tailored to this presumed interface.

The acquisition of the upper stage so as to satisfy this interface, and its qualification for use on the Space Shuttle are not treated in this study. They are considered to fall outside the realm of the spacecraft contractor's responsibility.

## 2.3 DEEP SPACE NETWORK

The DSN is designated as the ground terminal for all communication with the spacecraft from the time it leaves earth orbit until the end of the mission. During routine phases of the mission a commitment for coverage about 8 hours each day is expected, calling for substantial on-board storage of scientific data.

Which DSN antennas will be assigned for spacecraft-earth communication is not established; however, for transmission of maximum required data at maximum range, the radio link is sized assuming a 64-meter DSN antenna.

## 2.4 SPACECRAFT DESIGN

The Mission Requirements Documents give numerous requirements for spacecraft design. The most significant of these are performance requirements responding to ground rules given above.

One important design requirement also imposed is that "attitude stability shall be achieved through the gyroscopic effect from a spinning mass." Thus, a spin-stabilized spacecraft design is required.

# 3. THE MISSIONS

## 3.1 SEQUENCE OF EVENTS

The overall sequence of events from STS separation through extended (2 year) mission operation is depicted in Figure 3.1-1. Omitted from this figure is the final boost to a safe orbit to meet the Planetary Protection Procedure (PPP) requirements.

The initial events (following STS separation and carrying through upper stage firing to the initial precession maneuver that places the spacecraft into its interplanetary attitude), will last some $\sim$75 minutes, and will be sequencer controlled. The sequencer itself is initiated some 10 minutes after separation from the STS.

A detailed breakdown of these operations, including event times, is given in Figure 3.1-2.

The launch period for the 1988 opportunity was determined by minimizing the sum of 1) the interplanetary injection velocity from a 160 nmi circular earth orbit and 2) the injection velocity into a 300 km circular orbit at Mars. By interpolating the 1988 "pork-chop" plots (from Mission Design Data, JPL Technical Memorandum 33-736, Volume II, September 1, 1975 by Andrey B. Sergeyevsky) the curves of Figure 3.1-3 were obtained. Summing these to obtain the minimum total velocity produced the curves of Figure 3.1-4. As indicated minimum velocity was obtained for the ten day period from 7347 to 7357 (Julian dates). A single arrival date of 7555 seems to best fit the data, which, when located back on the pork-chop plot of $C_3$ and $V_\infty$ appears as in Figure 3.1-5. A summary of the resulting launch and Mars arrival parameters for this launch period is presented in Figure 3.1-6.

## 3.2 THE EARTH-MARS SEGMENT

A summary at the midcourse trajectory correction maneuvers (TCM's) is presented in Figure 3.2-1. Note that we have assumed a minimum of three will be required. The first of these will correct for launch errors, both in direction and total impulse. These will result from errors in timing (probably small) on STS deployment, STS trajectory and sequencer initiation, the variation in total upper stage impulse, and the directional error, induced by precession of the spacecraft plus upper stage spin axis

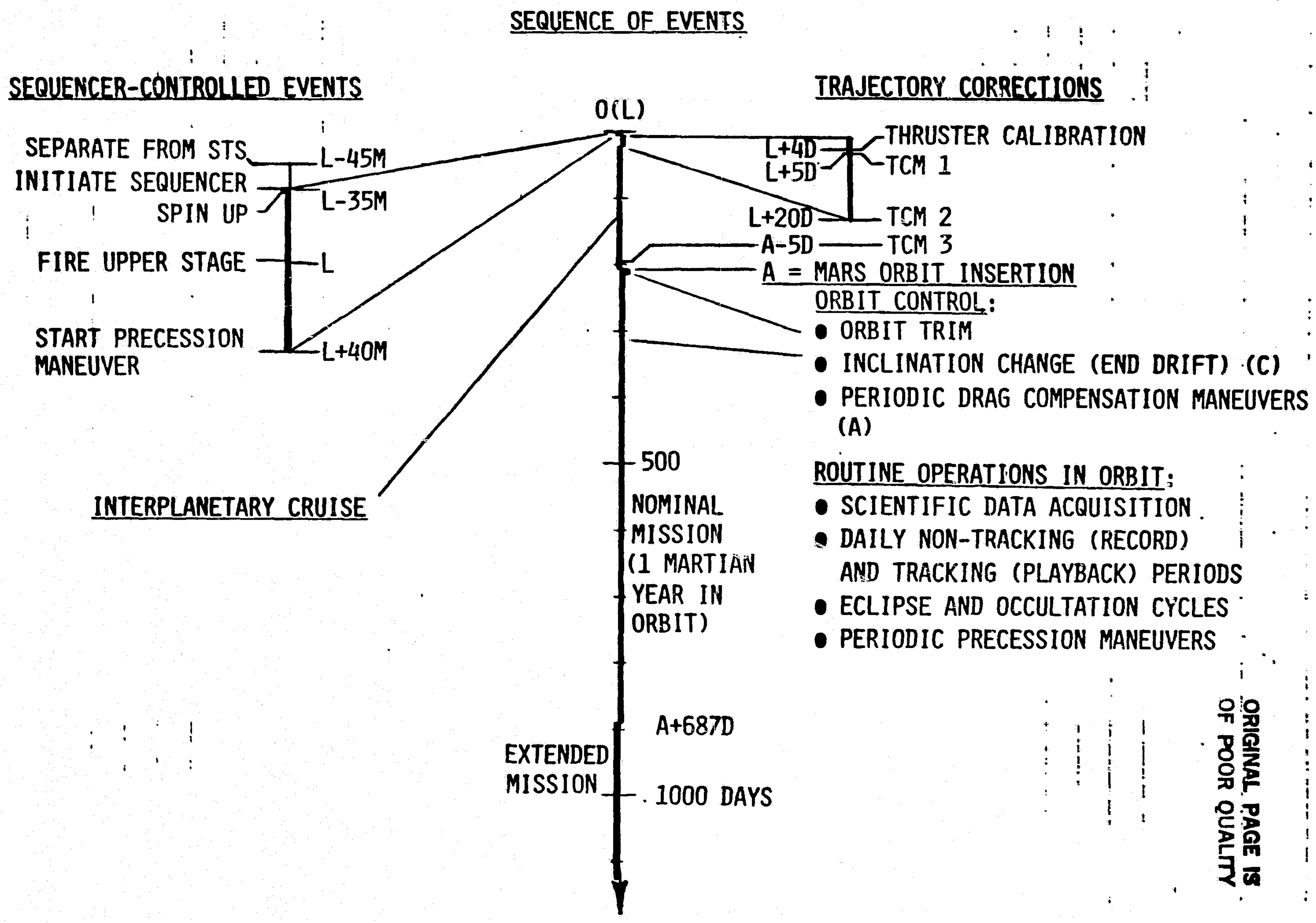


Figure 3.1-1

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## SEQUENCER-CONTROLLED EVENTS (●)

| EVENT | SEQUENCER TIME (MIN.) | TIME FROM L (INJECTION) |
|---|---|---|
| (SEPARATE PL FROM STS) | | L-45 |
| (INITIATE SEQUENCER) | 0 | L-35 |
| ● ENABLE STS-CRITICAL COMMANDS TO BE DIRECTED FROM COMMAND PROCESSOR | 0.5 | L-34.5 |
| ● START SPIN-UP | 1 | L-34 |
| ● ENABLE ANC | 3 | L-32 |
| ● STOP SPIN-UP | 7 | L-28 |
| ● INHIBIT ANC | 33 - | L-2 - |
| ● FIRE UPPER STAGE | 33 | L-2 |
| (UPPER STAGE BURNOUT) | 35 | L (INJECTION) |
| ● ENABLE ANC | 35 + | L + |
| ● INHIBIT ANC | 39 | L+4 |
| ● SEPARATE SC FROM UPPER STAGE | 39 + | L+4+ |
| ● START SPIN-DOWN | 50 | L+15 |
| ● STOP SPIN-DOWN | 53 | L+18 |
| ● START PRECESSION MANEUVER (PER PARAMETERS PRE-STORED IN CEA) | 75 | L+40 |

Figure 3.1-2

EARTH-MARS TRAJECTORY DEPARTURE AND ARRIVAL VELOCITIES

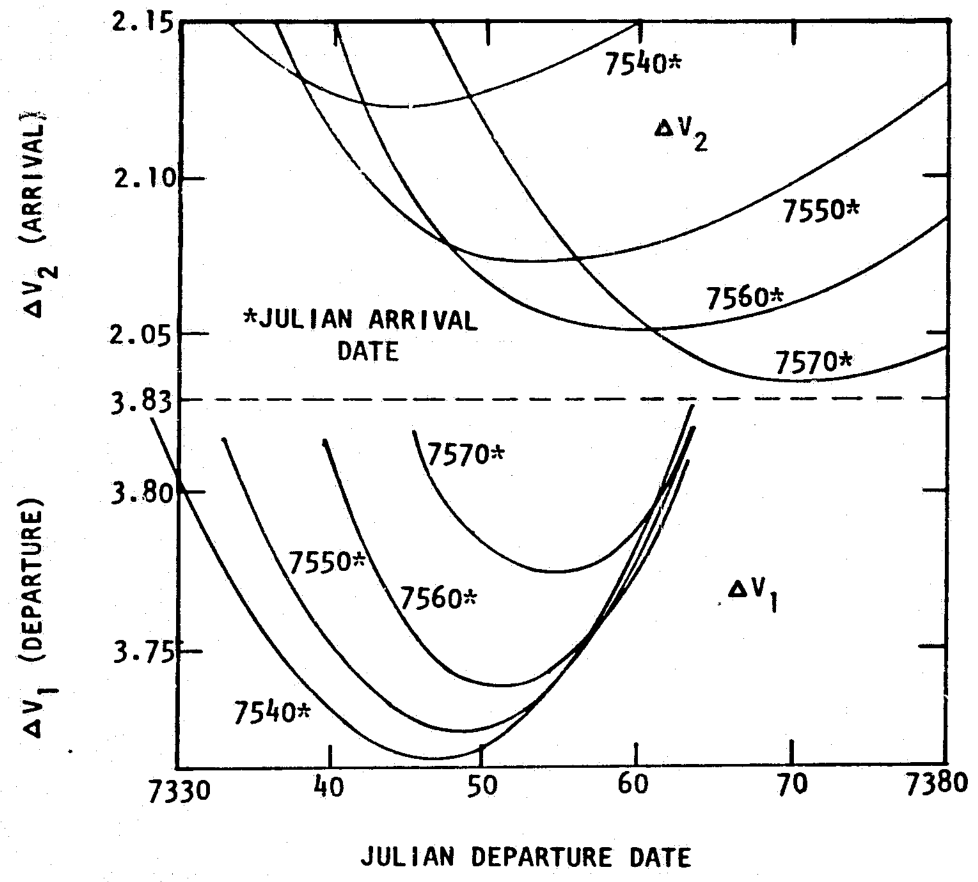


Figure 3.1-3

EARTH-MARS TRAJECTORY MINIMIZATION OF TOTAL VELOCITY

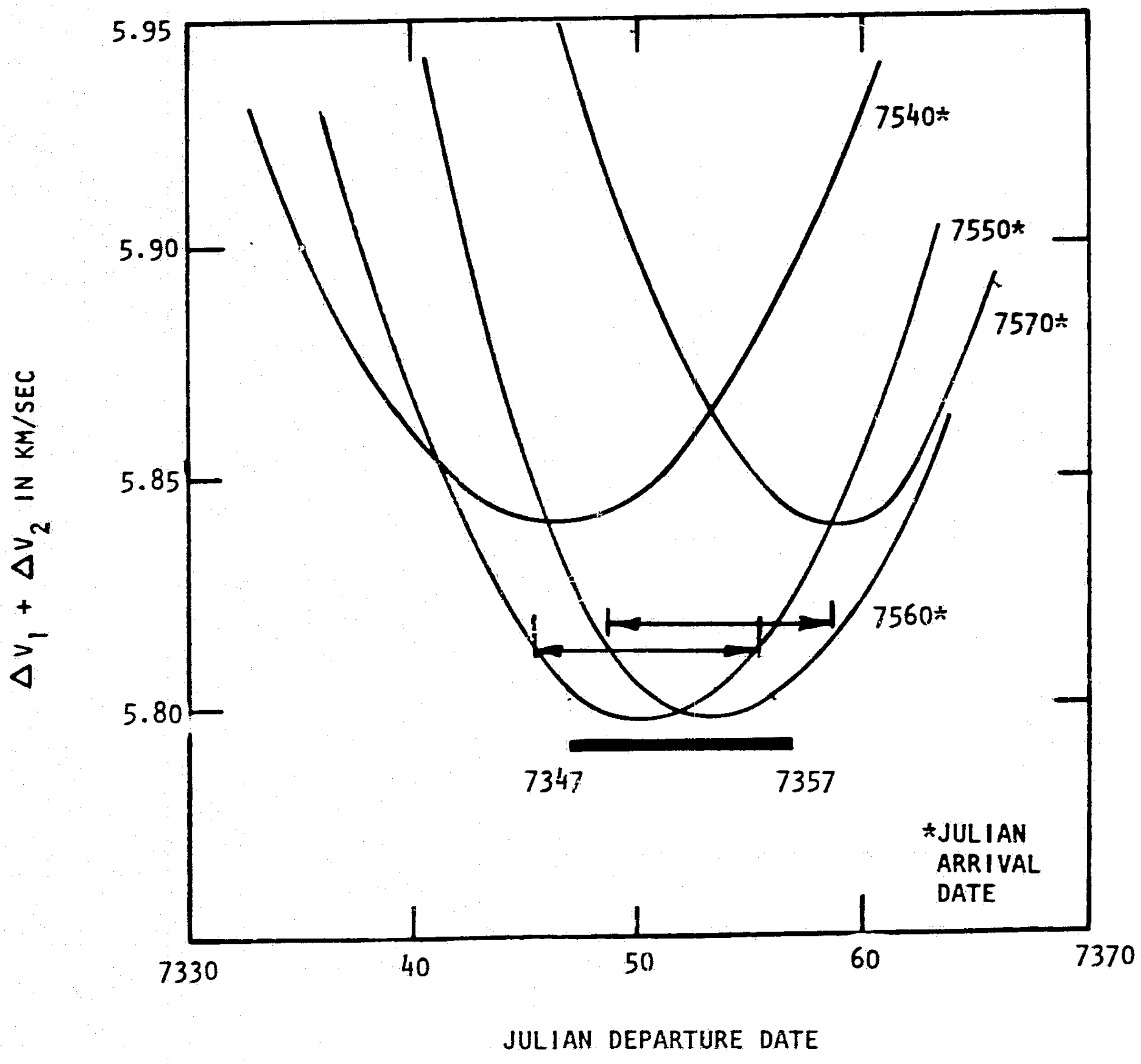


Figure 3.1-4

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SELECTED INTERPLANETARY TRAJECTORIES

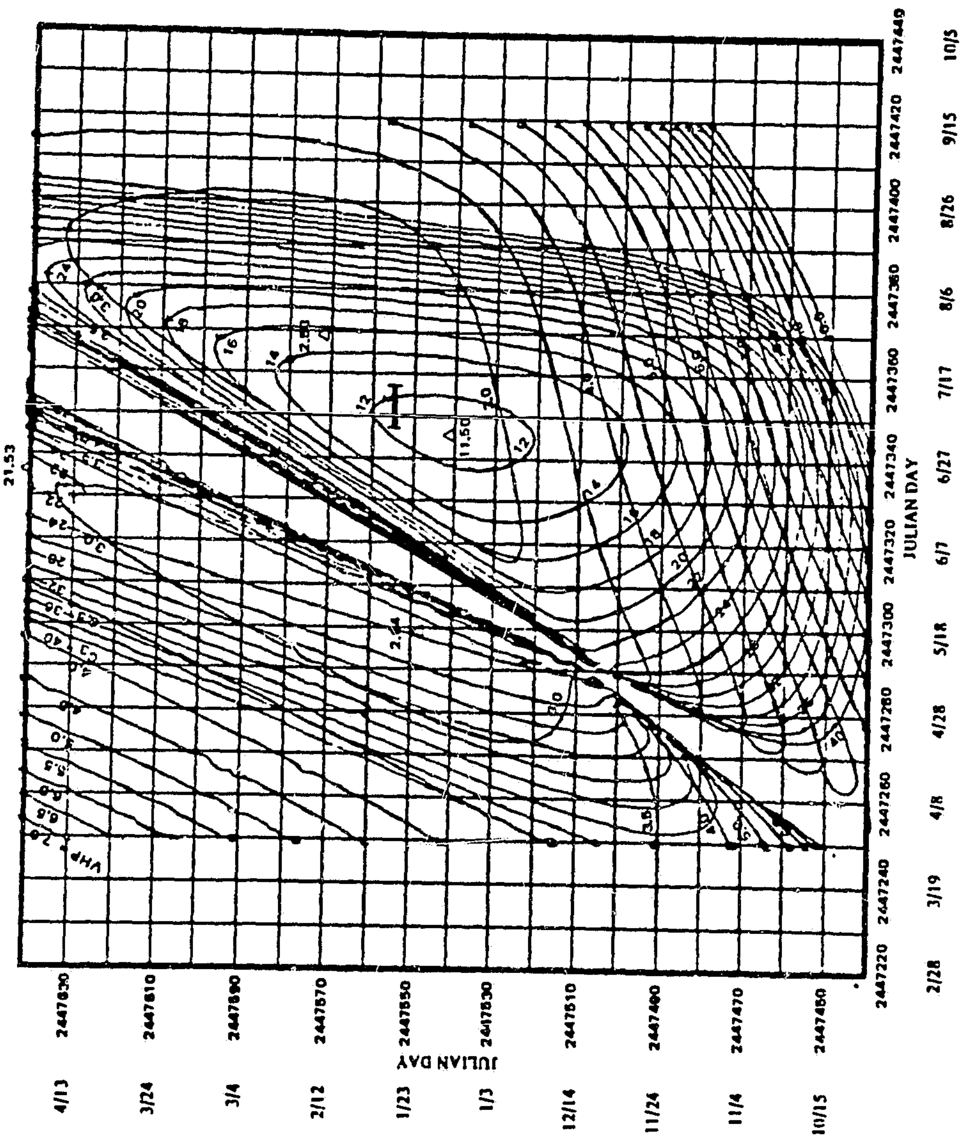


Figure 3.1-5

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## EARTH-MARS TRAJECTORY RESULTS

| | 10-DAY LAUNCH PERIOD | | | |
|---|---|---|---|---|
| | BEGINNING | MIDDLE | END | |
| LAUNCH DATE | 244 7347<br>88/07/04 | 7352<br>88/07/09 | 7357<br>88/07/14 | (JULIAN DATE)<br>(CALENDAR DATE) |
| ARRIVAL DATE | ← | 244 7555<br>89/01/28 | → | (JULIAN DATE)<br>(CALENDAR DATE) |
| TRIP TIME | 208 | 203 | 198 | DAYS |
| INJECTION ENERGY | 11.9 | 11.85 | 12.35 | $KM^2/S^2$ |
| ARRIVAL $V_\infty$ | 2.62 | 2.60 | 2.575 | KM/S |
| $\Delta V_1$ (AT EARTH) | 3.735 | 3.73 | 3.75 | KM/S |
| $\Delta V_2$ (AT MARS) | 2.075 | 2.065 | 2.06 | KM/S |
| TOTAL $\Delta V$ | 5.81 | 5.795 | 5.81 | KM/S |
| ZAP ANGLE | $103^0$ | $104^0$ | $105^0$ | |

Figure 3.1-6

## MIDCOURSE TRAJECTORY CORRECTION MANEUVERS

- MINIMUM OF THREE TCM'S RECOMMENDED
  - FIRST TCM NEAR EARTH TO CORRECT FOR LAUNCH ERRORS (PRIMARILY A VELOCITY CORRECTION) AT ~ L + 5 DAYS
  - SECOND TCM PERFORMED AFTER DETERMINING INTERPLANETARY TRAJECTORY AT ~ L + 20 DAYS
  - THIRD TCM PERFORMED ~ 1 WEEK BEFORE ENCOUNTER (PRIMARILY A POSITIONING CORRECTION)
- TIMING OF SECOND AND THIRD TCM'S DEPENDS ON ACCURACY OF TRAJECTORY. IN WORST CASES ADDITIONAL TCM'S MAY BE NEEDED.
- LATER TCM'S REQUIRE APPROPRIATE "MIS-AIMING" TO MEET PLANETARY PROTECTION PROCEDURE REQUIREMENTS
- "TURN-AND-BURN", USING 0.1# THRUSTERS TO TURN Z-AXIS IN THRUST DIRECTION, 5# THRUSTERS TO BURN. BURN DIRECTION (+Z OR -Z) DETERMINED BY REQUIREMENT TO KEEP EARTH IN +Z HEMISPHERE OF S/C FOR COMMUNICATION PURPOSES
- FOLLOWING BURN, RETURN TO ORIGINAL ORIENTATION WITH S/C SPIN AXIS NORMAL TO ECLIPTIC USING 0.1# THRUSTERS.



Figure 3.2-1

between STS deployment and upper stage firing plus the initial tip-off error.

An analysis of the above noted sources of error indicates that the attitude error of 2° (3 sigma) and upper stage impulse uncertainty of 0.75% (3 sigma) are the prime contributors. With an incremental velocity requirement of 3737 m/s this leads to an error ellipsoid (3 sigma) of 28.03 m/s along the velocity axis and 130.4 m/s transverse to the velocity axis. Reducing these to one sigma values, propagating them to components at $V_\infty$, then combining them into 99% certainty values along three axes yields a final magnitude of 91.1 m/s uncertainty.

A second TCM will be performed some ~ 20 days later based on initial determinations of the interplanetary trajectory. This correction is expected to be small and, as with the first, will be a velocity correction. The necessity for a third TCM will be a function at the accuracy with which the first two are performed and the uncertainty limit in the trajectory determination relative to the known Mars orbital elements.

For all early maneuvers the PPP requirements can be ignored. That is, the uncertainty circle for a Mars impact is so large that the area represented by the capture radius is less than $10^{-4}$ of the uncertainty circle. Thus all early TCM's will aim for a planetary impact. As the mission progresses the situation gradually changes until the last TCM when a specific miss distance is included in the maneuver for MOI positioning.

During the interplanetary phase the spacecraft will be oriented with its spin axis approximately perpendicular to the ecliptic plane. Communications will be effected through an S-band omni for uplink and X-band bicone for downlink. All TCM's will be performed with the Earth in the +Z-hemisphere of the spacecraft. If the range and antenna patterns are such that the required maneuver orientation is in a low-gain configuration the maneuver will be performed as a pair of maneuvers at more favorable antenna orientations, the vector sum of the two maneuvers equalling the total required maneuver. The orientation for the first and by far the largest maneuver will be performed close to the Earth when link budgets are high and no offpointing will be necessary regardless of the required thrust direction. All other TCM's will be small and thus the propellant penalty for a two burn TCM can effectively be neglected. Further discussion of antenna usage and data rate capabilities will be found in Sections 6.8 and 6.2.2.

Each of the first two TCM's will be performed by the following sequence of events:

1. Uplink command to spacecraft for appropriate orientation, including command verification
2. Precession of spacecraft to appropriate thrust orientation using 0.1 pound thrusters
3. Measurement of spin axis orientation using star sensor data
4. After verification of correct orientation, command for thrust
5. TCM execution using axial 5# thrusters
6. Command spacecraft back to original (ecliptic normal) orientation
7. Precession of spacecraft to original orientation using 0.1 pound thrusters

The third TCM will likely be performed in the cruise attitude, firing axial thrusters continuously for the longitudinal component and transverse thrusters in pulses for the lateral components.

## 3.3 MARS ORBIT INSERTION AND ORBIT DRIFT

The events surrounding Mars orbit insertion (MOI) are depicted in Figure 3.3-1 for the 1988 launch opportunity with data included for comparison for the 1990 and 1992 opportunities as well. The relevant angles, velocities, and distances for injection into a 300 km circular orbit are presented in Figure 3.3-2. Particularizing Figure 3.3-1 for the specific Climatology and Aeronomy orbits requires slight changes in $\Delta V_2$ and MOI orientation angle, as discussed below. Following MOI and determination of the orbital parameters achieved in the insertion, orbit trim procedures will be performed to achieve the desired orbital elements. In general this could require a trim on inclination, periapsis, apoapsis, and, in some cases, nodal position. However for the Mars mission the PPP requires something more based on the necessity of keeping the probability of surface impact to a low value (tentatively noted as $10^{-4}$).

## MARS ORBIT INSERTION

| | |
|---|---|
| MOI-48 HOURS | ORIENT S/C TO MOI ATTITUDE ( $\sim 38^{\circ}$ TO EARTH LINE) USING 0.1# THRUSTERS |
| MOI-46 HOURS | CHECK ATTITUDE, PERFORM ADDITIONAL FINE PRECESSION |
| | SPIN UP S/C TO $\sim$40 RPM |
| | CHECK ATTITUDE, PERFORM FINAL PRECESSIONAL TRIM |
| MOI-24 HOURS | |
| MOI-20 HOURS | TRANSMIT TIME OF MOI MOTOR IGNITION TO S/C |
| | CHECK TRAJECTORY UNDER INFLUENCE OF MARS GRAVITATION<br>CONSIDER CHANGING TIME OF IGNITION |
| MOI | S/C TRANSMITS SUCCESSFUL MOI MOTOR FIRING |
| MOI + 6 MIN | S/C ENTERS EARTH OCCULTATION REGION |
| MOI + 8.3 MIN | NEWS OF MOI REACHES EARTH (MILD CHEERING) |
| MOI + 46 MIN | S/C EMERGES FROM EARTH OCCULTATION |
| MOI + 54.3 MIN | NEWS OF PREDICTED EMERGENCE REACHES EARTH (WILD CHEERING) |
| MOI + $\sim$ 2 HOURS | S/C SPUN DOWN TO 9 RPM |
| MOI + $\sim$ 4 HOURS | S/C ORIENTED WITH SPIN AXIS PERPENDICULAR TO ORBIT PLANE WHILE INITIAL ORBITAL PARAMETERS ARE DETERMINED |



Figure 3.3-1

## 1998 THROUGH 1992 LAUNCH OPPORTUNITIES

- THE 1988 MISSION FAVORS A TYPE I TRAJECTORY WHILE BOTH 1990 AND 1992 MISSIONS FAVOR TYPE II TRAJECTORIES. THE FOLLOWING ARE MIDDLE-OF-WINDOW CHARACTERISTICS:

| | 1988 LAUNCH | 1990 LAUNCH | 1992 LAUNCH | UNIT |
|---|---|---|---|---|
| LAUNCH DATE | 88/07/09 | 90/08/27 | 92/09/26 | CALENDAR DATE |
| ARRIVAL DATE | 89/01/28 | 91/08/18 | 93/08/31 | CALENDAR DATE |
| TRIP TIME | 203 | 356 | 339 | EARTH DAYS |
| INJECTION ENERGY | 11.85 | 15.53 | 11.92 | $KM^2/S^2$ |
| ARRIVAL $V_\infty$ | 2.60 | 2.79 | 2.48 | KM/S |
| $\Delta V_1$ (AT EARTH) | 3.73 | 3.89 | 3.73 | KM/S |
| $\Delta V_2$ (AT MARS) | 2.065 | 2.16 | 2.01 | KM/S |
| TOTAL $\Delta V$ | 5.795 | 6.05 | 5.74 | KM/S |
| ZAP ANGLE | $104.2^0$ | $54.1^0$ | $72.6^0$ | DEG |
| LVI ANGLE | $0.1^0$ | $35.2^0$ | $14.3^0$ | DEG |
| MOI SOLAR DISTANCE | 1.533 | 1.643 | 1.591 | A.U. |
| MOI EARTH DISTANCE | 1.230 | 2.480 | 2.317 | A.U. |
| MOI ORIENTATION ANGLE* | $81^0$(N), $38^0$(S) | $52^0$(N), $58^0$(S) | $77^0$(N), $109^0$(S) | DEG |

* WITH RESPECT TO EARTH (NORTHERN AND SOUTHERN APPROACHES)

Figure 3.3-2

Both Mission Requirements Documents state an altitude uncertainty of 100 kilometers at the time of MOI. Based on PPP requirements and orbital drag calculations an aim point of 350 km has been chosen for the Climatology Mission, 250 km for the Aeronomy. Thus, in worst case, achieving the desired orbit altitude would require going from 450 circular to 300 km circular in the Climatology case and 350 km periapsis to 150 periapsis in the Aeronomy case. (These translate to 67.0 m/s and 25.0 m/s respectively, and the propellant requirements are so sized as noted in Section 6.3.3.) Thus, following initial orbital element determination altitude adjustments will be made for both missions, again using the 5-pound thrusters and before boom extension or platform despinning.

## 3.4 CLIMATOLOGY ORBIT

### 3.4.1 Baseline

The Climatology Orbit requirements, as stated in PM-2000 are as follows:

1. Altitude: 300 kilometers, maintained within ±50 kilometers
2. Inclination: Near 90° for maximum planetary coverage
3. Orientation to Mars-Sun Line: Orbit plane to remain within 22.5° to 45° (1:30 to 3:00 orbit)
4. Nodal Precession Rate: Use Mars' oblateness to maintain the orientation requirement stated in 3
5. Design Lifetime: Nominal 687 Earth days (one Mars year), but not precluding 1374 Earth days

For a 300 km orbital altitude, requirement 4 leads to an orbit inclination of 92°.65 a figure based on planetary data supplied as Attachment A to the Statement of Work as follows:

| | |
|---|---|
| Planet Radius | 3397.51 km |
| Gravitational Parameter | 42828.2866 $km^3/s^2$ |
| $J_2$ Harmonic | 0.001964 |

However, requirement 3 precludes, in general, the possibility of injecting directly into this inclination. Rather an initial inclination different from sun synchronous must be chosen. After a suitable drift period (for the orbit plane to achieve the required orientation relative to the sun) the inclination is changed to 92°.65 using 5# thrusters at equator crossing. The velocity increment required for each degree of plane change is 59.3 m/s, and the drift rate for each degree away from sun-synchronous is 0.197 degrees/day.

With the arrival geometry shown below the minimum drift period is to the anti-solar direction, i.e., orbit insertion with inclination $<$ 92°.65, since even in the worst case sun location the angular drift is 83° in this direction compared to the best case value of 90° drift to the sub-solar direction.

A determination of the Martian equation of time, including effects of both orbit eccentricity and planet inclination is shown in Figure 3.4-1. The broken curve in this figure shows the equation of time if only orbit eccentricity is included. Note that in the more exact calculation the longitudinal range of the sun is 23°, somewhat larger than the 22.5° stated in the requirements document. For the determination of precise orbit injection we have taken the sun position to drift between 22°.0 and 45°.0. Using as a guideline the difference in arrival date for a Type I and Type II trajectory, i.e., 140 days, an initial orbit inclination of 90° was chosen as a compromise between drift period and inclination change requirement. With this inclination the actual drift period using graphical construction techniques on Figure 3.4-2 was found to be 146 days following a 28 January 1989 arrival. The velocity increment required to make the plane change is 157 m/s.

With the above considerations in hand, the sequence described in the top of Figure 3.4-3 can now be delineated.

Not mentioned on this figure, but included in Appendix B, are the sequence of steps taken to put the spacecraft in its operational mode. Following injection the spacecraft spin axis is processed to place it normal to the orbit plane, the platform is despun, the HGA pointed toward the earth and the GRS boom deployed. With a retrograde drift of the spacecraft relative to the sun direction the thermal radiator of the GRS will remain shaded from the sun allowing the GRS to operate during the entire drift period. Other Climatology instruments can operate during this

MARS EQUATION OF TIME

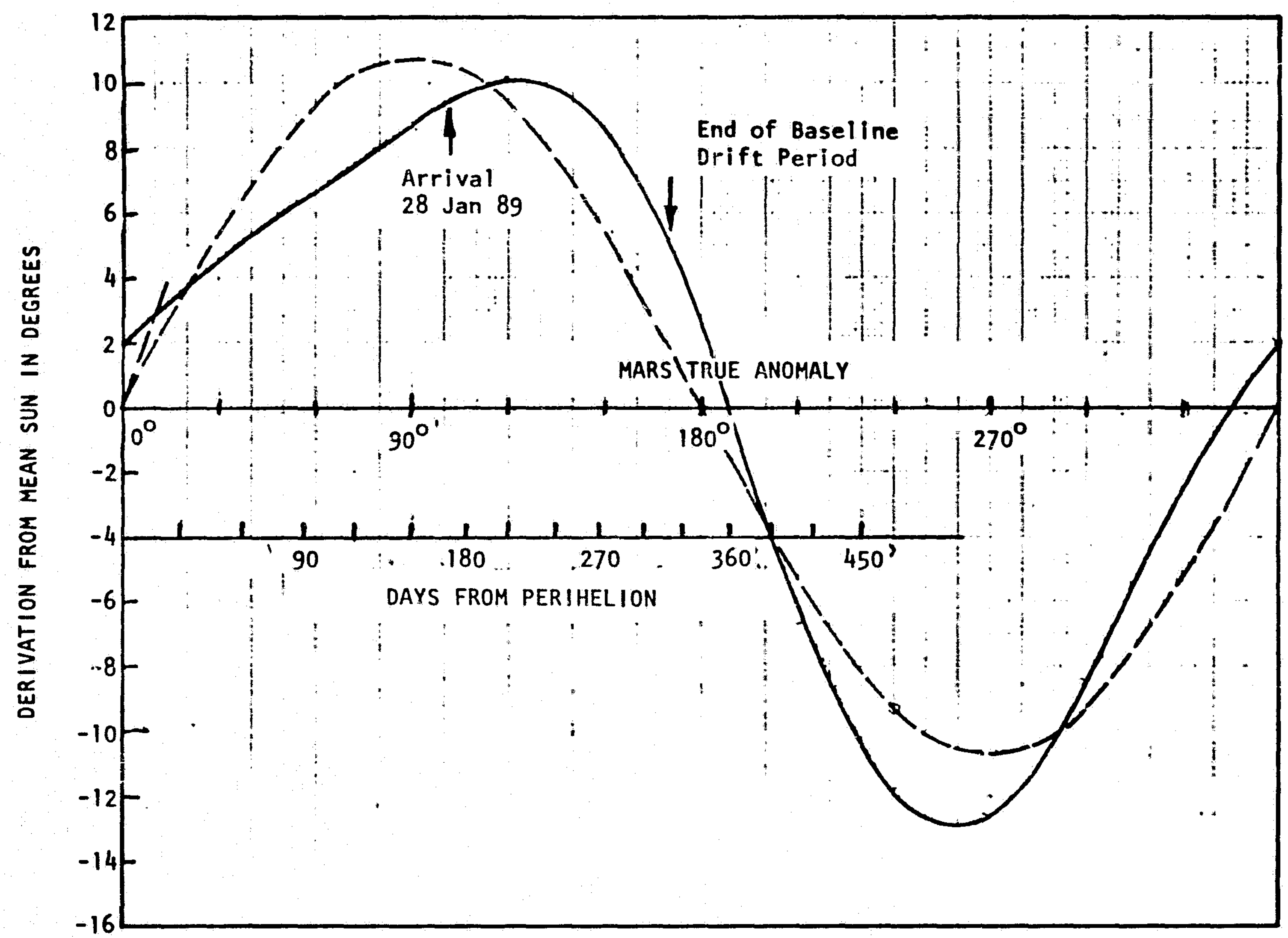


Figure 3.4-1

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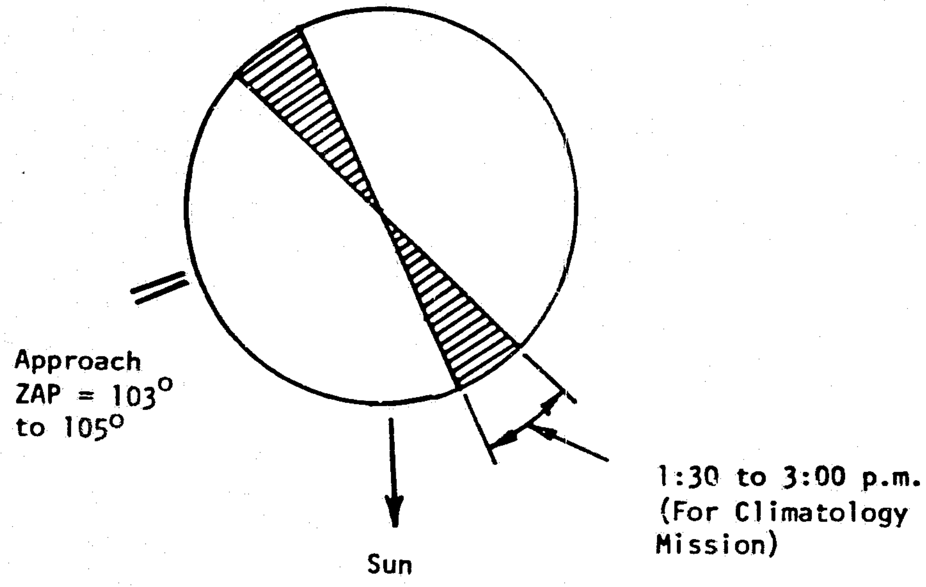
## MARS ARRIVAL GEOMETRY




## BASELINE MANEUVERS

### Climatology

1. ENTER AT $i = 90^{o}$, 350 km
2. WAIT 146 DAYS FOR SUN LINE TO DRIFT TO REQUIRED TIME
3. CHANGE $i$ TO $92.6^{o}$, 300 km

### Aeronomy

1. ENTER AT 250 km BY 3 MARS RADII ALTITUDE
2. LOWER PERIAPSIS TO 150 km

Figure 3.4-2

## ORBIT TRIM AND ADJUSTMENT

### CLIMATOLOGY MISSION

- INITIAL AIM POINT OF 350 KM WITH $\pm 100$ KM UNCERTAINTY REQUIRES TWO BURNS TO CIRCULARIZE AT 300 KM ALTITUDE
- USING TRACKING DATA AND ONBOARD SENSORS INITIAL ORBIT WILL BE DETERMINED WITHIN $\sim 2$ DAYS
- "TURN-AND-BURN" SEQUENCE WILL BE PERFORMED FOR APOAPSIS AND PERIAPSIS BURNS. (WORST CASE BURN DURATION $\sim 10$ MINUTES USING PAIR OF 10 POUND THRUSTERS.)
- FOLLOWING ATTAINMENT OF 300 KM CIRCULAR ORBIT AT $90^{\circ}$ INCLINATION, ORBIT DRIFT PERIOD OF UP TO 146 DAYS IS REQUIRED TO ATTAIN CORRECT SUN-ORBIT PLANE ANGLE. INSTRUMENTS CAN BE TURNED ON DURING THIS PERIOD.
- AT END OF DRIFT PERIOD, BURN IN $\pm Z$ DIRECTION ACHIEVES SUN SYNCHRONOUS ORBIT AT $92.65^{\circ}$ INCLINATION. $\Delta V$ OF 157 M/S REQUIRES 45 MINUTES OF BURN TIME. TO KEEP BURN WITHIN $\pm 15^{\circ}$ OF EQUATOR (FOR PROPELLANT EFFICIENCY) REQUIRES SEPARATE BURNS OVER $\sim 5$ ORBITS.

### AERONOMY MISSION

- SIMILAR TO CLIMATOLOGY BUT ONLY ONE INITIAL BURN, TO ACHIEVE 150 KM PERIAPSIS, IS NEEDED.
- NO DRIFT PERIOD REQUIRED - OPERATIONAL ORBIT ACHIEVABLE WITHIN $\sim 2$ DAYS

Figure 3.4-3

period as well, obtaining data at different local times. Such operations will permit scientists to obtain some data on the effects of diurnal variations and sun aspect on Martian surface properties.

During the course of a two year mission several effects can be expected to perturb the orbit. One of these, atmospheric drag, can be expected to be large for the Aeronomy orbit but also might be of concern to the 300 km Climatology orbit. While a detailed consideration of drag effects is more appropriately to be found in Section 3.5, we note here that, using the same methods described in 3.5, a velocity increment of 9 m/s is required to maintain the 300 km orbit over a two year operational period.

At the end of the operational lifetime of the satellite a final boost into a safe orbit is required to meet the Planetary Protection Policy. According to PM-2000 sufficient fuel for 100 m/s incremental velocity should be included in the propellant budget. For a 300 km circular orbit this is enough propellant to achieve a final circular orbit of approximately 525 km altitude.

### 3.4.2 Orbit Option

Appendix C of the Statement of Work denotes the requirements for an alternate Climatology orbit that uses Mars oblateness to rotate the orbit plane at the rate of 720° per Martian year. The altitude of the orbit is to remain at 300 km. The above nodal precession rate, with the physical constants quoted in 3.4.1 above, dictates an orbit inclination of 84°.69. Since there is no longer any requirement for drifting to an appropriate local time this inclination can be assumed at orbit insertion, thereby reducing the maneuvering velocity requirement by 157 m/s from the baseline mission. In a very real sense, operations in the orbit option are the same as in the drift mode of the baseline mission. Particular geometrical characteristics of the two orbits will be discussed in Section 3.6 below.

## 3.5 AERONOMY ORBIT

### 3.5.1 Baseline

The Aeronomy Orbit requirements as stated in PM-2001 are summarized as follows:

1. Periapsis Altitude: 150 km
2. Apoapsis Altitude: Greater than 3 Mars' radii but less than 5
3. Inclination: Greater than 75° but less than 95° to the Mars equator
4. Nodal and Apsidal Motion: Mars oblateness shall be used to control an orbit with the following one year objectives:
   a) Periapsis motion of at least 360°
   b) Place periapsis and apoapsis at the sub- and anti-solar points at least once
5. Design Lifetime: Nominal 687 Earth days, but not precluding 1374 Earth days

Figure 3.5-1 delineates the region of choice for orbit parameters that meet these requirements. Horizontal lines at 75° and 95° limit the inclination and vertical lines at 4 and 6 Mars radii limit the altitude. For the required 150 km periapsis altitude, contours of apsidal advance rate further limit the region of choice to be left of the -1.0 revolution per Mars' year curve. Also shown on the figure is the suggested "point source design" at 77.5° inclination, 3 Mars radii altitude of apoapsis.

Orbital calculations have been performed for a number of different inclinations but for the same apoapsis altitude in an attempt to meet requirement 4b as closely as possible. In general orbits with inclinations above 90° are always less successful than their supplements below 90°. Moreover, inclinations close to 90° seem not to be successful. Thus the search was narrowed to the 77° to 83° range, although choosing a higher apoapsis could possibly modify this result. In addition, carrying the calculations forward for a second operational year could lead to a different decision.

In any case, based on a one year mission and the 28 January 1989 arrival data the time history curves of Figure 3.5-2 were obtained. Shown is the angle between the sun direction and the periapsis vector as a function of time for orbit inclinations of 77.5° and 80.0°. Note that periapsis occurs ~10° closer to the anti-solar point and ~8° closer to the sub-solar point for the 30° orbit. Also of interest, a 79° orbit comes

MARS AERONOMY ORBIT

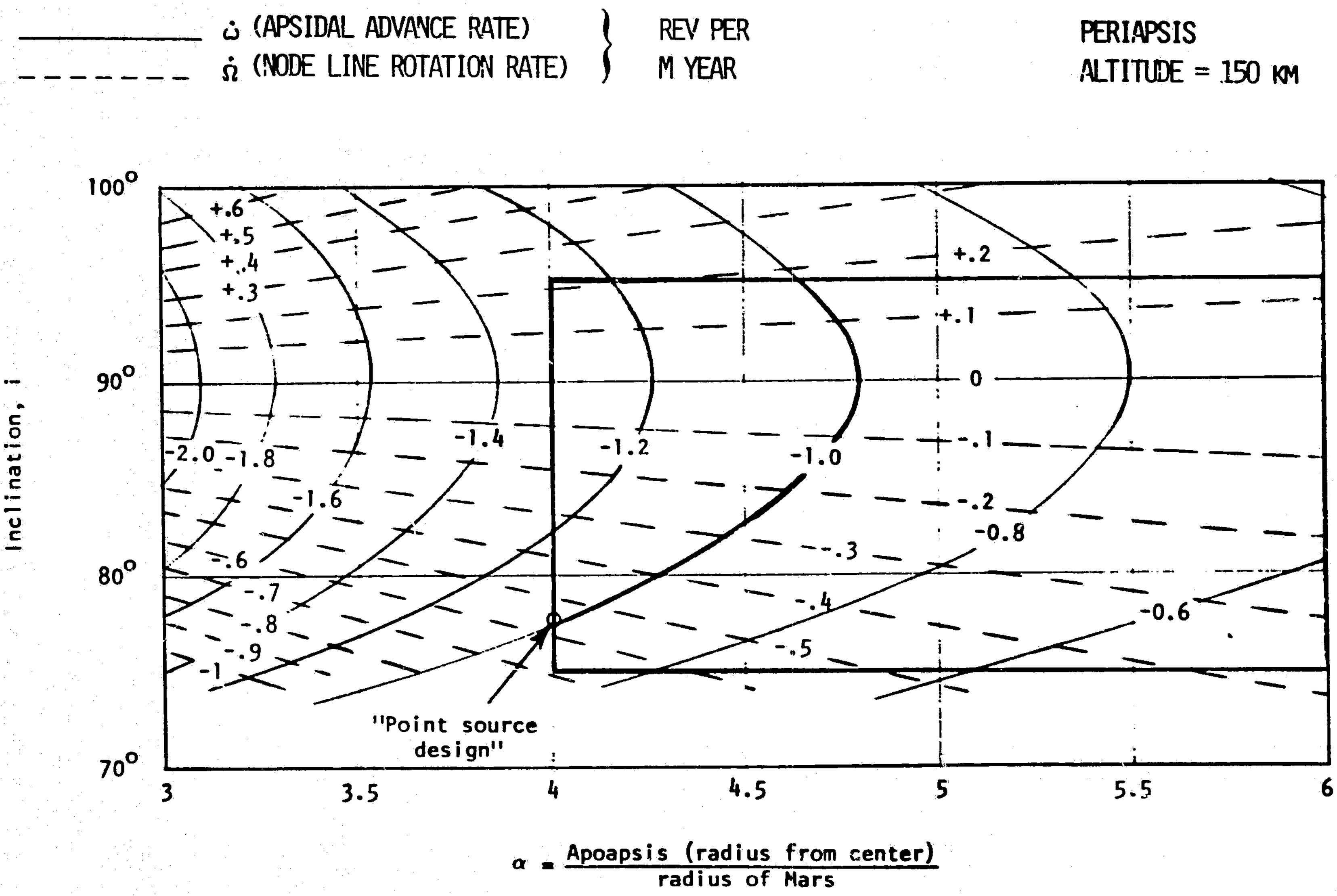



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Figure ? 5-i

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AERONOMY MISSION
TIME HISTORY OF PERIAPSIS LOCATION

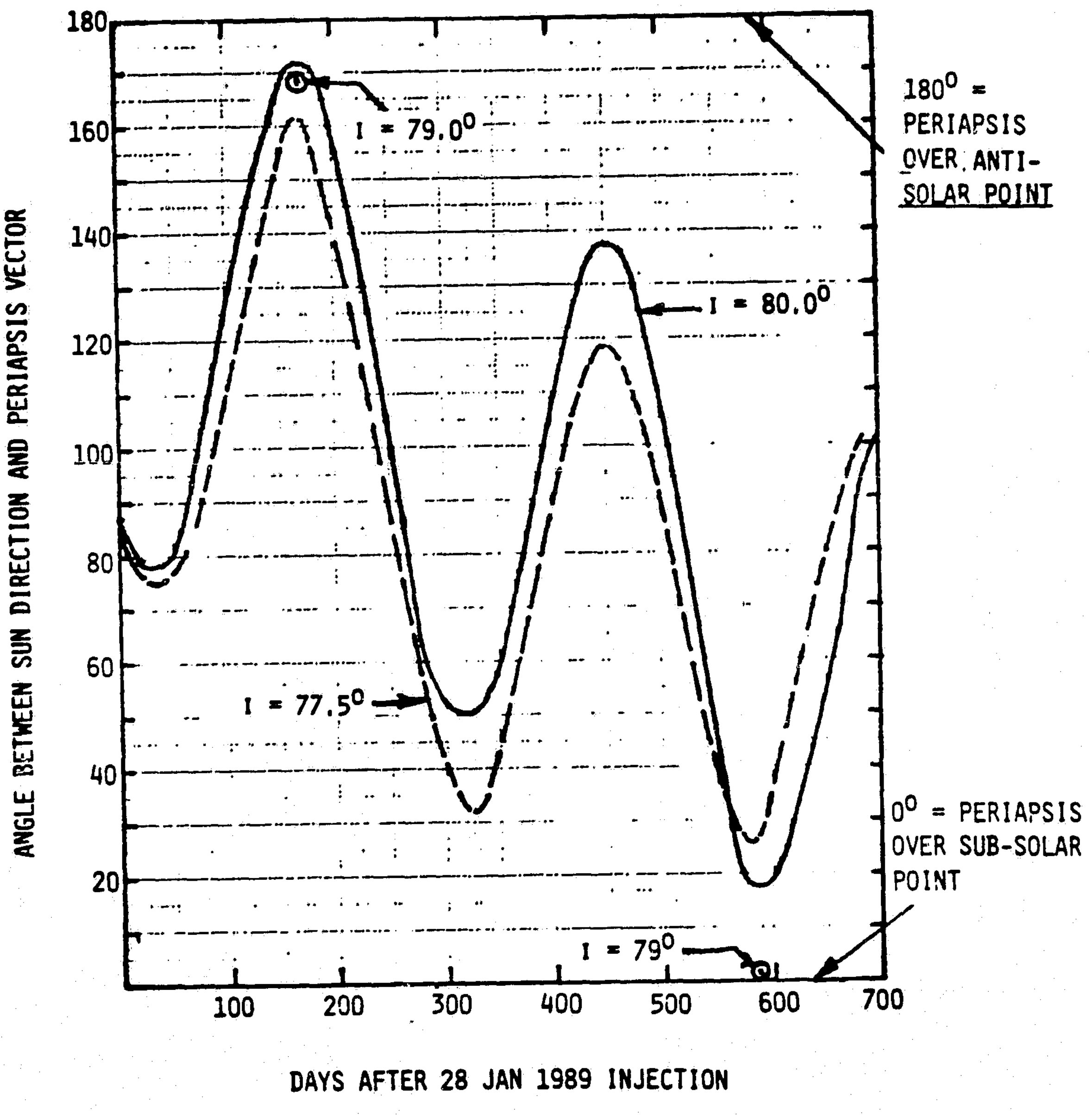


Figure 3.5-2

within 2° of the sub-solar point (some 15° closer than the 80° orbit and ~25° closer than the 77.5° orbit, while being only slightly further than the 80° orbit at the anti-solar point. Other trials would probably produce orbits that approach the objectives of 4b even more closely. For our purposes here we will take the 77.5° and 80° cases as closely representative of what might be the optimum orbit.

An additional complication to the orbit calculations is the effect of drag which will gradually change height of apoapsis and, to a lesser extent, periapsis. (Third body effects and solar pressure, as well as higher order Mars gravitational terms will also take their toll on any attempted precise choice of orbit parameters.) However, at 150 km, atmospheric drag, especially over a two year mission lifetime, can be considerable, leading to a large $\Delta V$ requirement to maintain orbit parameters.

In performing the drag calculations the requirements and assumptions of Figure 3.5-3 were used. A drag coefficient of 3 was taken as a mean between 4 for a perfectly elastic collision and 2 for complete thermalization of the impinging atmosphere. Assumption 5 indicates that the calculation was performed assuming a constant density at the periapsis altitude for one scale height, zero density above this. With these assumptions the results of Figure 3.5-4 were obtained. Note that increasing the periapsis altitude to 170 km decreases the $\Delta V$ requirement by a factor of five. However, such an increase would place the orbit above the region of interest for several of the experiments. This last result does, however, point to the possibility of periodically raising and lowering periapsis (by an apoapsis maneuver) to save propellant over the course of the mission. Another possibility is to allow apoapsis to decrease somewhat during the mission. For example, injecting into an initial orbit of 5 Mars' radii apoapsis altitude and allowing this to decrease to 3 Mars' radii would be equivalent to a 160 m/s savings in propellant requirement. A third option would be to allow apoapsis to decrease below 3 Mars' radii at the end of the mission, then boost to a safe orbit at 625 km periapsis by whatever apoapsis was finally attained.

For the Aeronomy Mission, the final boost to a safe orbit will consist of lifting periapsis to 525 km, the same altitude as was achieved with

## ORBITAL DRAG EFFECTS

REQUIREMENTS

1. MAINTAIN AERONOMY MISSION ORBIT ALTITUDE AT 150 $\pm$ 10 KM
2. PROPELLANT SIZED FOR 2M YEAR LIFETIME

ASSUMPTIONS AND METHODOLOGY

1. DRAG EQUATION: $\Delta V = \frac{C_D A}{2M} \rho V^2 \Delta \tau$
2. KNUDSEN NUMBER $>$ 100, I.E., MOLECULAR FLOW
3. DRAG COEFFICIENT EQUALS 3
4. ORBITAL PERIAPSES REMAIN CONSTANT AT 150 KM OR 300 KM
5. FOR AERONOMY MISSION, DRAG EFFECTS OCCUR WITHIN ONE SCALE HEIGHT OF PERIAPSIS
6. ATMOSPHERIC DENSITY OF "ATTACHMENT B"

Figure 3.5-3

## ORBITAL DRAG EFFECTS
## RESULTS

CLIMATOLOGY: $\Delta V$ = 9.4 METERS PER SECOND

AERONOMY: TIME INTERVAL ($\Delta\tau$) WITHIN ONE SCALE HEIGHT OF PERIAPSIS:

215 SEC PER ORBITAL PERIOD OF 6.78 HOURS

(1.06 X $10^6$ SEC IN TWO M YEARS)

$\Delta V$ = 430 METERS PER SECOND

$$\left(\Delta\tau = \sqrt{\frac{8a^2h}{\mu e}}\right)$$

TRADE STUDY: INCREASE OF AERONOMY PERIAPSIS TO 170 KM DECREASES $\Delta V$ REQUIREMENT BY ~ FACTOR OF FIVE

Figure 3.5-4

100 m/s in the Climatology Mission. Such a boost requires a 34 m/s velocity increment for the Aeronomy mission.

## 3.6 GEOMETRIC CHARACTERISTICS

The geometric characteristics of the Climatology and Aeronomy orbits consist of the sun-spacecraft angle, earth-spacecraft angle, and eclipse periods. These characteristics are relevant as they refer to the solar cell and battery charge-discharge cycles and to the required HGA pointing angle.

In contrast to the rather simple behavior of the baseline mission, the optional mission eclipse profile, as shown by the broken curve in Figure 3.6-1, shows six peaks per Martian year. Since the orbit is designed for a 720° retrograde nodal motion in a year and the sun moves 360° forward, the sun, as viewed from the spacecraft, will progress three full revolutions during the year. Thus the solar aspect will pass through 90° relative to the spacecraft spin axis necessitating a flip of the spacecraft axis six times to keep the solar panel illumination angle less than 90°. Note that every ~115 days the satellite remains out of eclipse for ~ 25 days. Thus the Climatology optional mission will have over 20% fewer eclipses than will the baseline mission.

A similar set of eclipse curves for the Aeronomy mission is depicted in Figure 3.6-2. Eclipse durations for both the 77.5° and 80° inclination orbits are shown as a function of time following orbit injection. Note that the curves are plots of calculations performed at five day intervals so narrow peaks might be missed in some rapidly varying sequences.

Data on sun aspect and earth pointing angles has also been plotted for the 77.5° and 80° cases. These are shown in Figures 3.6-3 and 3.6-4. The solid curves show the solar aspect with a spin axis flip at 90° to keep the sun angle favorable with respect to the solar array. The broken line shows the earth angle relative to the spin axis and represents the pointing angle of the high gain antenna. Note that at 562 days in the 77.5° orbit the HGA must be capable of pointing at an angle of 134° from the +Z axis. For the 80° orbit this angle reaches a maximum of 126.5° at 602 days representing a 7½° less stringent pointing condition. Let us, however, study this issue further.

# ECLIPSE DURATIONS FOR MARS ORBITER

## CLIMATOLOGY MISSION

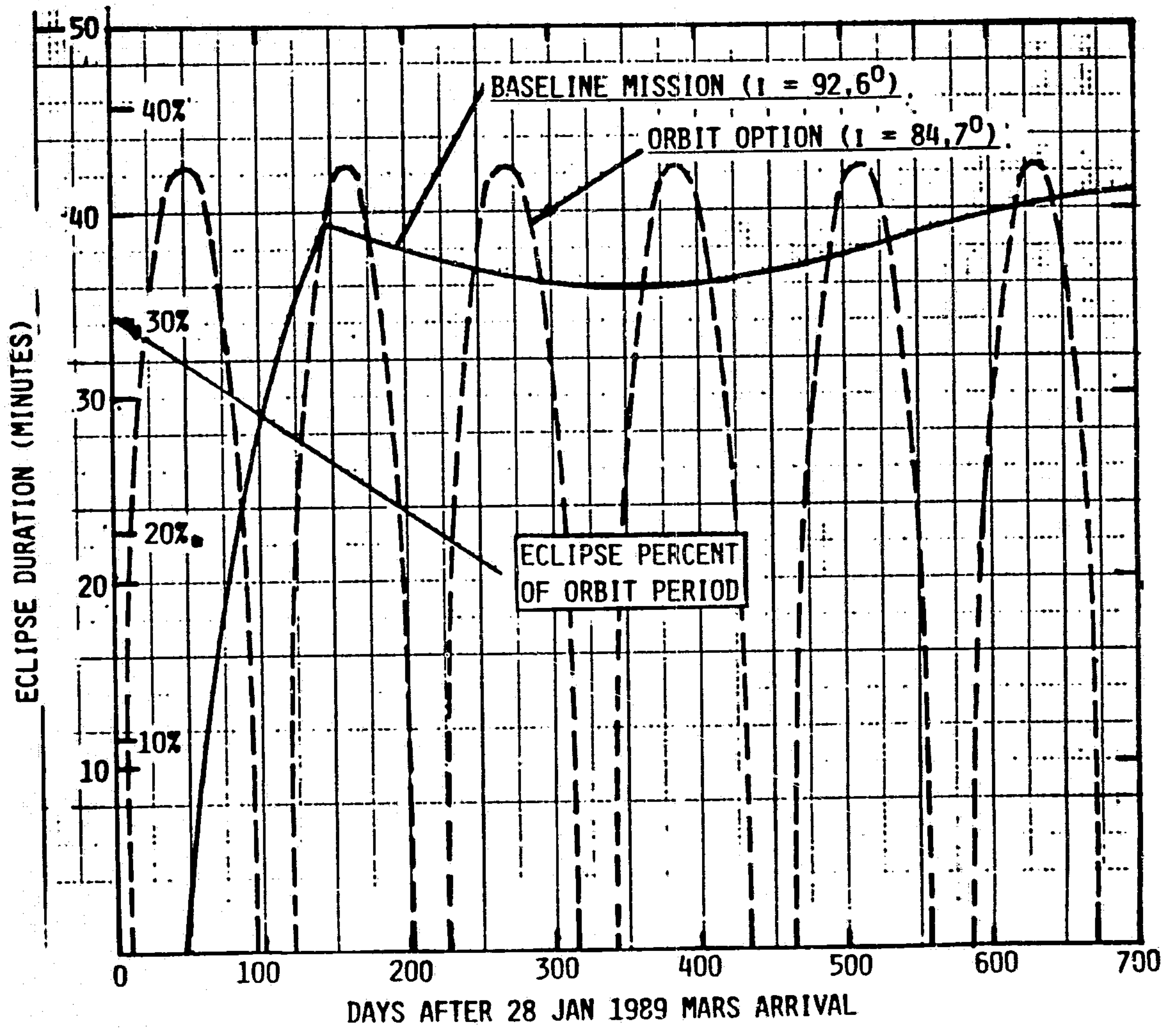


Figure 3.6-1

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## ECLIPSE DURATIONS FOR MARS ORBITER

## AERONOMY MISSION

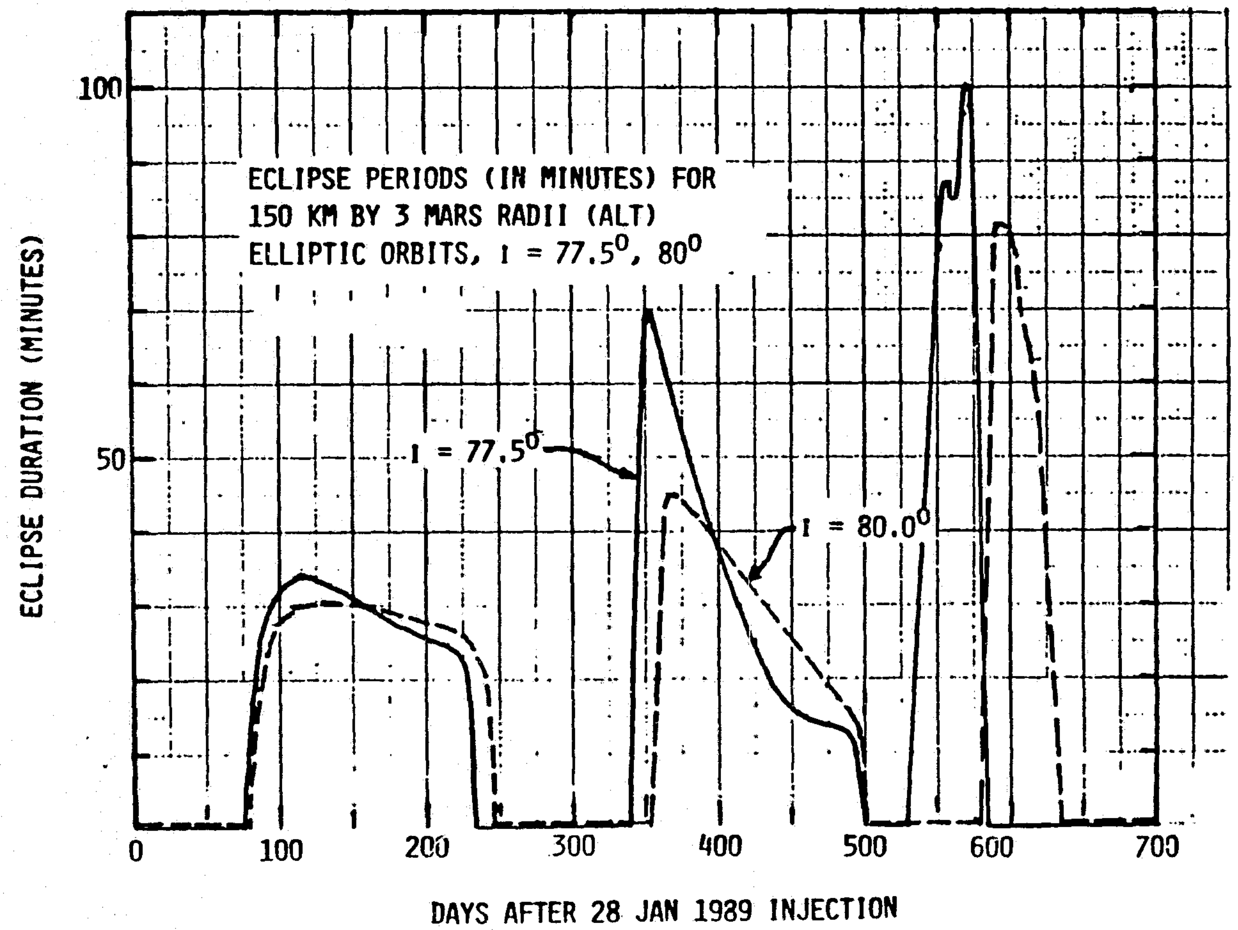


Figure 3.6-2

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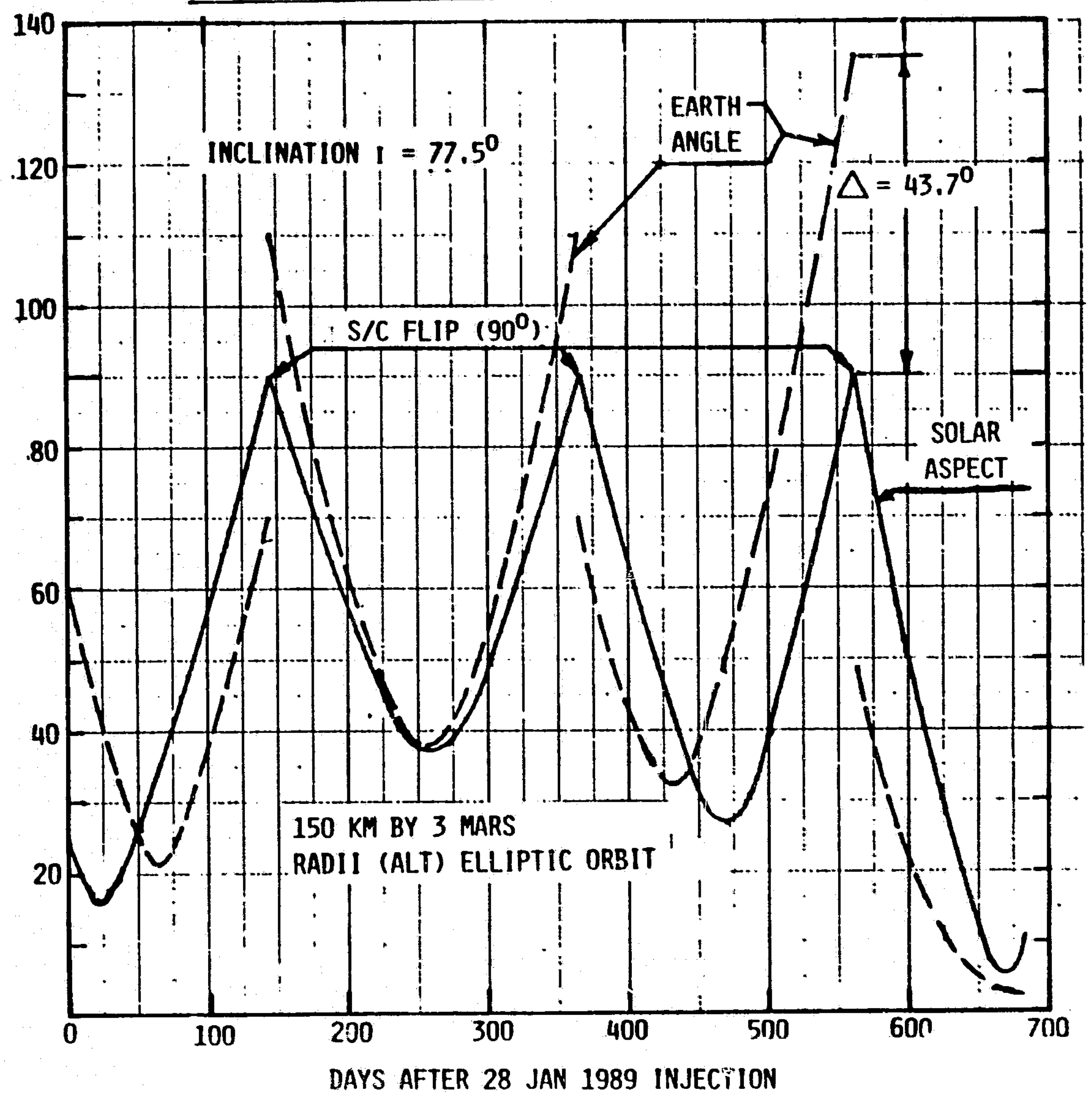
## AERONOMY MISSION

### SUN ASPECT AND EARTH POINTING ANGLES 77.5° ORBIT




| DAY | Δ |
|---|---|
| 145 | 17.1° |
| 365 | 18.9° |
| 562 | 43.7° |

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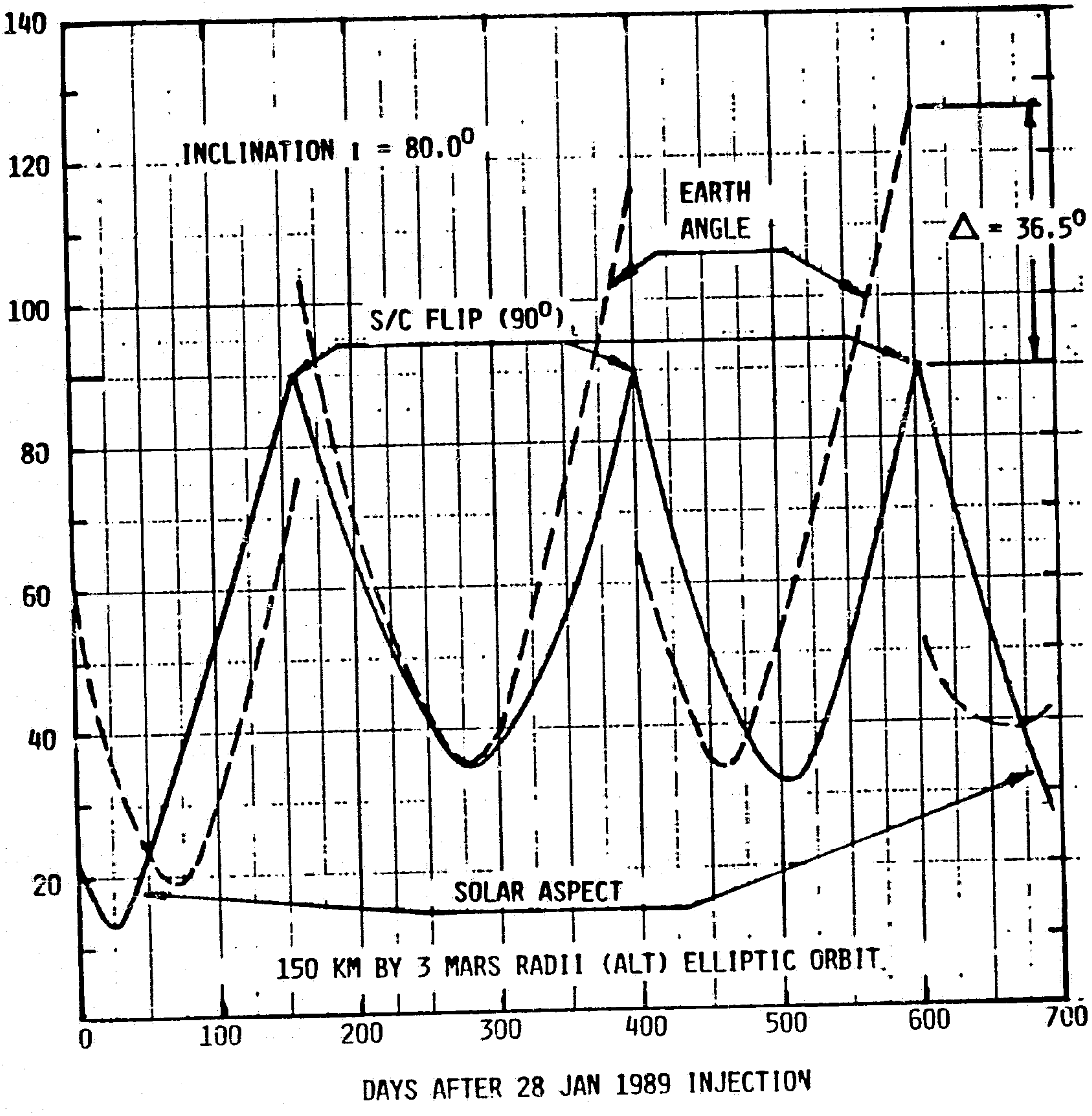
BOTH ANGLES MEASURED RELATIVE TO ORBIT NORMAL, I.E., TO S/C SPIN AXIS

Figu 3.6-3

AEROWOMY MISSION

SUN ASPECT AND EARTH POINTING ANGLES 80.0° ORBIT



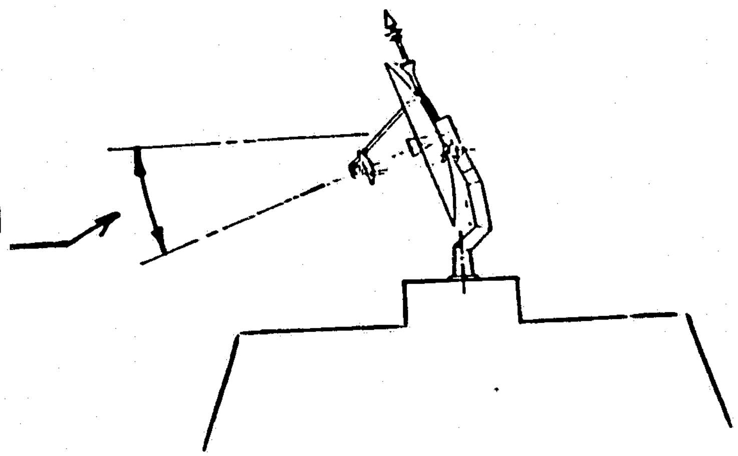

| DAY | Δ |
|---|---|
| 155 | 16.1° |
| 397 | 23.6° |
| 602 | 36.5° |

BOTH ANGLES MEASURED RELATIVE TO ORBIT NORMAL, I.E., TO S/C SPIN AXIS

Figure 3.6-4

In Figure 3.6-5 a small sketch indicates the HGA depression angle, the angle ▲ tabulated in Figures 3.6-3 and 3.6-4. As noted in 3.6-5, a large depression angle requirement, as in the Aeronomy Mission, implies a long HGA mast to avoid the spacecraft structure. Such a long mast leads to an unfavorable moment-of-inertia ratio and possible spacecraft instability. A design goal was to limit this angle to not more than 20°, despite the 36.5° required for the 80° orbit, or even the 43.7° of the 77.5° orbit. A solution was provided by flipping the spacecraft as the Earth angle passed through 110° rather than as the sun angle passed through 90°. The analysis of this strategy is presented in Figure 3.6-6. In essence, a favorable Sun-Mars separation compensates for an unfavorable solar array attitude. In the orbits in question the solar constant is 45% and 41% above its minimum value at the time of spacecraft flip. The product of this increased solar constant and the decreased solar array output is given as the last row in the table of Figure 3.6-6. Note that in both instances the final array output is greater than its minimum value, this latter being the basis for its design. Note, however, that the 80° orbit does have a three times more comfortable margin than the 77.5° case. Thus the HGA antenna mast has been designed for a 20° depression angle. (Actually 25° was used as a design value for the spacecraft clearance, but some instruments may protrude into this field-of-view. In any case, a greater than 20° depression angle is clear of obstruction.)

## HIGH GAIN ANTENNA LOOK ANGLE REQUIREMENT

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- CLIMATOLOGY MISSION DEPRESSION ANGLE $<20^{o}$
- AERONOMY MISSION IN GENERAL CASE, DEPRESSION ANGLE CAN APPROACH $47^{o}$

PROBLEM: LARGE AERONOMY DEPRESSION ANGLE REQUIREMENT IMPLIES LONG HGA MAST WITH CONSEQUENT POOR MOMENT-OF-INERTIA RATIO.

DESIGN GOAL: LIMIT DEPRESSION ANGLE TO $20^{o}$

Figure 3.6-5

## HIGH GAIN ANTENNA LOOK ANGLE REQUIREMENT

(CONTINUED)

SOLUTION:

FLIP SPACECRAFT AT $20^{\circ}$ HGA DEPRESSION ANGLE. USE FAVORABLE SOLAR DISTANCE TO OFFSET LOWER ARRAY EFFICIENCY AT ASPECT $<90^{\circ}$

ANALYSIS

| | $I = 77.5^{\circ}$ | $I = 80.0^{\circ}$ |
|---|---|---|
| DAYS AFTER ARRIVAL* | 535 | 575 |
| SOLAR CONSTANT RATIO | 1.451 | 1.408 |
| SUN ASPECT ANGLE | $113.5^{\circ}$ | $110.3^{\circ}$ |
| ARRAY OUTPUT RATIO+ | 0.701 | 0.748 |
| RELATIVE ARRAY OUTPUT | 1.017 | 1.053 |

OBSERVATIONS

- SMALL CHANGE IN ORBIT INCLINATION CAN HAVE MARKED EFFECT ON POINTING ANGLE REQUIREMENTS.
- BOTH $77.5^{\circ}$ AND $80.0^{\circ}$ ORBITS PROVIDE SATISFACTORY ARRAY PERFORMANCE, BUT $80.0^{\circ}$ IS MORE COMFORTABLE.

---

* INDICATES DAY WHEN HGA DEPRESSION ANGLE EQUALS $20^{\circ}$

\+ INDICATES OUTPUT RELATIVE TO MINIMUM OUTPUT IN THE $0^{\circ}$ TO $90^{\circ}$ SUN ASPECT ANGLE RANGE.

Figure 3-6

# 4. SCIENTIFIC REQUIREMENTS

## 4.1 SCIENTIFIC OBJECTIVE OF MISSIONS

The Climatology Mission will map seasonal variations of dust, water, carbon dioxide ($CO_2$) and other atmospheric constituents to locate volatile constituent reservoirs on the surface of Mars and to determine transport patterns from pole to pole. This requires a long enough mission life to assure that Martian seasonal variations are observed over the entire planet. Because of the unpredictibility of the dust storms, a life longer than the extended mission's 1,374 earth days would be desirable, at least in monitoring mode. The required sun-synchronous orbit provides a good measure of seasonal variations and helps in the interpretation of nadir and limb-scanning instrument data but poor diurnal variation data will be obtained. Only during the drift period (see Section 3.4.1) will different local times be sampled on the Martian surface. The Climatology orbit option (Section 3.4.2) is specifically designed to overcome this limitation but at a cost. The cost of having to unravel seasonal from diurnal changes on the planetary surface.

The Aeronomy Mission must obtain a complete sampling of the Martian upper atmosphere, ionosphere, and solar wind interaction region for all local times over the largest possible range of altitudes. The sampling should range from a lower altitude limit set by drag effects and orbit lifetime restrictions (due to planetary quarantine) to an upper altitude limit set by the orbit period, the desire to achieve the largest possible sampling volume, and orbital precession rates. On the planet's sun side, the solar wind may penetrate to an altitude of 300 km. On the antisun side, the interaction region between the solar wind and the Martian ionosphere extends for an unknown distance (at least several planetary radii) behind Mars. The shape of this interaction region, which is similar to the Earth's magnetosphere, is either tear or bullet-shaped. The Aeronomy Mission must measure atmospheric and ionospheric densities, species, distributions, and temperature; hot plasma energy spectra; plasma wave frequencies and amplitudes; and magnetic field configurations.

## 4.2 PAYLOAD COMPLEMENTS

The baseline Climatology Mission includes three instruments, the pressure modulated infrared radiometer (PMR), the frost infrared spectrometer (FIS), and the gamma ray spectrometer (GRS). The three optional missions add five additional instruments to the possible payload complement, each option being made up of a different mix of the eight. These five additional are the ultraviolet ozone ($UVO_3$), the ultraviolet atomic hydrogen photometer (UVHP), the radar altimeter (RA), Fabry-Perot interferometer (FPI), and multi-spectral mapper (MSM). The complement of instruments on each mission is shown in Figure 4.2-1.

The gamma ray spectrometer provides mapping of the major surface and subsurface elements on a resolution scale of several hundred kilometers. Reflectance spectrometry (FIS and RA) provides information on the abundance composition, and distribution of certain major minerals on a much finer scale as well as provide surface structure details. The infrared (PMR) and ultraviolet ($UVO_3$, UVHP) sensor systems will give detailed information about the state of the Martian atmosphere, its vertical structure, its dust content, and the nature of atmospheric processes. (The UVHP, for example, provides data on the loss rate of hydrogen, and thus water, from Mars.) Atmospheric wind directions and velocities will be measured with the FPI. The MSM provides high resolution spectra (both spectral and spatial) of the Martian surface from the UV to IR range of the spectrum. Thus, for example, the rationale for payload option 3 becomes clear in that the MSM can replace much of the functional performance of the UV and IR instruments.

## 4.3 PAYLOAD REQUIREMENTS

A summary of the Aeronomy and baseline Climatology Mission requirements is given in Figure 4.3-1. The Climatology payload option requirements are summarized in Figure 4.3-2. Additional details of data acquisition rates for all missions is given in Figures 4.3-3 and 4.3-4.

Additional requirements, only hinted at in the tables, include thermal radiators, look and scanning angles, minimization of spacecraft interface, and operational modes.

In this regard, a key requirement of the GRS is providing a clear field of view for the thermal radiator that does not view the planetary surface, spacecraft, or Sun during the mission lifetime. This radiator has a FOV of 105° by 180°. The GRS also requires a low radiator background and therefore must be boom mounted away from the spacecraft.

## CLIMATOLOGY MISSION INSTRUMENT COMPLEMENTS

| INSTRUMENT | BASELINE | OPTION 1 | OPTION 2 | OPTION 3 |
|---|---|---|---|---|
| PMR | X | X | X | X |
| FIS | X | X | X | |
| GRS | X | X | X | X |
| $UVO_3$ | | X | | |
| UVHP | | X | | |
| RA | | X | | X |
| FPI | | | X | |
| MSM | | | | X |

Figure 4.2-1

## SUMMARY OF PAYLOAD REQUIREMENTS

| | AERONOMY DEPENDENT | CLIMATOLOGY REQUIREMENT |
|---|---|---|
| MASS (KG) | 53.5 | 37 |
| POWER (W, MAX) | 46.5 | 59 |
| POINTING (DEG) | 0.5/0.1 | 1°/0.2° |
| RAM VIEWING | ±30° | -- |
| DATA RATE (BPS, MAX) | 2048 | 1284 |
| THERMAL (°C) | -20 TO +40 | -20 TO +40 |
| RADIATORS | NONE | THREE |

Figure 4.3-1

## CLIMATOLOGY PAYLOAD OPTIONS

### INSTRUMENT REQUIREMENTS

| OPTION | MASS (KG) | POWER (W) | | DATA | |
|---|---|---|---|---|---|
| | | DAY | NIGHT | DAY | NIGHT |
| BASELINE | 37 | 59 | 56 | 1284 | 1164 |
| C1 | 50 | 77.5 | 74 | 1456 | 1272 |
| C2 | 57 | 71 | 61 | 1540 | 1164 |
| C3 | 53 | 67 | 55 | 3264<br>(20K)* | 1264 |

THERMAL: ALL CATEGORIES OF INSTRUMENT TEMPERATURE REQUIREMENTS ARE IDENTICAL FOR EACH PAYLOAD OPTION

LOOK ANGLES: OPTION C1 REQUIRES VIEWING FORWARD LIMB
OPTION C2 REQUIRES LIMB VIEWING AT $45^{0}$ AND $135^{0}$ FROM RAM DIRECTION
OPTION C3 REQUIRES THERMAL RADIATOR NOT VIEWING SUN, PLANET OR S/C

POINTING STABILITY: OPTION C1 REQUIRES LIMB PULSE WITH $0.08^{0}$ ACCURACY
OPTION C2 REQUIRES 3-AXIS STABILITY OF $0.5^{0}$, KNOWLEDGE OF $0.1^{0}$
OPTION C3 REQUIRES 3-AXIS STABILITY OF $0.08^{0}$

* POSSIBLE MAXIMUM FOR SPECIAL GRS + MSM OPERATING MODE, BUT AVERAGE 24-HOUR RATE NOT INCREASED.

Figure 4.3-2

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## SCIENCE DATA ACQUISITION RATES (B/S)

### CLIMATOLOGY MISSION

| | BASELINE | OPTION 1 | OPTION 2 | OPTION 3 | |
|---|---|---|---|---|---|
| PMR | 140 | 140 | 140 | 140 | |
| FIS | (120) | (120) | (120) | | (DAY ONLY) |
| GRS | 1024 | 1024 | 1024 | 1024 | |
| $UVO_3$ | | 64 | | | |
| UVHP | | 8 | | | |
| RA | | 100 | | 100 | |
| FPI | | | (256) | | (DAY ONLY) |
| MSM | | | | (1500 -12000) | (DAY ONLY; 1000 B/S AV. OVER 24-HRS.) |
| TOTALS: | | | | | |
| DAY | 1284 | 1456 | 1540 | 2680-13,180 | |
| NIGHT | 1164 | 1336 | 1164 | 1264 | |
| AVERAGE OVER 24-HRS. | 1224 | 1396 | 1352 | 2264 | |

Figure 4.3-3

## SCIENCE DATA ACQUISITION RATES (B/S)

### AERONOMY MISSION

| ORBITAL PHASE | IONOSPHERE | IONOSHEATH | APOAPSIS | |
|---|---|---|---|---|
| HOURS PER ORBIT | 0.500 | 1.500 | 4.689 | |
| INSTRUMENT | | | | |
| NMS/CWT | 256 | 0 | 0 | |
| TIMS | 256 | 128 | 0 | |
| ETP | 256 | 128 | 64 | |
| RPA/DM | 512 | 128 | 64 | |
| MAG | 128 | 128 | 128 | |
| EFD | 128 | 128 | 128 | |
| SWPA | 128 | 128 | 128 | |
| UVS | 128 | 128 | 0 | |
| FPI | 256 | 128 | 0 | |
| TOTAL BIT RATE (B/S) | 2048 | 1024 | 512 | 741.6 (AVG) |
| TOTAL BITS PER ORBIT (MB) | 3.686 | 5.530 | 8.643 | 17.859 |
| TOTAL BITS PER 32 HOURS (AVG)(MB) | | | | 85.44 |
| (MAX)(MB) | 18.43 | 27.65 | 40.55 | 86.63 |

Figure 4.3-4

Thermal radiators are also required for both the PMR and FIS, the former having an internal active Sterling cycle cooler, the latter a passive radiator. Either of the radiators for these instruments can view the planetary surface and even occasionally the sun if necessary. For Climatology payload option 3, the MSM requires a thermal radiator with more severe restrictions, again, as with the GRS, not facing the spacecraft, planet, or sun.

The look and scanning angles of the various instruments are summarized in Table 4.3-1. Note that the requirements for the FPI differ between the Climatology and Aeronomy Mission. That is, on the Climatology Mission the FPI must scan in altitude from the planetary surface at the limb to the altitude of the spacecraft. On the Aeronomy Mission the FPI does not scan but instead looks out in a horizontal plane, i.e., in a plane, at spacecraft altitude, parallel to the plane which is tangent to the Martian surface at the point of spacecraft nadir.

Minimization of spacecraft outgassing is requirea, both to keep gaseous materials clear of the line of sight of viewing instruments and from condensing on cooled detectors. A number of experiments on the Aeronomy Mission require special care in spacecraft design and instrument location to minimize the effects of spacecraft interference with the natural environment. The TIMS and RPA, for example, require that spacecraft-produced electric fields must be quite small so thermal ions and electrons are not accelerated.

The EFD, mounted some distance from the spacecraft body, samples plasma undistributed by the passage of the spacecraft. The SWPA samples particles with energies in the electron volt range. It should be placed on the edge of the spacecraft away from non-conducting materials to avoid disturbing the observed pitch angle distributions.

Spacecraft-produced electromagnetic interference (EMI) is of particular concern for the EFD and the MAG. Standard measures must be taken to assure electromagnetic compatibility with the distribution of power and data signals on the spacecraft. Interference can be reduced by properly placing these two instruments. EFD antennas must be well removed from the RPA, EFD, SWPA, and TIMS so locally produced waves do not dominate ambient signals. For the magnetometer, the spacecraft's magnetic field are typically much larger than the fields to be measured (particularly for Mars where no magnetic field mapping has been performed). The magnetometer must be boom mounted to remove it from the effects of spacecraft magnetic fields.

Normal operational modes for the Climatology Mission include differences between daytime and nighttime operation. The Aeronomy Mission has three modes depending on the orbital epoch as compared to the time or periapsis. In addition, a special operational mode has been defined for Climatology Mission option 3. For this case, the MSM would be used in a high spatial resolution mode, sampling data of a particularly interesting area of the planetary surface. During this operation the data rate would climb to 12,000 bps. It is also envisaged that simultaneous high resolution data from the GRS would be useful, raising its data rate to perhaps 8,000 bps. However, this special mode would only persist for a fraction of an orbital period so that the average data rate from these instruments over a 24-hour period would not be increased.

# 5. SYSTEM ANALYSES (BASELINE MISSIONS)

## 5.1 SYSTEM ASPECTS CONSIDERED

### 5.1.1 Spacecraft

Three spacecraft configurations were considered in performing this study, as depicted in Figures 5.1-1, -2 and -3. These will be discussed below.

5.1.1.1 Configuration A. Configuration A (Figure 5.1-1), our final choice as the baseline, uses the same basic spacecraft bus for both missions and demonstrates one approach to satisfy all mission requirements at the lowest cost.

As previously discussed in Section 4, a spinning spacecraft whose equator crosses the nadir direction seems appropriate for certain experiments. A despun platform which can track either nadir or the ram direction and also point in other directions or even scan from the nadir to limb and back is desired for other experiments. These considerations lead to a common requirement for both a spinning and an appropriately despun experiment platform with the spin axis normal to the experiment.

The Climatology Mission requires a 300 km circular orbit which is slightly retrograde from polar keeping it sun synchronous on the average. The orbit plane remains between 22° and 45° from the sun. Therefore, a conical solar array on the spinning body with a half cone angle of about 20° optimizes the solar array output for the critical 22° orbit plane to sunline angle.

The 45° limit results in significantly shorter eclipses and hence greater average power. The Aeronomy Mission is an eccentric orbit with 150 km periapsis and 3 Martian radii apoapsis altitude. This orbit cannot be made sun synchronous. However, eclipses are not as critical because they are less frequent and on the average shorter in duration. As a result, the 20° cone angle is again near optimum with the sun in the hemisphere centered about the forward spin axis. When the sun crosses the orbital plane, the spacecraft can be flipped 180° to keep the sun in this favored hemisphere.

An earth pointing HGA is needed to meet the experiment data rates with reasonable transmitter power. Additional equipment includes low gain broad

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## SPACECRAFT (CONFIGURATION A)

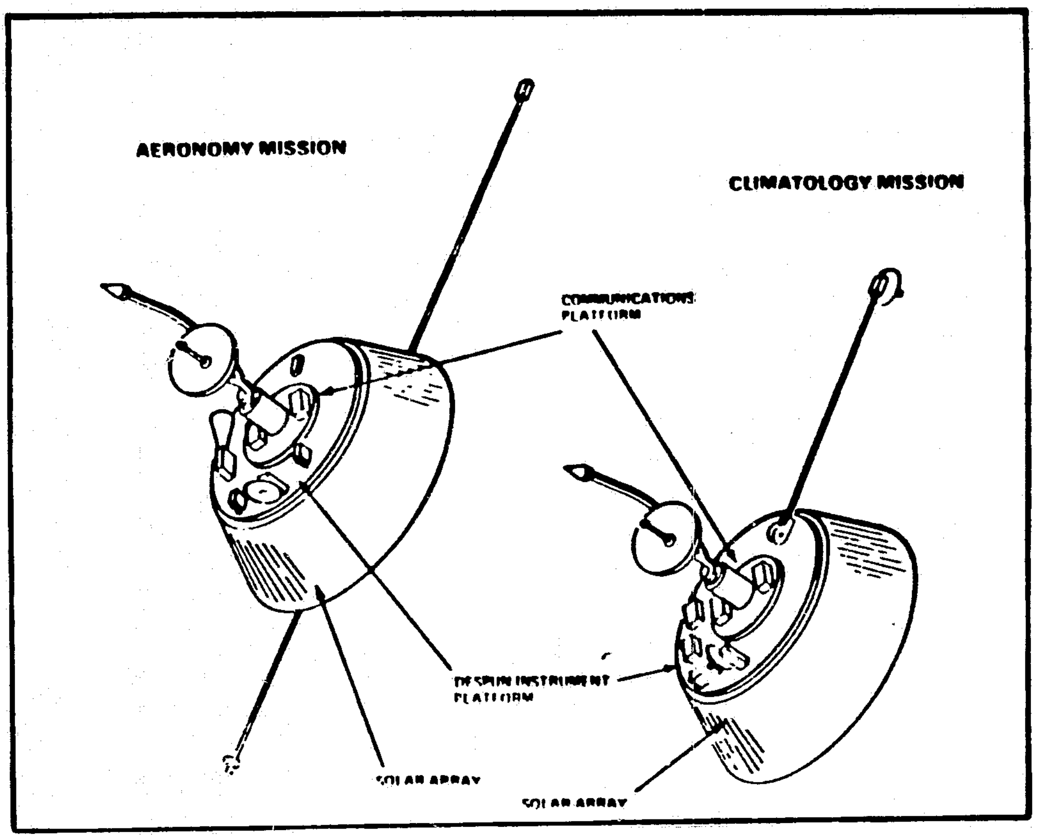
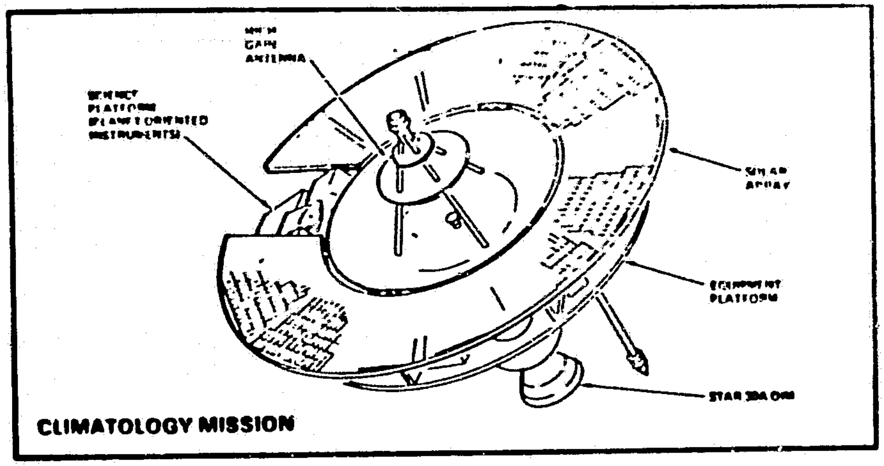



Figure 5.1-1

## SPACECRAFT (CONFIGURATION B)



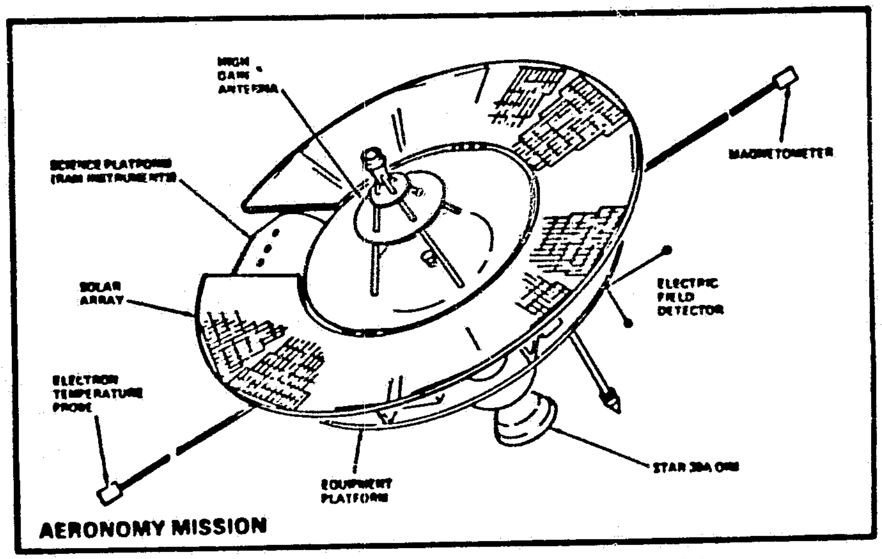




Figure 5.1-2

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## SPACECRAFT (FLTSATCOM OPTION)

EXISTING FLTSATCOM CONFIGURATION

| | |
|---|---|
| NORMAL ORBIT: | GEO-STATIONARY |
| LAUNCH: | 1978 (FIRST) |
| | ETR |
| | ATLAS-CENTAUR |
| | CENTAUR STANDARD FAIRING |
| SPACECRAFT WEIGHT: | 4170 LB |

IN ORBITAL OPERATION
THE SPACECRAFT IS 3-AXIS STABILIZED WITH THE BODY-FIXED ANTENNA POINTING CONSTANTLY AT THE EARTH AND THE SOLAR ARRAY ROTATED TO POINT AT THE SUN

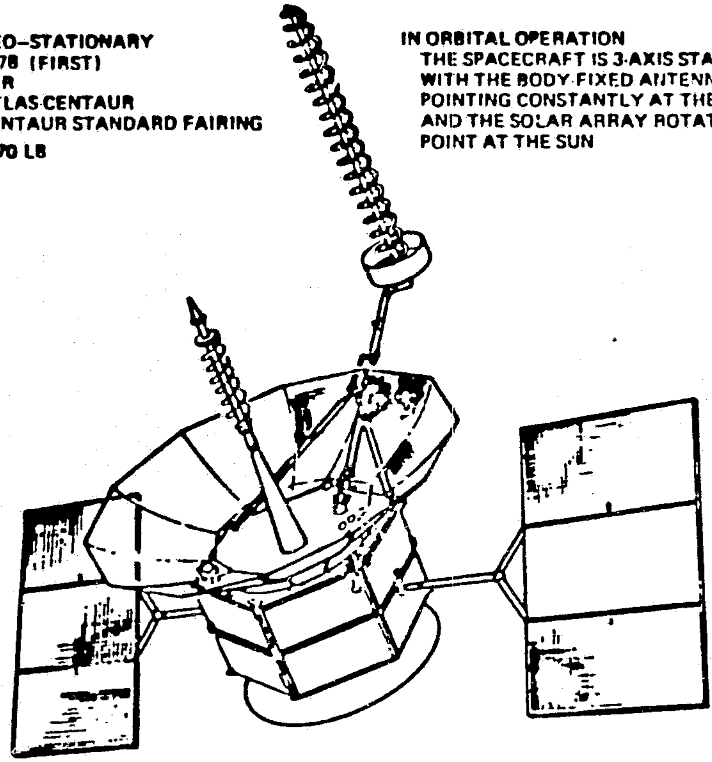

Figure 5.1-3

coverage antennas, orbital insertion propulsion, and remaining subsystems. The configuration which meets the requirements of both missions is shown in Figure 5.1-1.

The spacecraft bus features fore and aft compactness providing favorable (spin stable) inertia ratios. The conflict with providing high gain and omni antenna wide area coverage is resolved by the selected antenna location. The resulting 115° look angle from the forward spin axis is adequate for 2 Martian years. Calculations indicate that the inertia ratio margin is adequate to allow depressing the antenna to provide the required 115° angle. The fore and aft location of the peripheral equipment shelf can optimize the inertia ratios before and after Mars orbit insertion burn.

A despun experiment platform is provided as indicated in Figures 5.1-3 and -4. The despin mechanism is attached to the central cylinder which supports the Star 37 engine and is tied to the SRM-1 interstage. The platform extends to the solar array cage and meets all experiment FOV requirements. The platform allows Climatology instrument pointing in the nadir direction or slow scan from nadir to horizon. For the Aeronomy Mission, the platform could be programmed to allow pointing in the true ram direction at any time, or to point 90° from nadir or to be inertially stabilized in, for example, the periapsis ram direction. A further possibility, discussed later in greater detail, is uniformly rotating the platform at two or three times per orbit, an operation that would meet all the ram pointing requirements near periapsis.

The Aeronomy Mission spinning experiments mount aft of the solar array. Boom mounted experiments are counterbalanced both when stowed and deployed. The Climatology Mission gamma ray spectrometer is a special case in which the experimenter desires the boom to be despun. In this case, in principle, the spacecraft need be dynamically balanced only with the boom stowed. However, if the despun platform is ever inadvertently spun-up with the boom deployed, it may be impossible to again despin the platform because of the potential energy well induced by the boom. This leads to a desire to have a retractable boom, such as the Astromast chosen.

Configuration A provides independent HGA earth pointing and despun platform experiment pointing. This is achieved by a low rate azimuth drive mounted on the despun experiment platform. This drive supports a communications equipment platform which is mounted low to minimize adverse effects on the spacecraft inertia ratio. The HGA with its elevation drive and the omni antenna are on this platform along with the power amplifiers.

Further descriptions of Configuration A will be found in Section 6.

5.1.1.2 Configuration B. This section gives the rationale for the Configuration B alternate spacecraft design, its description and system performance.

The Configuration A spacecraft previously described is a straightforward design satisfying the requirements of both missions in the most cost effective manner. However, several aspects introduce elements of higher cost and complexity. To fully satisfy the programmatic objective of minimizing overall program cost, we have identified several areas of cost savings and simplicity compared with Configuration A. These are as follows:

- Decrease the number of spin/despin and gimbaled joints
- Decrease the complement of RF components and antennas
- Decrease the power requirements, and thus the battery and array weight
- Further reduce weight to use the PAM-A upper stage rather than the Intelsat VI injection stage
- Simplify ground operations

Configuration B has features exploiting each of the above areas. In particular, it is launchable by the PAM-A, with a Climatology system 14% weight margin and over 50% margin for the Aeronomy Mission. Use of the PAM-A instead of the Intelsat VI injection stage represents a major cost reduction, and eliminates depending on a licensing agreement to secure the STS deployed launch cradle.

The design contains significantly fewer components without sacrificing reliability, utilization of existing hardware, or operational simplicity. It does somewhat reduce the scientific instrument services such as scanning motion rather than pointing and reduction in data rate. Thus all instruments spin so they point in the optimum directions for only part of the spin cycle. Thus, for the Climatology Mission, the instruments cannot continuously take good data. For the Aeronomy Mission, the ETP, MAG, EFD, SWPA, and UVS are not seriously affected whereas the NMS, TIMS, RPA, and FPI can only periodically sample in the optimum pointing directions.

A more serious problem, however, is the inability of Configuration B to accommodate the viewing, look direction and thermal requirements of the GRS. Thus this instrument had to be omitted from Configuration B and therefore Configuration B was not considered as a viable option for the Climatology Mission.

The two predominant features of Configuration B which are different from Configuration A are: 1) the normal orientation is with the spin axis directed toward the earth, and 2) all instruments not compatible with spinning about the earthline at a fixed cone angle are located on a single gimbaled science platform. There is no despun platform. Specific features of this configuration follow.

The earth pointing attitude in normal operations permits high data rate downlink communication with reduced transmitter power via a large body-fixed antenna. Also, since the earth-spacecraft-sun angle is under 47° from launch plus 30 days through the end of the mission, the optimum solar array becomes a planar array normal to the spin axis. This doubles array efficiency and better utilizes STS cargo bay space.

The instrument mounting provisions are changed due to the different orientation and the deletion of a despun platform. Instruments mounted on the spacecraft body scan about the earthline at a fixed cone angle, usually 90°. The scanning motion will be the same, except about the earthline instead of about the orbit normal. We anticipate that such mounting, which is substantially the same as in Configuration A, will be favored for the ETP, MAG, EFD, SWPA, and UVS instruments of the Aeronomy Mission.

The other instruments which previously remained fixed in pointing direction and/or locked to Mars (nadir, for example) now spin with the spacecraft on a single controlled cone angle science platform.

The earthfacing aspect is a 2.0 m diameter HGA, surrounded by a solar array with a 3.7 m (145 inch) outer diameter. Over 10 inches clearance remain in the Shuttle cargo bay dynamic envelope into which the array could expand if necessary. Through an omitted array segment, the science platform points parallel to the spin axis (cone angle of 90° to 180°).

The HGA and the array are supported by N-struts from the equipment mounting platform. Propellant tanks and all spacecraft electronics are on the equipment platform. This platform, an annulus with a cutout, is supported by a conical shell connected to the Star 30A OIM attachment flange. The PAM-A upper stage adapter connects to the same flange.

Forward thrusters fire through a gap between the HGA and the array. Aft thrusters are located directly below. Circumferential thrusters on the spacecraft periphery are above the equipment platform at the Z station of the spacecraft center of mass after OIM firing. The annular platform has almost as much area as the array (7.15 $m^2$) providing ample space and mounting flexibility for all equipment. Equipment may be placed far enough from the Z-axis for a favorable moment-of-inertia ratio before and after OIM firing. This location will counterbalance the science platform equalizing moments-of-inertia about the principal transverse axes.

The gimbal mechanism and control permits the science platform mounted Aeronomy instruments to scan through the ram direction. Other Aeronomy instruments (EFD receiver and antenna, SWPA and possibly UVS) are on or near the equipment platform, and some (MAG and ETP) are on booms extended from the equipment platform.

5.1.1.3 The FLTSATCOM Option. A third option considered for the Mars Orbiter is the use of the FLTSATCOM spacecraft as a bus, with appropriate modifications in instrument accommodation, power, and attitude control to support the mission requirements. Since the only potential advantage in using such a spacecraft design is in cost savings, and another parallel TRW study effort is looking into this very issue, i.e., meeting the mission requirements at minimum cost, no design effort has been performed as a part of this study to utilize the FLTSATCOM bus.

5.1.1.4 Spacecraft Options Compared - Choosing Configuration A. In Figure 5.1-4 the advantages and disadvantages of each of the three configurations that we have considered are compared.

The first point to be made is that the FLTSATCOM option, as noted above, is being studied on another contract. Thus, if for no other reason,

# CONCEPTUAL DESIGNS

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| | OPTION PAIRS | | | |
|---|---|---|---|---|
| CLIMATOLOGY MISSION C | CONFIG. A | CONFIG. A (PAM-A+*XX) | CONFIG. A (SRM-1 OR PAM-A+*XX) | FLTSATCOM (SRM-1) |
| AERONOMY MISSION A | CONFIG. A (SRM-1) | CONFIG. A (PAM-A+*XX) | CONFIG. B (PAM-A) | FLTSATCOM (SRM-1) |
| ADVANTAGES | BEST FOR SCIENCE | | NO DESPUN PLATFORM OR ASSOCIATED EQUIPMENT (MISSION A) | (MISSIONS C, A) |
| | WEIGHT NOT CRITICAL | CHEAPER LAUNCH VEHICLE (MISSIONS C,A) | (MISSION A) | WEIGHT NOT CRITICAL<br>EXISTING SPACECRAFT<br>COST? |
| DISADVANTAGES | NEW SPACECRAFT DESIGN | | | MUST QUALIFY FOR STS<br>TECHNICAL DEFICIENCIES:<br>• POWER<br>• ATTITUDE CONTROL<br>• PROPELLANT<br>SCIENCE DEFICIENCIES:<br>• NON-SPINNING<br>• LARGE PANELS |
| | | (SOME WEIGHT CONTROL) | | |
| | DESPUN PLATFORM AND ASSOCIATED EQUIPMENT (MISSIONS C,A) | | (MISSION C) | |
| | | | MISSION A:<br>• GIMBALLED SCIENCE PACKAGE<br>• SCANNING DATA ACQUISITION<br>SACRIFICES COMMONALITY (SC + LV) | |

Figure 5.1-4

choosing the option would eliminate the possibility of having two completely different designs to compare and contrast in terms of costs and requirement satisfaction. Moreover, there are significant problems in using the FLTSATCOM bus that cannot be entirely overcome without some relaxation of science requirements. For example, the large solar panels may seriously influence the electric fields around the spacecraft, thereby modifying the natural charged particle environment. There is also some question of their residual magnetic field unless special wiring techniques are used, techniques which could have a significant cost impact. Having a non-spinning spacecraft would be satisfactory for the Climatology Mission, but would create instrument pointing problems for the Aeronomy Mission, especially with the SWPA which requires mounting on a spinning platform. Other problems include possible propellant deficiencies, especially for the Aeronomy Mission with its drag compensation requirement, and the more stringent attitude control required by some Climatology instruments as compared to the control required by the antennas on FLTSATCOM. For all these reasons, the FLTSATCOM option was eliminated from further consideration.

The next option under consideration is Configuration B. Since this configuration cannot support the GRS on the Climatology Mission, it has only been considered for the Aeronomy Mission. Immediately, a major disadvantage appears in that the designs for the two missions no longer are common, both in terms of spacecraft system and the launch vehicle. In addition, as noted above, the data taking periods of a number of Aeronomy instruments will be markedly reduced since there is no longer a despun platform. Thus further consideration of Configuration B beyond the system design was terminated, leaving Configuration A as the only viable option.

In studying Configuration A the possibility of launching with an alternate vehicle was considered. For example, the use of a PAM-A plus a Star motor as a second stage has not been ruled out. This option is discussed further in Section 5.1.2 below. Suffice it to say that the use of a PAM-A launch vehicle places some weight restrictions on the design (leading to higher costs) and almost certainly limits the launch to the 1988 opportunity, since 1990 requires substantially greater energy. Furthermore, the continued availability of the PAM-A and its STS launch cradle is open to question since no more PAM-A's are planned for the future.

For these reasons the baseline spacecraft configuration that will be described in the remainder of this report is Configuration A launched from the STS with the Intelsat VI injection stage.

### 5.1.2 Launch Vehicle Upper Stages

Consideration was given to the use of a launch vehicle upper stage other than the SRM-1. In particular a PAM-A was considered, as well as a PAM-A with various Star motors as a second stage. The PAM-A alone was found to be totally inadequate having a capability on the order of 800 kg for a Mars mission. Consideration of various second stages pointed up the fact that the currently designed PAM-A cradle has a loading capability of 4400 pounds based on its design for the Intelsat V launch. Discussions with the MacDonnell-Douglas Corporation indicated that the current cradle design was capable of supporting up to 5000 pounds provided care was taken with c.g. locations and reasonant frequencies. Furthermore the motor itself was being improved to yield a considerably higher specific thrust, $I_{sp}$. With these new figures the best performance was attainable with an off-loaded Star 37XF and a bipropellant spacecraft propulsion system, the latter being used for both MOI as well as the baselined monopropellant functions. Under all these assumptions the maximum spacecraft weight capability at upper stage separation was found to be 1322 kg. If the baseline weight margins are somewhat reduced the PAM-A/Star 37XF launch vehicle could be used for the Aeronomy Mission. However, as the spacecraft now stands, the Climatology Mission is just within the 1322 kg limit with no margin in either propellant or spacecraft weight. Thus, in this latter case, rather stringent weight budgeting would necessarily be instituted if the PAM-A option were chosen. With the combination of stretching the PAM-A to its presumed limits and placing strict weight control on the spacecrafts design the PAM-A was eliminated from further consideration as the baseline option. Nonetheless, nothing in the baseline design precludes the use of a PAM-A launch vehicle, i.e., under changing circumstances in availability, costs, etc., the PAM-A still represents a possible option.

## 5.2 SPACECRAFT ATTITUDE STABILIZATION

Attitude stabilization is closely tied to the entire configuration approach. Candidates associated with different configurations are:

| Configuration | Stabilization |
|---|---|
| Proposal A | Spin stabilized about orbit normal. Despun science platform. |
| Proposal B | Pure spinner, about earth line. Gimbaled science platform. |
| FLTSATCOM | 3-Axis control, but with momentum wheel providing loose stabilization about pitch axis. |
| --- | A momentum wheel which can absorb all angular momentum of a 5-rpm body to despin it. |

The final candidate is rejected, because the size of this momentum wheel is beyond current experience. The other three are all feasible. With the selection of Configuration A of the proposal, the options are narrowed. Details of spin rate, spin control, moment-of-inertia control, and nutation damping are covered in other sections, particularly Section 7.7.

## 5.3 ATTITUDE ASSUMED THROUGHOUT MISSION

The candidates for a dual spin spacecraft as defined by Configuration A are given in the following table:

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## ATTITUDE CANDIDATES

| Phase | Injection from earth and MOI | Other ΔV | Interplanetary transfer | Mars orbit |
|---|---|---|---|---|
| **SPIN AXIS ORIENTATION** | | | | |
| Candidates | $\boxed{\text{Parallel to } \Delta V \text{ vector}}$ | $\boxed{\text{Parallel to } \Delta V \text{ vector}}$ | $\boxed{\text{Perpendicular to ecliptic plane}}$ | $\boxed{\text{Perpendicular to orbit plane}}$ |
| $\boxed{\text{Selected}}$ | | $\boxed{\text{Retain Cruise Attitude}}$ | Parallel to earth line | |
| **DESPUN PLATFORM** | | | | |
| Candidates | $\boxed{\text{Locked}}$ | $\boxed{\text{Locked}}$ | $\boxed{\text{Locked}}$ | $\boxed{\text{Despun: Point, ram, nadir, limb}}$ |
| | Spun up | $\boxed{\text{Despun}}$ | Despun: Point HGA to earth | |
| | Despun | Spun up | | |

There are not too many choices here. For injection from earth and MOI it is clear that the spin axis must be parallel to the ΔV vector, and the despun platform must remain despun, both for its protection and to avoid large attitude changes during motor burn. We elected to keep the despun platform locked (spinning) until after MOI, and then unlock it permanently and despin it.

This precludes "spun up" (but unlocked) state of the despun platform for these maneuvers.

For other ΔV's we implement two approaches for the spin axis orientation. For large ΔV's (such as the first TCM or the orbit inclination change maneuver) the selection is to turn, burn (axial thrusters), and return, so the spin axis is parallel to the $\Delta V$ vector. For small $\Delta V$'s, especially at great ranges where it is less comfortable to precess the spacecraft away from its cruise attitude, $\Delta V$'s are performed in the cruise attitude: continuous axial firing, pulsed transverse firing.

There are two implementations of the despun platform, also. For $\Delta V$'s before MOI, the despun platform remains locked. After MOI, the despun platform remains despun. This has the advantage of keeping the HGA directed at the earth, but the disadvantage of a despun static imbalance during thruster firing which could lead to attitude errors. If analysis shows these errors to be excessive, it would be best to spin up the despun platform for $\Delta V$'s after MOI, at least for those maneuvers great enough to produce too large attitude errors.

For interplanetary transfer we have selected a spin axis orientation perpendicular to the ecliptic plane. This satisfies power, thermal, and communication requirements -- the latter by use of an S-band omni uplink and an X-band bicone with a fan beam downlink. The other option, spinning about the earth line, is also feasible, although it makes thermal control more difficult when near syzygy: earth and sun aspect angles are both $\sim 0°$. For communications, one could now use the spinning HGA, but its stowed attitude would have to be changed to point in the +Z direction.

## 5.4 DATA HANDLING AND COMMANDS

Some of the issues which arise in the selection of the data concept are:

- Command storage capacity and partitioning
- Packetizing
- Must tape recorders be rewound before playback?
- Utilization of two tape recorders (assuming no failure)
- Interleaving of real-time and stored data for downlink transmission
- Selection of record, playback, and downlink bit rates

For economy, we have chosen existing hardware for all components for this subsystem. (The tape recorder development is not yet complete.) This influences many of the choices above.

- Command storage capacity of 256 32 bit words is standard; it is expandible in 256-word increments. The selected equipment offers command partitioning between repetitive and responsive commands; it has been so employed on the Solar Mesospheric Explorer.

- The chosen equipment is applicable to time multiplexed data collection. Packetizing is not selected.

- For convenience of operation, we selected Model B which operates two tape recorders. They are on simultaneously during the tracking period only. They are read out backward from the record direction. For reliability it will either (a) be necessary to implement a one-recorder mode also, for use if one recorder fails, or (b) require the acceptance of degraded data acquisition if one recorder fails. (a) is the better choice, although we have not indicated it in this study.

  With the selected equipment, and almost no changes in details, Model C could be utilized. This would rewind the recorders and permit all-forward record and playback.

- Interleaving of real-time and played-back data is by the bit substitution, method discussed in Section 6.3.5.

- Record, playback, and downlink data rates are also selected and described in Section 6.3.5.

## 5.5 ANTENNA COMPLEMENT

As a result of the attitude selection and the decision to keep the despun platform locked until after MOI, the evolution of the antenna complement was straightforward. It is described in Sections 6.2.6 and 7.3.

The antenna complement is also a consequence of a number of other decisions involving the whole RF communications subsystem. These decisions and their rationale are discussed in Appendix A.

## 5.6 PROPULSION COMPLEMENT

### 5.6.1 Thrust Levels.

Two thrust levels are needed: a high thrust for (large) axial $\Delta V$, for spin up, and for automatic nutation control while the spacecraft and upper stage are attached; a low thrust for fine precession maneuvers.

Standard 1 lbf thrusters took too long for the first spin up maneuver, and they are probably inadequate for automatic nutation control. A larger existing thruster, 5 lbf, was selected.

Standard 1 lbf thrusters are a little coarse for fine precession maneuvers, but are probably satisfactory depending on the interpretation of the science pointing requirements. However 0.1 lbf thrusters, standard on a current spacecraft program were selected.

5.6.2 Number of Thrusters

Spin-stabilized spacecraft can be controlled with very few thrusters, at least in comparison with the 12 selected for this configuration. However, the following criteria were observed:

- Moments attained by couples
- Resultant forces along spacecraft geometrical axes
- Redundancy

These, plus the two thrust-levels discussed above lead to 12 thrusters. They are described in Section 6.2.7.

5.6.3 Tanks

For symmetry, at least three must be used. For valving redundancy it is best to have an even number. Four existing 22-inch tanks satisfied the propellant capacity requirement, and fit well on the equipment shelf. They were selected.

5.6.4 Propulsion Distribution

The set of valves employed between the tanks and the thrusters was picked to satisfy these criteria:

- Shuttle safety requirements
- Implementation of redundancy

The resulting system is described in Section 7.7. It satisfies the criteria.

"Redundancy" does not mean survival and successful completion of the mission under all circumstances. It doesn't protect against these conditions:

- If a tank sustains a major leak early in the mission (even if it is detected early, and isolated by command)
- If a thruster develops a significant leak, and it is not detected until too much propellant has been lost.

There is no good way to take steps to survive the first condition. It is possible to create an automatic protection routine to survive the second condition, but it would be complex, probably not fool-proof, and therefore not a good candidate for implementation.

# 6. SYSTEM CONCEPT (BASELINE MISSIONS)

This section describes the spacecraft system concept for the baseline Climatology Mission and the Aeronomy Mission. A single description will suffice, as the differences between the spacecraft for the two missions are few; they will be pointed out as those features are discussed.

## 6.1 CONFIGURATION

The physical configuration of the spacecraft is shown in Figures 6.1-1 (Climatology Mission) and 6.1-2 (Aeronomy). The Z-axis is the spacecraft spin axis. It also coincides with the longitudinal axis of the Intelsat VI upper stage, which utilizes a solid motor based on the SRM-1 (the large motor) of the IUS (Inertial Upper Stage).

The nominal attitude assumed by the spacecraft during earth-Mars transit is with the Z-axis perpendicular to the ecliptic plane. In the nominal attitude in Mars orbit, the Z-axis is perpendicular to the orbit plane, with +Z directed toward the side of the orbit plane the sun is on.

A functional block diagram of the spacecraft is shown in Figure 6.1-3. It identifies the various subsystems, and shows which elements are on the spinning section and which are on the despun.

### 6.1.1 Solar Array

The configuration is dominated by a body-fixed solar array in the form of a truncated cone. The greater diameter is 4.32 m (170 in.), and the lesser 2.79 m (110 in.). The height of the cone is 2.08 m (82 in.), leading to a half-cone angle of 20 degrees. The 20-degree angle is chosen to maximize power derived from the array considering that the sun may shine from any direction in the +Z hemisphere (0° < sun aspect angle, SAA < 90°).

The mathematical angle giving this maxi-min performance is 18.6 degrees, but because of increased reflections near grazing incidence angles, the larger value of 20 degrees is used.

Solar array areal efficiency is plotted in Figure 6.1-4 as a function of $\alpha$, the array half-cone angle, and $\theta$, the SAA.

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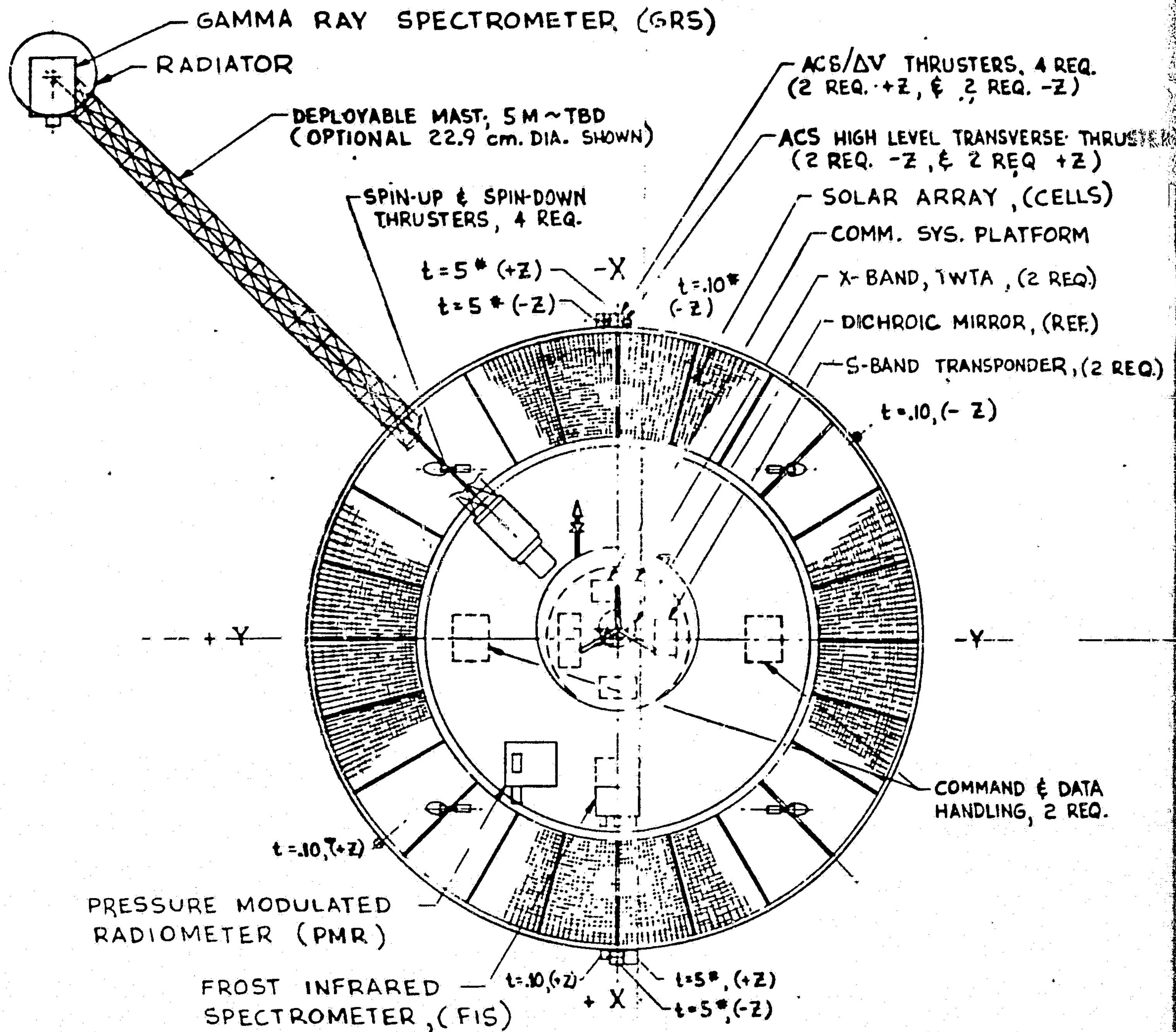
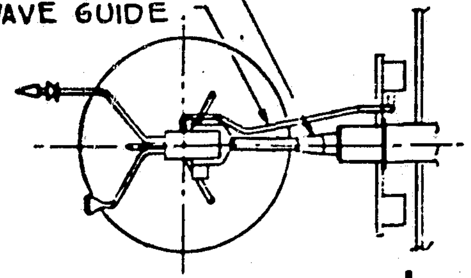
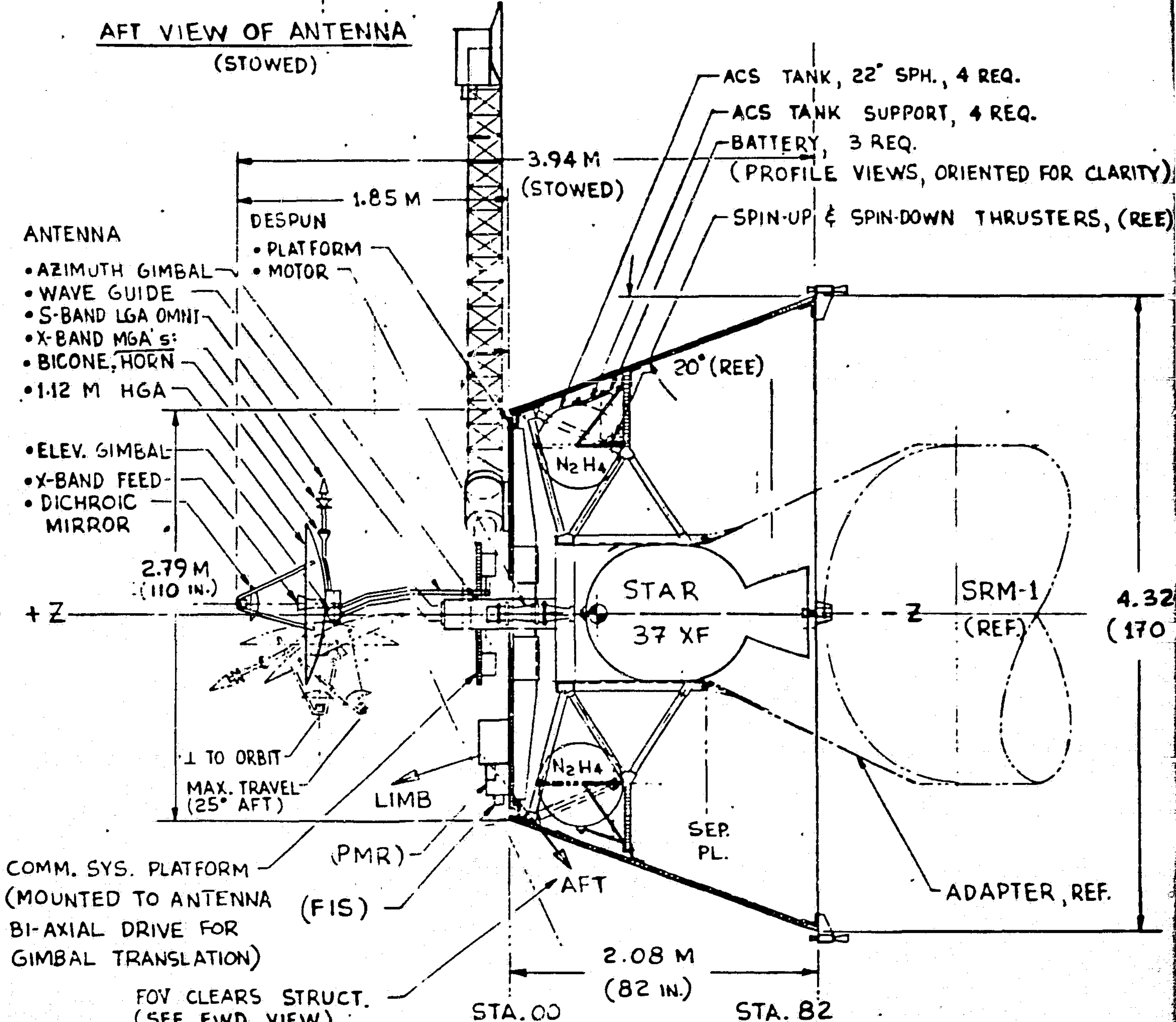



VIEW LOOKING AFT
(DESPUN PLATFORM)

FOLDOUT FRAME




AFT VIEW OF ANTENNA
(STOWED)




INBOARD PROFILE VIEW

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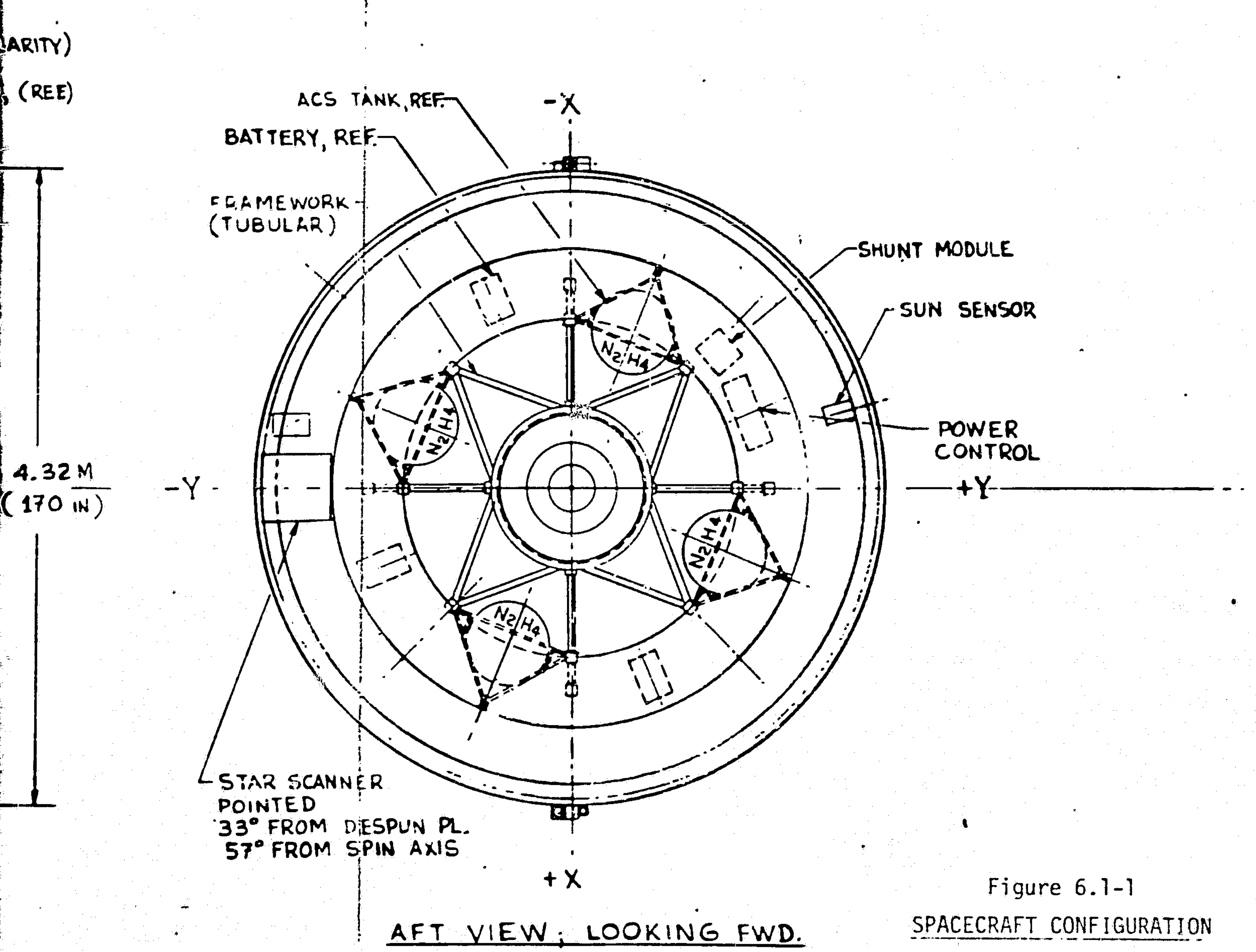
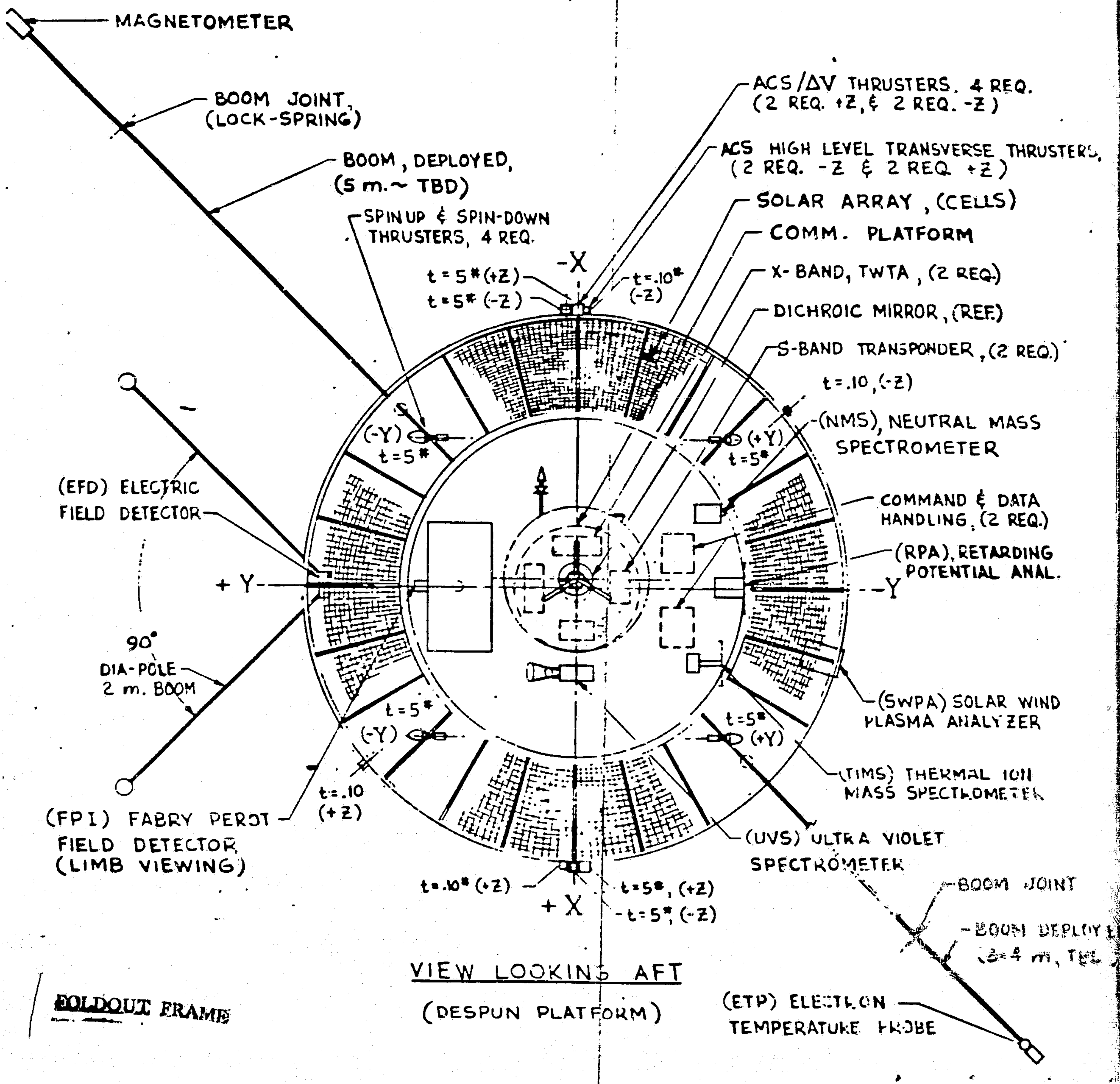



Figure 6.1-1
SPACECRAFT CONFIGURATION

CLIMATOLOGY

TRW

3 FOLDOUT FRAME




VIEW LOOKING AFT

(DESPUN PLATFORM)

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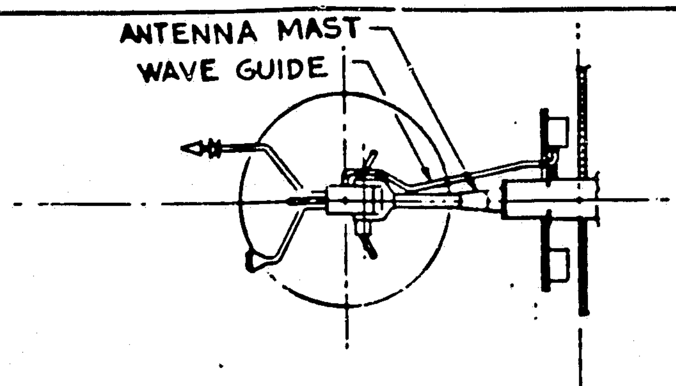
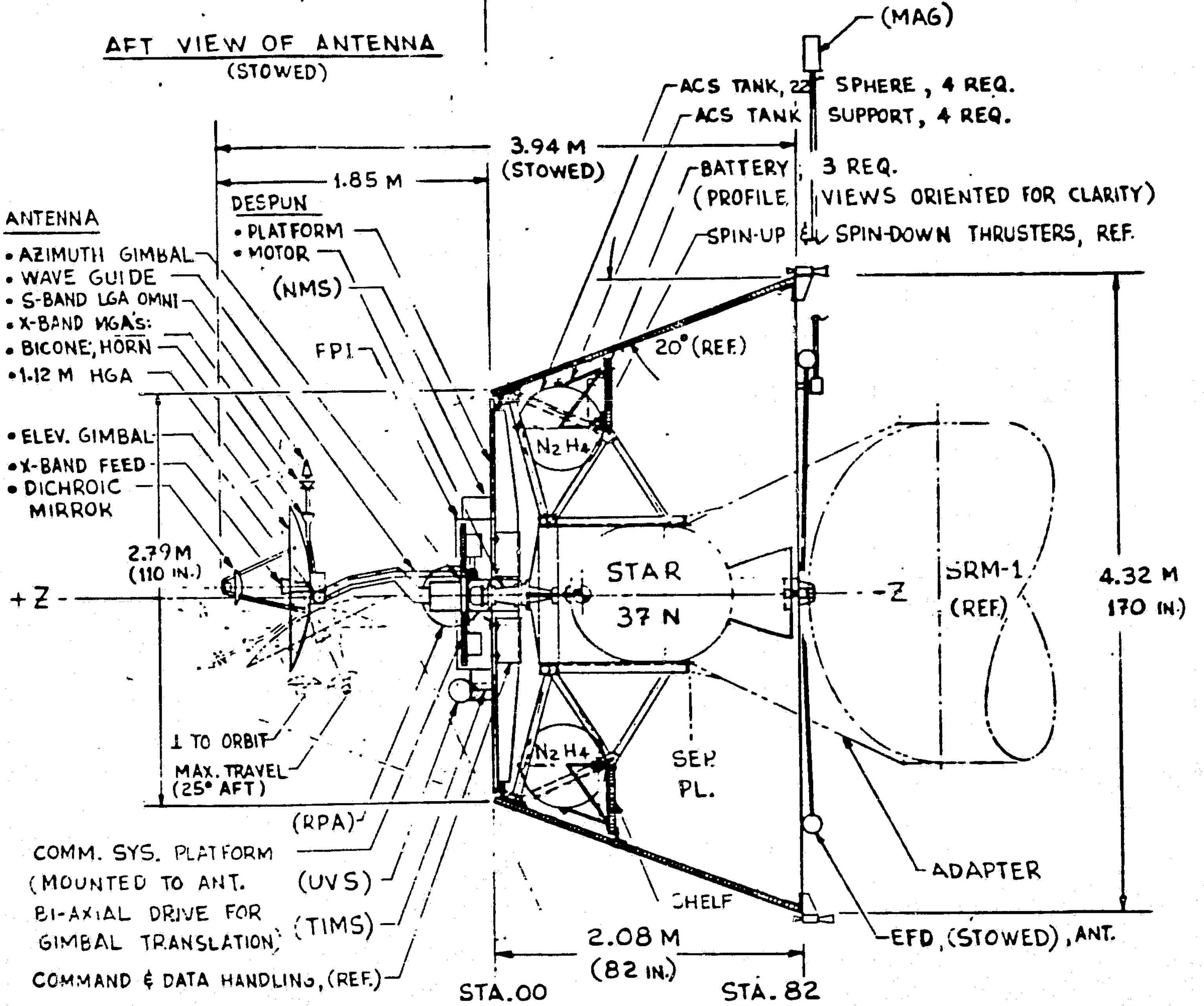



AFT VIEW OF ANTENNA
(STOWED)




INBOARD PROFILE VIEW

2 FOLDOUT FRAME

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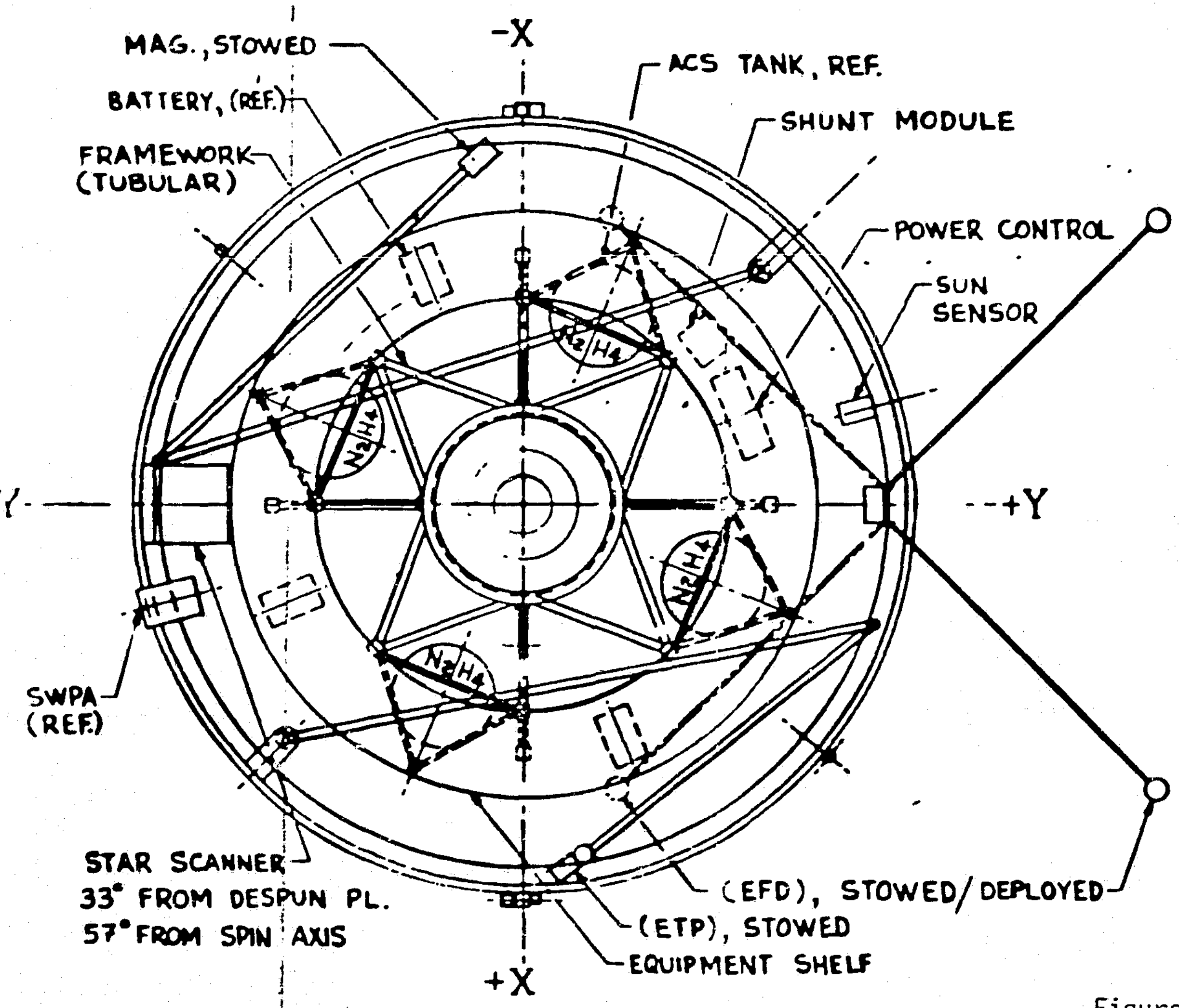


AFT VIEW; LOOKING FWD.

Figure 6.1-2
SPACECRAFT CONFIGURATION

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AERONOMY

TRW

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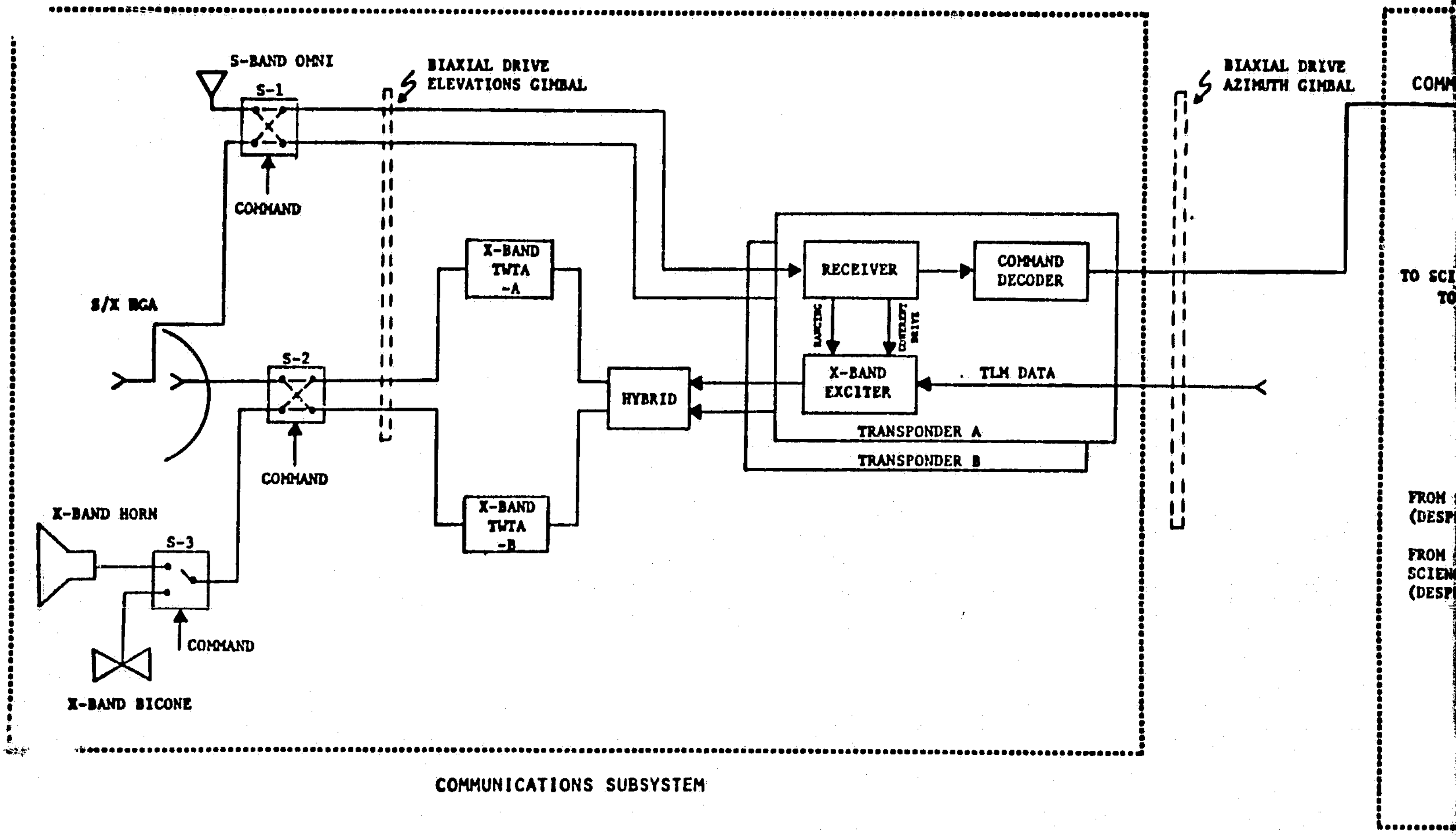
MARS O

DESPUN SECTION




COMMUNICATIONS SUBSYSTEM

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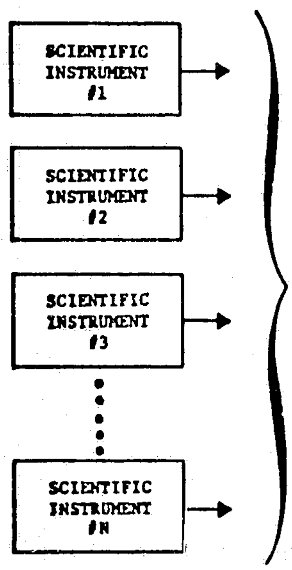


SCIENCE PAYLOAD

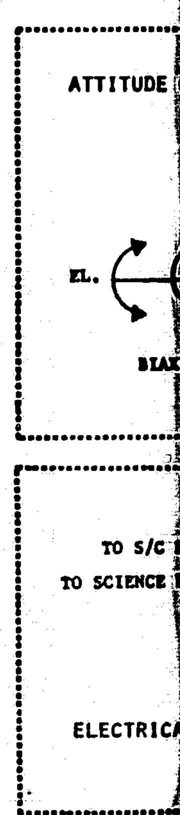


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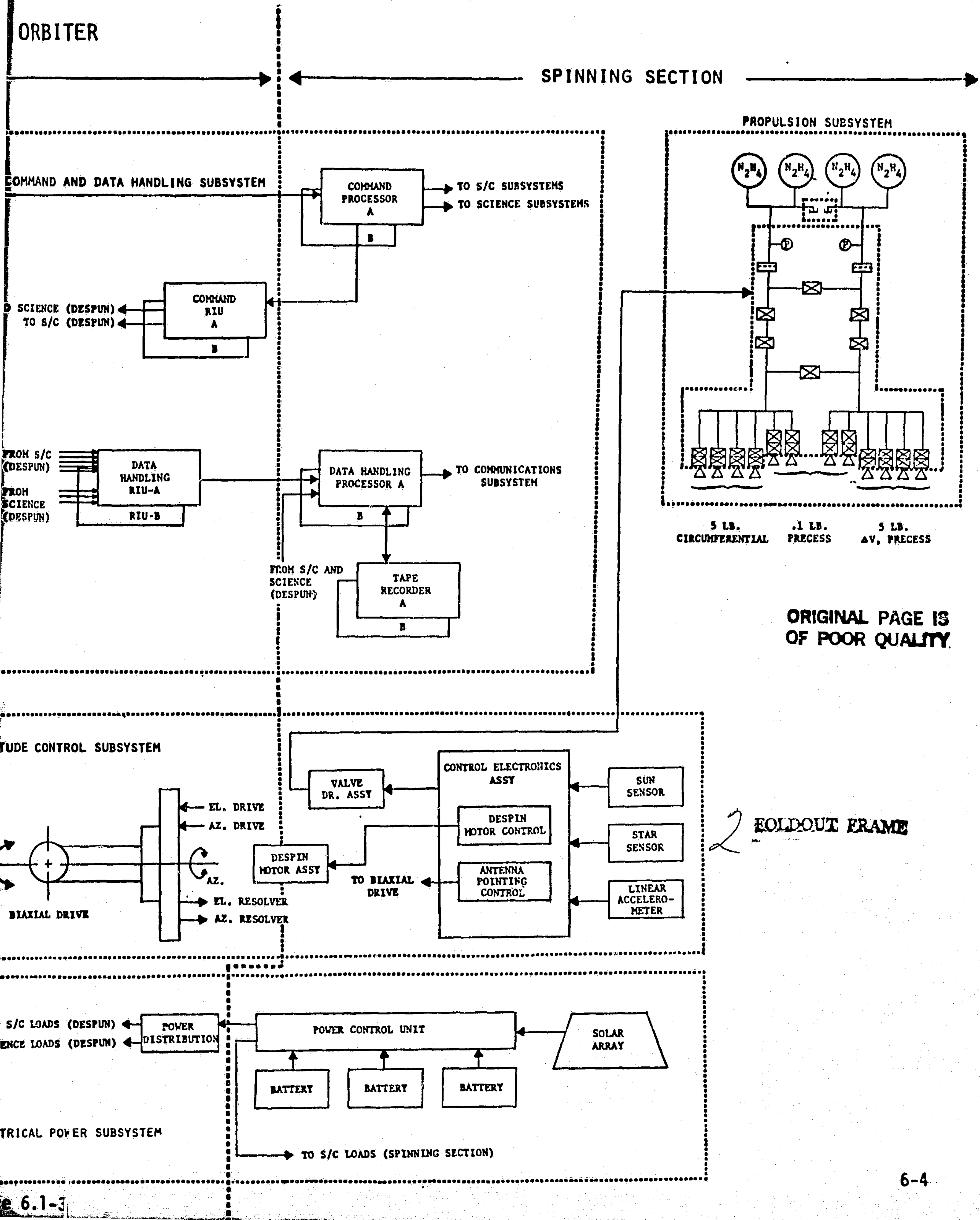
Figure

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e 6.1-3

EFFICIENCY OF CONICAL SOLAR ARRAY

$$\eta = \frac{Ai}{As} = \frac{\text{Area of intercepted sunlight}}{\text{Area of solar array}}$$

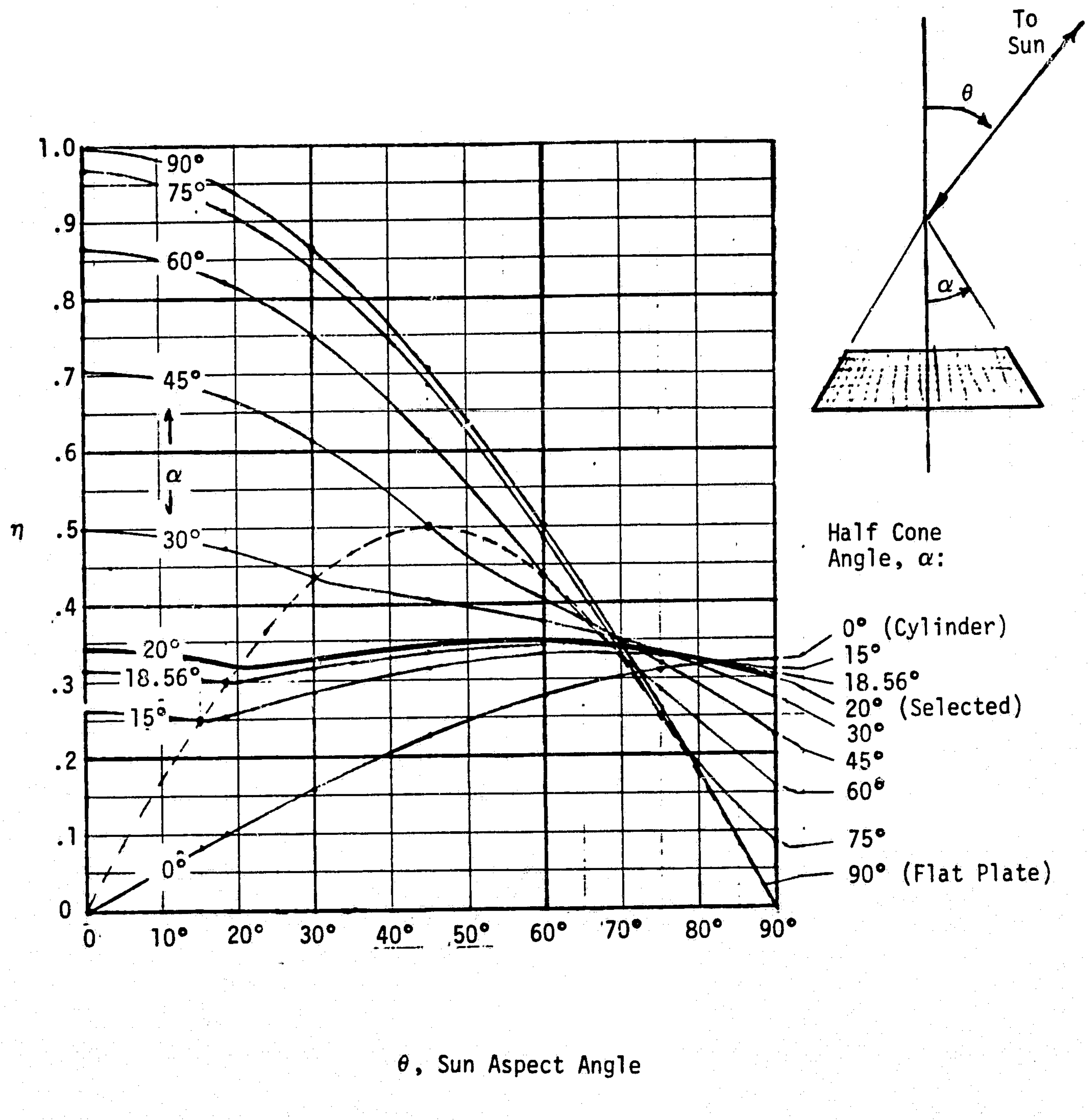


Figure 6.1-4

### 6.1.2 Despun Section

All elements forward of the array lesser diametrical plane (i.e., toward the +Z direction) and a cruciform structure just aft of it comprise the despun section, while the solar array, the equipment shelf and components mounted on it, and the entire propulsion system are in the spinning section.

Actually the despun section is not despun until after insertion into Mars orbit. For the entire earth-Mars transit is spins with the spinning section to which it is locked. This restraint is not removed until after the firing of the solid orbit insertion motor (OIM).

For the Climatology Mission all instruments are located on the despun platform, and its normal despun phase is nadir oriented. In the aft-looking view of Figure 6.3-1, nadir is down.

For the Aeronomy Mission, the instrument payload is divided. Most instruments, particularly those with objectives oriented toward the limb or ram direction, are located on the despun platform. However, two sensors are on booms deployed from the spinning section, and one uses an antenna attached to it. For the elliptical orbit of this mission, the despun platform may be positioned to follow the ram direction when in the vicinity of periapsis.

In addition to scientific instruments, the despun platform (both missions) carries all antennas and other components of the spacecraft's communications subsystem, and "remote" units of the command and data subsystems. These spacecraft antennas are double gimballed (azimuth and elevation) relative to the despun platform, to permit tracking the earth.

### 6.1.3 Equipment Shelf

On the spinning section, most spacecraft electrical and electronic components are mounted on the equipment shelf. This is an annular ring whose outer edge abuts and supports the solar array at a plane intermediate to the bounding planes. (The array is also supported by struts attaching to its lesser diameter, and stabilized by a ring at its maximum diameter.)

The equipment shelf carries the batteries and power control and distribution units of the electrical power subsystem, the $N_2H_4$ (hydrazine) propellant tanks, the telemetry and command processor, the tape recorders, and the attitude control electronics, as well as instrument electronics for instruments on the spinning section.

The inner edge of the equipment shelf is supported by struts connecting in a spoke-like arrangement to the central structural cylinder of the spacecraft which surrounds the solid orbit insertion motor (OIM).

### 6.1.4 Adapter to Upper Stage

The spacecraft-supplied adapter connects this same central cylinder to the solid motor of the Intelsat VI injection stage. It is a conical shell in design, and it is separated from the spacecraft -- and stays with the spent injection stage -- after injection from low earth orbit onto an earth-Mars transfer trajectory.

### 6.1.5 In the STS Cargo Bay

In the launch configuration, the spacecraft is supported from the injection motor, which, in turn, is tied to the Shuttle by its cradle via Shuttle sill and keel fittings.

All spacecraft booms are stowed. The despun platform is restrained to have no motion relative to the spinning section. The antenna azimuth and elevation gimbals are similarly locked with the high-gain antenna pointing in the XY plane. Furthermore, all these restraints remain in effect until after firing the OIM at Mars.

In this launch configuration, the maximum spacecraft diameter has over 5 inches (2.5 inches radius) clearance from the Cargo Bay with worst case deformation or vibration simultaneously by both elements. This represents margin into which the solar array could grow by extension, i.e., truncating the same cone at a greater maximum diameter, if additional power capacity became necessary.

Longitudinally, from the aft bulkhead of the Shuttle Cargo Bay, one foot are reserved for Shuttle functions, the injection stage and its ASE occupy ten feet, and the spacecraft occupies 3.9 m (13 feet). Of the 60-ft. Cargo Bay length, 36 feet remain unoccupied by Mars Orbiter Mission elements at the forward end of the Cargo Bay.

## 6.2 SYSTEM DESIGN

This section describes spacecraft system design characteristics.

### 6.2.1 Spacecraft Attitude Stabilization

As noted earlier, the basic attitude stability comes from spinning the spacecraft. While the large solid motors are firing (injection stage at earth, OIM at Mars) the spin rate will be in the range of 40 to 60 rpm. At other times in the mission, including interplanetary cruise and orbital operations, the spin rate will be picked at a point in the 5 to 10 rpm range.

To establish unconditional spin stability (i.e., a nutational disturbance will not grow spontaneously) the moment of inertia about a transverse axis, $I_t$, must be less than the moment of inertia about the spin axis, $I_z$, preferably less than 0.9 $I_z$.

If is a further goal, when the despun section is despun, that $I_t < I_{z(s)}$, where $I_{z(s)}$ is the spin moment-of-inertia of just the spinning section. Meeting this goal insures unconditional spin stability when the despun section is despun. It is a sufficient condition, but not necessary. Stability can be attained, even if $I_t > I_{z(s)}$, by having enough viscous damping on the despun section.

### 6.2.2 Attitude Assumed Throughout Mission

Figure 6.2-1 shows the attitude of the spacecraft throughout the mission. At injection from earth and at insertion into Mars orbit, both accomplished by large solid motors, the +Z-axis points in the direction of the required $\Delta V$ vector, tangential or anti-tangential to the local trajectory.

During interplanetary cruise, the nominal orientation has the Z-axis perpendicular to the ecliptic plane. When in orbit about Mars, the nominal orientation has the Z-axis perpendicular to the orbit plane.

There are some deviations from the nominal attitudes. When velocity maneuvers are to be performed with the liquid propulsion system, particularly for large $\Delta V$'s, it is desirable to orient the spacecraft spin axis in the direction of the $\Delta V$, to execute the velocity maneuver with axial thrusters. This is the case for the first two trajectory correc-

## ATTITUDE THROUGH MISSION

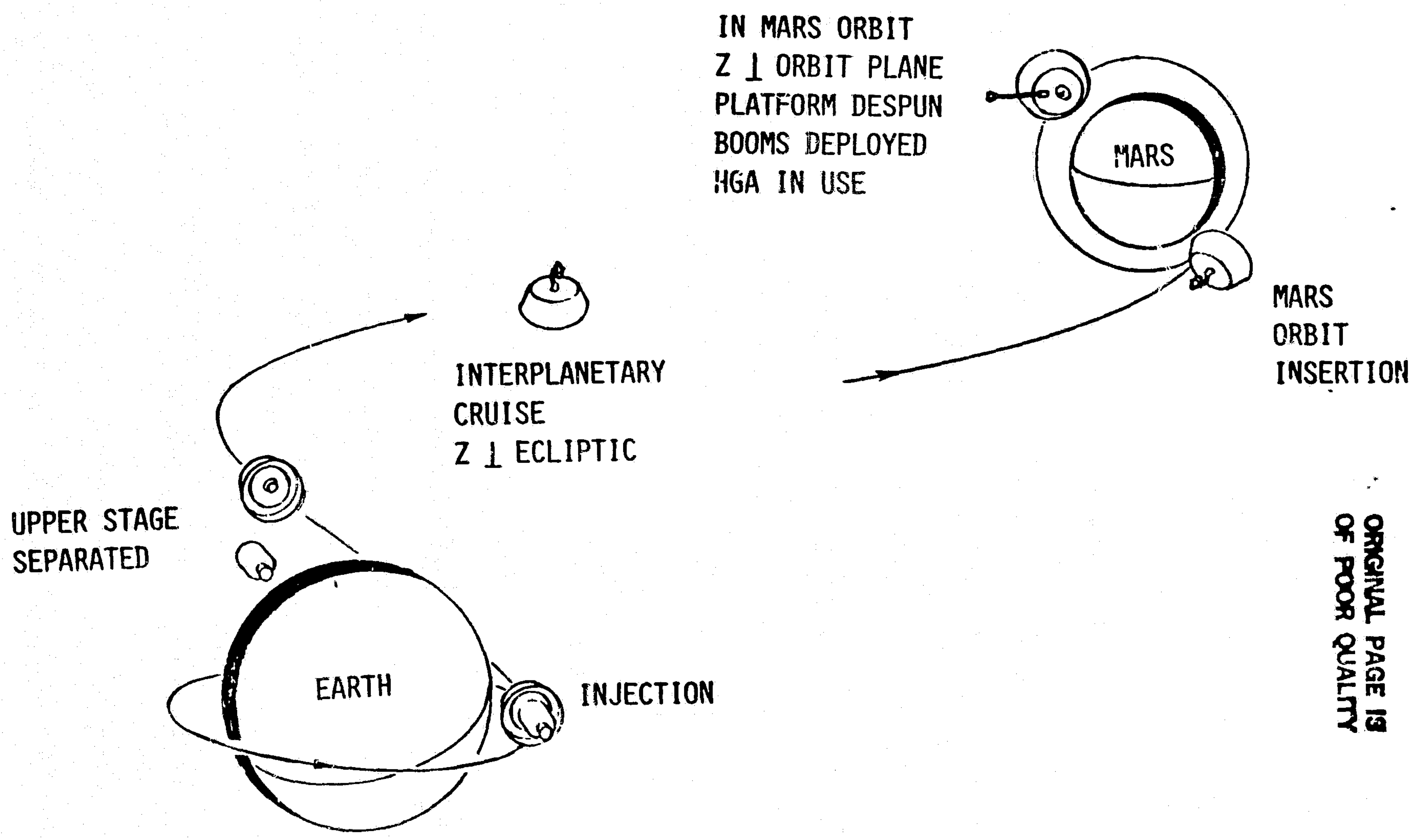


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Figure 6.2-1

tion maneuvers (TCM's) and probably for drag compensation maneuvers in orbit in the Aeronomy Mission. Early in the mission (before the first TCM) thruster calibration maneuvers will be performed, some of which require pointing the spin axis along the earth line; this is another deviation from the nominal orientation.

Other propulsive $\Delta V$ maneuvers may be performed without turning from the nominal attitude, if they are relatively small. Examples are the third TCM, ~5 days before arrival at Mars, and small orbit trim maneuvers at Mars.

The $\Delta V$ maneuver in the Climatology Mission to change orbit inclination at the end of the drift period of 100 - 150 days is directed perpendicular to the orbit plane. Even though it is a large maneuver (100 - 150 m/s) it can be performed by continuous firing of axial thrusters in the nominal spacecraft attitude.

Keeping the spin axis perpendicular to the orbit plane requires periodic precession, maneuvers, because orbit perturbations (primarily due to Mars' $J_2$, associated with its equatorial bulge) cause a continuous rotation of the orbit plane's node line in the equatorial plane. The frequency of such maneuvers depends on the rate of rotation (which differs for the three orbits: Climatology baseline, Climatology orbit option, and Aeronomy) and the tolerable misorientation. Figure 6.2-2 illustrates the effect, and determines precession maneuver intervals for these missions assuming a 1-degree maximum misorientation.

### 6.2.3 STS and Injection Operations

There are a number of events associated with injection by the upper stage onto the interplanetary trajectory which:

- are absolutely necessary to the success of the Mars Orbiter Mission (including the precise timing of their execution),
- must be executed before communications between the DSN and the spacecraft can be assured, and
- would represent a critical or catastrophic hazard to the Space Shuttle if executed before separation from the Cargo Bay

These events are listed in Figure 6.2-3; they include the firing of the injection motor and separating the spacecraft from the upper stage. The potential hazard to the Shuttle is obvious, and we interpret STS safety requirements as making it necessary to delay the enabling of these events

## PRECESSION MANEUVERS IN ORBIT

| MISSION | ORBIT INCLINATION | NODAL PRE-CESSION RATE | | ORBIT POLE ORIENTATION RATE | MAXIMUM INTERVAL BETWEEN MANEUVERS, T ERROR $\leq 1^{0}$ |
|---|---|---|---|---|---|
| | | $\frac{REV}{M.\ YR}$ | $\frac{DEG}{DAY}$ | $\frac{DEG}{DAY}$ | DAYS |
| C | $92.65^{0}$ | 1.000 | 0.524 | 0.523 | 3.82 |
| $C_0$ | $84.69^{0}$ | -2.000 | -1.048 | 1.044 | 1.92 |
| A | $77.5^{0}$ | -0.573 | - .300 | .293 | 6.83 |

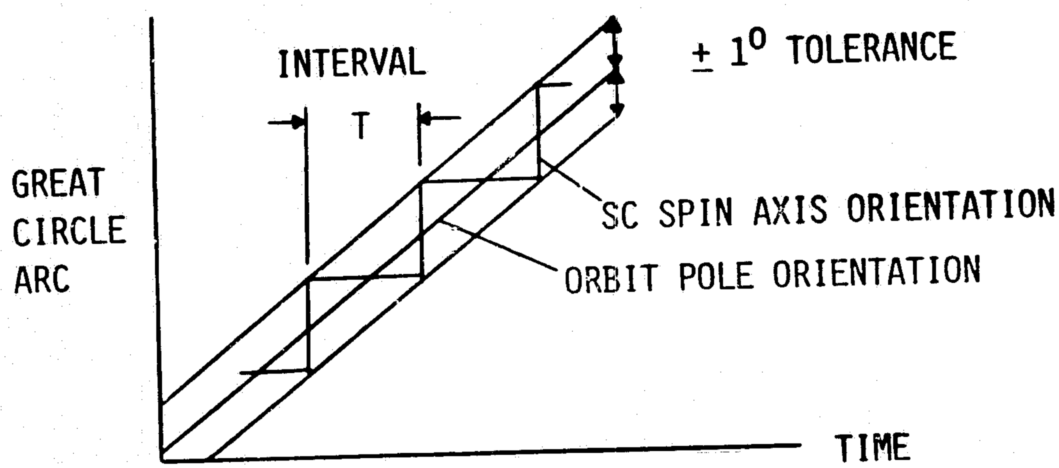


Figure 6.2-2

## SEQUENCER-CONTROLLED EVENTS (●)

| EVENT | SEQUENCER TIME (MIN.) | TIME FROM L (INJECTION) |
|---|---|---|
| (SEPARATE PL FROM STS) | | L-45 |
| (INITIATE SEQUENCER) | 0 | L-35 |
| ● ENABLE STS-CRITICAL COMMANDS TO BE DIRECTED FROM COMMAND PROCESSOR | 0.5 | L-34.5 |
| ● START SPIN-UP | 1 | L-34 |
| ● ENABLE ANC | 3 | L-32 |
| ● STOP SPIN-UP | 7 | L-28 |
| ● INHIBIT ANC | 33 - | L-2 - |
| ● FIRE UPPER STAGE | 33 | L-2 |
| (UPPER STAGE BURNOUT) | 35 | L (INJECTION) |
| ● ENABLE ANC | 35 + | L + |
| ● INHIBIT ANC | 39 | L+4 |
| ● SEPARATE SC FROM UPPER STAGE | 39 + | L+4+ |
| ● START SPIN-DOWN | 50 | L+15 |
| ● STOP SPIN-DOWN | 53 | L+18 |
| ● START PRECESSION MANEUVER (PER PARAMETERS PRE-STORED IN CEA) | 75 | L+40 |

Fiqu 6.2-3



C-2

until the Shuttle Orbiter has insured (by propulsive maneuver) that it will attain an adequate separation from the spacecraft when the injection stage is fired. Furthermore, to satisfy the two-fault-tolerant provisions, two independent enabling actions must occur.

Whether an independent (dual) sequencer is necessary on board the spacecraft to control these events, or a particular arrangement and usage of the stored command provisions within the spacecraft's command processor will suffice has not been determined. In either case we propose that the two independent enabling actions will be:

- a "breakwire" which opens when the upper stage/spacecraft is released from the Cargo Bay, and
- a specific radio command, sent from the Shuttle to the spacecraft, after adequate separation distance is assured.

### 6.2.4 Spacecraft Commanding

The spacecraft can respond to real-time commands which are executed as they are received, or to stored commands which are executed when the clock coincides with their associated stored time tag. In normal operations, nearly all commands would be stored.

Such commands would be loaded in batches transmitted from the ground to the spacecraft perhaps as frequently as daily, perhaps only once or twice per week. The heaviest demand for frequent commands will be during orbital operations at Mars when several cycles occur simultaneously:

- repetition at the orbital period:
  - instrument day/night operations (Climatology)
  - instrument periapsis operations and simultaneous tape recorder rate changes (Aeronomy)
  - tape recorder playback inhibit during occultation
  - programming the pointing of the despun platform and the HGA (azimuth gimbal)
- repetition at a daily rate (tracking, non-tracking)
  - tape recorder record and playback cycles
- periodically
  - command store contents transmitted
  - coherent operation for doppler or ranging
  - performance of precession maneuvers (See Section 6.2.2)

The selected command system has redundant command processors, each with a capability of 256 stored command slots, 32 bits each, and expandable to 512 commands if desired. To look at command rates and transmittal times, we consider the 512 command capacity. See Figure 6.2-4.

Selected uplink rates are 125 b/s (normal) and 7.8125 b/s (emergency). Insertion via the HGA will always support the higher rate. Via the S-band omni antenna at maximum range, the lower rate must be used. The times to load 512 commands at these rates are 2.2 and 35 minutes, respectively. (For ground test, an input rate of 1000 b/s can also be accepted, taking 16 seconds.)

Downlink bit rates for transmitting the command store contents for verification on the ground are 4000 b/s normal (but 2000 b/s, Aeronomy) and 16 b/s emergency, when the HGA cannot be used. Transmittal times are 4.1 seconds (8.2 seconds, Aeronomy) and 17 minutes, respectively.

### 6.2.5 Data Handling

Because it is desired to acquire scientific data continuously during the orbital phase of the mission, but ground station reception is committed for only one eight-hour shift each day, on-board storage of substantial data is necessary.

Two redundant 325 Mb tape recorders are used for this purpose.

Three models for the operating of the tape recorders are considered, and are illustrated in Figure 6.2-5. They differ in whether one or both recorders are routinely employed, and whether the tape recorder must be rewound between recording and playing back (and therefore whether data are played back in the same order as acquired or reversed). In all cases, real-time engineering and housekeeping data are interleaved with scientific data for downlink transmission.

In this study, Model B has been selected. However, the same equipment could serve in Model C. The same equipment with different record and playback rates could serve in Model A.

### 6.2.6 Antenna Complement and Use

The selected antenna complement is shown in Figures 6.1-1 and 6.1-2, and 6.2-6. The patterns of the low-and medium-gain antennas are represented in Figure 6.2-7.

## COMMAND LOADING AND DUMP

STORAGE OF COMMANDS AND TIME TAGS.

CONSIDER 512 COMMAND SLOTS x 32 BITS/COMMAND = 16,384 BITS

LOAD AND DUMP TIMES:

| | NORMAL | EMERGENCY |
|---|---|---|
| LOADING | 125 B/S | 7.81 B/S |
| | 131 S (2.2 MIN) | 2098 S (35 MIN) |
| READING OUT | 4000 B/S | 16 B/S |
| | 4.1 S | 1024 S (17 MIN) |

Figure 6.2-4

## DATA STORAGE AND PLAYBACK MODES

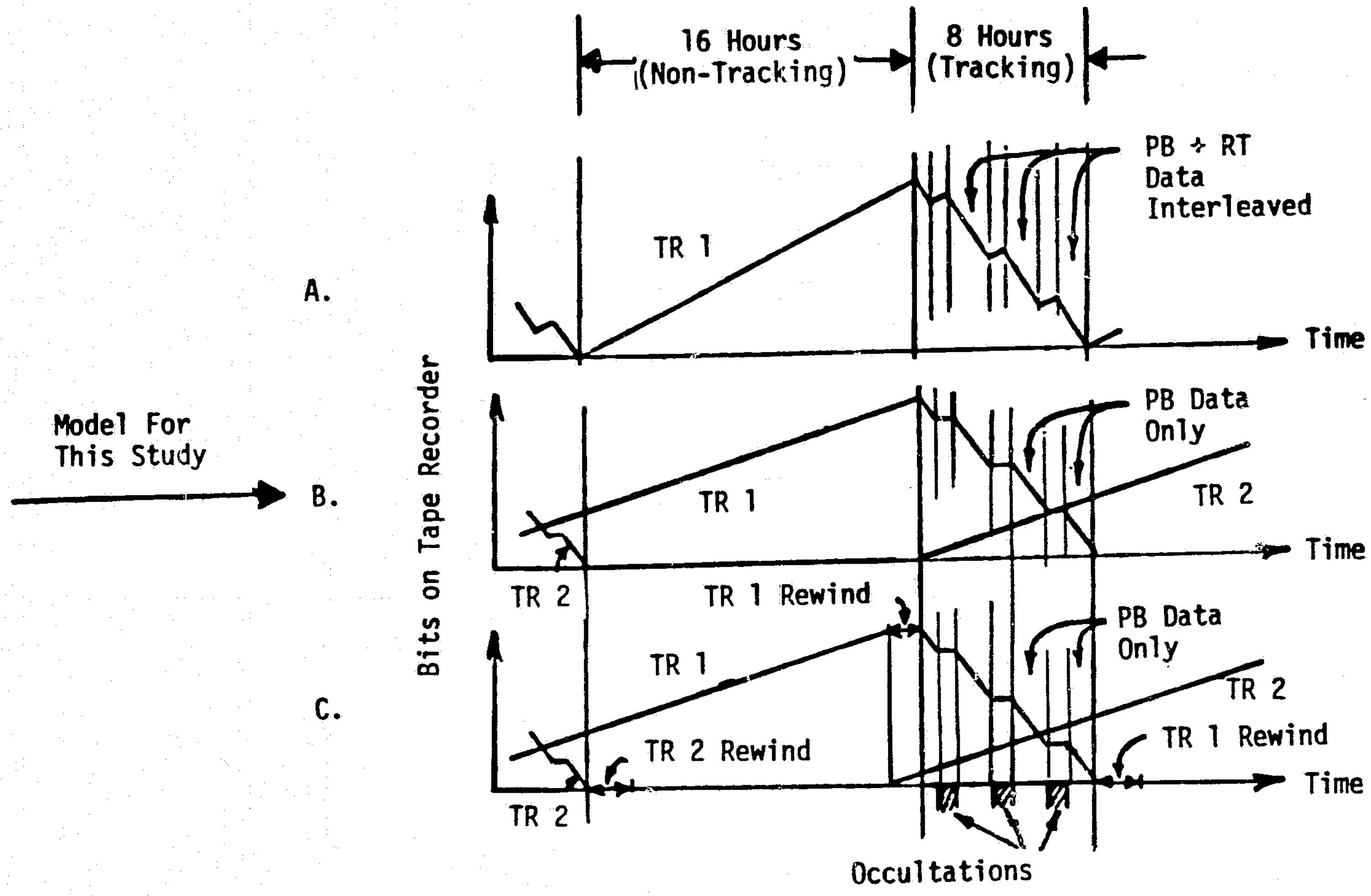


Figure 6.2-5

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## SPACECRAFT ANTENNA COMPLEMENT

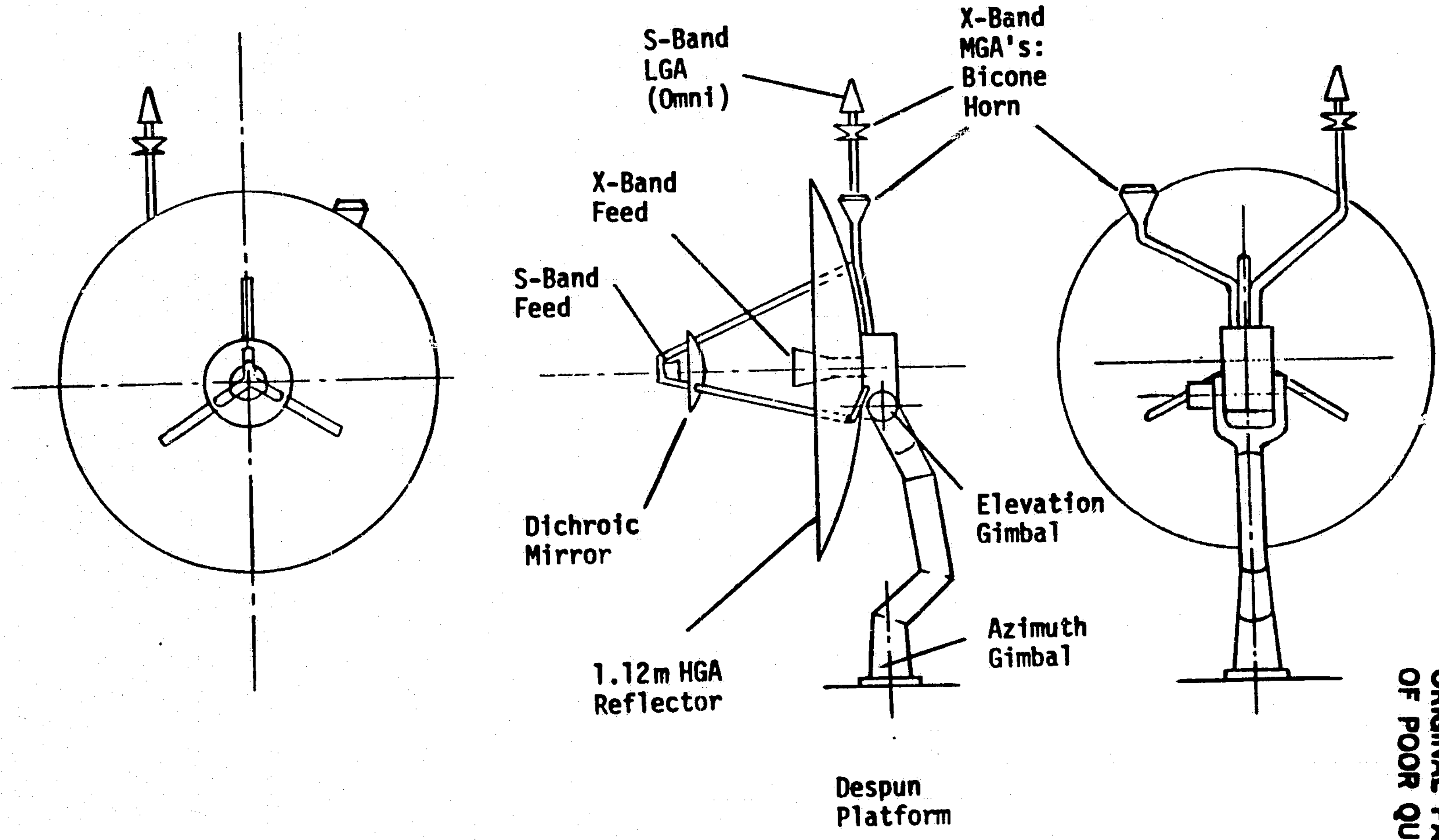


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Figure 6.2-6

## SPACECRAFT ANTENNA COMPLEMENT



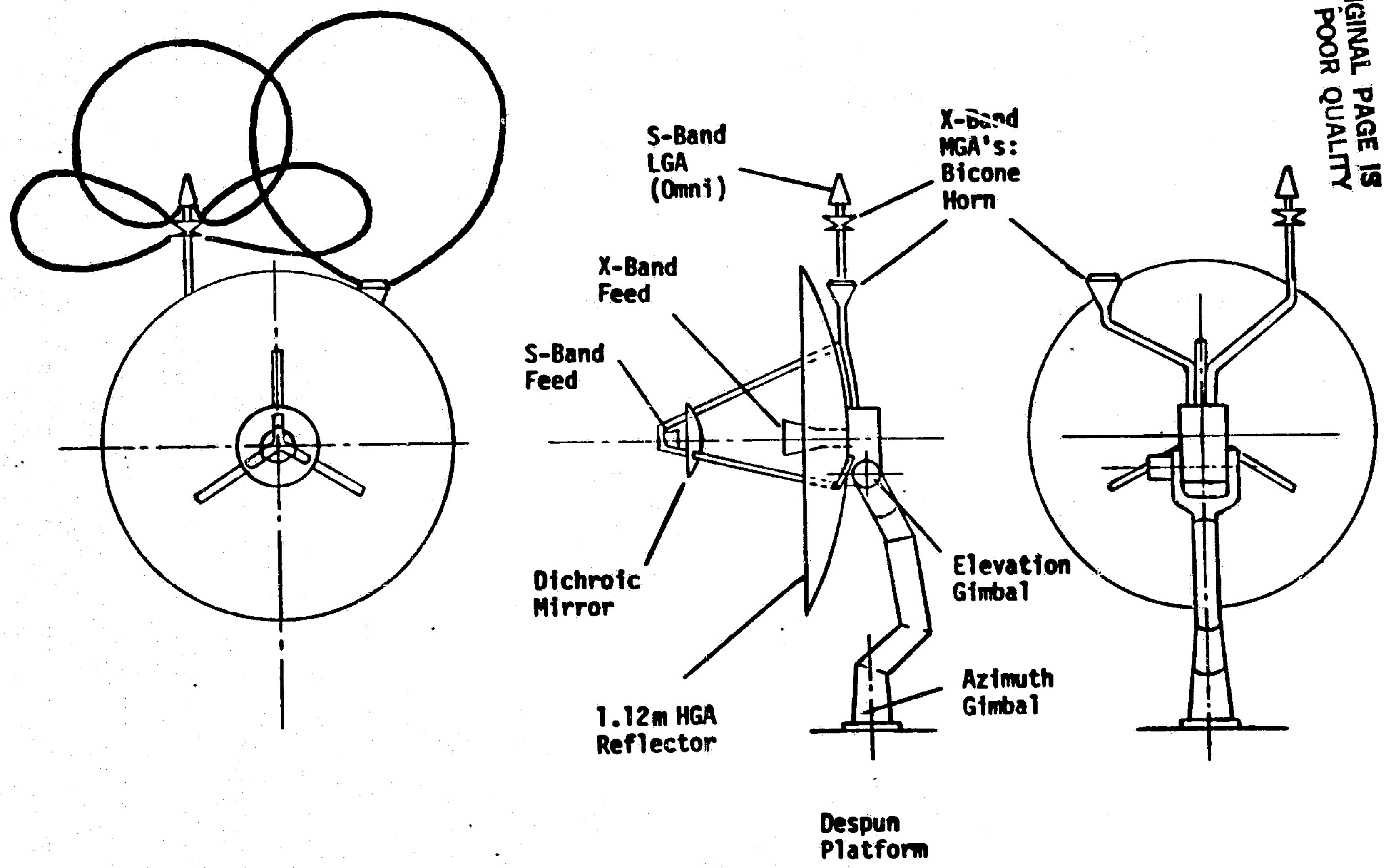


Figure 6.2-7

A high-gain antenna (HGA) is used for primary uplink (S-band) and downlink (X-band) communications when the spacecraft is in orbit at Mars. The same 1.12 m (44 inch) paraboloidal reflector from the DSCS II spacecraft is used for both functions. S-band reception is reflected from the dish, passing through a dichroic mirror to an S-band feed at the primary focus. The X-band feed is at a second (cassegrain) focus, created by reflection from the hyperboloidal dichroic mirror. This antenna has a gain of 36.9 dBi at X-band and 24.9 dBi at S-band.

The other antennas -- an S-band low gain omni and two X-band medium gain antennas (MGA's), a horn and a bicone -- are mounted to the base of the HGA reflector. They are intended for use when the HGA cannot be used, for example, before the despun section is despun. For such use, the elevation gimbal is in the launch position, as shown in the figure, with the HGA beam in the XY plane. Then all the low and medium gain antennas have patterns which are symmetric about the spacecraft Z-axis.

The coverage of these antennas is given in Table 6.2-1. Coverage is concentrated in the +Z hemisphere. In normal operation (except immediately after launch) the earthline will always be in this hemisphere.

When the HGA is despun and set at some other EAA than 90°, the omni and MGA's still have usable patterns. Instead of being symmetric with respect to the spacecraft Z-axis, their symmetry is about an axis that tilts from the Z-axis as the HGA deviates from the XY plane.

| Antenna | S-Band Omni | X-Band Horn | X-Band Bicone |
|---|---|---|---|
| Approximate coverage, earth aspect angle (EAA) | 0 to 120° | 0 to 60° | 60° to 100° |
| Maximum gain (approx.) at EAA = | 0 dBi<br>0° | 8 dBi<br>0° | 4.5 dBi<br>80° |
| Other gain (approx.) at EAA = | -3 dBi<br>90° | 4.5 dBi<br>40° | 1 dBi<br>60° or 100° |

Table 6.2-1

LGA AND MGA COVERAGE (HGA STOWED)

The HGA is double gimballed relative to the despun platform. The aximuth gimbal can travel ±270°; therefore it must be "unwound" once each orbit. The elevation gimbal can travel in the range -25° to +90° from its launch position (EAA's 0° to 115°). Depending on the height and location of instruments on the despun platform, there could be some interference with the antenna beam between EAA's of 110° and 115°.

### 6.2.7 Propulsion Complement

The complement of spacecraft propulsive components is shown in Figure 6.2-8. The solid OIM is shown in phantom on the spacecraft centerline, nozzle pointed in the -Z direction. This motor is the Thiokol Star 37XF for the Climatology spacecraft, and the lighter Star 37N for the Aeronomy spacecraft.

The monopropellant hydrazine system is shown in the figure. Hydrazine is stowed in four 22-inch spherical tanks mounted on (the +Z side of) the equipment shelf. There are 12 hydrazine thrusters on the spacecraft spinning section. Their functions, nominal thrust level, and location are shown in Table 6.2-2. Because blowdown pressurization is used, the actual thrust will vary above or below the nominal thrust according to the current pressure level.

The location of the transverse thrusters, above the equipment shelf (about 1.5 meters abive the base of the solar array) is picked to be at the same Z station as the center of mass after the OIM has been fired. This is also approximately where the propellant tanks are located, so the center of mass will move very little as propellant is depleted. Keeping the transverse thrusters at the level of the center of mass minimizes the precession incurred when pulsed thrusting is used for lateral $\Delta V$ components.

The 5 lbf axial thrusters can be used for coarse precession maneuvers for automatic nutation control. This thrust level is desirable to provide the necessary authority.

The 0.1 lbf axial thrusters, on the other hand, are intended to make fine precession maneuvers while exciting a minimum amount of nutation. This is done by letting each precession increment be the result of two couple pulses fired in sequence. The first couple pulse is achieved by

## ACS PROPULSION

(UPPER PLUMBING SHOWN)

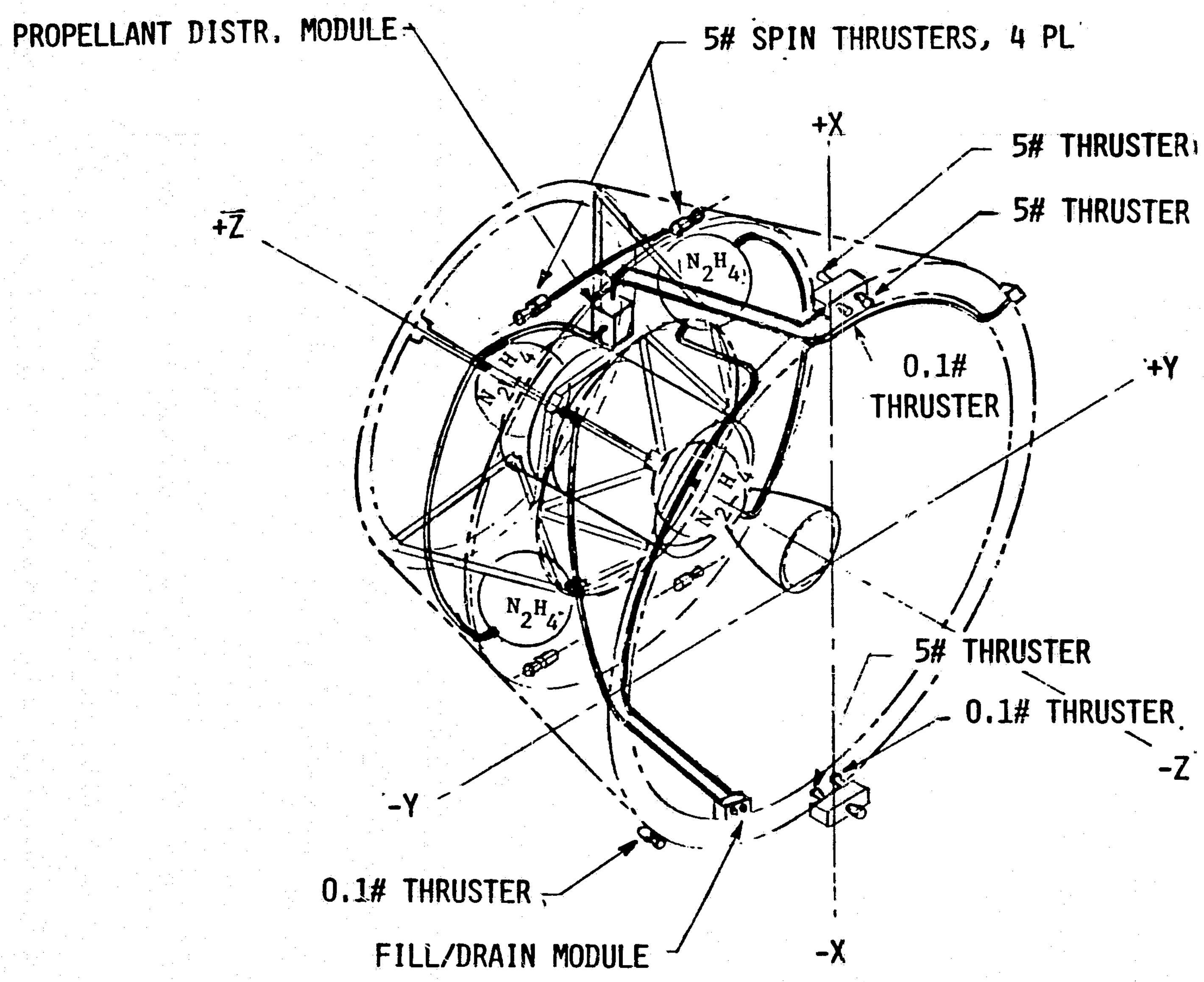


Figure 6.2-8

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one pair of ±Z thrusters, and the second by the other pair. The location of the thrusters on the array base ring and the time interval between pulses are selected so that the pair of pulses add in precession of the momentum vector, but cancel the induced nutation.

| Category/ use | Transverse/ spin control and lateral $\Delta V$ | Axial/ axial $\Delta V$ and coarse precession | Axial/ fine precession |
|---|---|---|---|
| Quantity | 4 | 4 | 4 |
| Thrust (lbf) | 5 | 5 | 0.1 |
| Location | above equipment shelf, at apertures in the solar array | base of solar array, ±X radials | base of solar array, 4 places |
| Pointing Direction | +Y (2)<br>-Y (2) | +Z (2)<br>-Z (2) | +Z (2)<br>-Z (2) |

## 6.3 SYSTEM PERFORMANCE

This section determines by analysis and presents the system performance of the Mars orbiter spacecraft. The distinction between system performance and subsystem performance is not always clear cut, and this section should be read in conjunction with the appropriate subsystem performance statements in Section 7.

### 6.3.1 Mass

Table 6.3-1 shows the allocation of mass for the Climatology and Aeronomy spacecraft, by subsystem. Each subsystem is split between spinning and despun sections.

EOL means end of life; in this condition there is no hydrazine remaining in the liquid propulsion system, and only the case of the spent OIM remains. Also the adapter between the spacecraft and upper stage and the associated separation fitting are absent from the spacecraft, having

## MASS ALLOCATIONS - EOL (KG)

| | CLIMATOLOGY | | AERONOMY | |
|---|---|---|---|---|
| | SPINNING | DESPUN | SPINNING | DESPUN |
| ELECTRICAL | 134.4 | 8.0 | 130.0 | 8.0 |
| COMMAND AND DATA HANDLING | 29.1 | 6.0 | 29.1 | 6.0 |
| THERMAL | 12.0 | 8.0 | 12.0 | 8.0 |
| COMMUNICATIONS | -- | 32.8 | -- | 32.8 |
| ATTITUDE CONTROL | 16.0 | 12.8 | 16.0 | 12.8 |
| PROPULSION | 25.6 | -- | 25.6 | -- |
| STRUCTURE AND MECHANISMS | 112.3 | 37.7 | 123.5 | 28.7 |
| INSTRUMENTS (EXCL BOOMS) | -- | 37.0 | 13.0 | 38.5 |
| STAR 37 MOTOR CASE | 62.6 | -- | 58.7 | -- |
| TOTALS | 392.0 | 142.3 | 407.9 | 134.8 |
| TOTAL SPACECRAFT | 534.3 | | 542.7 | |

Table 6.3-1

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remained with the upper stage after separation. The sequenced mass, considering propellant increments, is treated in Section 6.3.3.

Note that experiment booms which will be furnished by the spacecraft have been included in structure and mechanisms. For this reason we have deleted the 7 kg mass of the magnetometer boom from the Aeronomy instrument weights given in PM-2001, Table 5.10 (a), reducing the mass assigned to instruments from 58.5 to 51.5 kg. Structures and mechanisms includes two 5.1-kg booms, one for the magnetometer and one for the electron temperature probe, for the Aeronomy spacecraft.

### 6.3.2 Mass Properties

Figure 6.3-1 summarizes the mass properties of the spacecraft systems for the Climatology and Aeronomy missions. These properties include center of mass location, spin and transverse moments of inertia, and moment-of-inertia ratios.

The significant conditions are just before MOI, and at the end of life. We have picked worst cases in several regards, as follows:

- As boom deployment is generally stabilizing, we assume the booms remain stowed for all conditions
- As propellant in the $N_2H_4$ tanks is a stabilizing effect, we assume the minimum amount of propellant in the tanks consistent with the mission phase. For pre- or post-MOI, this amount is 88 kg (Climatology) or 142 kg (Aeronomy). At the end of the mission (EOL) it is zero.

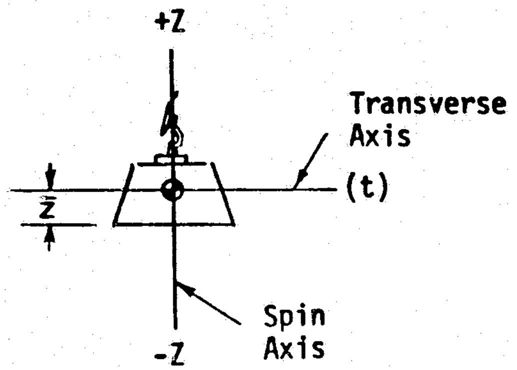
Pre-MOI is a designing condition for the all-spinning spacecraft, because the firing of the OIM is seen to improve moment-of-inertia ratio. For the condition with the despun section despun, the most adverse parameter is $I_t/I_{z(s)}$, where $I_t$ is the transverse moment-of-inertia, and $I_{z(s)}$ is the spin moment-of-inertia of just the spinning section. Here EOL conditions give the smallest margin in the moment-of-inertia ratio.

Pre-MOI, .909 is the most adverse (highest) ratio, $I_t/I_{z(tot)}$ for the baseline spacecraft. This provides satisfactory margin.

At EOL ratios of 0.934, 0.961, 0.946 are seen for $I_t/I_{z(s)}$. These are higher than desirable. The highest ratio corresponds to the Climatology spacecraft, payload Option 2. This Option puts the greatest science payload on the despun platform, 20 kg more than the baseline.

## MASS PROPERTIES

| CONDITION | MASS | C.M. ($\bar{Z}$) | SPIN M.O.I. $I_Z$ (KG $M^2$) SPINNING SECTION | DESPUN SECTION | TOTAL SC | TRANSVERSE M.O.I. $I_T$ (KG $M^2$) | M.O.I. RATIOS $\frac{I_T}{I_Z (S)}$ | $\frac{I_T}{I_Z (TOT)}$ |
|---|---|---|---|---|---|---|---|---|
| CLIMATOLOGY (BASELINE PAYLOAD) | | | | | | | | |
| PRE-MOI | 1349 | 1.25 | N/A | N/A | 954 | 867 | N/A | 0.909 |
| EOL | 527 | 1.54 | 673 | 111 | 783 | 628 | 0.934 | 0.802 |
| CLIMATOLOGY (PAYLOAD OPTION 2) | | | | | | | | |
| EOL | 547 | 1.56 | 673 | 127 | 799 | 646 | 0.961 | 0.808 |
| AERONOMY | | | | | | | | |
| PRE-MOI | 1092 | 1.31 | N/A | N/A | 1066 | 909 | N/A | 0.852 |
| POST-MOI | 674 | 1.51 | 930 | 107 | 1038 | 799 | 0.858 | 0.770 |
| EOL | 532 | 1.48 | 745 | 107 | 852 | 705 | 0.946 | 0.827 |




1. BOOMS ARE STOWED, ALL CONDITIONS (WORST CASE)
2. PRE- AND POST-MOI. PROPELLANT TANKS HAVE MINIMUM $N_2H_4$ FOR REMAINDER OF MISSION: CLIMATOLOGY 88 KG; AERONOMY 142 KG
3. EOL. PROPELLANT TANKS ARE EMPTY

Figure 6.3-1

We have determined that increase in $I_t/I_{z(s)}$ from 0.934 to 0.961 caused by the additional 20 kg on the despun platform could be canceled by adding 10 kg ballast at the perimeter of the equipment shelf. A corollary is that this moment-of-inertia ratio can be decreased by 0.027 by each 10 kg of ballast added.

With the large mass margins associated with this spacecraft/launch vehicle combination, the addition of ballast in this manner may be the most desirable way to improve stability margin. On the other hand, higher ratios, even exceeding 1.0, can be tolerated if adequate nutation damping is added to the despun section. This would be easy to implement particularly in the Climatology spacecraft, where the gamma-ray spectrometer boom can be easily arranged to drive such a damper.

Slight discrepancies in the total mass between Table 6.3-1 and Figure 6.3-1 occur. This is because the mass properties calculations were performed before the mass estimates of the subsystems were finalized.

### 6.3.3 Propulsion, $\Delta V$

Table 6.3-2 shows the velocity increment requirements for the two missions, for a 1988 launch, arranged chronologically from bottom to top.

Because both missions follow the same earth-Mars trajectory, the injection $\Delta V$ at earth (from low earth orbit) is the same, 3753 m/s. Mars orbit insertion (MOI) requirements differ because it takes more $\Delta V$ to bring the Climatology spacecraft into its 300 km circular orbit than to install the Aeronomy spacecraft in its elliptical orbit.

After insertion into Mars orbit, propellant requirements for the two missions differ. The inclination change maneuver, after the initial drift period, dominates the Climatology $\Delta V$'s, while drag compensation takes most of the Aeronomy's $\Delta V$ budget -- because of its low 150 km periapsis altitude. In fact, the calculated (worst case) $\Delta V$ to compensate for drag due to the estimated atmospheric density near 150 km altitude (430 m/s) exceeds the propellant capacity of the spacecraft design so this requirement is arbitrarily limited to 365 m/s.

The final boost from the nominal orbit to one with a long enough life to satisfy planetary protection requirements takes less $\Delta V$ for

## MISSION $\Delta$V REQUIREMENTS (M/S)

| | CLIMATOLOGY | AERONOMY |
|---|---|---|
| FINAL BOOST | 100 | 34 |
| DRAG COMPENSATION | 9 | 365 (430*) |
| ATTITUDE AND ORBITAL CORR'S | 50 | 50 |
| ORBIT FINALIZATION | 223 | 25 |
| TOTAL (ORBITAL MANEUVERS) | 382 | 474 |
| MOI REQUIREMENTS | 2075 | 1206 |
| TCM'S | 100 | 100 |
| INJECTION VELOCITY | 3753 | 3753 |

* 430 M/S DRAG COMPENSATION REQUIREMENT IS OUTSIDE PROPELLANT STORAGE LIMITATION OF BASELINE TANKS

Table 6.3-2

Aeronomy because it can be performed more efficiently at the apoapsis of the elliptical orbit.

Based on the $\Delta V$ requirements of Table 6.3-2 and the EOL mass of Table 6.3-1, a mass history from launch throughout the mission is constructed and shown in Table 6.3-3. This table incorporates several categories of margin. First, a ~20% margin is shown for the final spacecraft mass. Second, a 20% margin is added to all calculated masses of hydrazine propellant. Finally, the injection stage is shown capable of handling this greater spacecraft system with offloads (margins) of its solid propellant of 35 percent (Climatology) and 45 percent (Aeronomy). This verifies the assertion that there is a comparatively large mass margin for the mission. This margin can be assigned in a number of ways to reduce program cost and improve mission reliability.

The solid propellant offload for the MOI motor is not an additional margin. What it shows is the margin associated with the selection of the particular motor for MOI. If for some reason (greater spacecraft mass, or higher $V_{\infty}$ approaching Mars) this offload percentage shrank to zero or below, it would merely indicate that the next larger Star 37 motor should be substituted. But the validity of the overall table is not affected.

### 6.3.4 Power, Power Budget

The power requirements (payload and spacecraft subsystems) of the two missions are shown in Tables 6.3-4 and 6.3-5. These tables are based on several assumptions:

- Thermal control uses electric replacement heaters when experiments are turned off, particularly during interplanetary cruise
- In orbital operation, one TWTA and one tape recorder are on at all times
- Power to charge the batteries for use during eclipse time is added to the other power requirements during sunlight operation. This calculation is based on the maximum possible eclipse time for each mission.

The results which drive the system design are the full sun orbital operation: 522 W (Climatology) and 401 W (Aeronomy).

## PROPULSIVE BUDGETS AND MASS HISTORY (KG)

| | CLIMATOLOGY | AERONOMY |
|---|---|---|
| EOL | 534.3 | 542.7 |
| MARGIN | 109.5 (20%) | 113.2 (21%) |
| EOL ALLOCATION | 643.8 | 655.9 |
| MONOPROPELLANT (ORBITAL MANEUVERS) | 110.9 | 149.7 |
| 20% PROPELLANT MARGIN | 22.2 | 29.9 |
| POST-MOI | 776.9 | 835.5 |
| MOI PROPELLANT | 826.2 ( 6% OFFLOAD) | 445.2 (21% OFFLOAD) |
| PRE-MOI | 1603.1 | 1280.7 |
| MONOPROPELLANT (MIDCOURSE CORR'S) | 68.1 | 54.4 |
| 20% PROPELLANT MARGIN | 13.6 | 10.9 |
| POST SRM-1 LAUNCH | 1684.8 | 1346.0 |
| SRM-1 CASE | 638.0 | 638.0 |
| SRM-1 FUEL | 6250.6 (35% OFFLOAD) | 5340.7 (45% OFFLOAD) |
| STS DEPLOYMENT MASS | 8573.4 | 7324.7 |

Table 6.3-3

## POWER REQUIREMENTS FOR CLIMATOLOGY MISSION

(WATTS)

| Subsystem | Cruise Mode | Orbit Insertion | Orbital Operation | |
|---|---|---|---|---|
| | | | Full Sun | Eclipse |
| | | | Science On Transmission | Night Science On |
| Propulsion | < 1.0 | < 1.0 | < 1.0 | 1.0 |
| Science | 0 | 0 | 59 | 55 |
| Attitude Control | 6 | 6 | 10 | 10 |
| Command and Data Handling | 24 | 24 | 44 | 44 |
| Communication | 92 | 92 | 92 | 92 |
| Thermal Control | 121 | 121 | 46 | 34 |
| Electrical Power Conversion Loss | 2 | 2 | 12 | 12 |
| Electrical Power Distribution Loss | 4 | 4 | 28 | 8 |
| Total | 250 | 250 | 292 | 256 |
| Battery Charge | 30 | 30 | 230 | --- |
| Total | 280 | 280 | 522 | 256 |

Table 6.3-4

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## POWER REQUIREMENTS FOR AERONOMY MISSION

(WATTS)

| Subsystem | Cruise Mode | Orbit Insertion | Orbital Operation: Full Sun — Science On Transmission | Orbital Operation: Night Science On |
|---|---|---|---|---|
| Propulsion | 1.0 | 1.0 | 1.0 | 1.0 |
| Science | 0 | 0 | 47 | 47 |
| Attitude Control | 6 | 6 | 10 | 10 |
| Command and Data Handling | 24 | 24 | 44 | 44 |
| Communication | 92 | 92 | 92 | 92 |
| Thermal Control | 109 | 109 | 46 | 34 |
| Electrical Power Conversion Loss | 2 | 2 | 8 | 8 |
| Electrical Power Distribution Loss | 4 | 4 | 28 | 8 |
| Total | 238 | 238 | 276 | 244 |
| Battery Charge | 30 | 30 | 125 | --- |
| Total | 268 | 268 | 401 | 244 |

Table 6.3-5

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As developed in Section 7.2, the selected solar array for Climatology can produce 546 W, giving a 22 W margin, with cells occupying the full 82-inch height of the truncated cone. For Aeronomy the figure is 420 W, giving a 19 W margin, with cells occupying only 86% of the available substrate area.

These figures account for greater areal efficiency (0.346) for the Climatology Mission because the sun aspect angle (SAA) is restricted to the range 45 to 68 degrees after the drift period.* (See Figure 6.1-4.) The areal efficiency for the Aeronomy Mission (0.299) reflects the non-sun-synchronism of this orbit and the range of SAA's from 0 to 90 degrees.

Recognize that these power production figures presuppose three worst cases all happening simultaneously, which, in actuality, can't happen:

- Maximum sun-Mars distance
- Maximum eclipse time
- EOL-degraded solar cells

As a result, there is not likely to be a power pinch as severe as the 22 and 19 watt margins might imply. And if there is a pinch, it will be for a limited time, until conditions recede from the worst case.

For this reason, we identify tape recorder operational Model B (2 tape recorders on simultaneously during the tracking period) as a desirable mode, and which can be implemented if adequate power margin exists and if both recorders are operating well; otherwise operation must revert to Model A, which requires only one tape recorder.

### 6.3.5 Data

The data handling performance of the spacecraft system is tied closely to the data acquisition requirements of the scientific instruments.

Three other factors enter:

---

* During the drift period, the areal efficiency is less, but this is more than offset by the fact that the sun-Mars distance during this time is considerably less than its maximum value of 1.67 AU.

- The operation of tape recorders per Model B (Section 6.2.5)
- Use of the "bit-substitution method" for interleaving real-time engineering data with data played back from the tape recorder
- An "overhead" fraction of one-eighth

The bit substitution method is described in Figure 6.3-2. As data are recorded, space is reserved for the later insertion of real-time data when the tape recorder is played back. This reservation is by means of "fill" bits recorded in those positions on the recorder. Fill bits can be zeroes or ones in any combination.

When the tape recorder is played back, the fill bits are discarded. They are replaced by the real-time data it is desired to interleave. The fill bits and the real-time data occupy the same location in each data frame.

Figure 6.3-3 gives an example to show how this works in the Mars Orbiter Mission. Because operating Model B is selected, all science data transmitted to the earth come from the tape recorder, where they were recorded as acquired earlier, during the preceding 24 (or 32) hours. Thus the only real-time data to be interleaved are engineering (subsystems) and housekeeping (instrument status) data. We have selected an 1152-bit frame consisting of 1024 bits of science data and 128 bits of overhead. The overhead is divided into 88 bits (synch, identification, time tags, engineering and housekeeping) which are recorded with the science data, and 40 fill bits which are replaced later by real-time data (time tag, engineering, and housekeeping).

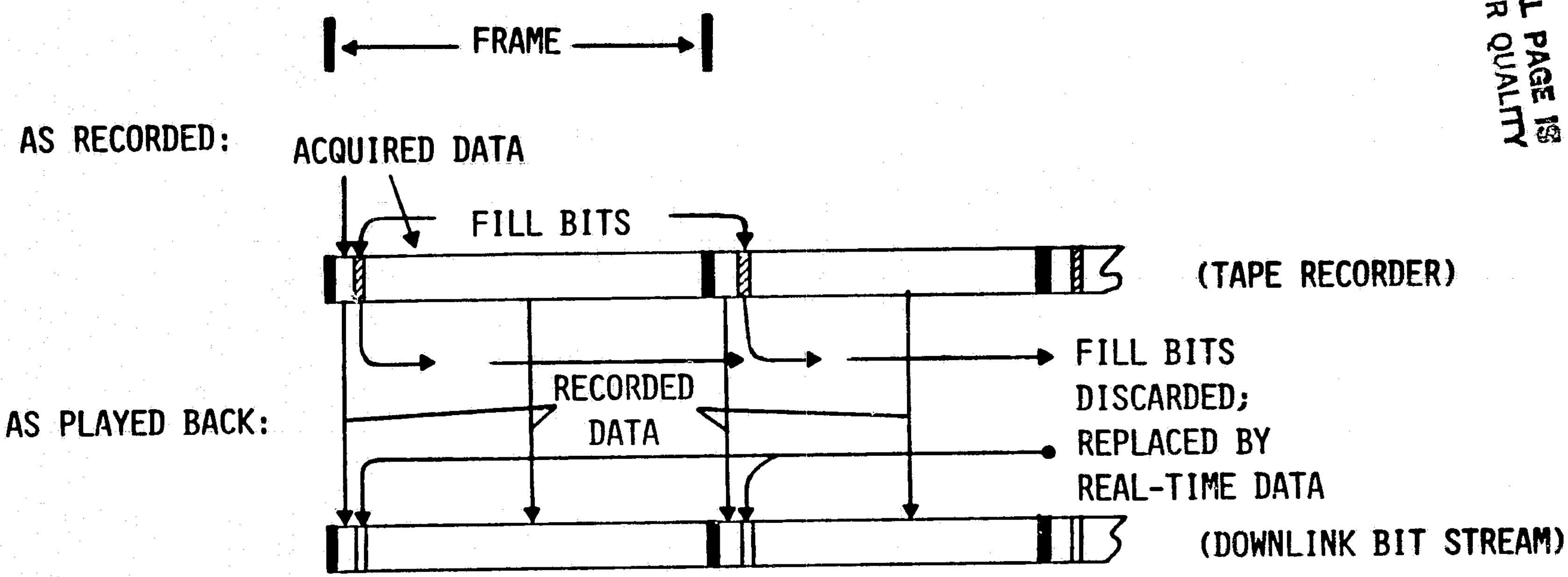
Because up to 32 hours recorded data can be played back in ~5 hours, the effective sampling rate of real-time engineering data is ~6 times as fast as the corresponding recorded data.

Returning to the required science data acquisition rates, these are summarized for the Climatology Mission (and its these payload options) in Table 6.3-6 and the Aeronomy in 6.3-7.

All these missions permit different science data acquisition rates at different times in the orbit. For three ($C, C_1, C_2$) we have not exercised this option, because day and night rates do not differ enough to warrant

## INTERLEAVING REAL-TIME DATA

(BIT SUBSTITUTION METHOD)




ADVANTAGES:

- MINIMUM DATA BUFFERS REQUIRED
- PLAYED-BACK AND REAL-TIME DATA TRANSMITTED IN BIT-FOR-BIT SYNCHRONISM
- TIME TAGS CAN BE INSERTED INDEPENDENTLY FOR RECORDED AND REAL-TIME DATA

Figure 6 3-2

## TELEMETRY SCIENCE DATA FRAMES

1152 BIT FRAMES:

| | WHEN RECORDED | WHEN PLAYED BACK |
|---|---|---|
| NON SCIENCE | | |
| SYNCH | 24 | |
| ID | 8 | |
| TIME TAG | 32 | 16 |
| ENG'G (SUBCOM) | 16 | 16 |
| HOUSEKEEPING (SUBCOM) | 8 | 8 |
| FILL* | 40 | |
| TOTAL NON SCIENCE | 128 | |
| SCIENCE | 1024 | |
| TOTAL FRAME | 1152 | |

$\frac{\text{TOTAL}}{\text{SCIENCE}} = \frac{9}{8}$ ; "OVERHEAD" $= \frac{1}{8}$

* FILL BITS ARE REPLACED BY REAL TIME DATA AS INDICATED, WHEN PLAYED BACK FOR DOWNLINK TRANSMISSION.

Figure 6.3-3

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## SCIENCE DATA ACQUISITION RATES (B/S)

CLIMATOLOGY MISSION

| | BASELINE | OPTION 1 | OPTION 2 | OPTION 3 | |
|---|---|---|---|---|---|
| PMR | 140 | 140 | 140 | 140 | |
| FIS | (120) | (120) | (120) | | (DAY ONLY) |
| GRS | 1024 | 1024 | 1024 | 1024 | |
| $UVO_3$ | | 64 | | | |
| UVHP | | 8 | | | |
| RA | | 100 | | 100 | |
| FPI | | | (256) | | (DAY ONLY) |
| MSM | | | | (1500 -12000) | (DAY ONLY; 1000 B/S AV. OVER 24-HRS.) |
| TOTALS: | | | | | |
| DAY | 1284 | 1456 | 1540 | 2764-13,264 | |
| NIGHT | 1164 | 1336 | 1164 | 1264 | |
| AVERAGE OVER 24-HRS. | 1224 | 1396 | 1352 | 2264 | |

Table 6.3-6

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## SCIENCE DATA ACQUISITION RATES (B/S)

### AERONOMY MISSION

| ORBITAL PHASE | IONOSPHERE | IONOSHEATH | APOAPSIS | |
|---|---|---|---|---|
| HOURS PER ORBIT | 0.500 | 1.500 | 4.689 | |
| INSTRUMENT | | | | |
| NMS/CWT | 256 | 0 | 0 | |
| TIMS | 256 | 128 | 0 | |
| ETP | 256 | 128 | 64 | |
| RPA/DM | 512 | 128 | 64 | |
| MAG | 128 | 128 | 128 | |
| EFD | 128 | 128 | 128 | |
| SWPA | 128 | 128 | 128 | |
| UVS | 128 | 128 | 0 | |
| FPI | 256 | 128 | 0 | |
| TOTAL BIT RATE (B/S) | 2048 | 1024 | 512 | 741.6 (AVG) |
| TOTAL BITS PER ORBIT (MB) | 3.686 | 5.530 | 8.643 | 17.859 |
| TOTAL BITS PER 32 HOURS (AVG)(MB) | | | | 85.44 |
| (MAX)(MB) | 18.43 | 27.65 | 40.55 | 86.63 |

Table 6.3-7

an additional tape recorder speed. For the third Climatology option ($C_3$) we do implement different record rates, because of the variety and range of acquisition rates associated with the multi-spectral mapper. For the Aeronomy Mission, different record rates are implemented because it would be very wasteful to record at the periapsis pulse rate over the whole orbit.

Based on the above, we can determine the requirements for tape recorder capacity in Table 6.3-8 (and compare the requirements with the selected recorder) and the requirements for record and playback rates and downlink transmission rates in Table 6.3-9.* The selected playback and downlink data rates have been rounded off upward:

- to get rates more simply related to each other
- to use the link capacity of the selected RF system, and
- to permit data return to occupy somewhat less than the full tracking period (unocculted portion)

Downlink rates are also selected for exclusively engineering data, which could be in a general engineering format or in a format for transmitting the contents of the on-board command store. These provisions would probably be used most for real-time data, but the highest of these rates corresponds to the lowest tape recorder playback rate, and could be used for recorded engineering data. The lower engineering data rates are for use when the link will not support the higher rates, particularly when the HGA cannot be used.

The "required" selected playback and downlink rates will return in one eight-hour shift all data collected in a 32-hour period, accounting for maximum earth occultations and receiver lock-up times. However, in the normal manner of scheduling ground stations of the DSN the normal non-tracking period would be 16 hours, requiring only 24 hours of data to be returned in one shift. The downlink rates, "most used: normal"

* Because of the bit substitution method of interleaving, the playback and downlink (before coding) rates are the same.

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## TAPE RECORDER CAPACITY

| MISSION | BASIS (32 HRS) | REQUIREMENT (MB) | CAPACITY (MB) |
|---|---|---|---|
| CLIMATOLOGY BASELINE | DAY | 166.4 | 325 |
| C1 | DAY | 188.7 | |
| C2 | DAY | 199.6 | |
| C3 | AVG | 293.4 | |
| AERONOMY | ~AVG | 97.5 | |

Table 6.3-8

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# TAPE RECORDER AND DATA TRANSMISSION RATES (B/S)

| | MISSION A | C | $C_1$ | $C_2$ | $C_3$ | |
|---|---|---|---|---|---|---|
| **TAPE RECORDER RECORD RATES** | | DAY RATE | DAY RATE | DAY RATE | | |
| ($\frac{9}{8}$ x SCIENCE ACQ. RATE) | 2304 (PERI) | 1444.5 | 1638 | 1732.5 | 14922 | D4 |
| | 1152 | | | | 8172 | D3 |
| | 576 (APO) | | | | 4797 | D2 |
| | | | | | 3109.5 | D1 |
| | | | | | 1422 | N |
| **TAPE RECORDER PLAYBACK RATES** | | | | | | |
| REQUIRED: MAX (32 HRS) | 5414 | 9989 | 11327 | 11980 | 17612 | |
| MOST USED: NORMAL (24 HRS) | 4147 | 7492 | 8495 | 8985 | 13210 | |
| SELECTED: | | | | | | DOWN-LINK DATA RATES |
| EXPLOITS HIGHER LINK CAP. | 12000 | 24000 | 24000 | 24000 | 36000 | |
| REQUIRED: MAX | 6000 | 12000 | 12000 | 12000 | 18000 | |
| MOST USED: NORMAL | 4500 | 9000 | 9000 | 9000 | 13500 | |
| TOLERATES LOWER LINK CAP. | 2000 | 4000 | 4000 | 4000 | 6000 | |
| **ADDITIONAL ENGINEERING FORMAT RATES (REAL TIME ONLY)** | 2000 | 4000 | 4000 | 4000 | 6000 | |
| A. ENGINEERING, OR | 250 | 250 | 250 | 250 | 250 | |
| B. COMMAND STORE DUMP | 64 | 64 | 64 | 64 | 64 | |
| | 16 | 16 | 16 | 16 | 16 | |

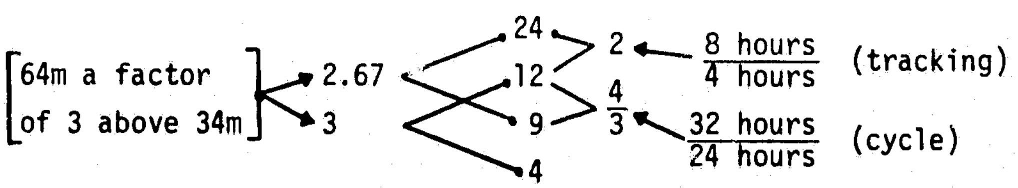
Table 6.3-9

reflect this reduced load. They are implemented to cater to the most probable conditions, to increase the utility of the link.

In addition, rates are implemented at twice and one-third the "required: max" rate. The higher one permits exploiting a higher link capacity which might exist. Examples: a 64-meter station is available when a 34-meter antenna would suffice; or communication ranges are small enough that even a 34-meter antenna can support the higher rate. This would permit a four-hour station assignment rather than the normal eight hours.

The lower rate (one-third) applies when conditions are reversed, when it is necessary to tolerate a lower link capacity. Suppose, near maximum range, the 64-meter antenna could do the job, but it isn't available. Rather than get no data, accept a lower rate which can be supported by the 34-meter antenna. This antenna has a gain 5 dB less than the 64-meter, so it can support one-third the data rate, and this is how the lowest playback rate is selected.

The ratios of these four playback rates are 24:12:9:4, with the rationale summarized by this table:




### 6.3.6 Communications

Uplink transmission should support command data rates of 7.8125 b/s (emergency) and 125 b/s (normal). Figure 6.3-4 shows the capacity of the selected uplink channels, via either the spacecraft's HGA or omni. The only situation where the emergency rate must be invoked is via the omni beyond 1.9 AU, which is not a normal condition. Capability via the omni would also be less than that shown if the earth aspect angle exceeded 90 degrees (with the HGA pointed in the XY plane). Gain would drop below -3 dBi, the 90° value, and the link would disappear when the omni is shielded by the spacecraft at FAA $\approx$ 145°. If the HGA is despun

## UPLINK DATA RATES

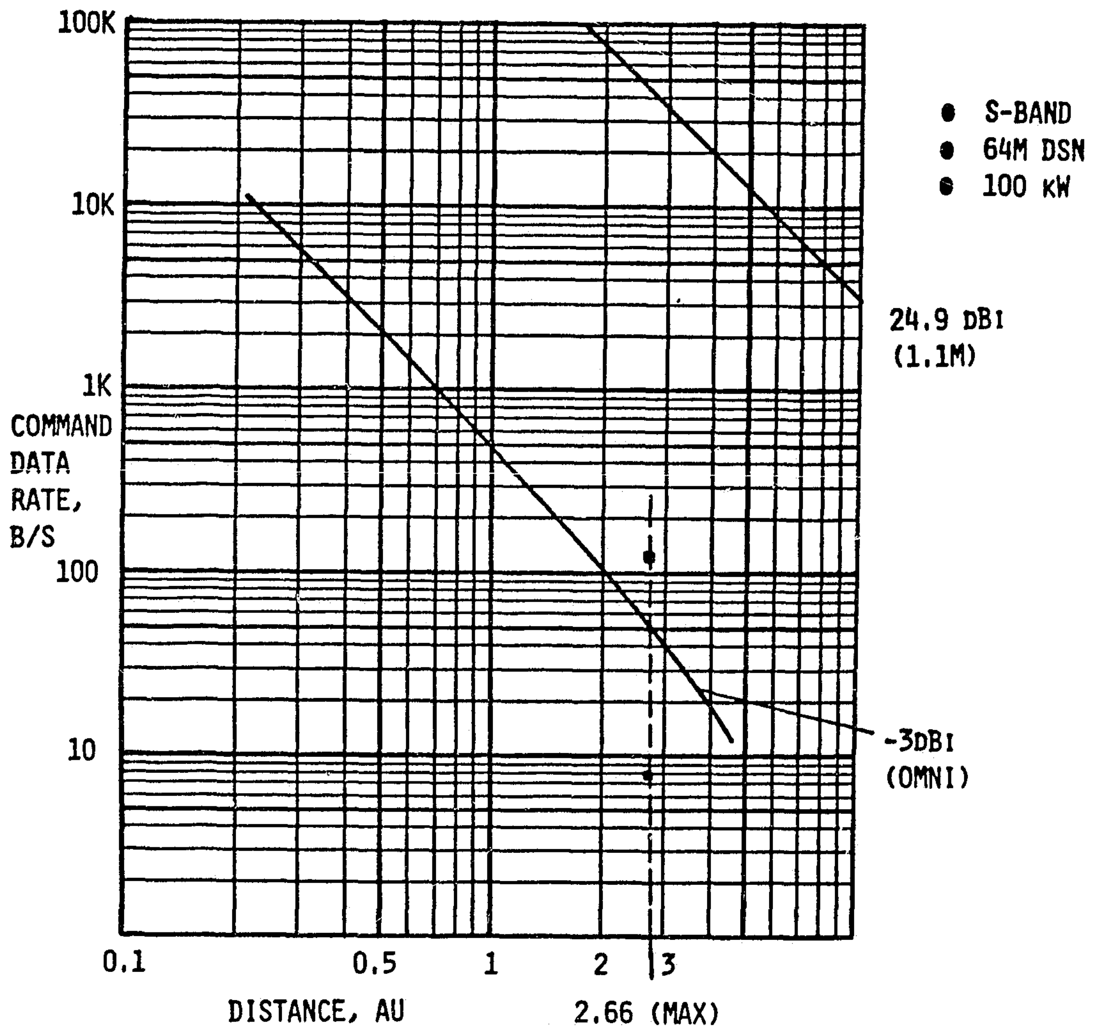



Figure 6.3-4

and pointed at the earth (but for some reason not receiving) then the omni will have a -3 dBi gain and command access can be established for recovery procedures.

Figure 6.3-5 shows downlink data capability via the HGA and via a MGA (horn or bicone) operating at a gain of +4.5 dBi. Critical requirements are also indicated. Conclusions: The selected HGA (1.12 m dish) satisfies the Aeronomy requirement (by over 3 dB), and it satisfied the Climatology baseline requirement as well as payload Options 1 and 2. To satisfy the data rate required by Option 3, a larger antenna is needed. The performance of a 1.5-meter HGA (up 2.5 dB) is shown; it easily satisfies the higher requirement. 1.5 meters is about the largest dish compatible with the physical characteristics of the selected spacecraft configuration.

Before the HGA is despun, downlink communications must be via one of the X-band MGA's on board. With a spacecraft antenna gain of +4.5 dBi, the performance barely meets the 16 b/s requirement at MOI, 1.23 AU from the earth. The MGA complement would provide this gain through the bicone at EAA = 90° (interplanetary cruise attitude) or through the horn at EAA = 38° (MOI attitude).

The capabilities shown in Figures 6.3-4 and 6.3-5 assume a $3\sigma$ adverse performance, and are supported by RF power budget tables in Section 7.3.

### 6.3.6 Pointing Accuracy

Instrument pointing accuracy in pitch (i.e., rotation about the spin axis) is achieved by controlling the phase of the despun platform. Relative to celestial directions are measured by the spinning star sensor, the required accuracy (0.08 degrees, $1\sigma$, for $C_1$ and $C_3$; 0.5 to 1 degree, $1\sigma$ for the baseline missions) is easily met. See Table 6.3-10. However, where instruments are to be pointed relative to a planet-oriented direction (nadir, limb, ram, for example), then the orbit determination accuracy enters the picture. An orbit determination accuracy analysis has not been made, so instrument pointing accuracy relative to the planet can not be validated.

An order-of-magnitude estimate suggests the orbit determination will support the 0.5 degree pointing requirement, but perhaps not the 0.08 degree.

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DOWNLINK DATA RATES

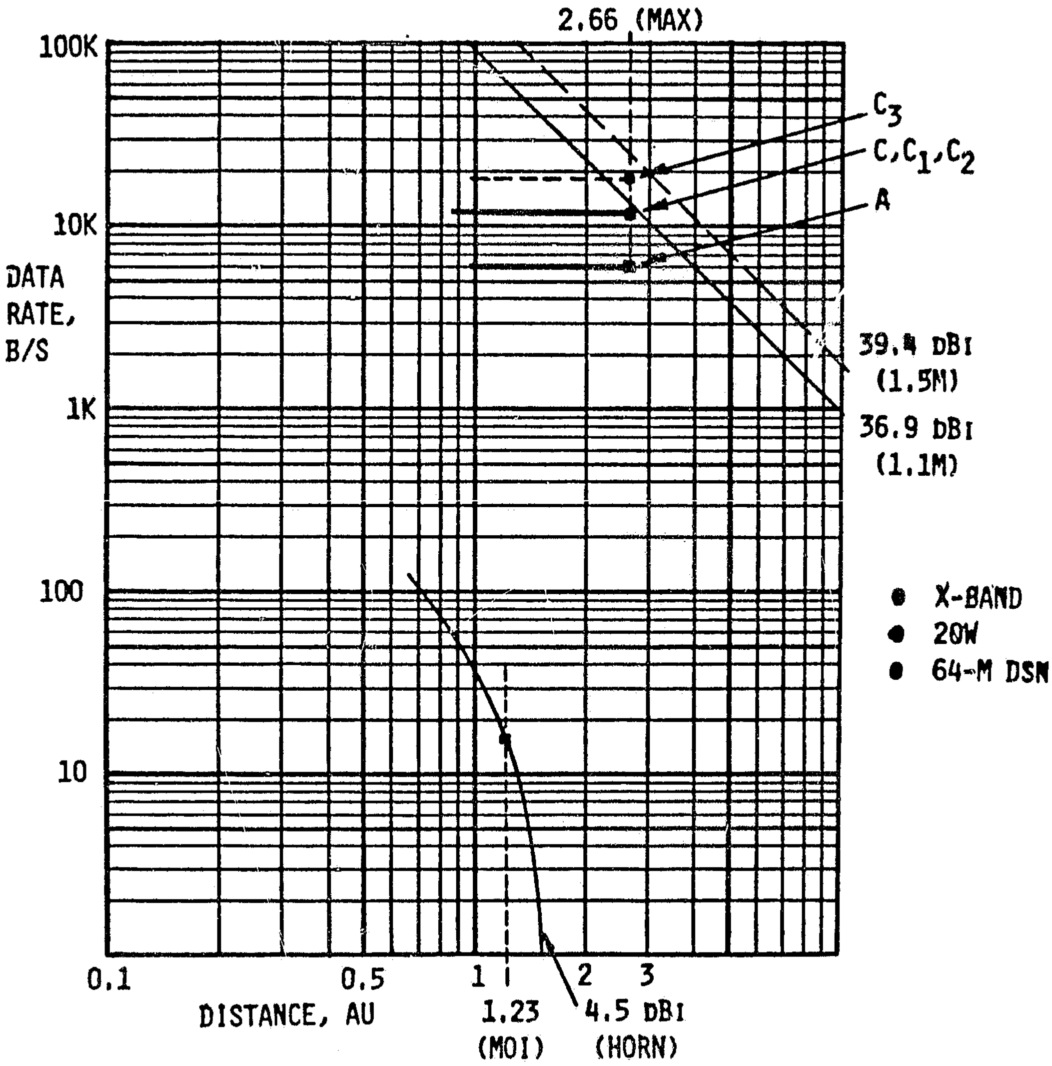


Figure 6.3-5

## INSTRUMENT P NTING ($1\sigma$)

REQUIREMENT - (BASED ON MOST SEVERE POINTING REQUIREMENT OF INSTRUMENT COMPLEMENT) (DEG)

| | ROLL | | PITCH | | YAW | |
|---|---|---|---|---|---|---|
| | C | K | C | K | C | K |
| BASELINE CLIMATOLOGY (PMR) | 1 | 0.2 | 1 | 0.2 | 1 | 0.2 |
| C1 (FIS + UVO3) | 1 | 0.2 | 0.08 | 0.08 | 1 | 0.2 |
| C2 (FPI) | 0.5 | 0.1 | 0.5 | 0.1 | 0.5 | 0.1 |
| C3 (MSM) | 0.08 | TBD | 0.08 | TBD | 0.08 | TBD |
| AERONOMY | 0.5 | 0.1 | 0.5 | 0.1 | 0.5 | 0.1 |

POINTING ERRORS

| SOURCE | ERROR | |
|---|---|---|
| | ROLL, YAW | PITCH |
| STAR SCANNER | $0.017^{\circ}$ | |
| STAR SCANNER TIMING | | $0.017^{\circ}$ |
| DESPUN PLATFORM ALIGNMENT AND TIMING | | $0.01^{\circ}$ |
| ALL OTHER ERRORS (ALLOWABLE) | $0.078^{\circ}$ | $0.077^{\circ}$ |
| TIGHTEST REQUIREMENT | $0.08^{\circ}$ | $0.08^{\circ}$ |

Table 6.3-10

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If analysis shows that orbit determination is inadequate to meet the required pointing accuracy, then horizon sensors can be considered, to provide direct sensing of limb and nadir directions.

Pointing in roll and yaw is achieved by first measuring, then controlling the orientation of the spin axis. With respect to a celestial direction this is done with the spinning star sensor and the fine precession thrusters, easily meeting the 0.08 degree requirement (the tightest one). However, if the direction of this axis is to be controlled relative to the orbit plane, again orbit determination accuracy enters the picture. This time the question is what the orbit plane orientation, whereas before it was where is the spacecraft in the orbit. Again, the analysis has not been made, but it is very likely that accuracy for orbit plane orientation is superior, because the perturbation rate is much slower.

For pointing the high gain antenna at the earth, it is pointing in celestial coordinates which must be controlled. Since the basic reference is the star sensor, orbit determination accuracy is not a factor. Table 6.3-11 shows that the dominant contributor is the azimuth or elevation gimbal angle error "($1\sigma$)" due to resolver linearity or resolution, or antenna mast distortion. The 0.25° ($1\sigma$) requirement is easily met.

## HIGH GAIN ANTENNA POINTING ACCURACY

REQUIREMENT: $\pm 0.75^{0}$ ($3\sigma$)
$\pm 0.25^{0}$ ($1\sigma$)

| SOURCE | ERROR |
|---|---|
| GIMBAL ANGLE ERROR (AZ. OR EL.)<br>RESOLVER LINEARITY, RESOLUTION, ETC. | $0.15^{0}$ |
| DESPUN PLATFORM ORIENTATION<br>(ROLL, YAW OR PITCH) | $0.08^{0}$ |
| ANTENNA POINTING ERROR ($1\sigma$)<br>(AZIMUTH OR ELEVATION) | $0.17^{0}$ |

Table 6.3-11

# 7. SUBSYSTEM DESIGNS (BASELINE)

## 7.1 STRUCTURAL SUBSYSTEM AND MECHANISMS

### 7.1.1 Structural Subsystem Functions

The spacecraft structural subsystem performs several important functions in addition to providing housing for the scientific payload and other spacecraft subsystems. Among these are:

a) It provides for static and dynamic loads and load paths throughout the various phases of the mission. In addition, the structure must be tolerant of all the environments encountered over the course of the mission without causing deleterious effects on other spacecraft subsystems and functions.

b) It accommodates and influences mass distributions and moment of inertia effects. The structural subsystem is of special importance in this regard in the case of a spin stabilized spacecraft.

c) It may provide surface area for primary source of spacecraft electrical power when mission is close enough to the sun that solar cells are suitable for the generation of power. In this case, the structural subsystem and resulting spacecraft size (surface area) is determined to a large extent by the amount of electrical power necessary to support mission requirements.

### 7.1.2 Structural Subsystem Requirements

The structural requirements for the Mars Orbiter are shown in Figure 7.1-1. The requirement for maximum longitudinal acceleration (ultimate) is based on thrust levels of the Star 37XF and 37F orbit insertion motors and Mars orbit insertion weight budgets for the orbiter. The requirement also includes a 1.2 x safety factor. Other structural requirements were obtained from the source references indicated in the figure.

In addition to the structural requirements indicated, the spacecraft configuration is required to provide an effective solar cell surface area of approximately 80 ft.$^2$ while remaining within the dynamic and thermal envelope of the STS cargo bay. These requirements are accomplished utilizing the truncated cone configuration discussed in Section 6.

### 7.1.3 Structural Subsystem Options

The spacecraft configuration was selected primarily as a result of overall scientific and systems considerations as discussed in Section 6. Having selected the configuration, structural options then included the following considerations.

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STRUCTURE REQUIREMENTS

| Parameter | Requirement | Source |
|---|---|---|
| Weight | 175 kg | Derived |
| Stiffness | 35 Hz Longitudinal<br>15 Hz Transverse | SOW PM-2001/2000 |
| Maximum Accelerations | 15.4g Longitudinal (Ultimate) | OIM Burn |
| | 6.0g Transverse (Ultimate) | Derived-STS Launch |
| Other | | |
| Safety | STS | STS-1123 |
| Integration | Scientific Payload | SOW PM-2001/2000 |
| | STS Launch Vehicle | JSC STS Volume XIV |
| Thermal, Electrical, RF Grounding/Shielding | Performance | PM-2001/2000 |

Figure 7.1-1

1) Accommodation of scientific experiments

2) Location, orientation and accommodation of other subsystems necessary to perform intended missions. (Propulsion tanks, thrusters, antennas, etc.)

3) Provisions for structural load paths and spacecraft rigidity within an allocated weight requirement. This includes the selection of materials and methods of support and attachment

4) Boom, mechanism and appendages options and selection

In each of these areas, system aspects remained a primary consideration since structural options are closely intertwined with the spacecraft configuration and vice-versa.

Boom and appendage options and trades for scientific instruments are shown in Figure 7.1-2.

Experiment Boom Comparisons: The Electrical Field Detector (EFD), is a dipole 2 meters in length and weight in the order of 0.5 kg. The configuration may therefore be made into one assembly that is rotated 180° for deployment from its stowed position. It is attached to the aft of the spinning section of the spacecraft. The MAG and ETP are single booms having to deploy weights of 2 kg -5 meters, and 5 kg -4 meters respectively. A bi-fold tubular 5.0 cm diameter boom, (one joint only), with an over center locking device may be stowed and deployed from the aft of the spinning section also.

The GRS weighs 14 kg and must be deployed 5 meters from the despun platform. Balancing is not felt to be required for the deployed condition.

The astromast and stem deployable booms are both considered as viable candidates for the GRS experiment since it may be desirable to have a retracting capability available.

### 7.1.4 Structural Subsystem Description

Given the truncated conical configuration with the electrical requirements and STS imposed envelope constraints previously stated, spacecraft areas and volumes available are as indicated in Figure 7.1-3.

After providing for the scientific instruments on the despun platform, on booms, or on the spinning section as dictated by the particular

EXPERIMENT BOOM COMPARISONS

| TRADE / IMPACT — CANDIDATE | Method Of Deployment | Modules Of Rigidity | Compatibility: Despun Instl. | Compatibility: Spin - Instl. | Stowage Factor | Weight | Retractable Capability | Cost |
|---|---|---|---|---|---|---|---|---|
| 1. Bi-Fold (2" Section) | Torsion Release and Spring Lock | Good | Must Instl to Clear Experiments | Exc. | Exc. | 7.0 | Poor | Very Low 10k |
| 2. Astromast (3" Dia.) | Motor Driven Lanyard | Good | Exc. | Must Provide Mounting Structure | Retract Capability Exc. | 9.0 | Exc. | Very High 250k - 400k |
| 3. Stem, (Spar) (4") | Positive Motor Reel | Fair/Poor | Exc. | Must Provide Mounting Structure | Retract Capability Exc. | 7.0 | Exc. | High 90 - 150k |
| 4. Tubular/ Telescopic 2½" | $GN_2$ Pressure | Unstable (Tolerance Buildup) | Fair | Must Provide Mounting Structure | Fair | 11 | Poor | Low |
| 5. Multi-Fold (Sections) | Multi-Spring Hinge | Fair | Fair | Poor Stowage | Fair/Poor | 12 | Very Poor | Low |

Figure 7.1-2

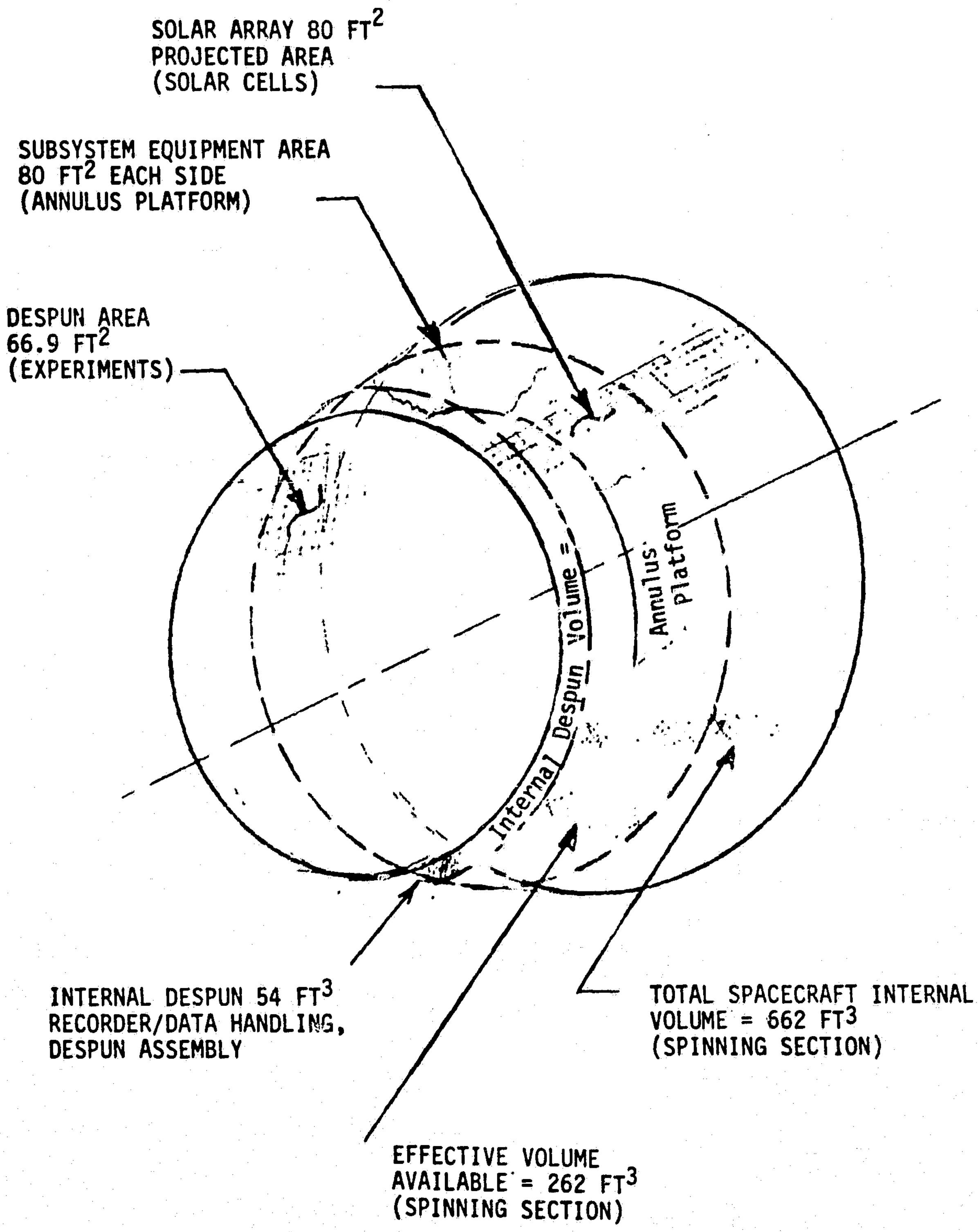
SOLAR ARRAY 80 FT$^2$
PROJECTED AREA
(SOLAR CELLS)
SUBSYSTEM EQUIPMENT AREA
80 FT$^2$ EACH SIDE
(ANNULUS PLATFORM)
DESPUN AREA
66.9 FT$^2$
(EXPERIMENTS)
Internal Despun Volume =
Annulus platform
INTERNAL DESPUN 54 FT$^3$
RECORDER/DATA HANDLING,
DESPUN ASSEMBLY
TOTAL SPACECRAFT INTERNAL
VOLUME = 662 FT$^3$
(SPINNING SECTION)
EFFECTIVE VOLUME
AVAILABLE = 262 FT$^3$
(SPINNING SECTION)

Figure 7.1-3

requirements of each instrument, subsystems were located partly in consideration of their effect on establishing the most favorable moment of inertia about the spin axis of the spacecraft. Independent of this consideration however, it was recognized that equipments would be located on both the despun platform and also on an equipment platform located on the spinning section at approximately the center of mass of the spacecraft. The spinning equipment shelf is supported by (and provides support to) the solar array substrate (at the outer diameter) and the spacecraft tubular framework (at the inner diameter). The framework struts are attached to the OIM cylindrical housing. The OIM cylinder is also attached to the solar substrate through four forward tubular struts and two attachment rings. The combination of tubular framework, and cylindrical housing, along with equipment shelf to solar substrate load paths provides a lightweight rigid structure for axial and transverse loads. (See Figure 6.1-1).

The spinning section is married to the despun section through the despun mechanism which includes the despin motor, bearings and slip rings. The despun mechanism is mounted to the spinning assembly through a (relatively) lightweight ring/cruciform assembly attached through bearings and a housing to the upper portion of the OIM cylinder.

The despun platform is attached to the spacecraft via the despun mechanism motor such that the despun platform is driven so as to remain stationary with respect to the spinning part. During Earth-Mars trajectory the spinning and despun sections are securely locked together via snubbers located around the outer periphery of the despun platform.

Mechanical and structural component descriptions and estimated weights used in the structure are shown in Figure 7.1-4.

Truss tubular framework structure is of 2 inch diameter, 0.030 inch wall thickness aluminum. The OIM cylinder is titanium with a 0.040 inch wall thickness. The despun platform and solar substrate are made up of 0.005 inch fiberglass facesheets with hex-cell aluminum alloy honeycomb core.

Comparisons of the Mars Orbiter structural design with comparable space structures previously built and tested indicates that structural natural frequencies should be above the 15 Hz transverse, 35 Hz longitudinal requirements.

STRUCTURAL AND MECHANICAL SUBSYSTEM

EQUIPMENT LIST

| Component | Qty. | Unit Wt. kg | Total Wt. kg | Unit Pwr. Watts | Dimensions cm | Heritage | | |
|---|---|---|---|---|---|---|---|---|
| | | | | | | Program | Vendor | Status |
| STRUCTURE/MECHANICAL | | | | | | | | |
| Solar Array Substrate | 1 | 35 | 35 | 0 | $\left.\begin{matrix} d_1 = 280 \\ d_2 = 432 \end{matrix}\right\} \times 208$ | DSCS II | --- | Adaptation of Existing Technology |
| Primary Structure: Rings, Struts | 1 | 31.2 | 31.2 | 0 | $\left.\begin{matrix} d_1 = 280 \\ d_2 = 432 \end{matrix}\right\} \times 208$ | DSCS II | --- | |
| Equipment Platform (Spinning) | 1 | 24 | 24 | 0 | d = 325 | DSCS II | --- | |
| Despin Assembly Support Structures | 1 | 6 | 6 | 0 | d = 280 | DSCS II | --- | |
| Miscellaneous Attach Brackets (Includes Isolators) | 50 | .06 | 3 | 0 | 1.2 x 4 | DSCS II | --- | |
| Despun Platform Assy (Cruciform & Rings) | 1 | 26 | 26 | 0 | 280 dia. | DSCS II | --- | |
| OIM Adapter | 1 | 3.8 | 3.8 | 0 | dia. = 100X 100L | DSCS II | --- | |
| Integration (Attachments & Brkts) | Gross | 6 | 6 | 0 | --- | DSCS II | --- | |

Figure 7.1-4

| Component | Qty. | Unit Wt. kg | Total Wt. kg | Unit Pwr. Watts | Dimensions cm | Heritage | | |
|---|---|---|---|---|---|---|---|---|
| | | | | | | Program | Vendor | Status |
| STRUCTURE/MECHANICAL (Continued) | | | | | | | | Adaptation of Existing Technology |
| GRS Boom (Climatology) | 1 | 9 | 9 | 20 | 7.5 x 500 | Voyager | Astro Research | |
| Upper Stage/SC Adapter | 1 | 18.8 | 18.8 | 0 | $\left.\begin{array}{l} d_1 = 102 \\ d_2 = 229 \end{array}\right\} \times 165$ | --- | -- | |
| Upper Stage Separation 2 Marmon Type Ring Clamps 2 Frangible Bolts | 4 | 1.75 | 7 | Negligible | | --- | -- | |
| OIM Attachment | 4 | 1.5 | 6 | Negligible | 102 dia. x X2 Ring | --- | -- | |
| | CLIMATOLOGY = | | 175.8 | | | | | |
| Booms & Appendages | 3 | 5.1<br>5.1 | 10.2 | 30 | 400<br>400<br>216 | | | |
| Total Vehicle Weight | | | 166.8 | | | | | |
| | AERONOMY = | | 177.0 | | | | | |

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Large diameter provides substantial volume and area growth or availability for large experiments that might be considered.

Excess capability matchups that might be considered are listed in the table below:

CAPABILITY GROWTH

| Available Growth | Antenna (HGA) (Larger) | Experiments (Increase) | Electrical Power (Increase) | OIM Star 37 | ACS (Increase Prop.) | Upper Stage SRM-1 (Increase Caplty.) | Mission |
|---|---|---|---|---|---|---|---|
| Weight | X | X | X | X | X | | X |
| Volume | X | X | X | X | X | | X |
| Performance (Martian Orbit) | | | | | | X | X |
| Solar Array (add skirt) | X | X | X | X | X | X | X |

Figure 7.1-5

## 7.2 ELECTRICAL POWER SYSTEM

### 7.2.1 Requirements - Function

The Mars Orbiter electrical power and distribution subsystem (EDPS) provides the following functions for the Climatology and Aeronomy missions:

- An uninterrupted source of electrical power compatible to user needs for the duration of the mission. This also includes the control, processing and storage of power as required.
- Protection of loads from damage due to power subsystems and other load failures
- Commands and telemetry for power subsystem monitoring and control

### 7.2.2 Requirements - Mission and Power

Table 7.2-1 shows the Climatology and Aeronomy mission characteristics which influence and impose EPDS performance requirements. The number of sun occultations establish battery usage which is approximately 8725 discharge/charge cycles. The extended 1 year mission would increase the discharge/charge cycles to 17,550. For the Aeronomy mission about 1000 battery discharge/charge cycles are envisioned. The extended 1 year mission would increase the required battery cycles to about 2000. The length of orbital periods and sun occultations and associated charging time are typical of Earth orbital missions. Because of the maximum Mars-Sun distance of 1.67 AU, the solar array specific performance at Earth is reduced approximately 60 to 65% at Mars. At 1.37 AU, the minimum Mars-Sun distance, which occurs every 22.6 months, the specific performance of the solar array is reduced to about 50 to 55% to that established at Earth. To provide solar array power for user load requirements at the end of the mission and in particular at 1.67 AU, the available power at Earth is about 3 times that required at Mars. Excess power at Earth, and until EOM but at a lower level, must be dissipated which would require a shunt limiter type system. A limiter system is also required during Mars orbital operations to limit system bus voltage to 35 VDC. In view of the distance from the sun, (1.67 AU), solar array operating temperatures at Mars ($-38^{o}C$) are reduced from that of Earth ($+11^{o}C$) thereby causing a swing in the voltage for peak power. In addition, as the spinning S/C emerges from sun

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| Mission Characteristics | Climatology | Aeronomy |
|---|---|---|
| Mission Duration | 195 day average transit plus 687 earth days | 195 day average transit plus 687 earth days |
| Orbital Period | 1.89 hours | 6.68 hours |
| Sunlight Time | 1.19 to 1.29 hours | 5.01 to 6.68 hours |
| Sun Occultation Time | 0.6 to 0.7 hours | 0 to 1.67 hours |
| Number of Sun Occultations | 8725 | 1000 |
| Orbital Altitude | 300 km | Periapsis 150 km |
| Maximum Mars Distance From Sun | 1.67 AU | 1.67 AU |
| Minimum Mars Distance From Sun | 1.37 AU | 1.37 AU |

Table 7.2-1

MISSION CHARACTERISTICS THAT EFFECT
POWER SUBSYSTEM CONFIGURATION
PERFORMANCE AND DESIGN

occultations, the solar array temperatures are estimated to be about -118°C, causing a significant voltage swing.

Tables 7.2-2 and 7.2-3 show the Climatology and Aeronomy spacecraft power requirements in a worst case condition for various key phases of the missions. Sequencing of the loads may reduce the total requirement. Battery sizing is based on sun occultation power levels in combination with mission life requirements related to battery discharge/charge cycle requirements of both missions. Solar array sizing is based on worst case end of mission power requirements with full science and transmission on. During the interplanetary cruise mode, trickle charging of the batteries is mandatory for ensuring a fully charged battery prior to usage.

### 7.2.3 Power Subsystem Options

The selection of the Mars Orbiter power subsystem components has been based primarily on the applicability and adaptability of the subsystem elements to perform within mission requirements and constraints with emphasis on design/hardware inheritance. A summary of these options, showing advantage and disadvantage and selection is shown on Table 7.2-4.

Although the basic use of radioisotope generators offer some significant mass advantages for this type of mission (trading off with solar arrays and batteries), operational hazards in addition to interference with the gamma ray spectrometer operation preclude its usage. Solar cell arrays and batteries are compatible with the selected scientific payload objectives. However, growth factors are limited due to the conical spacecraft configuration.

Nickel hydrogen batteries have been given consideration and compared to nickel cadmium batteries. Although slightly heavier depths of discharge are allowed, offering some small mass advantage, the lack of Ni $H_2$ battery flight experience at this writing, combined with extremely high costs, dictate the selection of the tried and proven, less costly nickel cadmium batteries. Ni $H_2$ batteries are also characteristic of increased volume, about 2:1 over Ni Cd batteries.

In terms of voltage control, series limiters, regulators, voltage limiters were briefly compared with partial shunt limiters. Because of

POWER REQUIREMENTS FOR CLIMATOLOGY MISSION

(WATTS)

| Subsystem | Cruise Mode | Orbit Insertion | Orbital Operation | |
|---|---|---|---|---|
| | | | Full Sun | Eclipse |
| | | | Science On Transmission | Night Science On |
| Propulsion | < 1.0 | < 1.0 | < 1.0 | 1.0 |
| Science | 0 | 0 | 59 | 55 |
| Attitude Control | 6 | 6 | 10 | 10 |
| Command and Data Handling | 24 | 24 | 44 | 44 |
| Communication | 92 | 92 | 92 | 92 |
| Thermal Control | 121 | 121 | 46 | 34 |
| Electrical Power Conversion Loss | 2 | 2 | 12 | 12 |
| Electrical Power Distribution Loss | 4 | 4 | 28 | 8 |
| Total | 250 | 250 | 292 | 256 |
| Battery Charge | 30 | 30 | 230 | --- |
| Total | 280 | 280 | 522 | 256 |

Tabl .2-2

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POWER REQUIREMENTS FOR AERONOMY MISSION

(WATTS)

| Subsystem | Cruise Mode | Orbit Insertion | Orbital Operation | |
|---|---|---|---|---|
| | | | Full Sun | |
| | | | Science On Transmission | Night Science On |
| Propulsion | 1.0 | 1.0 | 1.0 | 1.0 |
| Science | 0 | 0 | 47 | 47 |
| Attitude Control | 6 | 6 | 10 | 10 |
| Command and Data Handling | 24 | 24 | 44 | 44 |
| Communication | 92 | 92 | 92 | 92 |
| Thermal Control | 109 | 109 | 46 | 34 |
| Electrical Power Conversion Loss | 2 | 2 | 8 | 8 |
| Electrical Power Distribution Loss | 4 | 4 | 28 | 8 |
| Total | 238 | 238 | 276 | 244 |
| Battery Charge | 30 | 30 | 125 | --- |
| Total | 268 | 268 | 401 | 244 |

Table 7.2-3

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## POWER SUBSYSTEM OPTIONS

| Component | Advantages | Disadvantages | Selection |
|---|---|---|---|
| • Primary Power Source<br>Radioisotope Thermo-electric Generators | • Solar Panel Sun Orientation Not Required<br>• Boom Shadows Do Not Effect Performance<br>• Batteries Not Required For Eclipse (Note: Small Battery May Be Used, To Save Isotopic Fuel, For Peak Transmission Periods)<br>• Less System Weight | • Possible Operational Hazard During Ground Handling<br>• Not Compatible With On-Board Science (Gamma Ray Spectrometer) | |
| Solar Cell Array | • Non-Hazardous<br>• Compatible With S/C Science<br>• Tried and Proven | • S/C Orientation Required<br>• Sizing Limitations | Selected |
| • Secondary Power Source (Batteries) | | | |
| Nickel Hydrogen | • Reduced Weight<br>• Heavier Depth of Discharge | • Increased Volume (2 to 3 to 1)<br>• Costs High<br>• Lack Of Flight Experience | |
| Nickel Cadmium | • Tried and Proven<br>• Costs Less | | Selected |
| • Power/Voltage Control | | | |
| Series Limiter | • No Need To Dissipate Excess Power<br>• No Input Voltage | • Forces Operation Away From Maximum Power Point<br>• Highest Solar Array Area Requirement<br>• Double Power Conversion Required | |
| Partial Shunt Limiter | • Maintains Bus Voltage Within Limits<br>• Maximum Use Of Proven Design<br>• Eliminates Need For Radiator | | Selected |
| Voltage Limiter | • Maintains Bus Voltage Within Limits<br>• Maximum Use of Proven Design | • Requires Dissipation Of Excess Solar Array Power | |

Table 7.2-4

the selection of a conical array on a spinning spacecraft, the 8 to 10% power conversion loss considered with series limiters, regulators, etc., in actuality requires about a 9.3 to 1 increase in conical array area due to the combination of the array geometry (0.299) and loss in power at 1.67 AU, approximately 65%. (For example, on a 300 watt load, the power conversion loss would be about 20 to 30 watts.) To provide for the geometrical loss, the conical array area must be increased by a factor of 3.34 (1/0.299). In addition, the solar constant is reduced to $0.358 \left(\frac{1 \text{ AU}}{1.67 \text{ AU}^2}\right)$, thereby requiring a further increase in total conical solar cell array, by a total factor of 9.3 to 1 (3.34/.358). The voltage limiter system requires a radiator for dissipating excess heat at Earth, approximately 1 kw due to low power demands. At the 1.67 AU operation, as the S/C emerges from the sun occultations, the solar array operating temperature has been estimated to be about -118$^{o}$C but increases to a steady state temperature of -38$^{o}$C. Excess power to be dissipated at Mars (1.67 AU) for limiting the bus voltage to 35 VDC is about 20 watts at worst case conditions for either the partial shunt or voltage limiter approach. The partial control system has been selected because there is no requirement for a radiator.

### 7.2.4 Electric Power Subsystem Description

A simplified functional block diagram of the baseline MO direct energy transfer system is shown in Figure 7.2-1. Conical solar arrays with body mounted solar cells and coverslides provide user's load requirements and battery charging power in sunlight from launch (after ejection from the shuttle) to the end of the mission.

Nickel cadmium batteries provide energy (power) during launch until spacecraft sun acquisition and during planetary sun occultations or when the load momentarily exceeds the capability of solar array. (Note: the length and the number of sun occultations and power levels vary between the Climatology and Aeronomy missions.)

Solar array power is routed to the power control unit (PCU). The PCU contains three modular battery chargers, redundant command decoding logic, shunt regulator error amplifier (REA), power distribution, fault isolation and telemetry circuits. The main bus voltage at the PCU output is 23.5 to 35.0 VDC. Power is distributed directly to loads through fuses

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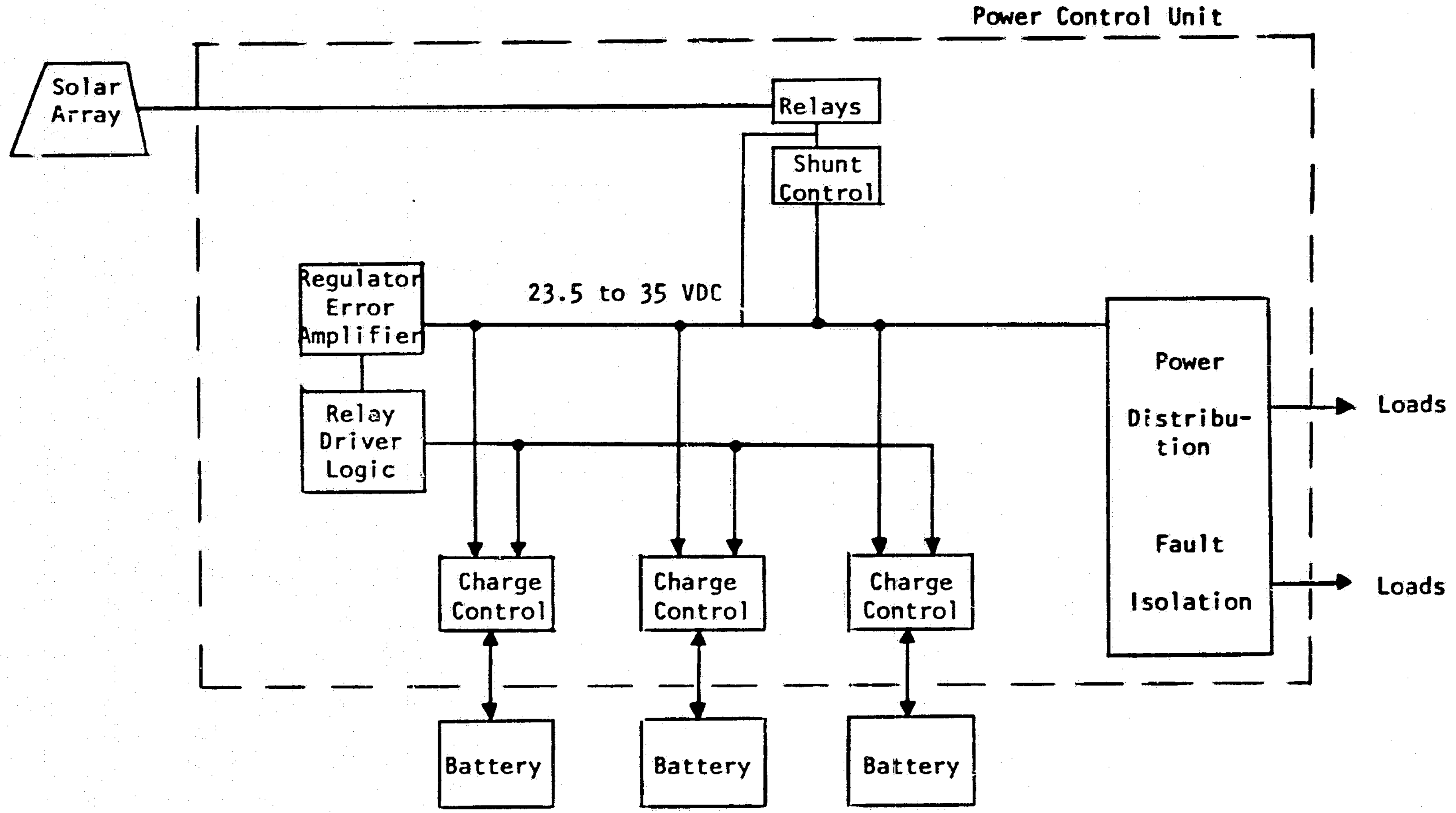


Figure 7.2-1

MARS ORBITER ELECTRICAL POWER SUBSYSTEM BLOCK DIAGRAM or through fused outputs to power switching units. The shunt limiter is designed to limit the upper bus voltage at 35 VDC. Shunt limiting is accomplished by sequentially shunting excess current through the power transistors of the shunt element assemblies. Presented herein are brief descriptions of the selected baseline subsystem components.

Solar Cell Array

The solar cell array is a conical type with a worst case solar cell projected area of 7.4 $m^2$ for the Climatology mission and 6.4 $m^2$ for the Aeronomy mission. Figure 7.2-2 shows the conical solar cell configuration. The conical solar cell array design provides power at all sun angles in the forward hemisphere. Although the originally proposed solar cell showed an 11.3% efficiency, AMO, 28°C, (TDRSS inheritance) a change to a cell of higher efficiency in a low intensity and low temperature environment is required, in particular for the 1.67 W Sun-Mars distance operation. Solar cells with 13.3 to 14.2% efficiency factors (at 1 AU) are currently being flown on Earth orbital missions. Testing at low intensity and low temperature has been performed but limited (Reference 1).* Assuming similar coefficients, the effects of using a 2 ohm-cm BSR solar cell 2 x 4 cm for these missions show an increase of approximately 18% in performance (AMO) over that stated in the proposal in addition to the increase in performance at Mars.*

At Earth (BOL), the specific power is 156 $W/m^2$, at 11°C, and assuming a conical array solar cell packing factor of about 80%. At 1.67 AU, the solar intensity is reduced to about 48.5 $mW/cm^2$ and the solar cell temperature to -38°C. Considering the Viking Mars orbiter solar cell array degradation of about 5%, and sun angles, the specific power of the conical array is reduced to about 73.6 $w/m^2$ and 66.1 $w/m^2$ for the Climatology and Aeronomy Missions, respectively.

The solar cell series-parallel network consists of 63 cells wired in series for both type missions. Consideration has been given to providing sufficient battery charging voltage at Earth with minimum wasted

* Reference 1. NASA TM 78253, "Characterization of Three Types of Silicon Solar Cells for SEPS Deep Space Missions," Vol. 1, NASA Marshall, Huntsville, Alabama, Page 79.

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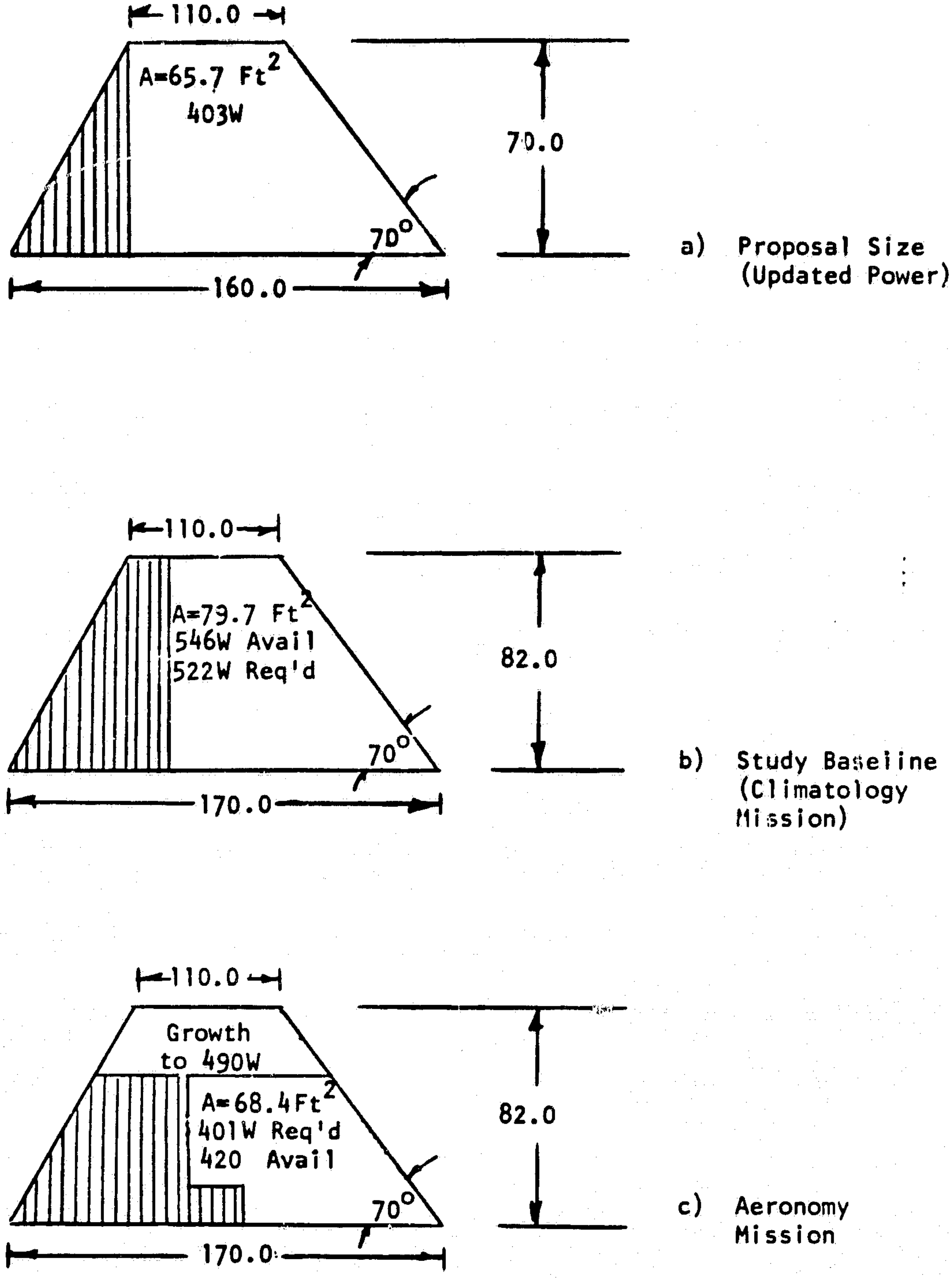


Figure 7.2-2

SOLAR ARRAY CONFIGURATIONS

power at Mars while limiting the solar cell array output voltage to about 36.5 VDC. The number of parallel strings will differ for each mission because of the difference in load requirements. For the Climatology mission, 143 parallel strings would suffice for providing 546 watts (End of mission) while 123 parallel strings would be sufficient for the Aeronomy mission. The maximum number of parallel strings that could be installed on the conical configuration is about 143, corresponding to about 546 W at 31.8 VDC. The total number of solar cells established for the Climatology mission is 30,130 and 26,060 for the Aeronomy mission.

Batteries

Three 15 AH 22 cell nickel cadmium batteries flown on the DSCS-II spacecraft are being used for the Climatology and Aeronomy missions. Nominal battery discharge voltage is 26.4 VDC. Battery capacity discharged during the Climatology 0.7 hour sun occultation period is 6.8 AH corresponding to a 15.0% depth of discharge. Battery charging occurs during the 1.2 hour sun period. A requirement is to provide for about 8725 discharge/charge cycles. The extended one Martian year mission would double this requirement. Failure of one battery will increase the depth of discharge to about 22.6% which is reasonably shallow enough to complete the basic mission on two batteries.

Failure of one battery in a two 15 AH battery system would increase the depth of discharge to about 42.5%, considered too marginal for the required number of discharge/charge cycles of the Climatology mission. Increasing the rating (capacity) of the batteries to operate at shallower depths of discharge (with two batteries), would enhance cycle life but would increase battery mass and decrease reliability on the basis that three batteries are better than two.

For the Aeronomy mission, the sun occultation time will vary from 0 to 1.67 hours during the 6.68 hour orbit. The number of sun occultation battery discharge periods has been estimated to be about 1000. Battery capacity discharged during the maximum 1.67 hour sun occultation is about 15.4 AH corresponding to a depth of discharge of 34.3%. This assumes the TWT is on during the eclipse operation. Battery charging occurs during the minimum 5.01 hour sun period. Failure of one battery would increase the

D of D to about 51.4%, reasonably shallow enough at a temperature of 0 to 10°C for the 1000 discharge/charge cycles required.

During interplanetary cruise, trickle charging of the batteries occurs at the C/50 rate where C = total battery capacity of 45 AH.

Power Control Unit

The power control unit selected for the Mars Orbiter has been used on the HEAO spacecraft series. The PCU contains command processing, telemetry, undervoltage, power distribution and fault isolation functions in addition to the shunt regulator error amplifier and relay driver logic circuitry. Individual charge/discharge controls for each of three batteries are also located within the PCU.

The relay driver logic circuitry receives parallel data from the Secondary Command Decoders and provide pulse outputs to control and configure the battery charge controls.

The battery charge controls provide the circuitry to sense battery voltage, as a function of temperature, and control current as required. Eight commandable voltage levels are provided for adjustment of recharge fraction. The charge controls automatically respond to battery over temperature to reduce charging current. Command capability for reconditioning is also provided.

An undervoltage sensor monitors the main bus voltage and generates a signal to disconnect non-essential loads when bus voltage decreases to a level indicative of low battery capacity.

The power distribution circuitry routes load power through current monitors and fuses to the experiments and spacecraft loads.

Shunt Limiter

The primary bus shunt limiter consists of the following elements: a) the regulator error amplifier (REA), b) the shunt drive assembly (SDA), c) the main bus filter network, d) the shunt disconnect relays, and e) the linear output power transistors and associated power resistors on the two shunt element assemblies (SEAs). A block diagram of the shunt limiter is shown in Figure 7.2-3. The shunt limiter is designed to limit the main bus voltage to 35.0 VDC. The shunt limiter utilizes solar array taps which

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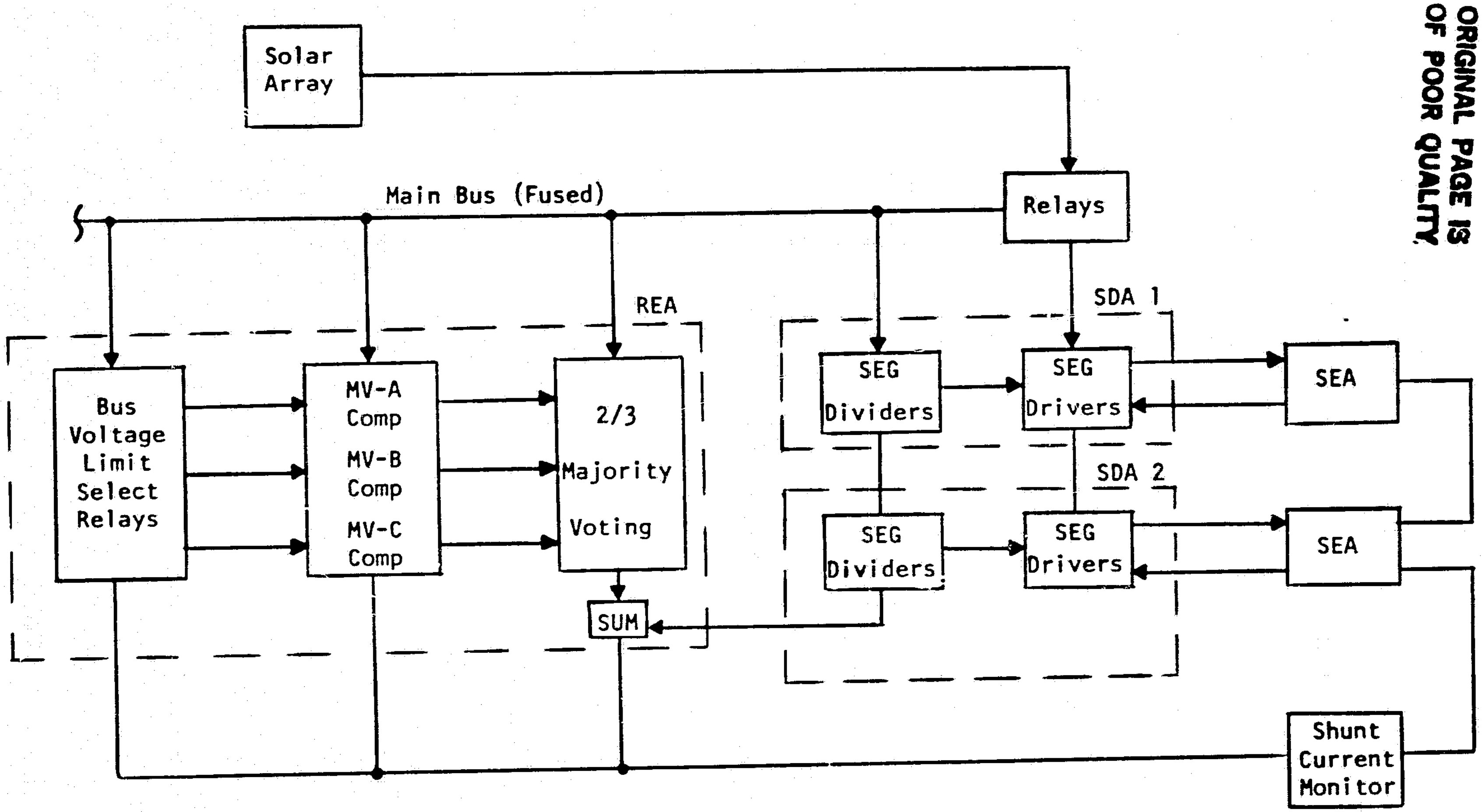


Figure 7.2-3

MARS ORBITER SHUNT LIMITER BLOCK DIAGRAM are located at some point up the solar array strings as required to ensure bus limiting during sun light conditions. Shunt limiting is accomplished by sequentially shunting current through the power transistors of the SEAs. When the solar array power is less than the required load power, the shunt limiter is in standby, and the main bus voltage and impedance characteristics are determined by the combination of the battery, load and solar array characteristics. When the solar array power is greater than the load power the shunt limiter is active and produces voltage limiting and main bus impedance control.

Power Distribution

Primary power is routed from the PDU located within the PCU and on the spinning section through slip rings to the despun platform for science and remaining subsystems usage. Figure 7.2-4 shows the basic approach for providing power to the despun platform. Using subsystems located on the spinner are provided power direct by way of the Electrical Integration Assembly.

Each output to the loads is protected by plug in fuses and fusistor modules located within the PCU/PDU. The distribution system also consists of 4 power buses for subsystem, telemetry, ordnance and to experiments.

Mass and Inheritance

Table 7.2-5 summarizes the EPDS mass, quantities and inheritance of components tentatively selected for the baseline configuration. Basically all components have flight proven experience, although the selected solar cell has not been flown in the low intensity, low temperature Martian orbit environment. Modifications of inherited components are minimal. The total EPDS mass is about 142.4 kg and 138.8 kg for the Climatology and Aeronomy missions, respectively. The end of mission specific power is about 3.5 w/kg which is somewhat heavier than that experienced in present Earth orbital missions. This is due to the loss of power at Mars and the use of inherited hardware (PCU, PDU) not designed optimally for these missions.

Commonality has been established for both mission power system designs. Differences in mass are related to the solar panel area due to the reduced power levels of the Aeronomy mission.

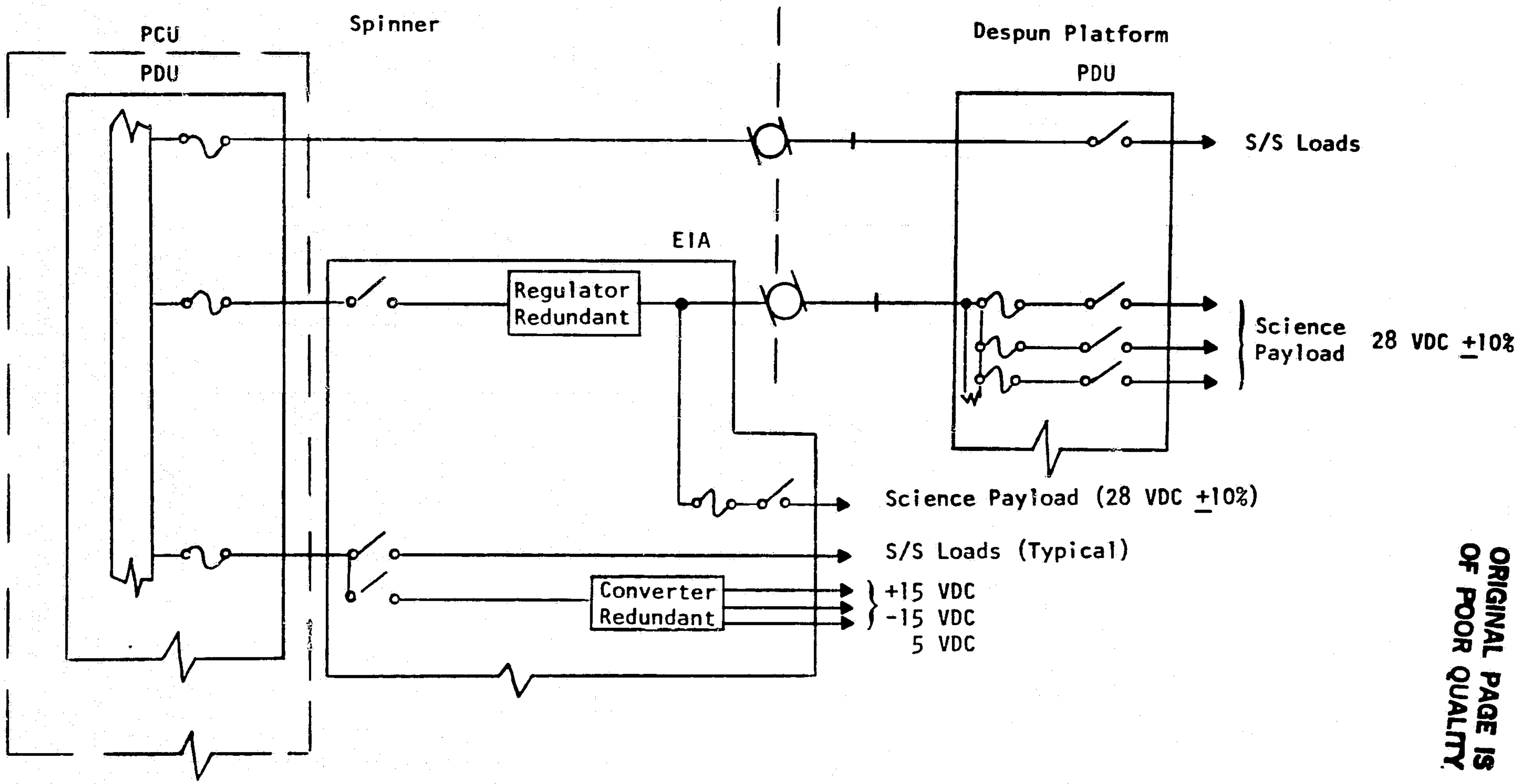


Figure 7.2-4

DESPUN PLATFORM AND SPINNER SECTION
BASIC POWER DISTRIBUTION AND FAULT
ISOLATION CONCEPT FOR THE MARS ORBITER

Table 7.2-5

POWER SUBSYSTEM EQUIPMENT LIST

| Component | Quantity | Mass (kg) | Inheritance |
|---|---|---|---|
| • Solar Cell, Coverslide | | | |
| Climatology | | | |
| 7.4 $m^2$ Effective | 24.8 $m^2$ | 30.4 | GRO, LANDSAT, Classified |
| Aeronomy | | | |
| 6.4 $m^2$ Effective | 21.4 $m^2$ | 26 kg | GRO, LANDSAT, Classified |
| • Batteries | 3 | 51.9 | DSCS II |
| (Nickel Cadmium) | | | |
| • Power Control | 1 | 13.7 | HEAO (Modified) |
| • Shunt Limiter | 1 | 6.4 | DSP-14 (Modified) |
| • Electrical Distribution | 1 | 10.5 | DSCS II (Modified) |
| • Harness | 1 | 29.5 | DSCS II (Modified) |
| | Total Mass | 142.4 kg | Climatology |
| | | 138.8 kg | Aeronomy |

### 7.2.5 Performance

Power profiles of the Climatology and Aeronomy missions are shown in Figures 7.2-5 and 7.2-6, respectively. Sequencing of subsystem loads may reduce power levels below that shown which is a worst case condition. The available solar array power at 1.0 AU (Earth), 1.37 AU (closest Sun-Mars distance), 1.52 (arrival Sun-Mars distance) and 1.67 AU (farthest Sun-Mars distance) is also shown. Sufficient power is available throughout all phases of both missions at worst case conditions. As shown in Figure 7.2-5, (Climatology Mission) the available solar array power capability at (Earth) is about 1150 w and is reduced to approximately 725 w at 1.37 AU, Mars closest distance to the sun. As the spacecraft proceeds to 1.67 AU, Mars farthest distance from the sun, the solar array output is reduced to approximately 546 w. For the Aeronomy Mission at the 1.67 AU Mars-Sun distance about 420 watts is available (required 401 w) with growth availability to about 490 w.

The areal efficiency ($\eta$) of 0.299 (Section 7.2.3) is for a worst-case aspect angle ($\theta$) of 90°. That is, the sun line is in the orbit plane. Since this condition never occurs in the Climatology Mission (baseline orbit), more power is available. The efficiency ranges between 0.325 and 0.350 for $\theta$ between 0 and 68 degrees (mission limits), and between 0.347 and 0.350 for $\theta$ between 45 and 68 degrees (mission limits after drift period). See also Figure 6.1-4. The Climatology power available at 1.67 AU, 546 W in Figure 7.2-5 reflects the factor $\eta$ = 0.347.

In the drift period, when $\eta$ can have intermediate values, the power available reflects intermediate values of $\eta$, rising to the value of $\sim$.347 when the drift period ends, and increasing sun-Mars distances, rising to 1.66 AU when the 155-day drift period ends. In addition, considerably less than the allocated 5% EOL degradation will have occurred this early in the mission. Combining these factors indicates that during the drift period of the Climatology Mission there is never less power available than the 546 W indicated in Figure 7.2-5.

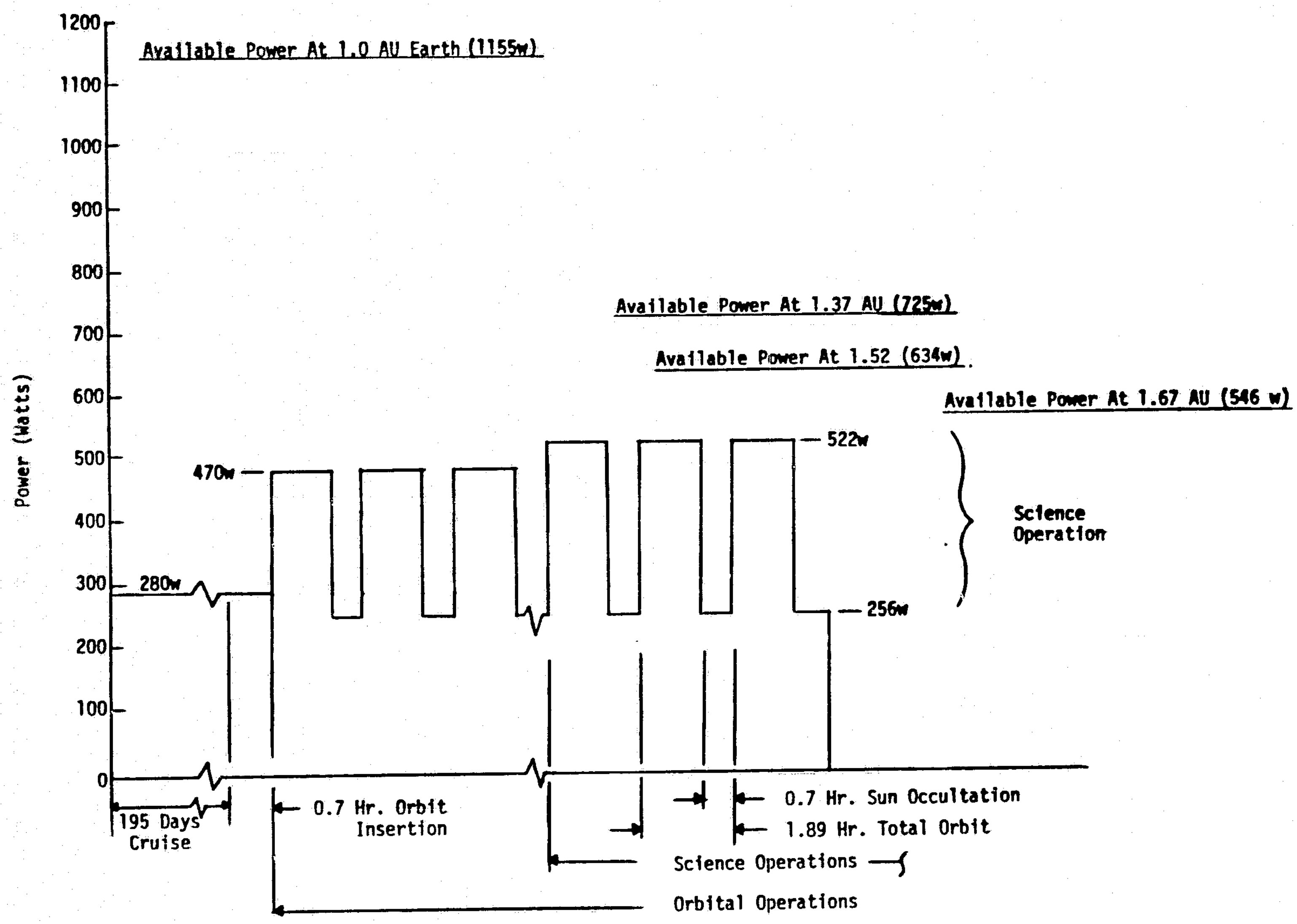


Figure 7.2-5

POWER REQUIRED VS AVAILABLE POWER FOR THE CLIMATOLOGY MISSION

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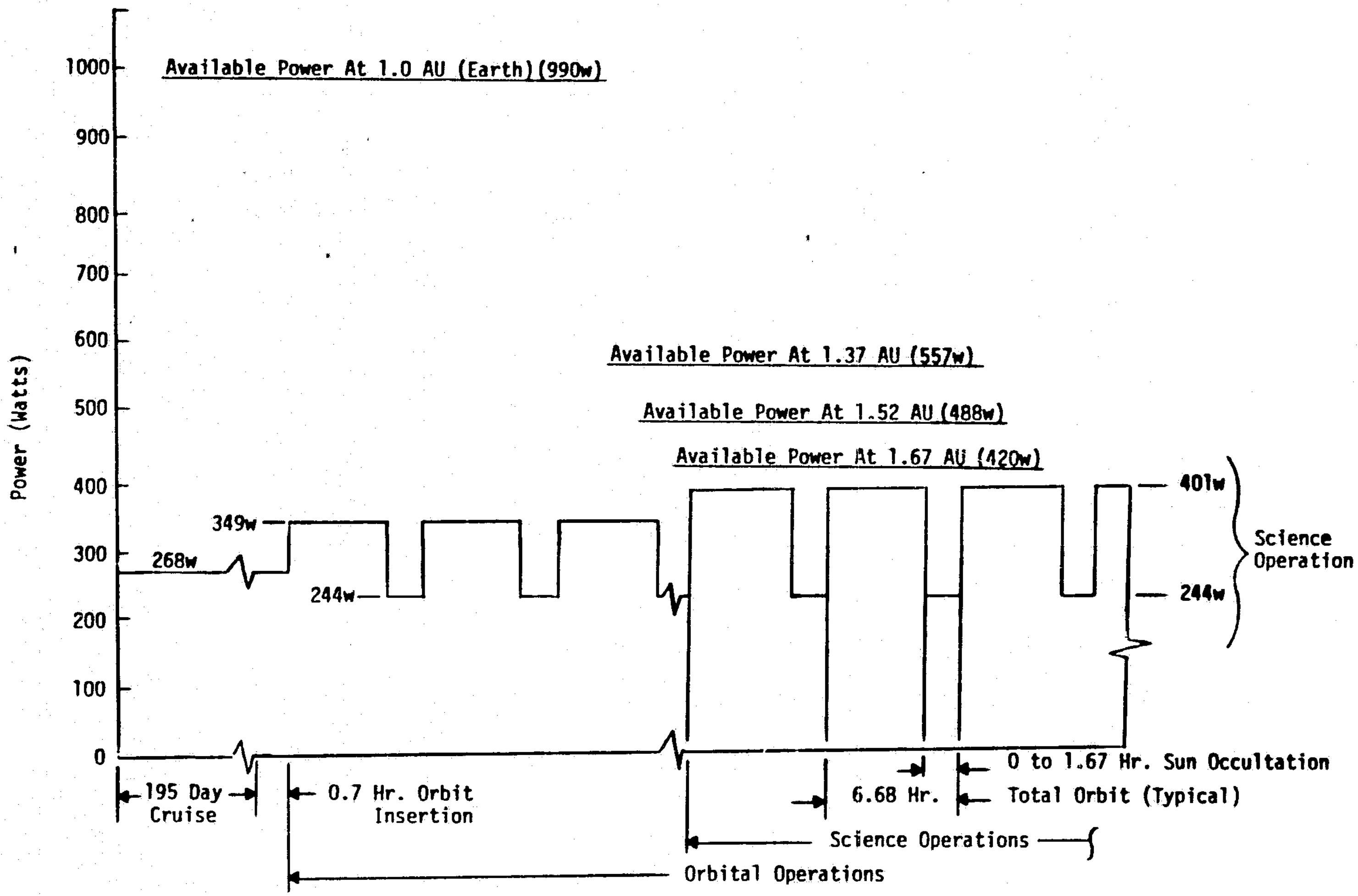


Figure 7.2-6

POWER REQUIRED VS AVAILABLE POWER FOR THE AERONOMY MISSION

## 7.3 COMMUNICATION SUBSYSTEM

### 7.3.1 Subsystem Functions

The communication subsystem contains the equipment required to perform the following functions:

a) Receive and detect command modulation data on an S-band uplink carrier transmitted by the Deep Space Network (DSN).

b) Receive and demodulate ranging modulation on an S-band uplink carrier transmitted by the DSN. (However, ranging modulation is not a mission requirement.)

c) Phase modulate science and engineering data onto an X-band carrier for downlink transmission to the DSN.

d) Phase modulate turnaround ranging modulation onto an X-band carrier for downlink transmission to the DSN.

It can perform functions (c) and (d) in either a two-way coherent mode of operation or a one-way non-coherent mode. In two-way operation, the X-band downlink carrier frequency is derived from the received S-band uplink carrier with a coherency ratio of 880/221. This enables range-rate (doppler) measurements to be made at the DSN on the turned-around carrier, as well as range measurements via the turnaround ranging signal. In one-way operation the downlink carrier is derived from an auxiliary (non-coherent) oscillator contained in the communication subsystem.

### 7.3.2 Subsystem Requirements

The first five columns of Figure 7.3-1 list the critical communications performance requirements according to the phase of the mission (Column 1). The minimum and maximum spacecraft-to-earth distances for each phase are given in Column 2. The spacecraft attitude is implied by the earth aspect angle (angle between the +Z axis of the spacecraft and the earth line) given in Column 3. The earth aspect angle has a range of values for a given mission phase if it undergoes change during the course of that phase (example: in the immediate post injection attitude, the spacecraft-earth line, is rotating rapidly) or if attitude during the phase is not determinable in advance (example: midcourse trajectory maneuver).

## MARS ORBITER COMMUNICATIONS CHARACTERISTICS

| SELECTED MISSION PHASES | RANGE (AU) | EARTH ASPECT ANGLE | LINK | BIT RATE REQ'D (B/S) | SC ANTENNA AND GAIN | BIT RATE CAPABILITY (B/S) |
|---|---|---|---|---|---|---|
| INJECTION AND POST INJECTION | 0-.001 | $90^0$-$180^0$ | S ↑ | NONE | -- | -- |
| | | $-130^0$ | X ↓ | NONE | -- | -- |
| TCM 2 | 0.04 | $0^0$-$100^0$ | S ↑ | 125 | OMNI (-3DB) | 120,000 |
| | | | X ↓ | 250 | HORN OR BICONE (+3DB) | 7,400 |
| MARS ORBIT INSERTION | 1.23 | $38^0$ | S ↑ | 7.8 | OMNI (0DB) | 500 |
| | | | X ↓ | 16 | HORN (+4.5 DB) | 16 |
| SCIENCE PLATFORM SPINNING ↑ DESPUN ↓ | | | | | | |
| ORBIT INCLINATION CHANGE | 2.0-2.2 | $40^0$-$60^0$ | S ↑ | 125 | HGA (1.1M) | 62,000 |
| | | | X ↓ | 16 | HGA (1.1M) | 19,600 |
| ORBITAL OPERATIONS (INCLUDING ΔV'S IN NORMAL ATTITUDE) | 0.5-2.65 | $0$-$120^0$ (A, $C_0$) | S ↑ | 125 | HGA | 43,000 |
| | | $0$-$105^0$(C) | X ↓ | 6000(A) / 12000(C, $C_1$, $C_2$) | HGA (1.1M) | 13,500 |
| | | | | 18000($C_3$) | HGA (1.5M) | 25,000 |
| EMERGENCY | 0-2.65 | ? | S ↑ | 7.8 | OMNI (-3DB) | 54 |
| | | | X ↓ | ? | ? | ? |

Figure 7 1

Column 4 indicates whether the requirement pertains to the uplink (implemented by S-band) or the downlink (implemented by X-band). Column 5 gives the bit rate required during the mission phase.*

The last two columns show the antenna selection and estimated performance, discussed in Section 7.3.5.

The minimum data rates required are 7.8 bits/sec uplink and 16 bits/sec downlink. These are for periods when only engineering data are significant and when attitude and antenna availability keep link capability low.

Note that during "Orbital Operations" large downlink data rates are required. These are based on data handling analyses given in Section 6.3. The high-gain antenna must be despun and aimed at the earth to accommodate these data rates: 6000, 12000, and 18000 bits/sec for the various mission options.

### 7.3.3 Communications Subsystem Options

Options examined reflect choices in several areas, not all independent:

1. Whether spacecraft antenna(s) should be located on the spinning section as well as on the despun section.

2. Which frequency (S- or X-band) should be used for the primary downlink.

3. (Having selected X-band), whether any S-band downlink be implemented.

4. If there should be communications to the TDRSS for the injection and immediate post-injection phase.

5. What RF links between the Shuttle Orbiter and the spacecraft should be implemented.

---

A = Aeronomy Mission

C = Climatology Mission

$C_0$ = Climatology Mission (Orbit Option)

$C_1,C_2,C_3$ = Climatology Mission (Payload Options)

## COMMUNICATIONS CHARACTERISTICS

1. ALL SPACECRAFT ANTENNAS ARE ON DESPUN SECTION
   - MAY INHIBIT COMMUNICATIONS FROM INJECTION UNTIL SEQUENCER-INITIATED PRECESSION (ABOUT ONE HOUR)
   - SAVES ANTENNA, DIPLEXER, TRANSPONDER
2. USE X-BAND FOR THE PRIMARY DOWNLINK
   - S-BAND WOULD BE PROHIBITIVE (POWER, ANTENNA SIZE) TO RETURN SCIENCE DATA AT DISTANCES REQUIRED
   - CONSISTENT WITH DSN TREND TO EMPHASIZE X-BAND RECEPTION
3. DELETE S-BAND DOWNLINK
   - X-BAND PERFORMANCE IS BETTER, AT 20W VS 5W
   - SAVES S-BAND POWER AMPLIFIER, 2 DIPLEXERS
4. DELETE COMMUNICATIONS TO TDRSS
   - PER CHANGE TO WORK STATEMENT
5. RESTRICT STS LINK TO ONE-WAY (STS → SC)
   - ENABLES SEQUENCER START

Figure 7.3-2

The selections in these areas, and summary reasoning is given in Figure 7.3-2. A more conprehensive analysis and rationale is given in Appendix A.

Other options have to do with transmitter power, antenna complement, antenna location, component location, and redundancy.

Transmitter Power. X-band traveling-wave tube amplifiers (TWTA's) are available with output powers of 10, 20, and 40W. We selected 20W as a compromise between data requirements and the spacecraft power budget.

Antenna Complement. Having made the frequency selections (2 and 3) in Figure 7.3-2, the antenna complement, in effect, is separated into an S-band group and an X-band group. However, high gain antennas are required for downlink during normal orbital operations, and for normal commanding (125 bits/sec) during the same operations, so one antenna, a 1.12-meter paraboloidal reflector, is selected with both an S-band and an X-band feed. (For Climatology Payload Option 3, this reflector is increased to 1.5 meters.)

From launch until just after Mars orbit insertion (MOI) the despun section remains spinning, and the HGA is tethered in its launch position -- pointing 90 degrees from the Z-axis. During this period, communications are via medium- and low-gain antennas whose patterns are symmetric about the Z-axis.

From among a number of possible complements, a triad -- S-band omni (log conical spiral) and two X-band medium-gain antennas, a horn parallel to the X-axis and a bicone to cover the spacecraft's equatorial region -- has been selected.

Antenna Location. With all antennas on the despun platform, there is a choice of location of attachment points:

- below the azimuth gimbal
- beyond the azimuth gimbal, but below the elevation gimbal
- beyond the elevation gimbal

Criteria for selection include:

- Compactness and simplicity of coax cable and waveguide routing
- Exclusion of mutual field-of-view conflicts between antennas; maintaining the desired antenna patterns.
- Avoiding shadowing the solar array
- Keep masses as much as possible toward the -Z direction, to minimize the ratio of transverse to spinning moments of inertia

The selected configuration, all antennas beyond the elevation gimbal, serves the last two objectives. It has the potential to satisfy the first objective, but the TWTA's selected location, below the elevation gimbal, dilutes this advantage. As to the second criterion, the selected configuration avoids almost completely any mutual interference. However, after arrival at Mars and the HGA elevation gimbal is released, as the HGA elevation angle departs from 0° (90° earth aspect angle) the omni and MGA's patterns deviate from axial symmetry. However, this is of concern only in emergencies and other cases when the HGA can not be used. A candidate fault protection routine will restore the HGA to 90° aspect angle to optimize the omni and MGA patterns.

Component Location. With the antennas selected and located as described above, the ideal location for the electronic components of the communications subsystem -- particularly the TWTA's -- is above the elevation gimbal at the back of the HGA reflector. However, such a location jeopardizes moment-of-inertia stability.

For this reason the selected location of the transponders, TWTA's, and hybrids is a small platform at the base of the HGA mast. This platform is about at the location of the azimuth gimbal, but topologically beyond it. Switch S-3 is on the back of the HGA reflector; S-1 and S-2 can be either near the S-3 location, or on the same platform as the transponders and TWTA's.

Redundancy. The choice here is standard for communications subsystems: the antennas are not redundant (except to the extent that antenna patterns overlap); transponders and TWTA's are redundant and cross-strapped.

### 7.3.4 Subsystem Description

The subsystem consists of redundant NASA Standard Deep Space transponders, redundant X-band traveling wave tube amplifiers (TWTA's), S-band and X-band antennas, an X-band hybrid, RF switches, and interconnecting coaxial cables and waveguides. Each transponder contains an S-band receiver, command detector, S-band exciter, X-band exciter, DC power converters, and telemetry/command conditioning circuits. All subsystem equipment is mounted on the spacecraft despun platform. The subsystem block diagram is shown in Figure 7.3-3 including the interfaces with the Command and Data Handling (C&DH) subsystem.

S-band uplink communication with the spacecraft transponders is available through either a conical spiral type of omnidirectional antenna, providing cardioid-type coverage, or a narrow-beam, parabolic, high gain antenna (HGA). The output of the omni antenna goes through the contacts of RF transfer switch, S-1, to the receiver in transponder A. The HGA output signal goes through the other contacts of switch S-1 to the receiver in transponder B. Since both receivers are powered on in-flight, these two communication links are always operational. In the event of a transponder failure, cross-strap switch S-1 enables connection of the redundant transponder to the active antenna by means of a command stored in the C&DH subsystem. The receiver assembly in each transponder processes a phase-modulated uplink carrier from either the 34-meter or 64-meter DSN nets. Modulation on the uplink carrier consists of a biphase-modulated command subcarrier and/or baseband ranging signals. Prior to signal processing, the receiver phase locks an internal frequency source to an unmodulated, frequency-swept input carrier and maintains phase lock when the carrier is phase modulated. The resulting phase coherent reference tone is used to demodulate the command subcarrier and ranging modulation (if employed) on the input carrier, and to provide a coherent drive signal to the X-band exciter.

The receiver incorporates both coherent and non-coherent automatic gain control (AGC). The former provides AGC and a telemetered lock indication when the receiver is phase-locked. The latter provides gain limiting in both the locked and unlocked state, resulting in a self-adaptive carrier loop bandwidth and thus improved carrier tracking at low input signal-to-noise ratios.

The demodulated command subcarrier is supplied to the command detector for coherent recovery of the baseband command data. The biphase-modulated subcarrier is sampled, A/D-converted, and digitally processed by microcomputer techniques. The resulting command data bit stream is outputted to the C&DH subsystem for command decoding. A bit timing signal (clock) and in-lock indication are also provided. The command detector can process data at any one of nine different bit rates, selectable by a set of contact closure type commands provided by the C&DH subsystem.

The demodulated ranging signal, comprising a pseudo-random noise (PRN) code, is supplied to the X-band exciter where it is phase-modulated on the exciter RF drive for downlink transmission. Spacecraft science and engineering data are supplied to the X-band exciter by the C&DH subsystem in the form of a biphase-modulated telemetry subcarrier. This is phase-modulated on an RF carrier for X-band downlink transmission. The exciter consists of a phase modulator and a chain of frequency multipliers which convert the RF drive to X-band. The RF drive can be either phase-coherent with the S-band uplink carrier or non-coherent, under the control of a commandable switch. The former is supplied by the receiver when it is phase-locked, the latter by a crystal-controlled auxiliary oscillator. The RF drive input is automatically switched to the auxiliary oscillator whenever the receiver is unlocked.

Downlink communication with the DSN is provided at X-band only. The outputs of both X-band exciters go to a hybrid power splitter which drives redundant TWTA-type power amplifiers. One TWTA is always connected through RF transfer switch S-2, to the X-band feed of the HGA and the other, through switches S-2 and S-3, to either of two medium-gain antennas (MGA's), a horn facing the +Z direction or a biconical horn (bicone) providing toroidal coverage.

Subsystem equipment masses, power consumption, and heritage are presented in Table 7.3-1.

COMMUMICATIO., SUBSYSTEM

EQUIPMENT LIST

| Components | Qty. | Unit Wt. kg | Total Wt. kg | Unit Pwr. Watts | Dimensions cm | Heritage | | |
|---|---|---|---|---|---|---|---|---|
| | | | | | | Program | Vendor | Status |
| S/X Transponder | 2 | 2.8 | 5.6 | 13.1 | 19.8 x 14.0 x 8.4 | NASA Std. | Motorola | Purchased Component |
| S-Band Transfer Switch | 1 | 0.28 | 0.28 | -- | 4.5 x 3.3 x 6.4 | Pioneer 11 | -- | |
| S-Band Omni Antenna | 1 | 1.0 | 1.0 | -- | -- | HEAO, Pioneer | -- | Use As Is |
| X-Band TWTA | 2 | 5.2 | 10.4 | 72 | 36.7 x 15.2 x 11.4 | DSCS II | Watkins Johnson | Purchased Component Modified |
| X-Band Transfer Switch | 2 | 0.34 | 0.68 | -- | 6.1 x 5.5 x 3.5 | DSCS II | -- | Purchased Component |
| X-Band Hybrid | 1 | 0.17 | 0.17 | -- | 4.5 x 2.5 x 1.9 | DSCS II | -- | Adaptation of Existing Technology |
| S/X-Band HGA | 1 | 10.2 | 10.2 | -- | 44" Dish | DSCS II | -- | New Dual Feed |
| X-Band Horn | 1 | 2.0 | 2.0 | -- | -- | DSCS II | -- | Adaptation of Existing Technology |
| X-Band Bicone | 1 | 1.0 | 1.0 | -- | -- | -- | -- | |
| RF Cables and Conn. | 1 set | 1.0 | 1.0 | -- | -- | -- | -- | Assemble To Fit |
| X-Band Waveguide | 1 set | 0.5 | 0.5 | -- | -- | -- | -- | |

Table 7.3-1

### 7.3.5 Communications Subsystem Performance

Subsystem performance is shown in three ways:

- Data rate capability vs range is shown in curves in Figures 6.3- and 6.3- for uplink and downlink.
- Sample points in these curves are verified by communications link power budgets, Tables 7.3-2 to 7.3-4.
- Capability vs requirements are indicated by the last two columns of Figure 7.3-1.*

Downlink performance about meets its requirement of 16 bits/sec via the X-band MGA horn at a range of 1.23 AU at the time and spacecraft attitude of MOI. This requires an antenna gain of 4.5 dBi at 38° off the antenna axis, with 20W transmitted power.

The HGA is sized to provide the data rate for orbital operations at the maximum earth-Mars range of 2.65 AU. The selected 1.15-m reflector from the DSCS-II program provides over 12,000 b/s, compared with a requirement of about 6,000 b/s (Aeronomy Mission) and 10,000 to 12,000 b/s (Climatology Mission, baseline payload and payload options 1 and 2). A larger antenna is necessary to satisfy Climatology payload Option 3, 18,000 b/s.

* These performance figures assume the use of the 64-meter antenna at the Deep Space Station. For uplink performance, the transmitted power is 100 kW.

Table 7.3-2

COMMUNICATIONS POWER BUDGET:

UPLINK, S-BAND, DSN 64 (100kW) TO SC OMNI

(At Maximum Range)

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| Parameter | Nominal Value | Adverse Tolerance | Comments |
|---|---|---|---|
| 1. Frequency, MHz | 2113 | | |
| 2. Transmit Power, dBm | + 80.0 | 0.5 | 100 kW, JPL 810-5 |
| 3. Antenna Gain, dBi | 60.7 | 0.7 | JPL 810-5 |
| 4. Antenna Pointing Loss, dB | 0.2 | 0.0 | $e = 0.02°$, $\phi_{3dB} = 0.15°$ |
| 5. EIRP, dBm | +140.5 | 0.9 | RSS Tolerances |
| 6. Atmospheric Loss, dB | 0.2 | 0.0 | |
| 7. Space Loss, dB | 270.9 | 0.0 | 2.66 AU |
| 8. Polarization Loss, dB | 0.0 | 0.0 | |
| 9. Rec'd Isotropic Signal Power, dBmi | -130.6 | 0.9 | RSS Tolerances |
| 10. Antenna Gain, dB | - 3.0 | 0.0 | Omni Antenna |
| 11. Antenna Pointing Loss, dB | 0.0 | 0.0 | Omni Antenna |
| 12. Receive Losses | 2.2 | 0.3 | SR-141, Components (RSS) |
| 13. Rec'r Input Signal Power, dBm | -135.8 | 0.9 | RSS Tolerances |
| 14. Rec'r Noise Temperature/ Noise Figure, dB | 5.5 | 0.0 | |
| 15. Rec'r Input Noise Spectral Density, dBm/Hz | -168.5 | 0.0 | |
| 16. Rec'd Signal-to-Noise Spectral Density Ratio, dB Hz | 32.7 | 0.9 | RSS Tolerances |
| Carrier Channel | | | |
| 17. Carrier Modulation Loss, dB | 9.4 | 0.0 | $\beta_C = 1.8$ rad, $\beta_R = 0$ |
| 18. Carrier Loop Thresh. Noise BW, dB Hz | 12.6 | 0.7 | $2B_{LO} = 18$ Hz ± 20% |
| 19. Carrier-to-Noise Pwr. Ratio, dB | 10.7 | 1.1 | RSS Tolerances |
| 20. Adverse CNR, dB | 9.6 | | |
| Telemetry Channel | | | |
| 21. Telemetry Modulation Loss, dB | 1.7 | 0.0 | $\beta_C = 1.8$ rad, $\beta_R = 0$ |
| 22. Required $E_b/N_0$, dB | 10.5 | 0.0 | |
| 23. Hardware Degradation, dB | 2.0 | 0.0 | |
| 24. Data Rate Capability, dB | 18.5 | 1.1 | RSS Tolerances [70.8 b/s] |
| 25. Adverse Data Rate Cap., dB | 17.4 | | [55.0 b/s] |

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Table 7.3-3

COMMUNICATIONS POWER BUDGET:

DOWNLINK, X-BAND, SC HGA TO DSN 64m

(At Maximum Range)

| Parameter | Nominal Value | Adverse Tolerance | Comments |
|---|---|---|---|
| 1. Frequency, MHz | 8415 | | |
| 2. Transmit Power, dBm | + 43.0 | 0.0 | 20W |
| 3. Transmit Loss, dB | 0.7 | 0.2 | SR1-141, WR-112 |
| 4. Antenna Gain, dBi | 36.9 | 0.4 | DSCS II Test Data |
| 5. Antenna Pointing Loss, dB | 0.1 | 1.1 | Pointing Error = 0.75° (3$\sigma$) |
| 6. EIRP, dBm | + 79.1 | 1.2 | RSS Tolerances |
| 7. Space Loss, dB | 282.9 | | 2.66 AU |
| 8. Atmospheric Loss, dB | 0.2 | 0.0 | |
| 9. Rec'd Isotropic Signal Power, dBmi | -204.0 | 1.2 | RSS Tolerances |
| 10. Antenna Gain, dBi | 70.6 | 0.6 | JPL 810-5, 10° Elev. |
| 11. Polarization Loss, dB | 0.1 | 0.1 | |
| 12. Antenna Pointing Loss, dB | 0.1 | 0.0 | |
| 13. Rec'r Input Signal Power, dBm | -133.6 | 1.3 | RSS Tolerances |
| 14. System Noise Temperature at Rec'r Input, °K | 43.0 | 5.0 | 10° Elev. |
| 15. Rec'r Input Noise Spectral Density, dBm/Hz | -182.3 | 0.5 | |
| 16. Rec'd Signal-to-Noise Spectral Density Ratio, dB Hz | 48.7 | 1.4 | RSS Tolerances |
| Carrier Channel | | | |
| 17. Carrier Modulation Loss, dB | 11.2 | 0.0 | $\beta_T$ = 1.3 rad, $\beta_R$ = 0 |
| 18. Carrier Loop Thresh. Noise BW, dB Hz | 10.8 | 0.0 | $2B_{LO}$ = 12 Hz |
| 19. Carrier-to-Noise Pwr. Ratio, dB | 26.7 | 1.4 | |
| 20. Adverse CNR, dB | 25.3 | | |
| Telemetry Channel | | | |
| 21. Telemetry Modulation Loss, dB | 0.3 | 0.0 | $\beta_T$ = 1.3 rad, $\beta_R$ = 0 |
| 22. Required $E_b/N_o$, dB | 4.4 | 0.0 | BER = $10^{-5}$, R = ½, K = 7 |
| 23. Hardware Degradation, dB | 1.2 | 0.2 | |
| 24. Data Rate Capability, dB | 42.8 | 1.4 | RSS Tolerances [19.1 kb/s] |
| 25. Adverse Data Rate Cap., dB | 41.4 | | [13.8 kb/s] |

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Table 7.3-4

COMMUNICATIONS POWER BUDGET:

DOWNLINK, X-BAND, SC MGA TO DSN 64m

(At Mars ORBIT Insertion)

| Parameter | Nominal Value | Adverse Tolerance | Comments |
|---|---|---|---|
| 1. Frequency, MHz | 8415 | | |
| 2. Transmit Power, dBm | + 43.0 | 0.0 | 20W |
| 3. Transmit Loss, dB | 0.7 | 0.2 | SR-141, WQ-112 |
| 4. Antenna Gain, dBi | + 4.5 | 0.0 | |
| 5. Antenna Pointing Loss, dB | 0.0 | 0.0 | |
| 6. EIRP, dBm | + 46.8 | 0.2 | RSS Tolerances |
| 7. Space Loss, dB | 276.2 | 0.0 | 1.23 AU |
| 8. Atmospheric Loss, dB | 0.2 | 0.0 | |
| 9. Rec'd Isotropic Signal Power, dBmi | -229.6 | 0.2 | RSS Tolerances |
| 10. Antenna Gain, dBi | 70.6 | 0.6 | JPL 810-5, 10° Elev. |
| 11. Polarization Loss, dB | 0.0 | 0.0 | |
| 12. Antenna Pointing Loss, dB | 0.1 | 0.0 | JPL 810-5 |
| 13. Rec'r Input Signal Power, dBm | -159.1 | 0.6 | RSS Tolerances |
| 14. System Noise Temperature at Rec'r Input, °K | 43.0 | 5.0 | |
| 15. Rec'r Input Noise Spectral Density, dBm/Hz | -182.3 | 0.5 | |
| 16. Rec'd Signal-to-Noise Spectral Density Ratio, dB Hz | 23.2 | 0.8 | RSS Tolerances |
| Carrier Channel | | | |
| 17. Carrier Modulation Loss, dB | 1.6 | 0.0 | $\beta_T \approx 1.0$ rad, $\beta_R = 0$ |
| 18. Carrier Loop Thresh. Noise BW, dB Hz | 10.8 | | $2B_{LO} = 12$ Hz |
| 19. Carrier-to-Noise Pwr. Ratio, dB | + 10.8 | 0.8 | |
| 20. Adverse CNR, dB | + 10.0 | | MRD PM-2000, Par. 8.6.3.2.1 |
| Telemetry Channel | | | |
| 21. Telemetry Modulation Loss, dB | 5.0 | 0.0 | $\beta_T \approx 1.0$ rad, $\beta_R = 0$ |
| 22. Required $E_b/N_0$, dB | 4.4 | 0.0 | BER = $10^{-5}$, R = ½, K = 7 |
| 23. Hareware Degradation, dB | 1.2 | 0.2 | |
| 24. Data Rate Capability, dB | 12.6 | 0.8 | RSS Tolerances [18.2 b/s] |
| 25. Adverse Data Rate Cap., dB | 11.8 | | [15.1 b/s] |

## 7.4 COMMAND AND CONTROL SUBSYSTEM

### 7.4.1 Subsystem Functions

The command and control subsystem (CCS) receives all command messages from the communications subsystem, validates commands, rejects anomalous commands, and transfers commands to the appropriate payload instruments and spacecraft subsystems according to time-tag information contained in each message. Commands are classified in two categories; 1) real-time commands for immediate execution, and 2) stored commands to be stored in the command buffer for subsequent transfer to their destination. Either mode can accommodate either quantitative (serial digital) and discrete (pulse) commands. Both real-time and stored command processing can be performed independently without timing conflict. Stored commands are used to execute spacecraft and science events in the absence of a ground link or can be initiated by on-board events. Command receipt verification and storage status are telemetered to the ground station via the data handling and communications subsystems.

### 7.4.2 Subsystem Requirements

The CCS is required to validate and execute commands initiated in the Mars Orbiter Mission Control Center and received from the spacecraft communications subsystem via a Deep Space Network (DSN) ground station. Table 7.4-1 identifies the type and number of commands for each science instrument and spacecraft subsystem required for the Climatology and Aeronomy missions. Command requirements for the four mission phases; I) Launch Phase, II) Interplanetary Cruise Phase, III) Mars Arrival/Instrument Deployment Phase, and IV) Nominal Orbital Mission Phase will be defined in the following paragraphs.

7.4.2.1 Launch Phase (L to L + 20 days). The CCS must be capable of storing and executing the upper-stage/spacecraft spin-up and motor firing command sequence. This sequence can be activated as a result of release from the shuttle bay and thereby will maintain all crew safety requirements imposed by the STS. In addition, the CCS must provide for commands to separate the spacecraft from the upper-stage, effect desired spin-up and nutation control, and configure the on-board communications subsystem for Deep Space Network (DSN) acquisition. Spacecraft checkout and command

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| Instrument | Discrete Commands | Serial Digital Commands |
|---|---|---|
| 1. Pressure Modulated Radiometer (PMR) | 4 | 1 |
| 2. Frost Infared Spectrometer (FIS) | 2 | 1 |
| 3. Gamma Ray Spectrometer (GRS) | 2 | 1 |
| 4. Ultra Violet Ozone ($UVO_3$) | 2 | 0 |
| 5. Ultra Violet Hydrogen Photometer (UHVP) | 2 | 0 |
| 6. Radar Altimeter (RA) | 2 | 1 |
| 7. Fabrey Perot Interferometer (FPI) | 2 | 1 |
| 8. Multi-Spectral Mapper (MSM) | 4 | 1 |
| Note: All Instruments on Despun Platform | | |
| **Subsystem** | | |
| 1. Command and Data Handling | 28 | 3 |
| 2. Communications | | |
| 3. Attitude Control | | |
| 4. Propulsion | 36 | |
| 5. Power | | |
| 6. Thermal Control | 8 | |

AERONOMY MISSION

| Instrument | | |
|---|---|---|
| 1. Neutral Mass Spectrometer (NMS) | 2 | 1 |
| 2. Thermal Ion Mass Spectrometer (TIMS) | 4 | 0 |
| 3. Electron Temperature Probe (ETP)* | 2 | 0 |
| 4. Retarding Potential Analyzer (RPA) | 2 | 1 |
| 5. Magnetometer (MAG)* | 4 | 1 |
| 6. Electric Field Detector (EFD)* | 2 | 0 |
| 7. Solar Wind Plasma Analyzer (SWPA)* | 2 | 1 |
| 8. Ultra Violet Spectrometer (UVS) | 3 | 0 |
| 9. Fabrey Perot Interferometer (FPI) | 2 | 1 |

* Located on spinning section, all other instruments despun.

Subsystem

Same as Climatology Mission unless otherwise noted.

Table 7.4-1

COMMAND LIST

loading prior to DSN acquisition can be accomplished via the shuttle/spacecraft interface while in the shuttle bay. There is no spacecraft to TDRSS link capability.

Spacecraft orientation is maintained by the Control Electronics Assembly (CEA) of the Attitude Control System (ACS). Commands to refine orientation and spin-rate can be sent in real-time from Mission Control to the CEA via the CCS as are thruster calibration sequences. For Trajectory Correction Maneuver parameters (TCM1 and 2) also can be stored in the CCS and executed by time tag information associated with each maneuver. There are no requirements to store orientation commands other than the TCMs.

7.4.2.2 Interplanetary Cruise Phase (L + 20 days to A - 15 days). Since the despun platform is tethered to the spinning section and booms are not deployed during interplanetary cruise, extensive instrument control is not anticipated during this phase of the mission; however, instrument turn on, checkout and calibration can be accomplished if desired. Any such instrument control will be accomplished by real-time commands.

7.4.2.3 Mars Arrival/Instrument Deployment Phase (A - 15 days to A + 5 days). The spacecraft CCS will be used to store and execute commands for TCM 3. Following TCM 3, commands to orient the spacecraft for Mars Orbit Insertion (MOI), MOI motor burn, and post-burn reorientation, are to be stored and executed according to mission plan. Communications will be minimal since as yet the HGA is not available. Round trip communications transit time tends to minimize any real-time interaction during this phase. Following successful attitude orientation, commands to despin the platform, deploy experiment booms, and possible instrument turn-ons may be stored and executed.

7.4.2.4 Nominal Orbital Mission Phase (A + 5 days to A + 687 days). During this phase, the CCS will be used to store and execute the required commands to configure science instruments into proper data taking modes, operate the spacecraft tape recorders, and configure spacecraft Data Handling and Communications subsystems to handle various bit rates dictated by scientific objectives and link capability. Sequence storage capacity should adequately accommodate a minimum of 32 hours of storage without a two-way communication link to the DSN. Spacecraft operation for one week without an uplink from the ground would be desirable.

7.4.2.5 The CCS is required to provide sufficient monitor telemetry to verify overall subsystem performance. Specifically, command validation, invalid command rejection, and executed command status are to be available as telemetered data for ground verification.

### 7.4.3 Subsystem Options

7.4.3.1 The tradeoffs considered for the CCS in the study involved evaluation of operations performed on-board the spacecraft versus those performed on the ground, assessment of how new technology could effectively reduce power, size, and risk while performing desired functions, and evaluation of existing or "off-the-shelf" equipment to meet mission requirements. Evaluation of the mission data requirements revealed that observation and pointing scenarios are cyclical for both the Climatology and Aeronomy missions once orbital operations are established. Since all pointing requirements are to be achieved via the ACS, through ground computation, these, plus instrument data modes, tape recorder speeds, and data handling subsystem modes will have to be commanded as functions of time. These events can be stored in the CCS for a minimum of 32 hours. A simple file can be maintained and updated on the ground to be uplinked to the spacecraft during desired command sessions.

7.4.3.2 New technology advancements were considered during the component selection process and weighed against currently available equipment. Since desired advanced components are not flight qualified, and in many cases require development, this option felt to be impractical. However, there is some degree of confidence that new technology will be developed and flight qualified by time component selection is finalized for a Mars Orbiter to be launched in the 1988 opportunity. Possibly, newer equipments may be smaller, require less power and provide greater capability than the current selection.

An in-depth evaluation of existing flight qualified equipments revealed that the Gulton Industries Command Decoder/Processor suggested in the study proposal meet all the requirements imposed for both the Climatology and Aeronomy missions, and proved to be the most satisfactory choice.

The only other choices were components from Litton and Sperry which did not offer flight proven components for the total Command and Data Handling Subsystems and the NASA Standard C&DH manufactured by Fairchild which was found to be unsuitable for the Mars Orbiter missions.

7.4.3.3 A command link from the shuttle to the spacecraft after deployment from the shuttle cargo bay may be required to activate the stored spin-up/ injection sequences if a separation trigger is not practical or fail-safe. This would be accomplished by the Communications Subsystem and does not add any additional requirements on the CCS.

### 7.4.4 Subsystem Description

The CCS utilizes the same processor selection for both the Climatology and Aeronomy missions. The Gulton Industries DSP-8024 CD/P Command Decoder/ Processor is a specialized version of a standard line. This equipment was utilized for the Solar Mesophere Explorer and satisfies the Mars Orbiter command requirements. Since this equipment is of modular design, specific requirements for Mars Orbiter applications can be implemented at minimal cost. The command processor remote interface unit is made up of standard* modules selected from the DSP-8000 family series. The RIU can be custom implemented specifically for Mars Orbiter command processing requirements. Figure 7.4-1 is a block diagram of the DSP-8000/8300 series command decoder/ processors and illustrates the major components of the subsystem.

7.4.4.1 DSP-8001 Command Message Processor. The Command Message Processor (CMP) is central to every DSP-8000 Series CD/P. It contains, as shown in Figure 7.4-1, (1) the command verification logic, (2) the process state sequencer and (3) the system power supply. The CMP is itself functionally modular. It is comprised of three distinct physical modules corresponding to the three functional areas described in the following module descriptions.

7.4.4.1.1 Command Verification Logic Module. This module performs three basic functions. It:

a. Provides interfacing for the input signals

b. Performs validation tests upon the incoming messages, including preamble sync detection and word length check

c. Stores each incoming message for command output/memory storage as well as for telemetry monitoring

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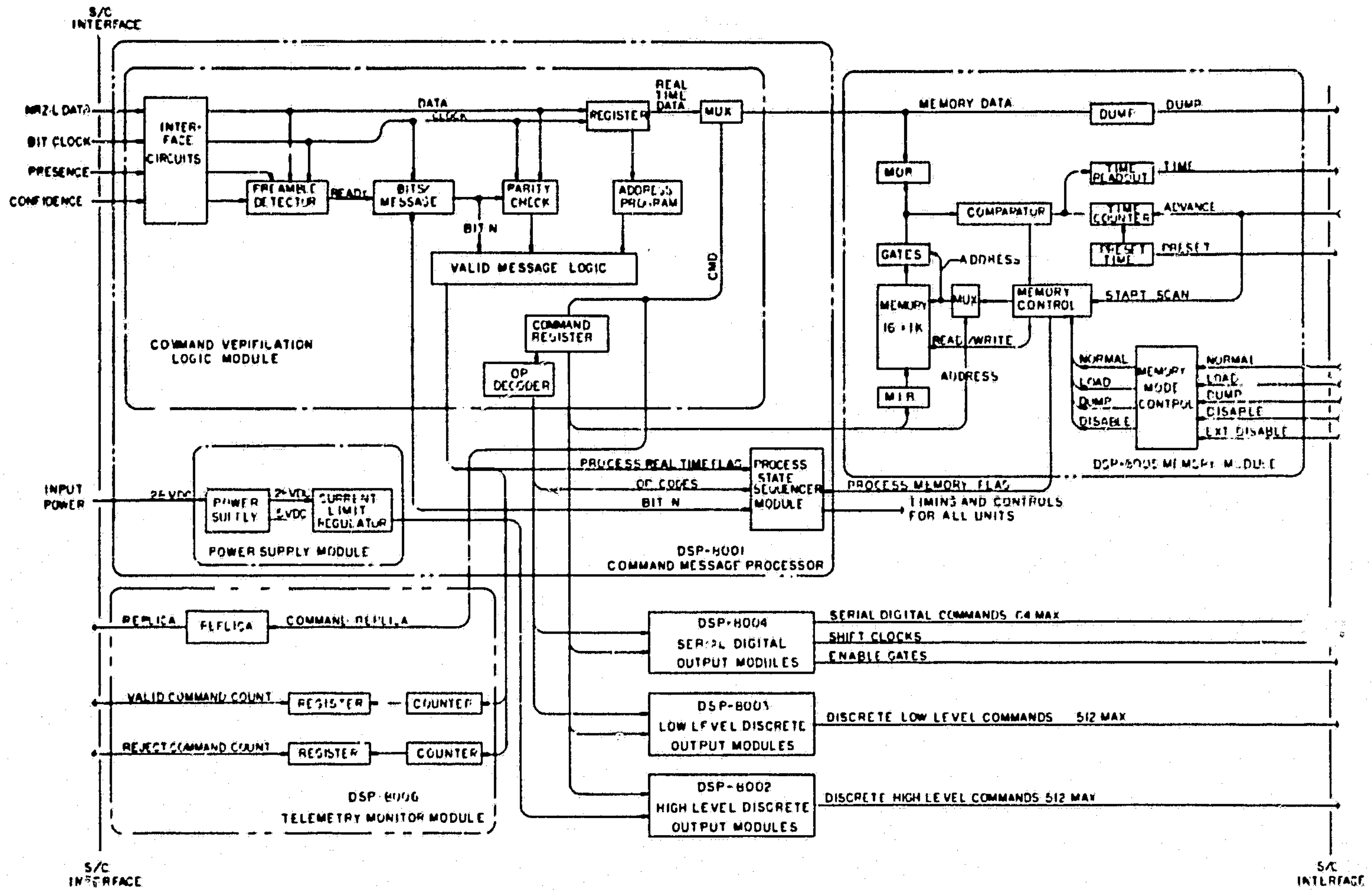


Figure 7.4-1

COMMAND AND CONTROL SUBSYSTEM BLOCK DIAGRAM

Compatibility with the NASA command message format is standard for any message length of up to 60 bits. A sample 60 bit NASA command message with 4 preamble bits is shown in Figure 7.4-2.

7.4.4.1.2 Process State Sequencer Module. This module provides central timing and control operations for the CD/P. It is the functional link between the incoming message detection/validation and the output commands, either real-time or delayed.

7.4.4.1.3 System Power Supply Module. This module provides the regulated power voltages required by the VD/P, limits command output circuit current and detects an undervoltage condition in order to prevent faulty system outputs and to maintain positive logic control. A key feature of the power system operation, which results in minimum overall system power consumption, is the implementation of a standby mode. This low power mode is in effect when no commands are being executed or received.

7.4.4.1.4 Command Execution Rate. Maximum bit rate - 2000 bps.

The command execution rate is variable with the incoming bit rate and the command message length.

7.4.4.1.5 Overall System Command Capacity. The number of commands which a DSP-8000 Series Command Decoder/Processor can receive and execute depends upon the number and type of command output modules which are included in each particular unit. However the ability of the CD/P to employ these output modules depends upon the incoming message format and the operations performed upon it by the DSP-8001 CMP. Thus the overall command capacity of the CD/P is a product of the command verification and control operations within the CMP in conjunction with the programmed decoding performed within each of the output modules.

| Feature | Characteristic |
|---|---|
| Discrete Command Capacity (High Level/Low Level) | Programmable to 32 unit increments to a maximum of 512 each |
| Serial Digital Command Capacity | Programmable in 16 unit increments to a maximum of 64 |

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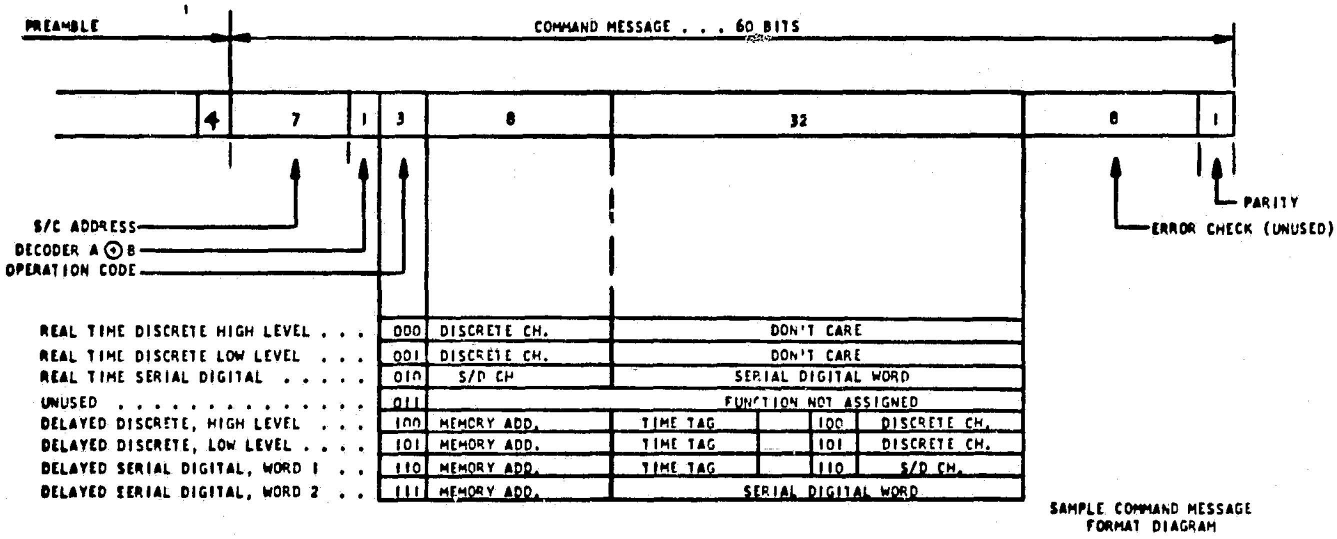


Note that a CD/P address selection bit allows the user to select between System A and System B in a redundant CD/P system configuration.

Figure 7.4-2

COMMAND MESSAGE

7.4.4.2 32-Channel High Level Discrete Output Module. The DSP-8000 High Level Discrete Output Module provides 32 separate high level discrete driver outputs, which are individually addressable via their associated address decoders. Each output supplies 210 mA of current at 28 Vdc to an external load, such as a relay. The output pulse duration is programmable via a gating signal generated in the DSP-8001 Command Message Processor.

The 32 discrete power gates are organized in a 2-level 4 x 8 matrix arrangement. The 1 of 32 output driver selection is accomplished with a 5-bit address code. The number of high level discrete output channels in the overall CD/P system is expandable in increments of 32 by the inclusion of additional DSP-8002 modules. Each of these modules carries its own unique jumper programmed address, which responds to a 4-bit address code at its input. The overall CD/P is therefore capable of executing 16 x 32 = 512 high level discrete output commands. It should be noted that due to the jumper programming on each module, any possible discrete command code up to a maximum of 512 can be decoded with no hardware design changes.

7.4.4.3 32-Channel Low Level Discrete Output Module. The DSP-8003 Low Level Discrete Output Module is configured in exactly the same 4 x 8 matrix arrangement as the DSP-8002. Low level discrete outputs are also expandable in 32 channel increments, via 1 of 16 jumper programming for module address, up to a maximum of 512 for a particular DSP-8000 Series CD/P system.

The basic difference between the DSP-8003 and the DSP-8002 is in the output drive characteristics. Each output from the DSP-8002 Low Level Discrete Output Module supplies 5 mA of current at 5.0 Vdc and sinks 2 mA of current at 0.4 Vdc and therefore dissipates much less power than the DSP-8002 module.

7.4.4.4 16-Channel Serial Digital Output Module

The DSP-8004 Serial Digital Output Module provides 16 separate serial digital data outputs, each with its own bit clock and enable gate outputs. The groups of three output lines each are individually addressable via their associated address decoders on the module. The serial data stream which is output is already contained in the input command message and is simply passed on through to the DSP-8004 data input lines. All output circuits supply 5 mA of current at 5.0 Vdc and sink 2 mA of current at 0.4 Vdc. Thus the DSP-8004, along with the DSP-8003, is a low power module.

The 1 of 16 output channel selection is accomplished with a 4-bit address code. The number of serial digital output channels in the overall CD/P system is expandable in increments of 16 by the inclusion of additional DSP-8004 modules. Each of these modules carries its own unique jumper programmed address. The overall CD/P is capable of executing 64 serial digital output commands. A serial digital control pulse, generated by the process state sequencer module in the DSP-8001, allows serial digital message commands to be distinguished from discrete message commands.

7.4.4.5 Memory Module. The DSP-8005 Memory Module operates in conjunction with the DSP-8001 Command Message Processor to provide for delayed command operation. Through the implementation of a RAM memory, a clock and a time comparator, it is possible to store up to 256 valid commands, each of which can be executed within up to 96 hours after their reception. A command message from memory is routed through the appropriate command output module in exactly the same manner as a real-time command.

Four types of mutually exclusive memory operating modes are provided.

DISABLE . . . No memory operations are permitted.

LOAD . . . . The command message is loaded into memory.

NORMAL . . . The CMP routes the delayed command message to the appropriate output module for execution.

DUMP . . . . The contents of memory plus memory address are output at the DUMP port.

The DSP-8005 module consists of two separate cards, one containing the memory and associated registers and the other containing the memory control. It should be noted that no hardware changes to the remainder of the CD/P system are required in order to incorporate the DSP-8005 Memory Module.

7.4.4.6 Telemetry Monitor Module. The DSP-8006 Telemetry Monitor module provides CD/P status information. Three types of status information are available in a standard module; (1) a replica of the latest executed command, (2) a count of the number of valid commands which the CD/P has executed, and (3) a count of the number of invalid commands which the CD/P has rejected.

| | | |
|---|---|---|
| REPLICA | . . . | Real time or delayed commands are stored in a special register for serial transfer to an external data device. |
| EXECUTE COUNT | . . . | An eight-bit counter accumulates the number of valid commands which have been executed since the CD/P power was applied. |
| REJECT COUNT | . . . | An eight-bit counter accumulates the number of invalid commands which have failed the validation tests since the CD/P power was applied. |

Transfer of all three types of data from the CD/P is controlled by externally supplied enable gate and bit clock signals. Data is available as serial digital outputs for external use.

7.4.4.7 Data Bus Interface Module. This module is not shown on Figure 7.4-1 and is not currently necessary to meet the Mars Orbiter requirements. Should the requirement arise, the module is easily added.

The DSP-8310 module contains all of the interface circuitry for receiving and transmitting to a party line bus system. Synchronization and bit by bit address and data verification is provided by a digital receiver which in turn performs a BIØ to NRZ conversion. It will output both serial and parallel data as well as a clock for the serial NRZ. This feature allows interface flexibility for numerous applications.

The DSP-8310 module employs a digital conversion technique rather than the traditional analog integration scheme. This initially will eliminate costly precision component parts as well as reduce lengthy calibration of the integrator time period. With the elimination of many discrete components, board area is minimized significantly, yielding a lighter more compact unit.

7.4.4.8 Command Equipments. Table 7.4-2 identifies the equipment complement for the complete Command and Data Handling subsystem. The Command Decoder Processor and Remote Interface Unit (first two items) are the command portion of the subsystem. (See Systems Block Diagram, Figure ).

COMMAND AND DATA HANDLING

EQUIPMENT LIST

| Component | Qty. | Unit Wt. kg | Total Wt. kg | Unit Pwr. Watts | Dimensions cm | Heritage | | |
|---|---|---|---|---|---|---|---|---|
| | | | | | | Program | Vendor | Status |
| Command Decoder Processor | 2 | 2.2 | 4.4 | 5.7 | 8.9 x 16.5 x 12.7 | TDRSS | Gulton | Purchased Component |
| Remote Interface Unit | 2 | 1.1 | 2.2 | 2.8 | 4.4 x 8 x 6.3 | TDRSS | Gulton | |
| Data Handling Processor | 2 | 3.85 | 7.7 | 10.5 | 8.9 x 26.6 x 12.7 | TDRSS | Gulton | |
| Remote Interface Unit | 2 | 1.9 | 3.8 | 5 | 4.4 x 10 x 6.3 | TDRSS | Gulton | |
| Tape Pecorder | 2 | 8.5 | 17 | 20.4 | 17.8 x 33 x 24 | ISPM | Odetics | |



Table 7.4-2

## 7.4.5 Subsystem Performance

The CCS performance capabilities are described in Table 7.4-3. This table compares the requirements as specified in Section 7.4.2 against the capabilities of the components described in Section 7.4.4.

| Function | Requirement | Performance |
|---|---|---|
| Instrument Cmds | Table 7.4-1 20 discr. 6 dig. | 7.4.4.1.5 64 dig. |
| Subsystem Cmds | Table 7.4-1 72 discr. 3 dig. | 512 ea. hi/lo discrete |
| Launch Sequence | 7.4.2.1 Undetermined | 7.4.4.5 256 x 64 bits |
| TCM Sequence | 7.4.2.1 | |
| Cruise Support | 7.4.2.2 | |
| Orbit Insertion | 7.4.2.3 | |
| Orbit Operations | 7.4.2.4 | |
| Monitor Telemetry | 7.4.2.5 | 7.4.4.6 Tlm. Mon. Module |
| Cmd. Validation | 7.4.2.5 | |
| Invalid Cmd. Reject | 7.4.2.5 | |
| Cmd. Recv. Rates | 7.8 bps & 125 bps | 7.8-1000 bps |
| Cmd. Process Rates | Undetermined | 7.4.4.1.4 2000 bps |
| Stored Cmd. Status | Memory Readout | 7.4.4.5 Memory Dump |
| Timing | | See Section 7.5 |

Table 7.4-3

COMMAND SUBSYSTEM PERFORMANCE

## 7.5 DATA HANDLING SUBSYSTEM

### 7.5.1 Subsystem Functions

The Data Handling Subsystem (DHS) gathers analog, bi-level, and serial digital data from payload instruments and spacecraft subsystems, formats the data and transfers the resultant bit stream to the spacecraft communications subsystem for downlinking to the DSN. Data formats selectable by command provide for transmission of engineering data only, engineering and science data, or memory read-out (contents of the command storage) data. Analog and bi-level discrete signals are converted to digital words. Variable data collection and transmission rates are available to accommodate link conditions throughout the mission. Formats are independent of the bit rate.

The DHS also provides storage for formatted engineering and science data for periods in the mission when a DSN station is unavailable. Stored data is played back during scheduled tracking passes and is interleaved with real-time spacecraft engineering and science instrument housekeeping data.

### 7.5.2 Subsystem Requirements

The DHS is required to collect data from each science instrument and spacecraft subsystem. Data types are analog, bi-level, and digital. The DHS must accept these various data types at various rates, format the data into data frames compatible with the ground data system, store data frames when required and transform the data frames into a serial bit stream to be transferred to the spacecraft Communications Subsystem for transmission to a DSN ground station. Table 7.5-1 identifies the data types received from the various instruments and spacecraft subsystems for Climatology and Aeronomy missions. Table 7.5-2 identifies the data rates for the science instruments during various mission phases. For the Climatology Mission, the 3712 bps requirement is an envelope of all instruments, not the requirement of any one payload complement.

7.5.2.1 Climatology Science Data Acquisition. Science data acquisition varies with day and night conditions on Mars. As shown on Table 7.5-2, four instruments (FIS, $UVO_3$, FPI and MSM) acquire data only when Mars is lighted (12 hours). The other instruments gather data continuously. Four formats are required for this mission; 1) Engineering only, 2) Night

| Instrument | Data Type: Analog | Digital | Bi-level |
|---|---|---|---|
| 1. Pressure Modulated Radiometer (PMR) | 0 | 1 | 2 |
| 2. Frost Infared Spectrometer (FIS) | 1 | 1 | 2 |
| 3. Gamma Ray Spectrometer (GRS) | 1 | 1 | 2 |
| 4. Ultra Violet Ozone ($UVO_3$) | 1 | 1 | 2 |
| 5. Ultra Violet Hydrogen Photometer (UVHP) | 1 | 1 | 2 |
| 6. Radar Altimeter (RA) | 1 | 1 | 2 |
| 7. Fabrey Perot Interferometer (FPI) | 0 | 1 | 4 |
| 8. Multi-Spectral Mapper (MSM) | 0 | 1 | 4 |
| Note: All instruments on despun platform | | | |
| **Subsystem** | | | |
| 1. Command and Data Handling | | 2 | 24 |
| 2. Communications | | | |
| 3. Attitude Control | | | |
| 4. Propulsion | 28 | | 20 |
| 5. Power | | | |
| 6. Thermal Control | 60 | | 8 |

AERONOMY MISSION

| Instrument | Analog | Digital | Bi-level |
|---|---|---|---|
| 1. Neutral Mass Spectrometer (NMS) | 0 | 1 | 3 |
| 2. Thermal Ion Mass Spectrometer (TIMS) | 2 | 1 | 3 |
| 3. Electron Temperature Probe (ETP)* | 2 | 1 | 3 |
| 4. Retarding Potential Analyzer (RPA) | 0 | 1 | 4 |
| 5. Magnetometer (MAG)* | 1 | 1 | 2 |
| 6. Electric Field Detector (EFD)* | 4 | 0 | 2 |
| 7. Solar Wind Plasma Analyzer (SWPA)* | 2 | 1 | 2 |
| 8. Ultra Violet Spectrometer (UVS) | 5 | 1 | 2 |
| 9. Fabrey Perot Interferometer (FPI) | 0 | 1 | 3 |

* Located on spinning section, all other instruments despun.

Subsystem

Same as Climatology Mission unless otherwise noted.

Table 7.5-1

HOUSEKEEPING AND ENGINEERING TELEMETRY LIST

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## SCIENCE DATA RATES

### CLIMATOLOGY MISSION
### (All Instruments are Despun)

| | | Rate bps | Special Mapping |
|---|---|---|---|
| 1. | Pressure Modulated Radiometer (PMR) | 140 | 140 |
| 2. | Frost Infared Spectrometer (FIS)* | 120 | 120 |
| 3. | Gamma Ray Spectrometer (GRS) | 1024 | 8000 |
| 4. | Ultra Violet Ozone ($UVO_3$)* | 64 | 64 |
| 5. | Ultra Violet Hydrogen Photometer (UVHP) | 8 | 8 |
| 6. | Radar Altimeter (RA) | 100 | 100 |
| 7. | Fabrey Perot Interferometer (FPI)* | 256 | 256 |
| 8. | Multi-Spectral Mapper (MSM)* | 2000 | 12000 |
| | Special Mapping (Not included in current options) | | |
| | TOTAL | 3712 | 20688 |

### AERONOMY MISSION

| | | Data Rates (bps)/Orbit Phase | | |
|---|---|---|---|---|
| | | Apoapsis | P+60 min. | P+15 min. |
| 1. | Neutral Mass Spectrometer (NMS) | 0 | 0 | 256 |
| 2. | Thermal Ion Mass Spectrometer (TIMS) | 0 | 128 | 256 |
| 3. | Electron Temperature Probe (ETP)** | 64 | 128 | 256 |
| 4. | Retarding Potential Analyzer (RPA) | 64 | 128 | 512 |
| 5. | Magnetometer (MAG)** | 128 | 128 | 128 |
| 6. | Electric Field Detector (EFD)** | 128 | 128 | 128 |
| 7. | Solar Wind Plasma Analyzer (SWPA)** | 128 | 128 | 128 |
| 8. | Ultra Violet Spectrometer (UVS) | 0 | 128 | 128 |
| 9. | Fabrey Perot Interferometer (FPI) | 0 | 128 | 256 |
| | TOTAL | 512 | 1024 | 2048 |
| | Non-Science Data (Sync, time, ID, Engr, HK) | TBD | 90 min.<br>360 min.<br>6 | 30 min.<br>120 min.<br>2 |

* Operate during MARS day only
** Spinning instruments

Table 7.5-2

Science and Engineering, 3) Day Science and Engineering, and 4) Command Storage Readout.

7.5.2.2 Aeronomy Science Data Acquisition. Science data acquisition varies with time from Mars periapsis. As shown on Table 7.5-2, most instruments have varying rates at different orbital positions. Five formats are required for this mission; 1) Engineering only, 2) Apoapsis Science, 3) P $\pm$ 90 minutes Science, 4) P $\pm$ 15 minutes Science, and 5) Command Storage Readout.

7.5.2.3 Climatology Science Data Storage. All Climatology science data is to be stored on the spacecraft and played back during scheduled DSN tracking periods. Data accumulation rates vary with day and night observation periods as shown on Table 7.5-2. Data playback rates vary with the available link capabilities throughout the mission. Planned tracking coverage is 8 hours per day throughout the science mission. A maximum tracking gap of 32 hours should be acceptable without loss of science data. There is no requirement for real-time science data return.

7.5.2.4 Aeronomy Science Data Storage. All Aeronomy science data is to be stored on the spacecraft and played back during scheduled DSN tracking periods. Data accumulation rates vary as a function of time from periapsis as shown on Table 7.5-2. All other requirements are the same as for the Climatology mission.

7.5.2.5 Spacecraft Timing. The timing clock for the spacecraft system, which is also used for the timing of all science instruments and spacecraft subsystems, is provided by the DHS. Clock stability and update requirements are TBD.

Science and engineering data are to be formatted into standard frames by the DHS. The frames and formats utilized must be compatible with DSN and mission control center processing capabilities.

The DHS must have sufficient monitor telemetry to provide ground operations with status and alarm information. There is no requirement for on-board failure recovery software.

### 7.5.3 Subsystem Options

The tradeoffs considering alternate approaches to the DHS design recommended in the study proposal involved the evaluation of the payload instrument set and spacecraft subsystem capabilities for each mission. Simplicity and low cost were stressed for this study.

7.5.3.1 The system architecture selected makes use of a main DHS on the spinning portion of the spacecraft and a remote DHS (RIU) on the despun platform. The subsystem employs Time Division Multiplexed data instead of packet telemetry in order to minimize user interface costs.

A new scheme to merge real-time engineering and housekeeping data with playback data is proposed as a result of the study. A multiplexer was eliminated and development of dual subcarriers was ruled out as expensive on the spacecraft design and required special ground capabilities. The scheme is described in the performance Section 7.5.5.2.

7.5.3.2 Tradeoffs made during the study indicate that a tape recorder is superior to bubble memories on semi-conductor storage units for storage and readout of Mars Orbiter data. Cost and flight history were on the side of recorders. Synchronous data transfer was selected over asynchronous rate buffering schemes which either complicate the spacecraft design or the ground data system design. All data return was chosen to be Last In First Out (LIFO). This simplifies operations and minimizes impacts of rescheduled playback passes.

7.5.3.3 As was the case with the Command and Control Subsystem, the Gulton Data Handling Subsystem was favored to be the most cost effective capability available which met both Climatology and Aeronomy mission requirements. The Odetics tape recorder developed for ISPM but modified for Mars Orbiter provides a most cost effective and reliable on-board data storage approach. Costs will be somewhat less than ISPM since only four playback rates are necessary.

### 7.5.4 Subsystem Description

The DHS utilizes the same processor selection for both the Climatology and Aeronomy missions. The Gulton series of equipment selected in flight qualified and of modular design. Expansion is easily accomplished at minimum cost. The data handling remote interface unit is made up of standard modules of the DSP-7000 series and can be custom implemented specifically for Mars Orbiter data handling requirements.

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Figure 7.5-1 is a block diagram of the DSP-7000/7300 series PCM System and illustrates the major components of the DHS.

7.5.4.1 The Central Processor Unit (CPU) is central to every DSP-7000 Series PCM. A CPU plus one or more add-on multiplexer modules comprises a functioning PCM system. The CPU contains, as shown in the General Block Diagram, (1) the word length/code generator control, (2) the PROM format control, (3) the sample/hold and ADC, (4) the premodulation filter output and (5) the system power supply. It can also be seen from the block diagram that the CPU is itself functionally modular. That is, the CPU is comprised of five distinct physical modules corresponding to the five functional areas listed above. The operation of these five CPU modules is described in the following paragraphs:

7.5.4.1.1 Word Length/Code Generator Control Module. This module generates all of the basic timing and controls for the CPU and other users. Its functions, which are listed below, are all programmable with the result that custom requirements are provided as standard features in the DSP-7000 Series PCM. Characteristics of this module are as follows:

| Feature | Characteristics |
|---|---|
| Oscillator | Crystal Controlled |
| Bit Rate | Jumper Programmable up to 600 kHz |
| Clock Selection | Automatic Internal or External |
| Word Length | PROM Programmable 6 to 12 bits |
| Sample/Hold | PROM Programmable portion of a Word Time |
| Sync Word Length | PROM Programmable up to 4n bits (n = bits per word) |
| Frame ID Count Length | PROM Programmable, 0 to 255 minor frames |
| Output Data Combiner | Combines ADC, digital, sync and ID data |
| Parity | Jumper Programmable, Odd, even or no parity |
| Output Codes | Jumper Programmable, NRZ-L, Bi-Ø |

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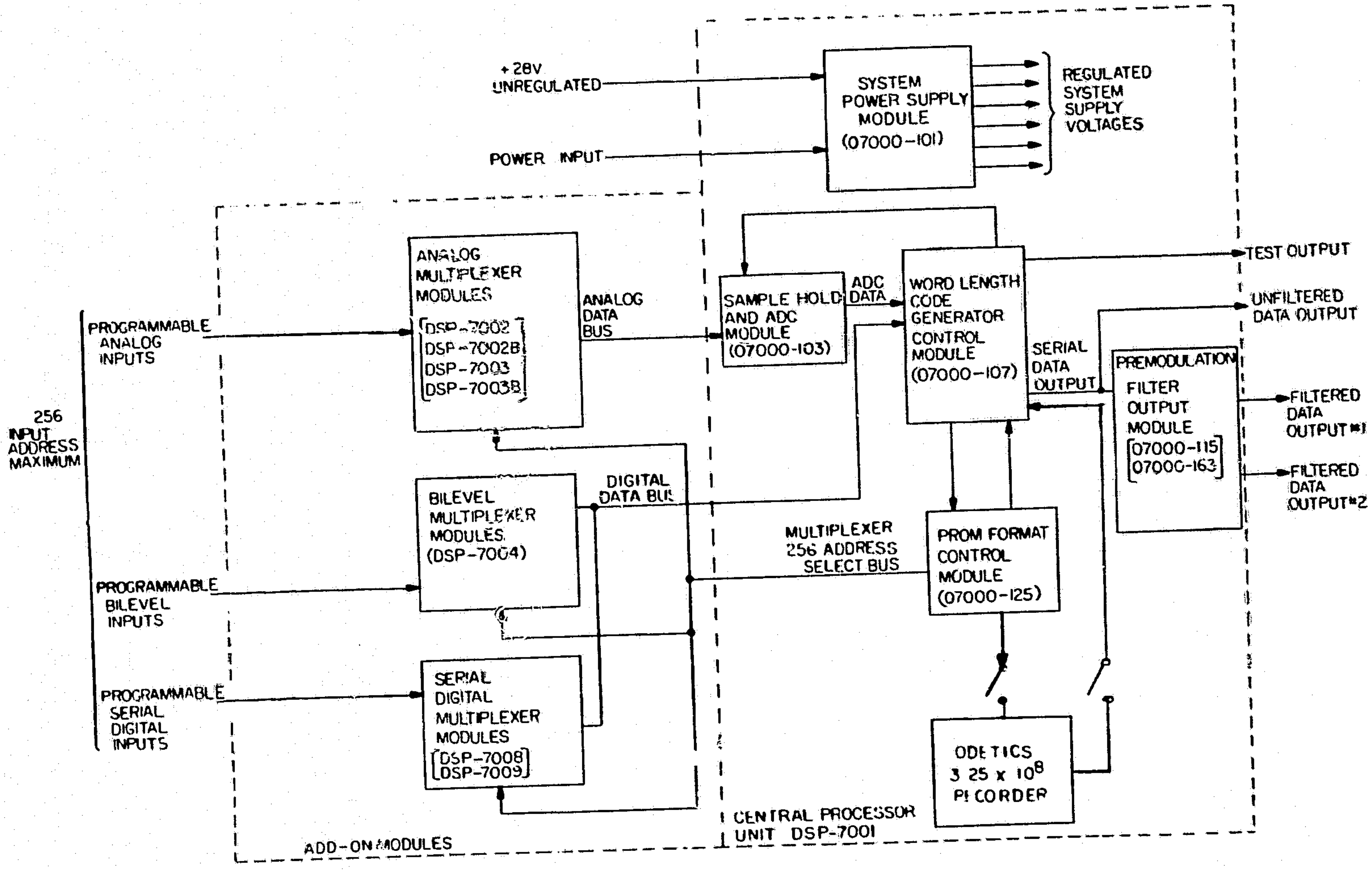



DATA HANDLING SUBSYSTEM BLOCK DIAGRAM

Figure 7.5-1

7.5.4.1.2 PROM Format Control Module.

This module formats the sampling and readout of all PCM data inputs. Because this formatting is completely under programmable PROM control, almost any format configuration within the PCM data input capacity is possible.

The format control circuitry is organized in two levels of programmable PROM selection designated as Master PROM and Slave PROM controls. As a result of this technique very large major frame formats, incorporating a high degree of super commutation and subcommutation flexibility, may be generated with a relatively small actual PROM capacity.

In the General Block Diagram, it can be seen that a single CPU can incorporate up to N = 4 PROM format control modules. Each module can format up to 256 data addresses. Where more than 256 data address points are sampled, additional modules must be added. Thus a PCM with four PROM formal control modules can sample and format up to a maximum of 1024 data address points. When N > 1, the additional format control modules are slaved to Module #1 via the Word Select signal. These additional modules do not expand the maximum major frame capability, which is 512 words/minor frame by 256 minor frames/major frame. They simply enable additional input data points to be sampled by the PCM. Characteristics of this module are as follows:

C-3

| Feature | Characteristics |
| --- | --- |
| Minor Frame Length | 512 Words Maximum |
| Major Frame Length | 256 Minor Frames Maximum |
| Major Frame Sync | 3 Fixed Words, plus ID Counter |
| ID Counter | 256 Words Maximum |
| Supercommutation | Controlled by Slave PROM 1<br>Maximum Rate = Word Rate |
| Supercommutation | Controlled by Slave PROMs 2, 3, 4<br>Maximum of 256 deep |
| Format | Random Sampling |
| Minor Frame Reset | Controlled by Master PROM |
| Major Frame Reset | Controlled by Slave PROM 4 |

Figure 7.5-2 illustrates a sample format.

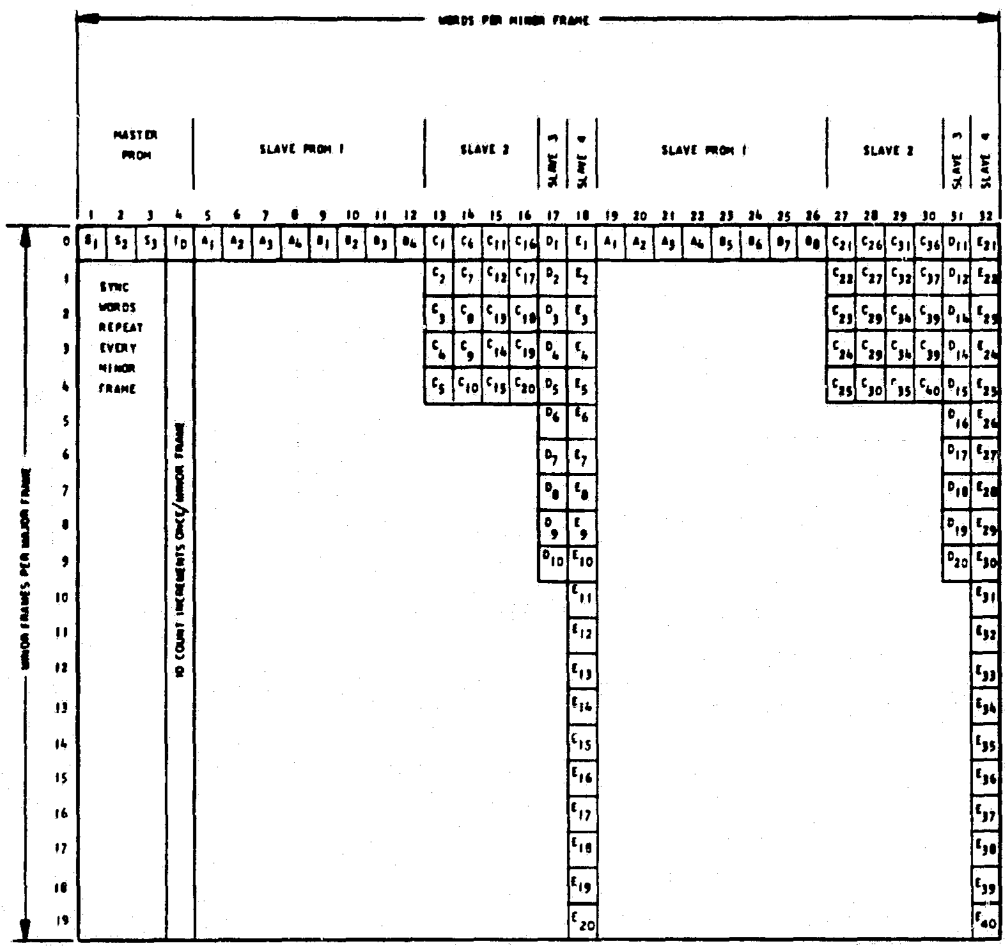
7.5.4.1.3 Sample/Hold and ADC Module. This module converts the multiplexed analog data into its binary representation, which is then output in serial digital form. It performs the three following basic functions.

a. Reduces common mode and ground loop errors to negligible levels. Also provides isolation of the input signal common returns from all system grounds.

b. Provides a sampling period during which sampling transients can settle out prior to the A/D conversion. Also provides a constant voltage to the ADC during the hold mode when the A/D conversion is performed.

c. Provides, as an output, an N-bit digital code (programmable from N = 6 to N = 12) representative of the amplitude value of the analog signal to a resolution of 1 past in $2^{N-1}$.

A sample frame format is included as a visual aid for clarifying DSP-7000 PCM formatting and to define the commonly used terms in a PCM format.




DATA LIST

| TYPE | QUANTITY | SAMPLE RATE | FORMAT TERM |
|---|---|---|---|
| S1 = 00000100 | 1 | 100 SPS | SYNC WORD 1 = 00000100 |
| S2 = 11001111 | 1 | 100 SPS | SYNC WORD 2 = 11001111 |
| S3 = 01011111 | 1 | 100 SPS | SYNC WORD 3 = 01011111 |
| ID = 0 - 19 | 1 | 100 SPS | ID COUNTER = 0 - 19 |
| A = SEHL | 4 | 200 SPS | SUPERCOMMUTATED PRIME WORD |
| B = DELL | 8 | 100 SPS | MINOR FRAME PRIME WORD |
| C = DEHL | 40 | 20 SPS | SUBCOMMUTATED WORDS |
| D = SEHL | 20 | 10 SPS | SUBCOMMUTATED WORDS |
| E = BL | 40 | 5 SPS | SUBCOMMUTATED WORDS |

MINOR FRAME RATE = 100 SPS

$$\text{MAJOR FRAME RATE} = \frac{\text{MINOR FRAME RATE}}{\text{MINOR FRAMES PER MAJOR FRAME}} = \frac{100 \text{ SPS}}{20} = 5 \text{ SPS}$$

WORD RATE = 100 FRAME PER SECOND X 32 WORDS PER FRAME = 3200 WPS

BIT RATE = 3200 WPS WORD RATE X 8 BITS PER WORD = 25.6 KBS

SAMPLE FORMAT

Figure 7.5-2

7.5.4.1.4 Premodulation Filter Output Module. This module uses low frequency active or high frequency passive filters to limit the spectral content of the output bit stream when the PCM output data is required to modulate an RF transmitter. Since this function is not always required, this module is optional in the DSP-7000 CPU.

7.5.4.1.5 System Power Supply Module. This module operates from an unregulated 28 Vdc power source and internally generates all necessary regulated voltages required by the overall PCM system through the implementation of a high efficiency switching pre-regulator and a non-saturating dc/dc converter.

7.5.4.2 High/Low-Level Analog Multiplexer Module. The DSP-7002 and DSP-7002B modules contain variations on the same basic multiplexer circuitry which multiplexes or "commutates" input analog signals onto a single output PAM bus to the ADC. The DSP-7002 module is a high-level analog multiplexer which can be configured to accept either 64 SEHL inputs or 32 DELL inputs. The DSP-7002B module is a low-level analog multiplexer configured to accept 32 DELL inputs. In this configuration, the multiplexer output is routed to a low-level amplifier on a DSP-7003B module prior to its inclusion on the PAM bus to the ADC.

In addition to the analog switches and switch drivers, these modules contain decode logic for translating the PROM format control address code into the selection of the analog input channel being sampled. Thus the input channel selected is actually programmed by the PROM format control. This is true for all DSP-7000 Series multiplexer modules.

7.5.4.3 Low-Level Analog Multiplexer with Amplifier Module. The DSP-7003 module contains a 16 channel low-level analog multiplexer with an accompanying low-level amplifier, whereas the DSP-7003B module contains only the low-level amplifier. This arrangement maximuzes low-level analog multiplexer flexibility. A LL amplifier is never configured with more than 64 DELL channels. Thus it is possible to configure a 16, 32, 48 or 64 channel multiplexer with a single LL amplifier through the judicious selection of DSP-7003, DSP-7003B and DSP-7002B modules. The LL amplifier output is routed onto the PAM bus to the ADC.

This module features an optional pre-selection technique which enables the low-level multiplexer/amplifier system to dwell on each input channel for several word times prior to sampling. Pre-selection is employed in high speed systems where the limiting factor on accuracy is usually the settling of the low-level analog inputs through the multiplexer and first stage amplifier. Thus this technique increases the settle time so that the desired accuracy can be maintained.

7.5.4.4 Bi-level Multiplexer Module. The DSP-7004 module multiplexer bilevel, or discrete, data inputs onto the system digital data bus. This operation is performed with a 64 point analog type multiplexer organized into an 8 x 8 matrix. Up to 8 discrete inputs are sampled in parallel and grouped into a single serial digital output word in which each input is represented by a single bi-level bit. A single DSP-7004 module can sample 8 such discrete input groups (i.e., 8 words of data).

Through the employment of analog type multiplexer switches and parallel precision voltage comparators highly accurate threshold level detection is obtained. It should be noted that all 64 inputs to a single module are completely independent of each other. The only constraint is imposed by threshold level requirements. There can be one threshold level (for all 64 channels) or two threshold levels (one for each grouping of 32 channels) on a module.

7.5.4.5 Quad Premodulation Filter Output Module. The DSP-7005 module contains four (4) low-lass filters which limit the spectral content of the output bit stream when the PCM output data is required to modulate an RF transmitter. This module is employed as an alternate to the dual premodulation filter module which is standard in the CPU when the capability for more than two output filters is required. This situation often arises in order to accommodate a system which requires bit change capability with no hardware modification. The power buffer output included in the standard CPU filter module is omitted in the DSP-7005 filter module in order to enable it to contain the four filter circuits.

7.5.4.6 Built-in Test Module. The DSP-7006 BITE module provides a quick and easy means of verifying that the in-flight operation of a DSP-7000 Series PCM unit is correct. Should there be a malfunction, this module will indicate which general area of the PCM system is malfunctioning.

This built-in monitor system employs eight (8) detection circuits whose logic level outputs can be sampled as eight (8) distinct bi-level inputs to the PCM or simply treated as test outputs. The significance of this approach is that the critical PCM system determinants, which themselves do not exist as simple logic levels, are detected and translated into logic levels which can be monitored as simple logic levels. These eight outputs are designated as Bit 1 through Bit 8, indicated as follows:

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| Bit No. | Indication |
|---|---|
| 1 | BITE "OR" - Overall PCM system operation check |
| 2 | Bits per word check |
| 3 | Words per minor frame check |
| 4 | Minor frames per major frame check |
| 5 | Sync pattern check |
| 6 | A/D conversion check |
| 7 | Bi-level high threshold check |
| 8 | Bi-level low threshold check |

7.5.4.7 Data Bus Interface Module. This module is not shown on Figure 7.5-1 and is not currently necessary to meet the Mars Orbiter requirements. Should the requirement arise, the module is easily added.

The DSP-7310 module contains all of the interface circuitry for receiving and transmitting to a party line bus system. Synchronization and bit by bit address and data verification is provided by a digital receiver which in turn performs a BIØ to NRZ conversion. It will output both serial and parallel data as well as a clock for the serial NRZ. This feature allows interface flexibility for numerous applications.

Another feature that yields flexibility is the transmission bit rate. This module is capable of acquiring real-time data from remote terminals and servicing telemetry control systems with party line bit rates between 200K to 3 Meg depending on terminal loading.

The DSP-7310 module employs a digital conversion technique rather than the traditional analog integration scheme. This initially will eliminate costly precision component parts as well as reduce lengthy calibration of the integrator time period. With the elimination of many discrete components, board area is minimized significantly, yielding a lighter more compact unit.

7.5.4.8 The DSP-7008 module multiplexes eight (8) independent serial digital data inputs onto the system digital data bus. Synchronous operation is obtained through the implementation of eight (8) sets of enable gates and PCM bit rate clocks generated by the PCM timing circuits and routed back to the customer data source to regulate the data input timing. As a result of this control operation, data is always input to the PCM synchronously and coincidently with its location in the PCM output data format. Thus the data is passed right on through the DSP-7008 multiplexer module via the system digital data bus to the PCM output data combiner. The number of bits in each data input is determined solely by each individual enable gate duration, which is always an integral number of PCM word lengths.

7.5.4.9 The DSP-7009 module multiplexes four (4) independent serial digital data inputs onto the system digital data bus. Each data input if acquired and stored asynchronously with its output slot in the PCM output data format. The interval for data acquisition is independent for each input and is established by a signal generated when the data register in question is emptied of its contents.

As a result of this asynchronous operation, which imposes no restrictions as to when the customer may generate his data, data storage is required. Thus the number of bits allowed per data input is determined by the DSP-7009 module storage capacity, which is independently set for each data input at either one or two 8-bit groups per PCM word. This data length flexibility is a valuable asset where there is considerable disparity in the nature of the various digital data sources being sampled by the PCM.

7.5.4.10 The Tape Recorder is an Odetics $3.25 \times 10^8$ recorder developed for the ISPM Project. The Mars Orbiter recorder will be the same version except have a standard maximum of 4 playback rates.

7.5.4.11 Data Handling Equipment. Table 7.4-2 (previous section) identifies the equipment complement for a complete Command and Data Handling subsystem. The Data Handling Processor, Remote Interface Unit, and Tape Recorder (last three items) are the data handling portion of the subsystem. All modules previously described are included in the processor and remote interface equipments. (See System Block Diagram, Figure 6.1-3.)

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### 7.5.5 Subsystem Performance

The DHS performance capabilities are described in Table 7.5-3. This table compares the requirements as specified in Section 7.5.2 against the capabilities of the components described in Section 7.5.4.

| Function | Requirement | Performance |
|---|---|---|
| Instr. Data Types | Table 7.5-1 5 or 16A, 8D, 20 or 24B | 256 total ADB, 7.5.4.2, 3, 4, 8, and 19 |
| Subsystem Data Types | Table 7.5-1 88A, 3D, 52B | 256 total ADB, 7.5.4.2, 3, 4, 8, and 19 |
| Instr. Data Rates | Table 7.5-2 3712 bps max | 600 KHz 7.5.4.1.1 |
| Data Formats | 7.5.2.1 Four | Four available 7.5.4.1.2 |
| Data Formats | 7.5.2.2 Five | Four available 7.5.4.1.2 (minus 1) |
| Data Storage | 7.5.2.3 24 hrs. = 215.3 megabits | 325 megabits 7.5.4.10 |
| Data Storage | 7.5.2.4 24 hrs. = 66.4 megabits | 325 megabits 7.5.4.10 |
| S/C Timing | 7.5.2.5 TBD stability | TBD Crystal 7.5.4.1.1 |
| Data Frame Format | 7.5.2.5 DSN/GDS compatible | Table 7.5-1 |
| Monitor Tlm. | 7.5.2.5 Status and Alarm data | Built in Test Module 7.5.4.6 |
| Comm. Compatibility | | Output Module 7.5.4.5 |

Table 7.5-3

DATA HANDLING SUBSYSTEM PERFORMANCE

7.5.5.1 For the Aeronomy Mission, a deficiency of one stored format capability exists with standard equipment. Stored format capability can be increased but cost is unknown. An alternate approach would be to combine the Engineering only format with the Apoapsis science format.

7.5.5.2 It is recommended that a standard frame size of 1152 bits be utilized for both missions. Each frame would contain 1024 bits of science data and 128 bits of non-science. The non-science is broken down as follows:

| | |
|---|---|
| Synch bits | 24 |
| I.D. | 8 |
| Time tag | 32 |
| Engineering | 16 |
| Science HK | 8 |
| Fill data | 40* |
| Total | 128 |

The 40 bits of fill are replaced by real-time data during Recorder playbacks. The DHS can accomplish this insertion of data in the fill slot. The 40 bits are comprised of 16 bits engineering, 8 bits housekeeping, and reduced time resolution of 16 bits.

7.5.5.3 Tables 7.5-4 and 7.5-5 reflect data taking possibilities for the Climatology and Aeronomy missions respectively. The Climatology shows how the baseline and all optional payload complements can be accommodated. For Option C3, a number of data acquisition bit rates have been implemented, one for night (when the MSM is not producing data) and four for day, assuming the MSM will produce data at four possible rates within the indicated range: 1,500, 3,000, 6,000, and 12,000 b/s. As long as the MSM observes its limit of 1,000 b/s, averaged over 24 hours, the selected tape recorder capacity and playback/downlink data rates will serve. A tape recorder record rate and a science data format -- allocating the 1,024 science bits per frame among the scientific instruments -- must be selected for each MSM data acquisition rate, but there is nothing sacred about the above assumed rates. By the same token, a higher acquisition rate, including MSM data at 12,000 b/s and GRS data at 8,000 b/s, can similarly by implemented for a "special mapping" mode. It only affects the science format and record rate for the time when this mode is active.

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Table 7.5-4

DATA PERFORMANCE SUMMARY - CLIMATOLOGY

| | Climatology Mission Option | | | | |
|---|---|---|---|---|---|
| | Baseline | C1 | C2 | C3 | |
| Basis | Day | Day | Day | Actual | |
| Science Data Acquisition Rate (b/s) (See Table 6.3-6) | 1284 | 1456 | 1540 | 13264 | D4* |
| | | | | 7264 | D3 |
| | | | | 4264 | D2 |
| | | | | 2764 | D1 |
| | | | | 1264 | N |
| Total Data Acquisition Rate (Science x 9/8) (b/s) | 1444.5 | 1638 | 1732.5 | 14922 | D4* |
| | | | | 8172 | D3 |
| | | | | 4797 | D2 |
| | | | | 3109.5 | D1 |
| | | | | 1422 | N |
| | | | | 2547 | AVG |
| Recorder Capacity (Mb) | 325 | 325 | 325 | 325 | |
| <u>In 32 Hours:</u> | | | | | |
| Data Stored (Mb) | 166.4 | 188.7 | 199.6 | 293.4 | |
| Margin (Mb) | 158.6 | 136.3 | 125.4 | 31.6 | |
| <u>In 24 Hours:</u> | | | | | |
| Data Stored (Mb) | 124.8 | 141.5 | 149.7 | 220.1 | |
| Margin (Mb) | 200.2 | 183.5 | 175.3 | 104.9 | |

* For "special mapping" mode involving GRS + MSM at 20,000 b/s, rate D4 can be increased.

Table 7.5-5

DATA PERFORMANCE SUMMARY - AERONOMY

| | Orbital Phase | | |
|---|---|---|---|
| | Ionosphere | Ionosheath | Apoapsis |
| Hours/Orbit | 0.5 | 1.5 | 4.689 |
| Science Data Acquisition Rate (b/s) (See Table 6.3-7) | 2048 | 1024 | 512 |
| Total Data Acquisition Rate (Science x 9/8) (b/s) | 2304 | 1152 | 576 |
| Orbital Average (Mb), 32 Hours | | 97.5 | |
| Recorder Capacity (Mb) | | 325 | |
| Margin (Mb) | | 227.5 | |

For both the Climatology and Aeronomy Missions there is considerable margin within the tape recorder capacity, particularly for the majority of instances when the DSN coverage cycle is 8 hours tracking, 16 hours non-tracking. This margin can be devoted to one or more of the following uses:

1. To take more high-rate burst data (Aeronomy Mission, and Climatology C3).
2. To go more than one day without a playback period.
3. To permit less than 8-hour downlink tracking periods.

## 7.6 THERMAL CONTROL SUBSYSTEM

### 7.6.1 Subsystem Functions

The functions of the Thermal Control Subsystem (TCS) are:

- to maintain the various instruments and spacecraft equipments at acceptable temperatures throughout the Climatology and Aeronomy Missions (i.e., pre-launch, launch/shuttle abort, interplanetary cruise, and orbital operation).
- to maintain acceptable thermal interfaces with the STS and launch vehicle upper stage.

### 7.6.2 Subsystem Requirements

The following lists the on-orbit and interplanetary cruise orbital parameter requirements, for which the baseline TCS is designed to meet:

- Interplanetary Cruise
  1. Distance from Sun: 1 AU to 1.53 AU
  2. Solar vector with respect to spacecraft equatorial plane: +30 to -30 degrees
- Climatology Mission
  1. 300 km altitude circular and sun-synchronous orbit
  2. Orbit time: 1.89 hours (0.7 hours maximum eclipse)
  3. Distance from Sun: 1.37 AU to 1.67 AU
  4. Solar vector with respect to orbit plane: 22 to +45 degrees
- Aeronomy Mission
  1. 150 km x $3R_m$ altitude elliptical orbit
  2. Orbit time: 6.68 hours (1.67 hours maximum eclipse)
  3. Distance from Sun: 1.38 AU to 1.67 AU
  4. Solar vector with respect to orbit plane: 0 to +90 (or +110) degrees

Storage and operating temperature requirements for the instruments and spacecraft equipments are listed in Figure 7.6-1. Acceptable thermal interface with the launch vehicle will be defined jointly with the launch vehicle contractor. This interface requirement normally minimizes thermal interactions between the spacecraft and launch vehicle, and can usually be

Figure 7.6-1

MARS ORBITER TEMPERATURE REQUIREMENTS

| EQUIPMENT | TEMPERATURE LIMITS (°C) | |
|---|---|---|
| | POWER OFF (min/max) | OPERATION (min/max) |
| COMMUNICATION SUBSYSTEM | | |
| 20 W. TWTA | -35/50 | 5/70 |
| S/X-BAND TRANSPONDER | | 0/50 |
| OTHERS | | 0/45 |
| C & DH SUBSYSTEM | | |
| TAPE RECORDER | | -10/45 |
| OTHERS | | 0/45 |
| ALTITUDE CONTROL SUBSYSTEM | | |
| DMA | | 0/50 |
| AZ/EL MOTORS | | 0/80 |
| OTHERS | | 0/45 |
| ELECTRIC POWER SUBSYSTEM | | |
| BATTERIES | | 0/10 |
| CONVERTERS | | -10/50 |
| OTHERS | | 0/45 |
| PROPULSION SUBSYSTEM | | |
| TANKS & LINES | | 5/45 |
| THRUSTER VALVES | | 5/45 |
| STAR 37 X F | | -7/43 |
| INSTRUMENTS | -40/50 | -20/40 |

met with the use of multilayered insulation blankets (MLI) and low thermal conductivity isolators.

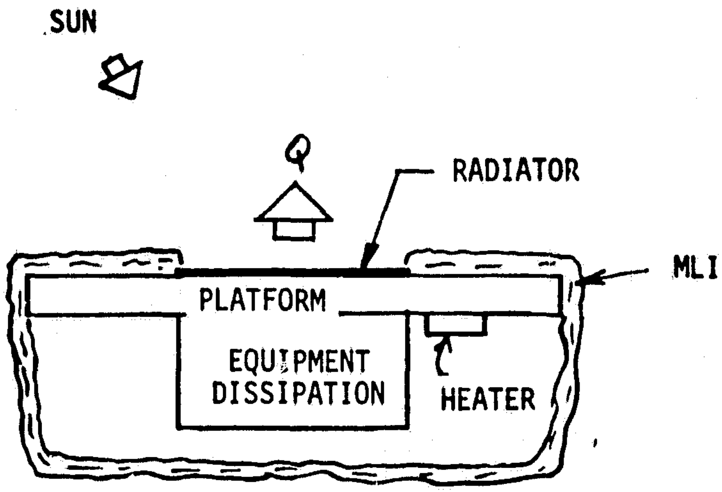
### 7.6.3 Subsystem Options

Figure 7.6-2 illustrates the major TCS options for the Communication platform. Similar TCS options also apply to the spinning platform and instrument platform. However, for the spinning platform, the louver option would not require a separate radiator due to the lack of direct solar irradiation. Each of these options is capable of maintaining acceptable spacecraft equipment temperatures throughout the Mars mission. The major advantage of the radiator/louver and radiator/variable conductance heatpipe (VCHP) systems is to conserve heater power when there are large power dissipation differences between operating modes. However, the radiator/louver and radiator/VCHP systems are more complex in construction, cost more, weigh more, and require development and additional test activities. The simpler radiator/heater option is chosen as the baseline TCS due to the following:

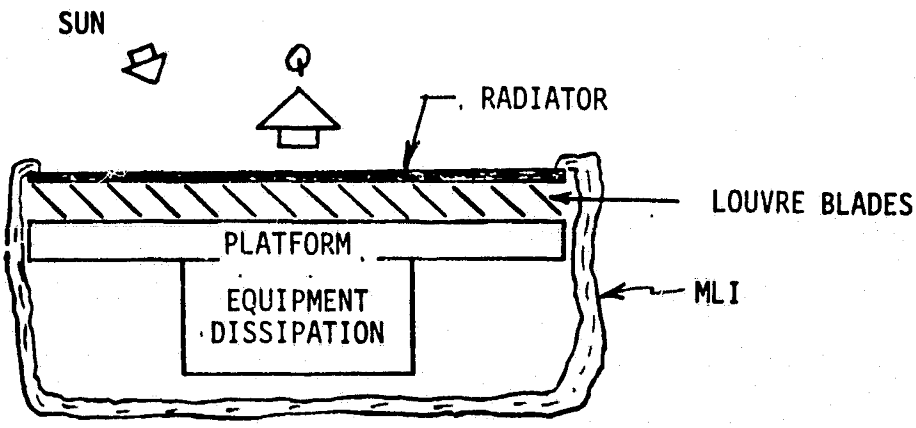
- For the spinning and instrument platforms, the equipments are spread out in a large area, and the radiator/heater option is more adaptable in maintaining individual equipment temperature control.
- For the communication platform, the power dissipation is quite constant throughout the mission with the TWTA's RF output (20 watts) being the only variable depending on whether the TWTA is in the transmit or non-transmit mode.
- Replacement heaters for the instruments provide operational flexibility.

### 7.6.4 Subsystem Description

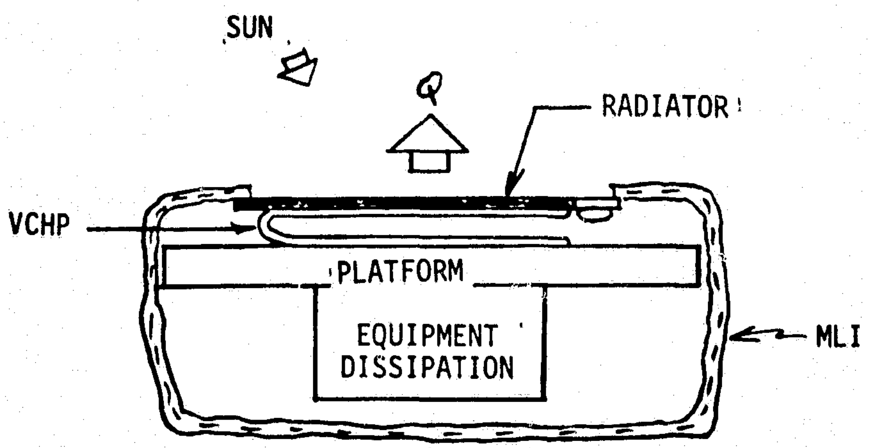
The TCS is primarily a passive system utilizing a combination of MLI, silvered-teflon tape radiators, thermal coatings, and thermal isolators of low thermal conductivity materials. These elements are supplemented by electrical heaters that are used to maintain acceptable operational environment during cold condition or to replace the equipment thermal dissipations where required to provide operational flexibility. The heaters are redundant and enabled by ground commands. When enabled, thermostats provide control of power to the strip heaters in those circuits




A. RADIATOR/HEATER OPTION




B. RADIATOR/LOUVRE OPTION



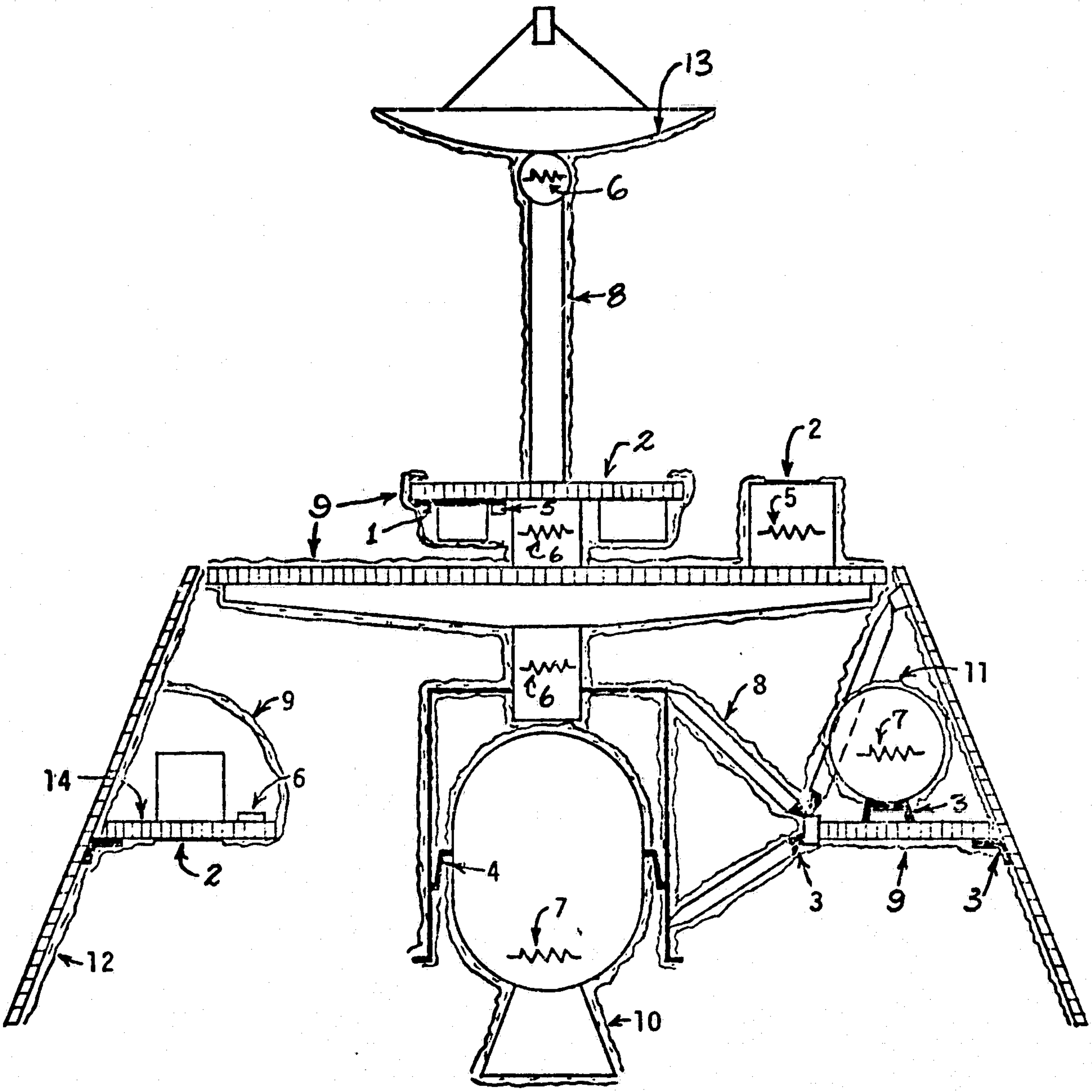

C. RADIATOR/VCHP OPTION

Figure 7.6-2

MAJOR THERMAL CONTROL SUBSYSTEM OPTIONS

used to maintain temperatures in a specified range. The thermostats are located in close proximity to the heater elements. Where overheating could be a problem due to possible thermostat failure (i.e., thruster valves and propellant line), redundant thermostat is provided. Heater status and temperature measurements are provided on telemetry to allow monitoring of the TCS performance.

Figure 7.6-3 presents a physical description of the baseline TCS. The spinning compartment is covered with MLI except those areas which are used to reject equipment heat dissipations; these areas are covered with 5 mil silvered-teflon tape which provides a low $\alpha/\epsilon$ surface finish. Thermostat controlled heaters are located on the platform for local temperature control. The batteries, because of their stringent temperature requirements (0 to 10°C), are isolated from their surrounding equipments with MLI, and have individual radiator and thermostat controlled heater. The communicationcompartment has an aluminum doubler measuring 0.37 $m^2$ in area and 0.4 cm thick placed between the TWTA and the mounting platform. The doubler is for spreading the heat dissipated by the TWTA. A heat replacement type heater is located on the doubler for maintaining the TWTA and other communication equipments at acceptable temperatures prior to their activation when the spacecraft is within the Shuttle bay and during launch. The exterior of the compartment is covered with MLI except the area used for heat rejection, which is covered with 5 MIL silvered-teflon tape. The solar array is covered with MLI in the back surface and mounted to the spinning platform with brackets of low thermal conductivity material (i.e., fiberglass); this is to provide a uniform solar array temperature and simultaneously isolating it from the spinning platform. The antennas are isolated from the despun platform with low thermal conductivity thermal isolators. The truss structure is covered with MLI, and thermally isolated from the different equipment platforms with low thermal conductivity isolators. The DMA and biaxial drive have their indicidual thermostat controlled heater, and are thermally isolated from the environment with MLI and thermal isolators. The propulsion equipments are isolated from its environment with MLI and thermal isolators, and utilize thermostat controlled heaters locating on the propellant lines, propellant tanks, thruster valves, and propellant distribution assembly to maintain hydrazine




FOLDOUT FRAME

Figure 7.6-3

BASELINE THERMAL CONTROL SUBSYSTEM

PHYSICAL DESCRIPTION

1. **Design:** 0.4 cm thick Aluminum Doubler.
   **Function:** To spread heat dissipated by the TWTA.

2. **Design:** Typically on platforms and instrument, 5 mil Silvered-Teflon tapes.
   **Function:** Radiating areas of low and stable $\alpha/\epsilon$ material to reject dissipated power at proper temperature.

3. **Design:** Brackets and thermal isolators of fiberglass and other low thermal conductivity materials.
   **Function:** To minimize conduction heat leaks from equipment platforms and propulsion components.

4. **Design:** 6 AL-4V Titanium alloy cone.
   **Function:** Low conductivity motor attachment structure to prevent excessive conduction heat soak into spacecraft after and during firing, and heat loss in orbit.

5. **Design:** Flexible Kapton strip replacement heaters for instruments and TWTA.
   **Function:** Provides operational flexibility.

6. **Design:** Flexible Kapton strip heaters and bimetallic thermostats for batteries, DMA, Az/El Drives, and selected areas of equipment platforms.
   **Function:** Maintain selected component temperature limits.

7. **Design:** Kapton tape heaters and bimetallic thermostats for tanks, lines, valves, and Orbit Insertion Motor.
   **Function:** Maintains propulsion components at selected temperatures.

8. **Design:** 10 layers, ¼ mil AL-Mylar sandwiched by inner 2 mil AL-Mylar and outer 2 mil AL-Kapton, spirally wrapped about booms and trusses.
   **Function:** Insulation to limit heat transfer with sun and space.

9. **Design:** 22 layers, ¼ mil AL-Mylar sandwiched by inner 2 mil AL-Mylar and outer 2 mil AL-Kapton, covered exteriors of equipment platforms.
   **Function:** Insulation to limit heat transfer with sun and space.

10. **Design:** 22 layers, 1/3 mil AL-Kapton sandwiched by inner 2 mil AL-Kapton and outer 3 mil AL-Kapton, covered exterior of Orbit Insertion Motor.
    **Function:** High temperature insulation to limit heat transfer with sun and space.

11. **Design:** 10 layers, ¼ mil AL-Mylar sandwiched by inner and outer 2 mil AL-Mylar on propellant tanks. 10 layers, ¼ mil AL-Mylar spinally wrapped about propellant lines and distribution assembly.
    **Function:** Insulation to maximize tank and line heater effectiveness.

12. **Design:** 10 layers, ¼ mil AL-Mylar sandwiched by inner 2 mil AL-Mylar and outer 2 mil AL-Kapton, on solar array back surface.
    **Function:** Insulates solar array from Spinning platform and provides uniform solar array temperaturs.

13. **Design:** Chenglaze white paint on reflector surface.
    **Function:** Provides acceptable temperature for reflector.

14. **Design:** Chenglaze black paint for equipment compartment interior.
    **Function:** High emittance coating to enhance radiation heat transfer in compartment interior.

2 FOLDOUT FRAME

above freezing temperature. The Orbit Insertion Motor (OIM) is isolated from its environment with MLI and Titanium thermal isolators, and electrical heater that operates from launch to orbit insertion is provided to maintain its temperature within acceptable range prior to its firing. The instrument platform and instruments are covered with MLI except for areas where instruments are mounted and areas needed as heat rejection radiators. The radiators are located on the forward surface and covered with 5 mil silvered-teflon tape. Replacement heaters locating either on the instrument or on the platform are used to provide operational flexibility. Figure 7.6-4 presents a list of the subsystem equipments together with their appropriate weight and program heritage.

### 7.6.5 Subsystem Analysis/Performance

Thermal analyses were performed to determine the radiator area and heater power required to maintain the instrument and spacecraft equipments within the acceptable temperature limits listed in Figure 7.6-1. Figure 7.6-5 presents the radiator area required. Figure 7.6-6 and 7.6-7 present the heater power required. Figure 7.6-8 presents the spacecraft power dissipations used in the analyses. Figure 7.6-9 presents predicted solar array temperatures during interplanetary cruise and on-orbit operations. Figure 7.6-10 presents predicted typical orbital operation temperatures for the instrument. Instrument temperature presented if for the orbit in which the solar vector formed an angle of 45 degrees with the orbit plane. For other orbits with different solar angles, temperature variation of slightly different magnitude is expected but at either higher or lower temperature levels.

### 7.6.6 Spacecraft Operational and Instrument Thermal Constraints

The angle between the solar vector and spacecraft spin axis should be limited to $\leq$ 60 degrees during interplanetary cruise for proper thermal control of the equipments with forward facing radiators (i.e., Communication platform and instruments). This limits the solar irradiation to the forward facing radiators at 1 AU to similar value experienced at the Martian orbit of 1.37 AU.

Three instruments with special radiators that require only space or space/planet views pose orientation problems; these are the Multi-Spectral

## THERMAL SUBSYSTEM EQUIPMENT LIST

| Component | Qty. | Unit Wt. kg | Total Wt. kg | Unit Pwr. Watts | Dimensions cm | Heritage | | |
|---|---|---|---|---|---|---|---|---|
| | | | | | | Program | Vendor | Status |
| Multilayered Insulation Blankets | 1 set | 16.9 | 16.9 | 0 | 500 $ft^2$ | Pioneer 10/11, HEAO, FSC, DSCS II | TRW | Assemble To Fit |
| Thermistors | 60 | 0.001 | 0.1 | 0 | N/A | Pioneer 10/11, HEAO, FSC, DSCS II | YSI | Purchased Part |
| Thermostatic Switches (Assumes Redundant Switches for Propulsion Components) | 96 | 0.0085 | 0.8 | 0 | N/A | Pioneer 10/11, HEAO, FSC, DSCS II | Sunstrand | Purchased Part |
| Kapton Strip Heaters (For Batteries, Platforms, AKM, DMA, EL/AZ Drives) | 22 | TBD | TBD | 0 | N/A | Pioneer 10/11, HEAO, FSC, DSCS II | Minco | Existing Technology Assemble To Fit |
| Kapton Tape Heaters (For Lines, Tanks) | 2 sets | TBD | TBD | 0 | N/A | Pioneer 10/11, HEAO, FSC, DSCS II | Minco | Assemble To Fit |
| Kapton Strip Heaters For Thruster Valves | 24 | TBD | TBD | 0 | N/A | Pioneer 10/11, HEAO, FSC, DSCS II | Minco | Assemble To Fit |

Figure 7.6-4

## RADIATOR AREA REQUIREMENTS

| Location | Area Requirement ($M^2$) | |
|---|---|---|
| | Climatology Mission | Aeronomy Mission |
| Instruments | 0.21 | 0.18 |
| Communication Platform | 0.37 | 0.37 |
| Spinning Platform | 0.16 | 0.15 |
| Batteries | 0.11 | 0.11 |

NOTE: Instrument radiator areas do not include those required to maintain detectors or interfaces at cryogenic temperatures.

Figure 7.6-5

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## HEATER POWER REQUIREMENT (CLIMATOLOGY MISSION)

| HEATER I.D. | POWER REQUIREMENTS (WATTS) | | | | |
|---|---|---|---|---|---|
| | PLANETARY CRUISE | ORBIT INSERTION | SUN OPERATION | | ECLIPSE |
| | | | SCIENCE ON TRANSMISSION | SCIENCE ON NO TRANSMISSION | SCIENCE ON NO TRANSMISSION |
| COMMUNICATION PLATFORM | 0 | 0 | 0 | 0 | 0 |
| SPINNING PLATFORM | 16 | 16 | 2 | 0 | 0 |
| BATTERIES | 0 | 0 | 21 | 21 | 0 |
| PROPULSION | 26 | 26 | 26 | 26 | 26 |
| STAR 37XF | 18 | 18 | 0 | 0 | 0 |
| El/Az DRIVES & DMA | 8 | 8 | 8 | 8 | 8 |
| INSTRUMENT | 59 | 59 | 0 | 0 | 0 |

Figure 7.6-6

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## HEATER POWER REQUIREMENT (AERONOMY MISSION)

| HEATER I.D. | POWER REQUIREMENTS (WATTS) | | | | |
|---|---|---|---|---|---|
| | PLANETARY CRUISE | ORBIT INSERTION | SUN OPERATION | | ECLIPSE |
| | | | SCIENCE ON TRANSMISSION | SCIENCE ON NO TRANSMISSION | SCIENCE ON NO TRANSMISSION |
| COMMUNICATION PLATFORM | 0 | 0 | 0 | 0 | 0 |
| SPINNING PLATFORM | 12 | 12 | 2 | 0 | 0 |
| BATTERIES | 0 | 0 | 21 | 21 | 0 |
| PROPULSION | 26 | 26 | 26 | 26 | 26 |
| STAR 37XF | 18 | 18 | 0 | 0 | 0 |
| El/Az Drives & DMA | 8 | 8 | 8 | 8 | 8 |
| INSTRUMENTS | 47 | 47 | 0 | 0 | 0 |

Figure 7.6-7

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## SPACECRAFT POWER DISSIPATIONS USED IN THERMAL ANALYSIS

| COMPONENT | PLANETARY CRUISE | ORBIT INSERTION | ORBITAL OPERATION | | |
|---|---|---|---|---|---|
| | | | SUN OPERATION | | ECLIPSE OPERATION |
| | | | SCIENCE ON TRANSMISSION | SCIENCE ON NO TRANSMISSION | SCIENCE ON NO TRANSMISSION |
| Climatology Mission | | | | | |
| Instruments | 0 | 0 | 59 | 59 | 59 |
| Communication Platform | 82 to 102 | 82 to 102 | 82 | 102 | 102 |
| Spinning Platform | 28 | 28 | 42 | 57 | 57 |
| Batteries | 30 | 30 | 0 | 0 | 48 |
| Aeronomy Mission | | | | | |
| Instruments | 0 | 0 | 47 | 47 | 47 |
| Communication Platform | 102 | 102 | 102 | 35 | 35 |
| Spinning Platform | 28 | 28 | 38 | 53 | 53 |
| Batteries | 30 | 30 | 0 | 0 | 44 |

Figure 7.6-8

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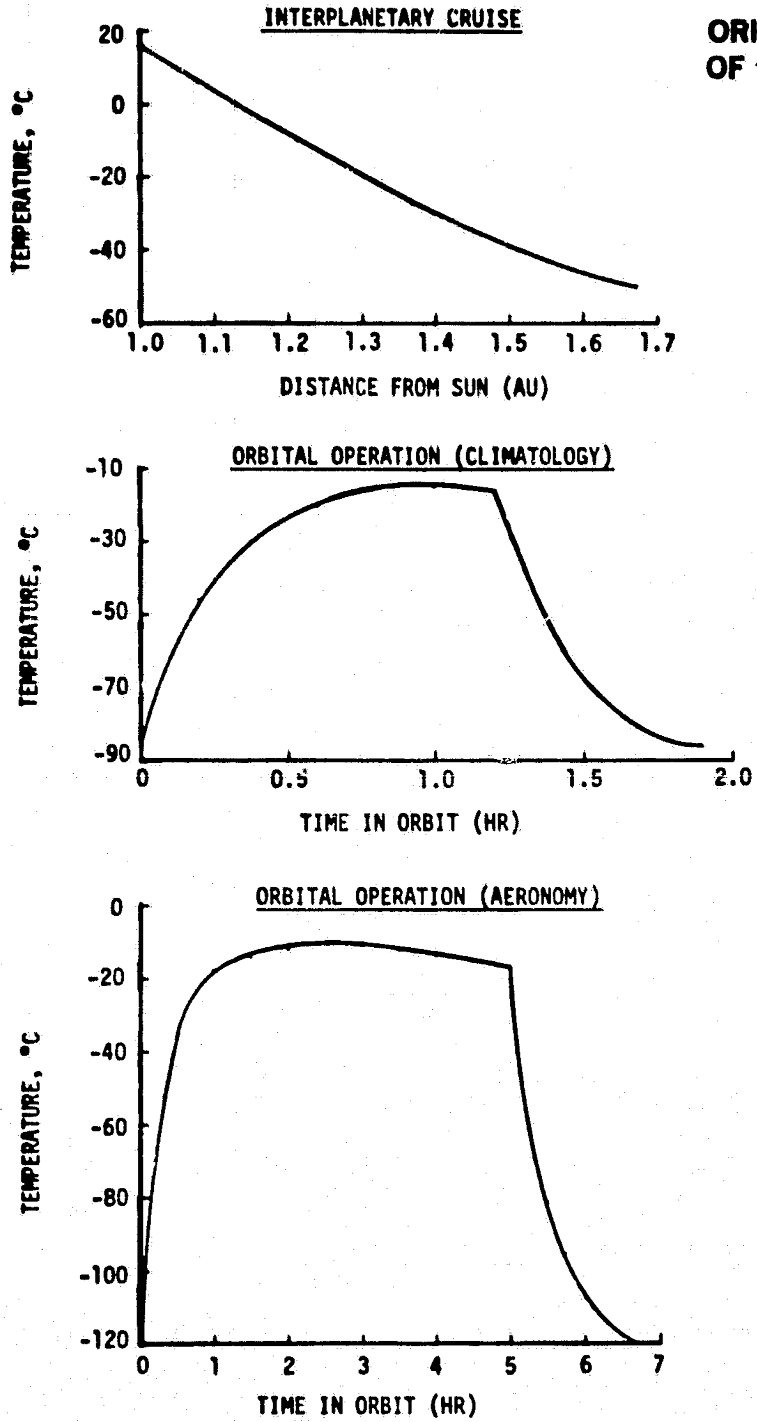


Figure 7.6-9

PREDICTED SOLAR ARRAY TEMPERATURE

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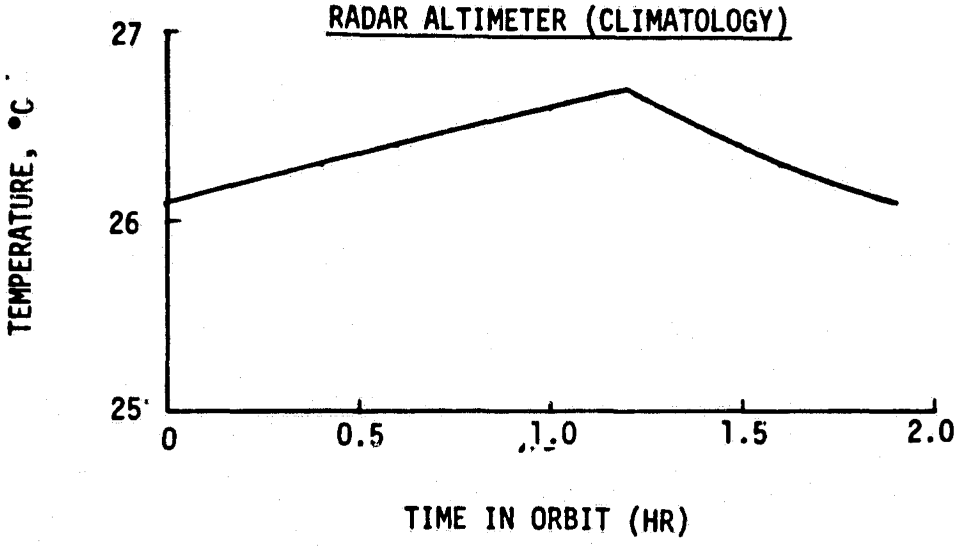


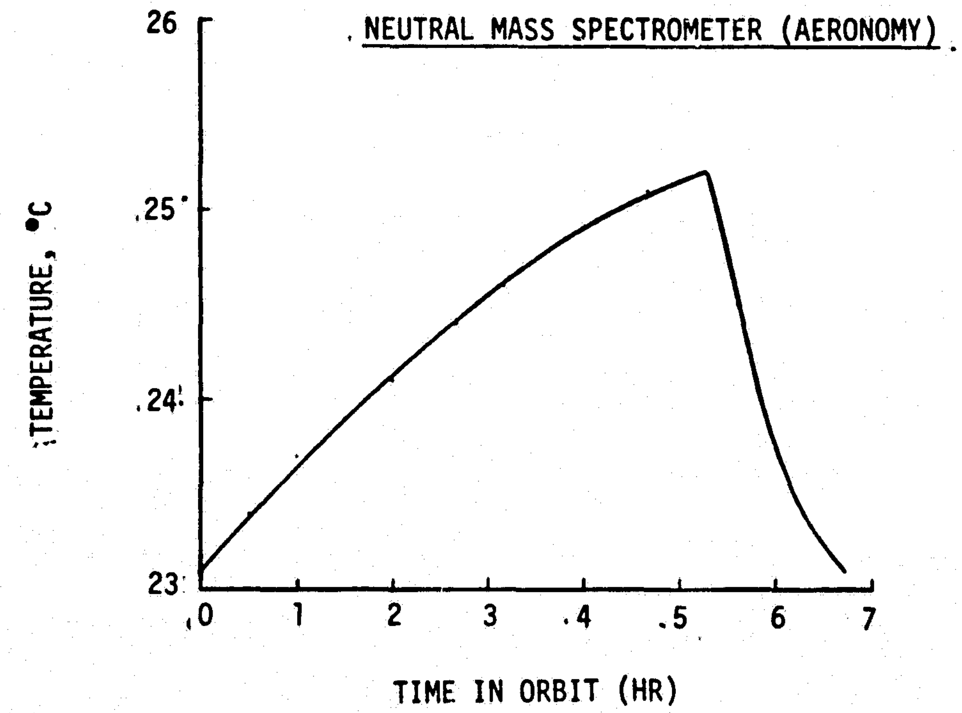
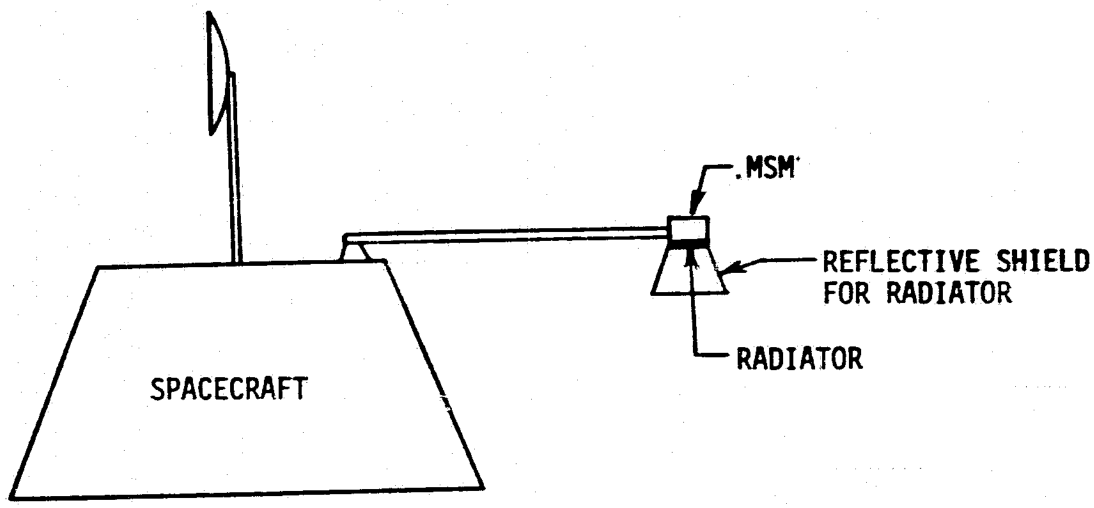




Figure 7.6-10

PREDICTED TYPICAL INSTRUMENT ORBITAL TEMPERATURE VARIATION

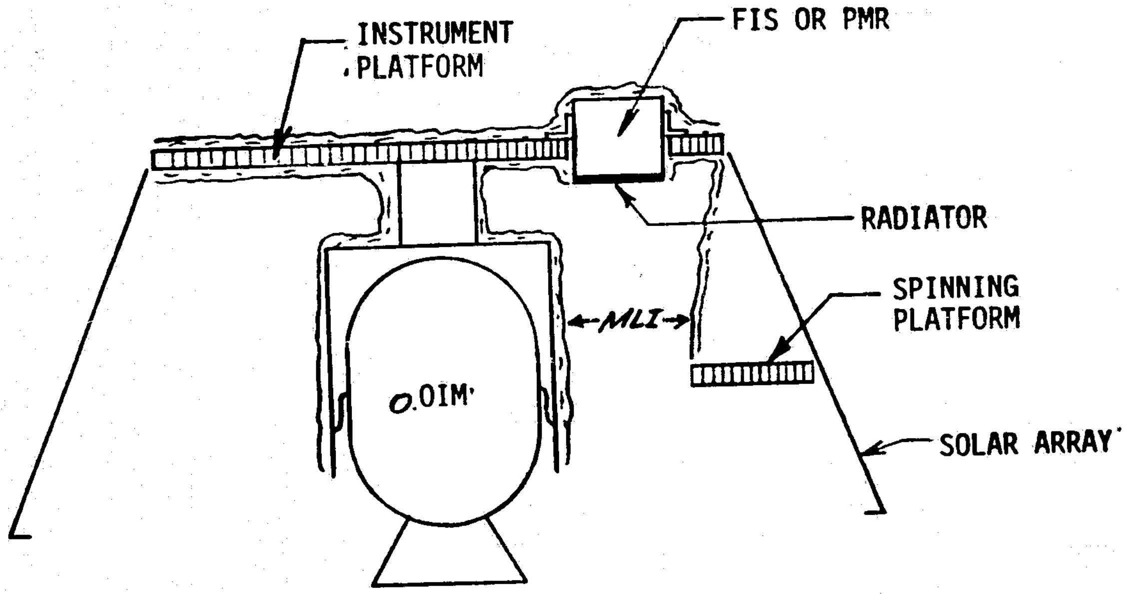
Mapper (on Climatology payload Option 3, only), Pressure Modulated Radiometer, and Frost IR Spectrometer. The MSM, which requires space only view, can probably be accommodated by having it mounted on a truss that extends outward similar to that required by the Gamma Ray Spectrometer (Figure 7.6-11). The PMR and FIS, which can view only space or planet, can probably be accommodated by having their radiators view the cavity in the aft-end of the spacecraft per Figure 7.6-12. The aft-end cavity composes of mainly MLI surfaces and space view. Very little equipment heat dissipation is rejected to this cavity since all the primary heat rejection radiators face the spacecraft forward end. Analysis showed the MLI surfaces in the aft-end cavity at or below the equivalent Martian surface temperature of about -60°C. It is therefore possible to have these instrument radiators that can view only space or planet view the aft-end cavity without compromising their operational performances.




(NOTE: REFLECTIVE SHIELD SHADES MSM RADIATOR FROM SPACECRAFT, PLANET, AND SUN.)

Figure 7.6-11

MULTI-SPECTRAL MAPPER THERMAL DESIGN




(NOTE: RADIATOR VIEWS ONLY SPACE AND AFT-CAVITY MLI, WHICH IS AT OR BELOW EQUIVALENT MARTIAN TEMPERATURE OF ABOUT -60°C.)

Figure 7.6-12

FROST IR SPECTROMETER AND PRESSURE MODULATED RADIOMETER THERMAL DESIGN

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## 7.7 PROPULSION SUBSYSTEM

### 7.7.1 Propulsion Function

The Mars Orbiter propulsion subsystem provides impulse and torque for spacecraft velocity and attitude control during the Climatology and Aeronomy Missions. Specific propulsion functions include:

- Interplanetary trajectory corrections
- Mars orbit insertion
- Orbit insertion trim, orbit maintenance, spin and attitude control
- Final quarantine maneuver (EOL orbit raising)

In addition, the spacecraft RCS will provide spin torque and nutation damping when the INTELSAT VI upper stage is mated to the orbiter.

### 7.7.2 Subsystem Requirements versus Capabilities

A summary of requirements versus capabilities for the subsystem is provided in Table 7.7-1. Included in this list are the requirements source and the method of compliance verification.

### 7.7.3 Subsystem Options and Trade Studies

Analyses were conducted to compare the propellant requirements for monopropellant and bipropellant subsystems. A baseline configuration was selected based on the results of trade studies. Factors including complexity, cost and technical risk were considered in selecting a baseline design. The results of these studies are summarized in Table 7.7-2.

### 7.7.4 Subsystem Description

A monopropellant (hydrazine) blowdown system using catalytic thrusters was selected for all propulsive tasks with the exception of Mars orbit insertion, which is to be accomplished using a solid propellant motor. The hydrazine system will operate at an initial pressure of 300 psia over a 3.0:1 blowdown range. The baseline design is illustrated schematically in Figure 7.7-1. Major subsystem components include:

- Four propellant tanks capable of supplying 135 pounds of hydrazine each over a 3.0 - 1 blowdown range
- A fill and drain module (FDM) for propellant/pressurant loading (see Figure 7.7-2)

## REQUIREMENTS VERSUS CAPABILITIES

| Requirement | Source | Capability | Verification |
|---|---|---|---|
| 1. Sufficient propellant allocation to accommodate: | PM-2000 and PM-2001 | Complies | Analysis/ Assessment |
| • Required mission maneuvers with reserves for 99.5% probability to correct errors | | Selected tank capabity can accommodate 245 kg of hydrazine per spacecraft | |
| • Corrections to cross-coupling effects on the basis of 0.5% effective thrust misalignment per thruster and mismatch of functionally paired thrusters | | | |
| • In-flight thrust calibration | | | |
| • Reserve capacity to include the uncommitted spacecraft contingency weight at time of delivery as spacecraft ballast | | | |
| 2. Spacecraft shall be designed such that from initial propellant loading preparatory to launch until end of mission life, no operating condition of the spacecraft will allow a loss of pressurant that could prevent continuous mass expulsion to the limit of the design propellant residuals, or that would compromise the predictability of performance | PM-2000 and PM-2001 | All welded propellant lines and redundant seals prohibit leakage of pressurant and propellant | Similarity/ Test |

Table 7.7-1

| Requirement | Source | Capability | Verification |
|---|---|---|---|
| 3. Expenditure of the total propellant from the maximum capacity (including reserve capacity) down to the design residuals shall not incur a spacecraft dynamic imbalance of (TBD) | PM-2000 and PM-2001 | Propellant tanks are manifolded to provide uniform tank depletion | Similarity/ Analysis |
| 4. No propellant within the subsystem shall be allowed to reach temperatures that would cause any part of the subsystem to exceed acceptance conditions | PM-2000 and PM-2001 | Thermal management will include heaters, multilayer insulation and surface coatings as required to prevent freezing or overheating of propellant | Similarity/ Analysis/Test |
| 5. The minimum tepmerature of the propellant supply system shall be demonstrated during system thermal vacuum tests to be at least 5$^{\circ}$F above the propellant freezing temperature | PM-2000 and PM-2001 | Propulsion subsystem will be tested during spacecraft thermal/vacuum tests | Test |
| 6. Heaters may employ command actuation of a steady state control, but duty cycle control by ON-OFF command modulation shall not be embodied in the design | PM-2000 and PM-2001 | Heaters will be thermostatically controlled. Duty cycling by ground command not employed | Assessment |

| Requirement | Source | Capability | Verification |
|---|---|---|---|
| 7. Each thruster shall be capable of operating in a pulsed or continuous firing mode | PM-2000 and PM-2001 | Complies | Similarity/ Analysis/Test |
| Each thruster shall be capable of firing at least (TBD) seconds cumulative duration in the continuous mode, (TBD) pulses in the pulsing mode, and (TBD) bed ambient starts | | | |
| 8. Thruster performance predictability shall be established by a ground calibration program and the following error allocation: | PM-2000 and PM-2001 | Thruster predictability established during qualification programs and acceptance tests | Similarity/Test |
| <u>Pulsed mode</u> | | | |
| Impulse uncertainty: (TBD) % | | Pulsed mode maneuver predictability = 1.8% | |
| Centroid error : (TBD) % | | | |
| <u>Continuous mode</u> | | | |
| Impulse uncertainty: (TBD) % | | Steady state maneuver predictability = 1.2% | Test/Assessment |
| Centroid error : (TBD) % | | | |

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| Requirement | Source | Capability | Verification |
|---|---|---|---|
| 9. Total number and placement of thrusters shall be such as to preclude single point failures catastrophic to the mission | PM-2000 and PM-2001 | Complies<br>Thrusters are oriented on spacecraft such that failure of a single thruster is non-catastrophic to mission | Similarity/ Analysis/Assessment |
| 10. Fill, drain and thruster valves shall have redundant seals | PM-2000 and PM-2001 | Complies<br>• 3 seals in F/D valve<br>• Dual seat thruster valve | Similarity |
| 11. Thruster control valves shall have redundancy for opening actions. Actuation shall only be executed by presence of electrical power. Absence of electrical power shall result in positive closure of valve | PM-2000 and PM-2001 | Complies<br>Dual coil for actuation. Positive closing provided by spring loaded poppet | Similarity |
| 12. Subsystem shall provide filtered propellant with a contamination level that shall not cause leakage exceeding 0.1 lb per year | PM-2000 and PM-2001 | Complies<br>10 micron propellant filter removes particulate matter | Similarity |
| 13. Propulsion assembly shall be designed to provide leak test capability. Propellant leakage shall not exceed 0.1 lb per year as demonstrated by gas leakage rate | PM-2000 and PM-2001 | Complies<br>Subsystem to be tested using helium mass spectrometer method | Similarity/Test |

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| Requirement | Source | Capability | Verification |
|---|---|---|---|
| 14. Propellant supply pressure shall be provided by telemetry with 2 psia minimum resolution and an error band of 1 percent of full scale maximum | PM-2000 and PM-2001 | Propellant pressure transducer has 2 psig resolution with 0.8% static error | Similarity |
| 15. External propellant tank temperature at a location that will provide values representative of pressurant temperature shall be provided by telemetry | PM-2000 and PM-2001 | Complies<br>Propellant tank temperature will be monitored by thermistor bonded to external surface | Similarity/ Analysis/Test |
| 16. Temperature of each thruster control valve shall be telemetered | PM-2000 and PM-2001 | Complies<br>Thruster valves are equipped with thermistors for temperature monitoring | Similarity |
| 17. Subsystem to comply with applicable safety requirements | PM-2000 and PM-2001 | Subsystem complies with requirements in STS safety document NHB 1700.7A | Assessment |
| 18. Operational lifetime $\sim$5 years | PM-2000 and PM-2001 | $\leq$10 years demonstrated capability | Similarity/ Assessment |
| 19. Axial thrusters operated in pairs shall be selected so that the thrust mismatch is less than 2% for the continuous mode of operation throughout the mission lifetime. | PM-2000 and PM-2001 | Thrusters can be matched as required. | Test/assessment |

| Trade Study | Approaches Considered | Approach Selected | Rationale for Selection |
|---|---|---|---|
| Propulsion method | 1. Monopropellant RCS with solid Mars orbit insertion motor<br>2. All bipropellant RCS and orbit insertion motor | Monopropellant RCS with solid motor | Less complex<br>Lower cost<br>More reliable (fewer components)<br>Less risk<br>Acceptable performance |
| Propellant tank configuration | 1. 4 spherical<br>2. 6 teardrop | 4 Spherical | Less complex<br>Fewer components required<br>Compatible with structural space allocation<br>Less cost<br>Less weight |

Table 7.7-2

PROPULSION TRADE STUDIES SUMMARY

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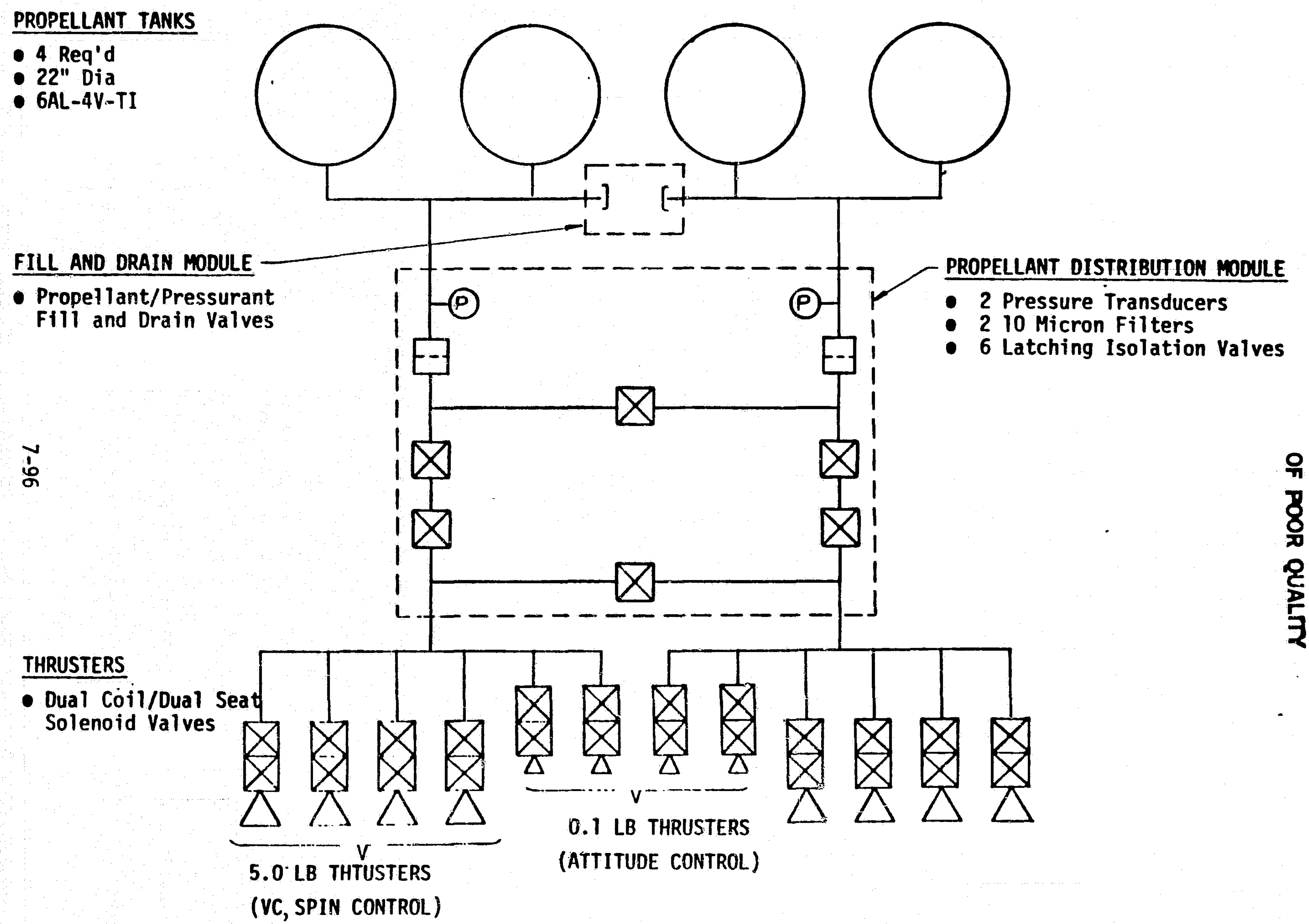


Figure 7.7-1

PROPULSION SCHEMATIC

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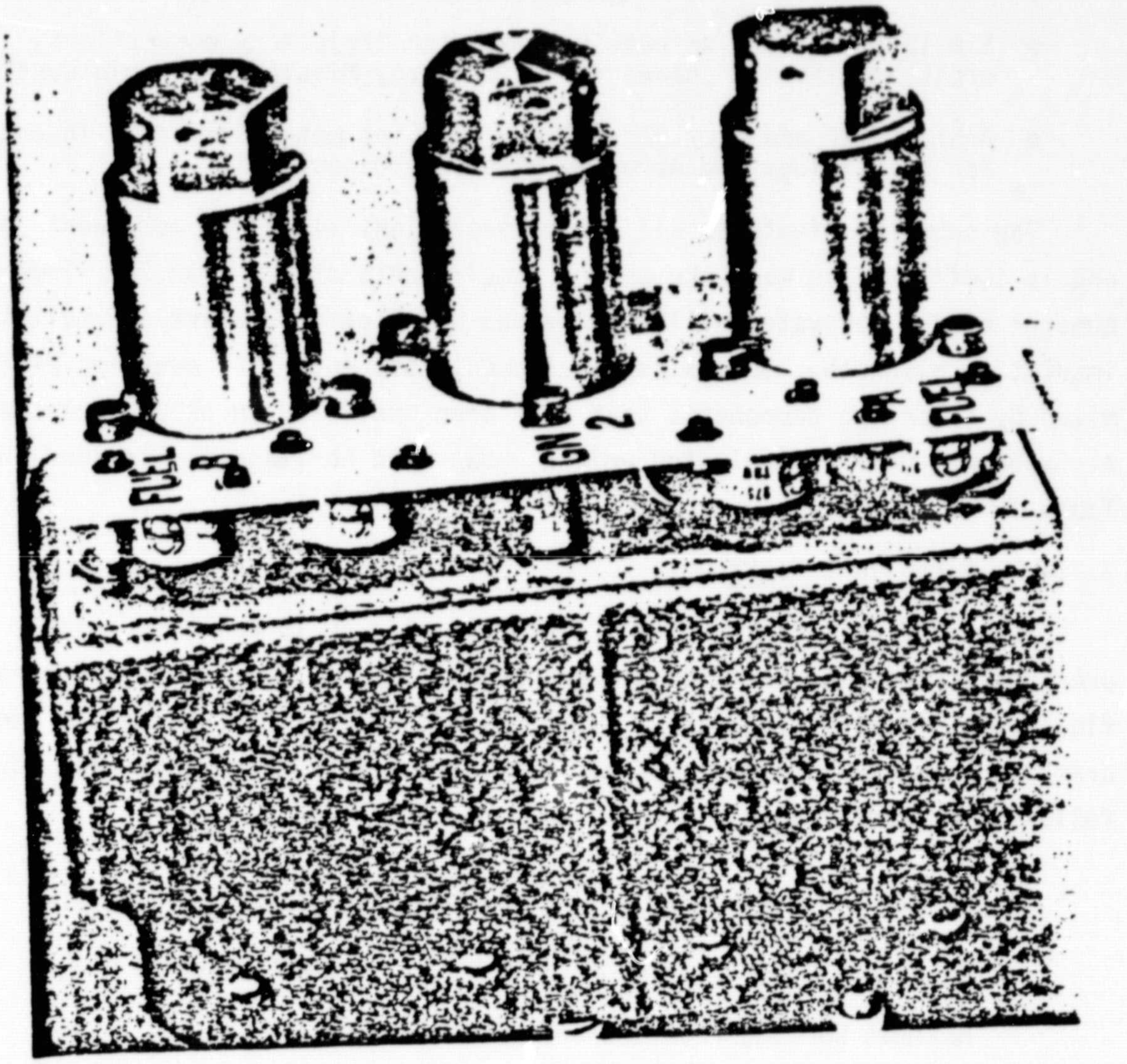

Figure 7.7-2

TDRSS FILL AND DRAIN MODULE

- A propellant distribution module (PDM) containing latching isolation valves, filters and pressure transducers (see Figure 7.7-3)

- 0.1 $lb_f$ thrusters to provide attitude control impulse

- 5.0 $lb_f$ thrusters to provide midcourse trajectory corrections, orbit inclination change for Climatology Mission, and spin control

- A Star 37XF and Star 37N solid propellant motor for orbit insertion for Climatology and Aeronomy Missions respectively

The subsystem features all welded propellant lines and component joints, and is configured to meet the safety requirements of MHB 1700.7A. Thermal control of the subsystem will be provided by electric heaters and multilayer insulation blankets. Subsystem development cost and risk have been minimized by selecting components that have been qualified on other spacecraft. A component summary, including weight, power and heritage is provided in Table 7.7-3.

### 7.7.4.1 Propellant Tanks

The propellant tanks are 22.0 inch diameter (0.25 wall) spherical pressure vessels manufactured by Fansteel. They are qualified, and were flown on the Gemini spacecraft. Internal pressurization ($GN_2$) and spacecraft spin ensure continuous expulsion of hydrazine. The tanks have the following characteristics:

| | |
|---|---|
| Volume | 5575 $in^3$ |
| Material | 6AL-4V titanium |
| Maximum Working Pressure | 300 psia |
| Proof Pressure | 500 psia |
| Burst Pressure | 700 psia |
| Pressurization Control | Blowdown |
| Weight | 3.6 kg (each) |

### 7.7.4.2 Propellant Pressure Transducers

Two pressure transducers are installed on the PDM to monitor tank pressure and provide an indication of remaining subsystem life. The transducer is manufactured by Gould Inc. (Statham) and is used on TDRSS and DSP.

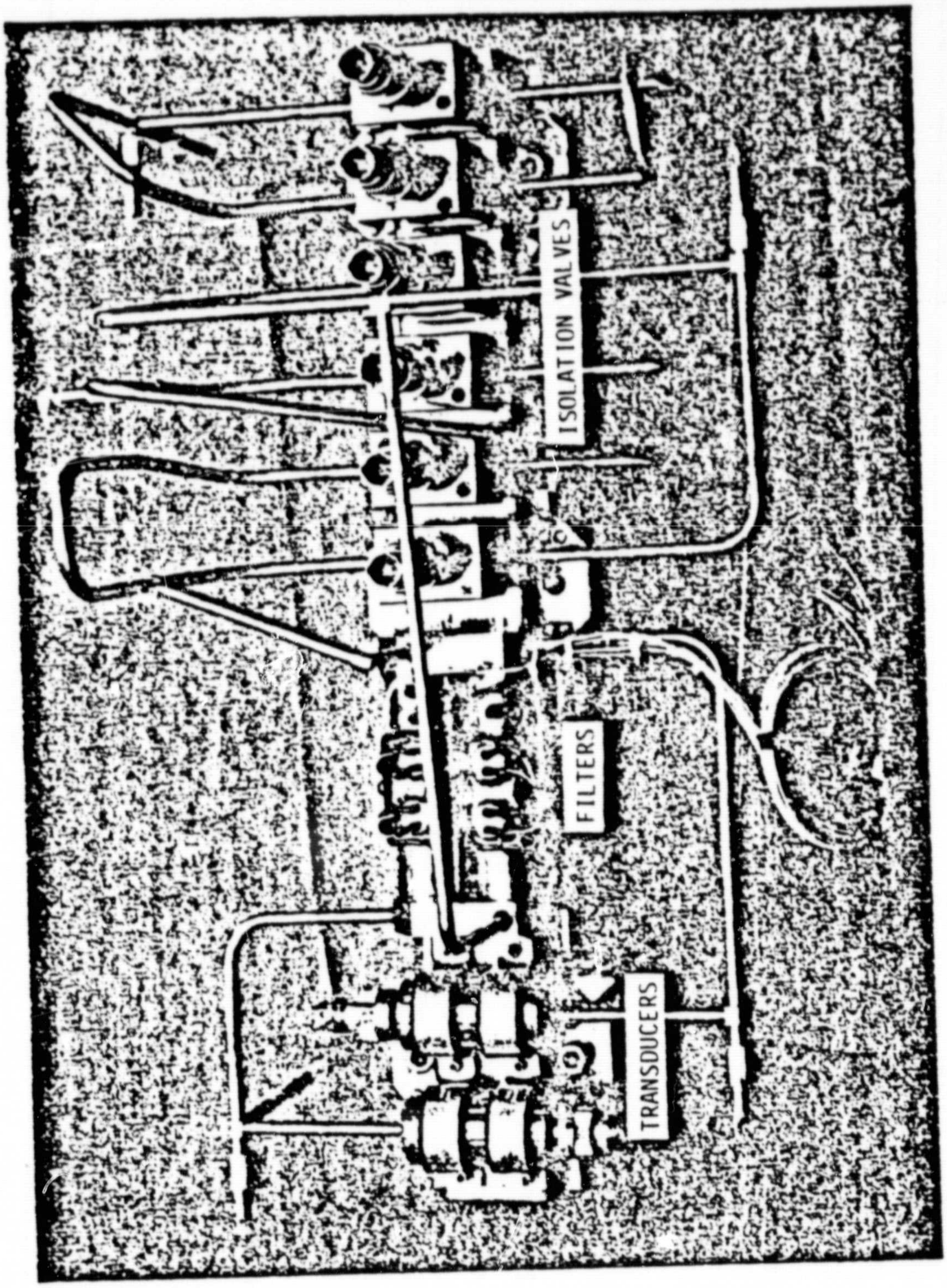


Figure 7.7-3

TDRSS PDM

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PROPULSION SUBSYSTEM

EQUIPMENT LIST

| Component | Qty. | Unit Wt. kg | Total Wt. kg | Unit Pwr. Watts | Dimensions cm | Heritage | | |
|---|---|---|---|---|---|---|---|---|
| | | | | | | Program | Vendor | Status |
| Propellant Tanks | 4 | 3.6 | 14.4 | N/A | 55.9 dia. | Gemini | Fansteel | Purchased |
| Pressure Transducers | 2 | 0.27 | 0.54 | .20 | -- | TDRSS | Gould | |
| Fill and Drain Valves | 2 | 0.14 | 0.28 | N/A | -- | TDRSS | Pyronetics | |
| Propellant Filters | 2 | 0.13 | 0.26 | N/A | -- | TDRSS | Wintec | |
| Latching Isolation Valves | 6 | 0.27 | 1.62 | 13.6 (Rated) | -- | TDRSS | Hydraulic Research | |
| 0.1 $lb_f$ Thrusters | 4 | 0.4 | 1.60 | 4.8 (Rated) | -- | TDRSS | --- | Existing Component |
| 5.0 $lb_f$ Thrusters | 8 | 0.56 | 4.54 | 7 (Rated) | -- | GRO | --- | |
| Propellant Lines and Supports | 1 set | -- | 2.3 | N/A | -- | -- | --- | Made To Fit |
| Star 37XF (Climatology) | 1 | -- | 796 (1) | N/A | -- | -- | Thiokol | Purchased |
| Star 37N (Aeronomy) | 1 | -- | 477 (1) | N/A | -- | -- | Thiokol | |

(1) Weight includes propellant required for orbit insertion.

Table 7.7-3

The unit is considered qualified based on similarity of the Mars orbiter requirements with previous TRW propulsion subsystems.

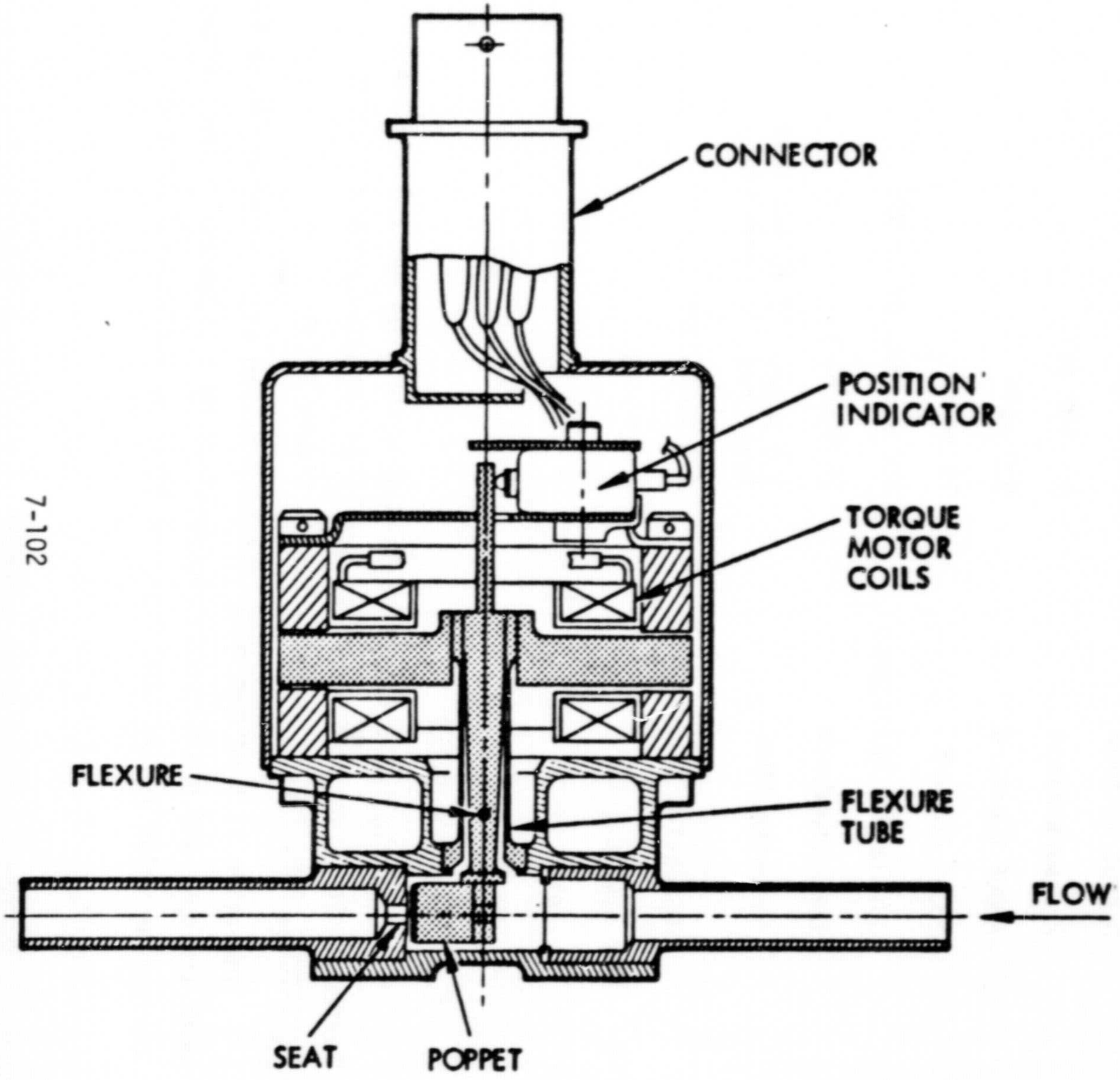
The unit consists of a machined diaphragm connected by a force rod to a cantilever beam where four active thin-film vacuum-deposited strain gages are attached with electronics for signal conditioning and amplification. The integral electronics package of welded construction provides input/output isolation, voltage regulation, output signal amplification (to 5 volts), output impedance of less than 150 ohms, and internal single point calibration for functional checkout and data reduction. The transducer characteristics are summarized below:

| | |
|---|---|
| Excitation Voltage | 22 - 33 VDC |
| Output Voltage | 0 - 5 VDC |
| Pressure Range | 0 - 400 psia |
| Proof Pressure | 600 psia |
| Input Current | 15 ma max. |
| Insulation Resistance | 100 megohms at 50 VDC |
| Load Impedance | 25,000 ohms |
| Response Time | 3.0 m sec, 10 to 90% with less than 10% overshoot |
| Operating Temperature | 4.4 to 48.9°C |
| Static Error | ± 0.8% FS |
| Total Error | ± 1.8% FS |
| Weight | 0.27 kg |

7.7.4.3 Isolation Valve

The latching isolation valve is manufactured by Hydraulic Research-Textron. It is the same as units qualified for use by TRW for HEAO, FLTSATCOM, DSP, ISEE-C and is shown in Figure 7.7-4.

The valve is a torque motor actuated design with latching forces in both the open and closed positions supplied by a permanent magnet circuit. The torque motor is isolated from the fluid by a flexure tube which is



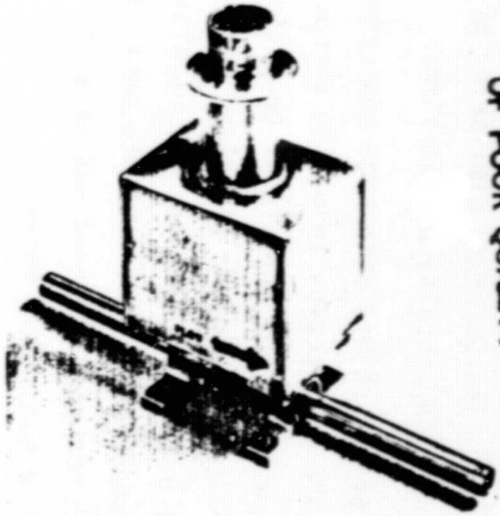





Figure 7.7-4

LATCHING ISOLATION VALVE

also the primary valve spring. The torque motor contains two non-redundant coils, wound in opposite directions. One coil is for opening the valve and the other coil is for closing the valve. With the valve in the closed position, the armature is held against the closed pole surfaces by permanent magnet attractive forces. The magnet force is significantly greater than the force required to deflect the flexure tube from a spring null position to the closed position.

To open the valve, a dc voltage is applied to the opening coil which establishes a magnetic field in opposition to that of the permanent magnet. This opposing field diminishes the attraction of the armature to the closed pole and increases its attraction to the open pole. When the total magnetic force is reduced to a level lower than the spring force, the armature moves off the closed poles and to the open poles where it is again retained by the permanent magnetic forces. An integral position switch is incorporated for positive indication of poppet position.

The characteristics of the valve are given below:

| | |
|---|---|
| Weight | 0.27 kg |
| Operating Pressure | 600 psig (max.) |
| Proof Pressure, Closed | 900 psig |
| Proof Pressure, Open | 1350 psig |
| Burst Pressure | 2400 psig |
| Internal Leakage @ 600 psig | 0.5 scc/hr $GN^2$ |
| External Leakage @ 600 psig | $1.0 \times 10^{-7}$ scc/sec $GH_e$ |
| Cycle Life | 5000 |
| Operating Temperature | 40° to 150°F |
| Flow Rate | 0.02 lb/sec |
| Closing Response | 50 ms (max.) |
| Opening Response | 50 ms (max.) |
| Pressure Drop | 36 psid at 0.02 lb/sec |
| Back Relief Pressure | 700 psid |

7.7.4.4 Fill and Drain Valve

The fill and drain valve, shown in Figure 7.7-5, is manufactured by Pyronetics, Inc. Valve opening and closing is actuated by turning the hexagonal nut to advance or retract the poppet. When the edge of the poppet contacts the valve seat, a metal-to-metal seal is formed. Redundant sealing is provided by an O-ring between the poppet and valve body, and a cap on the threaded end of the poppet. The entire assembly is covered by an O-ring sealed pressure cap. Hence the valve exceeds STS safety requirements by having three independent seals to prohibit hydrazine leakage.

7.7.4.5 Propellant Filter

Two propellant filters manufactured by Wintec, Inc., are located on the propellant distribution module. They are capable of removing all particles larger than 10 microns from the propellant before it passes through the isolation valves and propellant valves. Figure 7.7-6 shows the filter and its internal configuration.

7.7.4.6 Thrusters

TRW MRE-0.1 and MRE-5 thrusters will be used to provide impulse for all propulsive functions except Mars orbit insertion. The MRE-0.1 thruster was qualified on the FLTSATCOM program. The MRE-5 will be qualified for the GRO (Gamma Ray Observatory) spacecraft. The details of these thrusters are illustrated in Figures 7.7-7 and 7.7-8.

The basic components of each thruster are the propellant valve, a head end assembly, catalyst bed, chamber and nozzle assembly, and heaters. These components are discussed briefly below.

The head end assembly consists of a structural support that serves as a thermal barrier between the thrust chamber and the propellant valve, a feed tube and an injector plate. The feed tube is crimped to provide thruster-to-thruster repeatability, and contains a spiral loop to allow differential thermal expansion. The injector plate is configured to provide a uniform spray of hydrazine into the catalyst, which contains two sizes of Shell 405-ABSG catalyst. The upstream bed contains 18 to 20 mesh catalyst, and the lower bed contains 20 to 30 mesh catalyst.

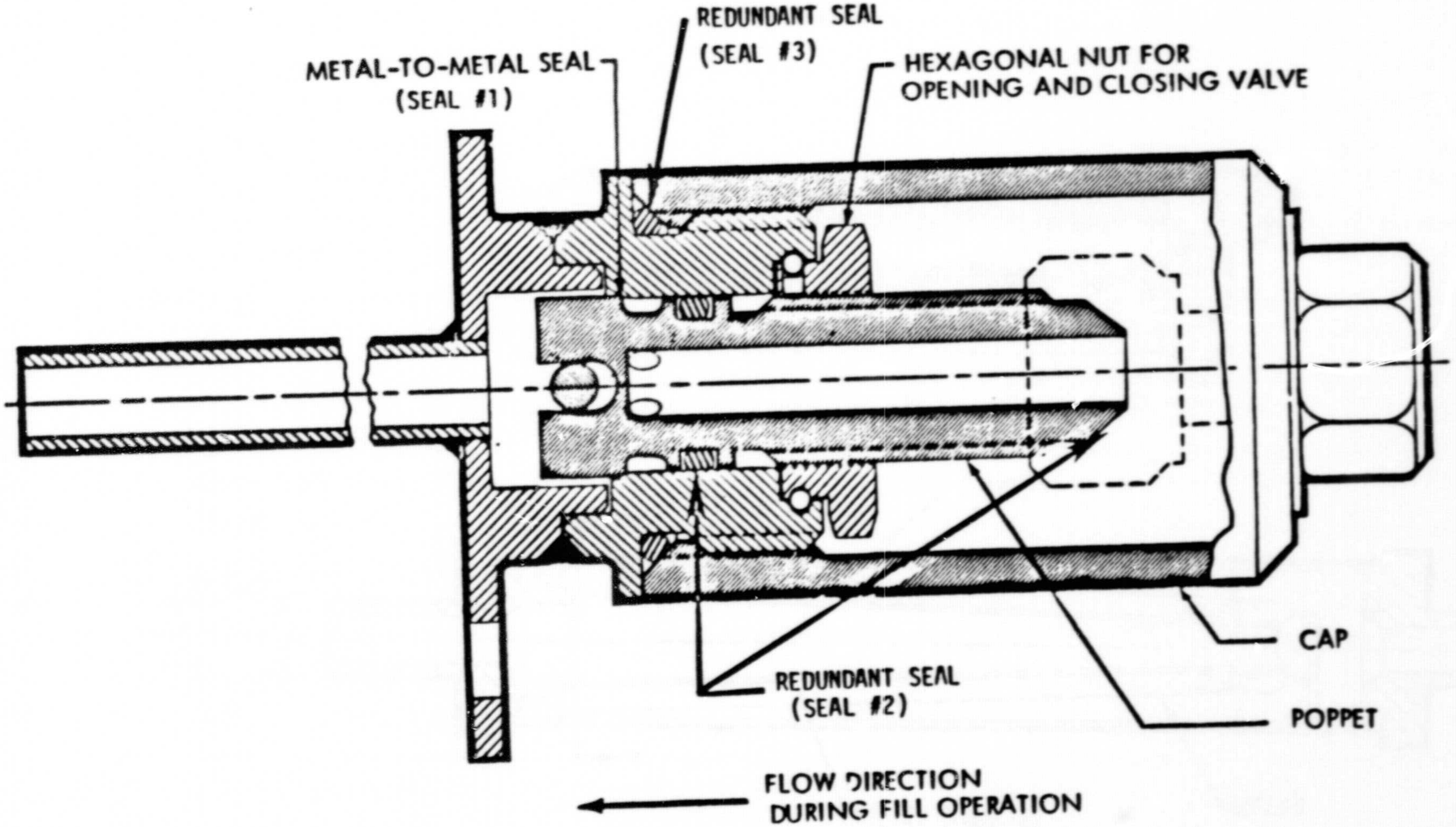



Figure 7.7-5

FILL AND DRAIN VALVE

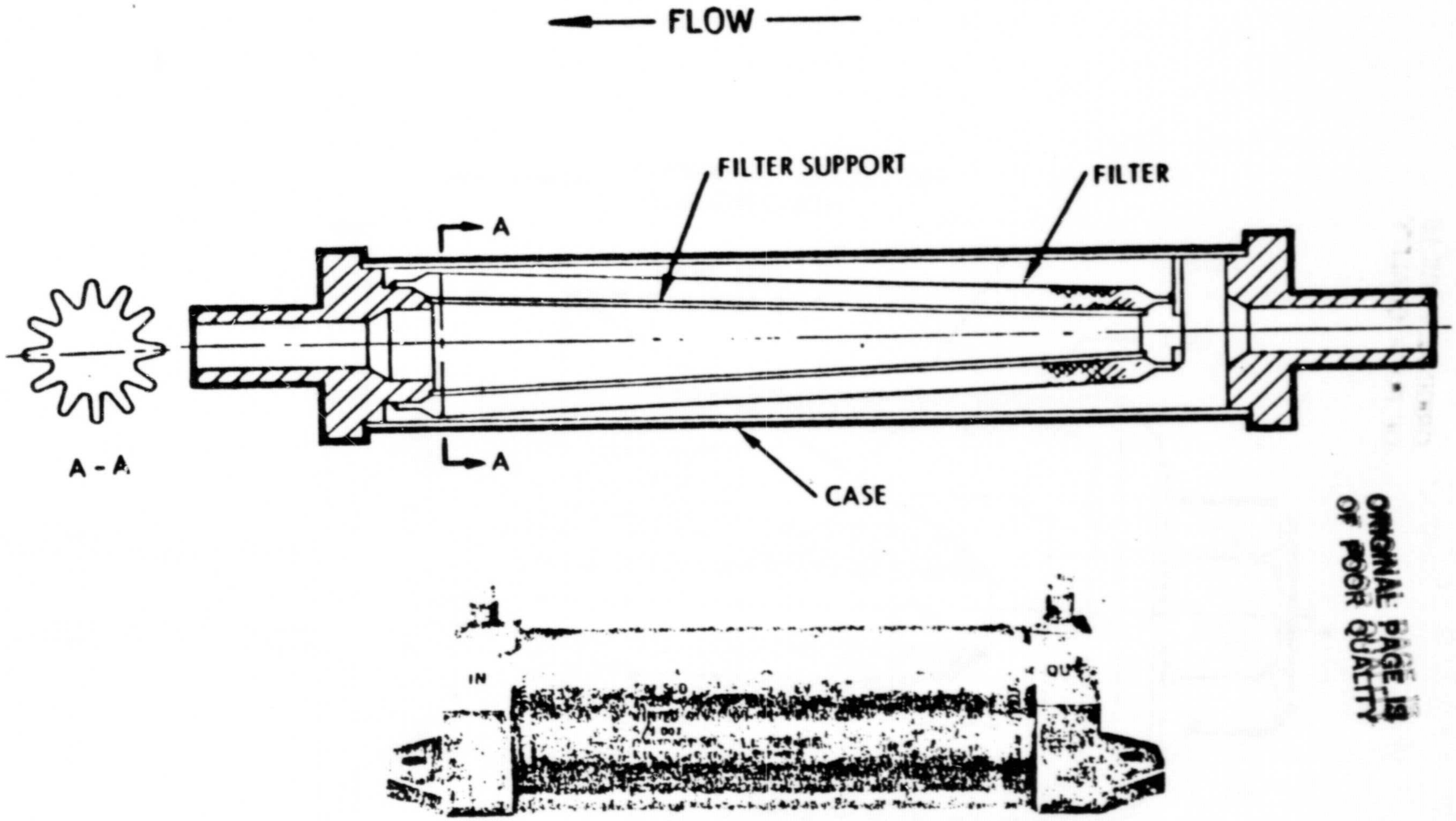



Figure 7.7-6

PROPELLANT FILTER

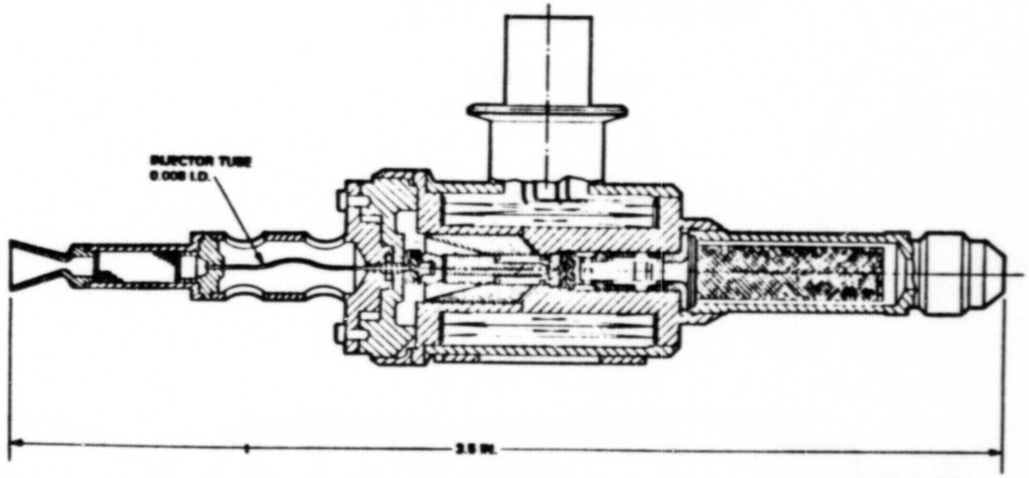


Figure 7.7-7

<u>CUTAWAY OF 0.1 LB THRUSTER</u>

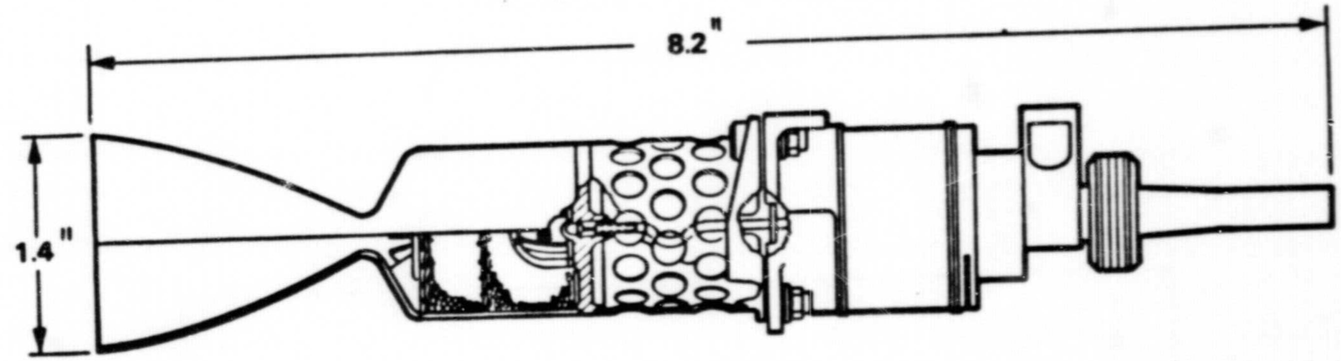



Figure 7.7-8

5.0 LB. THRUSTER CUTAWAY

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The thrust chamber, fabricated from Haynes-25 (L605) is welded to an L605 nozzle. Screens, also made of L605, are welded into the chamber to separate the two catalyst sizes, and to retain the catalyst within the chamber.

The propellant valves are dual coil, dual poppet coaxial design, incorporating redundant seals and coils. Each of the two poppets has an elastomeric seal (AF-E-411) and is spring loaded with a fail safe mode in the closed position. When either coil is energized, a magnetic force is created between the plunger and valve body, thereby opening the valve. When the coil is de-energized, the spring returns the poppet to the closed position. The downstream poppet is mechanically linked to the solenoid plunger, and provides redundant sealing with two seals in series. The valve inlet incorporates a tube for welding the thruster into the system and a mechanical joint that is used for ground testing.

Thruster Heaters. Each thruster is equipped with two catalyst bed heaters (connected in parallel for redundancy) and two valve heaters (primary and redundant). These heaters are described briefly in the following sections.

A. Catalyst bed heaters. Catalyst bed heaters are used to provide increased thruster life and increased "first pulse" specific impulse. The thrusters utilize the heater design qualified and flown on FLTSATCOM. These heaters are designed to operate at approximately 15 Vdc, or about 1/2 the bus voltage. In order to utilize the standard bus voltage, pairs of heaters will be connected in series.

Figure 7.7-9 is a cross section of the heater element assembly showing the "free standing" element as it is supported by the ceramic mandrel. The element is, in fact, floating in four holes running the length of the mandrel and is "stabilized" only by an alumina power which is vibration packed into the element cavity. Thus, no deflection stresses are generated as the element expands and contracts over a temperature range of ambient of 1750$^{o}$F.

The element wires are welded to gold-plated, platinum transition jumpers, which in turn are brazed to the nichrome sheath wires. During manufacture, the element cavity is evacuated and back-filled with dry argon gas.

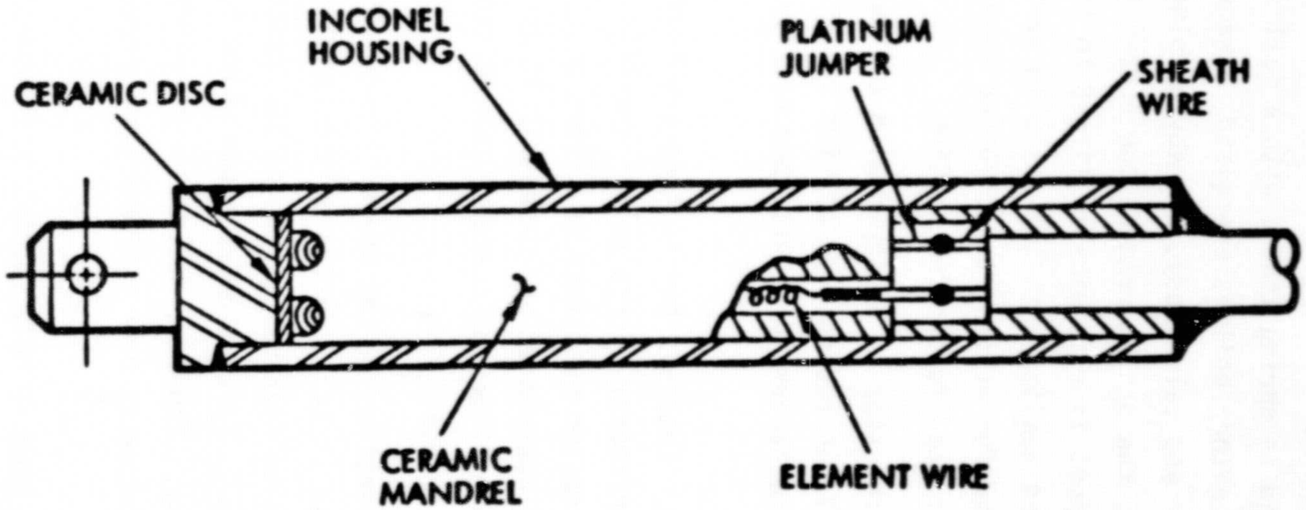



Figure 7.7-9

CROSS SECTION OF CAT. BED HEATER

In order to connect to the spacecraft power supply, the transition subassembly as shown in Figure 7.7-10 is utilized. The key element of this assembly is the glass header which serves as a hermetic feedthrough for the conductors. Allen PBX is applied to the sheath wires and pins to avoid any movement and consequent shorting to case. The lead wires are anchored to the heater assembly via knots imbedded in potting which transmit loads to the transition tube.

B. Valve heaters. Each propellant valve will be equipped with two valve heaters (similar to those used on HEAO and TDRSS). Figure 7.7-11 shows a cross section of the heater.

#### 7.7.4.7 Solid Propellant Motors

The Mars orbit insertion (MOI) maneuver will be accomplished using Star 37XF and Star 37N solid propellant motors for the climatology and aeronomy missions, respectively. Both are part of the Star 37 series of qualified motors manufactured by Thiokol. Although these engines deliver different performance, they are mechanically and electrically interchangeable. The characteristics of each motor are summarized below.

| | 37XF | 37N |
|---|---|---|
| Status | To be qualified for INTELSAT V | Qualified. Flown on Japanese 'N' L.V. |
| Average Thrust ($lb_f$) | 9000 | 8900 |
| Burn time (sec) | 62.5 | 38.3 |
| Total pre-burn inerts (kg) | 64.0 | 63.6 |
| Burned-out inerts (kg) | 62.7 | 62.0 |
| 100% propellant load (kg) | 877 | 560 |
| Total impulse | 566,000 lb-sec | 357,500 lb-sec |
| Specific impulse (sec) | 293.3 | 287.8 |

### 7.7.5 Interfaces

This section provides a description of the interfaces between the propulsion subsystem and the other spacecraft subsystems.

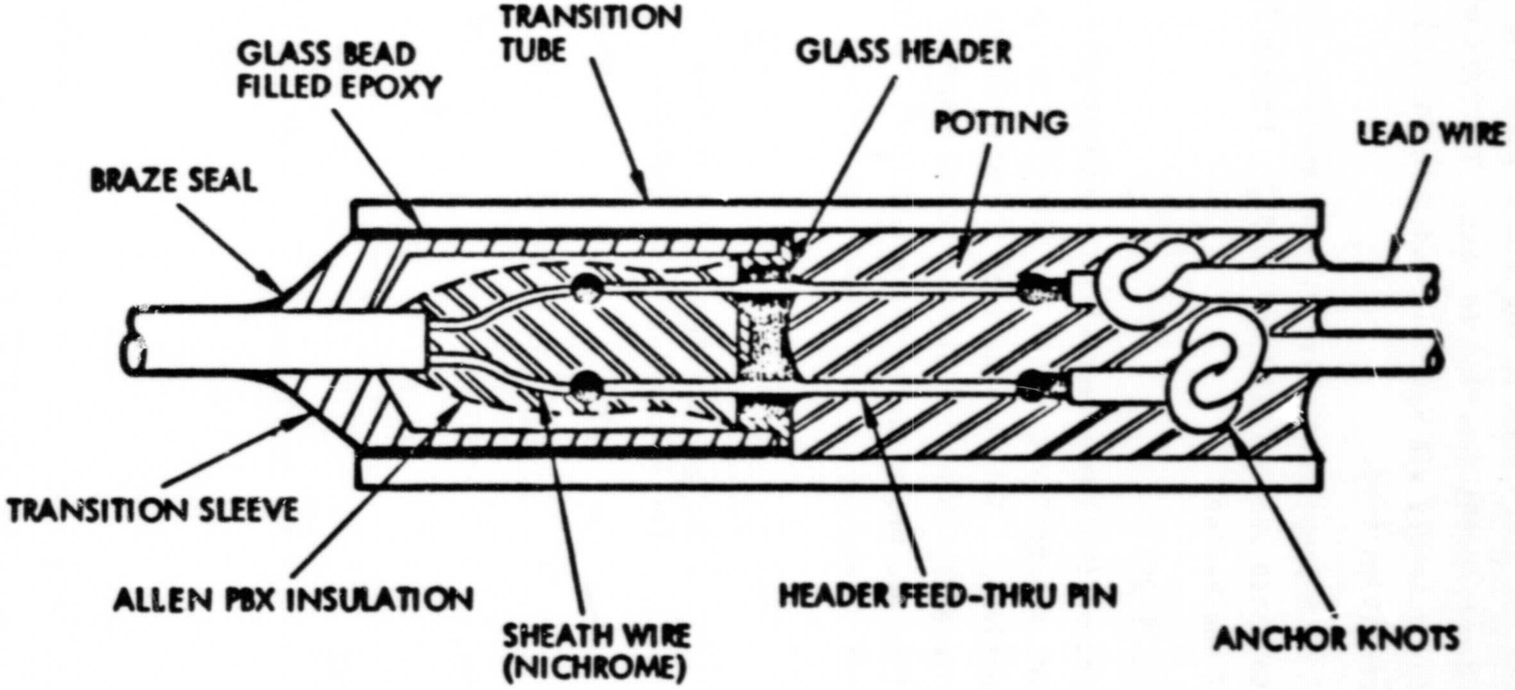



Figure 7.7-10

CROSS SECTION OF TRANSITION JOINT

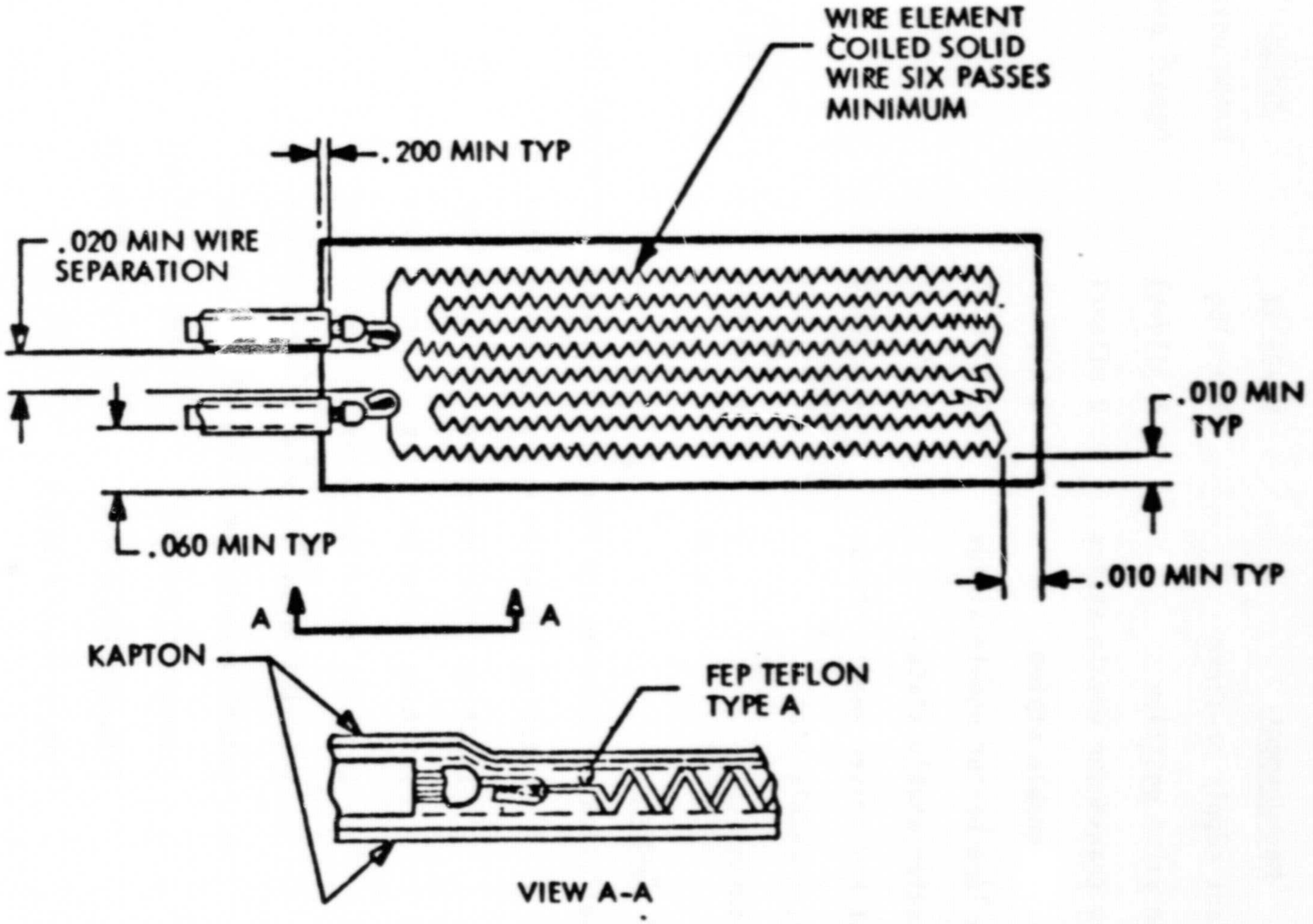



Figure 7.7-11

KAPTON VALVE HEATER

### 7.7.5.1 Telemetry

The following parameters will be available for ground evaluation of the propulsion system status and performance.

| Measurement | Quantity | Range |
|---|---|---|
| Propellant supply pressure | 2 analog | 0-400 psia |
| Latching valve position | 6 bilevel | Open/closed |
| Pressure transducer enable status | 2 bilevel | |
| Tank heater enable status | 2 bilevel | |
| PDM and line heater enable status | 2 bilevel | |
| Valve heater enable status | 2 bilevel | |
| Catalyst bed heater enable status | 4 bilevel | |
| FDM heater enable status | 2 bilevel | |
| Temperatures | 26 analog | $-50^{o}$ to $+250^{o}F$ |

### 7.7.5.2 Commands

The following commands are required to operate the propulsion subsystem.

| | | |
|---|---|---|
| Latching isolation valve | 1 | open/close |
| Latching isolation valve | 2 | open/close |
| Latching isolation valve | 3 | open/close |
| Latching isolation valve | 4 | open/close |
| Latching isolation valve | 5 | open/close |
| Latching isolation valve | 6 | open/close |
| Pressure transducer | 1 | enable/disable |
| Pressure transducer | 2 | enable/disable |
| Tank primary heaters | | on/off |
| Tank redundant heaters | | on/off |
| PDM/line primary heaters | | on/off |
| PDM/line redundant heaters | | on/off |

| | |
|---|---|
| Valve primary heaters | on/off |
| Valve redundant heaters | on/off |
| 1.0 lb cat. bed heaters | on/off |
| 5.0 lb cat. bed heaters | on/off |
| FDM primary heater | on/off |
| FDM redundant heater | on/off |

- Total = 36 commands

7.7.5.3 Power

Power requirements are summarized for individual components in Table 7.7-3. The subsystem is designed to operate over a voltage range of 22 - 34 VDC. On-orbit power requirements are TBD; however, it is anticipated that the mission average power requirement for the subsystem (thermal control excepted) will be less than 1.0 W.

7.7.5.4 Thermal

Spacecraft interface temperatures for propulsion components are TBD. Based on previous experience, however, it is anticipated that the following heater powers will be required for RCS components.

| Component | Watts (@ 28 VDC) |
|---|---|
| Propellant tanks | 5.0 |
| PDM | 2.5 |
| FDM | 1.0 |
| Propellant lines | 10.0 |
| Valve heaters | 3.5 |
| Catalyst bed heaters | 1.0 |

7.7.6 Subsystem Performance Data

Performance data for the 5.0 and 0.1 $lb_f$ thrusters are provided in Figures 12 and 13.

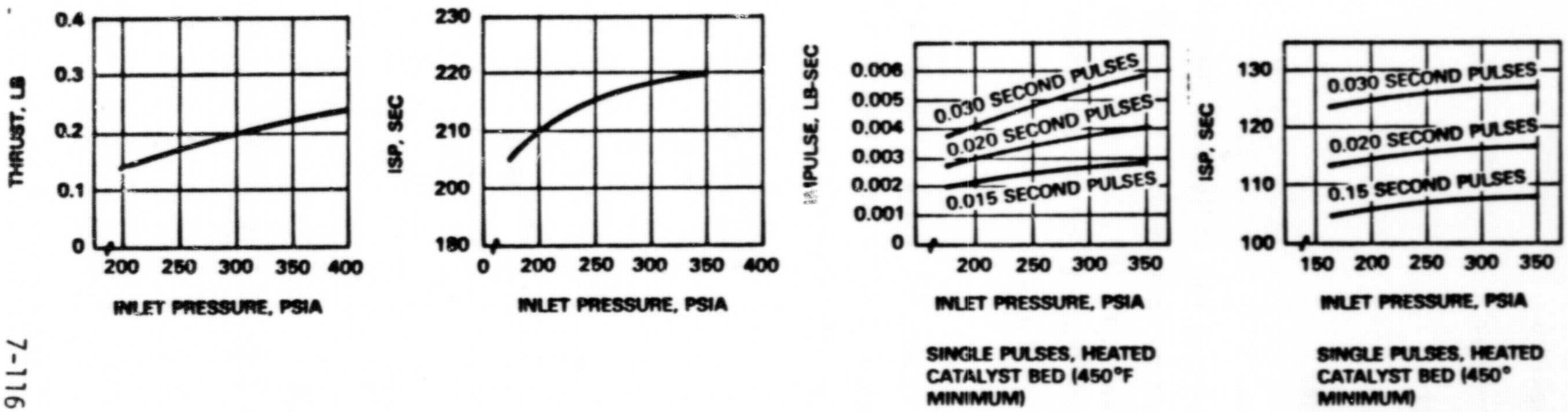



Figure 7.7-12

0.1 LB THRUSTER PERFORMANCE DATA

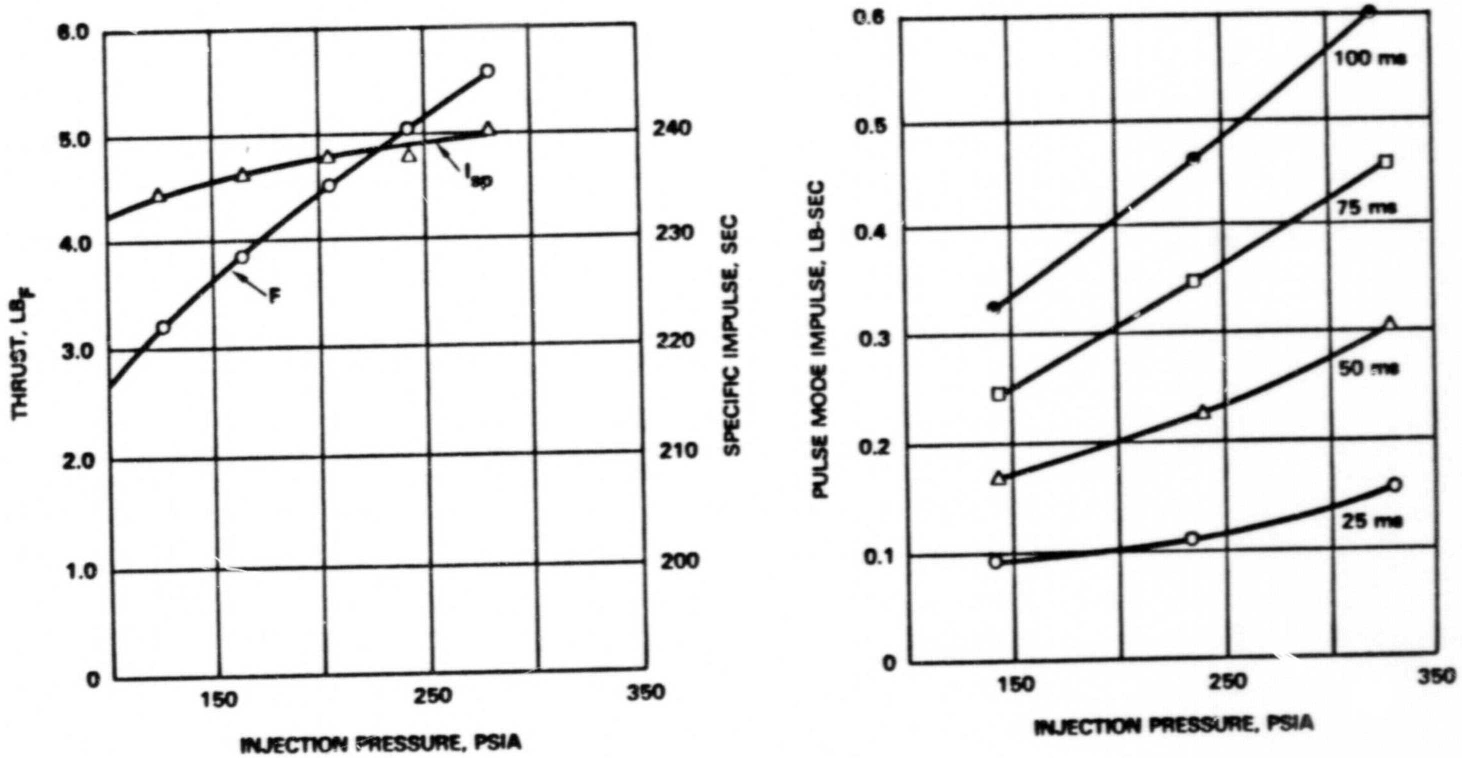



Figure 7.7-13

5.0 LB THRUSTER PERFORMANCE DATA

## 7.8 ATTITUDE CONTROL SUBSYSTEM

### 7.8.1 Subsystem Functions

The attitude control system is required to orient the spacecraft spin axis for transfer orbit injection motor firing, midcourse correction using high-level control jet firings, Mars orbit insertion using the solid propellant insertion motor, orbit adjustment using the high-level control thrusters, despun platform pointing for proper orientation of the scientific instruments, and coordination of spin axis direction, platform phase, and gimbal angles for pointing of the high gain antenna axis toward earth. During transfer orbit and trajectory correction maneuvers, medium gain antennas will be pointed along the positive Z-axis to provide optimum coverage patterns during this portion of the flight. In addition the ACS subsystem is required to control spin rate during all flight phases and to provide attitude sensing commensurate with the pointing precision requirements.

### 7.8.2 Subsystem Requirements

The attitude control accuracy required during each flight phase is governed by the greatest pointing error allowed for successful engine firings, communication antenna pointing, and scientific payload orientation. Table 7.8.2-1 lists these requirements. Spin speeds are adopted to provide effective compromises between structural integrity, thermal averaging, motor capacility, difficulty of pointing upset, propellant conversation and science payload accommodation.

The baseline Attitude Control Subsystem (ACS), includes the components listed in Table 7.8.2-2, configured as in Figure 7.8.2-1.

### 7.8.3 Subsystem Options

Alternatives to the sensor/actuator array shown in Table 7.8.2-2 includes the elements shown in Table 7.8.3-1. The reasons for the choice of baseline elements are noted in the table.

### 7.8.4 Subsystem Requirements

All ACS sensors and actuators, except for the Biax Drive Assembly, are mounted on the spinning section of the spacecraft.

Table 7.8.2-1

ATTITUDE CONTROL SUBSYSTEM

REQUIREMENTS

($1\sigma$)

| Flight Phase | Spin Speed (rpm) | Point Error (Deg) | Platform Error (Deg) | Antenna Error (Deg) | Req. Source | Pred. Perf. (Deg) |
|---|---|---|---|---|---|---|
| Deployment from STS | 2 | 0.6 | NA | NA | Fire Att. | |
| Spin Up | 2-60 | 1.0 | NA | 45 | Omni Ant. | 1.0 |
| Upper Stage Fire | 60 | 1.0 | NA | 45 | Omni/$\Delta V$ | 0.1 - 1.0 |
| Transfer Orbit | 60 | 5.0 | NA | 45 | Omni/$\Delta V$ | 1.0 |
| Midcourse Fire | 60 | 1.0 | NA | 45 | Omni/$\Delta V$ | 0.1 |
| Injection Fire | 60 | 1.0 | NA | 45 | Omni/$\Delta V$ | 0.1 |
| Orbit Trim | 60/9 | 1.0 | NA | 45 | Omni/$\Delta V$ | 0.1 |
| Scientific Operations | 9 | .07 | .07 | 0.25 | HGA/Exp. | .08 |
| Turnover | 9 | -- | NA | 0.25 | NA | 0.1 |
| Environmental Protection Fire | 9 | 1.0 | NA | 45 | $\Delta V$ | 0.1 |

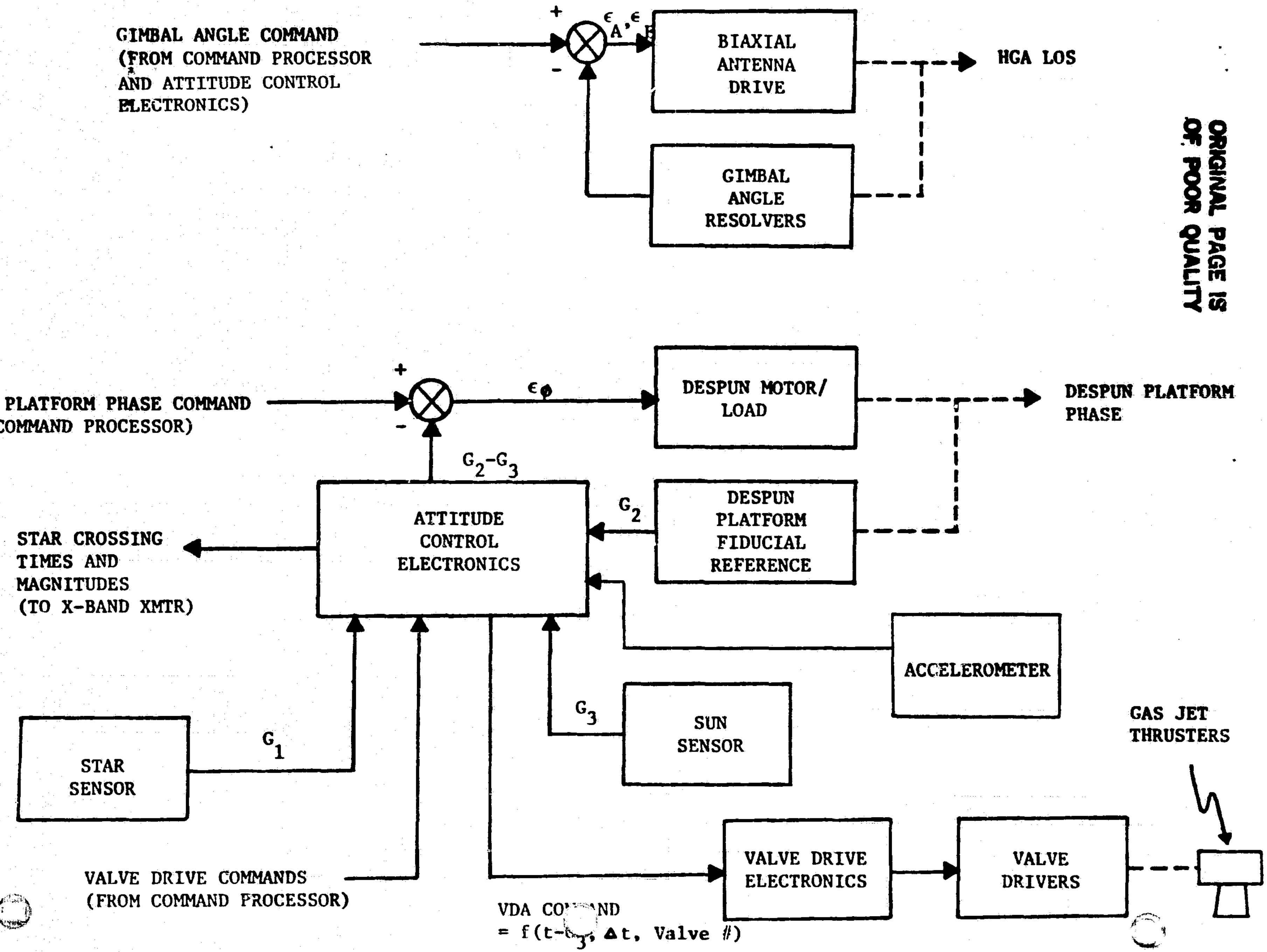
Table 7.8.2-2

ATTITUDE CONTROL EQUIPMENT LIST

| Component | Qty. | Unit Wt. kg | Total Wt. kg | Unit Pwr. Watts | Dimensions cm | Heritage | | |
|---|---|---|---|---|---|---|---|---|
| | | | | | | Program | Vendor | Status |
| Despun Mechanical Assy | 1 | 12.3 | 12.3 | 8 | 14.6 dia x 45 long | DSCS II | Ball Aerospace | Purchased Component |
| Biax Drive Assy | 1 | 6.5 | 6.5 | 42 | 17 dia x 34 long | DSCS II Landsat | | Modification of Existing Technology |
| Star Sensor | 1 | 2.95 | 2.95 | 1.9 | 49.5 x 45.7 x 10.2 | ISPM | Ball Aerospace | |
| Sun Sensor | 1 | 0.32 | 0.32 | 0.5 | 5.7 x 5.7 x 6.3 | Pioneer | Honeywell | Purchased Component |
| Control Electronics Assy | 1 | 4.6 | 4.6 | 10 | 35 x 20 x 23 | DSCS II | | Adaptation of Existing Technology |
| Valve Driver Assy | 1 | 1.4 | 1.4 | 2 | 18 x 23 x 5 | Pioneer | | |
| Wobble Damper | 1 | 0.23 | 0.23 | --- | 5.1 dia x 11.4 long | Pioneer | | |
| Accelerometer | 1 | 0.5 | 0.5 | --- | --- | --- | | Purchased Part |

FIGURE 7.8.2-1

ATTITUDE CONTROL SUBSYSTEM DIAGRAM

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| Item | To Replace | Reason for Not Adopting |
|---|---|---|
| Star Tracker | Star Sensor | Needs added gimbal drive or else mounting on despun platform + high-rate computation or data transmission. |
| Horizon Sensor | Earth Observation to Define Orbit Elements | No existing type has been applied in Mars mission -- observable horizon is uncertain; expected signal level is uncertain. |
| Tracking Antenna | Ground Computation of Earth Direction in Despun Platform Coordinates | Antenna feed on existing S/C is not monopulse type. Orbit rate is much smaller than coning rate for con-scan types. |
| Earth Sensor | Ground Computation of Earth Direction in Despun Platform Coordinates | Requires new design, additional mount to HGA -- would be required only if HGA pointing requirements were more severe. |

Table 7.8.3-1

REJECTED ACS OPTIONS

The sensors include a star scanner, a sun sensor assembly with internal redundancy, a despun platform fiducial reference (with respect to the spinning section), and biax drive gimbal axis resolvers. Both star scanner and sun sensor data are preprocessed, coded, and telemetered to earth for computation of spin axis pointing direction. Earth commands for phasing the fiducial reference properly with respect to either the sun sensor pulse or with respect to any star crossing pulse will maintain proper pointing of the despun platform. A gimbal angle command history will be parametrically transmitted to keep the high gain antenna pointed at earth. Spin axis maneuvers will require a transmitted parametric history of valve selections and operating times relative to sun-crossing pulse times.

The actuators include the despun motor, the biax assembly drive motors and the reaction control valve drivers. The valve drivers deliver vaporized hydrazine to pairs of thrusters to produce either "pure" forces or moments on the spacecraft. Degraded performance is available by using fewer thrusters. The thruster configuration includes 8 thrusters of 22 N nominal thrust (5 pounders) for spin, despin, and axial thrusting both forward and aft. Another 4 thrusters provide 1.87 N-m transverse moment for spin axis axis pointing control (0.445 N thrust x 4.2 m moment arm).

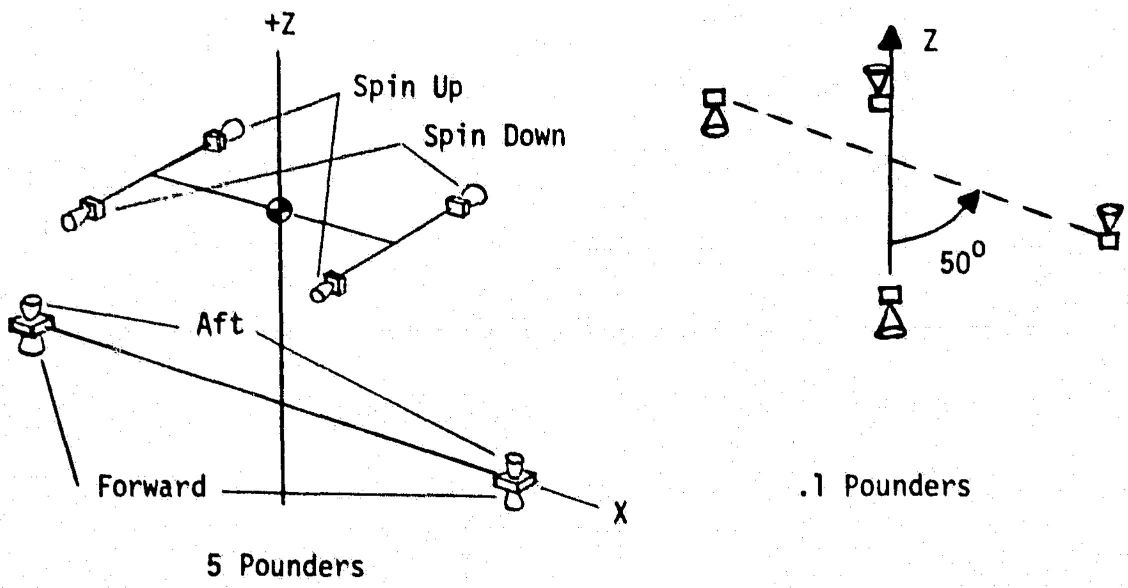



Figure 7.8.4-1

ACS THRUSTER ARRANGEMENTS

This torque level acting in 0.1 sec pulses on the nominal spacecraft momentum yields a minimum pulse-pair motion of about 0.023 deg at BOL, .031 deg at EOL. The low level thrusters will be spaced at 130$^{\circ}$ about the spin axis so as to minimize residual precession following firing of matched pulse pairs.

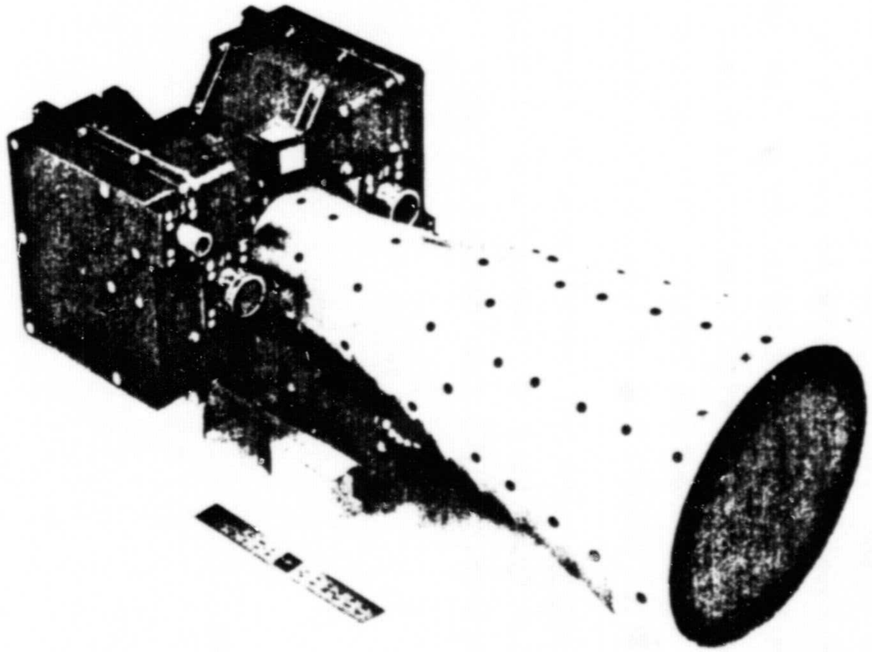
7.8.4.1 Star Scanner. The ACS Star Scanner assembly is a BASD type CS-203 modified as for the ISPM mission. The modified sensor is a solid state type, sensitive to stars as faint as 2.2 silicon magnitude at 60 rpm, and allowing determination of spin axis direction to within 0.017 degrees after averaging position signals for ten revolutions. The scanner mechanical layout is shown in Figure 7.8.4-2. A sunshade allowing operation to within 50 degrees of the sun is shown in Figure 7.8.4-3. The unit features redundant, independent detectors and associated electronics. The detector array, two isolated "V" pairs, is shown in Figure 7.8.4-4. There are 87 stars brighter than 2.2 MSi, allowing an average of 6.01 detections per revolution of a 10$^{\circ}$ wide swath inclined 47.5$^{\circ}$ to 57.5$^{\circ}$ to the spin axis with an unobstructed field, or 3.43 per revolution with a 43 per cent obstruction as in the climatology orbit (300 km) near Mars. Only two star detections per revolution are needed to yield the determination of spin axis direction to $\pm$ .017 degrees three sigma. Star magnitudes and crossing times are averaged on board with respect to sun crossing time as indicated by one of the two on board sun sensors. The averaged data is telemetered for ground station computation of spin axis direction.

7.8.4-2 Sun Sensor. The sun sensors shown in Figure 7.8.4-5 are Honeywell Radiation Center types used on Pioneer F and G spacecraft. Their characteristics are compared with MOS requirements in Table 7.8.4-1.

| | Value | Requirement |
|---|---|---|
| FOV | 174 degrees | 90 degrees |
| Range | .9 - 5.5 AU | 1 - 1.7 AU |
| Spin Speed | 2 - 85 rpm | 2 - 60 rpm |
| Accuracy | 2 - 30 minutes | none (on aspect) |
| Redundancy | 2 channels | none |

Table 7.8.4-1

SUN SENSOR CHARACTERISTICS



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# CS-203

## STAR SCANNER FOR IUS

SPECIFICATIONS

- FIELD OF VIEW: 4.6° ELEVATION
- SENSITIVITY: +1.56 MAGNITUDE
- SPIN RATE: 0.2 TO 1.0 DEG/SEC
- ACCURACY: 18 ARC SEC
- POWER CONSUMPTION: 6 WATTS
- DUAL REDUNDANT

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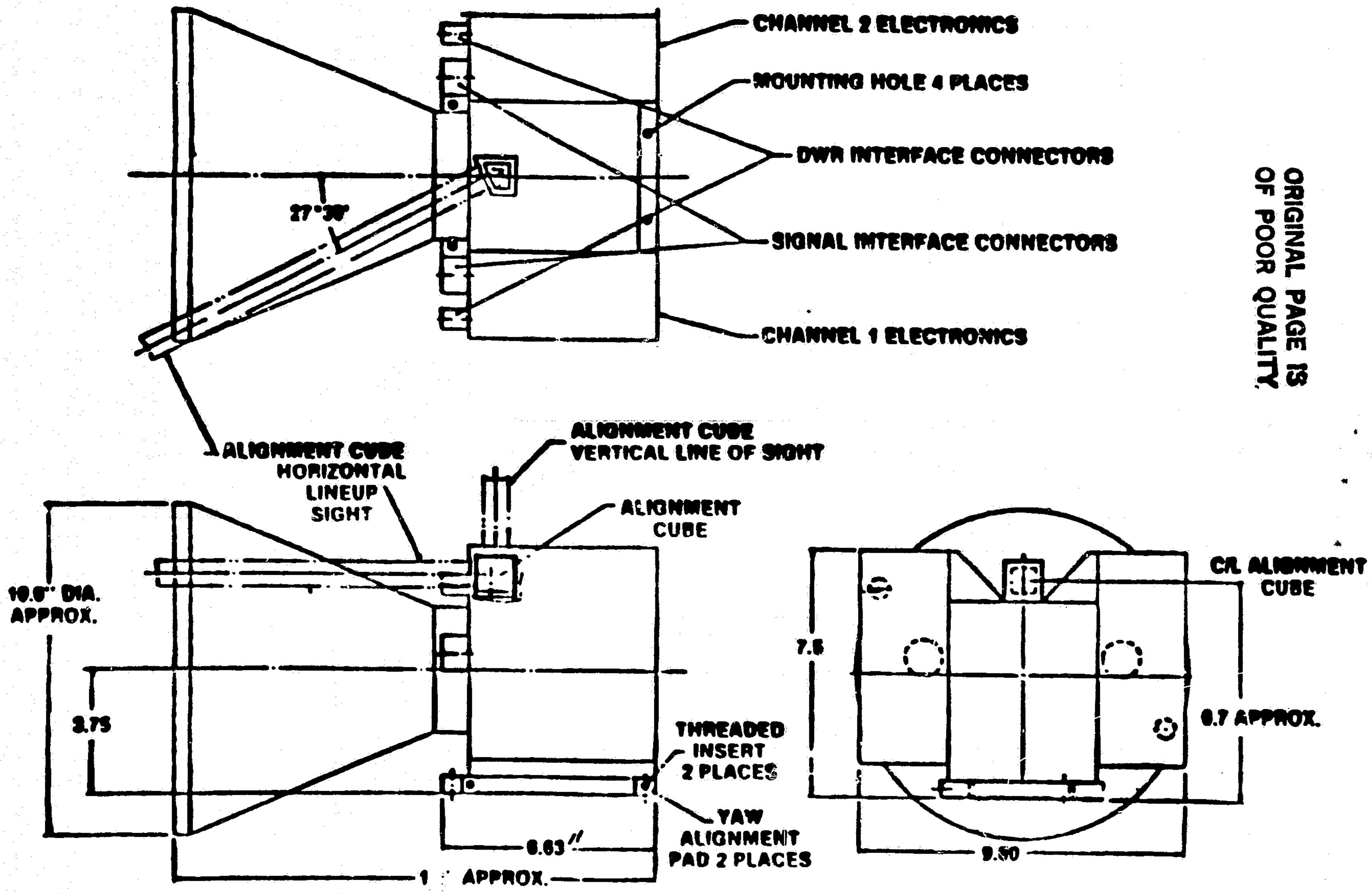


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Figure 7.8.4-2

STAR SCANNER MECHANICAL LAYOUT

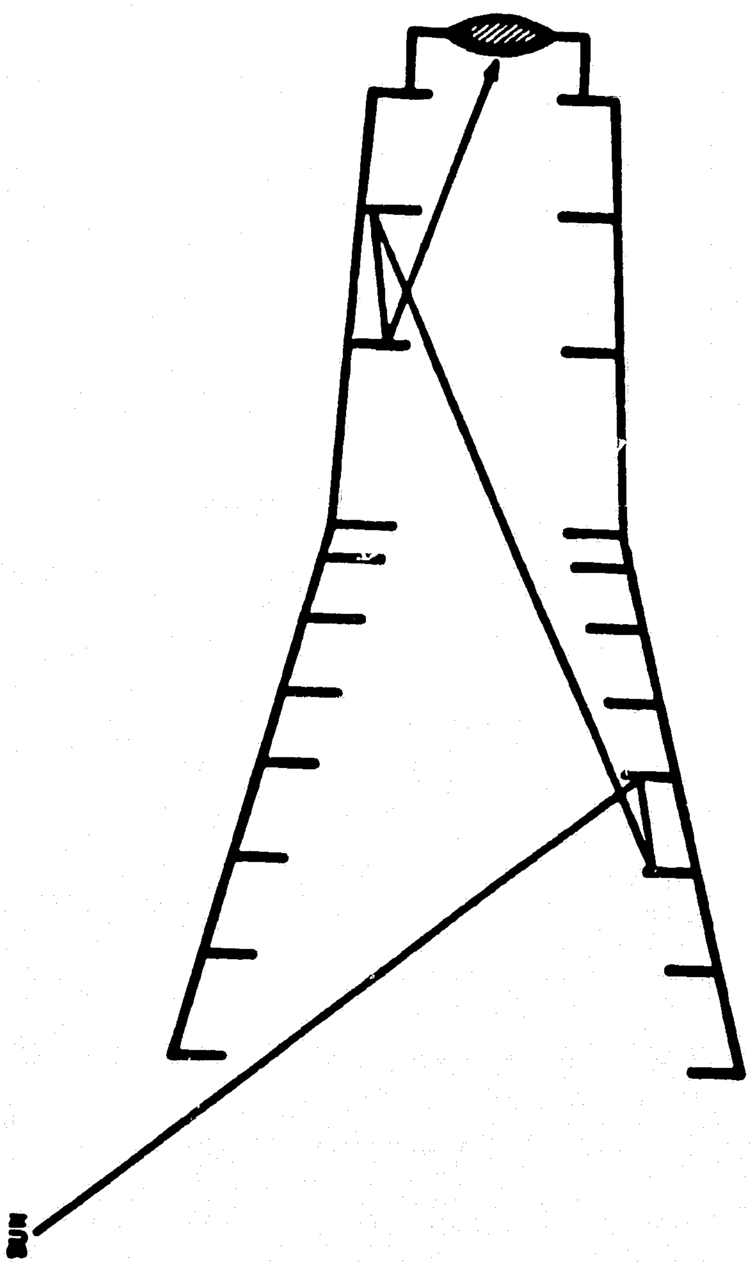


Figure 7.8.4-3

SCANNER SUNSHADE SCHEMATIC

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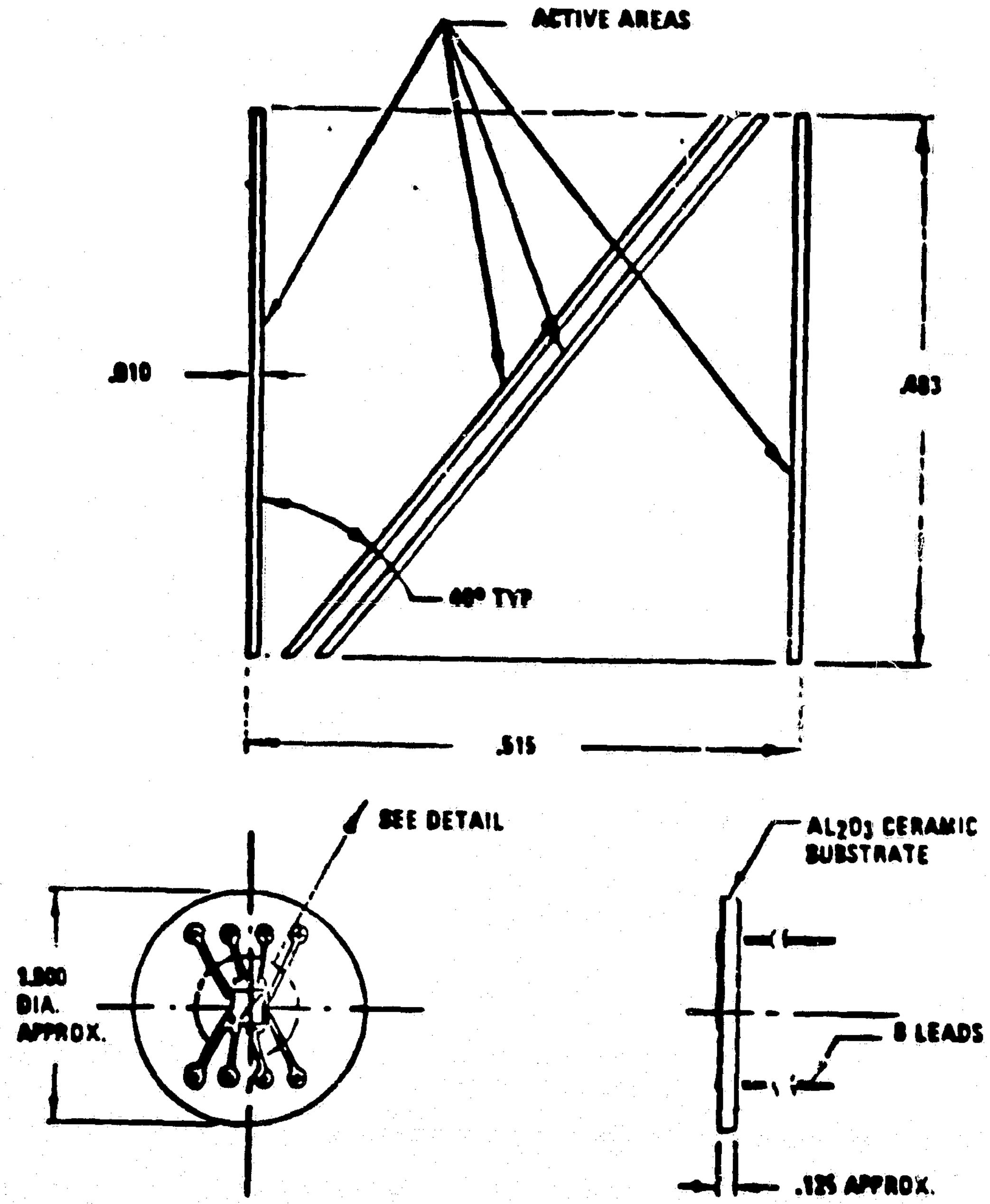


Figure 7.8.4-4

PHOTODETECTOR ARRAY

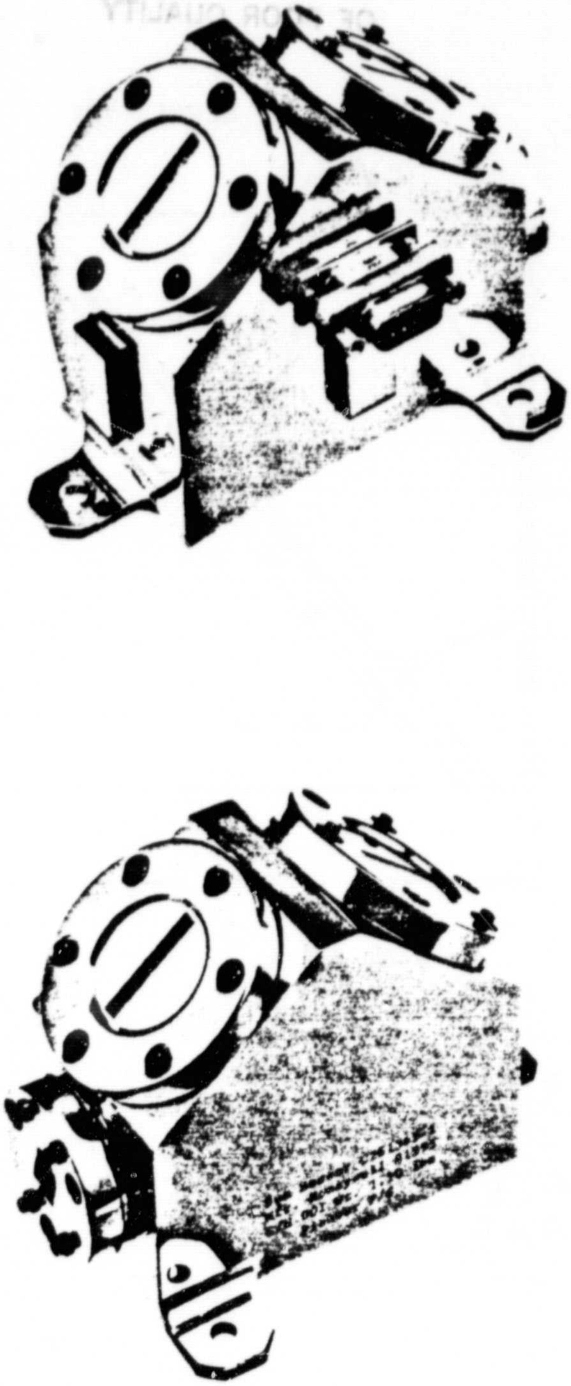

Figure 7.8.4-5

PIONEER F&G SUN SENSOR

7.8.4.3 Wobble Damper. The "linear" damper used as part of the mast mount on Pioneer is effective in removing momentum transverse to the spacecraft spin axis when mounted on either the spinning or despun portions of the spacecraft. Since the damper is designed to respond only to transverse velocity components, and since these components are identical for the two parts of the spacecraft, the damper will work equally well mounted on the despun platform for the climatology mission or mounted on the spinning shell of the aeronomy mission craft.

To obtain 99 per cent damping of transverse motion in ten minutes time, the damper must produce torque of

$$c \geq \frac{-400 \text{ kg - m}^2 \log .01}{600 \text{ sec}} \cdot \frac{\pi \text{ rad}}{180 \text{ deg}} = 0.054 \frac{\text{N-m}}{\text{deg/sec}}$$

Note that a passive damper will move the momentum vector toward the spacecraft axis of maximum moment of inertia. It must, therefore, be restrained during early flight when the spacecraft has a prolate inertial ellipsoid. During on-orbit operations, the spacecraft configuration always has its axis of maximum moment of inertia along the spin axis. The wobble damper is thus effective in forcing the momentum and spin axes into coincidence following on-orbit maneuvers.

### 7.8.5 Subsystem Performance

The shuttle deployment mechanism will be required to release the spacecraft/upper stage combination to the specifications shown in Table 7.8.5-1.

| | |
|---|---|
| Axial Spin Rate (Minimum) | 2 rpm |
| Tipoff Rate (Maximum) | .01 deg/sec |
| Separation Speed (Minimum) | .3 m/sec (1.0 ft/sec) |
| Axial Pointing Error (Maximum) | 0.6 degree |

Table 7.8.5-1

STS DEPLOYMENT REQUIREMENTS

The above specifications will allow MOS spinup to 60 rpm by firing two 22 N thrusters for 7 minutes, 50 seconds starting 200 seconds after deployment from STS. The sequence of events and pointing errors resulting are shown in Table 7.8.5.2.

| Event | Time min/sec | Spin Speed rpm | Shuttle Clearance m | Pointing Error deg |
|---|---|---|---|---|
| Deployment | 0/0 | 2 | 0 | .5 |
| Start Spin-up | 3/20 | 2 | 60 | 0.7 |
| Stop Spin-up | 11/10 | 60 | 200 | 1.0 |
| Fire Upper Stage | 45/0 | 60 | (established by STS-back-off) | 1.0 |

SEQUENCE AFTER DEPLOYMENT FROM SHUTTLE

Table 7.8.5-2

Following deployment, the spacecraft/upper stage spin axis will be moving about a cone of no more than 0.5 degrees about an inertial axis no more than 0.5 degrees from ideal upper stage thrust direction. The more than 30 minutes between end of spin-up and start of upper stage firing will allow telemetry of sun sensor pulses and star tracker pulse times and magnitudes to determine motion parameters and command firing of transverse jets for precession reduction and/or spin axis direction change.* 22 N thrusters will be fired, if necessary, in pairs to produce nearly 75 N-m torque about the spacecraft transverse axes. At 60 rpm and 60 degrees firing angle, a transverse pulse pair will change the transverse component of total momentum by nearly 25 N-m-sec. Total momentum under these conditions will be nearly 35,200 N-m-sec, so that a maximum duration pulse pair firing will be equivalent to a momentum vector direction change of .04 degrees with both longitudinal jet pairs operating, this can be done every revolution, so that the momentum direction can be changed at a rate of .04 deg/sec, or less than 30 sec to correct the greatest anticipated momentum pointing error with, perhaps, another half minute

* This assumes deployment attitude errors require correction, and the spacecraft implements two-way communications to facilitate it. This is not the baseline spacecraft design.

required to align the spin axis with the momentum vector (remove the transverse component of momentum). Jet selection and firing times will be ground commanded referenced to either a sun-crossing pulse from the sun sensor or else to a star crossing pulse from the star scanner. At 215 sec $I_{sp}$ for $N_2H_4$, the attitude correction process will consume at most

$$w_f = 60 \text{ sec} \times 2 \text{ jets} \times 22 \text{ N/jet} \div (9.80665 \tfrac{\text{N}}{\text{kg}} \times 215 \text{ sec}) = 1.25 \text{ kg}$$

The spin-up process will require a total impulse of

$$H_t = (60\text{-}2) \text{ rpm} \frac{2\pi \text{ rad}}{\text{rev}} \cdot \frac{1 \text{ min}}{60 \text{ sec}} \cdot 5600 \text{ kg-m}^2$$

$$= 34{,}000 \text{ N-m-sec}$$

which with a moment arm of 3.4 m (estimated), corresponds to a firing time of nearly 7 minutes 50 seconds, and a hydrazine expenditure of 9.82 kg.

Midcourse attitude maneuvers of up to 180 degrees are performed on the much lighter configuration following separation from the upper stage motor and may be conveniently combined with the on-orbit attitude maneuvering budget.

Assuming an average 730 kg-m$^2$ moment of inertia of the spun portion of the spacecraft about the spin axis, the nominal pulse pair change in spin axis direction will be

$$\Delta\theta_{NOM} = \frac{\Delta H}{H} = \frac{2 \text{ pulses} \times \text{time/pulse} \times \text{torque}}{\text{moment of inertia} \times \text{angle rate}}$$

$$= \frac{2 \times 0.1 \text{ sec} \times 0.44 \text{ N} \times 4.4 \text{ m} \times 60 \text{ sec/min}}{730 \text{ kg-m}^2 \times 9 \text{ rev/min} \times 2\pi \text{ rad/rev}} \times \frac{180^\circ}{\pi \text{ rad}}$$

$$= 0.033 \text{ deg/pulse pair}$$

Using differential firings of matched pulse pairs a resolution of perhaps 20% of the above value may be obtained, although that degree of precision in spin-axis pointing will not likely be required.

The range of values obtained for spin axis angular change based upon calculated decreases in moments of inertia of the spinning portion, and decreases in impulsive thrust over the course of the mission, are shown in Table 7.8.5-3.

The decreases in moments of inertia and thrust levels are due to the expenditure of propellant and propellant pressurization for the 3:1 blowdown propellant subsystem.

The gas consumption for the minimum spin axis direction change will be

$$\Delta w = \frac{\text{2 firings x impulse/firing}}{\text{specific impulse}}$$

$$= \frac{2 \times 0.44 \text{ N} \times 0.1 \text{ sec}}{9.80665 \text{ m/sec}^2 \times 215 \text{ sec}} = 4.17 \times 10^{-5} \text{ kg}$$

The specific hydrazine consumption is thus

$$\dot{w}_s = \frac{4.17 \times 10^{-5}}{0.033} = 0.0013 \frac{\text{kg}}{\text{deg}}$$

Assuming that no more than a total reorientation of five full rotations of the spin axis will be required during the spacecraft lifetime, the total hydrazine load for attitude control will be

$$W_T = 1800 \times 0.0013 = 2.34 \text{ kg}$$

of $N_2H_4$ at 215 sec average specific impulse.

Spindown of the injection configuration to 6.84 rpm will require another

$$W_{SD} = \frac{\text{momentum increment}}{\text{lever arm x specific impulse}}$$

$$= \frac{1000 \text{ kg-m}^2 \times (60 - 6.84) \text{ rev/min} \times 2 \text{ rad/rev}}{3.4 \text{ m} \times 215 \text{ sec} \times 9.80665 \text{ N/kg} \times 60 \text{ sec/min}}$$

$$= 0.78 \text{ kg}$$

ACS propellant requirements are thus assummarized in Table 7.8.5-4.

Table 7.8.5-3

CHANGE IN SPIN AXIS DIRECTION

(DEGREE/PULSE PAIR)

| | Beginning of Mission | End of Mission |
|---|---|---|
| Climatology | 0.054 | 0.017 |
| Aeronomy | 0.050 | 0.015 |

| Event | Requirement | $I_s$ ($kg\text{-}m^2$) | $w_f$ (kg) ($N_2H_4$ at 215 sec $I_{sp}$) |
|---|---|---|---|
| Spin Up | 2-60 rpm | 5600 | 9.82 |
| Active Damping | $\sim 2^o$ repoint | 5600 | 1.25 |
| Spin Down | 60-6.84 rpm | 1000 | 0.78 |
| Spin Axis Repoint | 1800 deg. | 730 | 2.34 |
| | | Total | 14.19 |

Table 7.8.5-4

ACS PROPELLANT REQUIREMENTS

7.8.5.1 Scientific Instrument Pointing and Control. Most severe pointing requirements for the instrument complement and different missions is shown in Table 7.8.5-5. These requirements, along with the various sources of pointing errors, establish the pointing error budget for the system. Since spin axis determination computations and despun platform phase commands are performed as ground operations, the error budget for the overall system is comprised of not only those errors associated with the space portion, but errors associated with ground processes as well.

For example, spin axis direction can be determined to within 0.017 degrees after averaging star scanners position signals for ten revolutions, as stated earlier. This value only considers the characteristics of the star scanner and does not account for other errors that may be associated with spin axis determination. In order to control pointing of instruments to $0.08^{o}$ in roll or yaw of the spacecraft velocity vector, other errors cannot exceed an rss value of 0.078 degrees.

In pitch, star scanner timing error is 0.017 degrees and despun platform fiducial accuracy is estimated to be 0.01 degrees. Other errors can then be as great as 0.77 degrees (rss) as indicated in the table.

7.8.5.2 High Gain Antenna Pointing Accuracy. In order to meet the communications link signal to noise and data rate requirements, a high gain antenna pointing error requirement of $\pm$ 0.75 degree ($3\sigma$) was established and subsequently used in all link analyses.

Since the error budget associated with pointing of the despun platform in roll, yaw or pitch provides for a pointing error of not more than 0.08 degrees and single axis gimbal angle error (azimuth or elevation) is estimated to be no more than 0.15 degrees, the resulting $1\sigma$, antenna pointing error is azimuth or elevation is 0.17 degrees as indicated in Table 7.8.5-6.

When both axes are considered, the $1\sigma$ pointing error is

$$\sigma = \sqrt{\sigma_A^2 + \sigma_e^2} = \sqrt{(0.17)^2 + (0.17)^2} = 0.24 \text{ degree}$$

or, within the antenna pointing accuracy requirement.

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Table 7.8.5-5

INSTRUMENT POINTING ($1\sigma$)

REQUIREMENT - (BASED ON MOST SEVERE POINTING REQUIREMENT OF INSTRUMENT COMPLEMENT) (DEG)

| | Roll | | Pitch | | Yaw | |
|---|---|---|---|---|---|---|
| | C | K | C | K | C | K |
| Baseline Climatology (PMR) | 1 | 0.2 | 1 | 0.2 | 1 | 0.2 |
| C1 (FIS + UV03) | 1 | 0.2 | 0.08 | 0.08 | 1 | 0.2 |
| C2 (FPI) | 0.5 | 0.1 | 0.5 | 0.1 | 0.5 | 0.1 |
| C3 (MSM) | 0.08 | TBD | 0.08 | TBD | 0.08 | TBD |
| Aeronomy | 0.5 | 0.1 | 0.5 | 0.1 | 0.5 | 0.1 |

| Source | Error: Roll, Yaw | Error: Pitch |
|---|---|---|
| Star Scanner | 0.017° | |
| Star Scanner Timing | | 0.017° |
| Despun Platform Alignment and Timing | | 0.1° |
| All Other Errors (Allowable) | 0.078° | 0.077° |
| Tightest Requirement | 0.08° | 0.08° |

Table 7.8.5-6

HIGH GAIN ANTENNA POINTING ACCURACY

Requirement: ± 0.75° ($3\sigma$)

± 0.25° ($1\sigma$)

| Source | Error |
|---|---|
| Gimbal Angle Error (Az, or El.) Resolver Linearity, Resolution, etc. | 0.15° |
| Despun Platform Orientation (Roll, Yaw or Pitch) | 0.18° |
| Antenna Pointing Error (Azimuth or Elevation) | 0.17° |

# 8. SYSTEM INTERFACES

## 8.1 SCIENCE INTERFACES

A summary of requirements for the two baseline missions and how the spacecraft design meets these requirements is given in Figure 8.1-1. Discussion of the various system performance parameters have been given in Section 6.3. In particular mass properties, data handling and communications, and pointing accuracy have been detailed. Other comments in thermal, viewing directions, and specific interface areas follow below.

The guidelines for meeting the instrument thermal requirements are summarized in Figure 8.1-2. The results of the thermal design meets the rather broad operating temperature limits of each instrument. However, it was felt from the beginning that individual instruments could not successfully operate if wide thermal fluctuations occurred on an hour-to-hour basis, even if these fluctuations were within the required temperature limits. This is especially true for optical instruments such as the FPI in which sizable thermal gradients across the instrument would most probably make the instrument unusable. Thus an effort was made to keep thermal swings between day and night to a minimum. For the Climatology Mission variation was kept to within ±1° over any orbit, with long term drifts over the Martian year somewhat greater. Estimates for the Aeronomy Mission were more difficult to make because of the wide variations in eclipse periods and distance from the planet surface. Under worst case conditions, the temperature drift over any orbit has been calculated as ±3°C.

The three thermal radiators required for the Climatology Mission presented additional design problems. The FIS and PMR radiator requirements are less stringent than the GRS requirement and this are more easily met. Since both radiators can view the planet surface we have assumed that they could also view a spacecraft surface that is at or below the average Martian surface temperature. In particular the whole aft end of the spacecraft is covered with multilayer insulation and all heat is radiated out the upper end and outside surfaces of the spacecraft. Because of the sun-synchronous orbit solar radiation will never enter the underside of the

## GUIDELINE FOR MEETING THE INSTRUMENT THERMAL REQUIREMENTS

1. PERFORM THERMAL DESIGN TO ACHIEVE ACCEPTABLE OPERATING TEMPERATURE FOR EACH INSTRUMENT

2. DETERMINE THERMAL GRADIENTS AND TEMPERATURE EXCURSIONS FOR EACH INSTRUMENT

3. IDENTIFY INSTRUMENT ACCOMMODATION THERMAL PROBLEMS AS AREAS FOR FUTURE STUDY

4. EACH INSTRUMENT WILL CONTAIN A THERMISTOR FOR MAKING POWER-OFF TEMPERATURE MEASUREMENTS

Figure 8.1-1

## SUMMARY OF SCIENCE ACCOMMODATION

| | AERONOMY | | CLIMATOLOGY | |
|---|---|---|---|---|
| | DEPENDENT | CAPABILITY | REQUIREMENT | CAPABILITY |
| MASS (KG) | 53.5 | NOTE 1 | 37 | NOTE 2 |
| POWER (W, MAX) | 46.5 | 84 (NOTE 3) | 59 | 73 (NOTE 4) |
| POINTING (DEG) | 0.5/0.1 | $0.08^0$ | $1^0/0.2^0$ | $0.08^0$ |
| RAM VIEWING | $\pm 30^0$ | $\pm 20^0$ | -- | -- |
| DATA RATE (BPS, MAX) | 2048 | 4096 | 1284 | 1540 |
| THERMAL ($^0$C) | -20 TO +40 | -15 TO +35[6] | -20 TO +40 | -15 TO +35[7] |
| RADIATORS | NONE | -- | THREE | NOTE 5 |

ALL POINTING DIRECTION, FIELD OF VIEW, AND MOUNTING REQUIREMENTS ARE SATISFIED IN BASELINE DESIGN.

NOTES:

1 - A 20% PAYLOAD MARGIN IS INCLUDED IN BASELINE DESIGN BUT UP TO AN ADDITIONAL 140 KG IS AVAILABLE DEPENDING ON PROPELLANT REQUIREMENTS FOR DRAG COMPENSATION

2 - A 20% PAYLOAD MARGIN IS INCLUDED IN BASELINE DESIGN BUT AN ADDITIONAL 37.6 KG IS AVAILABLE WITHIN STAR 37XF CAPABILITY

3 - A 38 WATT WORST CASE MARGIN IS AVAILABLE TO THE S/C. ADDITIONAL POWER IS EASILY OBTAINABLE BY INCREASING THE ARRAY AREA

4 - WORST CASE, ASSUMING EOM DEGRADATION

5 - GRS RADIATOR REQUIREMENT SATISFIED BY BOOM LOCATION. PMR AND FIS RADIATOR REQUIREMENTS ARE SATISFIED BY S/C THERMAL DESIGN. (RADIATORS SUPPLY SOLE SOURCE OF HEAT RADIATED FROM COMPLETELY BLANKETED AFT END OF S/C.)

6 - VARIATION OVER ANY SINGLE ORBIT IS $\pm 3^0$C

7 - VARIATION OVER ANY SINGLE ORBIT IS $\pm 1^0$C

Figure 8.1-2

spacecraft. Thus this area will be at or below the surface temperature of Mars since it views either the Mars surface or empty space. Thus the FIS and PMR radiators, which face into this region, will be viewing either space or MLI at a low temperature. Since there is no other source of heat in the aft region of the spacecraft, and the relative rotation between radiators and MLI presents almost the total MLI area to the radiators (thereby minimizing the possibility of local hot-spots), adequate cooling is provided.

An analysis of the GRS cooling requirements is best presented by reference to Figure 8.1-3. This figure represents the front hemisphere of the fields-of-view as seen from the center of the gamma ray spectrometer. (The back hemisphere is identical.) In the figure, the Martian surface is shown as the curve 23° below the local GRS horizontal. In the course of a day the sun follows a line from nadir to zenith and back again, but is blocked out (eclipsed) by the planet at the same 23° below the horizontal. In the course of a year the solar track moves back and forth within the indicated "SOLAR TRACK REGION." The GRS radiator FOV is shown on the figure with its requirement not to view the planetary surface or Sun. The problem now is to place the spacecraft within this figure without appearing in the radiator FOV. Since the GRS has been located on a 5-meter boom extending out from the despun platform the horizontal extent of the spacecraft is defined by two meridians in this figure at angles of 28° to the right and 24° to the left. The orientation of the boom on the despun platform relative to the ram direction determines where to place the lines of latitude defining the spacecraft location. In this direction the spacecraft entends some 46° for a 5 meter boom. Thus to keep the spacecraft as far away from the nadir viewing direction as possible and not impinge on the radiator FOV it was located as shown in the figure. In this position it does partially shield the planet surface from the instrument, although the region to the right is low in density, being mostly the total extent of the HGA locations. The far region to the left is also of relatively low density, containing only the spacecraft skirt and solar cells. In any case, as seen on the figure, the radiator FOV could be extended down another 5° and to left and right another 20° without viewing either sun or planet. With this increased area available, perhaps a notch could be made available in its FOV into

## ACCOMMODATION OF GRS FIELDS OF VIEW

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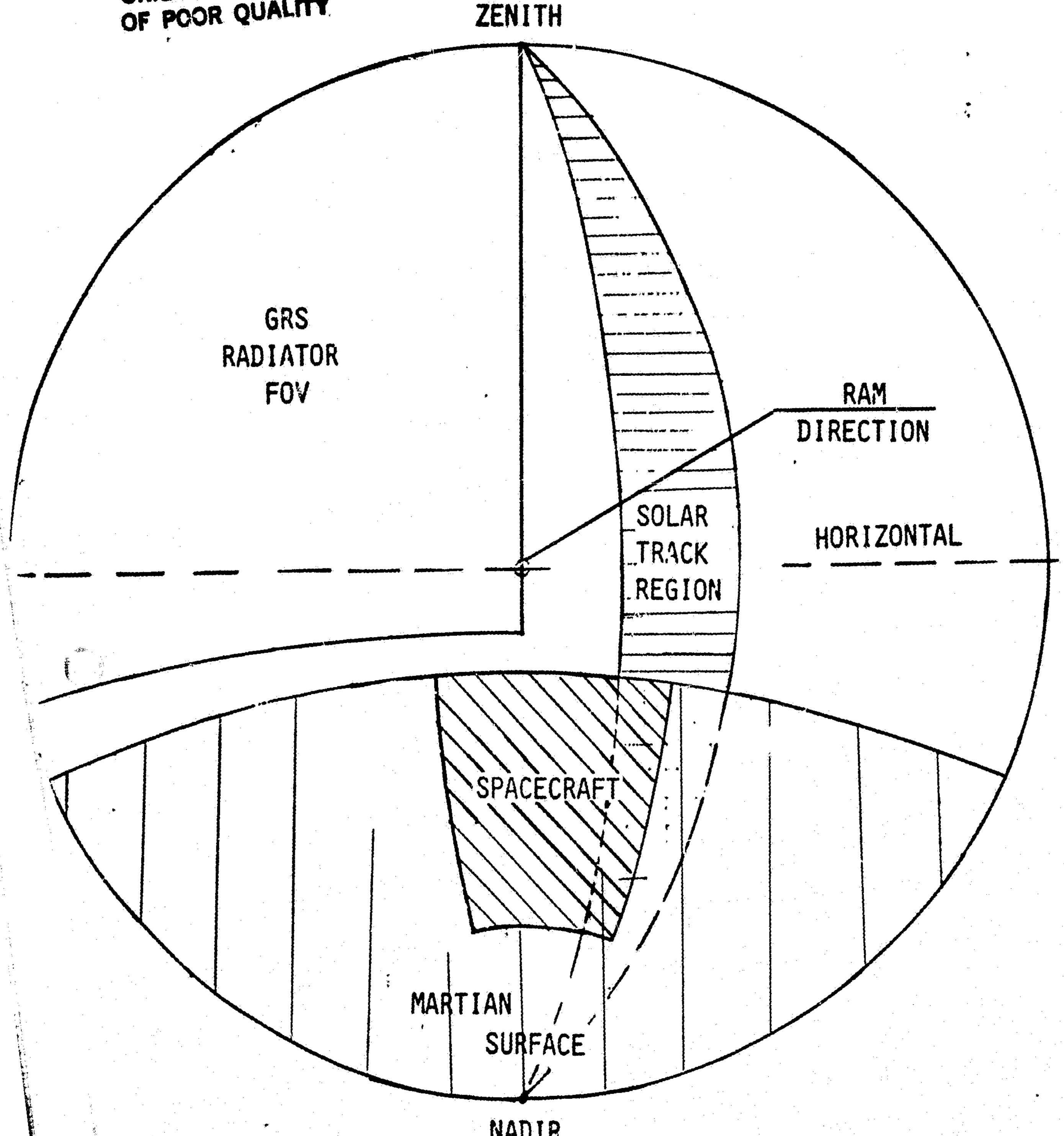


FIELDS OF VIEW AS SEEN FROM
CENTER OF GAMMA RAY SPECTROMETER

Figure 8.1-3

which the spacecraft could be moved, thereby removing the spacecraft from in front of the planet surface.

The requirement for the PMR to view the Mars polar regions from the non-polar Climatology orbit has been met by placing the PMR FOV at the edge of the despun platform, somewhat above its baseplane. For a 300 km orbit at 92.6° inclination this downward facing angle is 27°. With the FOV of less than 1° this look direction is easily met by the PMR placement.

The Aeronomy Mission has stringent requirements to minimize spacecraft interference effects (see Section 4). To meet these requirements special design efforts were instituted. For example, to achieve low spacecraft potentials, a significant portion of the spacecraft surface is covered with grounded conductors and non-conducting material areas are minimized. Solar cells, having non-conducting surfaces, are placed well away from the TIMS and RPA entrance apertures. Solar cell distribution has been made symmetrical to minimize irregularities in electric field patterns around the spacecraft and backwired to minimize residual magnetic fields. Outgassing is minimized in the vicinity of the NMS and TIMS to avoid non-ambient atmospheric contamination. Thrusters are so placed that dense-plume regions do not impinge on instruments or obscure their FOV's.

Requirements to point in the ram direction for the NMS, TIMS and RPA have been considered from two points of view. A straightforward approach is to rotate the despun platform on command, moving it so that it keeps within a few degrees of ram direction throughout the periapsis phase of the orbit. However, such an approach will require continual torquing of the despun platform based on predetermined attitude data, and will lead to slight changes in the spin period, possibly affecting data gathering and handling for the MAG, EFD, and ETP. Thus another approach suggests itself.

By rotating the despun platform an integral number of times per orbit the NMS, TIMS and RPA can be made to scan through the ram direction. For a circular orbit, one revolution per orbit, as with the Climatology mission, would keep the platform always pointing exactly ahead. For the highly elliptical Aeronomy orbit such a solution would not be successful in keeping the ram direction within the ±30° FOV on the instruments.

However, rotating at a rate of two or three times per orbit presents a viable alternative. In Figure 8.1-4 the pointing angle and pointing angle errors are shown as a function of time from periapsis. The solid curve in the top half shows the ram direction relative to the local horizontal for the 150 km by 3 Mars' radii orbit. The broken curves show the linear increase in angle with time for a constant rotation rate of twice and thrice per orbit, placing the phase with 0° in the ram direction at periapsis. The two solid curves on the bottom show the "mispointing" error. Note that for the two revolution per orbit case the mispointing error is within 20° for a full hour before and after periapsis. The three revolutions per orbit case matches almost exactly the ram direction near periapsis keeping the angular error at less than 5° for the ionosphere phase of the mission but getting way outside the 30° limit before the end of the ionosheath phase.

## 8.2 LAUNCH VEHICLE

The Space Transportation System puts the spacecraft into low earth orbit. The spacecraft and the Upper Stage are passengers on the Shuttle Orbiter.

The interface requirements of the Shuttle have been documented:

- JSC 07700, Volume XIV, "Space Shuttle System Payload Accommodations"
- NHB 1700.7, "Safety Policy and Requirements for Payloads Using the Space Transportation System (STS)"

The facets of the Shuttle most influential to the Mars orbiter spacecraft are dimensions, weight, data, and safety.

Of dimensions and weight, only the diameter of the Cargo Bay dynamic envelope is restrictive. Its diameter is 180 inches. We allow 5 inches on the diameter for all deformation and deflection on the payload side of the interface. The baseline solar array base diameter is 170 inches, leaving 5 inches for growth of the array.

Length and weight are not critical, as they do not approach the 60 foot and 65000 lb. capability. Of course it is desirable to restrain these two parameters to permit a shared payload on the Shuttle flight.

AERONOMY MISSION

POINTING IN THE RAM DIRECTION

ACHIEVING RAM POINTING CAPABILITY BY ROTATING INSTRUMENT POINTING PLATFORM AT UNIFORM RATE.

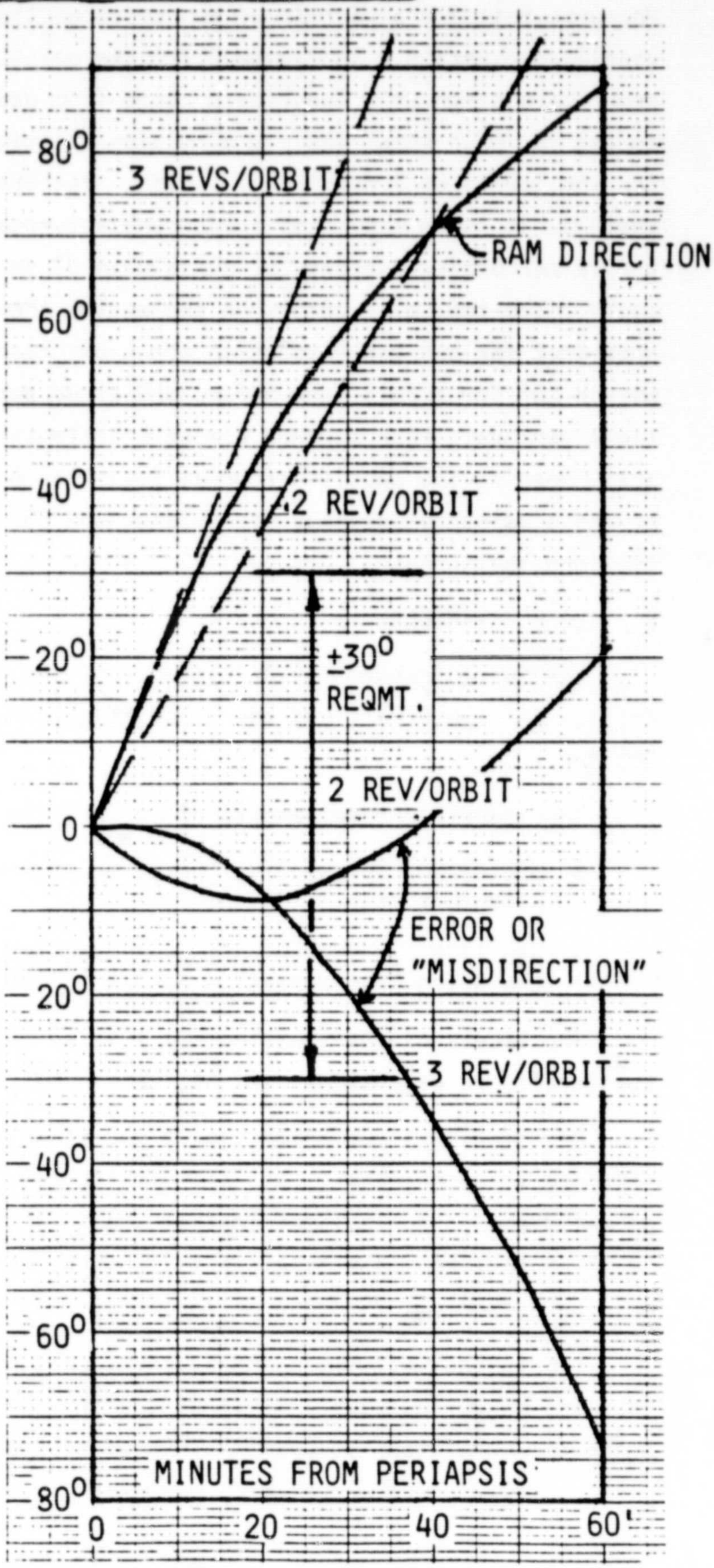


Figure 8.1-4

Shuttle data capabilities when the spacecraft is on board should be satisfactory for some simple operations: relay spacecraft engineering data to the ground; read out the contents of the spacecraft command store; load commands into the store as required. It may also be desirable for the spacecraft to execute commands to exercise it through various modes for test.

However, a most significant requirement is that of Shuttle Safety. If affects numerous aspects of the spacecraft design, including structure, propellant tankage and valving, appendages, and the RF system. Another area it affects is that of inhibiting the execution of commands which would jeopardize the one- and two-fault tolerant criteria for events which would be critical or catastrophic to the Shuttle if executed in the Cargo Bay. We have discussed earlier the proposed spacecraft approach for satisfying this safety requirement.

With regard to the Upper Stage, the Mission Requirement Documents offer a choice. The spacecraft concept of this report is compatible with the SRM-1 rocket motor, and specifically with its implementation as a spinning solid motor in the Intelsat VI perigee stage. Since this stage and its ASE have not yet been developed, we have assumed what some of the properties are:

Size: 91 inches diameter, 124 inches long

Propellant weight: 21,000 lbm

Deployment characteristics:

Upper Stage plus spacecraft are rolled out of the Cargo Bay by release from the cradle. This gives it both separation velocity and spin rotation at 2 rpm.

Deployment accuracy: Separated payload maintains desired orientation within ±2 degrees

Impulse accuracy: ±0.75 percent

| Operations | Responsibility |
|---|---|
| Up to and including deployment from the Cargo Bay | Upper Stage |
| Spinup from 2 to ~60 rpm | Spacecraft |
| Automatic nutation control: sensing and thrusting | Spacecraft |
| Electric power to ignite motor | Spacecraft |
| Separation: hardware and actuation | Spacecraft |
| Yo tumbling device, if necessary | Upper Stage |
| Telemetry | TBD |

## 8.3 TRACKING STATIONS

The spacecraft design is based on the following assumptions about the interface with the Deep Space Stations of the DSN.

DSN stations assigned to this mission must receive X-band, the only transmission from the spacecraft. They must transmit S-band, although S-band transmission is a must only when doppler tracking takes place and when commands are being sent to the spacecraft. Command loading normally takes no more than 2.2 minutes, and no more than 35 minutes in emergency; it is expected to take place perhaps as often as daily, perhaps no more than once or twice a week.

The doppler tracking based on X-band downlink coherent with the S-band uplink will be required at various intervals throughout the mission. During interplanetary cruise and during orbital operations, tracking may be required on a regular, but not too frequent basis -- perhaps one or two eight-hour passes per week -- for spacecraft orbit determination. At other times, particularly before and after trajectory correction maneuvers, it may be required more continuously. Right at the time of maneuvers, doppler information must be supplied for a real-time assessment of the spacecraft's propulsive performance. Certainly, near continuous coverage will be required in the first weeks of flight, and for ±2 days around Mars orbit insertion.

Note that ranging capability is included in the spacecraft, although this is not a mission requirement.

The assumed commitment of DSN stations to the project is one eight hour pass every 24 hours. For purposes of sizing the spacecraft's data storage and data transmitting capability, it is recognized that scheduling may not permit the same shift to be used each day, so the assumed maximum non-tracking period between two tracking periods is 24 hours, not 16 hours. This is primarily the commitment for downlink reception; as discussed above, uplink capability is not required for the full eight hour pass every day, and may not be required at all on some days.

The downlink data transmission capability at maximum range (2.66 AU) assumes a 64-meter antenna at the DSN station. However, there is no assumed commitment of a 64-meter antenna at all times in the mission. With the selected system, it is estimated that the 34-meter antenna would suffice out to 1.94 AU, nominal, or 1.65 AU, adverse tolerances. (If the maximum non-tracking period is limited to 16 hours, these ranges are extended to 2.24 and 1.91 AU, respectively.)

Even beyond these range limits, a 34-meter antenna will permit partial return of data. And when the range is less than the 34-meter limit, the availability of a 64-meter antenna will permit all data to be recovered in ~4 hours each day.

Technical details of the interface, for example antenna gains, power levels, carrier and subcarrier frequencies, etc., are taken from JPL Document 810-5.

## 8.4 MISSION OPERATIONS

This discussion of mission operations emphasizes the handling and treatment of information received from the spacecraft, and the generation and transmission of commands based on this information.

The mission operations must, of course, be compatible with the DSN support, as discussed in Section 8.3. The two activities must be planned and scheduled in mutual cooperation.

Emphasis here is on continuing and routine operations. Particular events such as injection from earth, TCM's and mass orbit insertion, are treated in Section 3.

8.4.1 Information From The Downlink Carrier

Doppler tracking of the downlink carrier provides the following information:

8.4.1.1 Orbit Determination (Interplanetary). Successively more refined estimates of the earth-Mars trajectory are deduced as the mission proceeds culminating in the best estimate of the arrival time and asymptote location (B) of the approach to Mars.

8.4.1.2 Orbit Determination (Mars Orbit). Successively more refined estimates of the elements of the orbit about Mars are calculated.

8.4.1.3 Thruster Performance. Doppler tracking permits calibrating and monitoring the performance of spacecraft thrusters.

8.4.1.4 Signal Strength. Carrier strength permits monitoring performance of spacecraft RF output, antenna characteristics and pointing accuracy.

8.4.2 Telemetry Reception

The telemetry channel of the downlink transmission provides the following data:

8.4.2.1 Science Data. Science data, the primary purpose of the mission, are received and forwarded to the investigators.

8.4.2.2 Experiment Housekeeping. Housekeeping data provide the state of health of the instruments and are used by mission planners as well as by the investigators.

8.4.2.3 Spacecraft Performance. Spacecraft engineering data permit analysis of all subsystem performance characteristics.

8.4.2.4 Attitude Determination. Star sensor data, included in the engineering transmission, permit determination on the ground of the orientation of the spacecraft's spin axis.

### 8.4.3 Mission Planning

The above information is utilized in planning subsequent spacecraft operations.

8.4.3.1 Trajectory Correction Maneuvers. Trajectory estimates (8.4.1.1) are used to plan trajectory correction maneuvers.

8.4.3.2 Orbit Change Maneuvers. Orbit estimates (8.4.1.2) are used to plan $\Delta V$ maneuvers to trim, adjust, and maintain the desired orbit at Mars.

8.4.3.3 Attitude Adjustment. Information on the orbit (8.4.1.2) and on the attitude (8.4.2.4) is used to plan attitude correction and adjustment operations. For the Aeronomy Mission (and Climatology optional orbit) this includes the planning of 180 degree flip maneuvers.

8.4.3.4 Propulsion Subsystem. Based on telemetry and the history of preceeding maneuvers, the state of the propulsion system is assessed periodically. Its performance to handle the above propulsive operations (8.4.3.1 to 8.4.3.3) is planned, and maneuver parameters as calculated.

8.4.3.5 Despun Platform Pointing. Based on orbit determination (8.4.1.2) and the scientific emphasis for succeeding orbits (feedback from 8.4.2.1) the despun platform pointing program is generated. Parameters to describe this program are calculated. The plan accounts for anticipated attitude changes (8.4.3.3).

8.4.3.6 HGA Pointing. Based on attitude measurements and planned changes (8.4.2.4 and 8.4.3.3) and the planned despun platform pointing (8.4.3.5) the high gain antenna pointing program is generated. (It includes a 360° "unwind" of the azimuth gimbal once per orbit.) Parameters to describe this program in terms of azimuth and elevation gimbal angles are calculated.

8.4.3.7 Spacecraft State. Spacecraft state changes, is necessary, are planned, based on analyses 8.4.2.2 and 8.4.2.3.

8.4.3.8 Experiment State. Experiment state changes are planned, based on scientific emphasis to be pursued (feedback from 8.4.2.1) and on experiment health data (8.4.2.2).

8.4.3.9 Tape Recorder and Bit Rate. The program of tape recorder record and playback cycles, and the bit rates to be employed for these cycles and for downlink transmission is planned, based on DSN station availability, occultation, predictions from 8.4.1.2, and planned science operations (8.4.3.8).

8.4.3.9 Battery. Occasional battery reconditioning operations are planned.

8.4.4 Commanding

8.4.4.1 Loading. Commands generated to effect the operations implied by Section 8.4.3 and their appropriate time tags are compiled and sent to the spacecraft to be loaded.

8.4.4.2 Verification. A command store dump is also commanded. If all commands check out, they will be left unchanged on the spacecraft for execution.

8.4.5 Typical Operations Cycles

Figure 8.4-1 illustrates typical 24-hour operating cycles for the Climatology and Aeronomy Missions. Tape recorder data state (increasing during record, decreasing during playback) are indicated, for a non-tracking period of 16 hours followed by an eight-hour tracking period. For the Climatology Mission, occultations and night/day alternations are shown, illustrating the 1.9-hour orbital period. For Aeronomy, the data bursts associated with periapsis passage (P) are shown, and a possible occultation near apoapsis.

8.4.6 Automatic Fault Response

In addition to the execution on schedule of each command loaded from the ground, the spacecraft will implement certain automatic fault response routines. These will enable the spacecraft to recover from situations that demand quick attention or involve loss of the communications links, up or down.

A list of candidate autonomous actions and the circumstances which would enable them is given in Table 8.4-1.

# TYPICAL 24 HOURS OF ORBITAL OPERATIONS

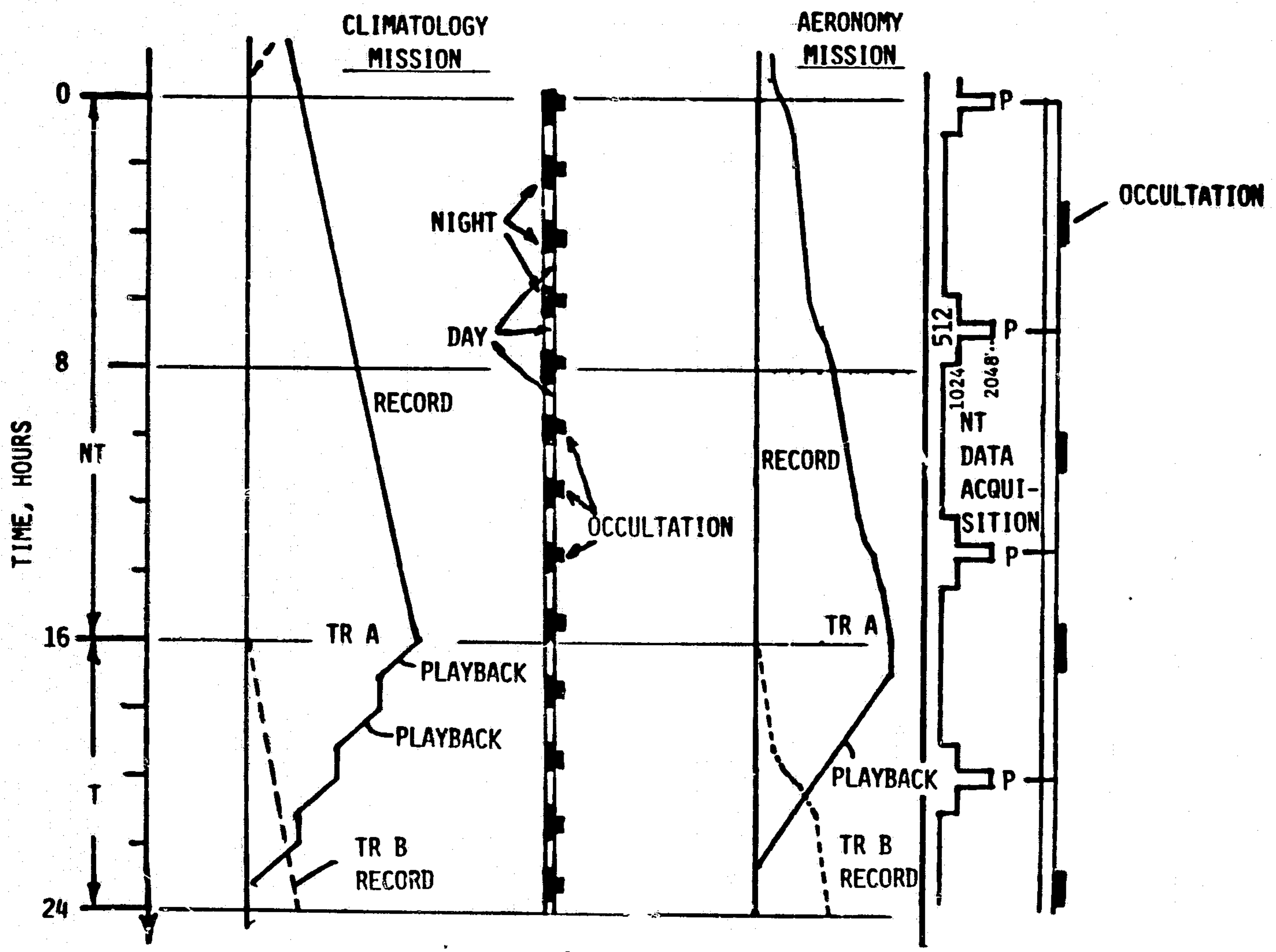


Figure 8.4-1

## AUTOMATIC FAULT RESPONSE

| | | | |
|---|---|---|---|
| ELECTRIC POWER | A. | IF EXCESSIVE CURRENTS ARE DRAWN | BLOW FUSES OR CIRCUIT BREAKERS |
| | B. | IF VOLTAGE DROPS TOO LOW | SHEDS LOADS |
| | C. | IF BATTERY CHARGE CURRENT OR TEMPERATURE IS EXCESSIVE | ADJUST CHARGE CURRENT |
| COMMANDS | | IF NO COMMANDS ARE RECEIVED IN A GIVEN PERIOD (~6 DAYS) | I) SWITCH RECEIVERS AND COMMAND PATH THROUGH CRITICAL COMMAND RECEIPT, WITH RESPECT TO THE 2 S-BAND ANTENNAS<br>II) SET HGA CONE ANGLE GIMBAL TO 90°. (ASYNCHRONOUS TO (I)) |
| ATTITUDE CONTROL | A. | IF ACCELEROMETER DETECTS NUTATION ABOVE A THRESHOLD | FIRE PRECESSION THRUSTERS AT APPROPRIATE TIME IN THE SPIN CYCLE. (ONLY WHEN ANC IS ENABLED. ORDINARILY ONLY WHEN ATTACHED TO INJECTION STAGE.) |
| | B. | IF DESPUN PLATFORM SPINS UP | SET HGA CONE ANGLE GIMBAL TO 90°. (ONLY WHEN THIS PROVISION IS ENABLED.) |

THERMAL CONTROL

(THERMOSTATS; MAY BE OVERRIDDEN ON; MAY BE OVERRIDDEN OFF)

Table 8.4-1

## 8.5 GROUND/LAUNCH

### 8.5.1 Ground/Launch Operations Requirements

The following launch operations requirements are addressed in subsequent sections.

- Identify any special arrangements (support, administrative, etc.) which are required to support launch operations at the ELS.
- Specify the activities and time duration and launch site activation and deactivation.
- Include a description of how the launch site operations will be managed, scheduled, and controlled.
- Describe the launch operations.
- Identify documentation to be provided.

### 8.5.2 Launch Activities

The flow diagram of Mars orbiter spacecraft (MO) launch activities is shown in Figure 8.5-1. The numbers appended to the flow boxes refer to descriptions of each task that are provided in Table 8.5-1 of the plan. The in-plant activities assure that the spacecraft is ready to launch when shipped so that the launch site flow is limited to those tasks necessary to verify that the spacecraft has survived the shipment, perform an additional end-to-end compatibility test with the ground segment and configure/service the spacecraft (ordnance installation, fueling, SRM mating, thermal closeouts). The remainder of the flow follows the prescribed STS sequence for payload vertical processing.

The principal facilities to be used at the launch site are the payload processing facility (PPF), hazardous processing facility (HPF), spin test facility (STF), vertical processing facility (VPF) and Launch Complex 39. A central PPF at the Cape Canaveral Air Force Station (CCAFS) will serve as the initial spacecraft receiving and checkout facility and the ground station for the EGSE test and checkout equipment used to support all prelaunch functional and operational interface testing. The HPF is used for fuel loading and pressurization. It is expected that the HPF will be either ESA 60A or a new facility. The MDAC Delta Spin Test Facility will be used for mating the MO spacecraft to the I-VI SRM perigee stage. This facility cannot presently support dynamic spin balance tests of the mated MO/SRM (weight limitation). Modifications to this facility or an alternate facility will be available in the required time frame, if a mated dynamic spin balance test is deemed necessary.

The cargo integration test equipment (CITE) in the VPF will be used for orbiter compatibility verification. These facilities satisfy the spacecraft requirements with the possible exception of minor changes for power, data lines, etc. Detailed requirements will be provided in the Launch Site Support Plan MO/STS PIP Annex 8).

The hardline EGSE (spacecraft unbilical console, solar array simulator, ACS Test Set, etc.) are set up and validated for the functional test of the spacecraft in the PPF. The spacecraft umbilical console is moved with the spacecraft to each facility since it is required to trickle charge batteries. The remaining EGSE (I&T computer, baseband equipment, instrument computers, etc.) are set up and validated in the test control center PPF and remain in place throughout the launch operations.

### 8.5.3 Special Launch Support Requirements

The period of launch site utilization is approximately 11 weeks, which includes receiving and checkout of support equipment and post-launch shipping preparations. The PPF that serves as the MO test control center is required for the entire period. The HPF and the STF are required for approximately 1 week each. Use period for the VPF and launch complex is planned to be nominal but is dependent upon other cargo elements, the orbiter timeline, etc.

No special or unusual support requirements are identified. Standard equipment and support requirements are typical of the following:

- Cargo lift trailer and tugs for transporting the spacecraft.
- Trucks, fork lifts to support receipt of C-5A shipment.
- Portable crane (10 Ton).
- C-5A landing support.
- Standard office supplies and reproduction services.
- Propellant cleanliness verification.
- Dry nitrogen.
- Coordination by KSC of inter-organizational activities.
- Medical facilities.
- Radioactive source control.

Detailed requirements will be provided in the Launch Site Support Plan MO/STS PIP Annex 8).

### 8.5.4 Management of Launch Site Operations

The assistant project manager for spacecraft integration and test is the TRW test director for launch site activities. He directs the overall spacecraft activities and is the TRW management interface with ARC, KSC, JSC and other agencies.

In addition to the launch bar chart schedule, daily schedules are prepared several days in advance and updated daily if required. These schedules identify each task to be performed, the responsible individual, and the required test procedure. Short morning meetings are held to assess the previous and upcoming days activities. Any problems, conflicts or significant delays encountered during the day's activities will be reported to management in real time so that appropriate action can be taken. Interface/coordination meetings with other agencies are fully supported. The launch team remains established, with key personnel available, until 24 hours after launch.

### 8.5.5 Documentation

All spacecraft launch activities are performed to detailed test procedures provided to ARC 30 days prior to start of the activity (90 days for propellant loading). Procedures containing hazardous activities (hoisting, fueling, ordnance installation and AKM mating and balance tests are reviewed and the activities witnessed by safety personnel. Requirements are provided for KSC preparation of the CITE operations support. Reports are provided to ARC within 60 days of activity completion.

In addition to the documentation of activities that are to be implemented, an abort recovery plan and abort recovery procedures will be prepared. Four abort procedures will be prepared for the following conditions: (1) abort prior to launch, (2) abort to primary landing site, (3) abort to secondary landing site, and (4) abort to contingency landing site. The first abort procedure is a "back-out" procedure which covers removal of the spacecraft from the orbiter and return to the VPF. The abort recovery plan will cover the last three abort cases following the orbiter lanuch. It will identify the support elements required to recover the spacecraft after a post-flight abort to any of the specified landing sites. These elements include TRW and NASA contractor personnel, spacecraft and GFE ground support equipment, and facilities required at the landing site and upon return to TRW or NASA KSC. The plan will include a coordinated task flow diagram and integrated schedules for each landing site. The plan will also specify the relationships and responsibilities of all agencies involved in planning and conducting post-flight abort activities.

C-4

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## KSC FLOW DIAGRAM FOR MARS ORBITER (MO)

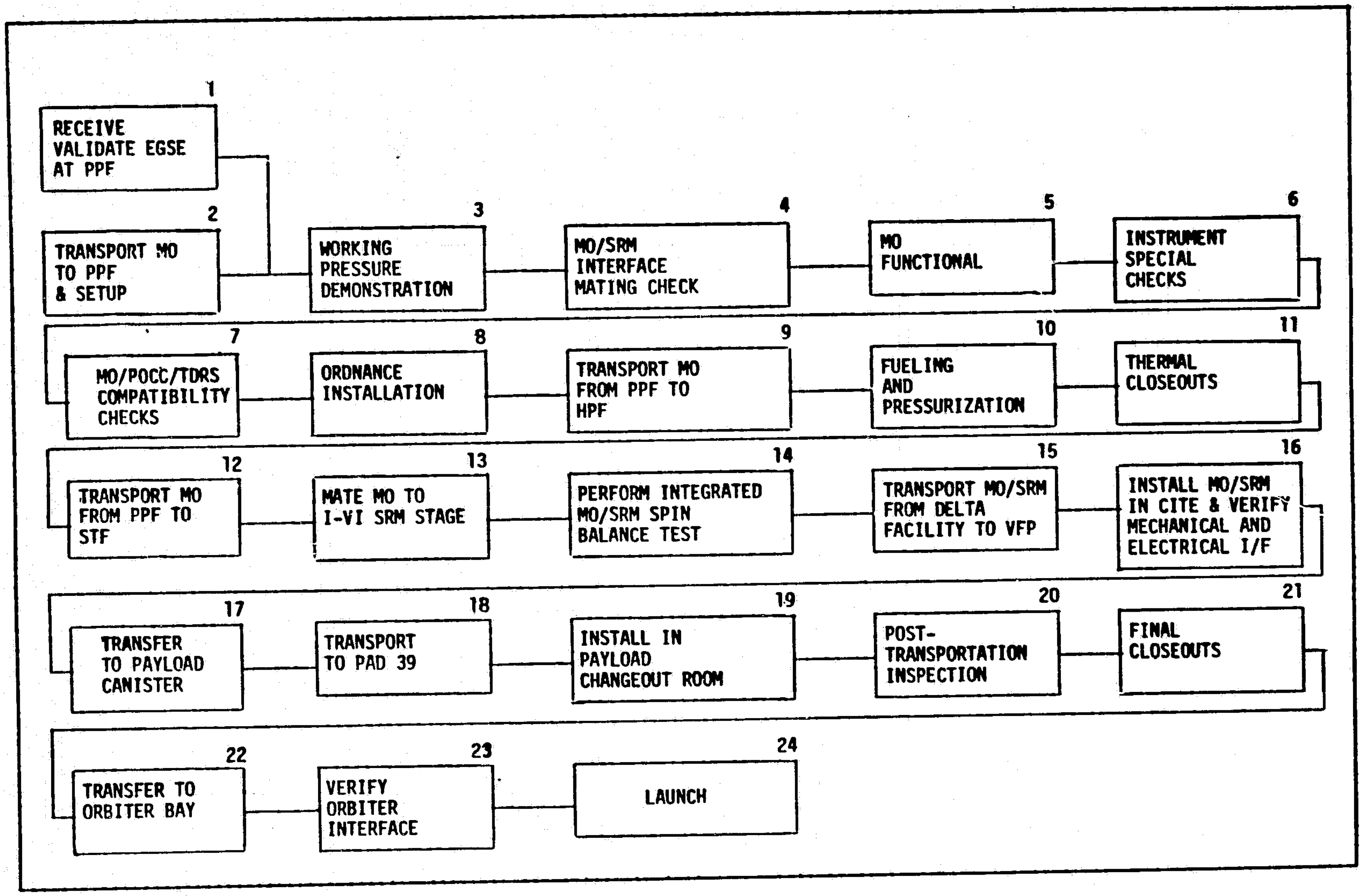


Figure 8.5-1

## SUMMARY DESCRIPTION OF MO PRELAUNCH/LAUNCH VERIFICATION OPERATIONS

| FLOW | TASK | PURPOSE | DESCRIPTION | SUPPORT EQUIPMENT |
|---|---|---|---|---|
| 1 | RECEIVE AND VALIDATE EGSE AT PPF | TO SET UP AND VALIDATE EGSE FOR SUPPORT OF SPACECRAFT TESTING | THE OSTC EQUIPMENT IS OFFLOADED FROM THE VANS, MOVED INTO THE PPF, UNPACKED, INSPECTED, AND VALIDATED. INSTRUMENT SUPPORT EQUIPMENT IS SET UP AND VALIDATED BY INSTRUMENT PERSONNEL. FACILITY CHECKOUT IS PERFORMED IN CONJUNCTION WITH THE ABOVE TO ASSURE AVAILABLE POWER FACILITY DATA HARDLINES, ETC. | VALIDATION UNITS, METERS, POWER SUPPLIES, OSCILLOSCOPES, ETC. |
| 2 | TRANSPORT MO TO PPF AND SET UP | TO LOCATE AND PREPARE MO FOR TEST | THE SPACECRAFT IS TRANSFERRED FROM THE C5A ONTO THE NASA CARGO LIFT TRAILER (CLT) AND TOWED TO THE PPF WHERE IT IS OFFLOADED. THE SHIPPING CONTAINER IS REMOVED AND THE SPACECRAFT INSPECTED. OTHER FLIGHT HARDWARE AND GSE ARE TRANSFERRED BY KSC TRUCKS FROM THE C5A TO THE PPF. | HORIZONTAL HANDLING AND TRANSPORTATION FIXTURE, SPACECRAFT SHIPPING CONTAINER, PORTABLE TEMPERATURE CONTROL UNIT, DYNAMIC INSTRUMENTATION SYSTEM. |
| 3 | WORKING PRESSURE DEMONSTRATION | TO SATISFY KSC SAFETY REQUIREMENT FOR FIRST PRESSURIZATION AT KSC | A PORTABLE BLAST SHIELD IS SET UP AND THE SPACECRAFT PROPULSION SUBSYSTEM IS PRESSURIZED REMOTELY TO WORKING PRESSURE WITH $GN_2$. THE PRESSURE IS THEN REDUCED TO 50 PSI FOR FUNCTIONAL TESTING. | PRESSURE AND PROPELLANT LOADING UNIT, PROPULSION CONTROL AND MONITOR UNIT, PORTABLE BLAST SHIELD. |
| 4 | SPACECRAFT FUNCTIONAL | TO VERIFY THAT SPACECRAFT SUBSYSTEMS AND INSTRUMENTS OPERATING IN THE SYSTEM FUNCTION CORRECTLY AND HAVE SUFFERED NO SHIPMENT DEGRADATION | THIS OPERATION WILL EXERCISE THE SPACECRAFT THROUGH SIMULATED LAUNCH, INJECTION, AND ORBIT MODES FOLLOWED BY DETAILED SUBSYSTEM PERFORMANCE MEASUREMENTS. SPACECRAFT AND INSTRUMENT MODE OPERATIONS ARE MONITORED DURING THE TEST EXERCISE TO DETECT ANY ANOMALOUS CHARACTERISTICS BASED ON PRIOR BASELINE DATA. | EGSE, IGSE |
| 5 | MO/SRM INTERFACE MATING CHECK | TO VERIFY SATISFACTORY MECHANICAL INTERFACE BETWEEN MO/SRM ADAPTER INTERFACE | THE SPACECRAFT INTERFACE ADAPTER RING IS REMOVED FROM THE I-VI SRM PERIGEE STAGE AND TRANSPORTED TO THE MO PPF. USING FEELER GAUGES AND SPECIAL TOOLING THE INTERFACE BETWEEN THE MO AND THE SRM ADAPTER RING IS VERIFIED USING THE FLIGHT ADAPTER CLAMPING HARDWARE. | INTERFACE ADAPTER TOOLS GAUGES ETC. |

Table 8.5-1

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SUMMARY DESCRIPTION OF MO PRELAUNCH/LAUNCH VERIFICATION OPERATIONS (CONTINUED)

| FLOW | TASK | PURPOSE | DESCRIPTION | SUPPORT EQUIPMENT |
|---|---|---|---|---|
| 6 | INSTRUMENT SPECIAL CHECKS | TO PERMIT THE PRINCIPAL INVESIGATORS TO VERIFY INSTRUMENT FUNCTIONAL CHARACTERISTICS | THE SPACECRAFT IS POWERED ON AND COMMAND AS REQUIRED BY INSTRUMENT PERSONNEL (IN ACCORDANCE WITH PROCEDURE CONSTRAINTS). SPACECRAFT HARDLINE TELEMETRY DATA IS MONITORED TO DETECT ANOMALOUS CONDITIONS AND TO PROVIDE CONFIDENCE THAT NO DEGRADATION OCCURRED DUE TO SHIPMENT. | EGSE, IGSE |
| 7 | MO/POCC/TDRSS COMPATIBILITY CHECKS | TO VERIFY THE COMMAND AND TELEMETRY WITH TDRSS AND THE GROUND STATIONS | DURING THIS TEST THE SPACECRAFT IS EXERCISED THROUGH A MISSION SIMULATION THAT EXERCISES ALL THE COMMANDS NECESSARY FOR THE MISSION THROUGH LANUCH/ASCENT, DEPLOYMENT, AND ACTIVATION ON-ORBIT. THIS TEST VERIFIES THE COMMAND AND TELEMETRY LINK WITH THE POCC VIA TDRSS. THE TEST IS DIRECTED BY THE POCC AND IS MONITORED BY THE EGSE. COMMANDS AND TELEMETRY ARE TRANSMITTED VIA RF LINK TO MILA AND FROM THERE TO TDRSS AND THE POCC. | RECEIVE AND REPEATER ANTENNAS, EGSE, IGSE |
| 8 | ORDNANCE INSTALLATION | TO INSTALL ORDNANCE | THE ORDNANCE IS TRANSPORTED TO THE PPF AND INSTALLED IN THE SPACECRAFT BUT NOT CONNECTED TO THE FIRING CIRCUITS. FINAL ELECTRICAL CONNECTION WILL TAKE PLACE IN THE ORBITER AFTER NO-VOLTAGE AND BRIDGE WIRE CHECKS ARE MADE. | SPACECRAFT WORK STANDS, ORDNANCE SUITCASE |
| 9 | TRANSPORT MO FROM PPF TO HPF | TO MOVE TO A FACILITY ACCEPTABLE FOR HYDRAZINE FUELING | THE HORIZONTAL PROTECTIVE COVER IS INSTALLED AND THE PORTABLE TEMPERATURE CONTROL UNIT IS ACTIVATED. THE SPACECRAFT IS HOISTED ONTO THE KSC CLT AND TOWED TO THE HPF. IT IS PRESENTLY PLANNED THAT THE SPACECRAFT REMAIN ON THE CLT DURING FUELING OPERATIONS. THE SPACECRAFT UMBILICAL CONSOLE IS ALSO MOVED TO THE HPF TO PERMIT BATTERY TRICKLE CHARGING DURING PERIODS WHEN FUEL IS NOT BEING LOADED. | HORIZONTAL PROTECTIVE COVER, PORTABLE TEMPERATURE CONTROL UNIT, CLT, UMBILICAL CONSOLE |

Table 8.5-1

SUMMARY DESCRIPTION OF PRELAUNCH/LAUNCH VERIFICATION OPERATIONS (CONTINUED)

| FLOW | TASK | PURPOSE | DESCRIPTION | SUPPORT EQUIPMENT |
|---|---|---|---|---|
| 10 | FUELING AND PRESSURIZATION | TO FUEL THE PROPULSION SUBSYSTEM AND TO PRESSURIZE IT. | THE PRESSURE AND PROPELLANT LOADING UNIT IS TRANSPORTED FROM THE FUEL FARM TO THE HPF. EACH OF THE TANKS IS LOADED WITH THE SPECIFIED WEIGHT OF HYDRAZINE FUEL AND PRESSURIZED WITH DRY NITROGEN. ONCE PRESSURE STABILITY IS ACHIEVED THE FILL/DRAIN VALVE IS CLOSED AND CHECKED FOR LEAKAGE. THE PPLU IS DISCONNECTED AND TRANSPORTED BACK TO THE FUEL FARM FOR CLEANING. EACH THRUSTER IS CHECKED FOR LEAKAGE WITH THE HYDRAZINE VAPOR DETECTOR. | PPLU, HYDRAZINE VAPOR DETECTOR, TEST STANDS, EGRET PRESSURIZATION UNIT |
| 11 | THERMAL CLOSE-OUTS | TO CONFIGURE THERMAL CLOSEOUTS AT PROPULSION SUBYSTEM. | THE THERMAL INSULATION IS INSTALLED IN THOSE AREAS WHICH REQUIRED ACCESS FOR FUELING AND PRESSURIZAION. | |
| 12 | TRANSPORT MO FROM PPF TO DELTA SPIN | TO RELOCATE MO TO STF FOR SRM MATING | THE HORIZONTAL PROTECTIVE CAVE IS INSTALLED, THE PORTABLE CONTROL UNIT IS ACTIVATED AND THE SPACECRAFT IS TOWED TO THE STF. | HORIZONTAL IN-PLANT TRANSPORTER, HORIZONTAL PROTECTIVE COVER, PORTABLE TEMPERATURE CONTROL UNIT. |
| 13 | MATE MO TO I-VI SRM STAGE | SPACECRAFT MATING TO I-VI SRM PERIGEE STAGE | DELTA TEST FACILITY PERSONNEL WORKING WITH TRW SPACECRAFT CONTRACTOR PERSONNEL PERFORM S/C SRM MATING OPERATIONS. | MDAC DELTA SPIN TEST FACILITY |
| 14 | PERFORM INTEGRATED MO/SRM SPIN BALANCE TEST | TO SPIN BALANCE SPACECRAFT AND THIRD STAGE | DELTA TEST FACILITY PERSONNEL PERFORM STATIC/DYNAMIC SPIN BALANCE TEST MATED MO/SRM PERIGEE STAGE. | NOTE: PRESENT STF CANNOT SUPPORT THE MO/SRM WEIGHT FOR DYNAMIC SPIN BALANCE TESTS. IF THIS TEST IS REQUIRED FACILITY MODIFICATIONS WILL BE REQUIRED. |
| 15 | TRANSPORT MO FROM STF TO VPF | TO PERMIT SITE TESTING | THE HORIZONTAL PROTECTIVE COVER IS INSTALLED, THE PROTABLE TEMPERATURE CONTROL UNIT IS ACTIVATED, AND THE SPACECRAFT IS TOWED ON THE CLT TO THE VPF. FOLLOWING CLEANUP AND PREPARATIONS' IN THE AIR LOCK THE CLT IS MOVED INTO THE HIGH BAY. | HORIZONTAL IN-PLANT TRANSPORTER, HORIZONTAL PROTECTIVE COVER, PORTABLE TEMPERATURE CONTROL UNIT, CLT. |

Table 8.5-1

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SUMMARY DESCRIPTION OF PRELAUNCH/LAUNCH VERIFICATION OPERATIONS (CONTINUED)

| FLOW | TASK | PURPOSE | DESCRIPTION | SUPPORT EQUIPMENT |
|---|---|---|---|---|
| 16 | INSTALL IN CITE AND VERIFY SPACECRAFT INTERFACES | TO VERIFY MECHANICAL AND ELECTRICAL INTERFACES WITH ORBITER. TO ESTABLISH TOTAL CARGO CONFIGURATION | THE SPACECRAFT/SRM IS HOISTED TO VERTICAL USING THE FACILITY CRANE AND POSITIONED IN THE TEST STAND. INSPECTION OF THE HGA COAXIAL CONNECTION IS PERFORMED. CITE CONNECTIONS ARE MATED AND THE INTERFACE FUNCTIONS ARE VERIFIED. TRW WILL PROVIDE REQUIREMENTS FOR KSE PREPARATION OF THE CITE PERFORMANCE SUPPORT. | VERTICAL HOIST BEAM/SLING, EGSE |
| 17 | TRANSFER TO PAYLOAD CANISTER | TO PREPARE FOR TRANSPORT TO PAD 39 | THE MO/SRM IS INSTALLED INTO THE PAYLOAD CANISTER AND THE INSTRUMENTATION AND ENVIRONMENTAL CONTROL SYSTEM CONNECTED. THIS IS A KSC TASK SUPPORTED AS NECESSARY BY TRW. | CANISTER AND TRANSPORTER |
| 18 | TRANSPORT TO PAD 39 | TO PREPARE FOR INSTALLATION IN THE ORBITER BAY | THE CANISTER IS TRANSPORTED TO THE PAD IN THE VERTICAL POSITION. | PAYLOAD CANISTER, TRANSPORTER |
| 19 | INSTALL IN PAYLOAD CHANGEOUT ROOM | TO PREPARE FOR INSTALLATION IN THE ORBITER BAY | THE CANISTER IS HOISTED INTO POSITION AT THE RSS DOORS BY THE 90-TON CRANE. THE INTERFACE BETWEEN THE RSS AND THE CANISTER IS SEALED AND THE RSS AND CANISTER DOORS ARE OPENED. THE PAYLOAD GROUND HANDLING MECHANISM (PGHM) IS TRANSLATED INTO POSITION, REMOVES THE OBSERVATORY FROM THE CANISTER, AND TRANSLATES IT BACK INTO THE RSS. THE CANISTER AND RSS DOORS ARE CLOSED AND THE CANISTER LOWERED TO ITS TRANSPORTER. THE SPACECRAFT UMBILICAL CONSOLE IS MOVED TO THE RSS AND CONNECTED TO MAINTAIN BATTERY TRICKLE CHARGE. | RSS HOIST EQUIPMENT, RSS EQUIPMENT |
| 20 | POST-TRANSPORTATION INSPECTION | TO VERIFY MO HAS NOT BEEN DAMAGED BY TRANSPORTING | A DETAILED VISUAL INSPECTION OF THE SPACECRAFT IS CONDUCTED. | |

Table 8.5-1

## SUMMARY DESCRIPTION OF PRELAUNCH/LAUNCH VERIFICATION OPERATIONS (CONTINUED)

| FLOW | TASK | PURPOSE | DESCRIPTION | SUPPORT EQUIPMENT |
|---|---|---|---|---|
| 21 | FINAL CLOSEOUTS | TO PREPARE MO FOR INSTALLATION INTO THE ORBITER | PROTECTIVE COVERS ARE REMOVED FROM AREAS THAT WILL BE INACCESSIBLE AFTER INSTALLATION IN THE ORBITER, AND INSULATION IS INSPECTED/CLOSED OUT. | |
| 22 | TRANSFER TO ORBITER BAY | TO PREPARE FOR LAUNCH | THE RSS IS ROTATED TO THE ORBITER, THE RSS AND ORBITER BAY DOORS ARE OPENED, AND THE SPACECRAFT/SRM IS INSTALLED IN THE ORBITER BY THE PGHM. | |
| 23 | VERIFY ORBITER INTERFACE | TO VERIFY ALL ELECTRICAL INTERFACE FUNCTIONS PRIOR TO LAUNCH | THE ELECTRICAL INTERFACES WITH THE ORBITER ARE MATED AND ALL INTERFACE FUNCTIONS (AFT FLIGHT DECK AND T-0 UMBILICAL) ARE CHECKED. THE SPACECRAFT IS THEN CONFIGURED FOR LAUNCH. FINAL ORDNANCE CONNECTIONS ARE MADE AND THE REMAINING PROTECTIVE COVERS ARE REMOVED. | |
| 24 | LAUNCH | TO PLACE MO IN ORBIT | AFTER FINAL ORBITER TASKS ARE COMPLETED THE LAUNCH TAKES PLACE. WHEN THE BAY DOORS ARE OPENED IN ORBIT THE MO TEMPERATURES ARE MONITORED/CONTROLLED VIA THE AFT FLIGHT DECK PAYLOAD PANEL. THE BATTERIES ARE CONNECTED TO THE BUS AND THE TRANSMITTER IS COMMANDED ON FOR GROUND COMMUNICATIONS VIA TDRSS. THE RMS IS MATED, THE UMBILICALS ARE EXTRACTED, AND THE MO IS PLACED IN FREE ORBIT AFTER WHICH SRM FIRING IS ACCOMPLISHED. THE LAUNCH SITE OPERATIONS TEAM WILL REMAIN INTACT, WITH KEY PERSONNEL AVAILABLE UNTIL 24 HOURS AFTER SRM FIRING. FOLLOWING ORBITER LANDING, SUPPORT WILL BE PROVIDED AS REQUIRED FOR GSE REMOVAL. THE GSE AND OTHER EQUIPMENT WILL BE PREPARED AND SHIPPED TO TRW AND INSTRUMENTER FACILITIES. | |



Table 8.5-1

# 9. CLIMATOLOGY ORBIT OPTION

## 9.1 SALIENT FEATURES

### 9.1.1 Objective

The baseline Climatology orbit is approximately sun-synchronous after the drift period. Thus the spacecraft's northbound equatorial crossing is always at the same local time of day (night), and the southbound at the same local time of night (day). This is good for tracking seasonal variations, where you might want to suppress diurnal variations.

The objective of the orbit option is to emphasize daily variations by having an orbit which is deliberately asynchronous to the sun, and in which the local time of equator crossing progresses through the entire 24 "hours".

### 9.1.2 Orbit Description

The baseline orbit (300 km altitude, circular) has an inclination of 92.65°. This gives a nodal advance rate of 1.0 revolutions per Martian year, the same as the average angular rate of the sun as seen from Mars.

The optional orbit is also 300 km in altitude and circular, but it is inclined at 84.69°. Instead of advancing, the node line now regresses at a rate of 2.0 rev/Martian year. Relative to the average sun, this is a regression (westward progression) of 3.0 rev/Martian year.

### 9.1.3 Properties

Some of the geometrical properties of the optional orbit are noted. The sun aspect angle (SAA) experiences a greater range. For a spacecraft whose +Z axis always is directed toward the side of the orbit plane the sun is in, the baseline orbit range is:

$$45° < SAA < 68° \text{ (after the drift period)}$$

For the optional orbit:

$$\sim 0° < SAA < 90°$$

The earth aspect angle (EAA) similarly has a range change. With the +Z axis still to the sunny side of the orbit, we have

baseline: $0° < EAA < 105°$, approximately
option $0° < EAA < 135°$, approximately

In this regard, the Climatology orbit option has characteristics closer to the Aeronomy orbit than to the Climatology baseline.

Eclipse durations for the baseline and optional orbits were presented and discussed in Section 3. As expected, there is much more variation in eclipse durations through the orbit option mission. Occultation patterrs may also vary more.

In addition, there is no preferred "initial" orientation of the orbit plane relative to the sunline, so there is no drift period at the beginning of the orbital phase.

### 9.1.4 Influences

The increased range of SAA's experienced means the array must operate at a lower areal efficiency than in the baseline. This will cause (when SAA = 20°, and when SAA = 90°) about a 10 percent reduction in the power produced at those times (there will be 18 such times in a Martian year). See the discussion in Section 6.1.

It will be necessary for the spacecraft to execute six 180° flip maneuvers in the course of the Martian year to keep the +Z axis on the right side of the orbit plane. In addition, normal precession maneuvers must be performed twice as frequently just to keep up with the regressing orbit plane.

The greater range of SAA's will also make the GRS radiator design more difficult, as the sun can shine from a greater range of directions.

The expansion of the range of EAA values makes it desirable to have a greater range of HGA elevation angles at ones command. This can be mitigated somewhat if EAA's are favored by letting the SAA go beyond 90°. This situation is similar to the Aeronomy Mission. But the frequency of such operations is now greater because of the greater orbit regression rate. Near Mars' aphelion, we may not be able to tolerate the further power reduction at SAA's beyond 90°.

A final influence is the observation that if there is no drift period at the start of orbital operations, then there does not have to be an inclination change maneuver. The $\Delta V$ which was dedicated to this maneuver in the baseline ($\sim$150 m/s) is no longer needed.

## 9.2 CHANGES TO SPACECRAFT

### 9.2.1 Solar Array

To compensate for the 10% reduction in power per unit area, the solar array area should be increased correspondingly. If this is done by extending the base of the truncated cone, the cone height of 2.08 meters will grow to 2.25 meters (7 inches higher), and the 4.32 meter (170 inch) maximum diameter will grow to 4.44 meters (175 inches). This will use up essentially all the available margin in diameter associated with the baseline design.

To accommodate the greater range of EAA's, there are two approaches. One is to increase further the solar array size to permit operation when the SAA > 90°, to favor lower EAA's. The other is to mount the HGA higher or otherwise alter its mounting to permit it to peer farther over the edge of the despun platform. We have not analyzed this situation to optimize the solution.

The moment-of-inertia ratios will have to be examined, also. Increasing the solar array height and raising the HGA can only aggravate it. Countermeasures will have to be examined, or the method of damping on the despun platform must be relied on.

The good news is that we do not have to carry as much propellant, because the inclination change maneuver has been deleted. The bad news is that, with this lesser propellant on board at pre-MOI, the moment-of-inertia ratios are further aggravated. (At EOL it doesn't matter; in either case we must be able to operate with the tanks empty and the platform despun.)

## 9.3 SCIENCE ACCOMMODATION

The principal effect on science accommodation is in thermal control. With the greater range of SAA's, some experiment thermal radiators will require a more careful design to exclude the sun from the field-of-view. It may be tough enough to solve this problem for the baseline; the optional orbit may compound it.

Other major science accommodation interfaces are unchanged. The measures of Section 9.2 meet the power requirements; the command, data, and RF link requirements and capabilities are unchanged; and the propulsive requirements are reduced. Structural (configurational) changes are necessary only for subsystem changes -- solar array and HGA -- and do not affect science performance.

# 10. SCIENCE INSTRUMENT COMPLEMENT OPTIONS

## 10.1 SALIENT FEATURES

The basic science requirements for the various Climatology payload options have been stated in Section 4. These are shown again as summarized in Figure 10.1-1. Figure 10.1-2 gives an overview of how the baseline spacecraft design meets these requirements. Areas of concern include power for all options, although option C1 is the most difficult, data transmission rate for C3, and thermal radiator accommodation for C3.

## 10.2 CHANGES TO SPACECRAFT AND MISSION PARAMETERS

The mass increase of up to 20 kg presents no problem for the baseline spacecraft. Ample margin has been included in the mass allocation as noted in Section 6.3.1. Should balance or moment-of-inertia ratios become a problem ballast added near the periphery of the rotating section on the c.g. plane could be added. With only minor structural changes a Star 37XE could be used in place of the baseline 37XF providing up to 50% increased total impulse.

Power requirements place a greater strain on the baseline design, however. The requirement for an additional 18.5 W in option C1 would consume almost all the margin in this resource and not permit the dual tape recorder operations mode. Careful study of solar aspect, eclipse periods, and Mars-Sun distance as a function of mission day might point the way to increased power availability using some duty cycling of instruments. However, the more straightforward approach is to increase the area of the array, extending the skirt downward in the -Z direction. The effect on mass and moment-of-inertia ratios could, again, be met with the large weight margins available in the baseline design.

The data handling and communication subsystems have been sized to accommodate the requirements of options C1 and C2 with no modification except in telemetry format. However, the higher data rate requirement of option C3 cannot be met with the baseline design, particularly at the largest Earth-Mars distances. The data requirement can be met in three ways. First, by increasing the HGA diameter to 1.5 m adequate margin is

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## CLIMATOLOGY PAYLOAD OPTIONS

## INSTRUMENT REQUIREMENTS

| OPTION | MASS (KG) | POWER (W) | | DATA | |
|---|---|---|---|---|---|
| | | DAY | NIGHT | DAY | NIGHT |
| BASELINE | 37 | 59 | 56 | 1284 | 1164 |
| C1 | 50 | 77.5 | 74 | 1456 | 1272 |
| C2 | 57 | 71 | 61 | 1540 | 1164 |
| C3 | 53 | 67 | 55 | 3264 (20K)* | 1264 |

THERMAL: ALL CATEGORIES OF INSTRUMENT TEMPERATURE REQUIREMENTS ARE IDENTICAL FOR EACH PAYLOAD OPTION

LOOK ANGLES: OPTION C1 REQUIRES VIEWING FORWARD LIMB
OPTION C2 REQUIRES LIMB VIEWING AT $45^o$ AND $135^o$ FROM RAM DIRECTION
OPTION C3 REQUIRES THERMAL RADIATOR NOT VIEWING SUN, PLANET OR S/C

POINTING STABILITY: OPTION C1 REQUIRES LIMB PULSE WITH $0.08^o$ ACCURACY
OPTION C2 REQUIRES 3-AXIS STABILITY OF $0.5^o$, KNOWLEDGE OF $0.1^o$
OPTION C3 REQUIRES 3-AXIS STABILITY OF $0.08^o$

* POSSIBLE MAXIMUM FOR SPECIAL GRS + MSM OPERATING MODE, BUT AVERAGE 24-HOUR RATE NOT INCREASED.

Figure 10.1-1

# CLIMATOLOGY PAYLOAD OPTIONS

## MEETING THE REQUIREMENTS

| | |
|---|---|
| MASS: | BASELINE CAPABILITY SIZED BY STAR 37XF.<br>MAXIMUM MASS INCREASE ALLOWABLE: 37.6 KG |
| POWER: | BASELINE CAPABILITY INCLUDES 24W MARGIN DURING SUN OPERATION |
| DATA: | BASELINE CAPABILITY INCLUDES 0.8 DB MARGIN IN WORST CASE |
| THERMAL: | INSTRUMENT REQUIREMENTS MET BY BASELINE DESIGN |
| POINTING STABILITY: | BASELINE ACHIEVES BETTER THAN $0.08^{\circ}$ ALONG ALL AXES |

Figure 10.1-2

assured for all Earth-Mars distances. Such an increase leads to further mass increases, greater extension of the antenna mast, etc. The second option is to reduce the data rate for those portions of the mission when the Earth-Mars distance is greater than 2.3 AU, since the baseline design accommodates the higher data rates out to this distance. (The possibility of doubling the transmitter's power would lead to major design changes in the whole spacecraft.)

Meeting the instrument FOV, pointing, and pointing stability requirements is within the baseline design with no changes although a horizon sensor would be desirable for option C1 in which both the $UVO_3$ and UVHP require pointing at the forward limb.

The thermal requirements for all options are identical with the baseline mission and can be met with no design changes. However, on option C3 the MSM requires a radiator with a clear FOV that does not view the planet, Sun, or spacecraft. The size of the radiator FOV is, as yet, TBD so that the difficulty in meeting its viewing requirement cannot be fully addressed. However, since the requirement is similar to that of the GRS boom mounting of this instrument may be necessary. In this case the pointing stability of 0.08° could present a serious design problem.

# 11. SUPPORTING RESEARCH AND TECHNOLOGY

There is no aspect of the spacecraft design -- system or subsystem -- which is beyond the current state of technology. For this reason, no hardware research and development program is necessary before program approval is sought.

There are, however, several areas of analysis where effort can be devoted to advantage, to advance primarily topics which are generally in the domain of system engineering.

These topics are outlined in Figures 11-1 to 11-3.

An additional future study area is the methodology of insuring the execution of events around injection which are time critical to the spacecraft and to the mission, without violating Shuttle safety provisions. We have outlined such a method in this report, but it needs to be fleshed out and tested against detailed Shuttle requirements.

## IDENTIFICATION OF FUTURE STUDY AREAS

1. ADAPTABILITY OF THE SPACECRAFT DESIGN TO THE 1990, 1992 MARS LAUNCH OPPORTUNITIES
   - PROPULSIVE MASS CHANGES
   - COMMUNICATIONS AT MOI
2. MORE DETAILED DESCRIPTION OF THE ATTITUDE CONTROL ELECTRONICS REQUIREMENTS AND ARCHITECTURE
   - THRUSTER FIRING CONTROL
   - PROCESSING OF SIGNALS FROM SENSORS
   - CONTROLLING THE DESPUN PLATFORM
   - CONTROLLING OF HGA POINTING

Figure 11-1

## IDENTIFICATION OF FUTURE STUDY AREAS

(CONTINUED)

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3. AUTOMATIC NUTATION CONTROL
   - PROPELLANT DAMPING EFFECTS
   - NUTATION SENSOR (ACCELEROMETER ?)
   - THRESHOLD
   - MAXIMUM AMPLITUDE; THRUSTER AUTHORITY
   - ALGORITHM FOR THRUSTER CONTROL

4. INJECTION ACCURACY OF THE UPPER STAGE
   - CAUSES: TIP-OFF UPON ROLLOUT<br>DISTURBING MOMENTS<br>SPIN-UP<br>FIRING<br>IMPULSE MAGNITUDE ERRORS
   - EFFECTS: SPACECRAFT $\Delta V$ TO CORRECT

Figure 11-2

## IDENTIFICATION OF FUTURE STUDY AREAS

(CONTINUED)

5. THE EFFECTS OF DYNAMIC IMBALANCE OF A HEAVY, BOOM-DEPLOYED UNIT (THE GRS):
   - PARTICULARLY DURING PROPULSIVE MANEUVERS
   - DURING OTHER EXTERNAL OR INTERNAL DISTURBANCES

6. ORBIT DETERMINATION ACCURACY
   - APPROACH TRAJECTORY TO MARS; AFFECTS:
     - NECESSARY ALTITUDE BIASING
     - $\Delta$V TO COMPENSATE FOR DISPERSIONS OF INITIAL ORBIT
   - IN ORBIT ABOUT MARS; AFFECTS:
     - ACCURACY OF OPEN-LOOP NADIR POINTING
     - RECONSIDERATION OF HORIZON SENSORS?



Figure 11-3

# 12. DESIGN DRIVERS

Sensitive design points are discussed in the Introduction, under Section 1.4, Major Study Results.

# APPENDIX A

# DISCUSSION OF LIMITING COMMUNICATIONS LINKS

MOS-82-Z201-008
Dixon
19 July 1982
Page 1



# Mars Orbiter Spacecraft

## Discussion of Limiting Communications Links

## 1. Limiting spacecraft antennas to despun section

### 1.1 Exposure

a. From injection until DSN visibility (1 to 8 hours) or until reorientation, whichever occurs later

b. For TCM's, if it is desired to use EAA $\leq 100^{\circ}$ to keep SAA $100^{\circ}$ for power or thermal purposes (unlikely)

c. In emergencies which radically misorient the spacecraft, e.g., executing a wrong precession command

d. At certain times in the $C_o$ or A missions, where EAA exceeds $105^{\circ}$ or even $120^{\circ}$. (But here, the HGA is prime.)

### 1.2 Evaluation

a. is the worst exposure. It takes faith in the ability of the spacecraft to orient itself based on the sequencer and stored commands

### 1.3 Saving

Assume we would have implemented $S\uparrow$ and $S\downarrow$ by an aft omni. We save an antenna, a diplexer, a transponder, a power amplifier(?), and having TLM modulation going to two exciters, by omitting the spinning antenna.

## 2. Using X-band for the primary downlink

### 2.1 Advantage

Assuming equal transmitted power and equal spacecraft antenna size (which reflects two important system constraints) the increase in data rate, using X-band, is $10.5 \pm 0.3$ dB, or x 11.2. Added pointing and weather losses may drop this to 9.2 dB, or x 8.3. This is a tremendous advantage. To put it differently, if we can just meet the data rate requirements at X-band with 20W and a 1.1 m antenna, then to meet it at S-band would take

| | |
|---|---|
| 20 W | 3.17 m |
| 40 W | 2.24 m |
| 89 W | 1.5 m |
| 166 W | 1.1 m |

Any of these combinations would appear to be prohibitive in the context of the present system design.

2.2 Phase Noise

When operating coherent, there is more phase noise for S-X turnaround than for S-S turnaround. However, this penalty may be very small at MOS conditions, and tolerable even if significant.

2.3 Ground Antenna Availability

Every DSN ground antenna of 34- or 64-m diameter contemplated for 1988-1996 receives X-band. Not every one receives S-band (or transmits S-band, either). Thus X-band downlink permits greater flexibility in scheduling ground stations. Of course, to get S-band commands in, some time each week, or maybe every day or every other day, must be scheduled on antennas accommodating S-band. The same is true for coherent tracking.

3. Deleting the S-band downlink capability

3.1 Ground Antenna Availability

As just noted, ground antenna availability is not in any way degraded if S downlink is deleted.

3.2 Spacecraft Antennas

For downlink, the HGA (X-band) is superior to the HGA (S-band) in data rate capability, but it has to be pointed more accurately toward the earth. This has been accepted in choosing this mode for the primary downlink transmission of scientific data.

For instances where the HGA cannot be used, and this includes the entire mission through orbit insertion at Mars after which the platform is despun, downlink communications depend on using antennas which have axially symmetric gain. Because of the varied spacecraft attitudes during these phases of the mission, some of which are statistically indeterminate, the solution we strive for is one which gives adequate communications capability over one hemisphere of coverage. Whether this is done via one spacecraft antenna or more, the comparison between S- and X-band is the same:

Spacecraft antennas have the same gain

Ground antennas have the same area

To a first approximation, data capability for a given transmitted power is independent of frequency, under these circumstances, and S- and X-band have comparable capability. However, because X-band is primary for downlink of scientific data via the HGA, we have chosen 20 W as the transmitted power for X-band, and have indicated 5 W for S-band. The advantage for X-band is about a factor of 4. Accounting for a slight lowering of ground station antenna efficiency and increased weather attenuation at X-band, the advantage is reduced to, perhaps, a factor of 3. Therefore, as in the case of the HGA, we have the potential for a superior communication path by using X-band provided antenna(s) can be selected to provide hemispherical coverage.

We have picked two X-band medium gain antennas, attached to the HGA mount. For the use of these antennas, the HGA must have its cone angle gimbal set to 90$^o$, making their patterns axisymmetric. One antenna is a bi-cone, with a fan shaped pattern emphasizing coverage at EAA's (earth aspect angles) from 50$^o$ to 100$^o$. The other is a horn with a broad pattern along the spin axis, emphasizing coverage at EAA's from 0$^o$ to 50$^o$.

We have not used a single X-band omni antenna in place of a single S-band omni, because (1) no such antenna has been developed; (2) there may be arcing between more closely spaced conductors at X-band, particularly at 20 W output. Furthermore, we can cover the desired hemisphere with greater gain (statistically) using two MGA's.

The critical period for downlink telemetry via the MGA's is when oriented for Mars orbit insertion. We indicate an EAA of 38 degrees, then, and assuming an antenna gain via the horn of + 3dB in this direction, the link can carry 16 bps. We consider 16 bps to be the requirement, although the mission could undoubtedly be performed if this were lowered to 8 bps.

### 3.3 Saving

By deleting S-band capability for downlink transmission, the following elements are deleted from the equipment list:

In the transponders:

| | |
|---|---|
| S-band power amplifier | (2) |
| S-band switch | (1) |
| Diplexer | (2) |

However we have added one X-band switch and one X-band MGA to get satisfactory coverage, particularly at MOI, which the S-band downlink system does not provide.

## 4. Deleting Communications to TDRSS and to STS

4.1 Statement of Work (TDRSS). The SOW has been changed to delete all requirements for communications between the spacecraft and the Tracking and Data Relay Satellite System (TDRSS). This recognizes that:

(a) The DSN transponder we are using may not be compatible with TDRSS communications; additional communications equipment may be necessary.

(b) A TDRSS communications requirement might constrain the launch opportunity in terms of launch time and injection modes (northbound or southbound)

(c) It might also call for a dedicated spacecraft antenna, and possibly an S-band power amplifier.

(d) The value of the link is minimal. It could serve only between deployment from the shuttle and exiting the TDRSS area of visibility about 20 minutes after injection. During this time, the chief value of the link would be for diagnosis of injection stage performance. (If the SC-STS link, next section, were restored, it would fill in much of this gap.)

## 4.2 STS

Communications between the Shuttle Orbiter and the separated spacecraft can be implemented using S-band channels. We have decided to require the spacecraft to accept uplink signals from the Shuttle, but not to implement a return downlink, although later analysis may indicate a desirability to add the downlink also. The pertinent factors are:

(a) We assume that the orientation of the spin axis of the spacecraft plus upper stage at the time of upper stage firing is established open-loop with sufficient accuracy. This accuracy emcompasses the propagation of the Shuttle orientation error at the time of release of the spacecraft plus upper stage, and degradations due to tip-off and spin up.

If subsequent analysis indicates that this process does not lead to sufficient accuracy, then a two-way communication link is necessary: downlink (SC to STS to ground) to report on the spacecraft attitude as determined by the on-board star sensor; and uplink (ground to STS to SC) to insert commands for the spacecraft to correct its attitude by firing precession thrusters in pulses. Both processes and the turnaround on the ground would take place between separation from the Shuttle and upper stage firing, a minimum of 45 minutes.

Note that this closed-loop orientation control protects against only the effects listed. The injection error would still include contributions from thrust vector misalignment and from upper stage impulse magnitude error.

(b) We propose that autonomous spacecraft actions necessary after separation from the Shuttle will be enabled by both of these events: "breakwires" triggered mechanically by the act of separation, and an r.f. signal from the Shuttle to the separated spacecraft. These two events (one alone is not enough) will start the spacecraft sequencer, which commands autonomously all actions necessary on the spacecraft until direct DSN-spacecraft communications are established, 0.5 to perhaps 6 hours after injection. The possibility that a downlink path should also be implemented would be that a return signal may be desirable to confirm the enabling.

(c) Implementation of these features is as follows: Uplink communications (ground to STS to SC) require no change to the spacecraft r.f. system. It is necessary to verify the geometry to assure visibility of the Shuttle via the spacecraft omni antenna. Downlink communications call for the partial restoration of equipment eliminated in Section 3; one S-band diplexer must be used. This will permit transmission from the output of the S-band exciter (125 mW) of one transponder via the omni antenna. This is adequate for the 75 n mi maximum range during the interval noted.

# APPENDIX B

## SEQUENCE OF EVENTS

## (TABLE)

## SEQUENCE OF EVENTS

| Date | Time From Milestone | Event | Operations and Remarks |
|---|---|---|---|
| 88/07/17±10d | L-10h | STS Liftoff | Spacecraft (SC) power and data are through STS Orbiter via Upper Stage (US) |
| | L-9h to L-7h | Cargo Bay Doors Opened | |
| | L-6h to L-2h | Checkout SC, Checkout US | |
| | L-1h | Start Deployment | US + SC ("Payload") are prepared to be separated from the STS Orbiter |
| | L-45m | Deploy Payload | Payload is separated. Depending on which US and ASE are used, the payload may be spun up before or after separation. Automatic nutation control (ANC) is initiated after separation or after spinup, whichever is later. The spin axis of the payload is oriented in the direction of the injection $\Delta V$ |
| | L-~10m to L+~15m | Possible Communication, SC to TDRSS | Would provide a downlink telemetry path covering US firing and separation |
| | L-2m | Ignite US | ANC is terminated |

## SEQUENCE OF EVENTS

(Continued)

| Date | Time From Milestone | Event | Operations and Remarks |
|---|---|---|---|
| | L | End US Firing | L = Launch. At this point all propulsive aspects of the launch process are completed. ANC may be restarted. SC is now on an earth-Mars trajectory |
| | L+3m | Separate SC From US | US may be tumbled by a yo yo despin device at L+4m to avoid possible collision with SC |
| | L+~25m | Establish Communications | This can be done when ground station visibility permits |
| | L+~30m | End of Eclipse | SC emerges from eclipse. The spinning sun sensor will now work |
| | L+~35m | Reduce Spin Rate | SC propulsively reduces its spin rate from ~60 rpm to ~5 rpm |
| | L+ 40m to L+ 90m | Orient SC | By propulsive precession pulses, spacecraft is reoriented from the injection attitude to an attitude more favorable for communications, solar power generation, and temperature control. |
| | L+4d | Calibrate Thrusters | Spacecraft attitude and velocity thrusters are calibrated for performance and for cross coupling effects |

## SEQUENCE OF EVENTS

(Continued)

| Date | Time From Milestone | Event | Operations and Remarks |
|---|---|---|---|
| 88/07/22±10d | L+5d | First Trajectory Correction Maneuver (TCM) | Spacecraft is precessed to the orientation of the desired ΔV vector and trimmed after attitude determination; axial thrusters are fired for predetermined time; trim pulses are fired as dictated by real time doppler measurements; SC is precessed back to cruise attitude. Desired ΔV vector has been determined by doppler tracking from L onward, by DSN stations |
| 88/08/06+10d | L+20d | Second TCM | Same as first TCM |
| | L+20d to A-15d | Interplanetary Cruise Phase | Communications are via SC's upper omni antenna. The despun platform remains spinning and tethered to the spinning section. No booms are deployed (except if necessary for moment-to-inertia control). Occasional precession maneuvers are made to maintain cruise attitude, spin axis ⊥ ecliptic. Periodic two way doppler tracking for orbit determination. Scientific instruments not requiring despinning or boom deployment may be turned on, checked out, and calibrated. |

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## SEQUENCE OF EVENTS

(Continued)

| Date | Time From Milestone | Event | Operations and Remarks |
|---|---|---|---|
| | L+ 70d | Syzygy | Earth passes between SC and sun |
| | A-15d to A- 5d | Approach Tracking | More frequent or continuous tracking to determine accurately the approach trajectory relative to Mars |
| | A-5d | Third TCM | Same as first TCM. This maneuver establishes the final trajectory to approach to the orbit insertion point at Mars |
| | A-2d | Orient SC to MOI Attitude | SC spin axis is precessed to the attitude for Mars orbit insertion. Increase spin rate. Attitude is measured accurately with the star mapper, and trimmed. Low-rate communications are maintained via the omni antenna. |
| | A-1d | Load Commands For MOI | |
| 89/01/28±10d | A | Fire Orbit Insertion Motor (OIM) | A = Arrival at Mars |
| | A+2m | OIM Burnout | SC is now captured in an initial elliptical orbit about Mars<br>Reduce spin rate. |

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## SEQUENCE OF EVENTS

(Continued)

| Date | Time From Milestone | Event | Operations and Remarks |
|---|---|---|---|
| | A+$\sim$1h to A+$\sim$1.8h (or when next visible from earth) | Orient to Nominal Orbit Attitude | This attitude, with the spin axis $\perp$ orbit plane, is used for scientific operations in orbit |
| | A+$\sim$2h, (or when next visible from earth) | Despin Platform | This permits despun instruments to be effective. It also permits the high-gain antenna to be aimed at the earth, for high data rate communications |
| | A+$\sim$2.5h (or when next visible from earth) | Deploy Experiment Booms | Permits deployed instruments to work |
| | A+1d to A+5d | Orbit Shaping (A) | (Aeronomy Mission.) A period of some 5 days in which orbit determination is alternated with orbit trim ($\Delta V$) maneuvers. The object is to modify the initial orbit to conform to the mission requirements within the stated limits. |

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## SEQUENCE OF EVENTS

(Continued)

| Date | Time From Milestone | Event | Operations and Remarks |
|---|---|---|---|
| | A+1d to A+ 100d | Orbit Shaping (C) | (Climatology Mission.) Since the direction of approach to Mars is well out of the plane of the desired sun-synchronous orbit, there will be a "drift period" of perhaps as great as 100 days before the desired orbit plane is attained. At the end of this period there may be large ΔV maneuvers to adjust periapsis altitude, to circularize the orbit, and to set the inclination to make the orbit sun-synchronous. Science data can be taken during the drift period. |
| 89/02 to 90/12 | A+5d to A+692d | Nominal Orbital Mission (A) | This is the period of one Martian year (687 days), the nominal orbital mission lifetime for scientific investigations. Orbital operations during this period include:<br>1) Normal operation of scientific instruments, and acquisition of data from them<br>2) Periodic (daily?) precession maneuvers to maintain the desired spacecraft attitude<br>3) Occasional small ΔV maneuvers to maintain the orbit within limits<br>4) A daily cycle of data acquisition and storage (16 hours) and data transmission (8 hours, less occultation periods)<br>5) Times in each orbital revolution when eclipse and occultation occur<br>6) Periodic (1 or 2 times per week) loading of commands from the ground into the SC's stored command memory |
| 89/05 to 91/03 | A+100d to A+787d | Nominal Orbital Mission (C) | |

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## SEQUENCE OF EVENTS

(Continued)

| Date | Time From Milestone | Event | Operations and Remarks |
|---|---|---|---|
| 90/11 to 92/11 | A+ 692d to A+1379d | Extended Orbital Mission (A) | A second Martian year of scientific investigations in orbit. Orbital operations continue as in first year |
| 91/03 to 93/01 | A+ 787d to A+1474d | Extended Orbital Mission (C) | |
| | After Extended Mission | Orbit Raising | (A) A $\Delta V$ maneuver near apoapsis to raise periapsis<br>(C) 2 $\Delta V$ maneuvers to raise the altitude of the circular orbit<br>These maneuvers are to observe the planetary protection policy: to leave the SC in an orbit which will not decay into Mars' atmosphere for several decades |

APPENDIX C

EQUIPMENT LIST

# MARS ORBITER CONCEPTUAL SYSTEMS DESIGN STUDY

EQUIPMENT LIST

AUGUST 20, 1982

SPACE & TECHNOLOGY GROUP

**KEY:**

E = Existing component

B = Build to fit (using existing technology)

M = Major } modification
m = minor } of existing component

D = Development under way; to be completed

MARS ORBITER STUDY

Subsystem: Mechanical & Structural

Date: 9/20/82

| Component | Qty. | Unit Wt. kg | Total Wt. kg | Unit Pwr. Watts | Dimensions cm | Heritage | | | | |
|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | | Program | Vendor | Status | Key | Remarks |
| STRUCTURE/MECHANICAL | | | | | | | | | | |
| Solar Array Substrate | 1 | 35 | 35 | 0 | $d_1 = 280$, $d_2 = 432$ } x 208 | DSCS II | --- | Adaptation of Existing Technology | B | |
| Primary Structure: Rings, Struts | 1 | 31.2 | 31.2 | 0 | $d_1 = 280$, $d_2 = 432$ } x 208 | DSCS II | --- | | B | |
| Equipment Platform (Spinning) | 1 | 24 | 24 | 0 | d = 325 | DSCS II | --- | | B | |
| Despin Assembly Support Structures | 1 | 6 | 6 | 0 | d = 280 | DSCS II | --- | | B | |
| Miscellaneous Attach Brackets (Includes Isolators) | 50 | .06 | 3 | 0 | 1.2 x 4 | DSCS II | --- | | B | |
| Despun Platform Assy (Cruciform & Rings) | 1 | 26 | 26 | 0 | 280 dia. | DSCS II | --- | | B | |
| OIM Adapter | 1 | 3.8 | 3.8 | 0 | dia. = 100X 100L | DSCS II | --- | | B | |
| Integration (Attachments & Brkts) | Gross | 6 | 6 | 0 | --- | DSCS II | --- | | B | |

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MARS ORBITER STUDY

Subsystem: Mechanical & Structural (Continued)

Date: 9/20/82

| Component | Qty. | Unit Wt. kg | Total Wt. kg | Unit Pwr. Watts | Dimensions cm | Heritage | | | | |
|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | | Program | Vendor | Status | Key | Remarks |
| STRUCTURE/MECHANICAL (Continued) | | | | | | | | Adaptation of Existing Technology | | ORIGINAL PAGE IS OF POOR QUALITY |
| GRS Boom (Climatology) | 1 | 9 | 9 | 20 | 7.5 x 500 | Voyager | Astro Research | | E | |
| Upper Stage/SC Adapter | 1 | 18.8 | 18.8 | 0 | $\left.\begin{matrix} d_1 = 102 \\ d_2 = 229 \end{matrix}\right\} \times 165$ | --- | -- | | B | |
| Upper Stage Separation 2 Marmon Type Ring Clamps 2 Frangible Bolts | 4 | 1.75 | 7 | Negligible | | --- | -- | | E | |
| OIM Attachment | 4 | 1.5 | 6 | Negligible | 102 dia. x X2 Ring | --- | -- | | | |
| | CLIMATOLOGY = | | 175.8 | | | | | | | |
| Booms & Appendages | 3 | 5.1<br>5.1 | 10.2 | 30 | 400<br>400<br>216 | | | | B | Fold out booms |
| Total Vehicle Weight | | | 166.8 | | | | | | | |
| | AERONOMY = | | 177.0 | | | | | | | |

MARS ORBITER STUDY

Date: 9/20/82

Subsystem: Thermal

| Component | Qty. | Unit Wt. kg | Total Wt. kg | Unit Pwr. Watts | Dimensions cm | Heritage | | | | |
|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | | Program | Vendor | Status | Key | Remarks |
| Multilayered Insulation Blankets | 1 set | 16.9 | 16.9 | 0 | 500 $ft^2$ | Pioneer 10/11, HEAO, FSC, DSCS II | TRW | Assemble To Fit | B | Standard Technology |
| Thermistors | 60 | 0.001 | 0.1 | 0 | N/A | Pioneer 10/11, HEAO, FSC, DSCS II | YSI | Purchased Part | E | |
| Thermostatic Switches (Assumes Redundant Switches for Propulsion Components) | 96 | 0.0085 | 0.8 | 0 | N/A | Pioneer 10/11, HEAO, FSC, DSCS II | Sunstrand | Purchased Part | E | |
| Kapton Strip Heaters (For Batteries, Platforms, AKM, DMA, EL/AZ Drives) | 22 | TBD | TBD | 0 | N/A | Pioneer 10/11, HEAO, FSC, DSCS II | Minco | Existing Technology Assemble To Fit | B | |
| Kapton Tape Heaters (For Lines, Tanks) | 2 sets | TBD | TBD | 0 | N/A | Pioneer 10/11, HEAO, FSC, DSCS II | Minco | Assemble To Fit | B | |
| Kapton Strip Heaters For Thruster Valves | 24 | TBD | TBD | 0 | N/A | Pioneer 10/11, HEAO, FSC, DSCS II | Minco | Assemble To Fit | B | |

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MARS ORBITER STUDY

Date: 9/20/82

Subsystem: Electrical Power

| Component | Qty. | Unit Wt. kg | Total Wt. kg | Unit Pwr. Watts | Dimensions cm | Heritage | | | | |
|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | | Program | Vendor | Status | Key | Remarks |
| Batteries (Ni-Cd) | 3 | 17.3 | 51.9 | --- | 20.3 x 14.4 x 34.5 | DSCS II | GE for cells | Existing Technology | E | |
| Power Control | 1 | 13.7 | 13.7 | --- | 20.3 x 26.5 x 61 | HEAO | -- | Adaptation of Existing Technology | B | New converter requirements |
| Shunt Limiter | 1 | 6.4 | 6.4 | --- | Part of Power Control | DSP | -- | | | |
| Electrical Distribution | 1 | 9.0 | 9.0 | --- | --- | --- | -- | | B | Spinning |
| Electrical Distribution | 1 | 1.5 | 1.5 | --- | --- | --- | -- | | B | Despun |
| Harness | 1 | 23 | 23 | --- | --- | --- | -- | Assemble To Fit | B | Spinning |
| Harness | 1 | 6.5 | 6.5 | --- | --- | --- | -- | | B | Despun |
| Solar Cells, Cover-slides, Etc: | | | | | | | | | | |
| Climatology | 1 set | 30.4 | 30.4 | --- | --- | GRO, Landsat | Spectrolab Etc. | Purchased Parts | E | (Concurrent Programs) |
| Aeronomy | 1 set | 22.4 | 22.4 | --- | --- | GRO, Landsat | Spectrolab Etc. | | E | (Concurrent Programs) |

MARS ORBITER STUDY

Subsystem: Command and Data Handling

Date: 9/20/82

| Component | Qty. | Unit Wt. kg | Total Wt. kg | Unit Pwr. Watts | Dimensions cm | Heritage | | | | |
|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | | Program | Vendor | Status | Key | Remarks |
| Command Decoder Processor | 2 | 2.2 | 4.4 | 5.7 | 8.9 x 16.5 x 12.7 | TDRSS | Gulton | Purchased Component | E | |
| Remote Interface Unit | 2 | 1.1 | 2.2 | 2.8 | 4.4 x 8 x 6.3 | TDRSS | Gulton | | E | |
| Data Handling Processor | 2 | 3.85 | 7.7 | 10.5 | 8.9 x 26.6 x 12.7 | TDRSS | Gulton | | E | |
| Remote Interface Unit | 2 | 1.9 | 3.8 | 5 | 4.4 x 10 x 6.3 | TDRSS | Gulton | | E | |
| Tape Recorder | 2 | 8.5 | 17 | 20.4 | 17.8 x 33 x 24 | ISPM | Odetics | | D | Development started under ISPM, based on NASA $10^8$ bit standard. Select record and play-back rates |

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MARS ORBITER STUDY

Subsystem: Attitude Control

Date: 9/20/82

| Component | Qty. | Unit Wt. kg | Total Wt. kg | Unit Pwr. Watts | Dimensions cm | Heritage | | | | |
|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | | Program | Vendor | Status | Key | Remarks |
| Despun Mechanical Assy | 1 | 12.3 | 12.3 | 8 | 14.6 dia x 45 long | DSCS II | Ball Aerospace | Purchased Component | E | |
| Biax Drive Assy | 1 | 6.5 | 6.5 | [illegible] | 17 dia x 34 long | DSCS II Landsat | | Modification of Existing Technology | M M | Same gimbal and actuator; different geometry |
| Star Sensor | 1 | 2.95 | 2.95 | 1.9 | 49.5 x 45.7 x 10.2 | ISPM | Ball Aerospace | ↓ | M | Light shade geometry |
| Sun Sensor | 1 | 0.32 | 0.32 | 0.5 | 5.7 x 5.7 x 6.3 | Pioneer | Honeywell | Purchased Component | E | |
| Control Electronics Assy | 1 | 4.6 | 4.6 | 10 | 35 x 20 x 23 | DSCS II | | Adaptation of Existing Technology | B | New combination of functions |
| Valve Driver Assy | 1 | 1.4 | 1.4 | 2 | 18 x 23 x 5 | Pioneer or FltSatCom | | ↓ | M | Match thruster complement |
| Wobble Damper | 1 | 0.23 | 0.23 | --- | 5.1 dia x 11.4 long | Pioneer | | ↓ | M | |
| Accelerometer | 1 | 0.5 | 0.5 | --- | --- | --- | | Purchased Part | E | |

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## MARS ORBITER STUDY

Subsystem: Communications

Date: 9/20/82

| Components | Qty. | Unit Wt. kg | Total Wt. kg | Unit Pwr. Watts | Dimensions cm | Heritage | | | | |
|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | | Program | Vendor | Status | Key | Remarks |
| S/X Transponder | 2 | 5.1 | 10.2 | 13.1 | 28.5 x 14 x 7.9 | NASA Std. | Motorola | Purchased Component | E | NASA Standard (DSN) |
| S-Band Transfer Switch | 1 | 0.28 | 0.28 | -- | 4.5 x 3.3 x 6.4 | Pioneer 11 | -- | ↓ | E | |
| S-Band Omni Antenna | 1 | 1.0 | 1.0 | -- | -- | HEAO, Pioneer | -- | Use As Is | E | |
| X-Band TWTA | 2 | 5.2 | 10.4 | 72 | 36.7 x 15.2 x 11.4 | DSCS II | Watkins Johnson | Purchased Component Modified | M | Reduce output power from 40 to 20% to fit SC power budget |
| X-Band Transfer Switch | 2 | 0.34 | 0.68 | -- | 6.1 x 5.5 x 3.5 | DSCS II | -- | Purchased Component | E | |
| X-Band Hybrid | 1 | 0.17 | 0.17 | -- | 4.5 x 2.5 x 1.9 | DSCS II | -- | Adaptation of Existing Technology | E | |
| S/X-Band HGA | 1 | 10.2 | 10.2 | -- | 44" Dish | DSCS II | -- | New Dual Feed | M | Existing dish new dual-feed required |
| X-Band Horn | 1 | 2.0 | 2.0 | -- | -- | DSCS II | -- | Adaptation of Existing Technology | M | pattern shaping? |
| X-Band Bicone | 1 | 1.0 | 1.0 | -- | -- | -- | -- | ↓ | M | pattern shaping? |
| RF Cables and Conn. | 1 set | 1.0 | 1.0 | -- | -- | -- | -- | Assemble To Fit | B | |
| X-Band Waveguide | 1 set | 0.5 | 0.5 | -- | -- | -- | -- | ↓ | B | |

MARS ORBITER STUDY

Date: 9/20/82

Subsystem: Propulsion

| Component | Qty. | Unit Wt. kg | Total Wt. kg | Unit Pwr. Watts | Dimensions cm | Heritage | | | | |
|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | | Program | Vendor | Status | Key | Remarks |
| Propellant Tanks | 4 | 3.6 | 14.4 | N/A | 55.9 dia. | Gemini | Fansteel | Purchased | E | ORIGINAL PAGE IS OF POOR QUALITY |
| Pressure Transducers | 2 | 0.27 | 0.54 | .20 | -- | TDRSS | Gould | | E | |
| Fill and Drain Valves | 2 | 0.14 | 0.28 | N/A | -- | TDRSS | Pyronetics | | E | |
| Propellant Filters | 2 | 0.13 | 0.26 | N/A | -- | TDRSS | Wintec | | E | |
| Latching Isolation Valves | 6 | 0.27 | 1.62 | 13.6 (Rated) | -- | TDRSS | Hydraulic Research | | E | |
| 0.1 $lb_f$ Thrusters | 4 | 0.4 | 1.60 | 4.8 (Rated) | -- | TDRSS | --- | Existing Component | E | |
| 5.0 $lb_f$ Thrusters | 8 | 0.56 | 4.54 | 7 (Rated) | -- | GRO | --- | | E | |
| Propellant Lines and Supports | 1 set | -- | 2.3 | N/A | -- | -- | --- | Made To Fit | B | |
| Star 37XF (Climatology) | 1 | -- | 796 (1) | N/A | -- | -- | Thiokol | Purchased | D | Current Star 37 Qualification program |
| Star 37N (Aeronomy) | 1 | -- | 477 (1) | N/A | -- | -- | Thiokol | | D | |

(1) Weight includes propellant required for orbit insertion.