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Mars Orbiter Study - Final Report
Volume 1, Summary

R. Drean
D. MacPherson
D. Steffy
T. Vargas
B. Shuman
K. Anderson
B. Richards

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Mars Orbiter Study - Final Report
Volume 1, Summary

Hughes Aircraft Company
El Segundo, California

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National Aeronautics and
Space Administration

Ames Research Center
Moffett Field, California 94035

CONTENTS

1. INTRODUCTION

This final report documents the results of study contract NAS2-11224 to define application of existing spacecraft to Mars climatology and aeronomy missions. The study focuses on minimizing Mars Orbiter cost and risk by

1) developing the basic spacecraft as part of the HS 376 communications satellite product line,

2) selecting subsystem units from other current product lines if they better meet Mars mission objectives without costly modification,

3) designing a common spacecraft for both the climatology and aeronomy missions, which allows significant cost savings for a two mission set,

4) using the existing integrated propulsion stage, simple frisbee deployment, and compact Shuttle launch configuration adopted by the Intelsat VI spacecraft,

5) minimizing operations cost by making the spacecraft simple to operate and providing autonomous control of specific critical functions, and

6) maintaining adequate performance margins.

Our experience in comparable missions, notably Pioneer Venus, allows us to understand the detailed requirements unique to deep space missions such as DSN compatibility and integration of complex science instruments. Attention to these details and extensive use of a currently-flying spacecraft bus gives the design sufficient maturity to validate cost estimates in this report.

1.1 Study Conclusions

The HS 376 spacecraft is our basis for Mars Orbiter designs. Of existing spacecraft design candidates, it clearly has the lowest cost due to its production line status: five launched and 22 under development with delivery during the next 5 years. We anticipate further sales, but even without them the production line will still be running when the Mars Orbiter begins hardware development in 1985-86.

The HS 376 can be launched on the Delta 3910/PAM-D, Delta 3920/PAM-D, Ariane/Sylda, and STS/PAM-D and also easily mates to the integrated propulsion stage. The spacecraft uses a telescoping solar panel to achieve high power output (up to 1100 watts beginning of life at Earth) while fitting within the payload envelopes of the three launch vehicles. This configuration features the simplicity and high reliability of a spinning spacecraft during the critical periods of Shuttle ejection, interplanetary trajectory injection, and Mars orbit insertion.

The HS 376 is a gyrostat when the solar panel is deployed and platform despun. The gyrostat technique, used previously on Hughes' Intelsat IV/IVA and Comstar spacecraft, allows onboard control of spacecraft wobble and nutation to create a stable platform well within the specified pointing needs of the science instruments.

HS 376 capabilities are particularly well suited for the Mars Orbiter: 1) the configuration requires only minimal changes; 2) solar panel power is adequate at the maximum solar distance of 1.67 AU with a modified solar cell layout for the lower solar intensity at Mars; 3) the large despun section easily accommodates the specified science instruments and new complement of communications, data handling, and command equipment; 4) attitude determination, stability, and platform pointing accuracy are well within the specified requirements; 5) aeronomy orbit insertion motor (OIM) interfaces are in place and the climatology OIM requires only a simple adapter; 6) the liquid propulsion subsystem requires no changes; and 7) the attitude control subsystem requires no changes other than direct substitution of Intelsat VI control electronics and added star sensor(s).

The spinning section of the Mars Orbiter designs contains HS 376 propulsion, power, and attitude control equipment with few changes. The revised despun section layout places a science platform on top of the spacecraft in place of the communications payload normally located there. The despun communications equipment combines GOES, PV, Galileo Probe, and other existing components for S- and X-band links. Data handling and command units, derived from GOES and Leasat, adequately support instrument and housekeeping requirements. Existing tape recorders provide sufficient storage capacity for a simple operating sequence.

Figure 1-1 shows how the HS 376 basic bus becomes the climatology and aeronomy orbiters. The spacecraft designs for the climatology and aeronomy missions are very similar. Instrument orientation requirements and mission geometry dictate the few differences between the climatology and aeronomy designs. The climatology mission specifies nadir-oriented science but the aeronomy mission science is ram-oriented. Different orbit, solar phase, and science operation requirements result in different spin axis orientations.

Mars Orbiter designs differ only in the science instrument layout, high gain antenna orientation, and orbit insertion motor size. Each mission uses the integrated propulsion stage with a different offload.

The Mars Orbiter spacecraft meet the objective of an existing design basis. Of the 33 unit types, 26 are currently in production, five require some modification, and only two are new. These new units, a remote command unit and antenna azimuth positioner, are straightforward developments and use only existing components.

Adequate margins for all spacecraft resources reduce the risk of significant cost increase to accommodate future changes of instrument or mission requirements. The injection and orbit insertion motors and liquid bipropellant capacity support the Mars mission requirements with a spacecraft mass that includes allocated contingency and additional reserve.

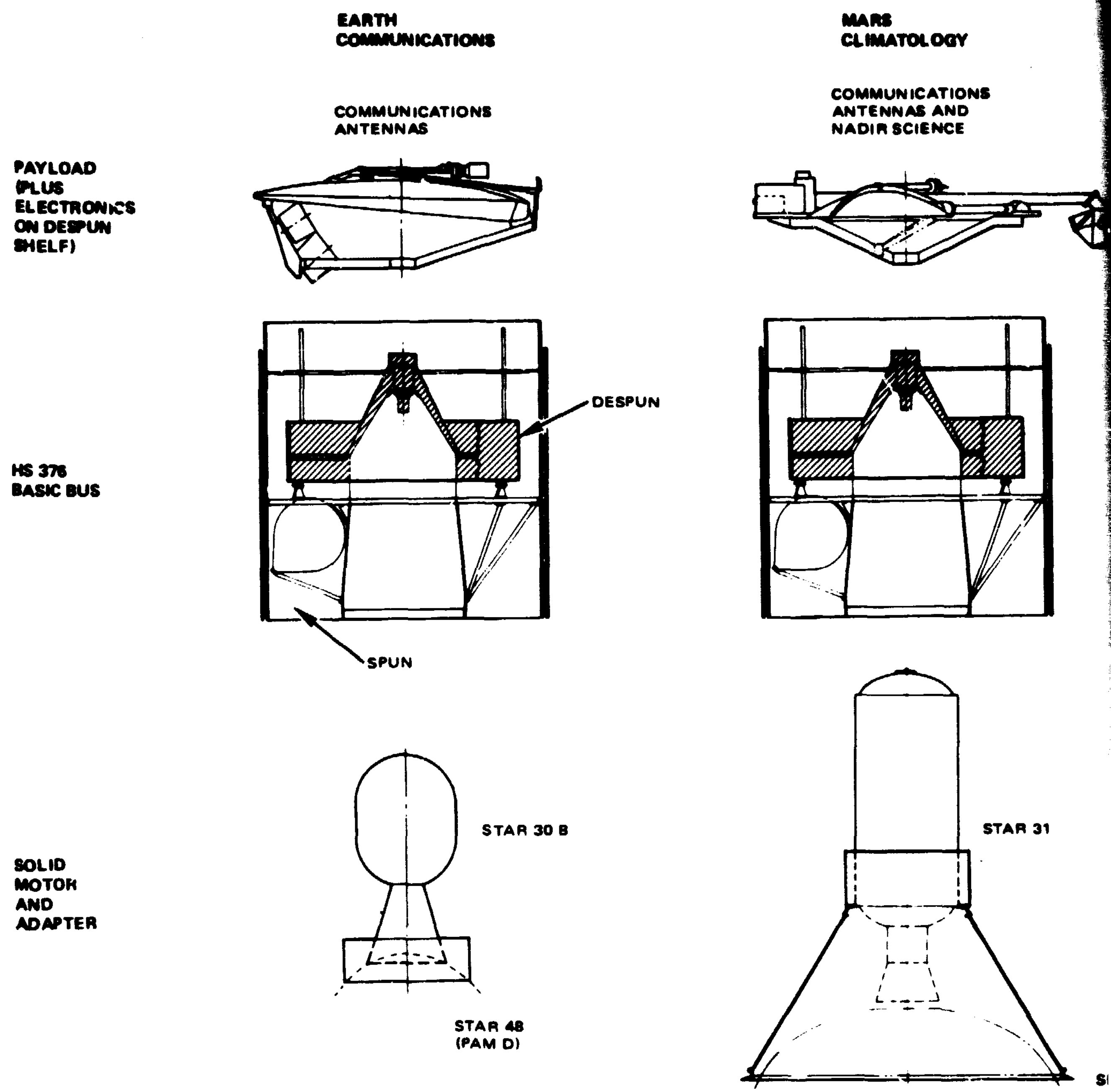

FIGURE 1-1. ADAPTATION OF HS 376 FOR MARS ORBITERS

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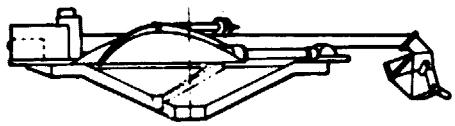
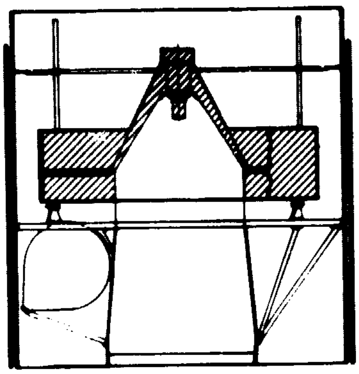
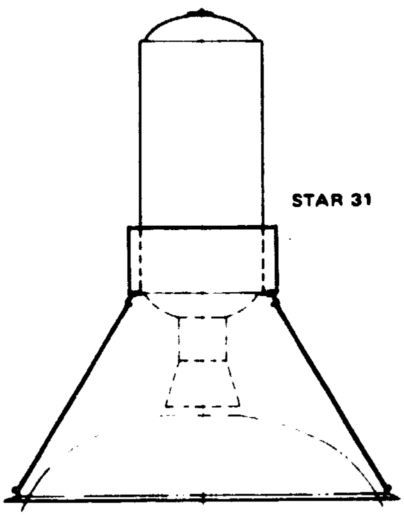
MARS CLIMATOLOGY

COMMUNICATIONS ANTENNAS AND NADIR SCIENCE

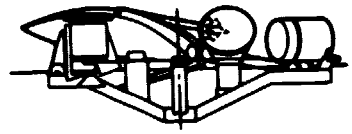


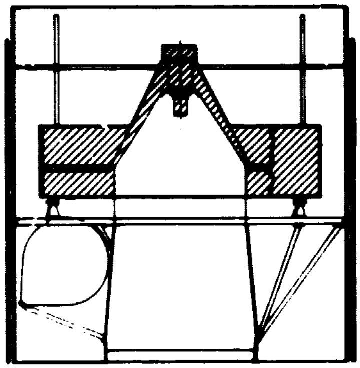


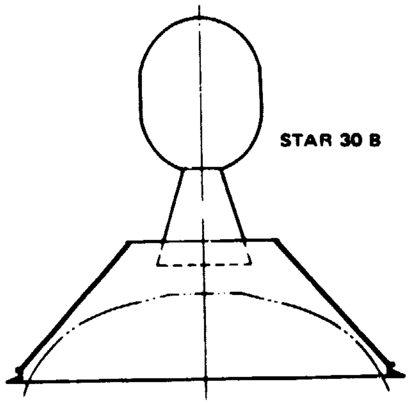


MARS AERONOMY

COMMUNICATIONS ANTENNAS AND RAM SCIENCE







S ORBITERS

The SRM-1 injection stage can inject increased spacecraft mass by reducing the propellant offload from the 21% and 37% values of the climatology and aeronomy missions. The STAR-31 (20% offload) and STAR-30B both provide excess orbit insertion capability which is spoiled by pointing bias; any mass increase could be accommodated by reducing this bias.

1.2 Recommendations for Further Study

The Mars Orbiter missions require no new technology. On the contrary, the high degree of heritage supports full scale development now. The simple integrated propulsion stage, which injects the spacecraft into its cruise trajectory following Shuttle launch, requires no further study.

No additional studies are required to verify the capability of achieving climatology and aeronomy mission objectives with minimum risk and cost. Hughes typically undertakes new fixed-price HS 376 programs with a degree of spacecraft design definition comparable to what now exists for the Mars Orbiters.

The validity of the program cost estimate and spacecraft design depend upon the similarity of the final science instruments and mission objectives to the present definitions and assumptions. Therefore, spacecraft design during a Phase B study requires further mission and science requirement definition including: 1) operation during the drift phase of the climatology mission, 2) subsolar and antisolar aeronomy mapping objectives, and 3) desired sampling density of the aeronomy mission. Review of all instrument interfaces and requirements, especially the layout and pointing of the FPI, the high sampling rate of the GRS and MSM, and the tight pointing requirement of the alternate climatology instruments, should verify their specifications.

Mission parameters requiring definition include the DSN frequency constraints, the option for data recovery during a DSN outage, the need for redundant data return (three tape recorders), the preferred launch year for a two mission sequence, and the STS thermal environment.

1.3 Final Report Organization

This final report consists of three parts. This part, Volume 1, summarizes the study results including mission design, science instrument accommodation, and spacecraft design for Mars climatology and aeronomy missions. Volume 2 provides additional detail on each of these topics, describes test and operations program elements, and makes recommendations for further definition of requirements before Phase B. Volume 3 describes programmatic assumptions and gives a cost estimate for development of the Mars Orbiter spacecraft.

2. DESIGN DRIVERS

We define design drivers as requirements or constraints which force a departure from existing hardware. The most significant design drivers require new hardware development while other design drivers only require some modification of existing hardware. The only significant new element of our Mars Orbiter designs, the despun remote command unit, is required because the LSI components which typically provide this function in existing units are no longer available. All other new elements including the HGA azimuth positioner and science support structure are low-risk developments and are not driven by specific requirements.

DSN compatibility and the general capability to control and process data from science instruments require the modifications to the communications, data handling, and command subsystems. Again, no specific design drivers are responsible.

The alternate climatology mission which specifies non-sun-synchronous operation becomes a significant design driver if science instruments must sample in twilight orbits (between 3 and 9 o'clock). The baseline Mars Orbiter configuration with spin axis normal to the orbit plane will not generate adequate solar power at these hour angles; twilight operation requires a different spacecraft design.

The optional climatology instruments may also become design drivers. The increased mass is within the spacecraft reserve. The increased power at worst requires a longer solar panel which is inexpensive and has no risk. The increased data rate of Option #3 requires increased tape recorder capacity and more significantly, requires a higher playback telemetry rate. This in turn requires a new communications antenna or higher power transmitter and larger solar panel.

The tight 0.08° pointing control requirement of the alternate instruments needs on-board attitude determination and upate or requires continuous DSN tracking and ground attitude determination and update.

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3. CLIMATOLOGY MISSION DESIGN

Table 3-1 lists the characteristics of the climatology mission interplanetary trajectory, drift orbit, and operational orbit. The following sections describe the climatology mission design.

3.1 Interplanetary

The baseline climatology orbiter launches during a 10-day window beginning 29 June 1988. The integrated propulsion stage (IPS) injects the separated spacecraft mass of 1924 kg (plus a 75 kg adapter) at a launch energy, C_3, of 11.64 km^2/s^2 into a Type I transit trajectory. The flight to Mars takes 194 days.

TABLE 3-1. CLIMATOLOGY BASELINE MISSION CHARACTERISTICS

Interplanetary Trajectory	
Launch date	29 June 1988
Arrival date	9 January 1989
Time of flight, days	194.69
C_3, km^2/sec^2	12.52
Declination of launch asymptote, deg	14.55
V_∞, km/sec	2.79
Declination of arrival asymptote, deg	3.70
TCM ΔV, m/sec	96 (3σ)
Drift Orbit	
Initial hour angle	7:45 a.m.
Drift rate, deg/day	1.1295
Time to full science operation capability, days	20
Time to 1:30 hour angle, days	129
Time to plane change, days	143
Plane change ΔV, m/sec	184 (3σ)
Operational Orbit	
Altitude, km	300 (circular)
Inclination, deg	92.64 (sun-synchronous)
Period, hr	1.893
Spacecraft attitude	Normal to orbit plane
Despun instrument orientation	Nadir
Local hour angle	1:30 to 3:00 p.m.
Orbit sustenance ΔV, m/sec	26 (3σ)
Planetary quarantine ΔV, m/sec	100
Final orbit, km	540 (circular)

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The spacecraft provides 96 m/s for trajectory correction maneuvers (TCMs). The calculated magnitude of the TCMs assumes 3σ errors in IPS performance and STS attitude, and failure of one STS vernier engine.

3.2 Approach and Orbit Insertion

The spacecraft arrives at Mars with an approach velocity, V_∞, of 2.79 km/s. The initial hour angle of the orbit plane is 7:45 a.m.

A B-plane analysis of the approach to Mars predicts 2.1° error in orbit plane inclination and 215 km error in insertion point altitude. Predicted capture orbit errors include 0.63% uncertainty in motor performance. About 35 m/s correction ΔV circularizes the orbit for nominal insertion parameters. Compensating for RSS targeting errors adds 150 m/s, so a total ΔV of 185 m/s corrects the capture orbit to 300 km circular.

3.3 Drift Orbit

The ellipticity of Mars' orbit causes a variation of hour angle for a sun-synchronous orbit. The desired 1:30 to 3:00 p.m. hour angle requires a 92.64° inclined orbit plane with a nodal rate of 0.524°/day. To reach the desired hour angle range from the 7:45 a.m. capture orbit, the orbit plane inclination is initially greater than 92.64° causing a higher rate of nodal precession. When the spacecraft position intersects the desired hour angle profile (Figure 3-1) a plane change maneuver decreases the nodal rate to the desired sun-synchronous value (0.524°/day).

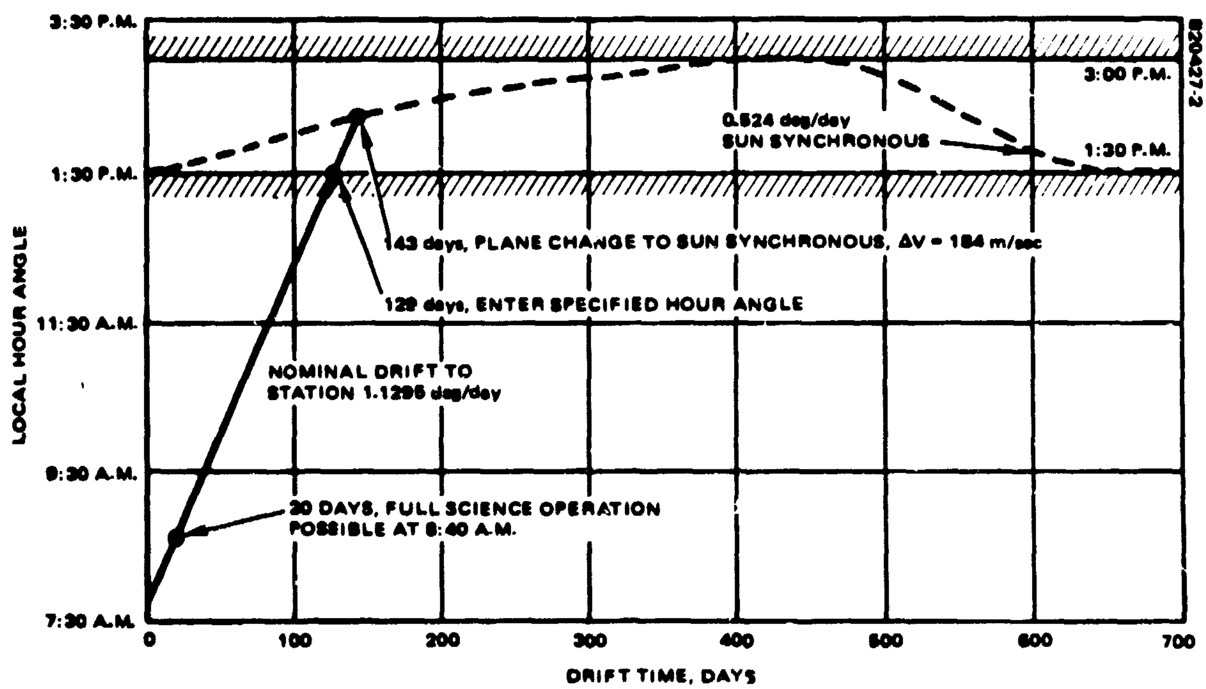

FIGURE 3-1. BASELINE ORBIT CONTROL STRATEGY

The magnitude of the plane change depends on the amount of maneuver propellant remaining after the TCMs and capture orbit trim and the amount to be saved for on-orbit functions. The 3σ TCM analysis allows a baseline plane change ΔV of 184 m/s with an initial drift rate of 1.1295°/day and a drift time of 143 days. After 129 days the orbit crosses the 1:30 p.m. hour angle and then remains within the specified science operating sun angle range. In most cases, lower actual TCM propellant use will leave propellant for a larger plane change. This allows a faster drift rate.

After 20 days of drift, at 8:40 a.m. and later hour angles, the available power permits full science instrument operation. Restricted duty cycle operation can begin immediately after circularization.

3.4 On Orbit

The 300 km altitude, circular, nearly-polar orbit, allows uniform instrument sampling during the entire Mars orbit. The spacecraft attitude remains normal to the orbital plane. The positioning of the instruments on the despun shelf accommodates continuous nadir tracking (Figure 3-2). The spacecraft body shades the instruments from the sun.

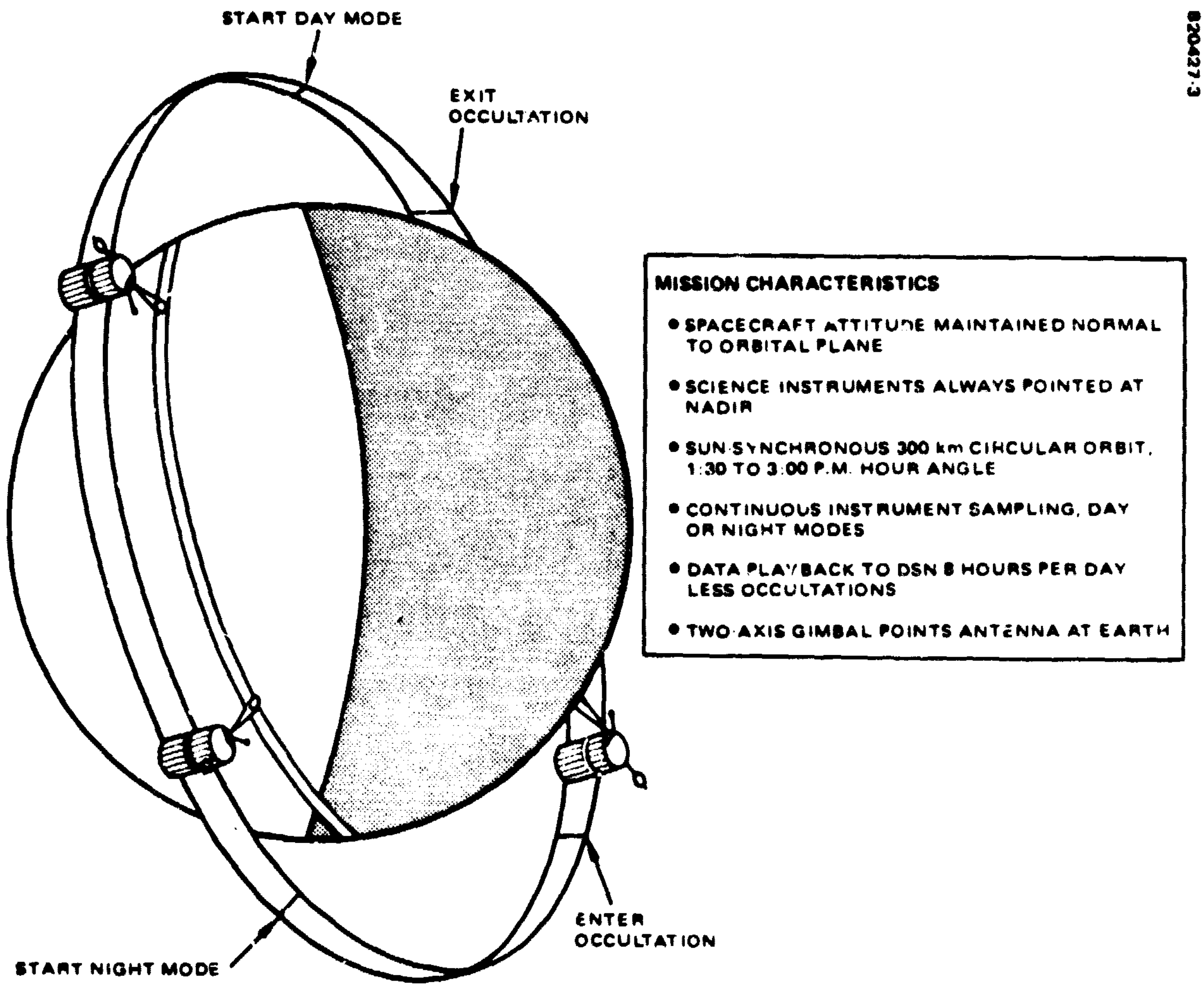

FIGURE 3-2. MARS CLIMATOLOGY ON-ORBIT OPERATION

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Figure 3-2 depicts the acquisition and playback sequence. Earth acquisition, command load, and orbit determination occupy part of the communications period. This communications period lasts for 8 hours per day less occultations (42 minutes maximum). Antenna elevation and azimuth gimbals adjust for the motion of the Earth's direction during the communications period. Orbit determination does not interfere with data playback and playback does not interrupt science instrument sampling and data storage. The recorders have enough capacity to tolerate a single DSN outage.

Solar and drag torques cause the spacecraft to precess away from the desired attitude. The specifications call for attitude control to within 1°. For the climatology mission, the solar and drag effects cause precession rates of 0.04°/day and 0.06°/day respectively requiring 0.9 kg of maneuver propellant per Mars year. Orbit sustenance, to correct the orbit decay due to the effect of drag, accounts for another 13 m/s per Mars year.

3.5 End of Mission

The spacecraft propellant supports the specified orbit for the nominal one Mars year mission and for an additional Mars year. If launch and orbit insertion errors do not reach the budgeted 3σ values propellant will remain for additional orbit operation as desired. At the end of the mission a planetary quarantine maneuver of 100 m/s raises the spacecraft into a very stable 540 km altitude circular orbit, high enough to eliminate drag perturbations. The spacecraft will remain in orbit for centuries, meeting Mars planetary protection requirements.

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4. AERONOMY MISSION DESIGN

Table 4-1 lists the significant characteristics of the aeronomy mission interplanetary trajectory and operational orbit. The following sections describe the aeronomy mission design.

4.1 Interplanetary

The baseline aeronomy orbiter launches during a 10 day window beginning 29 June 1988. The integrated propulsion stage (IPS) injects the separated spacecraft mass of 1344 Kg (plus a 60 Kg adapter) at a launch energy, C_3, of 11.64 km^2/s^2.

The spacecraft provides 96 m/s for trajectory correction maneuvers (TCMs). The calculated magnitude of the TCMs assumes 3σ launch errors due to uncertainties in IPS performance and pointing.

TABLE 4-1. AERONOMY BASELINE MISSION CHARACTERISTICS

Interplanetary Trajectory	
Launch date	29 June 1988
Arrival date	9 January 1989
Flight time, days	194.69
Launch Energy (C_3), km^2/sec^2	11.64
Declination of launch asymptote, deg	14.55
V-infinity magnitude, km/sec	2.79
Declination of arrival asymptote, deg	3.7
TCM magnitude, m/sec	96
Operational Orbit	
Periapsis altitude, km	150
Apoapsis altitude, km	10192 ($3R_M$)
Inclination, deg	77.5
Period, hr	6.68
Spacecraft Attitude	Normal to Mars orbit plane
Instrument orientation	Along periapsis velocity
Orbit sustenance ΔV, m/sec	126.4 (3σ)
Planetary quarantine ΔV, m/sec	100
Final periapsis altitude, km	1008

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4.2 Approach and Orbit Insertion

The spacecraft approaches Mars with an approach velocity of 2.79 km/s. B-plane analysis of the approach to Mars predicts 3σ errors of 2.1 degrees orbit plane inclination and 215 km insertion point altitude. Predicted capture orbit errors account for .63 percent uncertainty in motor performance. About 27 m/s correction ΔV lowers periapsis from 365 km to 150 km for nominal insertion parameters. Compensating for RSS targeting errors adds 83 m/s, so a total ΔV of 110 m/s corrects the capture orbit.

4.3 Subsolar and Anti-solar Targeting

The oblateness of Mars causes the line of nodes and line of apsides to rotate, resulting in a precession of periapsis about the planet. Periapsis should approach subsolar and anti-solar locations to best satisfy scientific goals. The geometry depends on orbit inclination, apoapsis altitude, and arrival date. Figure 4-1 follows periapsis latitude and longitude relative to the sun for two Mars years.

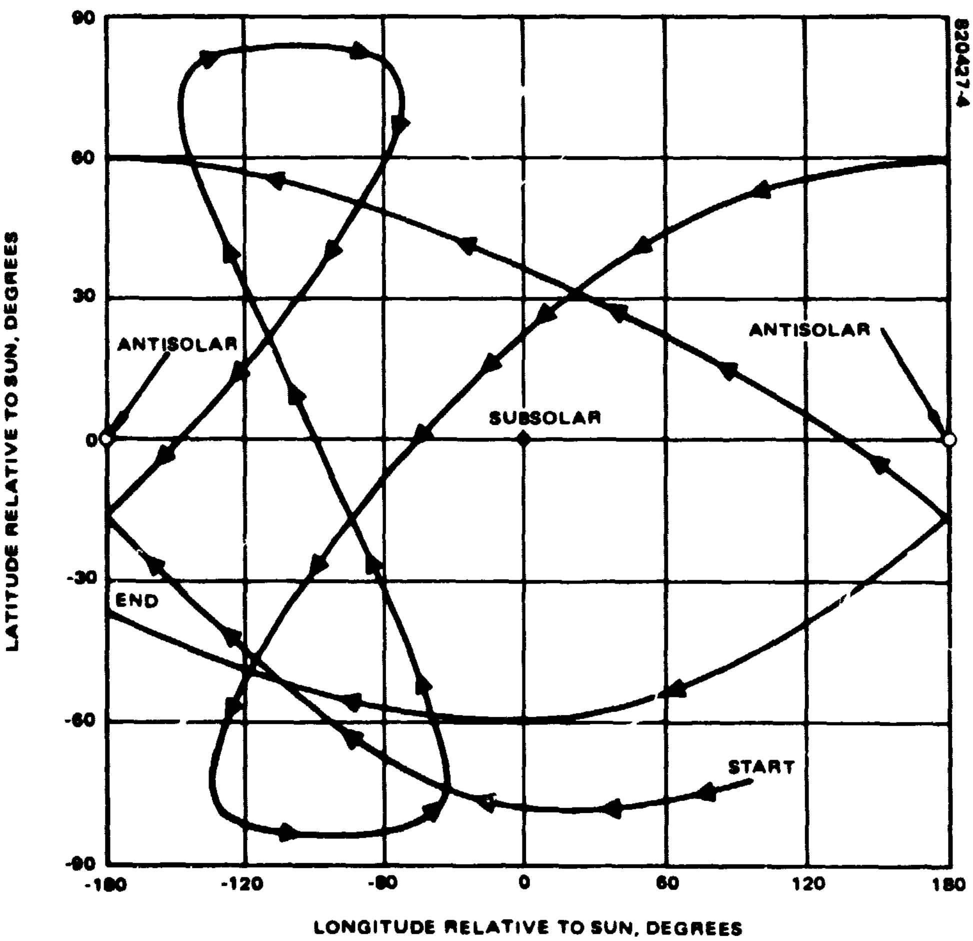

FIGURE 4-1. PERIAPSIS MOVEMENT DURING TWO MARS YEARS

During the first Mars year periapsis comes within 20 degrees of both subsolar and anti-solar points. The pattern does not repeat the second year. Adjusting the inclination can bring periapsis closer to the subsolar or anti-solar points. The elliptical, highly-inclined orbit provides a wide range of altitudes at solar wake crossing.

4.4 On Orbit

Figure 4-2 summarizes the on-orbit operation. The spacecraft spin axis attitude remains normal to the Mars orbit plane placing the sun continuously perpendicular to the solar panels for maximum power. A ram-oriented equipment shelf with zero to 90° elevation travel keeps the instruments pointed in the velocity direction at periapsis. Despun platform azimuth provides the other degree of freedom; flipping the spacecraft five times during the two Mars year mission reaches both positive and negative elevations.

The acquisition and playback sequence is similar to the climatology sequence described in Section 3.4. Earth acquisition, command load, and orbit determination occupy part of the communications period which lasts for 8 hours per day less

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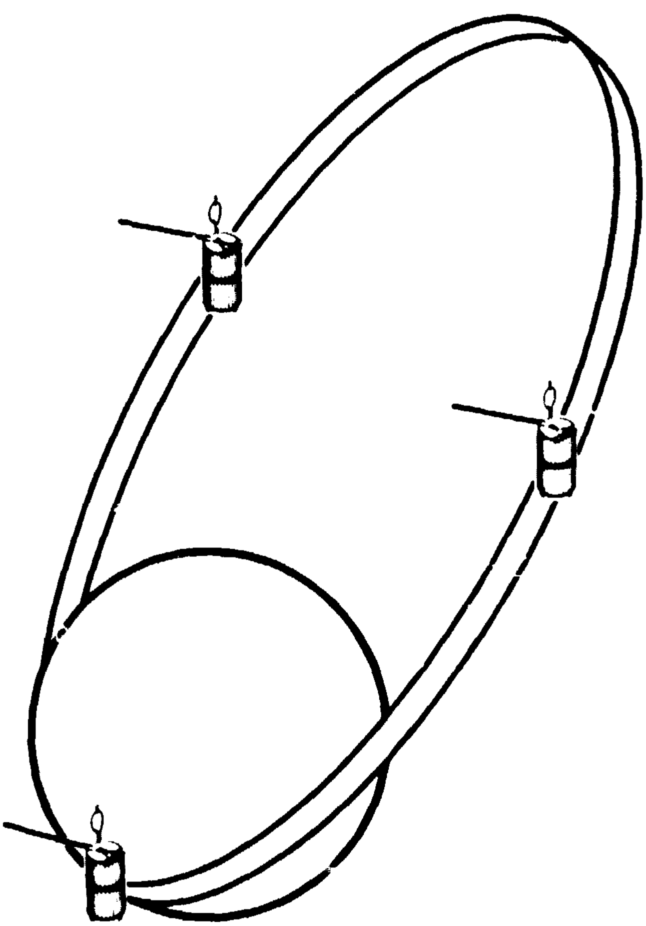

MISSION CHARACTERISTICS

- ELLIPTICAL ORBIT
  - ≈160 km PERIAPSIS ALTITUDE
  - ≈6.7 hr PERIOD
  - ≈80° INCLINATION
- MARS OBLATENESS CAUSES ORBITAL PLANE TO ROTATE IN LONGITUDE AND PERIAPSIS TO ROTATE IN LATITUDE – SUBSOLAR AND ANTI-SOLAR MAPPING AT VARIOUS ALTITUDES
- SPACECRAFT ATTITUDE MAINTAINED NORMAL TO MARS ORBIT PLANE
- SHELF INSTRUMENTS ALIGNED WITH RAM AT PERIAPSIS AND HELD FIXED THROUGHOUT ORBIT
- ALL DATA STORED, PLAYBACK TO DSN 8 HOURS PER DAY LESS OCCULTATIONS

FIGURE 4-2. MARS AERONOMY ON-ORBIT OPERATION occultations (70 minutes maximum). Antenna elevation and azimuth gimbals adjust for the motion of the Earth's direction during the communications period. Orbit determination does not interfere with data playback, and playback does not interrupt science instrument sampling and data storage. The recorders have enough capacity to tolerate a single DSN outage.

Solar and drag torques cause the spacecraft to precess away from the desired attitude. The specifications call for attitude control to within 1°. For the aeronomy mission, the solar and drag effects cause precession rates of .05 and 1.3 degrees per day, respectively, requiring 14.1 kg of bipropellant per Mars year for correction. Orbit sustenance, to correct the orbit perturbations due to drag, oblateness, and solar influence, requires 63 m/s per Mars year.

4.5 End of Mission

The spacecraft propellant maintains the specified orbit for the nominal one Mars year of operation and for the second Mars year of extended operation. The propellant available for additional orbit operation depends on the launch and orbit insertion errors; 99.7 percent of the time the budgeted propellant will exceed the amount needed. A 100 m/sec planetary quarantine maneuver at the end of the operational mission raises periapsis altitude to 1008 km to eliminate drag perturbations. The spacecraft will remain in orbit for centuries, with the specified environmental models, meeting Mars planetary protection requirements.

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5. SCIENCE INSTRUMENT ACCOMMODATION

The two Mars Orbiter spacecraft accommodate the climatology and aeronomy science instruments within the capabilities of existing and proven designs. This section summarizes instrument mechanical, electrical, and operational accommodation for the baseline missions and instrument complements and for the alternate climatology instrument payloads.

5.1 Mechanical Integration

Climatology Payload

The baseline climatology payload consists of a pressure modulated radiometer (PMR), frost infrared spectrometer (FIS), and gamma ray spectrometer (GRS). Three alternative payloads combine other instruments, as shown in Table 5-1. Table 5-2 lists the size, mass, orientation, and electrical requirements of each instrument.

TABLE 5-1. CLIMATOLOGY INSTRUMENT COMPLEMENTS

Baseline
Pressure modulated radiometer (PMR)
Frost infrared spectrometer (FIS)
Gamma ray spectrometer (GRS)
Option 1
Baseline instruments (PMR, FIS, GRS)
Ultraviolet ozone instrument (UV03)
Ultraviolet hydrogen photometer (UVHP)
Radar altimeter (RA)
Option 2
Baseline instruments (PMR, FIS, GRS)
Fabrey-Perot interferometer (FPI)
Option 3
PMR
GRS
RA
Multispectral mapper (MSM)

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TABLE 5-2. CLIMATOLOGY INSTRUMENT CHARACTERISTICS

Instrument	Mass, kg	Power, W	Dimensions, cm	Data Rate, bps	Instrument Orientation
Pressure modulated radiometer	18	35	35 x 30 x 20	140	Nadir
Frost infrared spectrometer	5	4	20 x 30 x 15	120	Nadir
Gamma ray spectrometer	14[1]	20	30 x 30 x 40	1024	Nadir
Ultraviolet ozone instrument	3.6	1.5	36 x 14 x 14	64	Forward limb
Ultraviolet hydrogen photometer[2]	1.2	3	20 x 3 x 3	8	Forward limb Nadir Zenith
Radar altimeter	9	16		100	Nadir
Antenna			39 dia.		
Electronics			20 x 10 x 10		
Fabrey-Perot interferometer	20	12		256	Two 10° cones at 45° and 135° from ram viewing side limb[4]
Optics			100 x 30 x 15		
Electronics			25 x 20 x 25		
Multispectral mapper	12	12	25 x 25 x 25	1000[3]	Nadir

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1 Boom mass not included
2 Data for three sensors
3 Average rate over 24 hours
4 Instrument optics provide 23° slant angle to limb

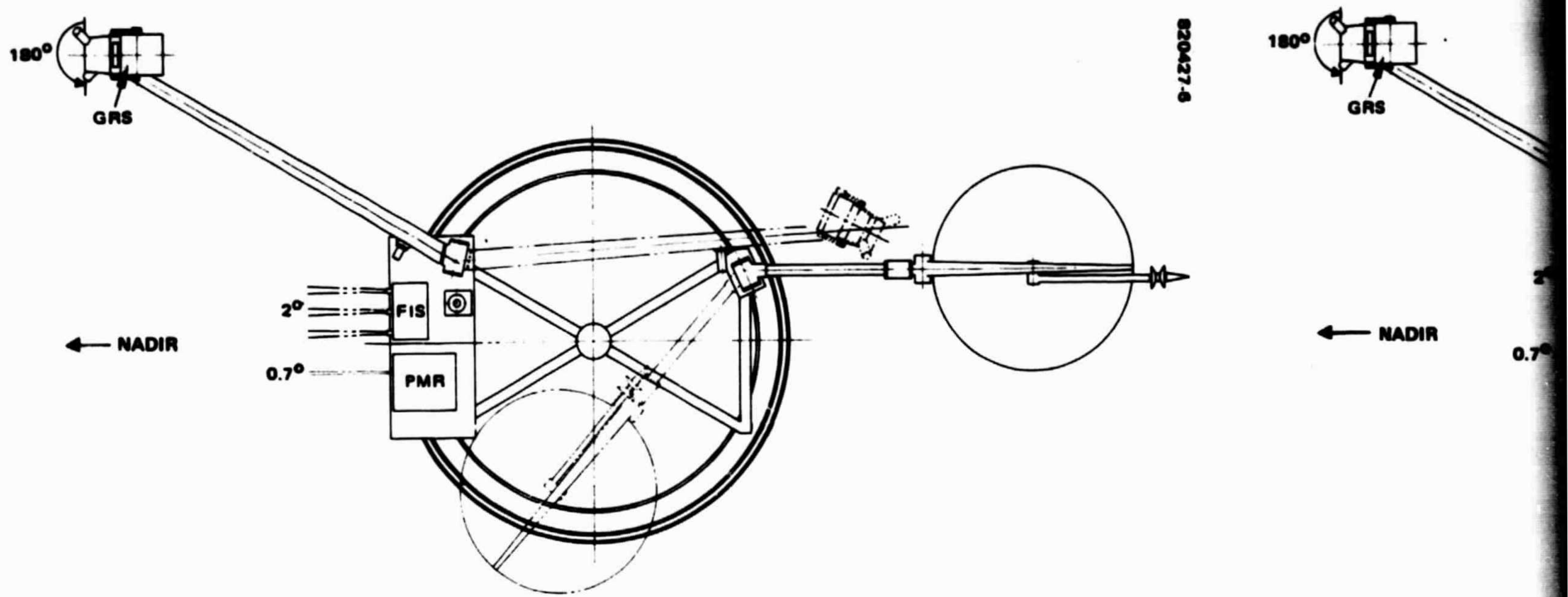
Figure 5-1 shows the science instrument layouts. Most instruments mount on the despun science shelf which tracks nadir. The alternate ultraviolet hydrogen photometer (UVHP) requires three sensors to view the forward limb, nadir, and zenith directions. The zenith sensor mounts on the despun X-beam on the opposite side of the spacecraft from the science shelf. The gamma ray spectrometer (GRS) extends more than one body diameter away from the spacecraft on a deployed single-segment boom. The FPI's one meter long axis lies parallel to the spacecraft's spin axis and extends through the thermal barrier down to the equipment shelf. An internal mirror inclines the field of view 23° towards nadir to view the limb as desired.

With the spin axis normal to the orbit plane, rotation of the despun platform once per orbit maintains nadir tracking. The spacecraft orientation shades the science shelf from the sun. One surface of each instrument never views the sun, fulfilling passive cooling requirements.

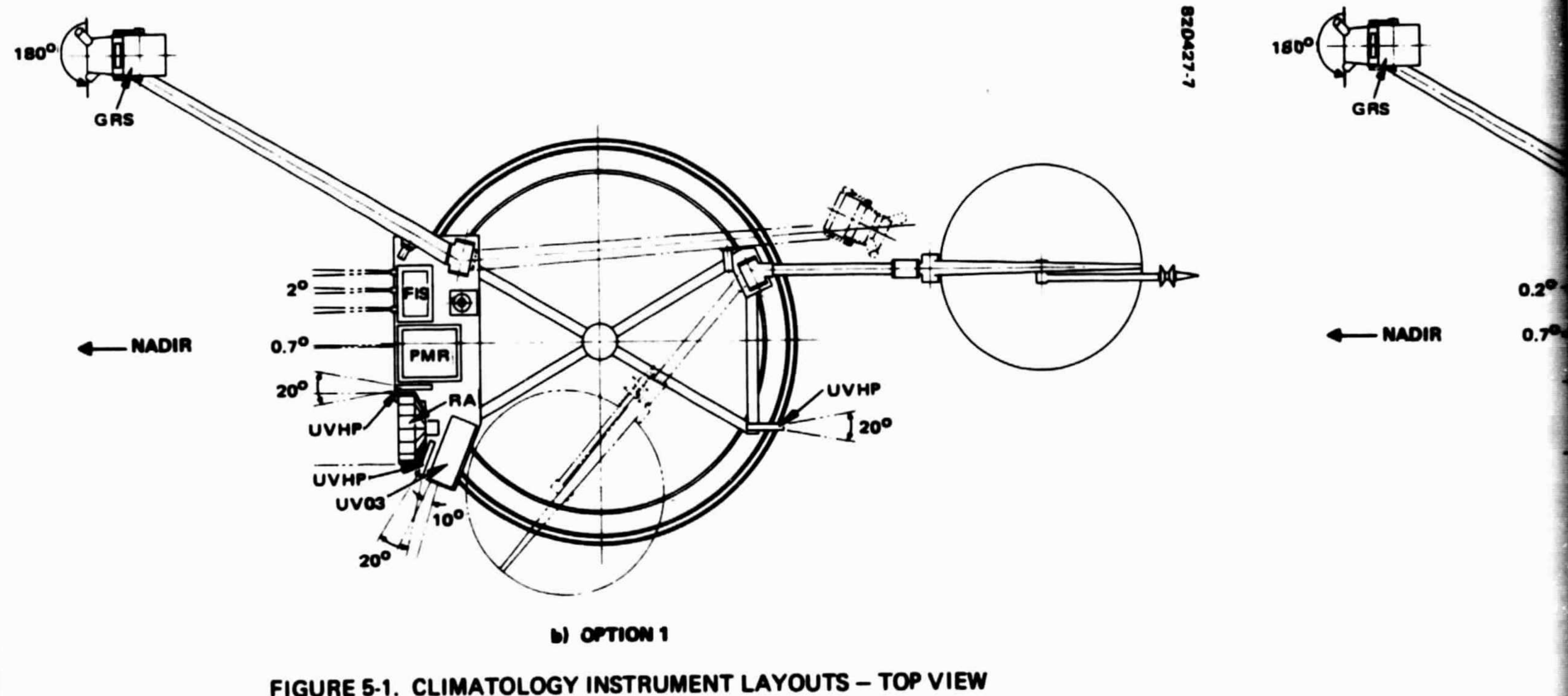
Aeronomy Payload

Table 5-3 lists the characteristics of the nine aeronomy instruments. Figure 5-2 shows the instrument layout. The five ram-oriented instruments (NMS/CWT, TIMS, RPA/DM, UVS and FPI) mount on a gimballed shelf which maintains ram alignment at periapsis. Three other despun instruments (EFD, ETP and MAG) attach directly to the X-beam support structure. The electron temperature probe and the magnetometer require three sensors to sample in orthogonal directions.

The three magnetometer sensors mount on the end of a 20-foot Astromast boom which is identical to the one flown on Dynamic Explorer. The solar wind plasma analyzer (SWPA) mounts on the spinning shelf (Figure 5-3). It samples the Mars environment through an opening in the solar array at the spacecraft spin rate of 55 rpm.



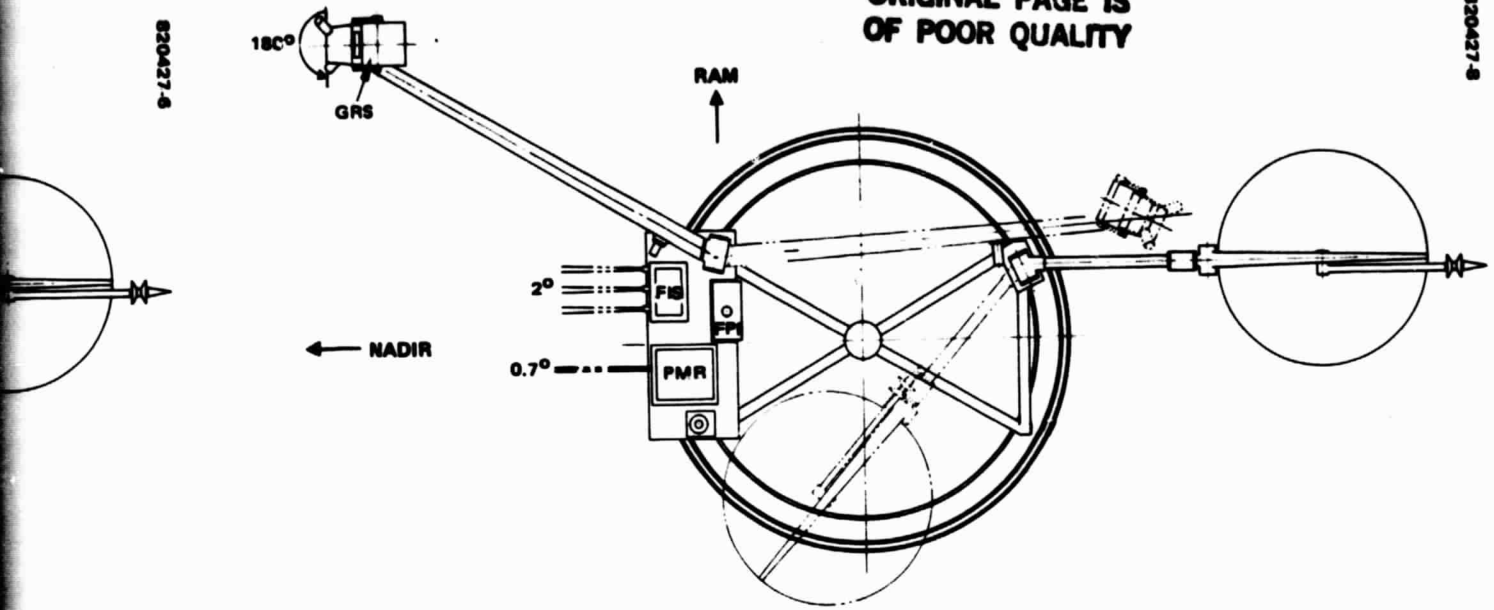
a) BASELINE



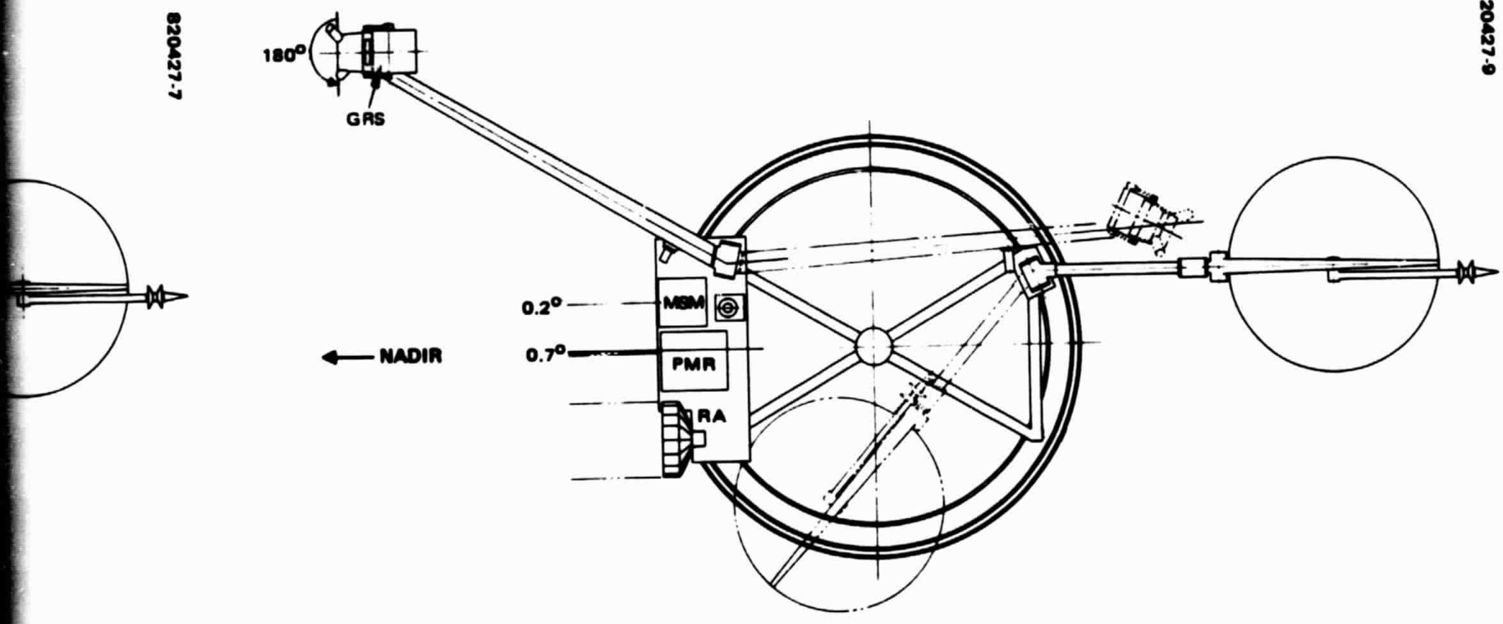
b) OPTION 1

FIGURE 5-1. CLIMATOLOGY INSTRUMENT LAYOUTS – TOP VIEW

FOLDOUT FRAME



c) OPTION 2



d) OPTION 3

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TABLE 5-3. AERONOMY INSTRUMENT CHARACTERISTICS

Instrument	Mass, kg	Power, W	Dimensions, cm	Data Rate, bps	Instrument Sensor Orientation
Neutral mass spectrometer					
Sensor	3.5	15	16 x 20 x 18	256	Ram
Electronics	5.5		16 x 16 x 8		
Thermal ion mass spectrometer	3	1.5	13 x 27 x 16	256	Ram
Electron temperature probe					
Sensors	3	3	40 x 0.5	256	3 orthogonal sensors; 1 sensor 11 spin axis
Electronics	3		6 x 6 x 6		
Retarding potential analyser	4.5	4	12 x 15 x 24	512	Ram
Magnetometer					
Sensors	0.8*	3	9 x 6 x 6	128	3 orthogonal sensors at end of boom
Electronics	2		11 x 22 x 15		
Electric field detector					
Sensors (2)	0.5	1	0.75 m/dipole	128	Dipole plane ⊥ spin axis
Electronics	1		7 x 19 x 8		
Solar wind plasma analyser	4	5	18 x 18 x 28	128	⊥ spin axis (spinning instrument)
Ultraviolet spectrometer	3	2	12 x 36 x 14	128	Forward limb
Fabrey-Perot interferometer					
Sensor	20	12	100 x 30 x 15	256	Side limb, 45° and 135° from ram
Electronics			25 x 20 x 25		
Total	53.8	46.5		2048	

*Boom not included

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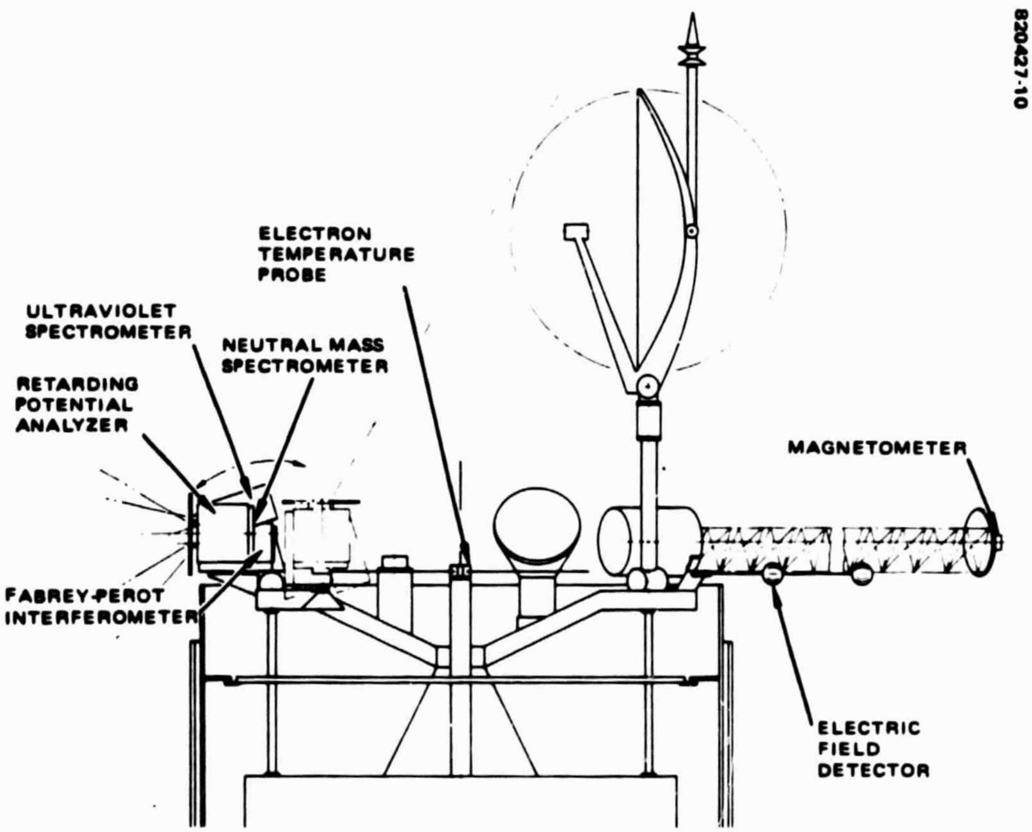

FIGURE 5-2. AERONOMY INSTRUMENT LAYOUT – SIDE VIEW

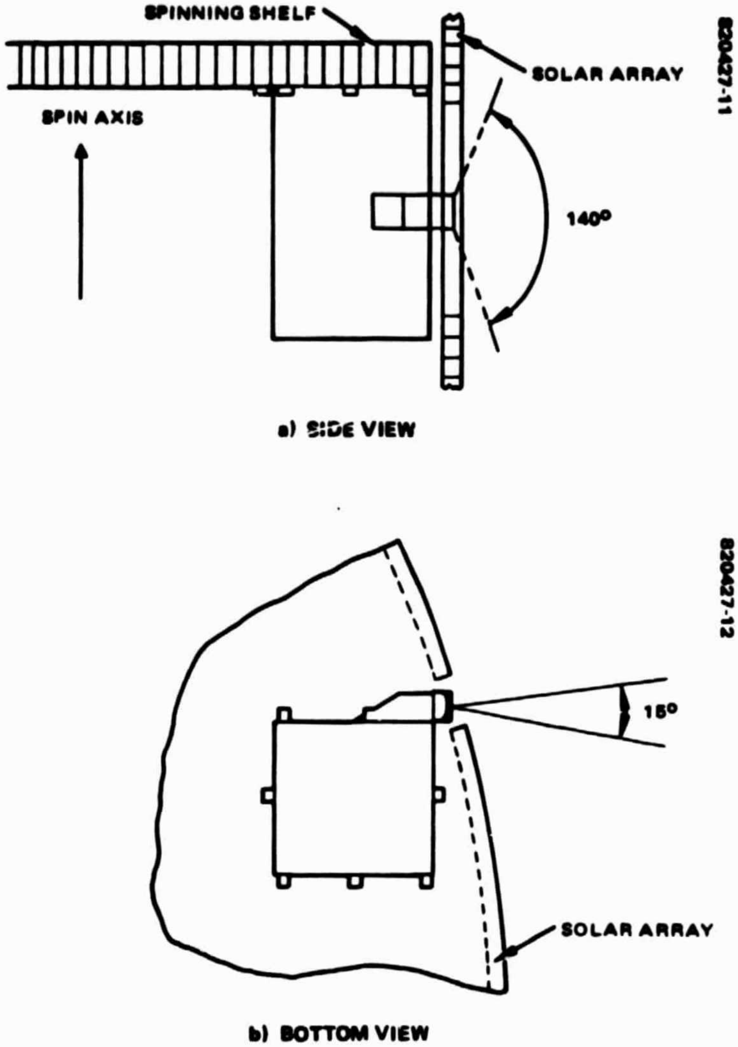

FIGURE 5-3. ARRANGEMENT OF SWPA ON SPINNING SHELF

5.2 Electrical Integration

Fully-redundant remote interface units mounted on the despun equipment shelf provide the necessary electrical connections. The instrument power interface unit (IPIU) distributes and regulates power from the despun bus to the instruments. The remote telemetry unit accepts status data from the instruments with serial data interfaces directly coupled to the central telemetry unit. The remote command unit distributes pulse and serial commands to the instruments.

The SWPA obtains power directly from the spun bus. Spinning remote telemetry and command units provide all other electrical interfaces for the SWPA.

The solar panel and batteries support continuous instrument sampling during sunlight and eclipse. The additional power of the alternate climatology payloads only requires a slightly longer solar panel; batteries and power electronics remain unchanged.

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5.3 Operating Sequence

All missions require continuous science sampling. The climatology instruments operate in a day/night sampling mode -- the FIS, UVO3, FPI and MSM do not collect data during the dark half of the orbit, as shown in Figure 5-4.

The aeronomy mission has three different sampling sequences. All of the instruments collect data at their maximum sampling rates in the periapsis region. During the outer atmosphere, or ionosheath, phase of the orbit, the NMS suspends sampling until the next periapsis phase, and the TIMS, ETP, and RPA/DM operate at reduced data rates. The apoapsis phase, which is most of the orbit, has the lowest total data rate. In this phase, the TIMS, UVS, and FPI suspend sampling, and the ETP and RPA/DM operate at further reduced rates. Figure 5-5 shows the aeronomy sampling sequence.

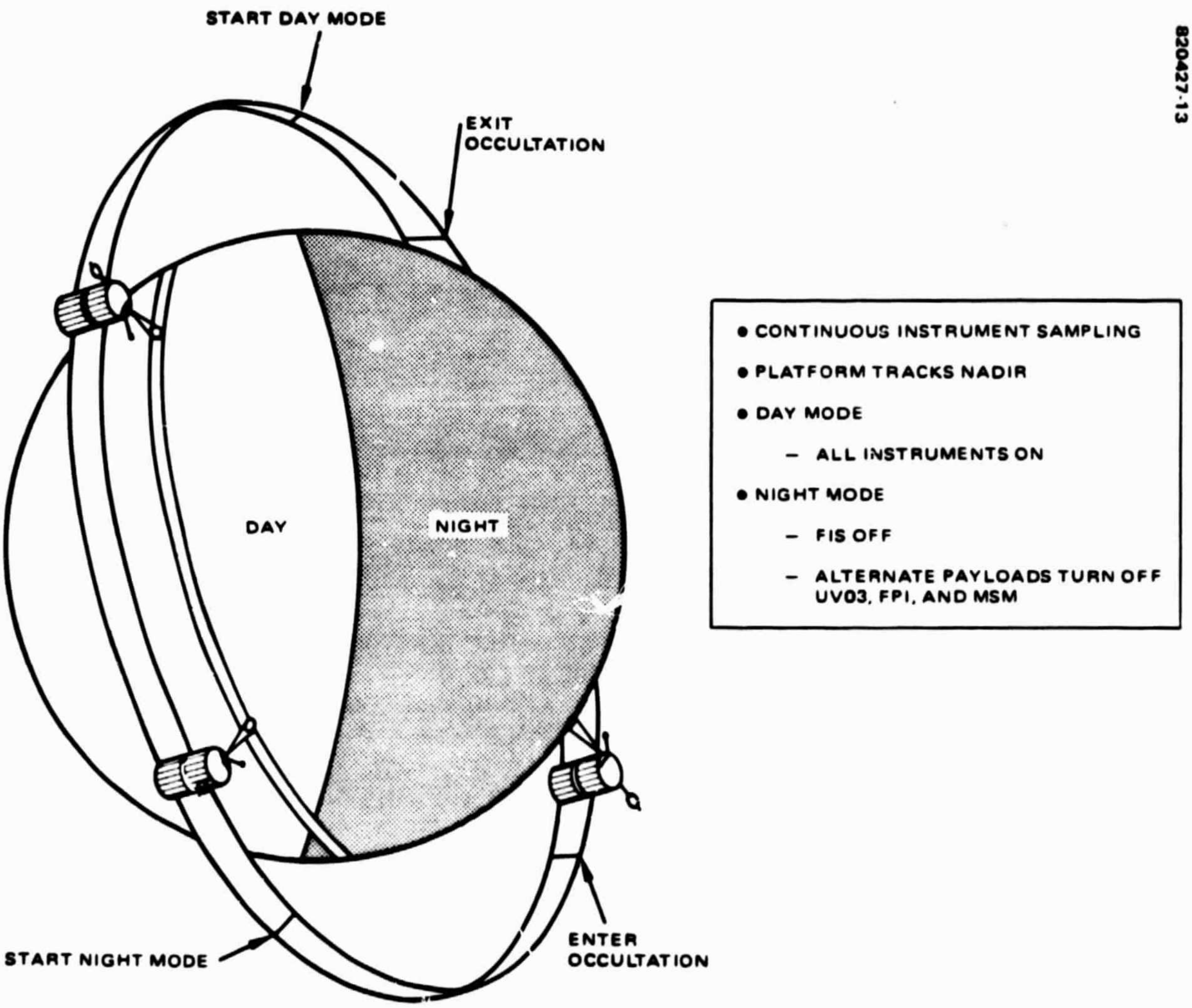

FIGURE 5-4. SCIENCE OPERATING MODES FOR CLIMATOLOGY MISSION

APOAPSIS MODE
- MAGNETOSPHERE SURVEY
- NMS, TIMS, UVS, AND FPI OFF
- RPA/DM ORIENTED FOR PERIAPSIS MODE

NEAR PERIAPSIS MODE
- IONOSHEATH SAMPLING
- NMS OFF; TIMS, ETP, RPA/DM, AND FPI REDUCED SAMPLING RATE
- SHELF INSTRUMENTS ORIENTED FOR PERIAPSIS MODE

PERIAPSIS MODE
- IONOSPHERE/UPPER ATMOSPHERE SAMPLING
- ALL INSTRUMENTS ON
- SHELF INSTRUMENTS ALIGNED WITH RAM AT PERIAPSIS

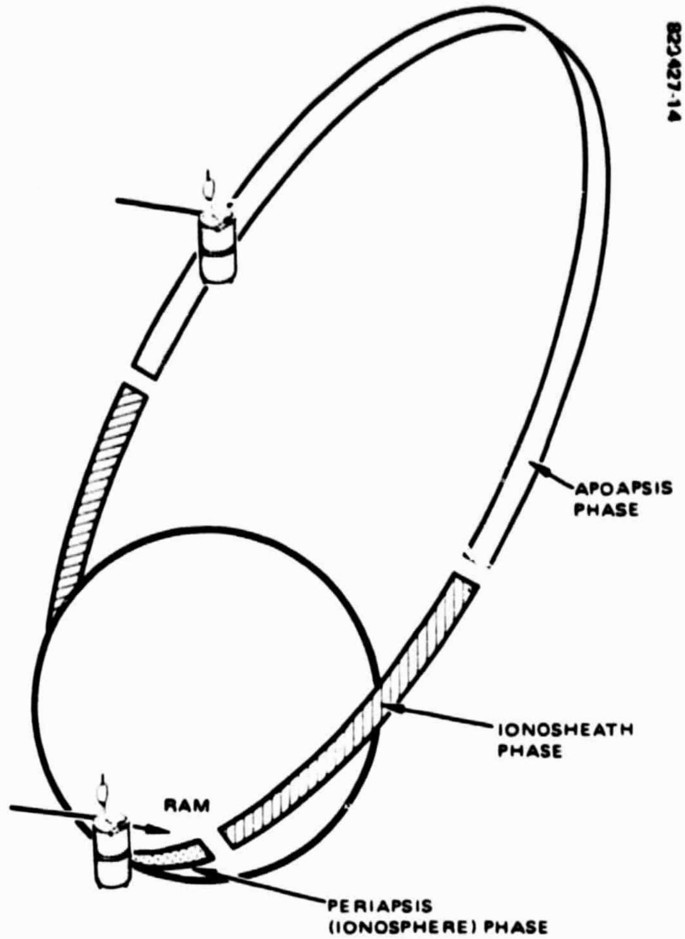

FIGURE 5-5. SCIENCE OPERATING MODES FOR AERONOMY MISSION

Each mission uses two tape recorders to maintain continuous instrument sampling with a third recorder as a backup. One recorder stores data for 24 hours (or 32 hours if needed to recover data during a DSN station outage) and then plays the data back during the subsequent DSN pass while the second recorder begins storing data for the next 24 hour period. Although the DSN is available for 8 hours a day, occultations, command load verification, and acquisitions shorten the availability to a minimum of 4.2 hours for the climatology mission and 5.6 hours for the aeronomy mission. These periods of DSN availability are long enough to return the stored data for the baseline missions and climatology payload alternates #1 and #2. Climatology alternate #3 requires a longer playback period and additional tape recorder capacity.

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6. SPACECRAFT DESIGN

The Mars Orbiter climatology and aeronomy spacecraft shown in Figure 6-1 both use the versatile HS 376 satellite design. The five flying and 22 ordered HS 376 satellites share a dual spin, gyrostat-stabilized configuration with telescoping solar panels. A bearing and power transfer assembly (BAPTA) connects the two halves of the spacecraft and transfers signals and power across the rotating interface on 26 slip rings. Delta, Shuttle, or Ariane boosters can launch the HS 376.

Table 6-1 highlights the features of the Mars Orbiter spacecraft. Subsystem descriptions follow this section. As shown in Figure 6-2, the despun section holds the science instruments, communications subsystem, data handling subsystem, and command subsystem, permitting direct electrical connections between all equipment necessary to gather, process, and transmit instrument data. This decouples the Mars science payload and the changed data handling, command, and communications subsystems from the remaining, generally unchanged, subsystems on the spinning section. Only the aeronomy mission SWPA instrument mounts on the spinning side of the spacecraft. Other changes to the spinning section of the HS 376 bus include a new solar cell layout optimized for conditions at Mars and a STAR-31 orbit insertion motor for the climatology orbiter. The Intelsat VI attitude control electronics unit replaces the HS 376 counterpart to allow pointing the despun platform relative to the sun.

Both orbiters carry identical, fully-redundant data handling systems with three 148.6 Mbit tape recorders. The microprocessor-based command subsystem, also mounted on the despun shelf, controls autonomous spacecraft functions such as recovery from lost uplink; solar panel deployment, battery charge monitoring, and overload protection, and supplies 20 serial and 436 pulse redundant commands to the spacecraft. The DSN-compatible 20 W communications subsystem uses GOES S-band transmitters, NASA standard transponders, and existing X-band TWTAs. Omni, bicone, and high gain antennas and the star sensor(s) complete the despun platform equipment.

The climatology spacecraft uses its despun platform to point the science instruments at Mars nadir. This causes the platform to rotate once per orbit in inertial space; a simple two-axis mechanism enables the HGA to track the earth. The spin axis is normal to the sun-synchronous orbit plane, resulting in solar incidence angles from 45° to 67.5°. The circular orbit permits use of sun sensor, horizon sensor and rate hold (backup) despin control modes.

The EFD, EPI, and boom-mounted magnetometer aeronomy science instruments attach to the despun platform directly. The NMS, TIMS, RPA/DM, UVS and FPI mount on a ram-oriented science shelf. The SWPA mounts on the 55 rpm spinning equipment shelf.

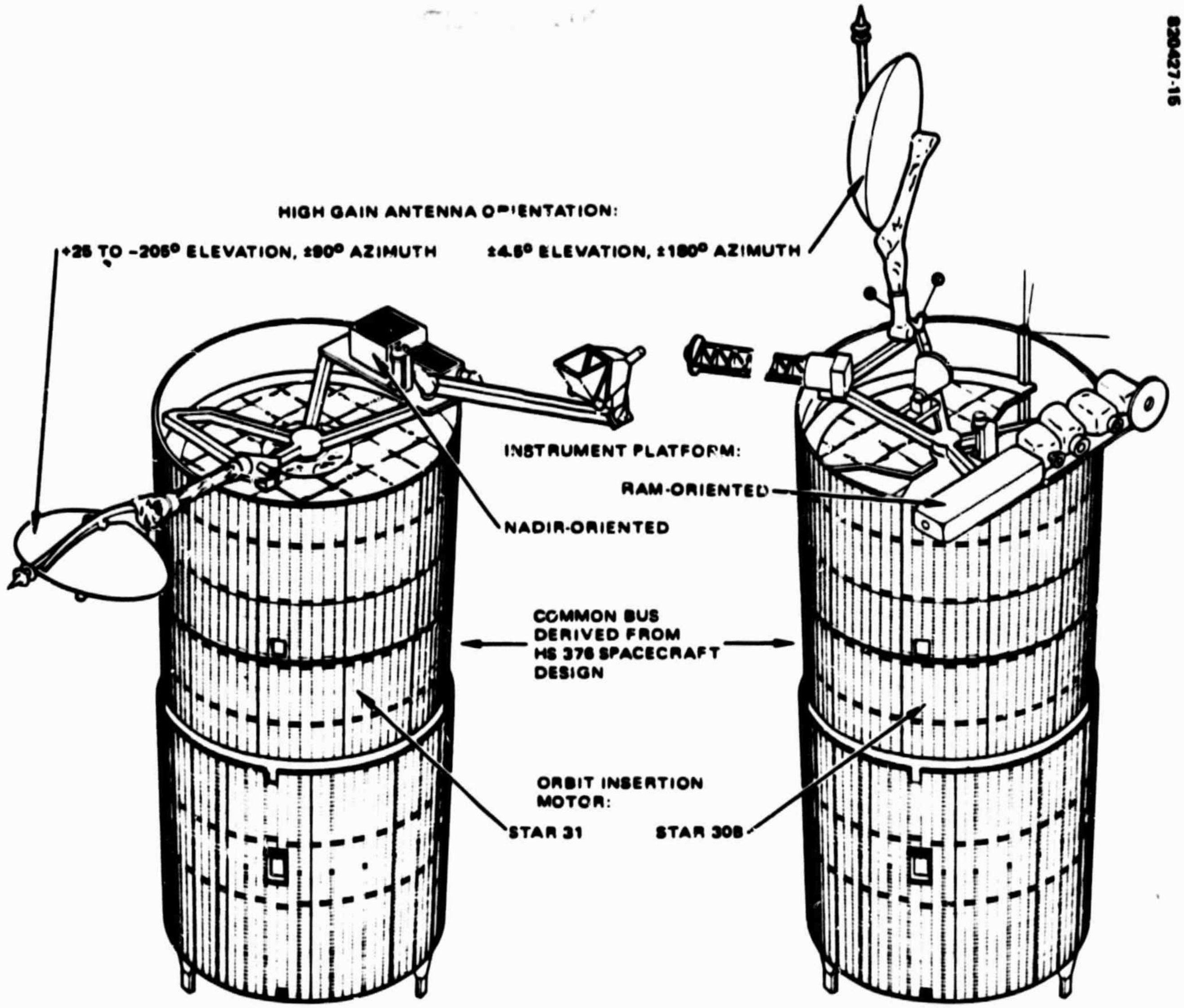

FIGURE 6-1. COMPARISON OF CONCEPTUAL CLIMATOLOGY AND AERONOMY ORBITERS – DEPLOYED

Elevation of the ram-oriented shelf and inertial azimuth rotation of the despun platform compensate for the orbit motion while correctly pointing the instruments and allowing the spin axis to remain normal to the ecliptic plane. Inverting the spacecraft after 180° of apsidal precession limits the required elevation travel of the ram-oriented shelf to 90°. The despun platform only rotates an average of 10° per month in inertial space to track the ram. This low rate simplifies pointing the HGA which must compensate for this rotation; the azimuth gimbal requires ± 180° motion; the elevation travel is less than ± 4-1/2°.

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TABLE 6-1. SPACECRAFT SYSTEM CHARACTERISTICS

Feature	Climatology Mission	Aeronomy Mission
Mission		
Orbit	300 km altitude, circular, sun-synchronous (1:30 to 3:00 pm)	150 km periapsis, 5.67 hr period
Attitude	Spin axis perpendicular to orbit plane	Spin axis perpendicular to Mars orbit plane
Orbit insertion ΔV	2284 m/sec	1305 m/sec
Launch/injection vehicle	Shuttle/I[illegible]	Shuttle/IPS
Science Accommodation		
Number of instruments	3	9
Science platform	Despun	Same (SWPA spinning)
Instrument orientation	Planet (nadir/limb viewing)	Ram pointing
Deployments	GRS boom	Magnetometer boom
Science telemetry data rate	8032 bps	4016 bps
Data handling/operations	All data stored; 4.2 hr playback in 32 hr maximum	Same, 5.6 hr playback
Thermal control	Passive	Same
Spacecraft		
Type	HS 376	Same
Physical characteristics		
Diameter, launch/on-orbit	2.5 m/6.7 m	2.3 m/8.2 m
Height, launch/on-orbit	2.8 m/4.9 m	2.4 m/6.7 m
Dry mass (including contingency)	614 kg	571 kg
Stabilization		
Spacecraft/injection stage (coast)	Spinning, active nutation control (ANC)	Same
Interplanetary cruise	Gyrostat with active nutation damping	Stable spinner
Mars orbit	Gyrostat with active nutation damping	Same
Life		
HS 376 spacecraft bus	7 to 10 yr	Same
Consumables	2 Mars yr on-orbit plus reserve	Same
Design reliability	Fully redundant	Same
Spacecraft Subsystems		
Structure		
Design	Dual load path: monocoque central structure with strut stabilization in outer load path	Same
Science instrument mounting	Separate despun science platform	Same with ram-oriented shelf
Launch vehicle loads	Qualified for STS/Delta/Ariane	Same
Communications		
Power, X band/S band	20 W/20 W	Same
Antennas	Despun high gain with dual gimbal; omni; bicone - all stowed at launch	Same
Data rate capability, worst case	8192 bps	Same
Data Handling		
Storage	148.6 Mbits	Same
Rate	8 to 8192 bps	8 to 4096 bps

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Table 6-1 (continued)

Feature	Climatology Mission	Aeronomy Mission
Serial data channels		
Spun	4 redundant	Same
Despun	20 redundant	Same
Analog/bilevel data channels		
Spun	252 redundant	Same
Despun	252 redundant	Same
Command		
Serial commands		
Spun	8 redundant	Same
Despun	12 redundant	Same
Pulse commands		
Spun	256 redundant	Same
Despun	180 redundant	Same
Attitude Control		
Sensors	Star, sun, horizon sensors	Same plus spinning star sensor
Despin control modes	Rate hold, Mars horizon sensor, sun sensor	Same
Pointing Accuracy	0.029° in sun mode; 0.061° in horizon mode	Same
Nutation damping	Thrusters or despun platform torquing	Same
BAPTA	HS 376, redundant, 26 slip rings	Same
Other mechanisms	HS 376, Pioneer Venus	Same
Power		
System	Dual regulated busses, overload protected	Same
Solar cells	High-efficiency K-7	Same
Main array power, worst case	387 W	467 W
Batteries	2 Westar Ni-Cd	Same
Capacity	19.5 A-hr each	Same
Depth of discharge	9.5%	0 to 15.2%
Propulsion		
MOI motor (case retained)	STAR-31	STAR-30B
System construction	Welded titanium	Same
Liquid propellant	232 bipropellant (usable)	Same
Thrusters	2 axial, 4 lateral	Same
Thermal Control	Passive; annular radiator, blankets and redundant heaters as needed	Same
Injection Stage		
Type	Integrated propulsion stage (IPS)	Same
Motor	CSD SRM-1 (7706 kg propellant)	Same (6128 kg propellant)
Deployment	Frisbee, 2 rpm	Same
Spin motors	2 STAR-6	Same

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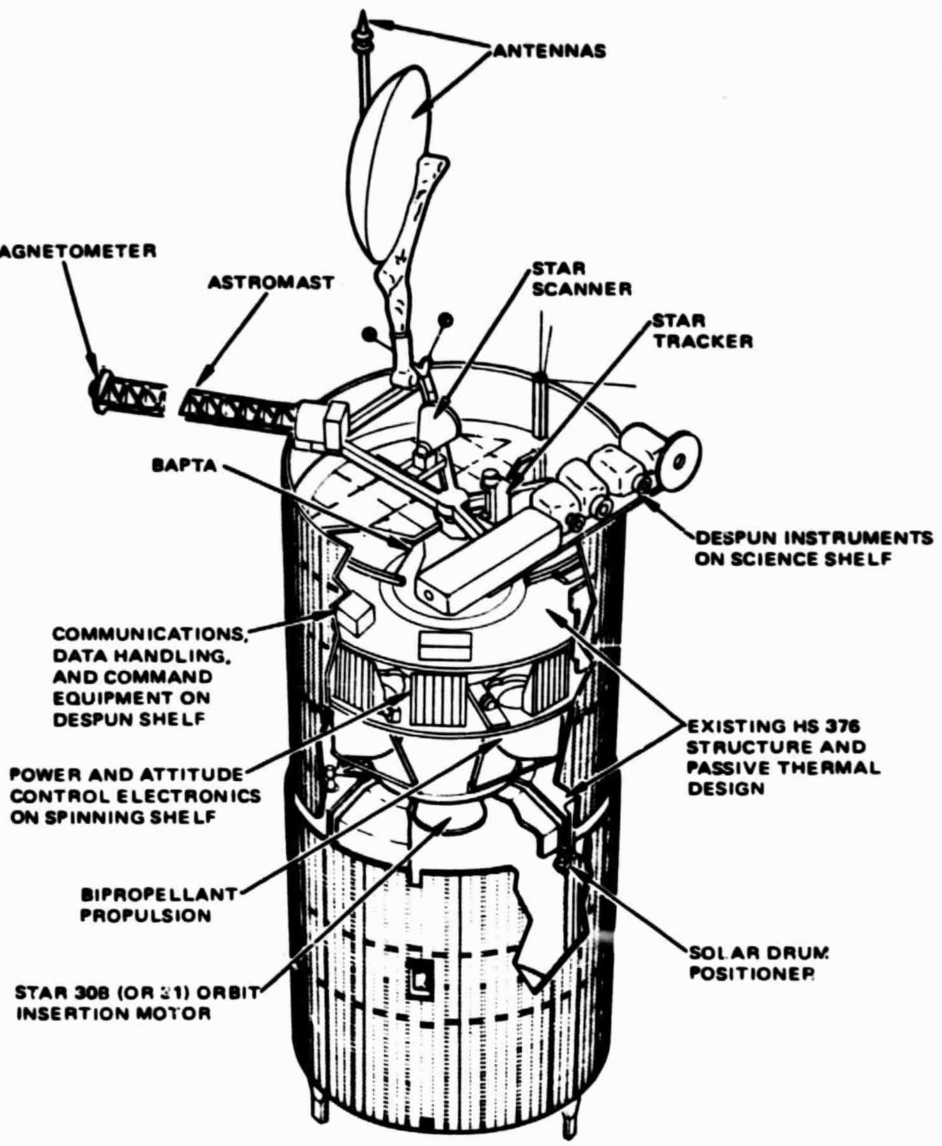

FIGURE 6-2. MARS AERONOMY ORBITER CUTAWAY

An offloaded integrated propulsion stage (IPS), attached to the spacecraft with a conical adapter, injects the spacecraft toward Mars. The spacecraft attached to the IPS motor deploys from the Shuttle using the frisbee technique patented by Hughes for Leasat, SAL, and Intelsat VI. Neither Mars Orbiter spacecraft has a direct mechanical interface with the Shuttle.

Mass Summary

Tables 6-2 and 6-3 summarize the mass of both spacecraft. The climatology orbiter mass is 614 kg with contingency, so the design mass of 650 kg includes 36 kg of reserve. The aeronomy orbiter, due to its lighter STAR-30B MOI motor case,

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TABLE 6-2. CLIMATOLOGY ORBITER MASS SUMMARY, KG

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Subsystem	Total	HS 376	Existing	In Development/ Modified	New
Structure	147.8	93.4	0.3	10.9	43.2*
Harness	25.2	3.1	0.0	13.2	8.9*
Communications	27.0	0.0	12.7	10.3	4.0*
Data Handling	41.6	0.0	6.3	35.4	0.0
Command	40.5	0.0	3.6	34.4	2.5
Attitude control	43.4	12.0	16.1	11.8	3.5
Power	93.3	91.3	2.1	0.0	0.0
Propulsion	22.6	0.8	4.2	17.6	0.0
Thermal control	20.3	11.5	0.0	3.7	5.0*
Balance mass	4.4	4.4	0.0	0.0	0.0
Solid motor case	73.6	0.0	73.6	0.0	0.0
Subtotal	539.9	216.5	118.9	137.3	67.1
Contingency	31.7	2.2 (1%)	2.4 (2%)	13.7 (10%)	13.4 (20%)
Bus Total	571.5				
Science	37.0				
Science contingency (15%)	5.5				
Spacecraft Total	614.1				
Mass reserve	35.9				
Design mass	650.0				

*No electronic parts

TABLE 6-3. AERONOMY ORBITER MASS SUMMARY, KG

820427-34T

Subsystem	Total	HS 376	Existing	In Development/ Modified	New
Structure	129.5	95.3	7.8	8.8	17.6*
Harness	26.3	3.1	0.0	13.2	10.0*
Communications	27.0	0.0	12.7	10.3	4.0*
Data handling	41.6	0.0	6.3	35.4	0.0
Command	40.5	0.0	3.6	34.4	2.5
Attitude control	49.9	15.8	18.8	11.8	3.5
Power	93.3	91.3	2.1	0.0	0.0
Propulsion	22.6	0.8	4.2	17.6	0.0
Thermal control	18.8	12.8	0.0	0.0	6.0*
Balance mass	4.4	4.4	0.0	0.0	0.0
Solid motor case	29.4	29.4	0.0	0.0	0.0
Total	483.5	252.9	55.6	131.5	43.6
Contingency	25.5	2.5 (1%)	1.1 (2%)	13.2 (10%)	8.7 (20%)
Bus Total	509.0				
Science	53.8				
Science contingency (15%)	8.1				
Spacecraft Total	570.8				
Mass reserve	29.2				
Design mass	600.0				

*No electronic parts

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TABLE 6-4. POWER BUDGET, W

Subsystem	Climatology		Aeronomy	
	Daylight	Eclipse	Daylight	Eclipse
Communications	76.1	25.2	76.1	25.2
Data handling	45.5	25.5	45.5	25.5
Command	15.0	15.0	15.0	15.0
Attitude Control	33.0	33.0	33.0	33.0
Power	1.9	3.3	1.9	3.3
Thermal control	10.0	20.0	10.0	20.0
Total	181.5	122.0	181.5	122.0
Contingency, 10%	18.1	12.2	18.1	12.2
Bus Total	199.6	134.2	199.6	134.2
Science	59.0	55.0	46.5*	21.3*
Battery charge (main array)	101.0	–	37.7	–
Spacecraft Total	360.6	184.5	283.8	155.5
Minimum main array output at 28 V	387.2	–	467.0	–
Power reserve/degradation allowance	27.6	–	183.2	–
Battery DOD	–	9.5%	–	0 to 15.2%

*Maximum power with periapsis in daylight, 90 min eclipse

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has a mass of only 571 kg including contingency. The 600 kg design mass leaves a reserve of 29 kg. The high percentage of existing components (72% by mass) provides unusually high mass confidence at this stage of the design. Contingencies of 1% and 2% allocated to HS 376 and other existing units, respectively, reflect manufacturing tolerances. Ten percent contingency is assigned to units under development or modified for this application; new units have a 20% contingency. The budget allocates 15% contingency for science instruments.

Power Summary

Table 6-4 totals the subsystem and instrument worst-case power requirements. The climatology power reserve (above 10% contingency) ranges from 27 to 240 watts depending on the sun angle and Mars-to-sun distance. The normal sun angle for the aeronomy orbiter results in higher output power. Both orbiters use about 16% of the panel area for a battery boost charge array; this panel area is not included in the budget. The climatology orbiter can begin full science operation after 20 days of drift orbit. Before then, instrument duty cycles must be restricted.

6.1 Structure/Harness Subsystem

The Mars Orbiter retains the basic HS 376 structure. The climatology motor mount, structural components for science instrument and antenna support, and deployment mechanisms are the only additions to the existing structure. The structural design gives a compact launch configuration and minimizes mass by using a dual load path. Figure 6-3 illustrates the principal structural elements.

The dual load path structure efficiently carries launch, injection and orbit insertion loads. The inner path consists of a monocoque thrust cone, the inner portion of the spinning shelf, and the BAPTA. Launch locks permit the eight tubular struts below the spinning shelf to carry the despun equipment loads through a second, outer load path.

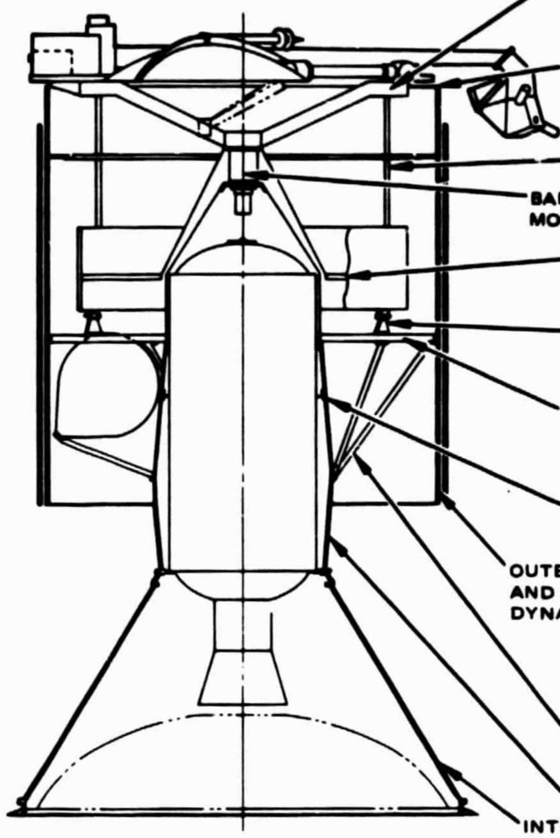

FIGURE 6-3. STRUCTURE SUBSYSTEM FEATURES

The HS 376 structure is qualified for horizontal configuration STS launch loads since it is already qualified for the more severe Delta launch vehicle loads. The 8g peak acceleration during injection on the integrated propulsion stage and the 10g climatology MOI load both are within the 12g design level of the HS 376 structure.

The climatology spacecraft includes an aluminum, semi-monocoque motor support which attaches the aft end of the long STAR-31 motor to the usual HS 376/PAM-D interface ring. This configuration takes the orbit insertion loads through structure designed for the higher load of the PAM-D perigee stage. At injection, the IPS adapter carries the STAR-31 mass directly. The aeronomy spacecraft is already qualified for the STAR-30B orbit insertion motor loads because it is identical to the HS 376 apogee kick motor.

The spun structure is unchanged from the HS 376, except the added STAR-31 climatology MOI motor support. The solar array substrates consist of aluminum honeycomb core with graphite and kevlar/epoxy face sheets. The spinning shelf, a honeycomb sandwich platform, supports all spinning electronic components and the battery packs. The four 21 in. conispherical tanks cluster around the central thrust tube below the spinning shelf. The propulsion support hardware includes a large bracket at the upper mount and tubular struts at the lower mount of the tanks.

The despun structure consists of a monocoque thrust cone, annular equipment shelves and a composite X-beam which supports the high gain antenna and science instruments. A graphite honeycomb instrument shelf mounts on the X-beam, opposite the HGA antenna support and provides a rigid science platform that resists thermal distortion. Crossmembers tie the ends of the X-beam, and bipods connect the beam assembly to the despun shelf. The beam load is distributed similarly for the normal HS 376 communications payload; the existing despun structure does not require any changes.

The Mars Orbiters deploy the high gain antenna mast, omni/bicone support and GRS boom using damped, spring-driven actuators and pyrotechnically-actuated pin pullers.

The HS 376 harness minimizes EMI while ensuring high reliability. The spacecraft structure provides a common ground plane. The existing HS 376 harness suffices for the spun section with minor changes reflecting the few unit substitutions. A revised portion of the despun signal and power harness accommodates the science instruments and communications and data handling equipment.

6.2 Propulsion Subsystem

The Mars Orbiters use a high performance liquid bipropellant propulsion subsystem which meets all the maneuver requirements of both missions. The subsystem configuration is under development for the SBS 1A HS 376 spacecraft. The aeronomy orbiter uses the HS 376 STAR-30B apogee kick motor for Mars orbit insertion. The higher energy requirements of the climatology mission require the STAR-31 motor for MOI.

As shown in Figure 6-4, the propulsion subsystem has two identical branches, each branch containing a fuel (MMH) and an oxidizer (N_2O_4) filter, four isolation latch valves, and two lateral and one axial thruster assembly. The isolation latch valves can disconnect any thruster from the pressurized fluid manifolds. The thruster arrangement is fully redundant; either branch of three thrusters can perform all mission operations. The axial thrusters are offset from the spin axis and canted 8° to minimize plume impingment on the aft drum. Canting the lateral thrusters provides radial and tangential thrust for ΔV and spin control maneuvers.

The 5000 cubic inch titanium tanks operate in a blowdown mode. The initial maximum operating pressure of 260 psia is less than half the 527 psia burst pressure, meeting the Shuttle safety requirement. The minimum tank pressure of 90 psia corresponds to a total blowdown ratio of approximately 2.9:1.

The six Marquardt R6C-1 5 lb_f thrusters have two nozzle configurations. The lateral thrusters use a 100:1 expansion one-piece columbium C-103 chamber and nozzle; the axial thruster extends the nozzle to an expansion ratio of 300:1. The R6C-1 can operate in steady state or pulse mode with a minimum impulse bit of 0.02 lb_f-sec. The steady-state I_{sp} at 260 psia pressure is 290 seconds. The INSAT and SAL programs have qualified the thrusters and demonstrated over 750 Kg throughput, which allows completing the mission with the loss of any thruster.

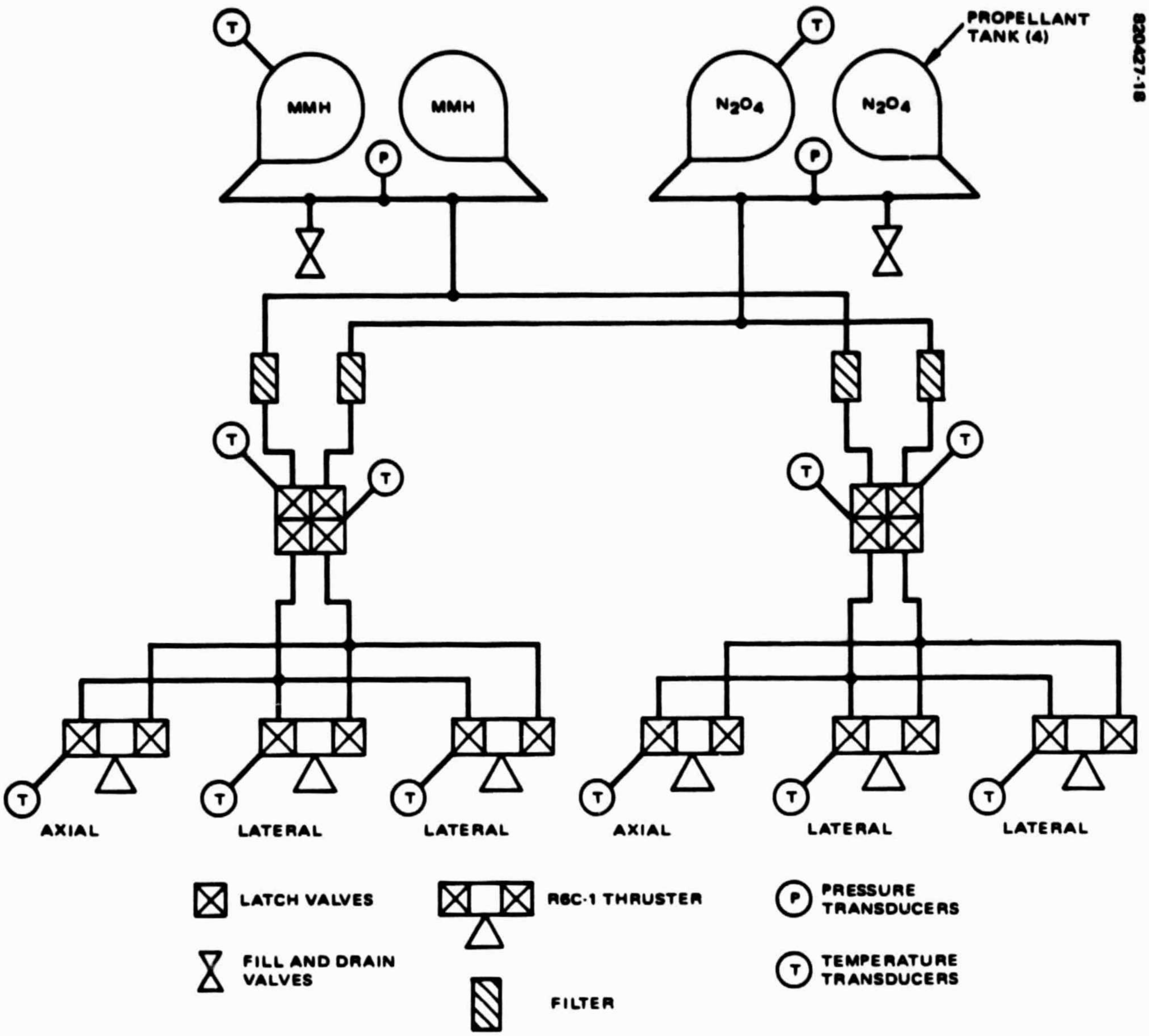

FIGURE 6-4. PROPULSION SUBSYSTEM CONFIGURATION

The subsystem design also includes redundant heaters, blankets, and low emittance tape wrap to avoid thermal operational constraints and maintain temperatures above the 10°C nitrogen tetroxide freezing point.

The Mars orbit insertion motors (STAR-31 for climatology, STAR-30B for aeronomy) are both manufactured by Thiokol using TP-H-3340 propellant. Thermal blankets maintain motor propellant temperatures between 40°F and 90°F and safe and arm devices accommodate Shuttle safety requirements. The STAR-31 motor is currently in production status as the third stage of the Scout launch vehicle. The Mars Orbiter application will require one sea level acceptance test to verify the performance of the 20% offload and 18" nozzle cut. The fully loaded STAR-30B is identical to all the motors purchased for the HS 376 product line and will share a common acceptance test.

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6.3 Communications Subsystem

The dual-frequency Mars Orbiter communications subsystem consists of antennas, NASA standard transponders and 20 W X- and S-band transmitters. Switching and use of parallel units results in a fully-redundant subsystem which meets the downlink data requirements for both missions. S- and X-band downlinks allow DSN flexibility and the option of dual-frequency radio science.

All of the communications equipment, identical for both missions, mounts on the despun side of the spacecraft. As shown in Figure 6-5, the NASA standard transponder receives the telemetry stream from the data handling subsystem. The transponder produces both S- and X-band signals. The S-band signal runs through one of two parallel S-band transmitters and then through a switch to the high gain or omni/bicone antennas. The TWTAs amplify the X-band signal and pass it through the high gain antenna for a second downlink.

The omni/bicone receives the comand uplink signal and sends it to the receivers within the transponder. The command detector then processes the uplink signal and sends it to the command processor unit.

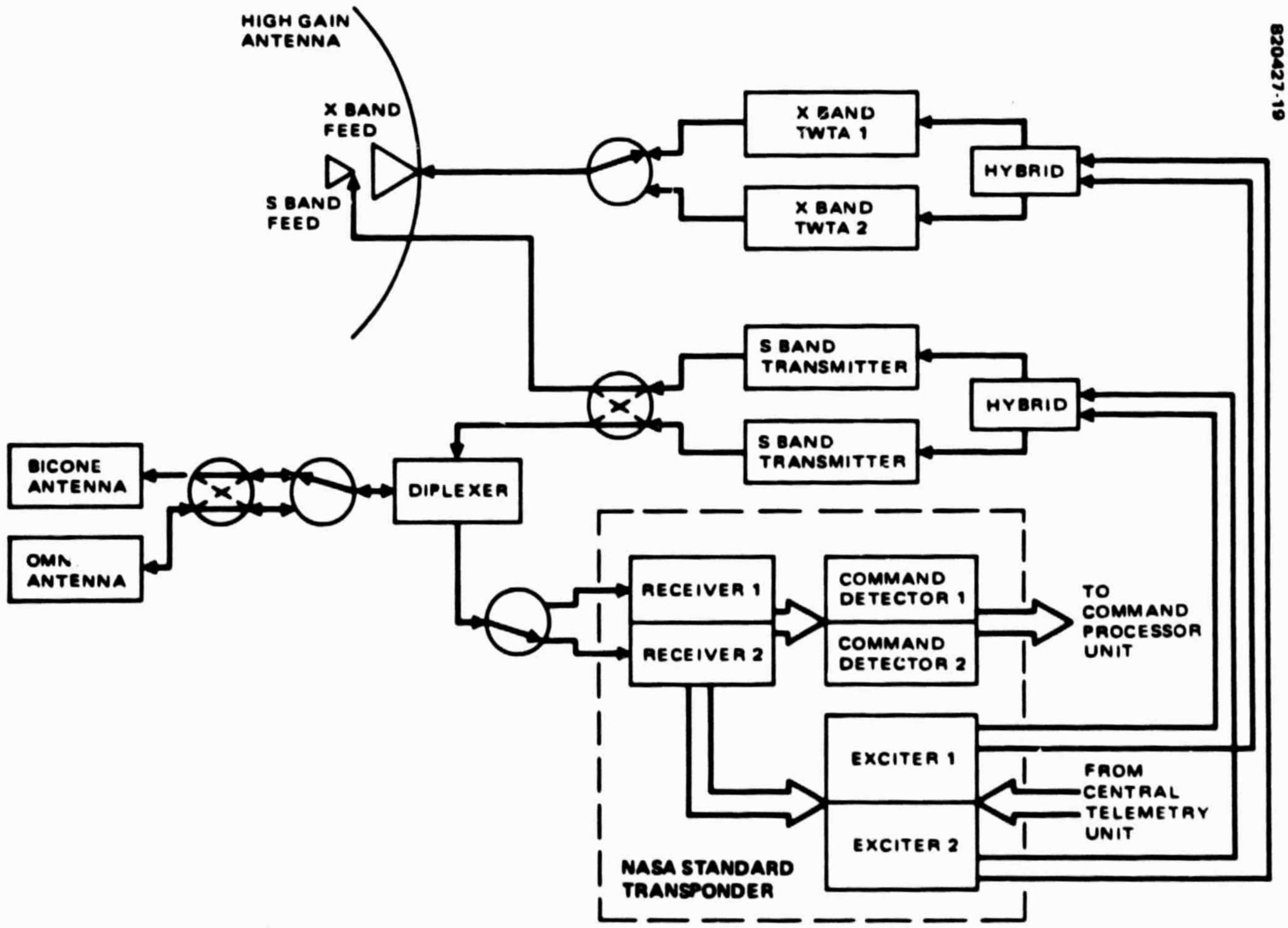

FIGURE 6-5. COMMUNICATIONS SUBSYSTEM FUNCTIONAL DIAGRAM

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TABLE 6-5. COMMUNICATIONS PERFORMANCE

High Gain Antenna (Telemetry, Maximum Range)		
Transmit frequency, MHz	2295	8415
Transmit power, W	20	20
Gain, dB	26.2	37.5
Beamwidth (-3 dB), deg	7.6	2.0
Downlink data rate, bps	2048	8192
Margin above adverse tolerance, dB	2.4	1.37
Bicone (Telemetry, Orbit Insertion)		
Transmit frequency, MHz	2295	
Transmit power, W	20	
Gain, dB	4	
Beamwidth (-6 dB), deg	30	
Downlink data rate, bps	16	
Margin above adverse tolerance, dB	2.0	
Bicone/Omni (Command, Maximum Range)		
Command frequency, MHz	2115	
Gain, dB	-2dB	
Command bit rate, bps	32	
Margin above adverse tolerance, dB	4.0	

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The Mars Orbiters use the 1.1 meter Galileo Probe relay antenna reflector. The feed, developed for the Pioneer Venus Orbiter (PVO), provides S- and X-band capability. The bicone antenna is modified from the Leasat design; the omni is identical to the PVO turnstile.

The GOES G and H 20 watt solid state S-band transmitters combine power and driver amplifiers. The X-band 20 watt traveling wave tube is identical to the DSCS III design manufactured by either Hughes or Watkins-Johnson. Hughes will integrate an electronic power conditioner with the TWT. The NASA standard transponder consists of a phase lock receiver, command detector unit, S-band exciter, and X-band exciter.

The Mars Orbiter communications subsystem supports all links with margin above adverse tolerance. Table 6-5 summarizes the link performance detailed in Volume 2. At maximum range, the X-band link can support 8192 bps with over 1 dB margin above adverse tolerance. The bicone antenna supports 16 bps S-band telemetry at the orbit insertion range of 1.48 AU. The S-band command uplink, based on 85 kw of transmit power, supports 32 bps with 4 dB margin above adverse tolerance.

6.4 Data Handling Subsystem

The data handling subsystem (DHS) contains two central and four remote units, flight-proven on the GOES spacecraft, and three tape recorders. The 148.6 Mbit tape recorders allow continuous storage of all instrument data.

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The DHS, shown in Figure 6-6, provides fully redundant data processing capability. A central telemetry unit (CTU) controls data collection through remote telemetry units (RTU) by supervisory bus instructions. Telemetry outputs include isolated redundant PCM streams to the two exciters and a separate uncoded and packetized stream with synchronous clock to the Shuttle signal interface unit (SIU). A 16-bit serial command configures the CTU format, bit rate, modulation index, and operating mode.

The CTU uses the GOES microcontroller and timing subassemblies with modified mission-unique circuitry and format generator firmware. The CTU produces a timing source to synchronize the command subsystem with the DHS and includes a multiplexer for 16 direct serial data channels.

The RTUs gather, format, and condition user telemetry and transmit the processed data to the CTU. The unit digitizes analog data with an accuracy of better than + 0.4%. Each of the four multiplexer modules within the RTU handles one serial and 63 analog or bilevel telemetry channels.

The Odetics DDS-3100 tape recorders have flown on the Earth Resources Budget Satellite (ERBS). Data transfer is a start/stop operation. A set of parallel 16 kbit buffers and switching controls connect the three tape recorders to the two CTUs. One buffer receives data from the CTU while a second empties data into the active recorder. The reproduce cycle reverses the buffering procedure. The second CTU also connects to the record and reproduce data buses, providing a fully-redundant cross-strapped architecture.

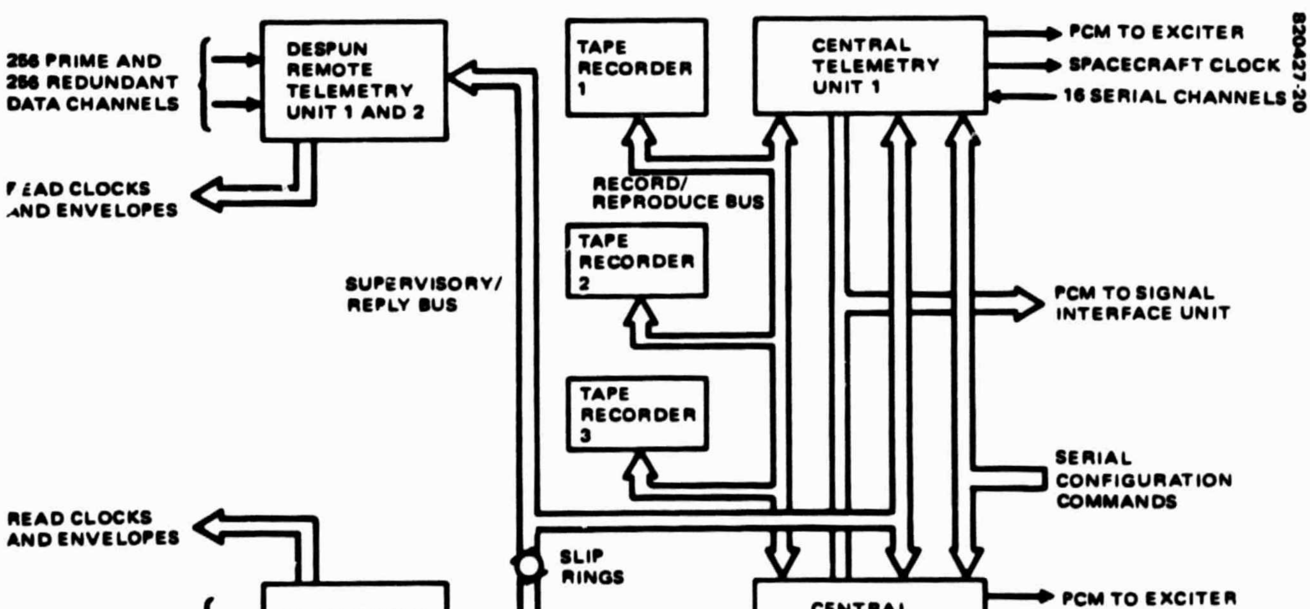

FIGURE 6-6. DATA HANDLING SUBSYSTEM FUNCTIONAL DIAGRAM

All the DHS components, except the spinning RTUs, mount on the despun platform near the instrument data sources and exciter/transmitter output equipment. The despun RTUs mount as close as possible to the instrument data sources to minimize the inaccuracy in digitized analog values.

The DHS supplies all serial, analog, and bilevel telemetry channel requirements of the Mars Orbiter subsystems and science instruments. The four serial and 252 redundant analog/bilevel channels of each redundant despun RTU, plus the 16 additional serial channels processed by each redundant CTU, easily meet the despun channel requirements. The RTU capability meets the spinning telemetry channel requirements.

6.5 Command Subsystem

The Mars Orbiter command subsystem, based on the Leasat design, uses a microprocessor-controlled central unit with distributed remote and driver units. The deep space mission requires added stored command logic. Autonomous control of time critical functions and recovery from lost uplink provide reliable spacecraft operation.

The command processor unit (CPU), located on the despun equipment shelf, contains a command demodulator which receives cross-strapped real-time commands from the transponder, (Figure 6-7). Modifying the Leasat CPU adds storage for 1024 32-bit commands. The subsystem executes these commands according to their time code, with the timing reference coming from the CTU-generated clock. CPU output buffers connect directly on a separate wire to each remote command unit. The remotes then distribute serial and pulse commands, including squib firing and stepper motor operation commands, through the driver units.

The Leasat remote units cannot be used because the main LSI component is no longer manufactured. The substitute spun remote unit from the classified HS 261 program resembles the Leasat design and supplies the negative command level required for the HS 376 spun subsystems. The new despun remote, based on existing components, provides the positive command level needed by the science instruments.

The squib, stepper motor, and valve driver units, from Intelsat VI, use Galileo Probe hybrid drivers to reliably generate current pulses. Parallel circuits provide redundancy of all functions. The number of outputs exceeds the needs of the Mars Orbiters, including operation of valves, thrusters, motors and pointing mechanisms.

The command subsystem autonomously handles time-critical functions and implements failure-recovery modes if the uplink is lost. The CPU monitors fault detection flags set by the CTU and overrides time critical operations, such as solar panel deployment and battery charging, if an anomaly occurs. In a similar manner, the subsystem also guards against bus overloads. If the uplink signal is lost, the CPU switches to the alternate transponder, and after an appropriate delay steps the high gain antenna. Ultimately, a sequence reorients the spacecraft to reacquire the link.

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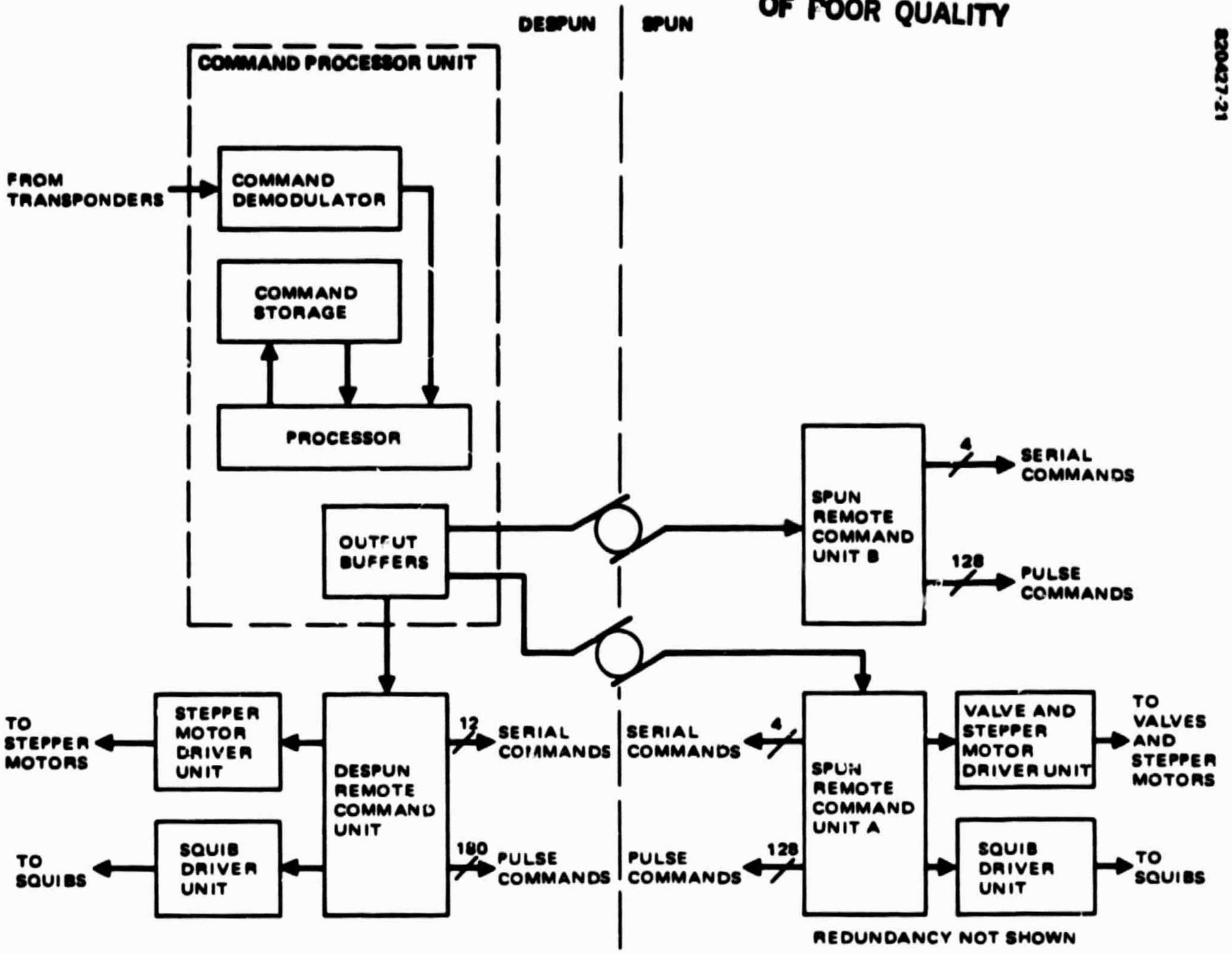

FIGURE 6-7. COMMAND SUBSYSTEM FUNCTIONAL DIAGRAM

6.6 Attitude Control Subsystem

The attitude control subsystem (ACS) provides velocity control, spin axis orientation and stabilization, spin rate control, and antenna and science instrument pointing. Active nutation control (ANC) using thrusters operates after ejection from the Shuttle while the injection stage is attached. After stage separation, the aeronomy spacecraft is spin stable until the platform is despun and the solar drum is extended in Mars orbit. The spacecraft then assumes a gyrostat configuration where despun platform torquing controls nutation, with ANC as a backup. The climatology orbiter also operates as a gyrostat during cruise.

Figure 6-8 shows the operation of the ACS. Ground control determines the orientation of the spin axis using telemetered sun, star, and Mars horizon sensor data. A pulsed axial thruster precesses the spacecraft attitude when required. Ground-commanded individual adjustment of the extended positions of the three racks supporting the deployed solar drum nulls any rotor imbalance. This reduces spin axis wobble to a maximum of 0.0005°.

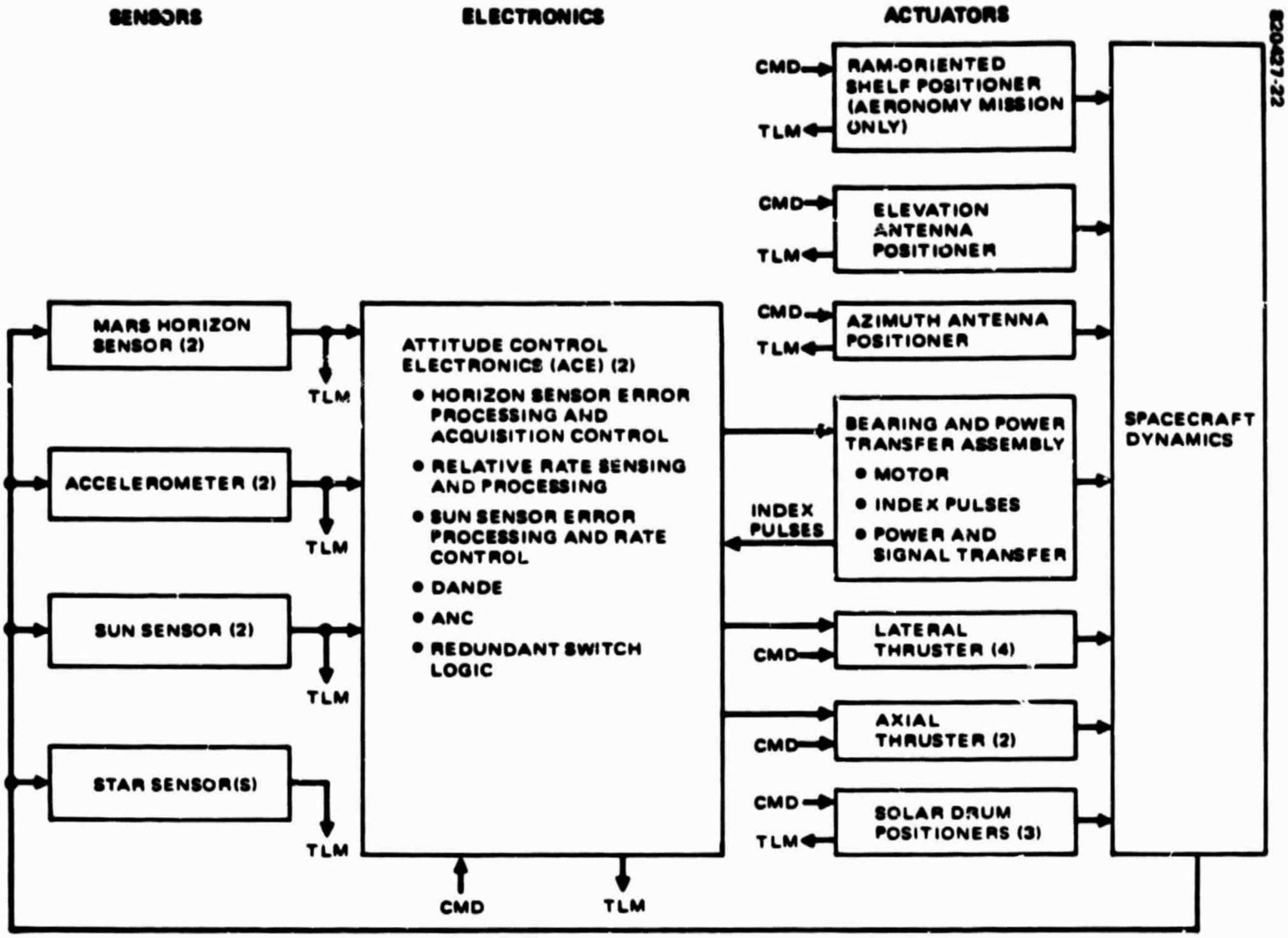

FIGURE 6-8. ATTITUDE CONTROL ELECTRONICS FUNCTIONAL DIAGRAM

Nutation Control

The spacecraft features two automatic modes of damping nutation. Both control loops use the same redundant pair of accelerometers to sense nutational motion. The active nutation control (ANC) loop fires thrusters to counter the nutation. The despin active nutation damping electronics (DANDE) controls nutation in all gyrostat modes. Adjusting the torque applied to the despin motor generates a component of transverse acceleration because of the coupling effect of despun platform product of inertia. DANDE control damps nutation to 0.002°.

The DANDE mode operates continuously in Mars orbit, with ANC available as high-level backup. ANC is enabled during platform despin, solar panel extension, and HGA and science instrument boom deployment. The ANC also limits the initial nutation of the spacecraft/IPS and maintains stability during the 45-minute coast phase after separation from the Shuttle and before injection motor firing. During interplanetary cruise the aeronomy spacecraft is passively stable with an inertia ratio greater than unity and requires no nutation control; DANDE controls nutation of the gyrostat-stabilized climatology orbiter during cruise.

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Despun Platform Control

The ACS points the despun platform using either horizon, sun, or relative rate mode. In horizon mode, data from the spinning Mars horizon sensors determine the pointing direction of the despun platform. For the climatology mission the platform tracks nadir as shown in Figure 6-9. The center of the detected Mars radiance pulse aligns with a master index pulse referenced to the instrument boresights. Ground commands can bias this pointing to reposition the instrument line of sight away from nadir if desired.

During the (gyrostat stabilized) climatology cruise and much of the elliptical polar aeronomy orbit, the spacecraft lacks a horizon reference. The Intelsat VI attitude control electronics (ACE) allow pointing the despun platform at any angle relative to the sun. The operation is similar to horizon mode.

For either mission, the spacecraft can use the relative rate mode whenever required. This mode controls the difference between platform and rotor spin rates. Since the average rotor rate is known from sun pulse data, the platform is despun to any commanded rate. The BAPTA includes a tachometer to provide this feature. An automatic toggle switches control to the redundant ACE unit if the relative rate exceeds acceptable limits. Because relative rate platform pointing uses no external reference, errors cause a slow drift.

Performance

Table 6-6 compares the ACS performance to the requirements. The electronics, sensors, and thrusters are fully redundant and extensively cross-strapped to ensure reliability. Manual override of all automatic control features (including nutation control) is possible. The command subsystem can update spin axis, HGA, and ram-oriented shelf pointing as required.

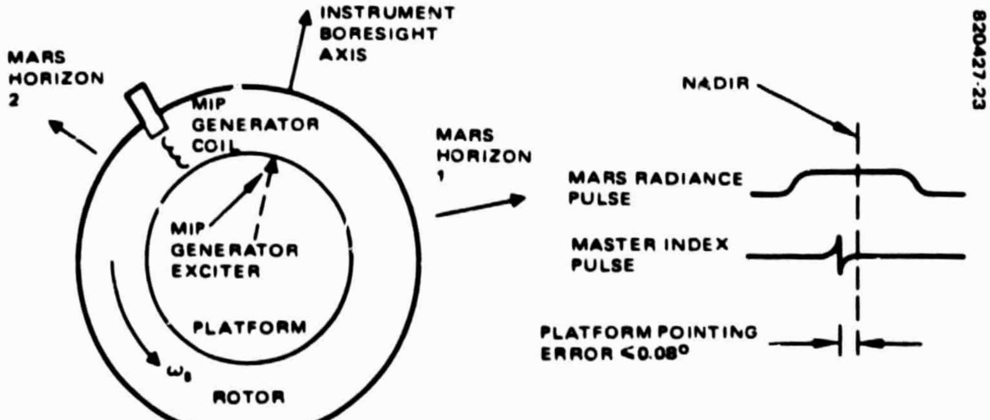

FIGURE 6-9. MARS HORIZON SENSOR REFERENCE FOR NADIR PLATFORM POINTING CONTROL

TABLE 6-6. ATTITUDE CONTROL SUBSYSTEM PERFORMANCE

Item	Requirement	Performance
Stability	Spin	Dual-spin (gyrostat); simple spinner during aeronomy cruise
Damping time constant	≤1 hr or as required	ANC – 3 sec DANDE – 21 sec, typical
Damping threshold	<0.1°	ANC, low gain – 0.15° ANC, high gain – 0.48° DANDE – 0.002°
Wobble	TBD	≤0.0005°
Spin axis pointing correction frequency	Ground-commanded maneuvers ≤1/wk	≤0.035°/week pointing uncertainty due to solar torque
Spin rate	TBD to 60 rpm	55 rpm nominal; 25 to 90 rpm possible; platform despun
Roll reference	≤1°	≤0.015°
Platform pointing control	1° climatology; 0.5° aeronomy, with reference	Horizon reference 0.029° Sun reference 0.061° Relative rate reference available
Spin axis pointing control	1° climatology; 0.5° aeronomy	Spacecraft can follow any spin axis pointing model using stored commands to <0.026°
Instrument pointing knowledge	0.2° climatology; 0.1° aeronomy	0.05°

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6.7 Power Subsystem

The spacecraft uses the HS 376 dual-bus power subsystem, shown in Figure 6-10. The extendible solar drum's main arrays provide 387 watts for the climatology orbiter and the 467 watts for the aeronomy orbiter, worst-case. Before its extension by three rack and pinion mechanisms, the aft (outer) solar panel drum generates all solar power. During eclipse two 19.5 A-hr nickel-cadmium batteries power the spacecraft.

The power subsystem features full redundancy. Two independent and balanced electrical busses distribute power to all units. The instrument power interface unit (IPIU) connects the science instruments to one of the busses. The Galileo Probe-derived IPIU switches power to the instruments and individually fuses each instrument. The IPIU also provides voltage regulation.

The solar array consists of two concentric cylindrical panels of K-7 high-efficiency solar cells. The revised cell layout compared to standard HS 376 layouts compensates for the decreased solar intensity at Mars. The cooler operating temperatures increase the cell output voltage, so each string is shortened. More parallel strings are available to compensate for the reduced solar intensity. The thermal radiator band around the forward drum, needed on HS 376 to radiate energy from the communications payload, is covered with solar cells in the Mars Orbiter designs. In sunlight, bus limiters automatically hold the bus voltage to 30 volts dc by shunting the lower voltage section of each array.

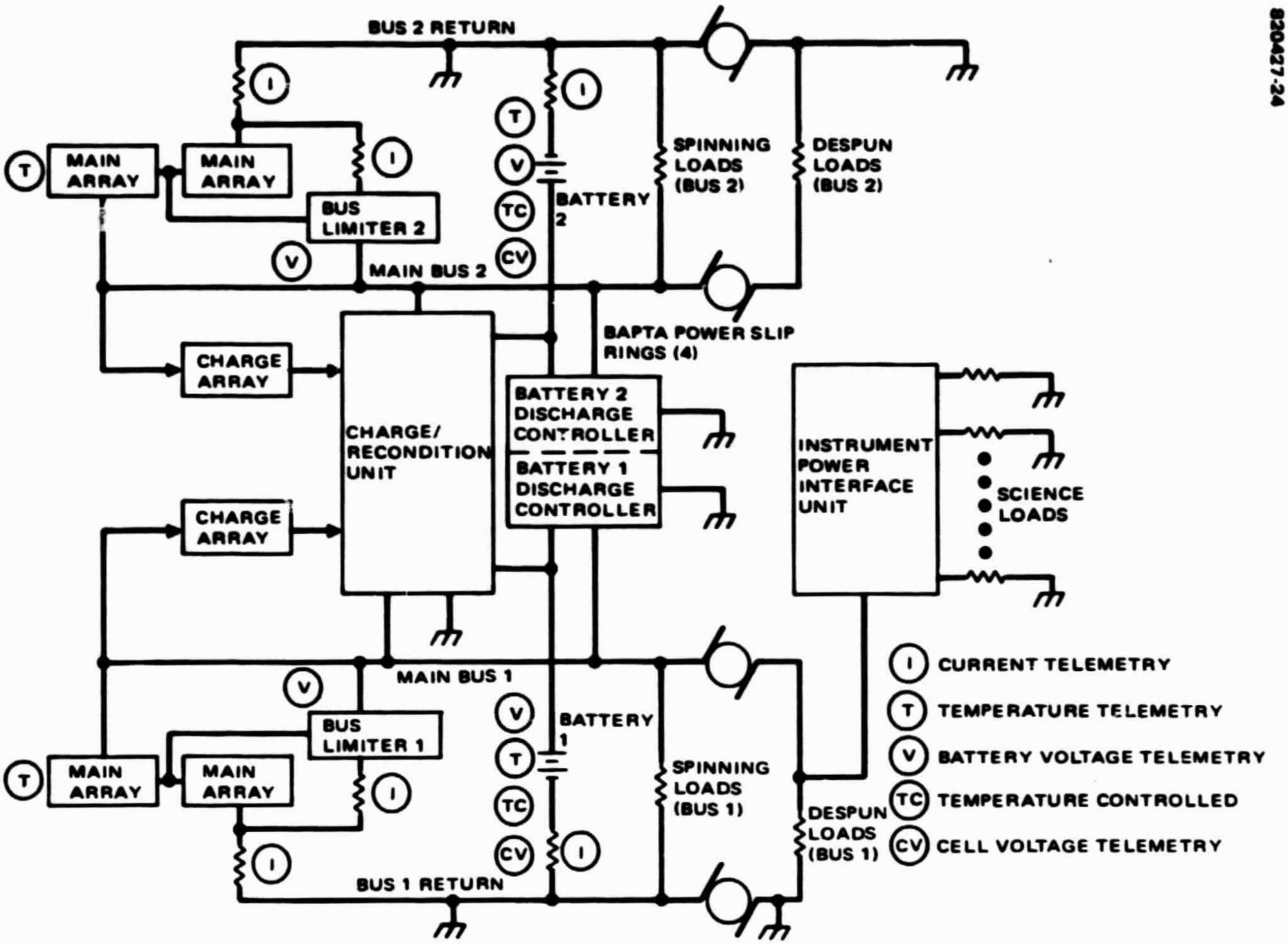

FIGURE 6-10. POWER SUBSYSTEM FUNCTIONAL DIAGRAM

6.8 Thermal Control Subsystem

Both Mars Orbiter spacecraft feature a passive thermal design. This design retains the Shuttle compatibility and component heritage of the HS 376 bus. The Mars thermal environment is colder than Earth and the internal dissipation of the conceptual Mars Orbiter is considerably less than typical HS 376 spacecraft. These factors result in the use of unit heaters, elimination of the HS 376 thermal radiator band, and added insulation inside the forward (inner) solar array.

Figure 6-11 shows the key features of the thermal control subsystem. A multi-layered aluminized Kapton blanket forms a despun barrier which closes the forward endplane. A 0.01 cm titanium thermal barrier closes the aft end and protects the spacecraft from the insertion motor plume. Sensors in critical locations throughout the spacecraft monitor the internal temperatures. Blankets cover the exposed portion of the STAR-31 climatology MOI motor, preventing thermal gradients.

Blankets with openings for instrument apertures and radiators cover the science instruments on the despun platform. The spacecraft body shades the climatology instruments from the sun.

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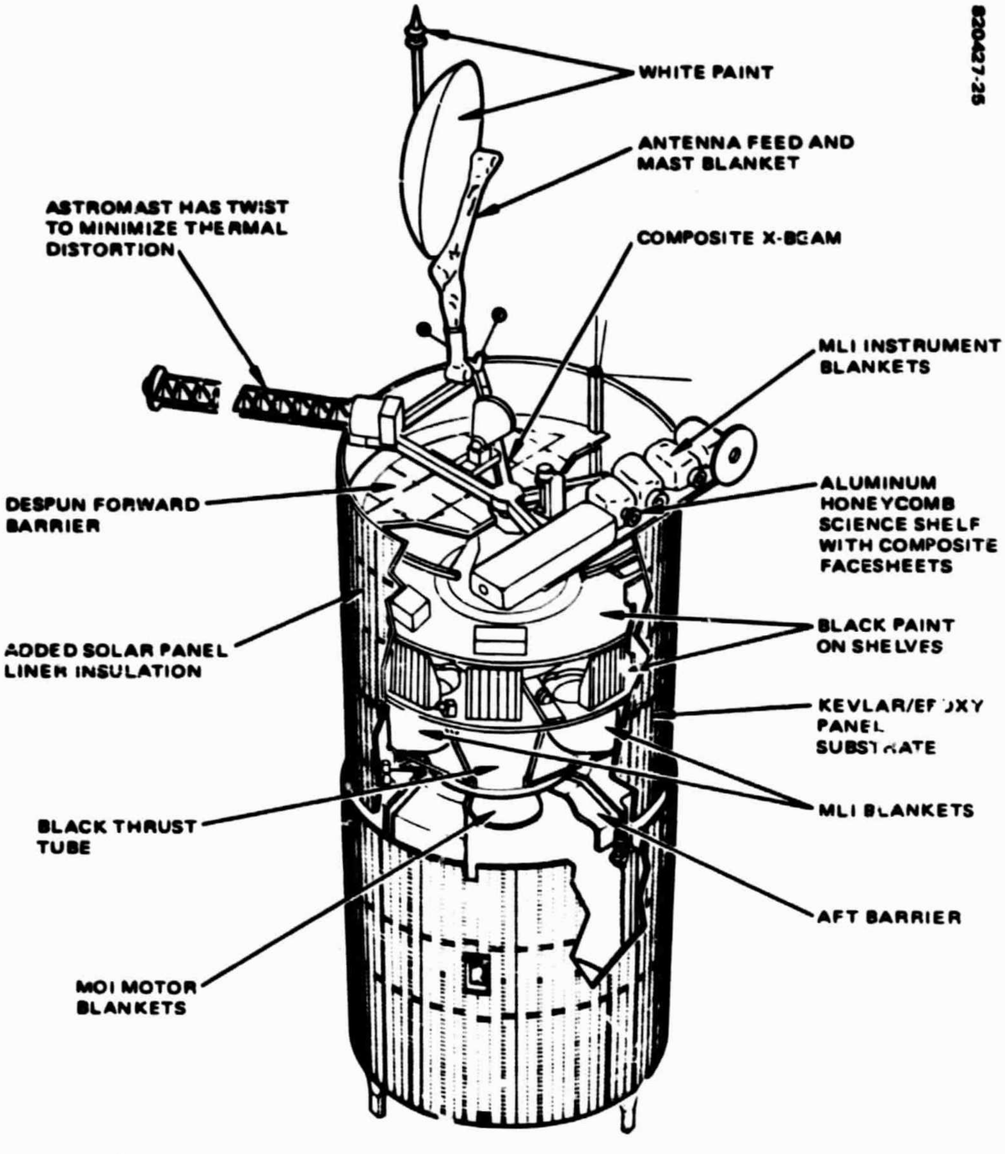

FIGURE 6-11. THERMAL CONTROL SUBSYSTEM COMPONENT LOCATIONS

Power bus regulation allows the passive thermal design to minimize component temperature excursions by holding the internal dissipation nearly constant. Added insulation inside the forward array isolates the equipment bay from the Martian cold and damps the thermal eclipse transients of the short-period orbits.

6.9 Integrated Propulsion Stage

The integrated propulsion stage (IPS) designed for Intelsat VI, injects either spacecraft on its trajectory to Mars. The launch configuration, shown in Figure 6-12, consists of the spacecraft, the IPS, and a cradle.

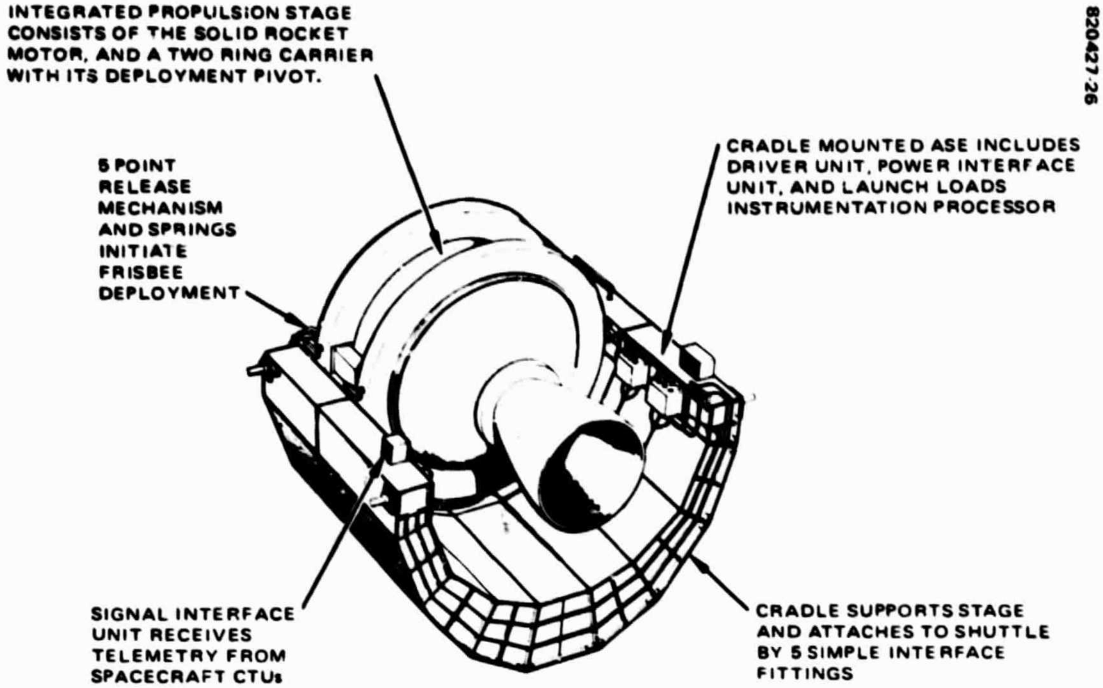

FIGURE 6-12. INTEGRATED PROPULSION SYSTEM FEATURES

The cradle supports the IPS and provides the mechanical and electrical interfaces with the orbiter. In addition, the cradle contains the frisbee ejection mechanisms that cause the ejected spacecraft/IPS stack to simultaneously move away from the Shuttle and rotate at 2 rpm. The complete cradle assembly remains in the orbiter after the ejection. After landing, checkout and refurbishment prepare it for the next launch.

The IPS consists of a solid rocket motor (SRM-1), its aluminum carrier support structure, and two STAR-6 solid rockets which spin up the ejected stage and spacecraft to 30rpm. Five moment-free attach points connect the carrier to the cradle. Two carrier outriggers provide reaction points for the frisbee ejection spring and pivot.

United Technologies Chemical Systems Division will qualify the SRM-1 for propellant loads from 4850 to 9700 kg. Figure 6-13 compares the capability of the fully loaded SRM-1 with the requirements of the Mars Orbiter missions. The energy demands of the climatology and aeronomy missions require propellant loads of 7706 kg and 6128 kg respectively.

Five Inteslat VI, four Leasat, and four SAL spacecraft will demonstrate the Hughes-patented frisbee deployment technique before it is needed for Mars Orbiter.

The design details of the integrated propulsion system are Hughes proprietary.

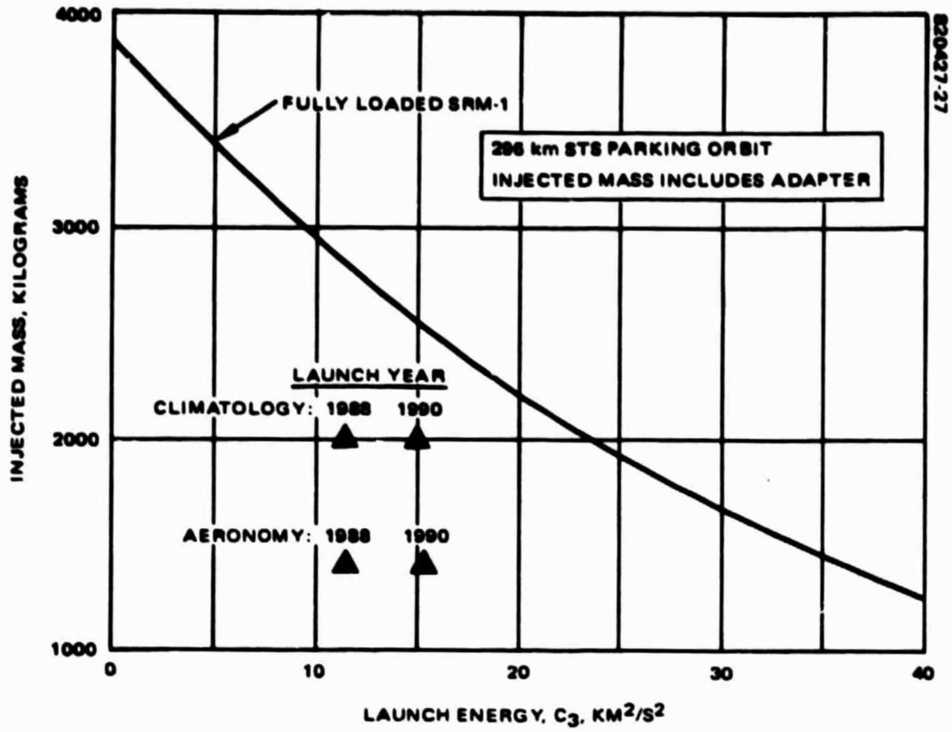

FIGURE 6-13. INTEGRATED PROPULSION STAGE CAPABILITY FOR MARS MISSIONS