Mars Observer Spacecraft Description

Dennis L. Potts*
Jet Propulsion Laboratory, California Institute of Technology, Pasadena, California 91109

The Mars Observer spacecraft implements the science and mission objectives for a planetary observer program with a design baseline evolving from existing, proven, flight subsystem designs and production techniques. The spacecraft conforms to a set of high-level functional requirements, allowing a development process with a high degree of flexibility in meeting performance, mission, and science requirements. The intent of the implementation approach is to procure a design-to-cost, reliable, production-type spacecraft that can accommodate the complement of science instruments and meet mission requirements with adequate margins.

Introduction

Overview of Mission and Science Objectives

THE Mars Observer mission provides a spacecraft platform in orbit around Mars such that the entire Martian surface and atmosphere may be mapped for approximately one Martian year. With a launch in late 1992, the spacecraft will be carried to its low, elliptical Earth parking orbit by the Titan III launch vehicle and then injected into the trans-Mars trajectory via the Transfer Orbit Stage (TOS). The three unique configurations of the spacecraft—launch, cruise, and mapping—are shown in Fig. 1. Following separation from the TOS upper-stage vehicle and attitude initialization, the spacecraft achieves its cruise configuration for minimum operations during its 11-month flight to Mars. During this cruise period, there will be a partial deployment of the solar arrays, the high-gain antenna, and the magnetometer/electron reflectometer and gamma-ray spectrometer science instruments. The spacecraft will achieve a near-circular, near-polar mapping orbit at a reference altitude of 378 km above the Martian surface. After final deployments and spacecraft and science instrument checkout, the full mapping mission will begin.

The mapping mission starts and ends at the autumnal equinox, allowing the study of seasonal variations on the planet over the mission of one full Martian year. Earth range for the mapping mission varies from 0.68 to 2.45 AU, with a perihelion of 1.38 AU and an aphelion of 1.67 AU. The mapping orbit is Sun synchronous at 2 p.m. local mean solar time. The beta angle (angle of the Sun out of the orbit plane) varies from 38 to 14.8 deg because of the eccentricity of Mars' orbit and its obliquity to the ecliptic.

The primary science objective is to conduct geoscience and climatology experiments on a global scale. Geoscience includes definition of the topography and gravitational field, determination of the elemental and mineralogical character of the surface material, and the establishment of the nature of the Martian magnetic field. Climatology experiments entail determination of the time and space distribution, abundances, sources, and sinks of volatile material and dust over a seasonal cycle, as well as exploration of the structure and other aspects of the circulation of the atmosphere.

Received Dec. 11, 1989; revision received April 17, 1991; accepted for publication April 29, 1991.

*Project System Engineer, Mars Observer Project, MS 264-627, 4800 Oak Grove Drive. Associate Member AIAA.

Science Payload Accommodations

The science payload consists of seven instruments, along with an ultrastable oscillator used for radio science, as defined in Table 1. These instruments complement one another, adding synergism to the full science payload. The spacecraft bus provides instrument fields of view, power, discrete and sequence command handoff, data return, pointing mechanical alignment, thermal control, contamination (purge) control, and ground electrical and performance testing for design verification at the system level.

Spacecraft Development Background

The Mars Observer mission was conceived as a pathfinder mission, based on a design-to-cost spacecraft development approach. The intent was to provide a production-type spacecraft system that would support generic and observer missions. Inherent in the approach was the desire to make

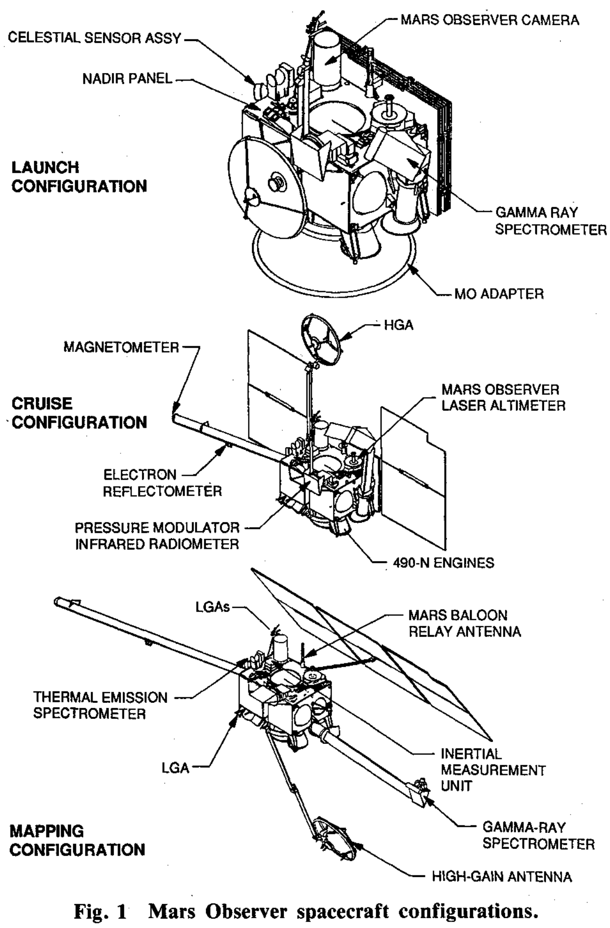

Fig. 1 Mars Observer spacecraft configurations.

Table 1 Science payload characteristics

Instrument	Measurements	Mass, kg	Power, W Orb. avg./pk.	Data record rates, b/s
Gamma-ray spectrometer	Primary/secondary gamma rays, bursts @ 200 keV–10 MeV and neutrons to 10 keV	24.4	16.3/16.3	655
Magnetometer/ electron reflectometer	Magnetic fields ranging from ±16 to 65,536 nT; electrons from 1 to 20 keV	5.4	4.6/4.6	324/648/ 1296
Mars balloon relay	Signal relay receiver interrogating @ 437 MHz; receiving @ 401–406 MHz	6.8	12.5/12.5	N/A
MO camera	Imaging via narrow/wide angles @ f/10, 3.5 m for 0.5–0.9 μm; f/6.5, 11.3 mm for blue @ 0.4–0.45 μm, and red @ 0.58–0.63 μm	21.1	22.8/29.8	700/2856/ 9120 (29,260)RT[a]
MO laser altimeter	Topography at 74-cm focal-length telescope receives 1.06-μm pulse with 1-nW sensitivity; 100-m footprint @ 2-m accuracy (maximum)	25.9	28.7/28.7	618
Pressure modulator infrared radiometer	Pressure-modulated CO_2 and H_2O channels plus 1 albedo @ 0.3–3.0 μm and 6 IR @ 7.0–46.5 μm	40.2	34.1/34.7	156
Thermal emission spectrometer	Interferometer @ 6.25–50-μm spectral range and radiometrics @ 0.3–100 μm	14.4	15.6/18.2	688/1664/ (4992) RT[a]
Ultrastable oscillator	Frequency reference source yields stability of 5×10^{-12} for 0.1-s integ period; 10^{-12} for 1.0-s integ period; 4×10^{-13} for > 10-s periods	1.3	3.0/3.0	N/A

[a]RT = real-time data rate, b/s.

maximum use of existing subsystem designs and design technologies; thus, significant importance was placed on maintaining a high degree of heritage with flight-proven equipment. The spacecraft, when integrated with an upper stage for trans-Mars trajectory injection, provides a capability to conduct multiple missions with a single baseline system design. The Mars Observer mission requirements were specified with the intention that they would encompass the requirements for all of the planetary observer missions.

A spacecraft design was thus pursued that would provide a maximum-capability platform to which a unique science complement could be selected, based in part on conformance to spacecraft capabilities, and accommodated by the spacecraft bus. The spacecraft performance has been derived primarily from the heritage from which it evolved. This provides a design constraint on the development and on the operational capabilities and requirements of the science instruments.

The spacecraft system derives its heritage from the Defense Meteorological Satellite Program (DMSP) and the TV and Infrared Observational Satellite (TIROS) Program satellites for most of the avionics elements (e.g., command and data handling, telecommunications, guidance and control, power supply electronics) and from the Satcom-K series of communications satellites for the structural, thermal control and some power generation elements. In general, subsystems and components from these existing spacecraft are being used for Mars Observer (MO) with well-understood changes. New developments unique to MO include a steerable high-gain antenna, a partial shunt assembly for power regulation and control, a bipropellant propulsion subsystem, a Mars horizon sensor assembly, an X-band transponder, and a command detector unit.

The spacecraft has been procured under a development contract with fixed-price terms. These terms impose a somewhat unusual system development in that the spacecraft rigidly adheres to a set of well-defined functional requirements. This gives the development contractor a high degree of flexibility in accomplishing the design requirements but places considerable constraint on the payload and other government-furnished property (GFP) by way of directed conformance. The spacecraft design is controlled through the use of contractually required documents, which are intended to completely characterize the design capabilities and the system/subsystem interfaces, and a series of formal and informal design reviews, intended to provide insight into the development process as well as conformance to requirements.

Spacecraft System Description

System Architecture

The spacecraft system architecture consists of nine hardware subsystems plus a software subsystem, which collectively accommodate the payload and meet the mission and science requirements. These subsystems are the following: 1) attitude and articulation control subsystem (AACS), 2) electrical power supply subsystem (EPS), 3) propulsion subsystem (PROP), 4) structure subsystem (STR), 5) thermal control subsystem (TCS), 6) command and data handling subsystem (CDHS), 7) telecommunications subsystem (TCM), 8) mechanisms subsystem (MECH), 9) cabling/harness subsystem (CBL), and 10) flight software subsystem (S/W).

System Operational Overview

The Mars Observer spacecraft will be placed into a low Earth orbit via a Titan III launch vehicle. Trans-Mars injection will be provided by a TOS upper-stage vehicle. The spacecraft accommodates four mission phases (launch, cruise, orbit insertion, and mapping) with three unique configurations (launch, cruise, and mapping), as illustrated in Fig. 1. During the launch phase of the mission, communications and telemetry will be provided by the TOS telecommunications subsystem. Utilizing the spacecraft's data storage subsystem, both real-time and recorded data of the launch and deployment sequences will be obtained. Following separation from the TOS and achievement of attitude initialization, the spacecraft autonomously assumes its cruise configuration for minimum operations during its 11-month flight to Mars. Early in the cruise phase the spacecraft will partially deploy its solar array, the high-gain antenna, the magnetometer/electron reflectometer, and the gamma-ray spectrometer booms.

The spacecraft maintains a slow, controlled roll during the cruise phase at 0.01 rpm, with the body-fixed solar arrays pointed off the Sun line at up to 60 deg for power management and thermal control considerations. Trajectory correction maneuvers are performed throughout this period to correctly target the spacecraft for a Mars orbit insertion maneuver, which places the spacecraft in a highly elliptical near-polar orbit about the planet. The orbit insertion phase continues with a series of maneuvers and orbital drift periods, spanning up to three months, to minimize fuel usage and to provide proper orbit-Sun geometry. Eventually, the spacecraft is lowered into its low circular mapping orbit, with a 2 p.m. (local Mars time) descending node for maximum science benefit. Once in this orbit, there will be an initial gravity calibration period to allow optimum navigation control during the mapping phase of the mission.

After the final maneuvering to the near-polar mapping orbit at a 378-km reference altitude, and following the gravity calibration period, final deployment of all extendable components is completed and a spacecraft/science instrument checkout is performed in preparation for the actual science mission. The spacecraft design accommodates the Earth and Sun range variations delineated previously while accommodating the science and engineering data transmission requirements throughout the mapping mission.

Table 2 Mars Observer summary mass breakdown

Current best estimates	Mass, kg
Spacecraft dry mass	
AACS	57
EPS	200
PROP	128
STR/adapter	219
TCS	39
CDHS	79
TCM	40
MECH	95
CBL	87
Balance	8
	952
Government-furnished property	
GRS	24.4
MAG/ER	5.4
MBR	6.8
MOC	21.1
MOLA	25.9
PMIRR	40.2
TES	14.4
USO	1.3
PDS	11.9
CDU(2 ea)	3.0
Reserve	11.6
	166.0
Propellants[a]	1369.0
BIPROP	1306.0
MONOPROP	63.3
Total estimate	2487
Current capacity	2572

[a]Propellant calculations based on 2487 kg of injected mass.

Spacecraft Requirements and Expected Performance

The basic performance requirements on the spacecraft are to provide a stable controlled platform for the acquisition of science data, to provide the necessary communications link for telemetry transmission to Earth, to maintain resource consistency with the upper stage and launch vehicle injection capabilities, and to accommodate science and bus configuration and operational requirements essential to meeting mission objectives.

For the Titan III launch vehicle, the spacecraft design is constrained to an injected mass of 2522 kg. While the current design is being specified at an injected mass of 2522 kg, the mission is being evaluated to identify potential launch vehicle capability improvements. Table 2 summarizes the mass breakdown and allocations of Mars Observer.

The spacecraft must provide power to the bus and payload elements continuously throughout the entire mission profile. Load and output vary throughout the profile, commensurate with mission phase, calibrations, maneuvers, and science operations. To accommodate a high solar array output during the early cruise phase, the spacecraft is pointed off the Sun line by up to 60 deg to limit the load on the shunt radiators. As the spacecraft recedes from the Sun, the solar arrays are aligned closer to the Sun line, and the spacecraft is held in an inertially controlled spin attitude. Once in the mapping orbit, the arrays and the high-gain antenna (HGA) are fully deployed and pointed toward the Sun and the Earth, respectively, via two-axis gimbals. Tables 3 and 4 show the load and mission power summaries for Mars Observer.

Telecommunications link performance is specified in terms of gain-to-temperature ratio (G/T) for the uplink and effective isotropic radiated power (EIRP) for the downlink. Unique specifications for the various mission (configuration) phases exist for MO, and those requirements and performance capabilities are shown in Table 5.

The spacecraft maintains adequate pointing control and provides sufficient telemetry to allow reconstruction of flight parameters to support science instrument observations. Spacecraft pointing knowledge is after-the-fact attitude reconstruction. The spacecraft bus provides data in its engineering telemetry stream sufficient to characterize nadir and HGA

Table 3 Mars Observer summary peak power breakdown

Mission phase	Max. load[a], W	Available solar array output, W	Margin, W
Launch	542	672	130
Inner cruise	614	740	126
Outer cruise	627	661	34
Orbit insertion	666	838	172
Gravity calibration	832	938	71[b]
Mapping			
Perihelion	951	1408	179[b]
Aphelion	1072	1147	45[b]

[a]Peak steady state. [b]Based on orbital average energy balance.

Table 4 Mars Observer battery depth-of-discharge performance

Mission phase	Predicted, %	Allowed, %	Margin, %
Launch	33.5	50	16.5
Maneuvers	36.4	50	13.6
Mapping			
Gravity calib.	15.3	27	11.7
Perihelion	17.3	27	9.7
Aphelion	21.1	27	5.9

Table 5 Mars Observer link performance (worst-case values)

Modes	G/T	EIRP	Requirement	Margin
	Uplink, dB/K			
Inner cruise	−22.8	—	−35.0	12.2
Outer cruise, drift, mapping	−11.5	—	−17.5	6.0
Emergency	−24.2	—	−28.2	4.0
	Downlink, dBw			
Inner cruise	—	20.2	17.5	2.7
Outer cruise, drift, mapping	—	51.4	51.4	0.0
Emergency	—	20.0	16.0	4.0

Table 6 Mars Observer primary pointing and stability margins (mapping phase: mrad, 3 σ)

Parameter	Axis	Predict	Requirement	Margin
	Nadir panel instrument pointing			
Control	Roll	2.45	10	7.55
	Pitch	1.99	10	8.01
	Yaw	2.24	10	7.76
Knowledge	Roll	1.93	3	1.07
	Pitch	2.61	3	0.39
	Yaw	1.00	3	2.00
	Nadir panel instrument stability			
0.5-s stability	Roll	0.40	0.5	0.10
	Pitch	0.45	0.5	0.05
	Yaw	0.50	1.0	0.50
12-s stability	Roll	2.58	3	0.42
	Pitch	1.49	3	1.51
	Yaw	0.87	3	2.13
	High-gain antenna pointing			
Control	Boresight	2.23	8.7	6.47
Knowledge	Boresight	2.23	3	0.77
Stability[a]	Boresight	1.87	3	1.13

[a]Based on a 300-s integration period.

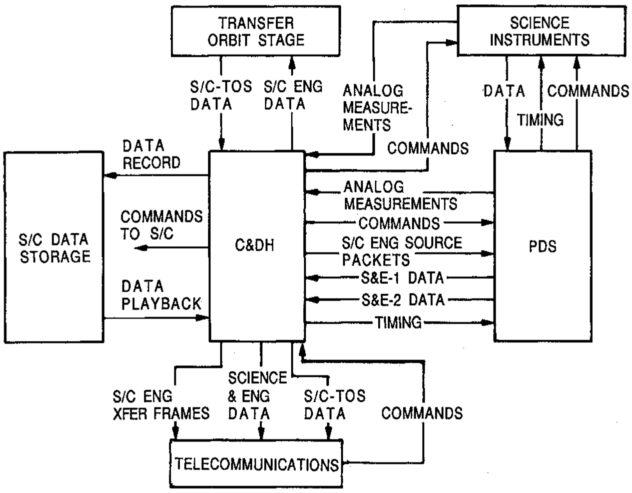

Fig. 2 Command and data flow diagram.

Table 7 Mars Observer maneuver execution margins (cruise through mapping: ΔV, 3 σ)

Parameter	Thruster thrust, N	Predict	Requirement	Margin
Proportional	490	0.07	2	1.93
Magnitude	22	0.38	2	1.62
Velocity	4.4	1.80	2	0.20
Error, %				
Fixed	490	0.031	0.05	0.019
Magnitude	22	0.002	0.05	0.048
Velocity	4.4	0.002	0.05	0.048
Error, m/s				
Side	490	7.93	25	7.07
Velocity	22	6.05	25	18.95
Error, m/s	4.4	7.17	25	17.83

pointing to within ±3 mrad (per axis, 3 σ) and science instrument boom pointing to within ±25 mrad (per axis, 3 σ). The spacecraft will be commanded to conduct maneuvers and flight-path control consistent with mission design requirements. Maneuver errors are either fixed, independent of impulse magnitude, or proportional, directly related to the impulse magnitude. Tables 6 and 7 provide the details of the pointing and maneuvering performance capabilities and margins.

Spacecraft pointing stability, defined as attitude angular excursion variations of the spacecraft body axes from the orbital reference coordinate system, is maintained over two time intervals, as delineated in Table 6. HGA pointing variations are controlled to within 3 mrad over any 300-s integration period.

The spacecraft telemetry data handling is accomplished through the engineering data formatter (EDF), located within the CDHS, the payload data subsystem (PDS), the telemetry subsystem, and the CDHS, as shown in Fig. 2. The spacecraft accommodates multiple data modes to coordinate and transmit science and engineering (S&E) data to the ground. Science data are Reed-Solomon (R-S) encoded, to reduce attendant transmission bit errors, prior to storage on the digital tape recorders or transfer to the telemetry subsystem for subsequent downlink. Table 8 shows the various combinations of data rates and data modes available upon ground command for playback of recorded data and engineering data, as well as for real-time science and engineering data.

The approach to system autonomy design has been to facilitate a low-complexity mission by minimizing spacecraft-to-instrument operational interfaces, providing for sufficient system performance margin, and maximizing spacecraft redundancy. The science instruments, by design, sequence autonomously, with minimal interaction with each other or with the spacecraft bus. Each instrument has its own internal sequencing and no complex validation process requirements, as is usually the case with interactive sequencing. The spacecraft accommodates commands, control, and major mode and state changes via the EDF, CDHS, and PDS. Science instrument commands are passed through the PDS, via the CDHS. As with most planetary spacecraft missions, Mars Observer is designed to operate autonomously, with onboard fault protection, for extended periods throughout all phases of the mission.

Spacecraft Subsystem Descriptions

Attitude and Articulation Control

The AACS provides, through the use of the appropriate planetary ephemeris and star catalog, the inertial reference for the spacecraft. The AACS provides the nadir pointing for the science instruments, momentum management control for induced disturbances, thruster control for attitude adjustments, and pointing of the solar panels and high-gain antenna. The AACS is able to provide closed-loop pointing control by means of a horizon sensor and celestial data provided by onboard sensors and uplinked ephemeris and star catalog data.

The AACS is based on the TIROS/DMSP design for a three-axis-stabilized, nadir-pointing, zero-momentum controlled spacecraft bus. With the exception of during maneuvers, when control is provided by thrusters, pointing control is maintained via a set of three reaction wheel assemblies. During the cruise to Mars and orbit insertion phases of the mission, the spacecraft is maintained in an inertially controlled roll at approximately 0.01 rpm. The attitude is maintained by periodically unloading accumulated momentum, using low-contamination hydrazine thrusters under autonomous software control. Once in the mapping orbit, a nadir-pointing attitude is maintained with similar momentum management and control.

A 4π-sr Sun sensor provides attitude reference after TOS separation and for reinitialization after an attitude anomaly. Inertial attitude sensing is done by a celestial sensor assembly (CSA), a fixed-star sensor that views a band of stars around the rotational axis. The CSA is used during the mapping phase to provide precise attitude knowledge. In the mapping orbit, spacecraft attitude control is maintained using a closed-loop Mars horizon-sensing system, operating at the CO_2 wavelength. Roll and pitch errors are used to maintain a nadir-pointing direction. An inertial measurement unit (IMU) contains gyros and accelerometers for measuring angular rates and linear accelerations of the spacecraft. These IMU measurements are used for fixed inertial pointing during maneuvers and to measure yaw attitude in the mapping phase.

Attitude determination software is built around two modules for sensor-based pointing: the horizon-sensor assembly (HSA) attitude determination software (HADS) and the CSA attitude determination software (CADS). During the cruise and orbit insertion phases, the CSA/IMU measurements and a stored spacecraft ephemeris are used by the CADS for pointing relative to the Earth and Sun. During the mapping phase, both software modules are active: HADS, using the Mars horizon-sensor assembly (MHSA) and IMU measurements for nadir-pointing control; and CADS, using CSA/IMU measurements for inertial pointing knowledge. CADS also controls pointing of the HGA toward Earth, using the stored planetary ephemeris. Backup control modes exist to operate in the mapping phase with a loss of either the MHSA or CSA, but they require stored spacecraft ephemeris to relate the nadir-pointing direction to the inertial reference.

Electrical Power Subsystem

The EPS provides the conversion, supply, regulation, and distribution of all electrical power required for the spacecraft bus and the payload. The EPS also provides for replacement and supplemental heat, maintenance of the batteries, pyrotechnic device power, and overload protection for the power bus.

The electrical power subsystem is based on the DMSP/TIROS series satellites, with slight modifications. The subsystem includes a six-panel photovoltaic solar cell array with an area of approximately 24.5 m^2, power conditioning and distribution electronics, and an autonomous battery energy management capability operated in a direct energy transfer configuration. The subsystem utilizes shunt regulation to control excess power and heat generated during low-load situations and a boost-voltage regulator system to maintain a constant bus voltage at 28 V dc $\pm 2\%$.

The solar arrays have cells configured in series and parallel, with redundant diode isolation, to control both temperature and current output. Two nickel-cadmium batteries, rated at 42 A-h each, provide the required capacity when operating during Sun eclipse. To ensure an operational lifetime of over 8000 cycles, the batteries are passively controlled to within $\pm 5°C$, and depth of discharge is limited. The batteries are discharged in parallel, but charged separately, to maintain each unit's current, voltage, and temperature levels. Battery reconditioning onboard is not a design capability, but it can be accommodated prior to launch via ground equipment.

Propulsion Subsystem

The PROP provides the propulsive capability to support the AACS throughout all phases of the mission, including trajectory correction maneuvers, Mars capture and orbit circularization maneuvers, and orbit maintenance. The PROP also accommodates the final maneuver to place the spacecraft in its quarantine orbit following mission completion. The PROP uses two propulsion systems, namely, a bipropellant system for large maneuvers and control and a monopropellant system for orbit sustenance, momentum unloading, and control during small maneuvers. The PROP also provides the pyrotechnic devices required for propellant isolation and deployment/release mechanism control.

All major maneuvers, including trajectory correction maneuvers (TCMs) and Mars orbit insertion (MOI) maneuvers, will be accomplished by a hypergolic bipropellant subsystem. Major ΔV impulses are provided using two of four 490-N (110-

Table 8 Science and engineering data modes and rates

	Mode ID: record/playback, b/s				Real time, b/s			
Downlink:	LRC[a]	MRC[b]	HRC[c]	POR[d]	RTL[e]	RTM[f]	RTH[g]	MBR[h]
S&E-1 rate:	3488	6976	13,952	3488	3488	6976	13,952	3488
GRS	665	665	665	—	665	665	665	—
PMIRR	156	156	156	—	156	156	156	156
MAG/ER	324	648	1296	—	324	648	1296	324
MOLA	618	618	618	—	618	618	618	—
TES	688	1664	1664	—	—	—	—	—
MOC1	700	2856	9120	—	1388	4520	10,782	2670
S/C engrg.	256	256	256	256	256	256	256	256
PDS engrg.	48	48	48	3072	48	48	48	48
Filler	1	1	1	128	1	1	3	2
Frame hdr.	32	64	128	32	32	64	128	32
S&E-2 rate:	—	—	—	—	34,261	34,261	34,261	—
TES					4992	4992	4992	—
MOC2					29,269	29,269	29,269	—
S/C engrg.		(N/A)			256	256	256	—
PDS engrg.					48	48	48	—
Filler					4	4	4	—
Frame hdr.					320	320	320	—

[a]Low-rate record @ 4000 sps (with R-S encoding).
[b]Medium-rate record @ 8000 sps (with R-S encoding).
[c]High-rate record @ 16,000 sps (with R-S encoding).
[d]Power-on reset mode @ 400 sps (with R-S encoding).
[e]40,000 low-rate real-time + 4000 LRC (with R-S encoding).
[f]40,000 medium-rate real-time + 8000 MRC (with R-S encoding).
[g]40,000 high-rate real-time + 16,000 HRC (with R-S encoding).
[h]Mars Balloon Relay mode.

lbf) engines. Four 22-N (5-lbf) thrusters are used for torque control as well as ΔV impulses. Two 1.07-m-diam fuel tanks and one 0.66-m-diam helium pressurant tank carry the 1300 kg of fuel plus 5 kg of pressurant required for the mission. Their total capacity is 1369 kg of propellants.

A hydrazine monopropellant system is used for orbit trim maneuvers during the mapping phase of the mission, for the quarantine maneuver, and for some attitude control functions, including momentum dumping. Besides being more adaptable to multiple-cycle operations, the hydrazine system offers a low-contamination environment for the science instruments. The monopropellant system uses 63 kg of fuel and can be used for quarantine orbit insertion if required. Four 0.9-N reaction equipment assemblies (REAs) provide reaction wheel desaturation and orbit trim maneuvers, whereas eight 4.4-N REAs provide Z-axis (nadir) control and trim maneuver authority. Two 0.48-m-diam tanks, with a total capacity of 84 kg, are used to contain the monopropellant. The propulsion subsystem, while incorporating considerable new development, is fully redundant operationally and is capable of meeting full mission requirements with only half of each system operational.

The bipropellant thrusters are located on the $-Z$ axis of the spacecraft, with control thrusters mounted on panels facing the velocity and antivelocity directions. The thrusters for the monopropellant system are also located on the velocity and antivelocity panels. Thruster operations are controlled during the mapping mission such that all pointing requirements can be met on a continuous basis.

Structure

The STR accommodates the spatial configuration and physical interfaces of the total spacecraft equipment and provides the basic load-carrying paths during ground, launch, and flight operations. The STR provides the interface with the TOS via the interface adapter and routes essential purge lines for prelaunch contamination control.

The MO spacecraft is configured around a Satcom-K-based bus with a central cylinder for primary load path transfer. The structure is primarily responsible for accommodating the launch vehicle interface, bus elements such as solar array and HGA booms, and the payload instruments and associated equipment such as booms and purge connections. For compatibility with the Titan III launch vehicle, the spacecraft is designed for a 2522-kg injected mass, including a total payload capacity of 166 kg.

A separation assembly located on the launch system adapter provides the necessary components for interface management with the TOS inertial upper-stage vehicle. The structure design is basically modular to maximize flexibility for design integration and payload accommodation activities. The structure and equipment are configured such that cabling runs and interconnections are optimized for integration.

Thermal Control

The TCS provides the capability to maintain flight equipment within allowable temperature limits throughout the mission. Thermal stability and minimization of thermal gradients are critical requirements that are satisfied by the TCS.

The TCS also derives its heritage from the Satcom-K communications satellite program. The spacecraft bus is required to provide the overall thermal control for the payload as well as the bus elements, whereas the instrument internal thermal design responsibilities are apportioned to the respective equipment suppliers. Although broad thermal control requirements have been established for the GFP, the spacecraft readily provides an operating environment for all equipment close to 20°C.

Thermal control is accomplished through an interrelated design of multilayer blankets, paint and tape, active and passive heaters, and localized radiators. The blankets utilized yield emissivities of 0.03–0.05. Internal configurations are constrained to aid in system thermal control functions and to minimize the need for supplemental heaters and temperature control devices.

Active and passive control is provided for specialized uses, such as thruster catalytic bed heaters, science instrument alignment, external (boom) cabling, dampers, gimbal drive electronics, and miscellaneous attitude control sensors.

Command and Data Handling

The CDHS acts as the focal point for orchestration of all major avionics functions of the spacecraft. This subsystem, which includes the main spacecraft computer and the spacecraft controls processor, provides for the receipt and transfer of commands for both the bus and the payload. It also verifies command validity; provides the basic timing reference for all spacecraft functions; collects, stores, and routes telemetry and signal data; and provides the control function for power management. Figure 2 depicts the functional elements and the overall data and command flow through the spacecraft system. The CDHS, which contains the flight software subsystem, has significant interaction with both the AACS and the telecommunications subsystems.

The CDHS is an all-digital subsystem that is based on the DMSP/TIROS design and incorporates the PDS as the data interface to the science payload. For commanding functions, the CDHS, which is used to control and configure the spacecraft, will receive incoming demodulated signals using the GFP command detector unit, perform onboard computations, and execute both externally received and internally generated commands for the bus and the payload. The CDHS also provides for the generation of precise spacecraft clock data. The CDHS houses all the software used for attitude control, spacecraft bus fault protection, and equipment state management. The data handling functions are to collect, format, route, store, and play back to the ground the following types of data: science instrument data, engineering data, command verification, CPU memory dumps, and combined science and engineering data, both in real time and from recorded mode.

The computational capability is provided by redundant microprocessors with identical software, consisting of 96K of RAM and 20K of ROM in the spacecraft controls processor (SCP) and 18K of ROM in the engineering data formatter (EDF) and provisions for autonomous switching in the event of a failure. Whereas the CDHS provides for convolutional encoding of transmitted telemetry, the PDS embodies the science and engineering transfer frame packets with Reed-Solomon coding to aid in bit error reduction.

The SCP, along with the controls interface unit (CIU), provides the control and computational functions for all modes of attitude determination and control; powered flight guidance and navigation; command decoding, checking, storage, and distribution; power management; failure diagnosis; and redundancy switching.

Storage is provided in the SCP memory for at least 1500 payload-dedicated commands for PDS control of science instrument functions, as well as 500 dedicated commands required by the bus itself. A time of execution is stored with each command to control onboard sequencing, and the duration of each load can be up to 144 h with a 1-s resolution. Up to 10 stored and real-time commands can be executed in any 1-s interval. The CDHS handles four different streams of science and/or engineering data for normal, emergency, and launch modes. Storage for recorded data is provided by three digital tape recorders (with one backup mechanical tape unit), each capable of recording 6.91×10^8 bits of information. The total required data storage capability is for 24 h at the maximum acquisition rate with a maximum interval of deep space station (DSS) coverage.

Telecommunications Subsystem

The telecommunications subsystem provides Deep Space Network-compatible X-band communications to and from

Earth for radiometric tracking, telemetry, commanding, and radio science. The spacecraft telecommunications equipment also accommodates a Ka-band feasibility demonstration experiment.

All spacecraft telecommunications will be handled using an X-band transponder and a government-provided command detector unit. The transmit power amplifiers are 40-W traveling-wave tubes, which are cycled each orbit for power considerations. The uplink command receivers are configured such that it is impossible to turn both off at the same time, since to do so could seriously jeopardize the mission.

Communications to and from the spacecraft are accomplished using a complement of one low-gain transmit antenna and two low-gain receive antennas for near-Earth operations and emergencies, plus a two-axis articulated high-gain antenna for mapping phase and late cruise and orbit insertion operations. The low-gain antennas are configured such that commands can be received over a wide range of spacecraft orientations. Commands can be received at the spacecraft at rates from 7.125 b/s (used primarily for emergency uplink conditions) to 500 b/s (used primarily for memory loading with the 70-m DSS). The nominal command uplink rate is 125 b/s. The transmit low-gain antenna is used for downlink immediately following launch, for initial acquisition, and during spacecraft emergency modes.

Two modes of navigational ranging are available aboard the spacecraft: 1) wideband turnaround, using signals originating at the ground stations; and 2) differential one-way ranging, using an onboard tone generator switched by command.

Mechanisms and Cabling

The MECH subsystem provides for the deployable and gimballed elements of the spacecraft bus and payload. These include a two-stage solar array deployment and gimballing system, a two-stage high-gain antenna deployment and gimballing system, a four-stage gamma-ray spectrometer boom deployment system, and a two-stage magnetometer/electron spectrometer science boom deployment system. The pyrotechnic devices (used to initiate deployment) are incorporated within this subsystem as well.

The instrument deployment mechanisms are canister boom devices that allow accurate positioning and orientation of the instruments away from the influence of spacecraft magnetic and radiation sources. Canister booms present low operational risk while providing partial deployments during the cruise phase. The solar panel deployment assembly consists of hinge assemblies, dampers, booms, and restraint devices that meet the requirements of launch and partial deployment during cruise.

Once deployments are completed, control of the solar array and the HGA is provided by means of two-axis gimbal devices for Sun and Earth pointing, respectively. The software for controlling the pointing of these bus elements, using planet ephemeris data, is contained in the onboard computer, the SCP.

The CBL consists of all spacecraft bus and intrapayload cabling, with few exceptions. Only those instruments with separate boxes that require interbox wiring deemed operationally critical will provide their own cabling.

Flight Software

The S/W is contained within the command and data handling subsystem. It provides the attitude control, data handling, power and thermal management, gimbal control, launch sequence control, maneuvers, fault protection, and pointing algorithms necessary to maintain autonomous flight operations. Hardware subsystem control functions specified by the various subsystems are incorporated within this subsystem.

The SCP flight software provides all of the functions to be implemented in the onboard computer. All functions are located in and operate from RAM, except the bootstrap and safe-mode software, which is contained in the nonvolatile memory of the SCP.

The software algorithms are directly derived from the DMSP/TIROS series satellites and supplemented to accommodate the unique aspects of a planetary mission spacecraft. The software coding is structured to be compatible with the 1750 architecture of the CDHS processors. Fault protection algorithms are contained in the software to correct for single-event upsets and soft errors. Memory can be isolated to allow a means to work around hard errors. The basic fault protection approach utilizes redundancy management as a first-line protection and then initiates graceful degradations through a multitiered system of anomaly modes to the safe-mode configuration, as required to protect the spacecraft.

Spacecraft Fault Protection

A combination of software redundancy management, hardware redundancy, and computer RAM backup logic provides autonomous onboard functions such as cold-start Sun-pointing control, inertial attitude determination, Mars-pointing attitude determination and acquisition, execution of preprogrammed maneuvers, and component failure detection and correction/replacement.

The mapping orbit phase attitude control modes, primary and backup, are entered autonomously, with ground-enabled preferencing. The primary mode uses IMU gyro rates and MHSA-derived roll/pitch measurements to provide yaw-gyro-compassing control to the local Mars-oriented coordinate reference frame. The controls are activated by means of reaction wheel assemblies. The onboard fault software also provides for autonomous pointing of the solar arrays and high-gain antenna during the cruise and mapping phases.

Having the mapping orbit ephemeris available on board makes possible two convenient backup modes: the CSA backup provides Mars pointing in the event of MHSA failure, and the MHSA can operate in a reduced-capacity state should the CSA fail in orbit. CSA operation, while requiring frequent periodic uplinks of planetary ephemeris data, allows for inherently better pointing due to reduced sensor errors, which is also why the CSA is used for attitude reconstruction.

In the event of loss of the high-gain antenna link, the spacecraft will enter the emergency mode, which provides for limited response over the low-gain antennas. The spacecraft remains in a nadir-pointing attitude if in the mapping phase, providing potentially rapid response and recovery.

In the event onboard inertial knowledge is lost, the spacecraft will enter the contingency mode. In this mode, the instrument covers are closed, the instruments are commanded to a safe state, and the spacecraft returns to the Sun-acquisition attitude control mode. The transmit low-gain antennas are selected, and the low command rate is selected by the uplink receivers.

Notwithstanding an approximate 25-day period for solar conjunction, the spacecraft is capable of redundancy management and fault protection actions necessary to provide, for at least 40 h, the minimum functions required for safe operations, including payload protection from abnormal spacecraft states or attitudes. A safe mode will be entered autonomously if both onboard processors fail their health-monitoring check process or if the redundancy management software fails to provide a minimum stable system configuration. The spacecraft can also be ground commanded, as for solar conjunction, into the safe mode.

In general, mode transitions follow a definite, orderly sequence of events. All permissible transitions are achievable through ground command. Autonomous transition to safe mode may be inhibited by ground command, as in the case of major flight maneuvers. However, in the event of onboard failures that disrupt normal ground-to-spacecraft communications, the spacecraft performs autonomous action that makes it always receptive to low-rate ground commands.

Summary

The Mars Observer mission represents a unique and challenging concept in the development and procurement of a spacecraft system in a system contract mode. Inherent in the approach is an underlying concept of design to cost, which sets the tone for both the management and technical development tasks and the corresponding evolution of the spacecraft and science instrument designs. The result is a spacecraft system that has evolved with strong influence from existing designs, and one that lends itself to a production-type development process. The Mars Observer spacecraft is fully a planetary spacecraft derived from an Earth orbital baseline, both in philosophy and in the finished product. Mars Observer is a near-flight-proven system that offers maximum capability for follow-on planetary observer missions.

Acknowledgment

The work described in this paper was carried out by the Jet Propulsion Laboratory, California Institute of Technology, under a contract with NASA.