Mars Observer Mission and Systems Overview

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The objective of the Mars Observer mission is the scientific investigation of the planet's surface, atmosphere, and gravitational and magnetic fields. The Mars Observer spacecraft provides a three-axis-stabilized platform for the payload system. This spacecraft is the first in a planned series of planetary orbiters that use Earth-orbiting design heritage adapted to planetary exploration mission objectives. The operation of the payload instruments is based on a new distributed concept, providing home-based commanding capabilities for the science investigators. The spacecraft will be launched during the September/October opportunity of 1992 and will travel approximately 11 months to Mars. During a two-year mapping period, the spacecraft will be placed in a nearly circular, low-altitude, Sun-synchronous orbit with a repeat cycle of 7 Martian days. At the end of the mission, the spacecraft will relay scientific information from scientific equipment, placed on the Martian surface, as part of a joint American, Soviet, and French experiment.

Introduction

THE objectives of the Mars Observer mission design are to facilitate a low-complexity mission both onboard the spacecraft and on the ground. The tasks encompass spacecraft operations from launch through the end of the mission. The approach is standardization of operations both on the spacecraft and on the ground, with routine performance of the majority of functions. A low-activity cruise period and a repetitive mapping orbit are planned with relaxed science data requirements. The required navigation accuracy is in line with previous mission technology, and a special one-week period is set aside to collect two-way coherent Doppler data that will enable accurate determination of Mars' gravity field.

The approach to system design is to minimize the spacecraft-to-instrument operational interface, to maximize spacecraft autonomy, and to provide for sufficient subsystem performance margin.

Minimizing spacecraft-to-instrument operational interfaces is achieved as follows: the scientific instruments operate autonomously with respect to the spacecraft and are noninteractive with each other, the spacecraft provides only major mode changes and parameter updates to the instruments, each instrument has internal sequencing, and no complex validation process is required as is the case with highly interactive sequences.

As a result of the interplanetary communication delay time of up to 40 min and a total communication outage of up to 30 days during solar conjunction, the spacecraft is designed to operate autonomously, independent of ground control for long periods of time. By providing autonomous mode changes, onboard fault protection, and autonomous spacecraft subsystem operations, the ground operations costs are greatly reduced.

The Mars Observer spacecraft design has sufficient operating performance margin to minimize the competition for spacecraft resources such as power, data handling, and telecommunication capabilities. The minimized competition simplifies operation planning, sequencing, and analysis on ground.

Science Requirements

The basic Mars Observer mission objectives include geoscience measurements of the gravity and magnetic fields and climatology measurements of the Martian atmosphere and surface. The seven science instruments carried on the spacecraft[1] and their measurement objectives are summarized in Table 1.

The gamma-ray spectrometer (GRS) will measure the basic elements of the Martian surface with a 360-km spatial resolution and a spectral resolution between 0.61 and 1.22 keV. The GRS will be able to detect the character of cosmic gamma-ray bursts (200 keV–10 MeV) and neutrons (0–2.5 MeV) within this spectral range.

The magnetometer/electron reflectometer (MAG/ER) will detect global magnetic fields over a wide dynamic range (16–65 nT) and will detect the surface magnetic fields by measuring the distribution of the ambient electrons in the range of 1 eV–20 keV.

The Mars Observer camera (MOC) will map the surface and atmosphere of Mars with a resolution of 250 m/pixel at nadir and 2 km/pixel at limb. A higher resolution image (1.4 m/pixel) will also be available at sampling intervals.

The pressure modulator infrared radiometer (PMIRR), a nine-channel infrared radiometer in the 0.3- to 46.5-μm region, will determine the vertical profiles of temperature, water vapor, pressure, dust, and atmospheric condensates, employing high radiometric precision and pressure modulation techniques.

Radio science (RS) measurements will be performed to determine the atmosphere refractivity profiles within a 200-m resolution. The ultrasable oscillator (USO) utilized for the experiment transmits at 4.7-MHz frequency. The measurements will be performed via the spacecraft's X-band transponder.

The thermal emission spectrometer (TES) will measure the composition of Mars' surface and perform atmospheric observations. The infrared spectral measurements will be in the 6.25- to 50-μm region and the radiometric measurements in the 3.9- to 100-μm and 0.3- to 3-μm bands.

The Mars Observer laser altimeter (MOLA), consisting of a Nd:YAG laser and a telescope assembly, will output repetitious pulses to determine the global topography and geophysical and geological structures and processes on Mars.

The science instruments and their measurement characteristics specify the mapping orbit selection requirements[2] as follows.

The global mapping of mineralogical resources, the gravity field determination, the determination of distribution of volatile materials, and the mapping of the atmosphere structure and circulation require low-altitude orbits. The low-altitude orbit provides higher spatial resolution and improved signal-to-noise ratio.

The measurement of the seasonal variation of volatiles needs elimination of diurnal variations and nearly the same lighting angle and local time of day for each revolution. A near-Sun-synchronous orbit provides this feature.

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Table 1 Science instruments and measurement goals

Instruments and principal investigators	Measurements	Observations
Gamma-ray spectrometer W. Boyton, Univ. of Arizona	Surface elemental composition; seasonal variations in polar caps; atmospheric density; soil water content; spectrum and character of cosmic ray bursts	Resolution test in cruise, annealing if required (2–7 days) followed by detector cool down; calibration in drift; continuous observation, mapping
Magnetometer/ electron radiometer M. Acuna, Goddard SFC	Global magnetic field; surface magnetic features; Mars-solar wind interaction; energy spectrum and angular distribution of electrons	Calibration and limited operation during drift; continuous operation during mapping orbits
Mars Observer camera M. Malin, Arizona State Univ.	Measurement of surface and atmosphere of Mars; global monitoring; monitoring of polar cap formation	Possible health and status check and bakeout during cruise; continuous operation during mapping
Pressure modulator infrared radiometer D. McCleese, JPL	Vertical profiles of temperature, water vapor, pressure, dust, atmospheric condensates	Continuous operation during mapping; regular calibration during mapping
Radio science L. Tyler, JPL	Gravitational field: atmosphere refractivity profiles	During occulations when DSN available in mapping; test during cruise
Thermal emission spectrometer Ph. Christensen, Arizona State U.	Composition of Mars' surface; atmospheric observations: aerosal, pressure, dust storms, polar cap	Possible test during cruise; continuous operation during mapping
Mars Observer laser altimeter D. Smith, Goddard SFC	Topography; geophysical and geological structures and processes	Possible test, cruise; continuous ranging to the surface during mapping
Mars balloon relay J. Blamont, CNES, France	High-resolution surface data	Relay of data from Mars surface to spacecraft; extension of mapping phase

The near-circular polar orbit chosen for this mission permits uniform viewing and spatial resolution for all latitudes and longitudes. This feature facilitates comparison of measurements taken at different locations on Mars.

Repeating ground track is desirable with quick return times for most observations. This prevents navigational error buildup and eliminates the requirement for large swath overlaps. In addition, maintenance of a nearly constant altitude above the oblate Martian surface facilitates instrument design and data processing.

There are no regular science data collection requirements during the cruise phase of the mission. Limited calibration requirements are indicated in Table 1. Regular science data collection will begin in mapping orbit.

The nadir orientation of the spacecraft during the mapping phase of the mission supports continuous data collection. Twenty-four hours of continuous data recording and daily playback through the Deep Space Network (DSN) is planned. In addition, every third day, high-rate, real-time data will be returned.

The Mars Observer mission will be extended to accommodate the joint American, French, Russian experiment. The French-built Mars Balloon Relay (MBR) on the Mars Observer spacecraft will interrogate scientific equipment released on the Martian surface by a Soviet spacecraft and will transmit the data to Earth.

The earliest planned date for the first descent landing module deployment is Sept. 24, 1995. There will be no balloon deployment as originally planned. The Mars Observer spacecraft will support the experiments by receiving the 400- to 407-MHz signals from the experiments on the Martian surface and transmitting the data received to Earth. It will be determined on an orbit-by-orbit basis which spacecraft's (the Mars Observer or the Russian spacecraft) transmitter/receiver system best views the equipment on the surface of Mars. The MBR will most likely be the primary system because of the good ground coverage provided by the Mars Observer Sun-synchronous, low-altitude orbit.

Mission Description

Four mission phases are defined for the description of the various activities during the Mars Observer mission: launch, cruise, orbit insertion, and mapping. The mission scenario[3,4] is depicted in Fig. 1.

Launch

The Mars Observer spacecraft will be launched on a Titan III/Transfer Orbit Stage (TOS)[5] from the Cape Canaveral Air Force Station. The Titan III launch vehicle will deploy the mated TOS/spacecraft configuration in low Earth orbit, and the TOS will boost the spacecraft into Mars transfer trajectory.

The Titan III is a commercial launch vehicle derived from the Titan 34 series of boosters. It is a two-stage liquid core vehicle with two strap-on solid rocket motors (SRMs). The stage 0 SRMs generate an average of 12,420,000-N thrust. The lower core bipropellant stage (stage 1) generates a 2,429,000-N thrust. The upper core stage (stage 2) uses a single engine, with an average of 463,000-N thrust capability. The avionics of Titan III, located on stage 2, provide autonomous guidance/navigation and flight control functions. The payload, consisting of the Mars Observer spacecraft and the TOS, is located above stage 2 in the payload fairing.

The TOS upper stage is a three-axis-stabilized single-stage propellant vehicle with a ring-laser inertial guidance system. The spacecraft is mated to the TOS via an adapter in its launch configuration, as shown in Fig. 2. During launch, the solar panels and the high-gain antenna (HGA) are folded against the spacecraft's body, and the two science booms are retracted.

Prior to liftoff, the Titan III and TOS navigation systems are initialized and the spacecraft is powered. The spacecraft reaction wheels and gyros are turned on to prevent damage during ascent, and the spacecraft propulsion system and pyrotechnics are inhibited. The SRMs are ignited at liftoff. The booster performs a roll maneuver after liftoff to achieve the correct azimuth. After stage 1 ignition, the SRMs separate from the booster core. After stage 1 separation and stage 2 ignition, the payload fairing is jettisoned and the TOS telemetry system is enabled. The second stage burns until the desired elliptical parking orbit is achieved. The total time from liftoff to parking orbit injection is 900.5 s.

The joint spacecraft and the attached upper-stage TOS remain in parking orbit until injection into transfer orbit opportunity occurs. The parking orbit is less then 1 rev, and the length of time spent in the parking orbit depends on the time and day of the launch.

TOS injection burn is initiated between 31.4–38.1 min after liftoff. The TOS injection burn lasts about 2.5 min. During the burn, the TOS RF system transmits combined spacecraft and TOS data to an airborne range instrument aircraft (ARIA). The spacecraft also records these data for later transmission. A few minutes after TOS burnout, the TOS orients the spacecraft to a preferred attitude for DSN acquisition. The spacecraft separates from TOS at 53.5 min after liftoff. After separation, the TOS performs a 2-m/s maneuver to avoid collision and contamination of the spacecraft and is consecutively deactivated.

Immediately after separation from the TOS,[6] the solar array and the HGA are partially deployed in the cruise configuration, as shown in Fig. 3. Four of the solar panels are deployed on the $+Y$ face of the spacecraft. The HGA is partially deployed to have a clear field of view in the $+Y$ direction.

Initially, after separation from the TOS, the spacecraft is kept in an inertial hold mode for a maximum of 2 h to achieve initial acquisition by the DSN. A single separation attitude has been defined that has a favorable geometry relative to the Sun (for adequate spacecraft power) and the Earth [to permit communication with one low-gain antenna (LGA) for all days in the launch period]. The spacecraft will be in view of Canberra about 8.6–14.3 min after TOS burnout, facilitating early acquisition. The DSN provides lockup within 30 min after the spacecraft is in view. Initial acquisition is performed by a small wide-beam antenna that is capable of signal acquisition up to a few hours after injection. This antenna is used to point the 34-m narrow-beam DSN antennas for subsequent collection of two-way Doppler data. About 30 min of Doppler tracking data is collected to update the spacecraft ephemeris. Telemetry is downlinked to assess the health of the spacecraft and to verify deployment.

After initial acquisition has been completed, the spacecraft is commanded to start its attitude initialization sequence. Subsequently, through a programmed roll maneuver about the Sun line, the spacecraft establishes its three-axis attitude state vector. In this configuration, the celestial sensor assembly (CSA) acquires the star transit data necessary to update its inertial measurement unit (IMU) reference. After establishing a three-axis attitude reference, the roll maneuver is halted at a point where the spacecraft Y axis is in the Earth-spacecraft-Sun plane, and the rotation is then transferred to the Y axis.

Cruise

The cruise phase lasts for about 11 months and is divided into two subphases based on spacecraft pointing and telecommunication: the inner cruise and the outer cruise phase.

During the inner cruise phase of the mission (the first three to four months of the trans-Mars flight), the spacecraft is close enough to Earth so that full communication is available using the LGAs. During this phase, the solar arrays, which are sized for the much weaker solar flux at Mars, provide excess power if the face is oriented on the Sun. Therefore, the spacecraft will maintain an inertial attitude for the spacecraft $+Y$ axis with a predetermined angular offset from the Sun, generally in the direction of the Earth for good LGA coverage. The angular offset varies as a function of date and will be con-

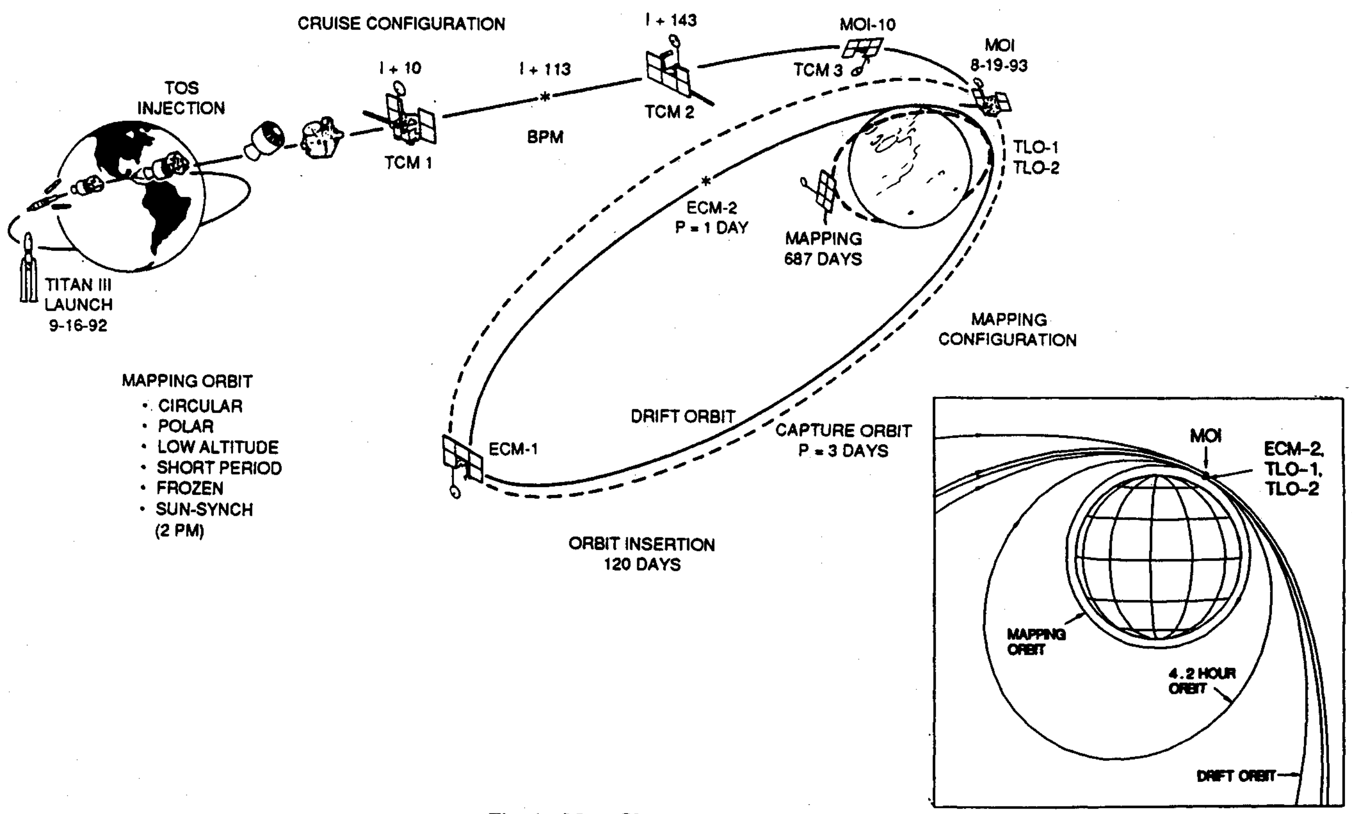

Fig. 1 Mars Observer mission scenario.

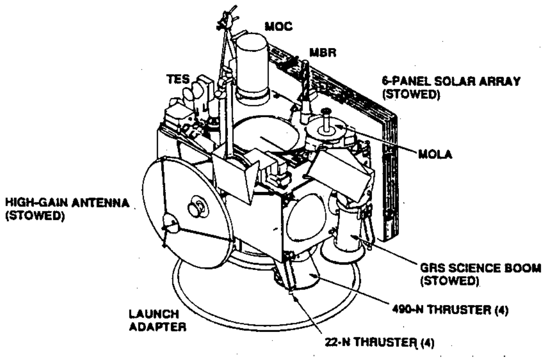

Fig. 2 Mars Observer spacecraft in launch configuration.

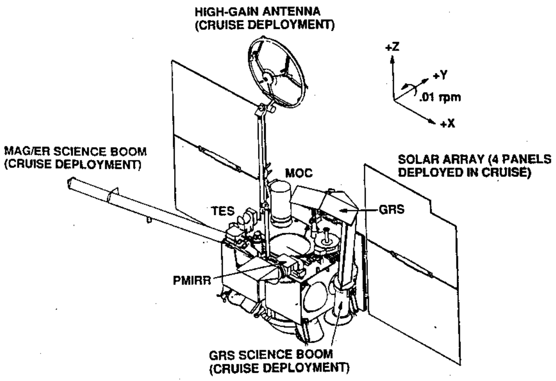

Fig. 3 Mars Observer spacecraft in cruise configuration.

strained between 65- to 30-deg Sun incidence angles. The selected $+Y$ axis vector within this range insures continuous Earth line-of-sight visibility from the primary low-gain receive antenna and the LGA transmit antenna. The star catalog will be updated every four weeks, or whenever a spacecraft attitude change is required for communications.

After communication with the LGA is no longer adequate to support a 250-bps telemetry rate, the spacecraft attains its outer cruise phase configuration. The spacecraft $+Y$ axis will be slewed into coincidence with the Earth line, with the HGA employed for normal communication. This will occur at the completion of the inner cruise phase, on Jan. 3, 1993.

During the outer cruise phase, the excess solar array input is no longer a problem and the HGA and the solar array are both Earth pointed; the Sun angle varies on the solar array. As a backup, the LGA link can always provide low bit-rate communication at any point in the mission.

Throughout the cruise phase, the normal operating mode of the spacecraft (other than during maneuver periods) will be the slow 0.01-rpm rotation about the spacecraft Y axis (the solar array normal). Since the slow rotation about the Y axis provides the scan motion for the CSA, the primary attitude mode in the cruise phase of the mission, the spacecraft can remain in this Earth-oriented attitude, except during ΔV maneuvers, until the mapping orbit is attained.

Following the autonomous launch phase, ground commanded activities will include changing battery charge rate to trickle charge, activating power redundancy management, and rotating the HGA to full cruise deployment. Partial deployment of the MAG/ER and GRS booms completes the spacecraft's configuration for cruise. The tape recorders will be repacked following pyroevents. After the full cruise configuration is achieved, there are no special checkout procedures required for the spacecraft. Engineering data will be received daily. Telemetry will be monitored for all subsystems, and subsystem trend analysis will be performed routinely. No special commanding for spacecraft operation (except for maneuvers) will be required.

Prior to the first bipropellant maneuver, the thruster heaters' operation will be verified, the bipropellant lines will be primed, and the bipropellant system will be pressurized. Celestial updates are not required during maneuvers. The attitude reference is derived entirely from the state vector, resident in the spacecraft's central processor unit at the initiation of the maneuver. The tape recorder operates during the maneuvers, and the recorded telemetry can be played back after a maneuver on command.

During the cruise phase, to save fuel, the heliocentric transfer trajectory includes a deterministic broken-plane maneuver (BPM).[7] The BPM is performed for the first four days of the 24-day injection period. Injection occurring after the fifth day utilizes only ballistic trajectories.

Trajectory correction maneuvers (TCMs) will be performed for correction of upper-stage injection errors, maneuver inaccuracies, and unmodeled forces. The first TCM, TCM-1, is performed 15 days after launch and corrects for injection errors. TCM-1 could be a fairly large maneuver, up to 90 m/s. TCM-2, TCM-3, and TCM-4 will be small maneuvers to make up for the execution error of the previous maneuver and to adjust for Mars arrival conditions, respectively.

Orbit Insertion

The most critical event during the entire maneuver sequence is the Mars Orbit insertion (MOI) maneuver. During all other maneuvers, if any anomaly is detected, such as a large attitude control error or a large difference between the measured and calculated Sun vector, the maneuver abort will be activated. The MOI, however, must occur at a precise time. During MOI, all abort criteria are disabled and the spacecraft system is provided with sufficient autonomous redundancy management capabilities to complete the maneuver. Automatic switching is provided to the backup bipropellant thrusters, and the backup computer will be capable of continuing the maneuver in the event of a computer error. In addition, limits will be set on the thruster timer to back up the accelerometer measured ΔV cutoff logic.

For the large finite burn maneuvers, the MOI and the two transfer-to-low-orbit maneuvers (TLOs), a steering profile of a constant rate about a fixed inertial axis was selected. The finite burn maneuvers were designed to minimize the maneuver gravity losses experienced by the spacecraft. During the maneuver burns, the spacecraft is oriented such that the solar array provides sufficient power to the spacecraft. The thrust vector is aligned with the spacecraft Z axis. The Z axis rotates about a specified pitch axis during the maneuver. The orientation of the X and Y axes of the spacecraft with respect to the pitch vector gives the desired solar orientation. For example, during each of the two TLO burns, the thrust vector turns about 22 deg. The initial and final orbits are coplanar.

According to present plans, MOI will place the spacecraft into a 72-h highly elliptical capture orbit, in which it stays for approximately 21 days. A drift orbit will follow with one-day orbital periodicity. The length of the drift orbit will vary between 63.6 and 42.7 days depending on the launch date. After a second drift period in a 4.2-h orbit lasting for 10 days, the spacecraft will be placed into a mapping orbit with a 2-h period.

The Earth-pointing cruise configuration will be maintained throughout the orbit insertion phase. Only limited science activity is planned during the drift orbit for the GRS and MAG/ER, as indicated in Table 1. For approximately two more weeks, the cruise configuration will be maintained in the mapping orbit to facilitate gravity calibration.

Mapping

Once the mapping orbit has been established and verified for accuracy, the solar array can be fully deployed (Fig. 4) and articulated to point at the Sun while the spacecraft's instrument panel is pointed to Mars. The GRS and MAG booms are fully extended. In the mapping configuration, the solar array tracks the Sun, the HGA tracks the Earth, and the nadir instrument panel remains pointed toward the Mars surface. In this configuration, the full science payload data acquisition and the communication link can be maintained.

After achieving the mapping orbit, the spacecraft and ground system will undergo a 10-day checkout period, until the spacecraft bus and the payload instruments are declared operational. Though the mapping orbit configuration has been attained at this point, an additional 30-day period is required to evaluate the gravity calibration data and to refine both the gravity model and the eccentricity of the orbit.

The mapping orbit will be a low-altitude, near-circular, near-polar Sun-synchronous orbit, with the day-side equatorial crossing at 2 p.m. mean solar time. The index altitude is 378 km, with the argument of periapsis and eccentricity chosen such that the altitude variations of the spacecraft are minimal. The mapping orbit will be controlled to a regular grid of equator crossing in Martian body fixed longitude.

A mapping cycle was selected based on the spacecraft completing 88 rev in the time that Mars completes seven rotations (Martian solar days: sols). The spacing between the consecutive ground tracks will be 28.6° westward. At consecutive mapping cycles, the ground tracks will not be repeated exactly. By adjusting the spacecraft's altitude slightly, the ground tracks will shift 58.6 km to the east for each repeat cycle. Ideally, if there would be no navigation uncertainties, this strategy would provide an overlapping buildup of coverage with equal ground track spacings of 3.1 km at the equator for a 550-sol supercycle.

The required accuracy of equator crossing relative to the nominal grid is ± 30 km. The spacecraft will provide equator crossing times maintained within a 2.5-s accuracy.

During mapping orbit, the spacecraft will be nadir oriented, using the Mars horizon sensor assembly (MHSA) and yaw gyrocompassing as the primary attitude control reference mode. The HGA will be articulated to point to Earth, and the solar array will be articulated to point to the Sun at a specified angle. The spacecraft will provide knowledge of inertial pointing for attitude reconstruction on the ground and will provide HGA pointing and stability for radio science experiments.

Early in the mapping phase, from Dec. 16, 1993 through Jan. 6, 1994, the Sun-Mars-Earth angle will be less than 2 deg, impairing command capabilities. The spacecraft will maintain its orbit autonomously during this period. Low-bit rate, limited real-time engineering telemetry may be possible when the DSN is in sight. No propulsive maneuvers are planned within two days of this period.

Navigation will provide weekly predictions of the spacecraft position over a 14-day period, with a 100-km downtrack, 8-km crosstrack, and 8-km radial accuracy, starting two weeks after exit from solar conjunction until the end of the mapping phase. An ephemeris will be generated and uplinked to the spacecraft each week to support the CSA backup attitude mode.

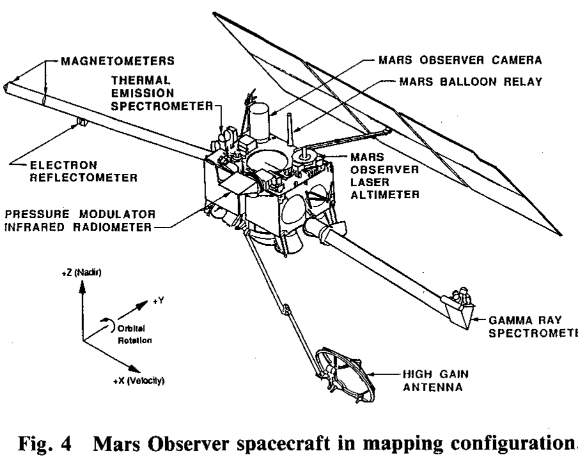

Fig. 4 Mars Observer spacecraft in mapping configuration.

Orbit trim maneuvers (OTMs) for the maintenance of the longitude grid will be performed biweekly and mostly will occur near periapsis, or occasionally near apoapsis, if required. Inclination correction maneuvers will occur at points far from periapsis or apoapsis and will be performed infrequently. The attitude control subsystem will provide active thruster control during these maneuvers. For the orbit adjust maneuvers, a specific yaw angle can be commanded prior to the maneuver.

The spacecraft's index altitude satisfies the NASA planetary protection requirements and complies with the international agreements not to contaminate Mars with terrestrial organisms. The spacecraft is required to survive in orbit until the year 2009 with a 99.9% probability and to survive thereafter until the year 2039 with a 95% probability. If systems resources permit, nothing precludes the spacecraft from continuing to operate beyond the planned mission.

Systems Design

Spacecraft Functional Design

The main function of the spacecraft is to provide a stable and controlled platform for the science instruments. The spacecraft's architecture is designed with an aggregate of the following subsystems: structure, cabling and harness, attitude and articulation control, electrical power supply, propulsion, thermal control, command and data handling, telecommunications, mechanisms, and flight software.

The structure subsystem supports all of the spacecraft bus components, the payload instruments, and the cabling manifold. The structure consists of the main rectangular bus and the various appendages such as the solar array, HGA, MAG, and GRS support booms, as shown in Fig. 4.

The harness subsystem provides the network for the electrical interconnects between the various components, boxes, and payloads on the spacecraft.

The attitude and articulation control subsystem (AACS) provides a three-axis-stabilized platform via closed-loop control, utilizing horizon sensor, star data, and ground uplinked ephemerides. Momentum management is maintained via hydrazine thrusters and reaction wheels.

The electrical power subsystem utilizes a photovoltaic solar cell array as the primary power source. During the launch and mapping phase, batteries supply additional power when no solar input is available. Shunt regulation controls excess power during the sunlit portions of the mission.

The propulsion subsystem supports the AACS for maneuvers and pointing while providing pyrotechnic capability for mechanism control. Interplanetary trajectory maneuvers are performed with bipropellant thrusters. Monopropellant thrusters are used for orbit control during mapping and pointing control during the cruise phase of the mission.

The thermal control subsystem provides active and passive control throughout the spacecraft with multilayered insulating blankets, paint, tape, active and passive heaters, and radiators. Local thermal control is provided to the science instruments, external cables, dampers, gimbal drives, and attitude control sensors.

The command and data handling subsystem provides the control functions of all spacecraft events, receives and transfers commands to the spacecraft components and the payloads, and manages and transfers telemetry data. Two programmable spacecraft computers provide for autonomous spacecraft operations, fault, and redundancy management.

The telecommunications subsystem provides telecommunications in X-band, wideband turnaround ranging, and one-way differential ranging. Various data bit rates from 250 bps (engineering data) to 40 ksps (combined science and engineering data) are accommodated to satisfy mission requirements. Emergency communications and recovery procedures are supplied in case the nominal link is out of service.

The mechanism subsystem provides for solar array deployments, MAG and GRS boom deployments, and HGA deployment. Gimbaled booms provide solar array and HGA phasing during the mapping phase of the mission.

The flight software subsystem provides for all of the functions associated with the flight computer, including autonomous operation, fault, and redundancy management. Degraded spacecraft modes, safe and emergency modes, are managed by the flight software subsystem. In addition, various subsystem and mission sequence management functions, such as attitude control, launch sequence management, etc., are implemented in flight software.

Autonomous Operation

The interplanetary communication delay time is up to 40 min roundtrip, and a total communication outage of up to 30 days can be expected during solar conjunction. The spacecraft is designed to operate independently of ground control during these communication outages. 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 acquisition, execution of preprogrammed ΔV maneuvers, and component fault detection and replacement.

The mapping phase attitude control modes, primary and backup, are entered autonomously, with ground enabled preferencing. The primary mode uses IMU gyrorates and MHSA-derived roll/pitch measurements to provide yaw-gyrocompassing control to the local Mars-oriented coordinate frame, using reaction wheel assembly (RWA) actuation. The spacecraft also provides autonomous HGA and solar array pointing once in Mars' orbit.

By making the mapping orbit ephemeris available onboard, two convenient backup modes are available. The CSA backup mode can provide Mars pointing in the case of a double failure of the MHSA or should better pointing stability be desired (such as during the seasonal dust storm periods when MHSA performance degradation is expected).

The onboard ephemeris is satisfactory for two weeks; however, due to the one-week delay in generating the ephemeris on ground, the mapping orbit ephemeris will be updated weekly. The star catalog, as in cruise orbit, will be updated every four weeks. No other routine ground activities are needed by the spacecraft, except for special OTM commanding and routine monitoring of the spacecraft's health and status.

The spacecraft is capable of autonomously entering a contingency mode whenever inertial knowledge onboard the spacecraft is lost. In this mode, the instrument shutters are autonomously closed and the spacecraft's returns to a Sun acquisition attitude control mode. The spacecraft will autonomously activate transmitter operations during the sunlit portion of the mapping orbit and during ingress and egress from eclipse.

Autonomous Sun avoidance will protect the MOC and MOLA payload instruments during the mission.

Fault Protection

In the Mars Observer spacecraft system design,[8] redundancy is employed to avoid single-point mission critical failures. Both functionally redundant and block redundant assemblies or units are used. The functionally redundant assemblies perform the same function using different hardware or software implementation. Block redundant assemblies or units perform the same functions using identical parts. Redundancy management is provided either autonomously onboard the spacecraft or by ground command. Autonomous onboard redundancy management is provided for the critical spacecraft bus functions. It is possible to override or disable autonomous onboard redundancy switching via ground command.

As a primary mission protection, the spacecraft is also capable to operate in degraded modes in the event of limited levels of onboard failure. In the degraded mode of operation, the primary scientific objectives can still be met but at the expense of loss of some scientific data. Some degraded operations can be achieved autonomously onboard the spacecraft, others result in an increase of mission operation complexity.

Degraded functional modes of operations are provided when nonredundant hardware functions exist and in the attitude control modes during mapping. Degraded operations are provided for the power system in case of partial solar array, battery cell, shunt failure, and the failure of battery temperature, charge and discharge sensors. The thermal subsystem is capable of accommodating limited temperature out of limits for equipment with variable duty cycles, and the command and data handling system can function with two or three tape unit failures.

In the event of major failures not handled by the redundancy management operations or degraded modes, the spacecraft will autonomously maintain, for at least 40 h, the minimum functions required for safe system operations, including payload protection from abnormal spacecraft states or attitudes. The safe mode will be entered if both onboard control processors fail their health monitoring check process, or the redundancy management software fails to provide a minimum stable system configuration, or the spacecraft receives a ground command to make the transition to safe mode.

In general, mode transitions follow a definitive orderly sequence of events required to achieve the transition between modes. All permissible transitions are achievable by ground commanding. Autonomous transition to safe mode will be inhibitable by ground command. In the event of an onboard failure that disrupts normal ground-to-spacecraft communications, the spacecraft provides autonomous onboard action that makes it receptive to low-rate commands.

Data Acquisition and Return

Mars Observer is the first deep space planetary mission that fully utilizes a distributed operations concept for data acquisition.

The repetitive science observation opportunities eliminated the need for feedback from data analysis to mission analysis—except in the case of atmospheric and seasonal variation measurements—which are the prime objectives for the Mars Observer mission. Mission operations are generally nonadaptive, thus reducing the complexity of the data acquisition events.

The fixed mounting of the payloads on the nadir-pointing platform provides nearly independent control for the science instruments. There is no articulating platform shared by instruments that would limit observation opportunities. Instrument commanding in general does not require inter-instrument coordination, and the spacecraft's command system is invisible to commands addressed to the instruments.

Consequently, the instruments can operate autonomously and can sequence themselves with minimum information from the spacecraft bus. The spacecraft bus function becomes decoupled from payload operation, and the scientists can interact directly (via data terminals in their home institutions) with the command uplink process. Science instruments will be controlled primarily by such noninteractive real-time commands.

The key features of the uplink process are based on reduced operation complexity. The bulk of the commanding is planned to be performed via stored sequence command loads and noninteractive real-time commanding. The anticipated interactive command loads are minimal, only 5% of the total.

Stored sequence command loads for the spacecraft will be constructed for 28-day intervals and will be uplinked every 3.5 days. In addition, every fourth uplink will include 2100 bytes of ephemeris data, and every eighth uplink will include 4300 bytes of star catalog data. The noninteractive real-time commanding will consist of about 4900 bytes per day, the largest portion of which is utilized for the control of the MOC.

The downlink strategy is based on relaxed data recovery requirements. Only 85% of data delivered by the science instruments needs to be recovered (it is expected that 95% will be actually recovered). This relaxed strategy leads to a flexible, less costly response to faults affecting the downlink process. A novel feature of the downlink process is that the data are distributed to the science investigators in the packet format produced by the instruments.

The science instruments use dedicated microprocessor-based techniques for data compression and buffering. Packet telemetry, along with the distributed character of the flight data system and the capability to buffer science data, eliminates coupling between the data rate of each science instrument's operating modes and the output data rate of the spacecraft's data handling system.

The downlink data rate is constant for extended periods of the mission, determined by the telecommunications link capabilities. Mainly the tracking coverage of the DSN limits the uplink and downlink activity.

During the prelaunch phase, any function capable of being updated by the ground or by the launch vehicle will be verifiable by telemetry. The spacecraft will be capable of recording and subsequently downlinking the upper-stage telemetry stream. The spacecraft will record TOS telemetry from liftoff of the Titan III through spacecraft/TOS separation. Spacecraft telemetry will be separately recorded from liftoff through spacecraft separation, deployment, and achievement of the inner cruise mode.

During the cruise and orbit insertion phases, the spacecraft will gather and return data sufficient to characterize the performance and health of the spacecraft bus. The planned periods of tape recorder operation are during all propulsive maneuvers, anomaly resolution activities, and routine testing of the tape recorders. The spacecraft will gather and return data to sufficiently characterize the results of payload calibrations. All data can be returned either in a real-time mode or non-real-time mode.

During the mapping phase of the mission, the spacecraft will continuously gather and store science and engineering data. The spacecraft will transmit each day, during a single DSN pass, 24 h of recorded science and engineering data. This data will be downlinked during 4.5 h on a noncontinuous, link available basis. Every third day, there will be an additional path available to downlink real-time science and engineering data during a 4.5-h ground station contact period.

Summary

Mission and systems design were discussed for the 1992 Mars Observer mission. The mission and system design is aimed at a low-complexity mission. Spacecraft system design facilitates autonomous operations, thus reducing ground activities. Science data gathering can be achieved on a continuous basis during the two-year mapping mission.

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