Mars Orbiter Mission spacecraft and its challenges

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Mars Orbiter Mission (MOM), India's first interplanetary mission, was launched on 5 November 2013 from the Satish Dhawan Space Centre (SDSC), Sriharikota, using PSLV XL C-25 rocket. MOM spacecraft was successfully inserted into the Martian orbit on 24 September 2014. One of the main objectives of the mission to Mars is to develop the technologies required for design, planning, management and operation of an interplanetary mission. Scientific objectives include exploration of surface features of Mars, morphology, topography, mineralogy and study of Martian atmosphere using indigenous scientific instruments. The spacecraft configuration is a balanced mix of design from flight-proven IRS/INSAT/Chandrayaan-1 bus. The configuration and subsequent design of the spacecraft had to take into consideration the many challenges it would face during its mission life. The spacecraft has been designed for interplanetary missions capable of operating in Earth burn, Mars transfer trajectory and Martian orbit environments. The major challenges of design are in thermal environment, radiation environment, power systems, communication systems, propulsion systems and on-board autonomy.

Keywords: Challenges, interplanetary mission, Mars Orbiter, spacecraft configuration.

Introduction

EXPLORATION of space to advance our knowledge of the Universe we live in, has always been an important component of the space science programme of the Indian Space Research Organisation (ISRO). India's solar system exploration programme was initiated in a major way in October 2008 with the launch of Chandrayaan-1, the first lunar mission of India. Nationally and internationally, Chandrayaan-1 generated unprecedented enthusiasm. In particular, it created excitement in the young minds of the country and firmly established that India can take on technological challenges. It established a sense of confidence in our ability to address new frontiers of space exploration. After the first lunar mission, ISRO is working on Chandrayaan-2, an indigenous lunar landing mission.

Next to Moon, in Earth's vicinity, Mars is a natural target of study in the solar system exploration programme. Of all the planets in the solar system, Mars has evoked the great human interest. Its orbit lies between the asteroid belt and the Earth. For ages humans have been speculating about life on Mars. The conditions on Mars are believed to be hospitable since the planet is similar to Earth in many ways. Like Earth, it has an atmosphere (though less dense and different in composition), water, ice and geology which interact with each other to produce the dynamic Martian environment. Mars has surface features reminiscent of the impact craters of the Moon as well as volcanoes, deserts and polar ice on the Earth. But, the question that is to be answered is whether Mars has a biosphere or ever had an environment in which life could have evolved and sustained.

Mars with its many similarities to Earth, is an important planet to understand the origin and evolution of the solar system, and in the not-so-distant future, will be the most probable candidate for human exploration. The carbon dioxide-rich atmosphere, absence of liquid water on the surface and absence of a protective global magnetic field are not perceived as deterrents for human settlement on Mars in a few decades from now. India certainly cannot afford to lag behind in its independent exploration of the red planet. With this in view, the Indian Mars Orbiter Mission (MOM) was undertaken primarily to demonstrate the prowess to venture into interplanetary space. This article provides details of the Mars Orbiter spacecraft configuration and the mission operations carried out.

Mission objectives

One of the main objectives of the first Indian mission to Mars is to develop the technologies for design, planning, management and operation of an interplanetary mission. Following are the major objective of the mission.

Technological objectives

- Design and realization of a Mars Orbiter with a capability to survive and perform Earth-bound manoeuvres, cruise phase of 300 days of travel, Mars orbit insertion/capture and on-orbit phase around Mars.
- Deep space communication, navigation, mission planning and management.

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- Incorporate autonomous features to handle contingency situations.

Scientific objectives

- Exploration of the surface features of Mars, morphology, topography, mineralogy and Martian atmosphere using indigenous developed scientific instruments.

Launch

The MOM spacecraft was launched on 5 November 2013, by ISRO's work-horse rocket PSLV in its extended form PSLV-XL, designated as PSLV-C25 mission, from the Satish Dhawan Space Centre, Sriharikota. The launcher injected the MOM spacecraft in 248 × 23,550 km elliptical orbit. Figure 1 shows a photograph of the launch.

Table 1 provides the realized parameters of MOM launch.

Spacecraft configuration

Mainframe elements

The spacecraft configuration is a balanced mix of design from the flight-proven IRS/INSAT/Chandrayaan-1 bus (Figure 2). Modifications required for Mars mission are in the areas of communication, power, propulsion systems (mainly related to liquid engine restart after nearly 10 months) and on-board autonomy.

Figure 1. MOM launch.

Table 1. Realized orbital parameters

Spacecraft mass	1337 kg
Launch date	5 November 2013
Lift-off time	14:38:26 IST
Separation time	15:22:43 IST
Perigee	248 km
Apogee	23,550 km
Inclination	19.27 degrees
Orbital period	6.831 h
Argument of perigee (AOP)	284.4 degrees

Table 2 lists the salient features of the space segment elements.

Mass budget

The spacecraft had to be configured with very tight mass budget of 1350 kg allocation. Excellent mass control and optimization measures effected the realization of the MOM spacecraft with a final mass of 1337.2 kg. Table 3 gives details of the mass of the spacecraft systems.

Major challenges

The configuration and subsequent design of the spacecraft had to take into consideration the many challenges it would face during its mission life. Some of the major challenges for which the MOM spacecraft systems/elements were designed are detailed in the following subsections.

Thermal environment

The spacecraft needs to cope with a wide range of thermal environment, from near Earth conditions with heat contributions from the Sun and the Earth (hot case conditions) to Mars conditions with eventual eclipses and reduced solar flux (cold case conditions).

The average solar flux in the Mars orbit is 589 W/sq. m, or about 42% of what is experienced by an Earth-orbiting spacecraft. As a result of the eccentricity of the Mars orbit, the solar flux around Mars varies by about 19% during the Martian year, which is considerably more than 3.5% variation near the Earth.

Albedo fractions are similar to that of the Earth, being around 0.25–0.28 at the equator and generally increasing toward the poles (Table 4). The effect of albedo and Mars IR loads are negligible due to eccentricity of the orbit around Mars (Figure 3).

Mitigation

The challenges posed by varying thermal conditions in various phases of the mission were mitigated to a large extent by adopting different attitudes shown (Figure 4).

Radiation environment

The MOM main frame bus elements and payloads are basically designed for interplanetary missions capable of operating in Earth-bound orbits (EBO), Mars transfer trajectory (MTT) and Martian orbit (MO) environments. The spacecraft bus components were selected with respect to accumulated dose of 6 krad for aluminium shielding of standard gauge 22 or below. Parts have been

Table 2. Salient features of MOM spacecraft

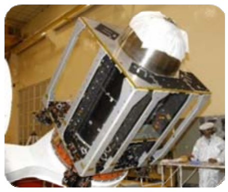
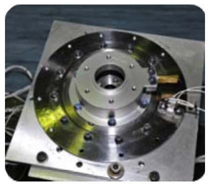
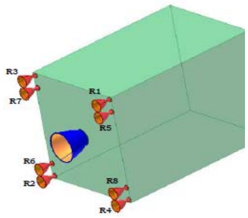
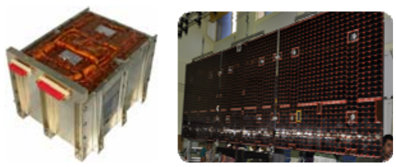
Feature	Specification	Photograph
Mass	1340 kg (upper limit + 0 kg and lower limit –3 kg from the nominal value of 130 kg)	
Structure	Aluminum and composite fibre reinforced plastic (CFRP) sandwich construction- modified I-1 K bus	 MOM primary structure with tank assembled
Mechanism	Solar panel drive mechanism (SPDM), reflector and solar panel deployment	 SPDM
Propulsion	Bi propellant system (MMH + N_2O_4) with additional safety and redundancy features for Mars Orbit Insation (MOI)	 Thruster configuration
Thermal system	Passive thermal control system	
Power system	Single solar array: 1400 × 1800 – three panels – 840 W generation (in Martian orbit); battery: 36AH Li-ion	 Battery and solar array
Attitude and orbit control system	AOCE (attitude and orbit control electronics): with MAR31750 processor Sensors: Star sensor (2 Nos), solar panel sun sensor (SPSS)-1 No, course analogue sun sensor (CASS)-9 heads, inertial reference unit and accelerometer package (IRAP) – 1 No. Actuators: reaction wheels (5 Nms capacity, 4 Nos), thrusters (22 N, 8 Nos), 440 N liquid engine	 Reaction wheels
TTC baseband and RF system	Telemetry (TM) and telecommand (TC): CCSDS compatible Baseband data handling (BDH) and solid state recorder (SSR): –16 + 16 Gb Communication (RF) systems: S-band for both TTC and data	 TTC package
TMTC package	Antennae: low gain antenna (LGA), mid gain antenna (MGA) and high gain antenna (HGA)	 Receiver

Figure 2. Spacecraft in stowed, deployed and launch configurations.

Table 3. Mass budget

Subsystem	Mass (kg)
Payloads	13.4
Power systems	58.2
TTC baseband systems	14.5
AOCS elements	60.3
Propulsion systems	87.3
Data handling system	6.0
RF systems	61.7
Thermal control systems	29.2
Spacecraft mechanisms	17.1
Structures	98.4
Assembly integration and testing elements	36.4
Total dry mass of the spacecraft	482.5
Pressurant	2.7
Propellant	852.0
Wet mass (lift-off mass) of S/C	1337.2

Table 4. Thermal environment

Entity	Perihelion	Aphelion	Mean
Direct solar (W/sq. m)	717	493	589
Albedo (sub-solar peak)	0.29	0.29	0.29
Planetary IR			
Maximum (W/m^2; near sub-solar)	417	315	390
Minimum (W/m^2; polar caps)	30	30	30

considered as directly suitable, if they have been evaluated successfully up to 12 krad (margin factor of 2).

Bus parts are capable of withstanding radiation events with a minimum threshold value linear energy transfer (LET): for single-event upset (SEU), LET > 40 $MeV.cm^2.mg^{-1}$ and for single-event latch ups (SEL), LET > 80 $MeV.cm^2.mg^{-1}$.

Power system

The power system has to support the spacecraft and payloads during various phases of the mission namely; launch phase, geocentric, heliocentric and Martian orbit phase. The power system consists of power generation, energy storage and power conditioning elements.

One of the major challenges in the design of the power system is power generation under low solar intensity due to the larger distance of the satellite from the Sun. The power generation in Mars orbit is reduced to nearly 50%–35% compared to Earth's orbit. Due to the eccentricity of Mars orbit, the power generation variation is nearly 15% in a Martian year. For example, if 1 W is the power generation in Earth's orbit, 0.35 W will be the power generation when Mars is at aphelion.

The power bus configuration comprises of a single wing of solar array with 7.56 m^2 area generating about 840 W when sunlit with normal incidence in Martian orbit, and it is a battery-tied single bus of 28–42 V bus.

The power requirement of various elements of the MOM spacecraft during different phases of the mission is listed in Table 5.

Solar array

The power source for the MOM is its single solar array wing having three solar panels which converts solar energy into electrical energy by photo-voltaic conversion. When the spacecraft is sunlit, the solar cells convert sunlight to electrical power to operate the electrical loads and charge the battery. Design of the solar panels of MOM is similar to the earlier IRS missions and according to standard space array design procedure. The main change in the design of the MOM solar panels with respect to earlier IRS mission is optimized solar array design to maximize power generation in the Martian orbit phase when the temperature of the panels would be low, while meeting the spacecraft load requirements during Earth-bound and heliocentric (cruise) phases when the temperature would be high.

Compared to Earth, Mars is away from the Sun and the distance between Mars and Sun varies from 206

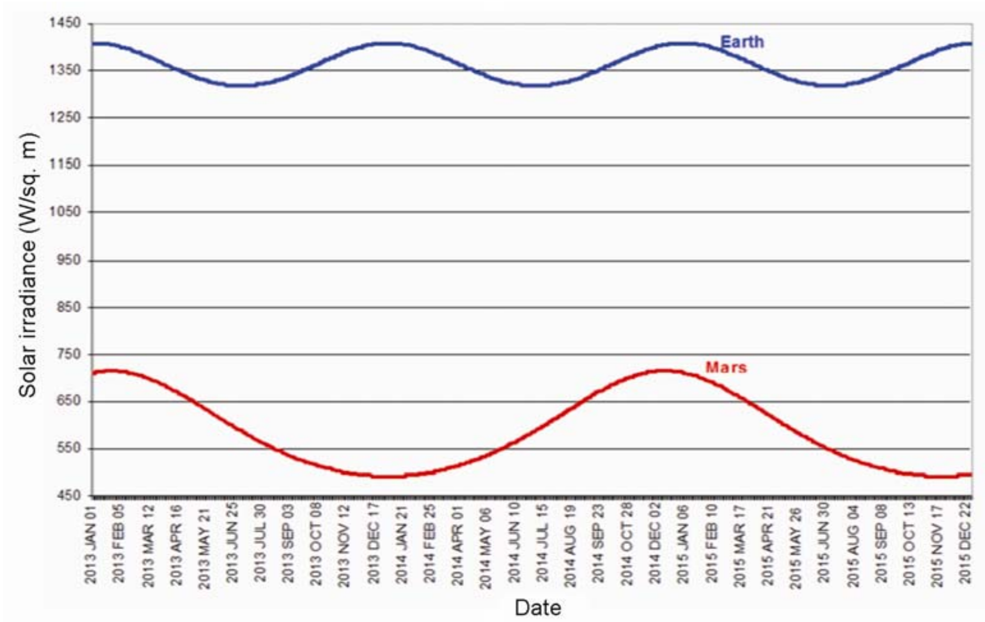


Figure 3. Solar irradiance variation.

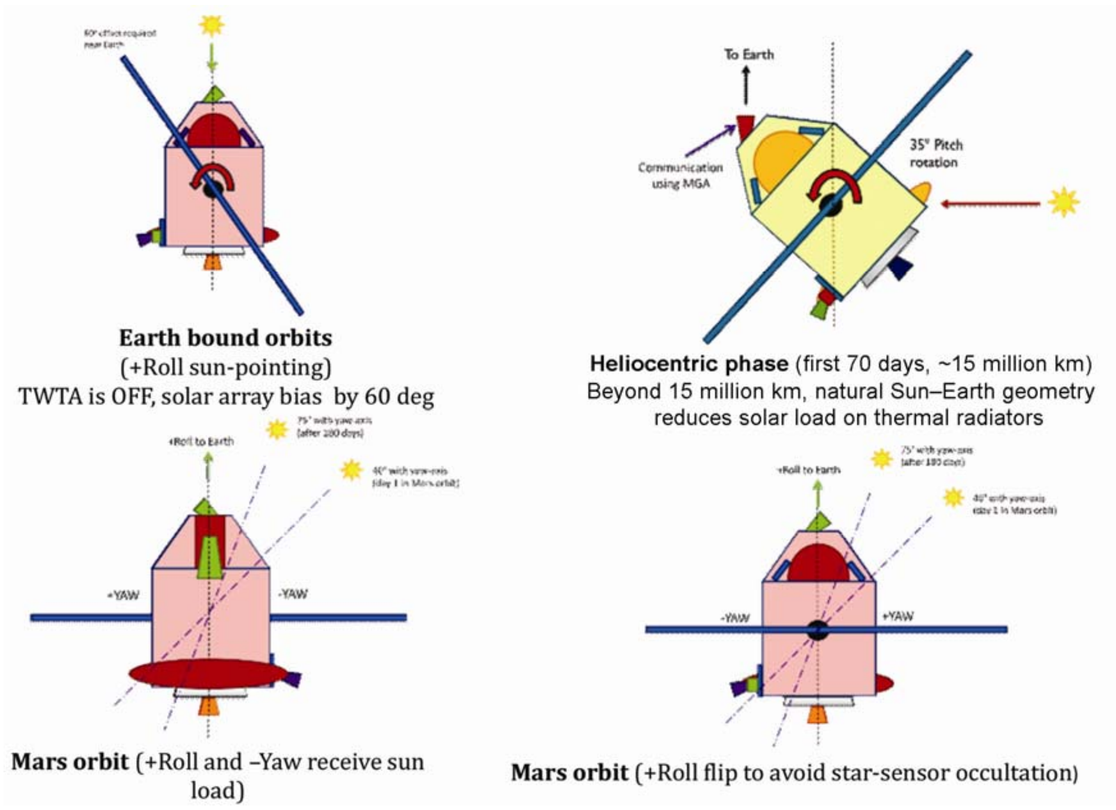


Figure 4. Spacecraft attitudes for thermal management.

(1.38 A.U.) to 249 (1.5 A.U.) million km. Due to this variation, the solar irradiance near Mars varies from 715 to 470 W/m^2 and solar cell operating temperatures vary from 2°C to –20°C. Figure 4 shows the solar irradiance variation in Earth and Mars orbits.

Solar array is a rigid, deployable, Sun-tracking, single-winged array (Figure 5). It consists of three solar panels of size 1.8 m × 1.4 m with the state-of-the-art triple junction solar cells having beginning of life (BOL) efficiency of 28.5% and capable of supporting the spacecraft load

Table 5. Power consumption of different subsystems of MOM

Subsystem	Launch pad	Before deployment	Earth phase	Heliocentric	MOI	Sunlit Image	Sunlit Download	Eclipse	Duty cycle (%) of Mars
Payload systems									
LAP	0.00	0.00	0.00	0.00	0.00	7.95	0.00	0.00	
MSM	0.00	0.00	0.00	0.00	0.00	6.17	0.00	0.00	
MENCA	0.00	0.00	0.00	0.00	0.00	28.80	0.00	0.00	
MCC	0.00	0.00	0.00	0.00	0.00	3.12	0.00	0.00	
TIS	0.00	0.00	0.00	0.00	0.00	5.88	0.00	0.00	
Power									
Distribution	0.00	0.00	10.50	16.50	18.80	17.00	18.80	0.00	100
Power control	5.00	5.00	5.00	5.00	5.00	5.00	5.00	5.00	100
Attitude and orbit control system									
AOCE-1	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	100
AOCE-2	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	100
NIN-10	8.66	8.66	8.66	8.66	8.66	8.66	8.66	8.66	100
Star sensor	0.00	0.00	11.00	11.00	24.40	11.00	11.00	11.00	100
IISU									
IRAP	54.40	54.40	38.60	38.60	54.40	31.00	31.00	31.00	100
Wheel & WDE-10	0.00	0.00	12.81	12.81	12.81	12.81	12.81	12.81	100
Wheel & WDE-20	0.00	0.00	12.81	12.81	12.81	12.81	12.81	12.81	100
Wheel & WDE-30	0.00	0.00	12.81	12.81	12.81	12.81	12.81	12.81	100
Wheel & WDE-40	0.00	0.00	12.81	12.81	12.81	12.81	12.81	12.81	100
Data handling system									
BDH & SSR	13.69	13.69	13.69	13.69	13.69	13.69	13.69	13.69	100
TMTC-10	23.55	23.55	23.55	23.55	23.55	23.55	23.55	23.55	100
TMTC-20	23.72	23.72	23.72	23.72	23.72	23.72	23.72	23.72	100
RF elements									
TWTA-1	0.00	0.00	0.00	440.00	440.0	0.00	440.00	0.00	
S-Band TM TX	0.00	28.80	59.50	28.80	28.80	0.00	28.80	0.00	100
S-Band RX	24.52	24.52	24.52	24.52	24.52	24.52	24.52	24.52	100
Thermal elements	25.00	100.00	166.00	150.00	150.0	145.00	141.00	205.00	100
Total subsystems power	196.94	300.74	454.38	853.68	885.18	424.69	839.38	415.78	

During MOI, TWTA is ON for initial 6 min and OFF due to occultation. For this initial 6 min, total load = 992 W. After 6 min, total load is 552 W till MOI is completed.

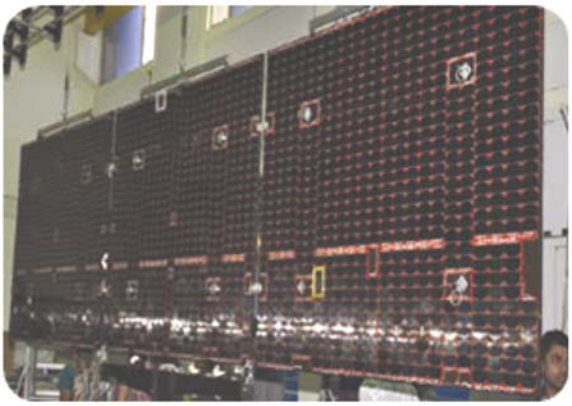


Figure 5. Solar array wing assembly.

Figure 6. 36 Ah lithium-ion battery.

and battery charging requirements in Earth-bound, heliocentric and Martian orbit phases.

Battery

A single 36 Ah lithium-ion battery in series–parallel (10S–24P) configuration is used, consisting of 18650HC type cells (Figure 6). The battery of MOM derive its heritage from Chandrayaan-1. The spacecraft loads are fully supported by battery during the major events like launch phase, initial attitude acquisition, eclipse, liquid engine burns near the Earth (EBN), Mars orbit insertion (MOI), safe mode and data transmission phases. Battery charging is done with the help of taper charge regulator (TCR).

Table 6. MOM antennae and their coverages

Antenna	Beam width (°)	Peak gain (dB)	Polarization	TM support (km)/ margin (dB)	TC support (km)/ margin (dB)
LGA	± 90	0	Left circular polarization (LCP) and right circular polarization (RCP)	1.4 million/2.4	30 million/2
MGA	± 40	7 3 dB @ ± 40°	RCP	40 million/2.4 (with MGA peak pointing to Earth)	110 million/2.3 (with MGA peak pointing to Earth)
HGA	± 2	31	RCP	400 million/5.4	400 million/10

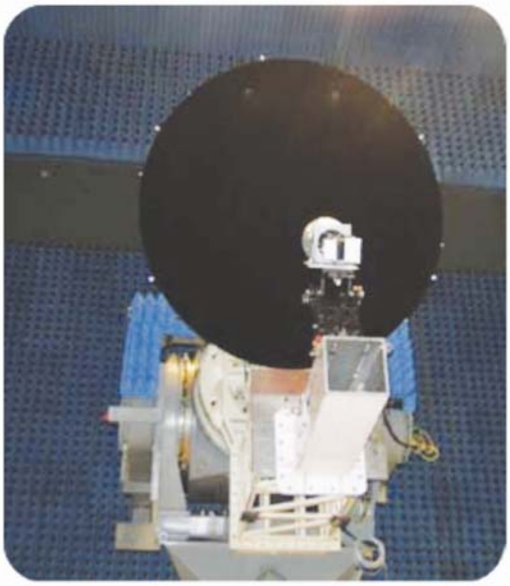


Figure 7. HGA in high power testing.

Battery is protected against over-discharge with the help of hardware emergency logic. In case of battery emergency, the protection logic opens the emergency (EM) relays to disconnect the battery from the power bus. However, S-band receiver, telemetry and telecommand (TTC), LAM supply and electro-explosive-device package (EED) are connected to the pole of the battery supply relay (K-relay), located before the emergency relay and hence continue to get power even after battery emergency.

Battery is fully charged before each discharge (at launch, near EBN, near earth eclipses, MOI, Martian eclipses).

Communication systems

The communication system for the Mars mission are responsible for the challenging task of communication management at a distance of nearly 200–400 million km. It consists of TTC systems and data transmission systems in S-band and a Δ-DOR transmitter for ranging. The TTC system comprises of coherent TTC transponders, TWTAs (travelling wave tube amplifiers), a near omni coverage antenna system, a high-gain antenna (HGA) system, medium gain antenna (MGA) and corresponding feed networks. The high-gain antenna system is based on a single 2.2 m reflector illuminated by a feed at S-band.

Antenna system

The antenna system consists of low-gain antenna (LGA), medium-gain antenna and high-gain antenna.

LGA consists two pairs of hemispherical coverage antennas mounted suitably on the spacecraft. Near spherical radiation coverage is obtained by placing two hemispherical coverage antennas with orthogonal circular polarization.

MGA with half power beam width of ±40° is designed for the MOM mission and this antenna is used to support TTC up to the injection of the spacecraft into the Mars orbit. MGA is used in case HGA loses its RF link due to reorientation of the spacecraft and during recovery modes.

In the Mars orbit, very high gain antenna system is required to transmit/receive the TTC or data to/from the Indian Deep Space Network (IDSN). Offset reflector geometry with 2.2 m diameter has been chosen for this application. Conical horn antenna with in-built septum polarizer is configured as feed for this reflector system. The dual circularly polarized feed enables this antenna to cater to both transmit and receive functions of the TTC system. Data transmission is also planned using HGA. Half power beam width of ±2° and gain of 31 dB are achieved for the HGA antenna system. The high power testing of HGA is shown in Figure 7.

Table 6 shows the antenna support capability in terms of distances with respect to 32 m IDSN station with 20 kW uplink.

Delta differential one-way ranging

A Δ-DOR package is employed to generate ranging tones for Δ-DOR measurement. Δ-DOR measurement is used to improve the orbit determination accuracy. It is incorporated as a part of RF system as Δ-DOR tones can be

Table 7. Payloads of MOM

Payload	Primary objective	Photograph
Lyman Alpha Photometer	Study of escape processes of the Martian upper atmosphere through deuterium/hydrogen	
Methane Sensor for MARS	Detection of methane	
Martian Exospheric Composition Explorer	Study of the neutral composition of Martian upper atmosphere	
MARS Colour Camera	Optical imaging	
TIR Imaging Spectrometer	Map surface composition and mineralogy	

downlinked along with TM data. This configuration allows to downlink TM data using IDSN, while Δ-DOR session is being carried out with JPL stations.

Propulsion systems

Propulsion systems embody the truly enabling technology for departing from the Earth and reaching Mars. The MOM propulsion system derives its heritage from the GEO mission and consists of a unified bipropellant system for orbit raising and attitude control. It consists of one 440 N liquid engine (LE-440) and eight numbers of 22 N thrusters. The propellants are stored in titanium propellant tanks each with a capacity of 390 litres. The tanks have combined storage capacity up to 852 kg of propellant. The 67 litre helium pressurant tank is used to pressurize the propellant. The 22 N thrusters are used for attitude control during various activities of the mission like orbit raising using liquid engine, attitude maintenance, Martian orbit maintenance (if any) and momentum dumping. As the critical operation of Martian Orbit Insertion (MOI) with liquid engine burn occurs after 10 months of the launch, suitable isolation techniques were adopted to prevent fuel/oxidizer migration issues.

Similar to conventional GEOSAT missions, the main engine was planned to be isolated after the Earth-bound liquid engine operations are completed. Liquid engine was isolated by operating pyro valves. On completion of cruise phase, the pyro valves were commanded OPEN, and propellant supply to liquid engine was re-established for MOI manoeuvres.

Since both the pressure regulators were isolated due to observed pressure rise in the propellant tanks during the cruise phase, MOI was carried out in the blow-down

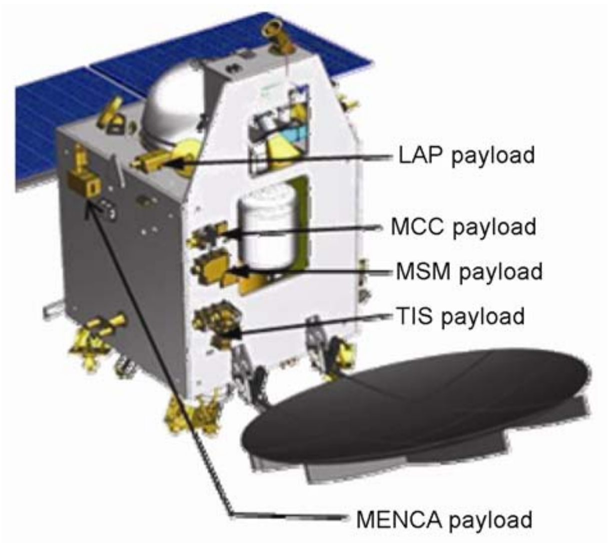


Figure 8. Payloads of MOM.

mode (i.e. the propellant was driven out of the tanks by the pressurant gas already existing in the tank ullages and reduction in the ullage pressure was not compensated by replenishment from the pressurant tank). Several ground simulation tests were conducted to simulate the on-orbit flight conditions of the propellant system. After satisfactory reviews, MOI was carried out in blow-down mode.

On-board autonomy

On-board autonomy refers to the capacity of the orbiter to make its own decisions about its actions. As the distance between the Mars Orbiter and Earth increases, the need for autonomy increases dramatically.

Given an average round-trip time (to and fro) from Earth to Mars of approximately 28.4 min (8.3–43 min), it would be impractical to micromanage a mission from the Earth. Due to this communications delay, mission-support personnel on Earth cannot easily monitor and control all the spacecraft systems in real time. Therefore, it is configured to use on-board autonomy to automatically manage the nominal and non-nominal scenarios on-board the spacecraft.

Autonomy logics manage the spacecraft when communication interruptions occur under following conditions:

- The spacecraft is occulted by Mars.
- Whiteouts/blackouts occur due to the Sun.
- Spacecraft enters safe-mode

Payloads

In order to meet the scientific objectives of the mission for exploration of the surface features of Mars, morphology, topography, mineralogy and Martian atmosphere, the Advisory Committee for Space Science (ADCOS), ISRO has selected five payloads for performing these studies (Table 7 and Figure 8).

Conclusion

The configuration and design of MOM spacecraft have worked perfectly well throughout all the phases of the mission. The excellent working of all the systems of the spacecraft has established the deep space mission heritage for these systems and the bus. The configuration and design of these systems/elements can also be adopted future interplanetary missions of ISRO.

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