INVITED PAPER

INDIAN INTERPLANETARY MISSIONS - A TECHNOLOGY PERSPECTIVE

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Abstract

The successful insertion of the Indian Mars Orbiter Mission in September 2014 in its maiden attempt along with the preceding Chandrayaan-1 mission has provided a huge boost to the global perception of Indias technological prowess. This technological achievement is built upon the decades of ISRO's experience in mission design, launch and operation of earth observation and communication spacecraft. The main driving factors for the Indian efforts in the direction of lunar and interplanetary missions were the optimal utilization of an existing launch system (PSLV) to achieve minimum energy orbit placement around the Moon and Mars with an innovative, highly elliptic orbit. An exposition of the mission planning, payload design, strategy and the technological features along with the differences with respect to the earth-orbiting missions is presented.

Introduction/Background

In 1980, India attained self-reliance in launch vehicles with the successful launch of SLV-3. Since then, India has achieved several milestones in the design and development of satellites for communication, earth observation and navigation, along with more powerful launch vehicle systems like the PSLV, GSLV and the upcoming GSLV Mk III. A unique aspect of the Indian Space Programme, which has been acknowledged the world over, has been the unwavering objective of societal development through cost-effective space technology. Once the necessary systems and infrastructure were in place, it was a natural transition for the Indian Space Programme to utilize the preceding decades of experience to move towards lunar and interplanetary missions.

The Advisory Committee for Space Science (ADCOS) was constituted by the Indian Space Research Organisation (ISRO) in 1980 to formulate recommendations on undertaking high quality and internationally competitive research and development. ADCOS set up a Science Panel on Planetary sciences to deliberate and chalk out the direction for Indian lunar and planetary exploration programmes and had identified mission to Mars as one of the important milestones. In order to work out implementable strategies for undertaking such missions, Study Teams were formed drawing experts from all major Centres and Units of ISRO. These teams addressed the launch strategies considering the existing capabilities, future launch opportunities to Moon and Mars, detailed mission options, spacecraft design and configuration challenges, possible scientific experiments, and deep space communication challenges. The findings were presented to national expert committees and their feedback was incorporated during mission planning and implementation.

Mission Strategy

Before embarking on a lunar or interplanetary mission, the following aspects need to be considered.

- Launch scenarios and capabilities
- Minimum energy launch opportunities
- Spacecraft design and configuration challenges
- Possible scientific experiments to augment the already existing understanding of the targeted celestial bodies
- Deep space network challenges for tracking, telecommand and communication

Critical analysis backed by a strong review mechanism, overall system understanding and a thorough analysis of the lessons learned by other space missions gave ISRO the confidence to undertake such complex missions. The ISRO Study Teams carefully studied the above aspects and arrived at a cost-effective mission strategy using existing capabilities at minimum development time for both the Chandrayaan-1 and Mars Orbiter Missions.

Table-1 shows the Mars launch opportunities that were considered for the Mars Orbiter Mission (MOM).

The earliest opportunity was in November 2013 and had the advantage that the total velocity impulse requirement is less by 380 m/s compared to the next opportunity in January 2016 [2].

Table-1 : Mars Mission Launch Opportunities

Departure Date	Flight Duration (Days)	Arrival Date	Argument of Perigee (degree)	Total Velocity Impulse (m/s)
30-11-2013	298	24-11-2013	299	2590
10-01-2016	275	11-10-2016	246	2970
17-05-2018	239	11-01-2019	121	2570

Launch Strategy

Before the first lunar mission, Chandrayaan-1, in 2008, our experience was limited to the injection of spacecraft into LEO, SSPO and GTO orbits using the PSLV and GSLV launch vehicles. The main factors that dictated the design of the Moon and Mars missions were to use the present capability of launch vehicles and to achieve the scientific objectives with minimum development time and cost. The detailed mission planning involved trade-off studies in payload optimization and the transfer trajectory determination in accordance with these requirements [1].

During the 1960s to the 1980s, Lunar missions by the space-faring nations employed the traditional direct transfer to the Moon, wherein the spacecraft is injected into a Lunar Transfer Trajectory in one go from a Low Earth Orbit. The traditional direct transfer has a velocity requirement of 3.1 km/s and normally requires a separate propulsion stage in the launch vehicle. Other unconventional methods employ either (a) highly elliptic phasing orbits, or (b) transfer via Lagrangian points to reach the Moon. Fig.1 illustrates the different methods of reaching the moon [1].

Significant reduction in launch cost can be obtained by minimizing the total delta velocity required to achieve the Transfer trajectory. Detailed trade-off studies showed that the best strategy is to reach the moon via an Elliptic Parking Orbit (EPO). The EPO can be a Geo-Transfer Orbit (GTO) itself; as the energy of GTO is considerably higher than that of a Low Earth Orbit (LEO), it leads to savings on the Transfer Trajectory requirements. Thus the launch vehicle itself provides considerable energy and that required from the spacecraft is correspondingly less[1].

ISRO had already placed communication and weather satellites in the Geosynchronous Transfer Orbit (GTO) using PSLV/GSLV. Planning the mission with GTO as the initial orbit has the advantage that the spacecraft design and its propulsion system comprising the Liquid Apogee Motor (LAM) is almost similar to that used for the existing spacecraft. The other method of transfer via Lagrangian points was not considered even though it was energy efficient, because the time taken to achieve Lunar Orbit Insertion (LOI) was months instead of days and the distance to be travelled was at least four times the lunar distance, putting a very high burden on the communication power requirements [1].

At the time of the Chandrayaan Mission in 2008, PSLV had already proved to be a reliable workhorse launcher with twelve consecutively successful missions. Moreover, ISRO had already gained experience in launching a spacecraft to GTO using PSLV in the PSLV-C4/METSAT mission. The strategy using GTO launch with PSLV required no major change in configuration or the development of additional stages and had similar propellant requirement as that of the PSLV-C4 mission; required short-development-time; had good payload capability with adequate margin; and had the advantage of proven launch scenario including all the range safety related issues completely tested during the PSLV-C4 mission.

A variant of the PSLV vehicle with 12 ton solid strapons (PSLV-XL) was used for the first time in the Chandrayaan-1 mission (Fig.2), and also the subsequent Mars Orbiter Mission (MOM) in 2013. The same variant is being regularly used to meet higher payload requirements for the other missions as well including the deployment of the IRNSS constellation.

Mission Planning

In an interplanetary mission design to Mars, the Earth parking orbit characteristics are so chosen to minimize the energy requirement for Trans-Mars injection and for Mars orbit insertion operations.

The PSLV trajectories for a regular GTO-type elliptic parking orbit mission and for the interplanetary mission to Mars have entirely different trajectories, as shown in

Fig.3. In its regular GTO missions, PSLV achieves about 178 degrees of Argument of Perigee (AOP) for suitable maximum payload. But the MOM demands an AOP of 299 degrees at the time of Trans-Mars injection, which is executed after several phasing orbits. Due to the perturbing forces from non-uniform gravity field, the Moon and the Sun, the initial Earth parking orbit characteristics keep changing. Accounting for these changes in the AOP, the initial Earth parking orbit AOP is fixed. These phasing orbits are determined depending on a chosen lift-off date. This implies that the launch vehicle is expected to achieve different AOPs for different lift-off dates. Values of required launch AOP, for one typical phasing orbit sequence, range from 275 to 288 degrees for lift-off dates between 28 October 2013 and 14 November 2013. Such a large AOP, which is different from those of the usual PSLV launches, is achieved by introducing a long coasting between the third stage (PS3) separation and the fourth stage (PS4) ignition that shifts the perigee location to the desired slot. A coasting of 1600 sec between PS3 separation and PS4 ignition was introduced for the Mars mission [2]. Two ship-borne terminals were commissioned to ensure visibility during PS4 ignition and satellite separation events.

Another parameter that must be ensured by the launch vehicle at the time of MOM injection is Right Ascension of Ascending Node (RAAN). This parameter fixes the launch vehicle trajectory/parking orbit crossing point on the equator with reference to an inertial axis (vernal equinox). RAAN also undergoes changes due to perturbing forces during phasing orbit evolution. It is because of the requirement of RAAN that the lift-off time of the launch vehicle has to be precise and the launch window available on the day of launch is narrow (1-5 min) compared to normal launches (30-60 min or more).

From the parking orbit, a velocity impulse of about 1470 m/s must be added to the existing orbital velocity at perigee to put the spacecraft in Trans-Mars phase/cruise phase. This velocity increment is imparted by splitting into several burns: (i) to reduce the finite burn loss, (ii) to provide flexibility for lift-off day of the launch vehicle, (iii) to validate the spacecraft systems before Trans-Mars phase, and (iv) to ensure visibility of the burn events.

The spacecraft approached Mars in a hyperbolic trajectory, typically with a velocity of 6.5 km/s with respect to the planet, almost 300 days after its launch. The last major manoeuvre was to retard the spacecraft typically by about 1.1 km/s. This reduction would enable it to enter an elliptical orbit around Mars (typically with a height of 80,000 km at apoapsis and 500 km at periapsis) [2]. Minor deviations from this target could have caused the spacecraft to either evade capture by Mars, or crash onto its surface. To achieve the target of precise insertion, sufficient autonomy and accurate sensors were required to be built into the spacecraft. The MOM trajectory is shown in Fig.4.

Spacecraft Configuration

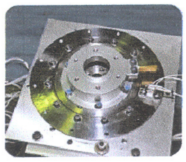
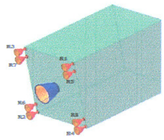
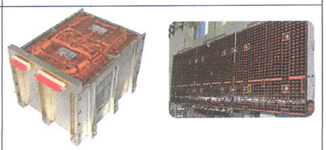
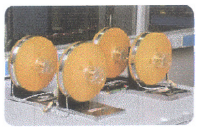
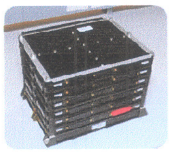
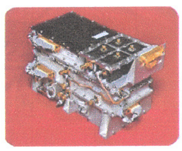
The spacecraft configuration is a highly optimized design based on the heritage derived from operational remote-sensing and communication satellites. While the requirements were less demanding in Chandrayaan-1, MOM required modifications in the areas of communication, power, propulsion systems (mainly related to liquid engine restart after nearly 10 months) and on-board autonomy [3].The MOM spacecraft had to be configured within the allocated mass of 1350 kg (1380 kg for Chandrayaan-1). A final mass of 1337.2 kg was achieved through tight mass control and optimization measures. Mission planning, executing various manoeuvres and operations, and controlling any small deviations in its course through mid-course corrections were the main challenges during the 300-day journey of the Mars Orbiter spacecraft. In addition, the mission demands Deep space communication/navigation along with onboard autonomy to handle contingency situations.

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 challenges posed by varying thermal conditions in various phases of the mission such as earth-orbiting phase, heliocentric phase and Mars Orbit phase, were mitigated to a large extent by adopting different attitudes during each of the phases of operation. Table-2 depicts the salient features of the MOM spacecraft [3].

Solar Array

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. This called for a change in the design of the MOM solar panels with respect to earlier

Table-2 : Salient Features of MOM Spacecraft

Feature	**Specifications**	**Photograph**
Mass	1340-3/+0 Kg	
Structure	Aluminum and Composite Fiber Reinforced Plastic (CFRP) sandwich construction-modified 1-1 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 MOI	 Thruster Configuration
Thermal System	Passive thermal control system	
Power System	Single Solar Array: 1400 x 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 Sub Sensor (SPSS) - 1 No, Course Analogue Sun Sensor (CASS) – 9 Heads, Inertial Reference Unit and Accelerometer Processing (IRAP) – I No. Actuators: Reaction Wheels (5Nms, 4 Nos), Thrusters (22N-8 Nos), 440N 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)	 TMTC Package

IRS missions - an 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 (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. The integrity of solar panel substrate and cell bonding under expected low temperature of about -170°C during eclipse had to be ensured.

Propulsion Systems

Propulsion systems for the Chandrayaan and the MOM missions derive heritage from the earlier INSAT/GSAT missions and consists of a unified bipropellant system for orbit raising and attitude control. It consists of one 440 N liquid engine 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 helium pressurant tank is used to pressurize the propellant. The 22 N thrusters are used for attitude control during various activities of the mission. 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 were 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 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.

Communication Systems

Communicating with the MOM spacecraft over distances of 200 - 400 million km presents challenges in the form of longer communication delays. 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. 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. 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. A Δ-DOR (Delta Differential One Way Ranging) package is employed to generate ranging tones for Δ-DOR measurement. Δ-DOR measurement is used to improve the orbit determination accuracy.

Onboard Autonomy

Given an average command 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 onboard the spacecraft. The thoroughly ground-tested, inbuilt method of autonomy on MOM is based on continuous watch, fault detection, isolation and reconfiguration, without disturbing the Earth-pointing attitude. 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.

Figure 5 depicts various views of the MOM spacecraft [3].

Indian Deep Space Network (IDSN)

Establishment of a Deep Space Tracking Station is a vital element to enable the undertaking of lunar and interplanetary missions. The already established ISTRAC / TTC and external S-band network can support slant range up to 100,000 km during journey towards moon orbit. Beyond this range during the mission profile and at lunar distance of approximately 400,000 km, IDSN is necessary both for TTC and payload data reception. The spacecraft position in orbit is determined using radio frequency ranging technique and computation of orbital parameters based on the measurement of range and rate of change of range. Two ground terminals one with 18m antenna and another with 32m antenna have been established at Byalalu village near Bangalore as a part of IDSN (Fig.6). Though 18m terminal was sufficient for Chandrayaan-1, the 32m antenna enabled the subsequent Mars Orbiter Mission.

Precise Insertion into Mars Orbit

The Mars orbit insertion is the most complex and critical operation in the mission. Because of the Mars-Sun-Earth geometry, the orbit insertion take place when MOM is in eclipse. The radio link between MOM and ground station also gets blocked as the spacecraft is on the other side of Mars at this critical juncture, and MOM must execute all functions autonomously.

As the spacecraft leaves Earth, the accuracy needed in terms of the overall attitude (orientation) targeting requirement to reach Mars is of the order of 0.01 arc sec [2]. In simpler terms, this is equivalent to shooting a 1 cm diameter coin placed at a distance of 200 km. Further, a propulsive error of even 1 m/s in the velocity imparted by the propulsion system can generate as much as 200,000 km error in the position, when the spacecraft approaches Mars. Closer to Mars, a delay of 30 sec in initiating the retarding burn would have resulted in a periapsis of 363,083 km [2].

In the initial five-phase orbital manoeuvres around Earth, the planned total velocity increment was 875.5 m/s, whereas the actual value realized was 873.43 m/s. This close match was achieved due to the accuracy of the ISRO-designed and built ceramic servo accelerometer. This sensor is sensitive to changes in acceleration at the level of one millionth of the gravity force we experience on Earth. This accelerometer was also used in several subsequent trajectory manoeuvres, thereby enabling accurate velocity computation and position.

Payload Considerations

As compared to Earth Observation (EO) missions, the factors and constraints within which a planetary mission and payloads have to operate are considerably different. The payload performance needs to be optimized with minimum resources while meeting demanding science requirements. In general, the design of scientific instruments for planetary missions is normally focused on the utilization of state-of-the-art technology and on the challenges of adapting it to the space environment and to the interfaces imposed by the spacecraft carrying the instruments itself. Due to constraints on the lift-off capability of launch vehicle and the requirement of relatively larger amount of onboard fuel for longer and multiple trajectory manoeuvres, payloads need to be lighter without compromising the performance for good science returns.

Due to launch vehicle envelope limitation and size of space craft bus dictated by fuel tank capacity, the real estate available for the payloads is less, necessitating the requirement for miniaturization to the extent possible. Miniaturized electronics through use of ASICs, integrated system on chip, smaller optics and focal plane assembly by maximizing the usage of enabling technologies are some of the measures that help in realizing compact, light weight payloads. Similarly, as compared to EO missions, planetary missions can lead to a situation where lesser solar flux is available to an orbiting spacecraft owing to the greater distance of the planet under observation to Sun as compared to earth-orbiting satellites. Hence the payloads have to be highly power efficient which is brought about by using less power hungry devices and electronics. The five compact scientific payloads onboard MOM adds up to a meagre 13.5 kg. Fig.7 gives the arrangement of the payloads in MOM.

The thermal environment for a planetary mission can be remarkably different from the temperature loads that a satellite in earth orbit normally encounters. The surface temperature of the planet under observation (e.g. average temperature on the surface of Mars is 77°C less than that of Earth), the in-orbit or on-surface (for lander/rover missions) thermal loads need to be accounted for while finalizing the thermal design of the spacecraft and payloads. Required thermal management is carried out by orbital energy balance modelling, usage of Multi-Layer Insulation (MLI) blankets, radiators, heat pipes, and through proper material selection and appropriate placement of payloads on the spacecraft. The radiation environment can also adversely affect the coatings on the optical components, thus bringing down the optical efficiency and consequently the overall system throughput. Using radiation hardened electronics, optical components and detecting elements and providing adequate radiation shielding are some of the approaches adopted for durable planetary missions.

Due to significantly larger distances involved between a satellite and the receiving ground stations for a planetary mission, there are considerable limitations (when compared to an EO satellite) on the transmission bandwidth, visibility, link margins etc. leading to a much tighter data transmission rate. The instrument design has to make use of onboard data processing and efficient data compression techniques to reduce the data volume and hence the requirements on data transmission speed.

Besides the payloads on an orbiter around a celestial body, the available resources can be even more stringent for instruments on a lander or a rover for surface-landing planetary missions. Hence the degree of miniaturization needs to be extremely high. For such missions (as indeed for orbiter missions) an integrated instrumentation approach may be most pragmatic and productive wherein the suite of instruments onboard complement/supplement each other to maximize the science returns for the resources used.

Conclusion

The next logical step in India's journey towards lunar and interplanetary space is to move towards descent and landing of a spacecraft on the lunar or planetary surface. This requires the mastering of a few critical technologies with commensurate increase in the payload capability of our launch vehicles. A typical mission would consist of a spacecraft housing the orbiter, lander and rover elements in a highly optimized package along with the required sensors and propulsion system to effect a precise and soft landing on the lunar or interplanetary surface. Such a mission to the moon would require high accuracy altitude and vertical velocity measurement provided by a Laser Altimeter, cameras for pattern matching and visual navigation, throttleable propulsion systems for soft landing and mechanisms for deployment of the landing gear and rover. The Navigation Guidance and Control system must be capable of carrying out autonomous descent and landing from the 100 km orbit using position, attitude sensors and, actuators to provide the required thrust during various phases and attitude control thrusters to maintain three axis control despite various disturbances arising because of thrust mismatch among the engines and CG offset. Once the Lander is at a height of 7.5 km (approx), the relative visual navigation approach is adopted, wherein the previously identified landmarks and absolute velocity (vertical and horizontal) along with the absolute altitude is taken as reference and the new trajectory to the identified site is worked out with the present state so as to land at the identified location. At a height of 100m when the Lander is hovering, the hazard camera looks for any hazards at the site below in real time and in the eventuality of hazard being present, the Lander has a capability to move to another safer site say 100-200m away and descend. This is enabled by Laser based inertial sensors and optical cameras with associated processing electronics.

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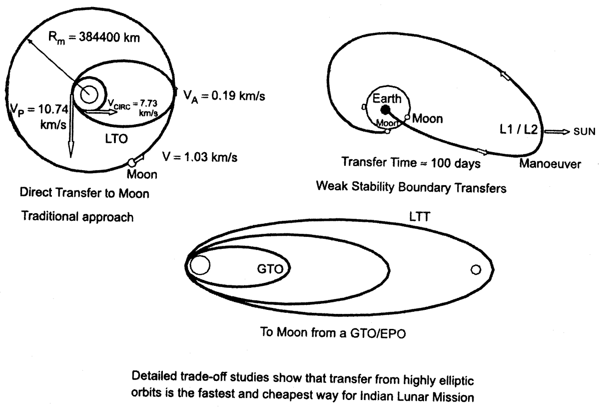

Fig.1 Different Methods of Reaching the Moon {Ref.1}

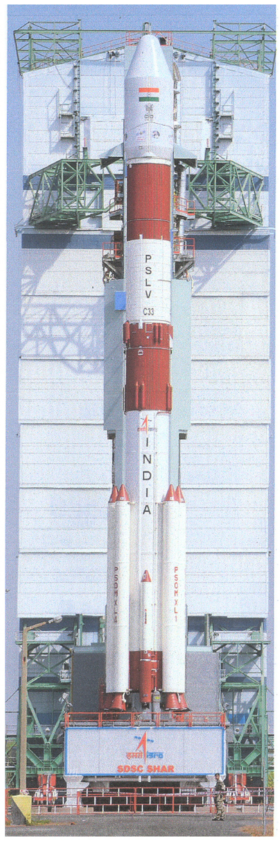

Fig.2 PSLV-XL Vehicle

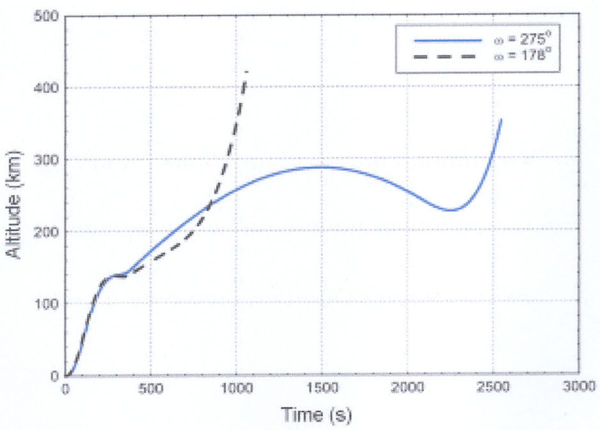

Fig.3 New Trajectory Design to Achieve the Required Argument of Perigee

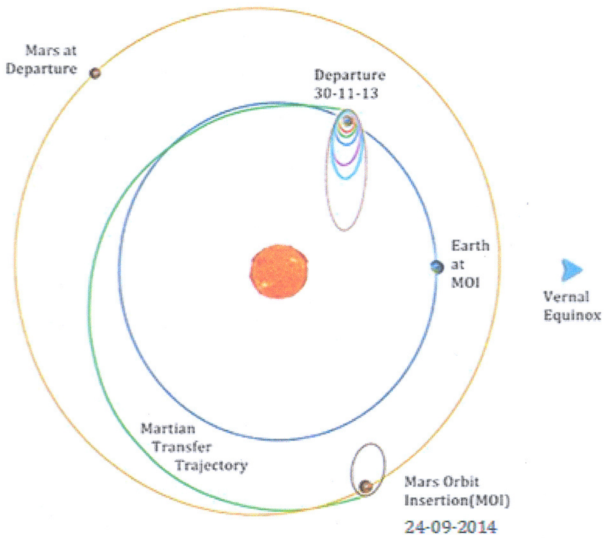

Fig.4 Mars Orbiter Mission Trajectory

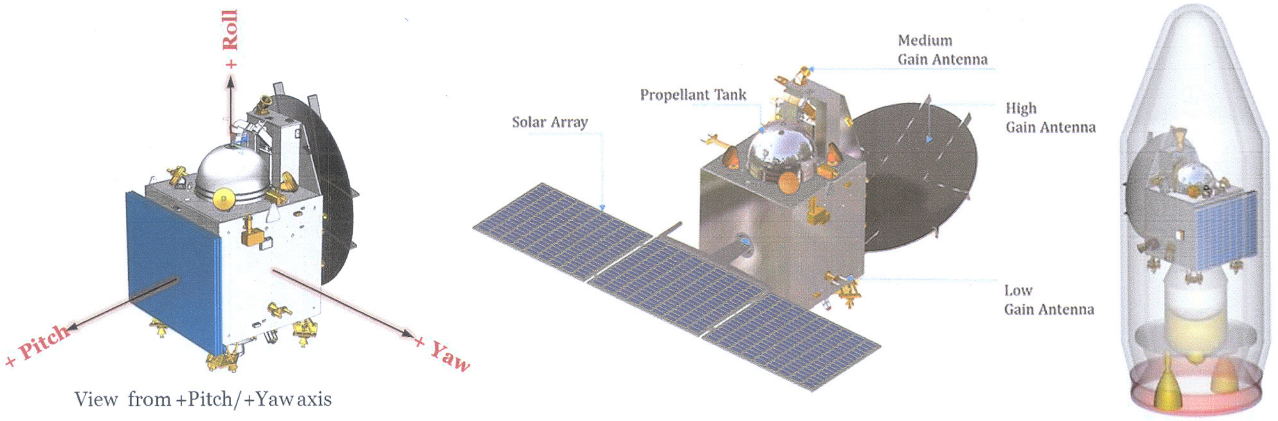

Fig.5 Different Views of the Mars Orbiter Spacecraft [Ref.3]

Fig.6 32m Antenna at the Indian Deep Space Network

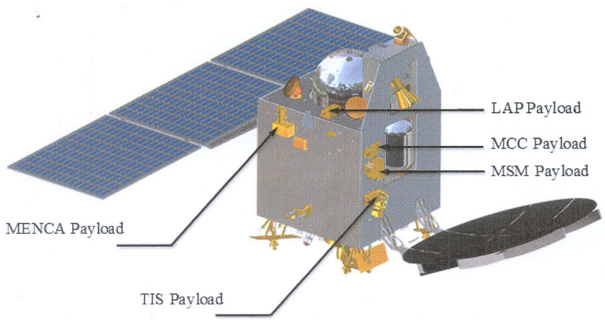

Fig.7 MOM Payloads [Ref.3]