MARS ORBITER MISSION

India's Triumphant Odyssey to Mars

BY GROUP-2

The exciting story of India's quest to explore Mars using a robotic spacecraft

IIT KANPUR
ASTRONOMY CLUB

Mars Orbiter Mission

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1 Introduction

1.1 Mars: Unending mysteries demand continued exploration

It is true that robotic spacecraft equipped with many scientific instruments have explored Mars in a detailed way. Besides, landers which have gently settled down on the surface of Mars have sent breath-taking pictures of the Martian surface. They have also sent weather reports from the surface of Mars and analyzed the soil and the rock samples of Mars for signs of extinct life. But many mysteries associated with Mars have not yet been resolved. Even today, scientists are very actively pursuing answers to questions like How microscopic life survived on Mars? Is there methane in the atmosphere of Mars and how was it generated? When and how Mars became a dry desert from a watery paradise? and many more. To seek answers for these important questions, further exploration of Mars is very necessary. This is the reason why humans are continuously launching spacecraft whenever the opportunity comes and India's *Mars Orbiter Spacecraft* is one amongst them.

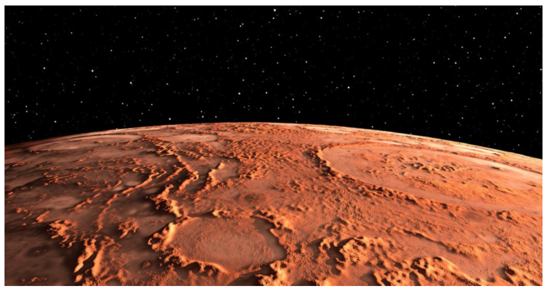

Figure 2: Surface of Mars

1.2 Mars Orbiter Mission

The *Mars Orbiter Mission (MOM)*, also called *Mangalyaan*, is a space probe orbiting Mars since 24 September, 2014. Mars Orbiter Mission was homegrown in India, built by a successful and skilled Indian aerospace establishment that is *Indian Space Research Organisation (ISRO)*. It was launched on 5 November, 2013 from the First Launch Pad at *Satish Dhawan Space Centre, Andhra Pradesh*, using a *Polar Satellite Launch Vehicle (PSLV) rocket C25*. The MOM probe spent about a month in Earth orbit, where it made a series of seven apogee-raising orbital manoeuvres before Trans-Mars injection on 30 November, 2013. It is India's first interplanetary mission and it made it the fourth space agency to achieve Mars orbit and the first nation in the world to do so in its first attempt. The main foreign contribution to this mission's success has been the irreplaceable communications services of the NASA's Deep Space Network, delivering Mars Orbiter Mission's messages to Earth.The mission is a "*technology demonstrator*" project to develop the technologies for designing, planning, management, and operations of an interplanetary mission. The spacecraft is currently being monitored from the *Spacecraft Control Centre at ISRO Telemetry, Tracking and Command Network (ISTRAC)* in Bengaluru with support from the *Indian Deep Space Network (IDSN)* antennae at Bengaluru, Karnataka.

2 Objectives

Any major effort undertaken should have a very clear goal or a set of objectives. Throughout human history, we see many examples of this. In the space field, this becomes very crucial because of the careful planning required to allocate the necessary human skill and money. The unimaginable speeds achieved and the temperature, forces and risks experienced during the journey of a rocket in space make this inevitable. Thus, only a few countries are successful in mastering various technologies necessary for spaceflight. It is a matter of pride that India is one of them. Following are the major objectives of MOM:

- Demonstration of India's capability to build a spacecraft capable of travelling to Mars and survive in an orbit around the red planet.
- Design and realization of a Mars orbiter with a capability to survive and perform Earth bound manoeuvres, cruise phase of 300 days, Mars orbit insertion / capture, and on-orbit phase around Mars.
- Deep space communication, navigation, mission planning and management.
- Incorporate autonomous features to handle contingency situations.

- Exploration of Mars surface features, morphology, mineralogy and Martian atmosphere by indigenous scientific instruments.
- Develop the technologies required for design, planning, management and operations of an interplanetary mission.
- Orbit maneuvers to transfer the spacecraft from an elliptical Earth orbit to a heliocentric trajectory and finally insert it into Mars orbit.
- Development of force models and algorithms for orbit and attitude computations and analyses.
- Navigation in all mission phases.
- To study sustainability of life on the planet.

3 Cost

India's space agency ISRO spent a mere $75 million(INR 450 Crores) to launch a small spacecraft bound for Mars, 140 million miles away. This was regarded as an outstanding achievement. The cost is relatively trifling compared to the other four nations that launched Mars missions, costing billions (Nasa's MAVEN Mission cost a whopping $671 Million).

Following, we analyse what was ISROs approach in achieving this level of cost-effectiveness:

Sr. No.	**System Description**	**Cost (INR in Crores)**
1	Space Segment	215.00
2	Ground Segment	70.00
3	Project Management/Contingency	45.00
4	Programme Elements	10.00
5	Launch Cost (PSLV-XL)	110.00
	Total	450.00

Figure 3: Breakdown of Cost

3.1 Modular Approach:

The use of Indian materials and producing engines in India has been the approach. For every successive launch, ISRO takes the base of the previous, proven launch technology, modifies and builds things on it. They use this modular tactic in other areas like the Payloads as well. This approach saved time and, more importantly, Millions.

3.2 Lesser number of Ground Tests:

ISRO conducted a fewer number of ground tests but extracted the best possible outcomes from it. These ground tests are time-consuming and expensive.

3.3 Tight Schedule and Longer Work hours:

ISRO completed this mission on a tight schedule of 15 months. Scientists usually had more extended workdays, as high as 18 and 20 hours; this saved much of the costs. Also, ISRO units work under multi-project environments, thus reducing the personnel cost for a particular mission.

4 Structure

4.1 PSLV: The muscle power to lift Mars Orbiter Spacecraft from the mother earth

The giant rocket, to be more precise, the 'launch vehicle' that lifted Mars Orbiter Spacecraft from the surface of the earth and put it into an orbit around the earth was Polar Satellite Launch Vehicle (PSLV). Before launching Mars Orbiter Spacecraft, this 'trusted workhorse' of ISRO had scored 23 successes continuously.

4.1.1 Body

- Height: 44 m
- Diameter: 2.8 m
- Number of stages: 4

4.1.2 Payload Capacity

- Payload to SSPO: 1,750 Kg

PSLV earned its title 'the Workhorse of ISRO' through consistently delivering various satellites to low Earth Orbits, particularly the IRS series of satellites. It can take up to 1750 Kg of payload to Sun-Synchronous Polar Orbits of 600 Km altitude.

- Payload to Sub GTO: 1,425 Kg

Due to its unmatched reliability, PSLV has also been used to launch various satellites into Geosynchronous and Geostationary orbits, like satellites from the IRNSS constellation.

4.1.3 Fairing

Payload fairing of PSLV, also referred to as its '*Heatshield*' weighs 1,182 Kg and has 3.2 m diameter. It has Isogrid construction and is made out of 7075 aluminum alloy with a 3 mm thick steel nose cap.

Separation Mechanism- The two halves of fairing are separated using a pyrotechnic device based jettisoning system consisting horizontal and vertical separation mechanisms.

4.1.4 Strap-on Motors

PSLV uses 6 solid rocket strap-on motors to augment the thrust provided by the first stage in its PSLV-G and PSLV-XL variants.

- Fuel: HTPB
- Max. Thrust: 719 kN

4.2 Mars Orbiter Spacecraft: India's first robotic messenger to Mars

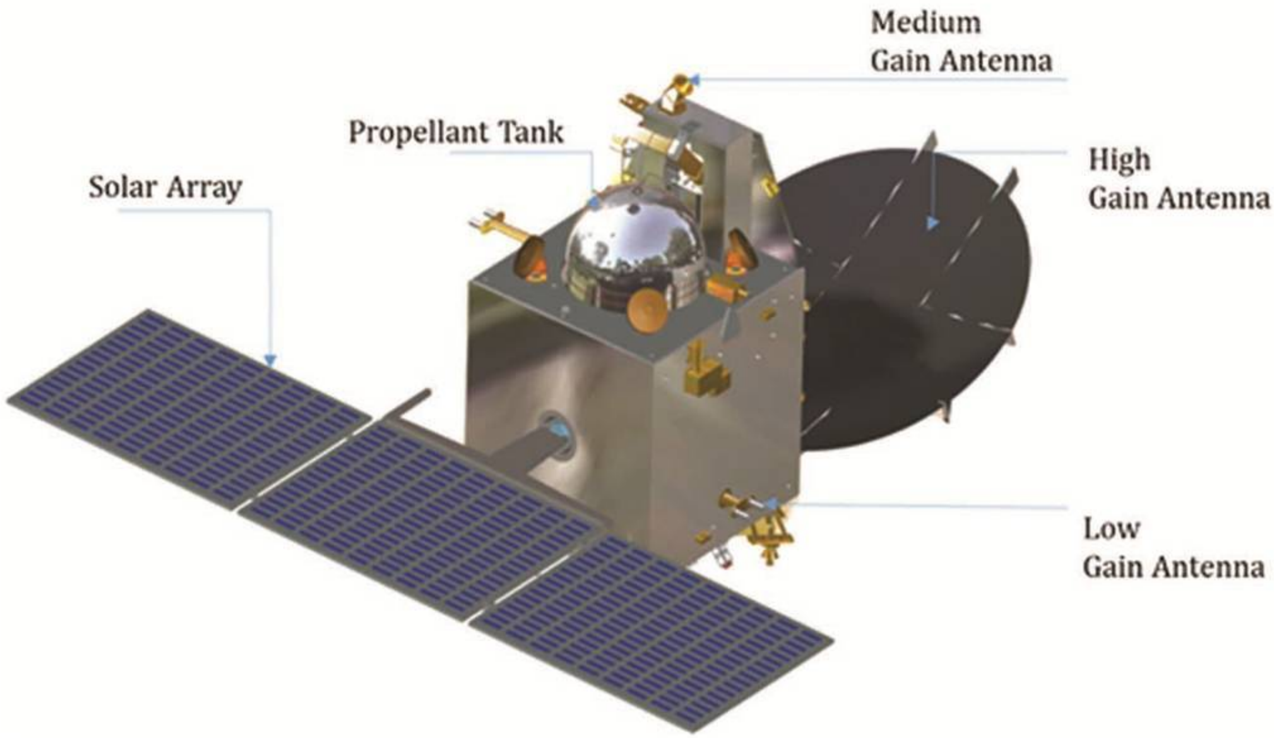

Figure 4: Mars Orbiter Spacecraft

The *Mangalyaan* spacecraft bus is cuboid in shape featuring composite and metallic honeycomb sandwich panels as well as the cylinder. Mangalyaan weighs in at 1,337 Kg. The spacecraft has a dry mass of 475 Kg including a payload mass of 15 Kg and it carries a fuel load of 852 Kg. The spacecraft is equipped with a single deployable solar array that consists of three panels – each being 1.4 by 1.8 m in size. Mangalyaan is equipped with a bipropellant Main Propulsion System and an Attitude Control System. The Propulsion System features two spherical propellent tanks each holding 390 liters of propellant. MOM is equipped with a propulsive Attitude Control System consisting of eight 22- Newton thrusters that also use *UMDH* and *MON-3* propellants. The thrusters also use a co-axial type Titanium alloy injector and a Columbian combustion chamber. Each 22 N thruster assembly weighs 0.8 Kg. In addition, the MOM spacecraft is equipped with four reaction wheels. Mangalyaan is equipped with a 2.2 m diameter High Gain Antenna which is a parabolic X-Band reflector antenna that is used for data downlink and command uplink.

5 Heat Shields

Object or vehicle used to deliver people or payload safely through the atmosphere of the planet (earth in case of returning) is called a **re-entry vehicle**. A Re-entry Vehicle could be a rocket, satellite, or a manned capsule. In our case it is the MARS- Orbiter itself.

Heat shields protect spacecraft from extreme temperatures and thermal gradients by two primary mechanisms; thermal insulation and radiative cooling, which respectively isolate the underlying structure from high surface temperatures, while emitting heat outwards through thermal radiation. The different types of heat shields used are:-

- Insulation blankets
- Insulation tiles
- Reinforced carbon-carbon
- Ablative heat shield

- Regenerative cooling

Figure 5: Heat Shield used in PSLV

In Mars Orbiter mission, for heat shields operating at 80K, aluminum has a cost advantage over copper as the material of construction assuming that labor costs for the two materials are the same. Highly pure alloys such as 1100-O aluminum have superior thermal properties, but that is more than offset by its higher price.

Usually for dwelling in terrestrial places, an alloy of copper and stainless steel is also used depending on the cost effectiveness of each metal.

And the rocket PSLV-XL-C25 was protected with the fairing that was typically made of aluminium or composite materials and incorporated blankets of acoustic absorbing materials to protect the spacecraft from the significant noise and high frequency vibration that occur during lift-off.

6 Spacecraft Thermal Control System

In spacecraft design, the function of the **thermal control system (TCS)** is to keep all the spacecraft's component systems within acceptable temperature ranges during all mission phases.

It is also necessary to keep specific components (such as optical sensors, atomic clocks, etc.) within a specified temperature stability requirement, to ensure that they perform as efficiently as possible. Thermal system is designed to cope with a range of thermal environment considering that average solar flux at Mars is 589 W/m2 (42% of flux at Earth orbit). The thermal control subsystem works in two ways:

- **Protects the equipment from overheating**, either by thermal insulation from external heat flux or by proper heat removal from internal sources.
- **Protects the equipment from temperatures that are too low**, by thermal insulation from external sinks, by enhanced heat absorption from external sources, or by heat release from internal sources.

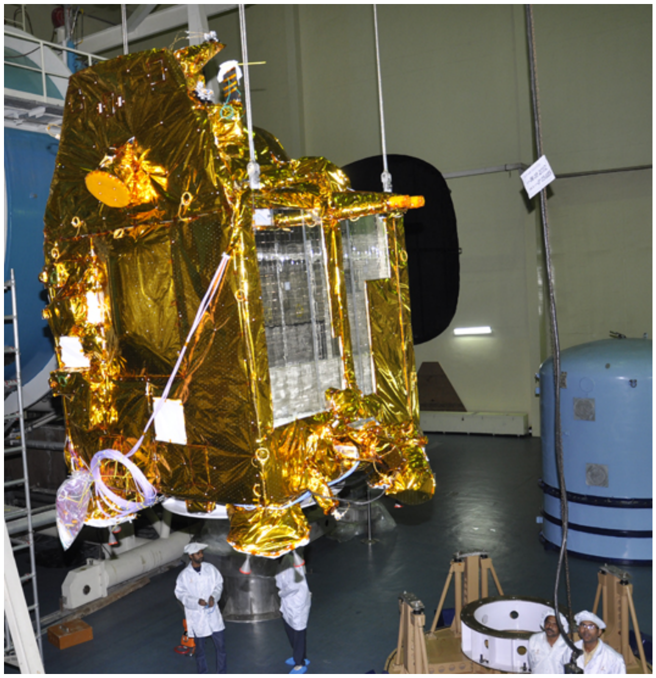

Figure 6: MLI

The various types of thermal control system include:-

- Coating
- Multilayer insulation (MLI)
- Heaters
- Heat pipes
- Radiators
- Sun shield

The kind of TCS **used in PSLV-XL C25 was Multi-layer Insulation(MLI)**. Spacecraft components such as propellant tanks, propellant lines, batteries, and solid rocket motors are also covered in MLI blankets to maintain ideal operating temperature. MLI consists of an outer cover layer, interior layer, and an inner cover layer. It can also provide a layer of defense against dust impacts, protecting delicate internal instruments and sensors from tiny particles of space debris.

For this mission, **radiators and sun shields were used** as well. Flat-plate radiators mounted to the side of the spacecraft reject heat by infrared (IR) radiation from their surfaces, While a Sun shield restricts overheating caused by sunlight hitting a spacecraft.

Also the metal gold is used that helps protect against corrosion from ultraviolet light and x-rays and acts as a reliable and long lasting electrical contact in onboard electronics.

7 Rocket Engine

The engine used in Mangalyaan mission's rocket, PSLV-XL c25 is known as **Vikas** (**VIK**ram **A**mbalal **S**arabhai). The Vikas is a family of liquid fuelled rocket engines conceptualized and designed by the Liquid Propulsion Systems Centre in the 1970s. The design was based on the licensed version of the Viking engine with the chemical pressurisation system. This engine is used to power the second stage of PSLV. The propellant loading in the PSLV rocket is 40 tons.

7.1 Technical Specifications

The engine uses up about 40 metric tons of Unsymmetrical dimethylhydrazine (UDMH) as fuel and Nitrogen tetroxide (N2O4) as oxidizer with a maximum thrust of 725 kN. An upgraded version of the engine has a chamber pressure of 58.5 *bar* as compared to 52.5 *bar* in the older version and produces a thrust of 800 kN. The engine is capable of gimballing.

In a gimbaled thrust system, the exhaust nozzle of the rocket can be swiveled from side to side. As the nozzle is moved, the direction of the thrust is changed relative to the center of gravity of the rocket.

LAM(Liquid Apogee Motor) provides 440 Newtons of thrust which equates to 44.87 Kilograms. The engine operates and an mixture ratio (O/F) of 1.65 and has a nozzle ratio of 160 providing a specific impulse of 3,041N*sec/kg. The engine's injector is a co-axial swirl element made of titanium while the thrust chamber is constructed of Columbium alloy that is radiatively cooled. Electron welding technique is used to mate the injector to the combustion chamber.

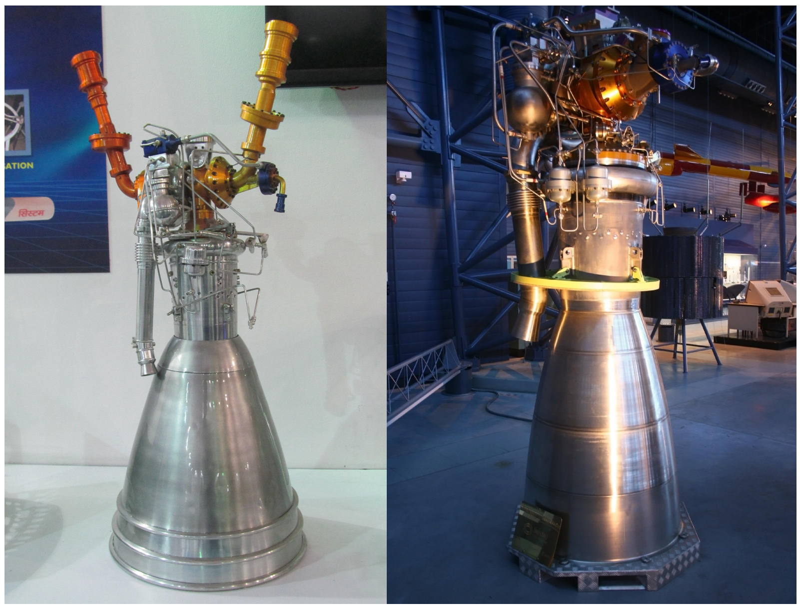

Figure 7: VIKAS engine used in MOM

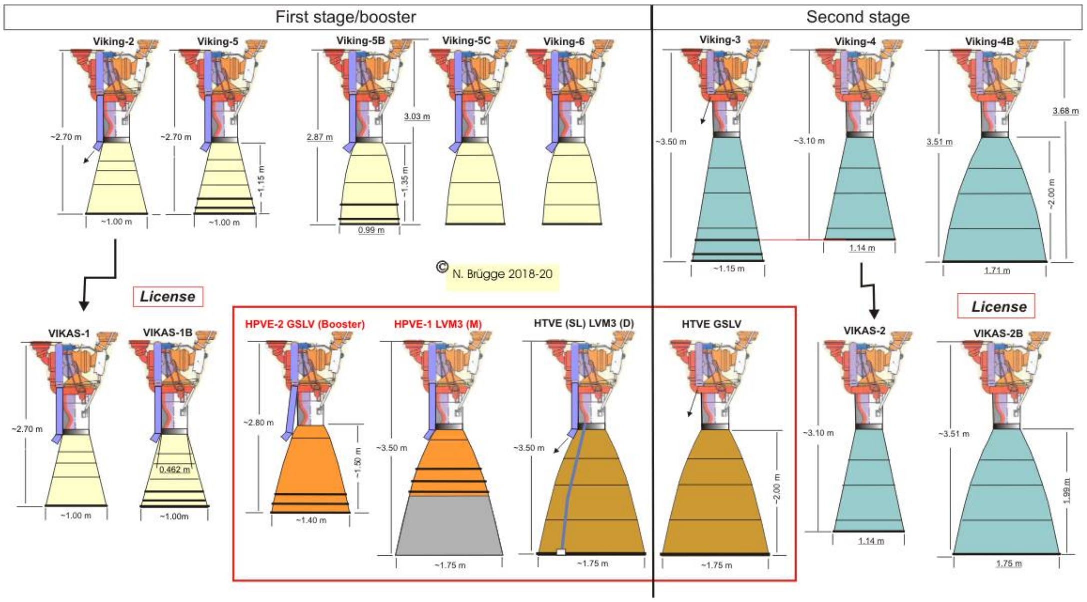

Figure 8: Evolution of the VIKAS engine

The engine can tolerate injection pressures of 0.9 to 2.0 MPa, propellant temperatures of 0 to 65°C, mixture ratios of 1.2 to 2.0 and bus voltages of 28 to 42 Volts. The engine is certified for long firings of up to 3,000 seconds and a cumulative firing time greater than 23,542 seconds.

8 Payload

The Mars Orbiter Mission carries five payloads to accomplish its scientific objectives. Three electro-optical payloads operating in the visible and thermal infra-red spectral ranges and a photometer to sense the Mars atmosphere and surface. One additional backup payload is planned in case of non-availability of the identified payloads.

8.1 Methane Sensor for Mars (MSM)

Figure 9: Methane Sensor for Mars.

MSM is designed to measure Methane (CH_4) in the Martian atmosphere with PPB accuracy and map its sources. Data is acquired only over illuminated scenes as the sensor measures reflected solar radiation. Methane concentration in the Martian atmosphere undergoes spatial and temporal variations. Hence global data is collected during every orbit.

8.2 Mars Color Camera (MCC)

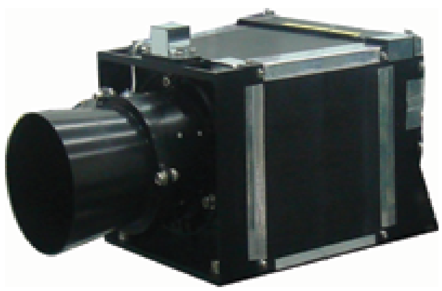

Figure 10: Mars Color Camera.

This tri-color Mars Color camera gives images and information about the surface features and composition of Martian surface. They are useful to monitor the dynamic events and weather of Mars. MCC will also be used for probing the two satellites of Mars – Phobos and Deimos. It also provides the context information for other science payloads.

8.3 Lyman Alpha Photometer (LAP)

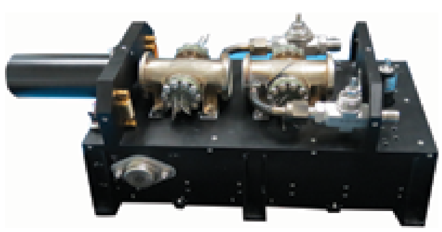

Figure 11: Lyman Alpha Photometer.

Lyman Alpha Photometer (LAP) is an absorption cell photometer. It measures the relative abundance of deuterium and hydrogen from Lyman-alpha emission in the Martian upper atmosphere (typically Exosphere and exobase). Measurement of D/H (Deuterium to Hydrogen abundance Ratio) allows us to understand especially the loss process of water from the planet.

The objectives of this instrument are as follows:

1. Estimation of D/H ratio
2. Estimation of escape flux of H_2 corona
3. Generation of Hydrogen and Deuterium coronal profiles

8.4 Mars Exospheric Neutral Composition Analyser (MENCA)

Figure 12: Mars Exospheric Neutral Composition Analyser.

MENCA is a quadrupole mass spectrometer capable of analysing the neutral composition in the range of 1 to 300 amu with unit mass resolution. The heritage of this payload is from Chandra's Altitudinal Composition Explorer (CHANCE) payload aboard the Moon Impact Probe (MIP) in Chandrayaan-1 mission.

8.5 Thermal Infrared Imaging Spectrometer (TIS)

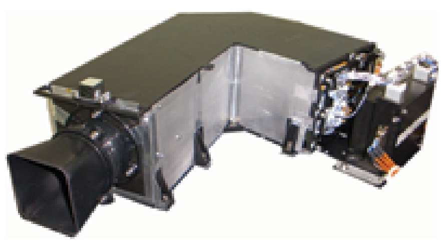

Figure 13: Thermal Infrared Imaging Spectrometer.

TIS measures the thermal emission and can be operated during both day and night. Temperature and emissivity are the two basic physical parameters estimated from thermal emission measurement. Many minerals and soil types have characteristic spectra in the TIR region. TIS can map surface composition and mineralogy of Mars.

9 Communication

To meet the challenging task of managing distance up to 400 million km S-band systems are kept for both TTC and Data transmission. Delta Differential One-way Ranging (D-DOR) Transmitter is provided for ranging to improve Orbit Determination accuracy. Antenna System consists of Low Gain Antenna, Medium Gain antenna and High Gain Antenna.

ISRO Telemetry Tracking and Command Network (ISTRAC) will be providing support of the TTC ground stations, communications network between ground stations and control center, Control center including computers, storage, data network and control room facilities, and the support of Indian Space Science Data Center (ISSDC) for the mission. The ground segment systems form an integrated system supporting both launch phase, and orbital phase of the mission.

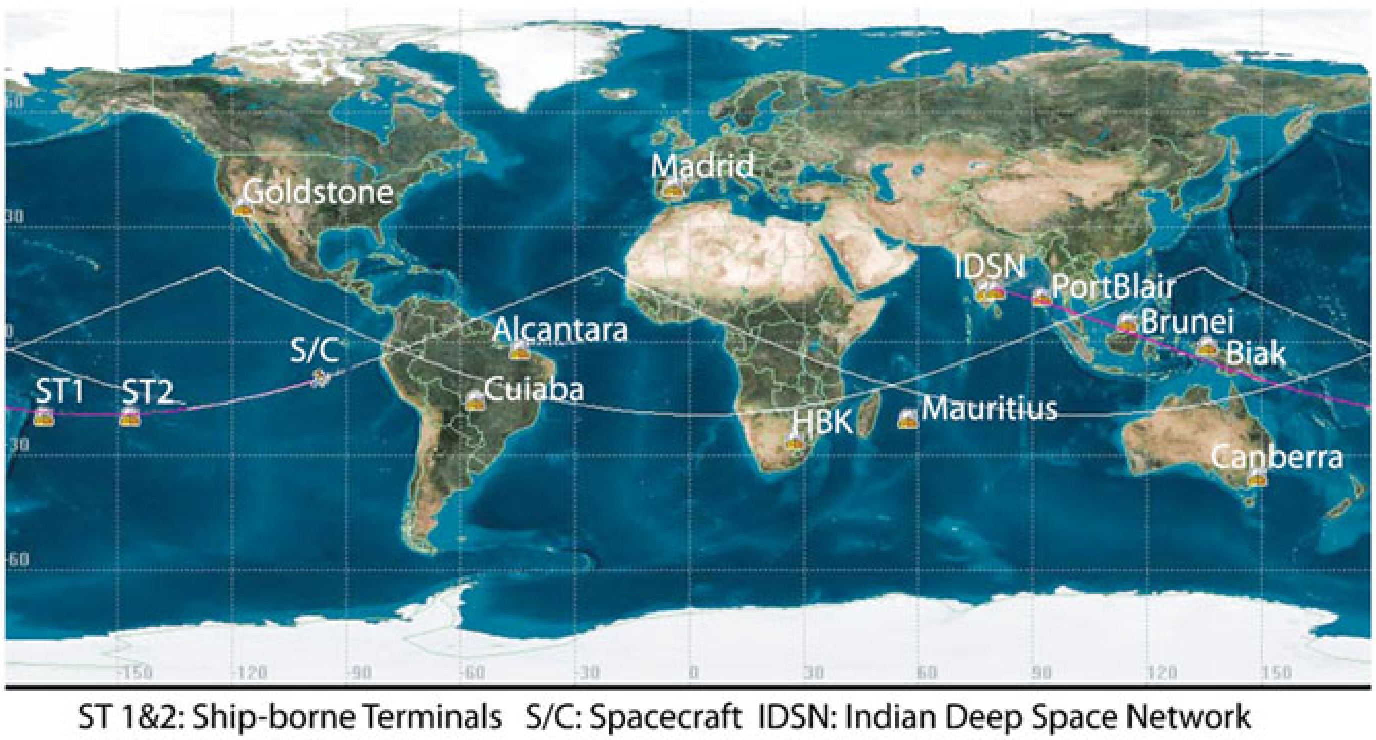

Figure 14:

9.1 Launch Phase

- The launch vehicle is tracked during its flight from lift-off till spacecraft separation by a network of ground stations, which receive the telemetry data from the launch vehicle and transmit it in real time to the mission computer systems at Sriharikota, where it is processed.
- The ground stations at Sriharikota, Port Blair, Brunei provide continuous tracking of the PSLV-C25 from liftoff till burnout of third stage of PSLV-C25.
- Two ships carrying Ship Borne Terminals (SBT) are being deployed at suitable locations in the South Pacific Ocean, to support the tracking of the launch vehicle from PS4 ignition till spacecraft separation.

9.2 Orbital Phase

- After satellite separation from the launch vehicle, the Spacecraft operations are controlled from the Spacecraft Control Centre in Bangalore.
- To ensure the required coverage for carrying out the mission operations, the ground stations of ISTRAC at Bangalore, Mauritius, Brunei, and Biak are being supplemented by Alcantara and Cuiaba TTC stations of INPE, Brazil, Hartebeesthoek TTC station of SANSA and the DSN network of JPL, NASA.

Communication with a deep space probe is done with the help of a 'Deep Space Network' or DSN. USA, India, Russia, European Union, China and Japan have their own DSN. Indian DSN (IDSN) facility is situated at Byalalu village near Bangalore. There are three antennas: 11 meter antenna, 18 meter antenna and 32 meter antenna. The 18 m antenna was mainly built for the Chandrayaan mission.

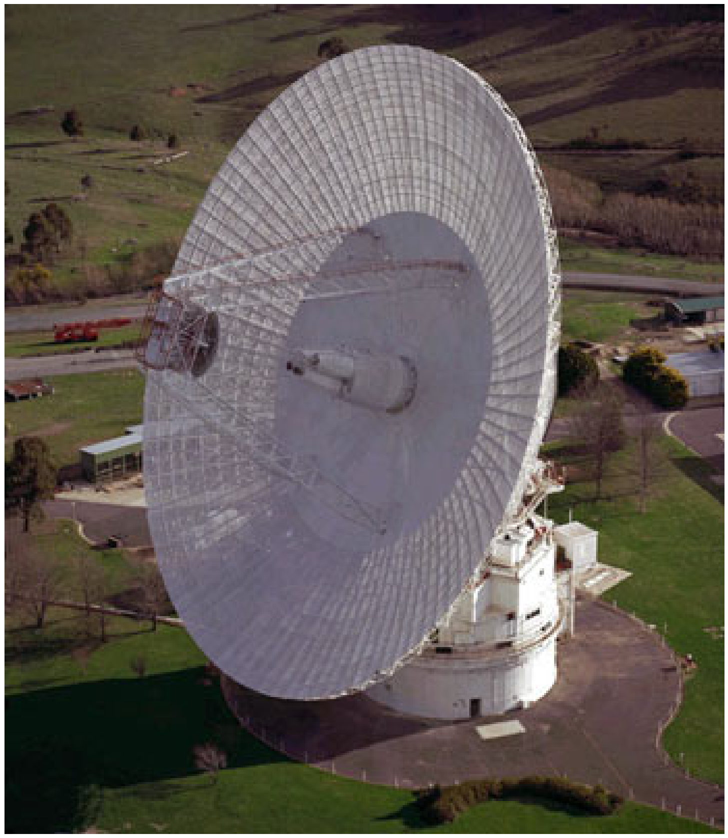

Figure 15:

The 32 m antenna is used for communication with deep space probes like Mangalyaan. These antennas are capable of communicating with the space probe for various purposes. A fibre optics / satellite link provides the necessary connectivity between the IDSN site and Spacecraft Control Centre / Network Control Centre. The station is also equipped for remote control from the ISTRAC Network Control Centre (NCC). But since India does not have multiple DSN facilities, NASA provides vital position data via its three stations when needed. It was required especially in time of Martian orbit insertion phase. NASA has 3 DSN facilities in 3 locations on Earth (each with multiple antennas.

10 Propulsion

10.1 Introduction

Being the first interplanetary mission of ISRO the Satellite had to traverse nearly 780,000,000 km. To overcome it ISRO used the PSLV – C25. It carried the Mars Orbiter Satellite (1337 kg) into a 250 km $\times$ 23500 km elliptical orbit. The Satellite was then further navigated to a hyperbolic departure trajectory and after this it traversed an interplanetary cruise trajectory before entering the intended orbit of 366 km $\times$ 80000 km around Mars.

10.2 PSLV - C25: The Launch Vehicle

PSLV is a four – stage vehicle with alternate Solid and Liquid propulsion stages. The booster stage along with the strap – on motors and the third stage are solid motors while the second and forth stages are liquid engines. PSLV – C25 / MOM employs the PSLV – XL version which has six extended strap – on motors attached to the first stage.

PSLV – C25 stages at a glance:

	STAGE-1	PSOM-XL	STAGE-2	STAGE-3	STAGE-4
Propellant	Solid (HTPB Based)	Solid (HTPB Based)	Liquid (UH25 + N2O4)	Solid (HTPB Based)	Liquid (MMH + MON-3)
Propellant Mass (Tonne)	138	12.2	42	7.6	2.5
Peak Thrust (kN)	4800	718	799	247	7.3 X 2
Burn Time (sec)	103	50	148	112	525
Diameter (m)	2.8	1	2.8	2.0	2.8
Length (m)	20	12	12.8	3.6	2.7

Figure 16: Some specifications of different stages of the launch vehicle PSLV - C25.

We see here that for lower stages in PSLV ISRO had used Solid – propellants (SRBs). Solid propellant of these stages is **HTPB** (Hydroxyl – terminated polybutadiene) [2] based. Nowadays it is widely used in propulsion both for solid propellants and hybrid fuels because it is easy to manufacture and provides good mechanical properties. It binds the oxidizing agent, fuel and other ingredients into a solid but elastic mass in most composite propellants systems. Thermochemical analysis has shown that the HTPB binder can be used as a solid fuel in hybrid rockets, granting higher specific impulse with respect to liquid propellants. From the above table we can see that the thrust provided by the STAGE – 1 is the maximum among the four stages. This stage was used to provide the main thrust to lift the PSLV – C25 off.

Though these SRBs provide great thrusts and are simple in mechanisms than the liquid propellant engines there are some disadvantages in using them. The SRBs committed the spacecraft to lift off and ascent flight to the orbit, without the possibility of launch or ascent abort, until the boosters have consumed their propellants [5]. In addition to this we cannot control the thrust of these SRBs. That's why SRBs are not suitable for using in higher stages.

For higher stages liquid propellants are much more suitable. Though they provide less thrust compared to solid propellants, here we have the liberty to control the flow of the propellant. For PSLV – C25 the 2nd stage and the 4th stage are liquid propellant based.

Both the stage used bipropellant system:

(a) for STAGE – 1 (UH25 + N_2O_4)

(b) for STAGE – 2 (MMH + MON-3)

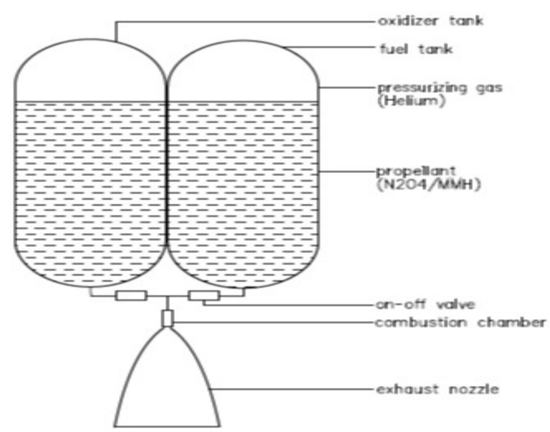

Figure 17: Gas pressurized N_2O_4/MMH liquid propellant rocket engine. [1]

UH25 + N_2O_4:

It was used in STAGE – 2 of PSLV – C25. **UH25** [6] was used as fuel and N_2O_4 (Nitrogen tetroxide) was used as oxidizer. UH25 is a mixture of 75% UDHM (Unsymmetrical dimethylhydrazine) and 25% hydrazine hydrate. It is hypergolic (propellants that ignite spontaneously on contact with each other and require no ignition source), easily flammable, toxic and corrosive. Some physical properties of the propellant:

- Specific impulse (vacuum) = 333 s, specific impulse (sea level) = 285 s.
- Maximum thrust = 799 kN
- Optimum oxidizer to fuel ratio = 2.61
- Temperature of combustion = 3,415 K
- Density = 1180 kg/m^3
- Oxidizer specifications: Density = 1431 kg/m^3, Freezing point = - 11 ℃, Boiling point = 21 ℃
- Fuel specifications: Density = 859 kg/m^3, Freezing point = - 57 ℃, Boiling point = 63 ℃

MMH + MON-3:

It was used in STAGE – 4 of PSLV – C25. It used **MMH** (Mono – methyl hydrazine) and **MON – 3** (a mixture of nitrogen tetroxide with approximately 3% nitric oxide) [4]. It is highly toxic, carcinogenic and hypergolic. Some physical properties of the propellant:

- Specific impulse (vacuum) = 340 s, specific impulse (sea level) = 292 s
- Maximum thrust = 7.3×2 kN
- Optimum oxidizer to fuel ratio = 2.27
- Temperature of combustion = 3,455 K
- Density = 1170 kg/m^3
- Oxidizer specifications: Density = 1370 kg/m^3, Freezing point = - 15 ℃
- Fuel specifications: Density = 880 kg/m^3, Freezing point = - 52 ℃. Boiling point = 87 ℃

10.3 Mars Orbiter Mission Spacecraft:

For the spacecraft ISRO used unified bipropellant hypergolic system (MMH +N_2O_4). The propulsion system consists of a 440 N Liquid engine and 8 numbers of 22 N thrusters [3]. It used MMH as fuel and N_2O_4 as oxidizer. Though these are toxic, because of their high specific impulse, extreme storage stability and hypergolic properties, this combination is extensively used in orbital manoeuvres, reaction controls and launch vehicle propulsion. One of the major challenges was that the liquid engine of the spacecraft had to be restarted after 10 months for Martian Orbit Insertion (MOI) manoeuvres. Some of the physical properties of the propellant used:

- Specific impulse (vacuum) = 336 s, specific impulse (sea level) = 288 s
- Net mass = 852 kg
- Optimum oxidizer to fuel ratio = 2.16
- Temperature of combustion = 3,385 K

- Density = 1200 kg/m^3
- Oxidizer specifications: Density = 1450 kg/m^3, Freezing point = - 11 ℃, Boiling point = 21 ℃
- Fuel specifications: Density = 880 kg/m^3, Freezing point = - 52 ℃. Boiling point = 87 ℃

10.4 Environmental Impact: Pollution and Climate Change

Space launches can have a hefty carbon footprint due to the burning of solid rockets and the hydrocarbons present in the liquid propellants. Apart from this rocket engines release trace gases into the upper atmosphere that depletes the ozone layer. Rocket soot accumulates in the upper stratosphere, where the particles absorb sunlight. This accumulation heats the upper stratosphere, changing chemical reaction rates and likely leading to ozone loss. Also, the "space junk" is a growing concern. Rockets which uses liquid hydrogen fuel as propellant produces water vapour. But for more efficiency we are leaning towards the solid and liquid propellants which are hydrocarbon based. Though rocket launches are relatively infrequent. The accumulated effect can be a threat in the future.

11 Staging

LAUNCH

The Mars Orbiter Mission probe lifted-off from the First Launch Pad at Satish Dhawan Space Centre (Sriharikota Range SHAR),Andhra Pradesh, using PSLV-C25 at 09:08 UTC on 5 November 2013.

PSLV-C25 which launched Mars Orbiter Mission Spacecraft has four stages using solid and liquid propulsion systems alternately.

- First stage (PS1)
- Second stage (PS2)
- Third stage (PS3)
- Fourth stage (PS4)

Besides these, the vehicle also used larger strap-on motors (PSOM-XL) to achieve higher payload capability.

Stage 1: At the moment of T-0, the PS1 Stage is ignited followed 0.5 seconds later by Boosters 1 and 2 and another 0.2 seconds later by Boosters 3 and 4 to create a total launch thrust of 700,600 Kgs. The remaining Boosters (5 and 6) are ignited at T+25 seconds when the vehicle is already 2.5 Km in altitude. Each of the Boosters burns for 49.5 seconds. The four ground-lit boosters are separated at T+1 minute and 10 seconds and fall into the Ocean. The air-lit boosters are jettisoned 22 seconds later enabling the PS1 stage to continue ascent on its own.

When the first stage has burned out, it separates from the second stage at T+1:53 followed by PS2 ignition an instant later. Staging occurs at approximately 58 Km in altitude.

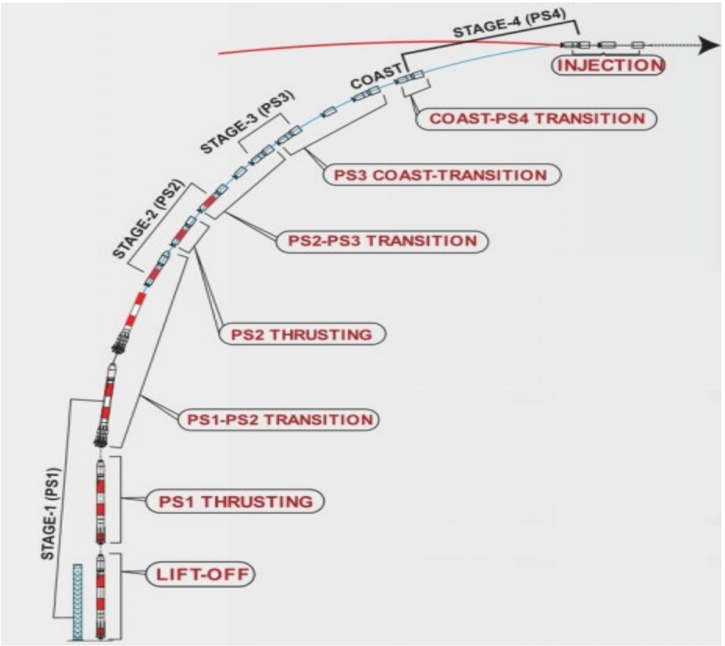

Figure 18: Mission Profile

Stage 2: During the second stage burn, the launch vehicle departs the dense atmosphere – allowing the vehicle to jettison its payload fairing at T+3:22 at an altitude of 113 Km. The second stage burns for about two minutes and 35 seconds before separating from the third stage that then ignites and assumes control of the flight at T+4:26.

Stage 3: The solid-fueled third stage burns for 112 seconds to boost the stack to a sub-orbital trajectory. After burnout of the PS3 stage, the stack begins a coast phase – initially holding onto the spent third stage before separating it at T+9:43 and continuing to coast uphill.

Stage 4: The coast phase allows the vehicle to fly uphill so that the fourth stage burn can increase the apogee altitude and also put a few Km onto the perigee to place the stack in a stable orbit. Once the stack reaches its desired altitude, the two L-2-5 engines of the fourth stage ignite at T+35 minutes on a burn of about 8.5 minutes to boost the stack into its Transfer Orbit. The Mars Orbiter Mission targets an injection orbit of 264 by 23,550 Kilometers at an inclination of 19.2 degrees.

12 Orbital Dynamics

12.1 Orbital Maneuver

After the Mars orbiter spacecraft was delivered to the orbit with a perigee of 264.1 km (164.1 mi), an apogee of 23,550 km (14,630 mi), and inclination of 19.20 degrees, the next task was to place the satellite in a transfer orbit from where it escapes Earth' Sphere of Influence, enter Heliocentric phase and finally comes under the Mars Sphere of Influence. It was achieved by doing six subsequent apogee-raising orbital Maneuver , firing the main liquid rocket engine, always when passing perigee and cutting-off engine after a short interval of time (burn time), each time increasing the apogee by several thousand kilometers. This was done to gradually build up velocity required to escape from Earth's gravitational pull in a fuel-efficient manner.

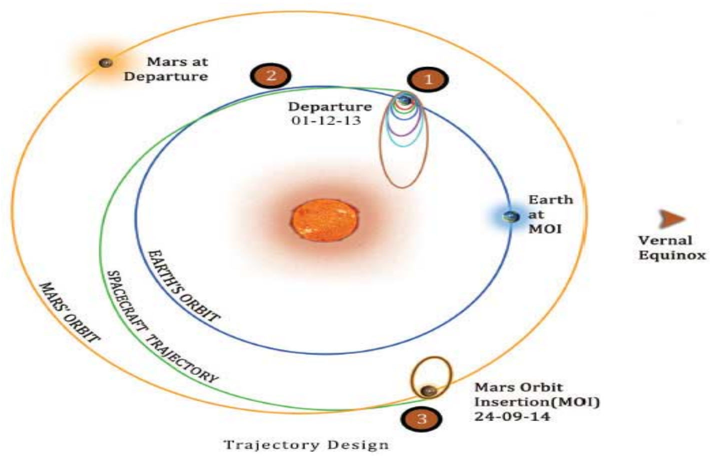

Figure 19: Mission Profile

Next, the spacecraft was transferred from Earth's orbit to Mars' orbit around the Sun, on a trajectory that forms an elliptical orbit (Hohmann transfer orbit) which is tangential to orbits of both planets. This was done by a final 7th burn to insert the spacecraft in a Heliocentric orbit, breaking away from Earth's gravity and on course to Mars. This journey from Earth to Mars can be divided into 3 phases:

(a) from the launch pad to the point where it is placed on transfer orbit, or the Geocentric phase
(b) journey on the transfer orbit, the Heliocentric phase
(c) from transfer orbit to final orbit around mars, the Areocentric phase.

12.2 Geocentric Phase

Orbit raising manoeuvres : Several orbit raising operations were conducted on 6, 7, 8, 10, 12 and 16 November

by using the spacecraft's on-board propulsion system and a series of perigee burns.

Figure 20: Apogee raising orbital manoeuvre

- The first orbit-raising manoeuvre was performed on 6 November 2013 at 19:47 UTC when the spacecraft's 440-newton (99 lbf) liquid engine was fired for 416 seconds. With this engine firing, the spacecraft's apogee was raised to 28,825 km (17,911 mi), with a perigee of 252 km (157 mi).
- The second orbit raising manoeuvre was performed on 7 November 2013 at 20:48 UTC, with a burn time of 570.6 seconds resulting in an apogee of 40,186 km (24,970 mi).
- The third orbit raising manoeuvre was performed on 8 November 2013 at 20:40 UTC, with a burn time of 707 seconds, resulting in an apogee of 71,636 km (44,513 mi).
- The fourth orbit raising manoeuvre, starting at 20:36 UTC on 10 November 2013, imparted a delta-v of 35 m/s (110 ft/s) to the spacecraft instead of the planned 135 m/s (440 ft/s) as a result of underburn by the motor. Because of this, the apogee was boosted to 78,276 km (48,638 mi) instead of the planned 100,000 km (62,000 mi).
- As a result of the fourth planned burn coming up short, an additional unscheduled burn was performed on 12 November 2013 that increased the apogee to 118,642 km (73,721 mi), a slightly higher altitude than originally intended in the fourth manoeuvre.
- The apogee was raised to 192,874 km (119,846 mi) on 15 November 2013, 19:57 UTC in the final orbit raising manoeuvre.

12.3 Heliocentric Phase

Trans-Mars injection : On 30 November 2013 at 19:19 UTC, a 23-minute engine firing provided the required Delta-v (Δv_1) to the spacecraft so that it escapes the Earth's Sphere of Influence and is placed into a Hohmann transfer orbit. The Hohmann Transfer Orbit or a Minimum Energy Transfer Orbit often uses the lowest possible amount of propellant which helps to cut down on energy costs. The spacecraft leaves Earth in a direction tangential to Earth's orbit and encounters Mars tangentially to its orbit. The flight path is roughly one half of an ellipse around sun. Eventually it will intersect the orbit of Mars at the exact moment when Mars is there too. This trajectory becomes possible with certain allowances when the relative position of Earth, Mars and Sun form an angle of approximately 44o. Such an arrangement recur periodically at intervals of about 780 days. Minimum energy opportunities for Earth-Mars occur in November 2013, January 2016, May2018 etc.

Calculation:

The total energy of the orbiter is the sum of its kinetic energy and potential energy, and this total energy also equals half the potential at the average distance a (the semi-major axis):

$$E = \frac{mv^2}{2} - \frac{GMm}{r} = -\frac{GMm}{2a} \tag{1}$$

$$\Rightarrow v^2 = \mu\left(\frac{2}{r} - \frac{1}{a}\right) \tag{2}$$

where,

v = speed of orbiter
μ = GM, the standard gravitational parameter of the primary body, the Sun
r = distance of orbiter from primary focus
a = semi-major axis

Therefore, the delta-v (Δv) required for the Hoffman transfer can be computed as follows,

$$\Delta v_1 = \sqrt{\frac{\mu}{r_1}}\left(\sqrt{\frac{2r_2}{r_1 + r_2}} - 1\right) \tag{3}$$

$$\Delta v_2 = \sqrt{\frac{\mu}{r_2}}\left(1 - \sqrt{\frac{2r_1}{r_1 + r_2}}\right) \tag{4}$$

where,

$$a = \frac{r_1 + r_2}{2} \tag{5}$$

r_1 = radius of the departure circular orbit corresponding to the periapsis of the Hohmann elliptical transfer orbit

r_2 = radius of the arrival circular orbit corresponding to the apoapsis of the Hohmann elliptical transfer orbit

Δv_1 = Delta-v required to enter the Hohmann transfer orbit from the Earth's orbit
Δv_2 = Delta-v required to enter the Mars' Orbit from the Hohmann transfer orbit

Thus,

$$\Delta v_{total} = \Delta v_1 + \Delta v_2 \tag{6}$$

And the time taken to transfer between the orbits by the Kepler's third law,

$$t = \frac{1}{2}\sqrt{\frac{4\pi^2 a^3}{\mu}} = \pi\sqrt{\frac{r_1 + r_2}{8\mu}} \tag{7}$$

Trajectory correction maneuvers : During the Heliocentric phase, the spacecraft travelled a distance of 660,000,000km, in which only 2 out of 4 planned trajectory correction maneuvers were performed to target the proper position for the important Mars Orbit Insertion Maneuver.

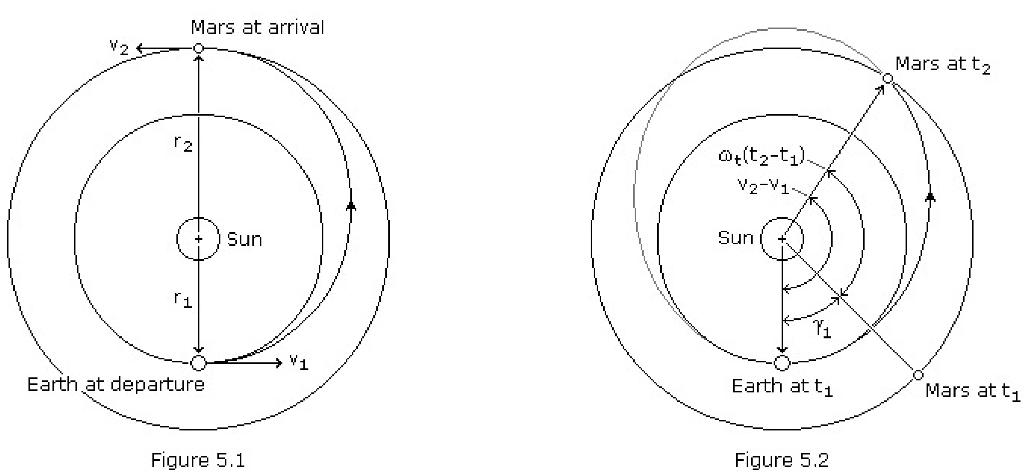

Figure 21: Heliocentric Phase

12.4 Areocentric Phase

Mars orbit insertion : The spacecraft arrives at the Mars Sphere of Influence (around 573473 km from the surface of Mars) in a hyperbolic trajectory. The 440-newton liquid apogee motor was test fired on 22 September at 09:00 UTC for 3.968 seconds, about 41 hours before actual orbit insertion. At the time the spacecraft reaches the closest approach to Mars (Periapsis), it is captured into planned orbit with a period of 72 hours 51 minutes 51 seconds, a periapsis of 421.7 km (262.0 mi) and apoapsis of 76,993.6 km (47,841.6 mi) around mars by impartingΔV-retro (Δv_2) which is called the Mars Orbit Insertion (MOI) manoeuvre and with this India achieved a roaring success in its very first attempt to place a spacecraft in an orbit around Mars.

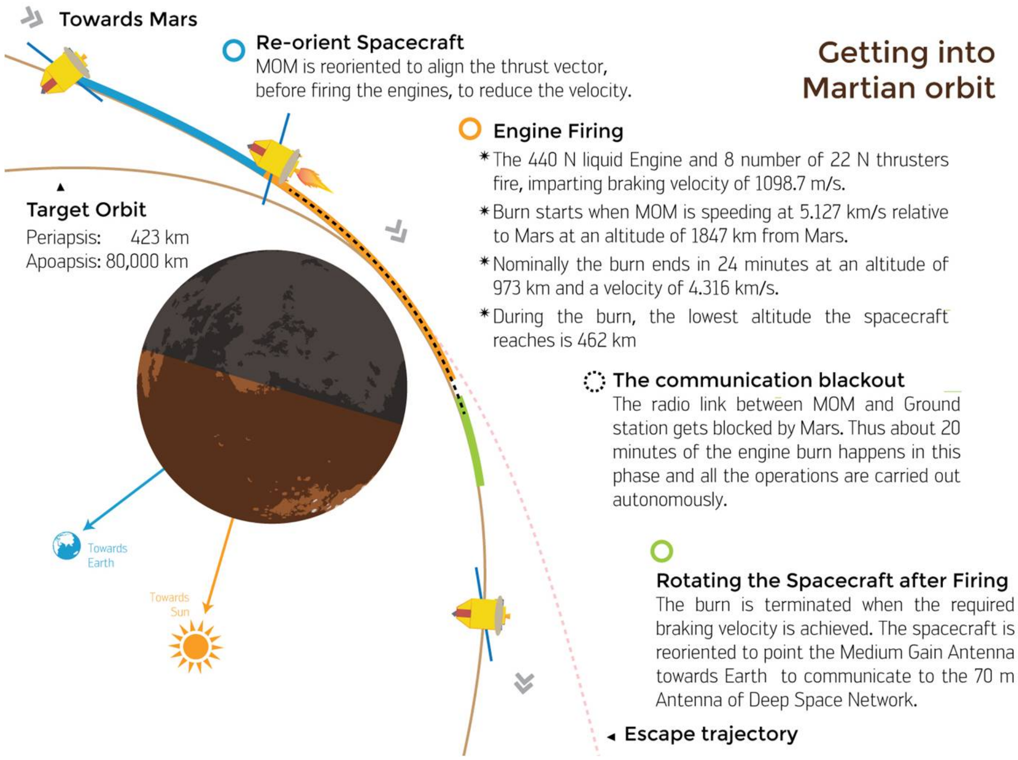

Figure 22: Areocentric Phase

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