Ulkayaan - A Rendezvous and Landing Mission to Comet 46P/Wirtanen

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by

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Abstract

46P/Wirtanen is a small short-period comet with a current orbital period of 5.4 years, with an estimated diameter of at 1.2 kilometres (0.75 mi). This report documents the preliminary mission design of rendezvous with comet 46P by a space probe, accompanied by a robotic lander which makes soft impact with the comet; this enables gathering of samples and data which help in the study of the comet and its origins. The launch window falls around April 2018 and the time for rendezvous is about 3 years. Selection of launch vehicle, in situ analysis and determination of landing site and soft landing strategies have been discussed in the report. FEM models of different components of lander and space probe have been created, and the required analysis and literature review have been presented in the report.

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Chapter 1

Introduction

1.1 Comet 46P/Wirtanen

Simply put, comets are snowy balls of frozen gases, rock and dust roughly the size of a small town. They usually have highly eccentric orbits and considerable inclination with the ecliptic. While approaching perihelion, they heat up and spew dust and gases, producing a visible atmosphere which can sometimes be larger than most planets, called the coma. The dust and gases form a tail that stretches away from the sun for millions of kilometers. Comets may not be able to support life themselves, but they may have brought water and organic compounds - the building blocks of life - through collisions with Earth and other bodies in our solar system. Thus comets are essential sources of information about the origins of the solar systems and provide valuable information about protoplanetary discs.

46P/Wirtanen is a small short-period comet with a current orbital period of 5.4 years. It

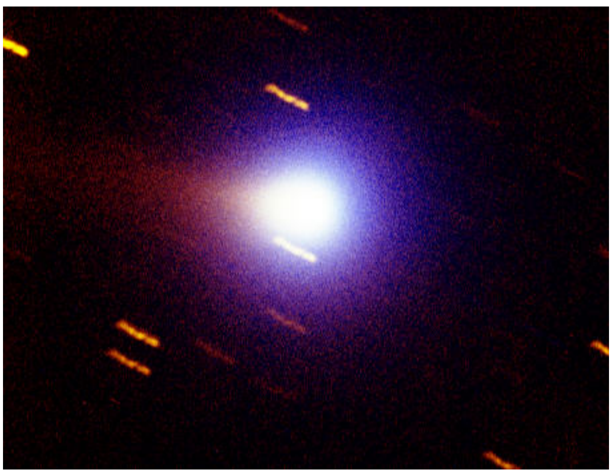


Fig. 1.1 Comet 46P/wirtanen

belongs to the Jupiter family of comets, i.e. which have an aphelion of around 5 to 6 A.U. Its diameter is estimated at 1.2 kilometres (0.75 mi).It has the following orbital perimeters:

- Orbit Eccentricity, e = 0.65920256
- Orbit inclination, i = 11.75713931^o
- Semi-major axis, a = 461,905,372 km
- Argument of perihelion, ω = 356.340205^o

It was discovered photographically on January 17, 1948, by the American astronomer Carl A. Wirtanen. The comet was the target for the proposed 2016 Comet Hopper mission, which reached the finalist stage in the Discovery program. It was also the original destination of the European Space Agency's Rosetta spacecraft mission, but an inability to meet the required launch window resulted in the change of the mission's target to comet 67P/Churyumov–Gerasimenko.

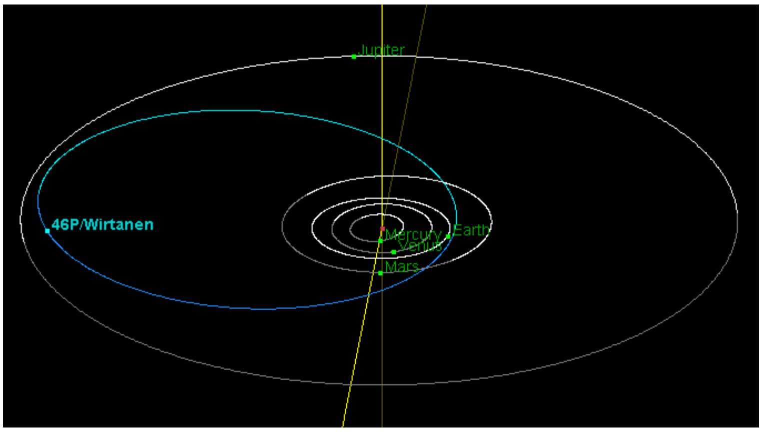


Fig. 1.2 Orbit of Comet 46P

1.2 Necessity for a cometary rendezvous

After the advent of spaceflight, the investigations into the structure and composition of comets has been more long than fruitful. Our current understanding of comets and their origin is based on the complementary information obtained by the missions like Vega 1 and Vega 2 for comet 1P/Halley and by the results provided by the more recent Deep Space 1 (comet 19P/Borrelly), Stardust (81P/Wild 2), Deep Impact (9P/Tempel 1), Stardust-NExT (9P/Tempel 1), and Extrasolar Planet Observation and Deep Impact Extended Investigation

(EPOXI) (103P/Hartley) cometary missions. The cometary nuclei and the surfaces, comae, and tails of Kuiper belt comets have been remotely observed and their nuclei have been approached. Ultraviolet radiation causes sublimation of cometary material from the cometary nucleus, providing great data for collection by these space probes. The general understanding of comets is furthermore and largely provided by remote observations in different wavelength regimes via ground-based or near-Earth space telescopes. Interstellar ice analogs have beeen produced in the laboratory by mimicking environments in which comets are believed to form and investigated extensively, also contributing to the information acquired about the chemical and physical properties of comets. However, the physical and chemical makeup of a cometary nucleus, which has been of paramount scientific interest since the first comet nucleus model was shaped by Fred L. Whipple, has not yet been observed directly. Today, considerable evolution and technological development in space experiment capabilities has opened the possibility of "taking the laboratory to the comet" rather than "bringing a sample back to Earth".

1.3 Mission Statement

The objective of the mission is to rendezvous with Comet 46P/Wirtanen, conduct in situ determination of landing site and send a robotic lander to the surface via soft landing. The launch will take place in the April of 2018 and the spacecraft will take around 3 years to rendezvous with the comet. The payloads have been selected to accomplish the following mission objectives:

- To study the nature of comet
- To study the details of formation of ice on comet
- To test William A. Bonner and Edward Rubenstein' hypothesis about amino acids

The mission has been divided into the following subsystems:

1. Orbit design
2. Launch vehicle selection
3. Orbiter (Structure and payloads)
4. Lander (Structure and payloads)
5. Thermal

6. Attitude Determination and Control Systems
7. Power systems
8. Communication systems and On-Board Computer

Chapter 2

Orbit Design

2.1 Trajectory Design

2.1.1 Objective

This chapter deals with the design of a trajectory that enables us to reach the comet 46P Wirtanen. The following approach was followed:

- Approach the problem with Hohmann Transfer.
- Design a precise trajectory using Lambert Conic approach.
- Compare the trajectory orbit's Δv with the JPL HORIZONS interface.
- Estimating the complete Δv budget.

2.1.2 Hohmann Transfer

The problem of trajectory design was initially approached with the Hohmann transfer for two non intersecting, coplanar, coaxial elliptical orbit. Here the coaxial and coplanar orbits were taken as an assumption, hence not the ideal case. As Hohmann Transfer is an explicit case of Lambert Problem.

The following results were obtained after doing the hand calculations for the Hohmann Transfer.

- This was done by feeding the values of the Earth's periapsis and Comet's Apoapsis into the Lambert Conic script and evaluating the end velocities.
- The total Δv came out to be 11.532 km/sec.

- This along with the plane change maneuvers and the necessary DSM(Deep Space Maneuvers) was far beyond our current propulsion system.
- So next we approach the problem with Lambert Conic

We came across the following limitations while using Hohmann Transfer.

- The orbit of the Earth and the Comet is neither coplanar nor coaxial with respect to the Earth's ecliptic, hence the above transfer can't be used for launch.
- The launch window as per this transfer requires specific location of Earth and the Comet in their respective orbits, hence the launch window is not always available.
- The Δv estimated by this approach was thus incorrect.

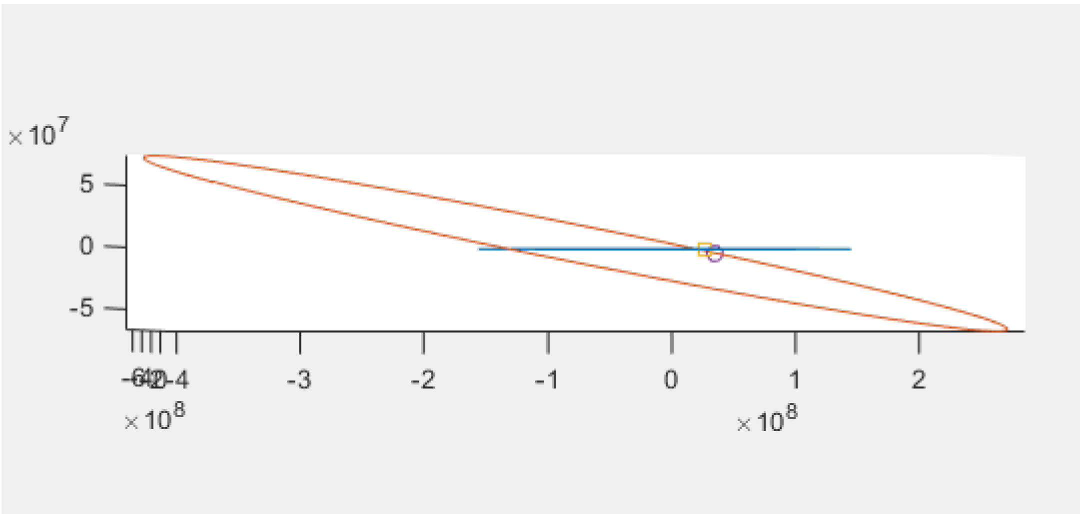


Fig. 2.1 The figure show the non co planarity of the two orbits.
The red colour shows the Comet's orbit and blue the Earth's orbit.

Lambert Conic Approach

To overcome the above mentioned limitations, a more precise trajectory was devised using the Lambert Conic Approach. A pre-coded Lambert Script named "Lambert-targeter for ballistic flight by (Izzo, and Lancaster, Blanchard and Gooding) was put in use. Following are the properties of the script lambert.m which is attached in the appendix.

- The script required the cartesian co-ordinates of the Earth and the Comet 46P which was requested from the JPL HORIZON web-interface, and was fed into the code.

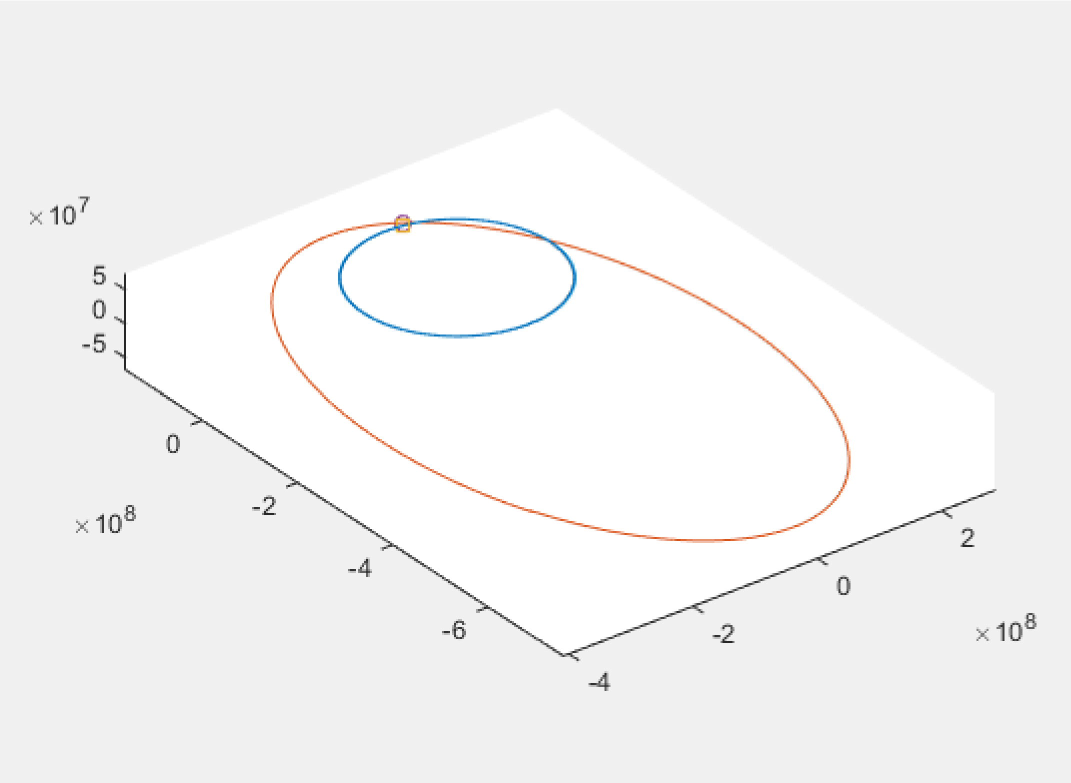


Fig. 2.2 The figure shows the non co- axial nature of the two orbits.

- The output of the script were two end velocities of the conic developed. In this approach, the gravitational force of the Sun was only considered and the same of the Earth and the comet was neglected.
- The end velocities named V1 and V2 were subtracted from the velocities of the Earth and the Comet respectively and then added to get the total Δv.
- The below mentioned Δv is excluding the velocity impulses required for the plane change maneuvers and the velocity impulse for putting the rocket in the EPO of 4000X200 kmXkm.

The following results were generated as code was run for different time of flight as being the variable.

As can clearly be seen, the minimum Δv along with the nearest launch window lies in the year 2018 as predicted by the code. The velocity impulses for the plane change maneuvers and the launch impulse have not been factored in the calculations. The plane change maneuvers cost was 3.451 km/sec at both for Earth as well as the Comet plane change. Hence, this approach also can't be put into consideration as our current propulsion system can't handle this much velocity impulse.

Table 2.1 Different Δv with the Time of Flight as variable.

Sl No.	Time of Flight	Total Velocity Impulse	Launch Window
	(Years)	(Km/sec)	(Date)
1	1	8.6025	03-02-2023
2	2	8.4715	21-01-2022
3	3	8.5316	13-03-2024
4	4	8.4796	31-03-2024
5	5	8.5487	08-03-2024
6	3.42	8.7029	12-12-2018

So we resort to the NASA's JPL trajectory Browser.

2.1.3 The Exact Design

The following approximations were made by the JPL browser for obtaining the optimized trajectory for rendezvousing the comet 46P.

- Only the Sun's gravitational force is taken into account.
- The script handles the problem using the Lambert conic approach.
- Lambert's problem for small-body heliocentric transfers can be stated as follows: given an Earth launch date and transfer duration to reach a small-body, determine the Keplerian transfer orbit connecting the two bodies.
- To determine which combinations of launch date and transit time yield low Δv, the two-dimensional time space is first discretized into all combinations of (tlaunch, tarrival) pairs for a given time step.
- For every pair, Lambert's problem is then solved to compute a transfer trajectory launching from Earth and arriving at the small-body at the given dates.
- The changes in velocity (Δv) required to leave Earth and arrive at the small-body can then be calculated.

The main advantage os using the browser trajectory over the modified script is that we get a reduced Δv using an Earth fly-by maneuver and one DSM(Deep Space Maneuver) to rendezvous with the comet.

The following table depicts the properties of the trajectory that was devised.

Including the safety margin of 0.2 km/sec the total Δv came out to be 8.74 km/sec.

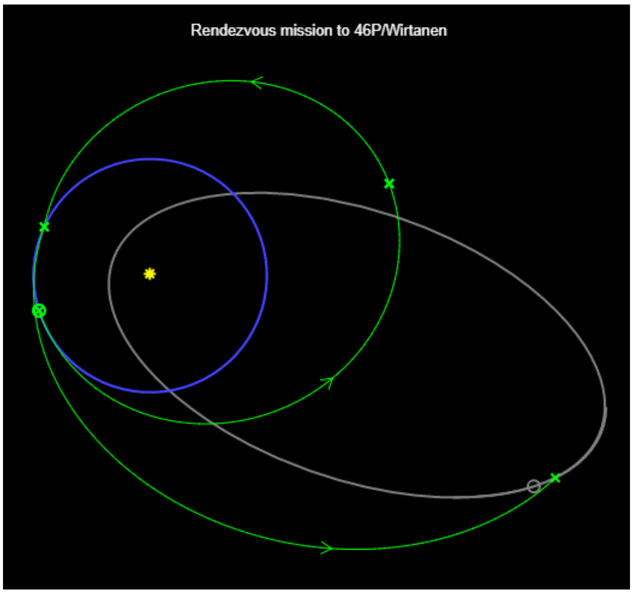


Fig. 2.3 The figure shows the exact trajectory as depicted by the JPL trajectory browser.

2.2 Post-rendezvous

2.2.1 Hibernation

Since the rendezvous happens near aphelion, the solar array does not generate sufficient power to maintain the spacecraft active due to the large distance to the Sun at aphelion. For this reason, a special hibernation mode is implemented, in which the majority of the spacecraft systems are deactivated. Some essential systems like tank heaters to avoid propellant freezing are to be kept on.
To generate minimal power, it is necessary to keep the spacecraft with the solar arrays pointing to the Sun. This is achieved by suspending the active attitude control systems but maintaining an axis perpendicular to the solar array always pointing toward the sun. Then the spacecraft is put to spin around this axis, where it will remain with the solar array reliably facing to the Sun, thanks to angular momentum conservation.

2.2.2 Approach

The approach strategy comprises of the following major steps:

1. Reduce relative velocity to 1 m/s. The total delta-V required for rendezvous is split in manoeuvres of decreasing size around the comet. The implementation errors of a manoeuvre are corrected in later manoeuvres.

Table 2.2 Overall Velocity Budget

Trajectory Itenary		
	Date	Velocity Impulse (km/sec)
Earth Departure	April-08-2018	4.36
1.01-year transfer		
DSM	April-14-2019	0.543
317-day transfer		
Earth Fly-by	Feb-25-2020	0.01
1.53-year transfer		
Comet Arrival	Sep-07-2021	3.83
3.42-year total mission		
post injection velocity impulse		4.39
Total velocity impulse		8.54

2. Improve comet ephemerides. Ground based comet ephemerides are not accurate enough for comet orbit insertion.

3. Bend progressively trajectory towards comet. Optical measurements of the comet with the orbiter cameras are required to determine relative trajectory.

The initial characterisation of the comet consists of maximum image coverage with variable viewing and illumination conditions to identify initial landmarks. The trajectory is designed such that the spacecraft remains on the illuminated side of the comet and comet characteristic like rotation, shape and mass can be estimated.

Since it is not possible to directly insert in elliptic orbit without precise comet mass knowledge, hyperbolic arcs are designed in the trajectory, beginning when the relative distance between orbiter and comet is 50,000 km. Each hyperbolic arc successively reduces distance between orbiter and comet, till the distance of least miss is reduced to 100 km.
Two triangular orbital trajectories, one at 100 km and other at 50 km, are designed around the comet with time period of around 3-4 days.

To find velocity budget of triangular maneuvers:

1. 100 km triangular orbit

    - Length of one side of triangle = 173.205 km
    - Time taken to cover one side of triangle = 1 day
    - Orbital velocity = 2.004 m/s
    - Assuming equilateral triangular orbit, δV to round one corner = 3.471 m/s

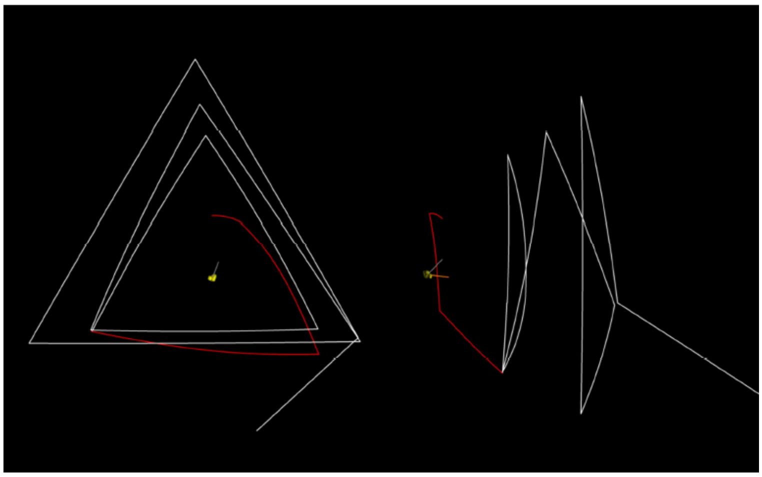


Fig. 2.4 Rendering of the triangular maneuvers around the comet

- Total δV = 10.41 m/s

2. 50 km triangular orbit

- Length of one side of triangle = 86.603 km
- Time taken to cover one side of triangle = 1 day
- Orbital velocity = 1.002 m/s
- Assuming equilateral triangular orbit, δV to round one corner = 1.735 m/s
- Total δV = 5.205 m/s

Total δV (including orbit reduction maneuver to 50 km) = 20 m/s approx

After this begins the global mapping phase to improve the navigation knowledge and extend landmark dataset. The spacecraft is put in quasi polar orbital arcs with radius less then 30 km. The details are as follows:

1. Two semi-circular arcs at 30 km from comet inclined 60 deg with respect to Sun direction.
2. One semi-circular arc on night side, decreasing to 20 km.
3. One 20 km circular orbit at terminator.

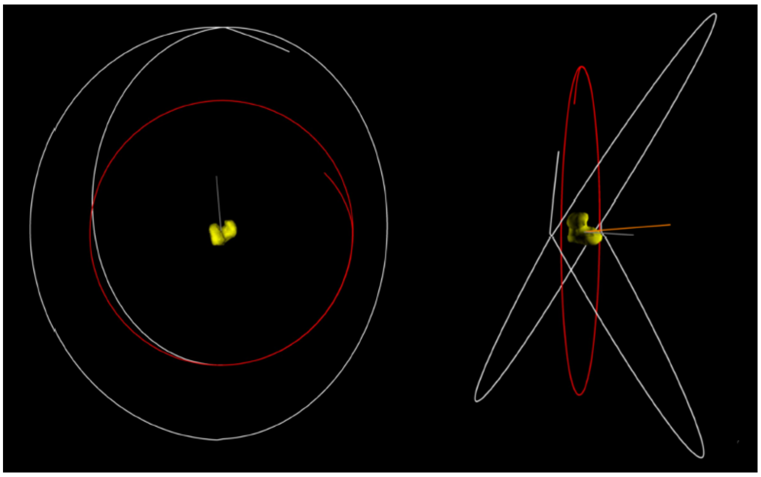


Fig. 2.5 Rendering of semi circular arcs and circular terminator orbit

Chapter 3

Launching the Spacecraft

One of the most crucial aspects in space mission design is the selection of the launch window. This is done by various optimization codes that minimize energy requirement to reach the target body, by using ephemerides. From such optimization, it is decided that the launch of the spacecraft should take place in the month of April, 2018.

The next obvious step is the selection of the launch vehicle and the launch location. In the design of this mission, we have assumed the launch to occur from SHAR, which is at a latitude of 13.8 degrees. In order to select a launch vehicle that meets the mission objective, it is necessary to determine the overall velocity budget. This has been done with the help of a code that uses the specific impulse and structural factor data of the various stages of multiple launch vehicles. See appendix.

The trajectory basically consists of launch and insertion into an Earth Parking Orbit (EPO). This is intended to be achieved by the penultimate stage of the vehicle. In order to ensure minimum energy transfer,the orbital plane of the comet and the spacecraft must be coplanar. We know that Earth is at an inclination of 23.4 degrees to the ecliptic while the comet orbit is at an inclination of 11.73813 degrees. Thus the inclination of the spacecraft with respect to Earth should be the difference, that is, 12.0187 degrees. But the safe azimuth range for SHAR is between 98-104 degrees and 130-135 degrees. This implies that for any launch from SHAR, the orbit inclinations can only be between 15.912-19.560 degrees and 41.9325-46.63089 degrees. The solution is then to launch the vehicle at an azimuth of 98 degrees and then incorporate a plane change maneuver.

Flexibility in the selection of the launch location would help us eliminate the requirement for a plane change maneuver.

Once the spacecraft is in the EPO (200x4000 km), the plane change maneuver is executed by the upper stage and the upper stage itself launches the spacecraft in a transplanetary orbit, hyperbolic with respect to Earth. Since the velocity impulse required to rendezvous with the comet is very high (8.74 km/s), a part of the velocity increment is achieved with the help of gravity assist maneuvers from then on. Mission Overview is as shown in the figure 3.1.

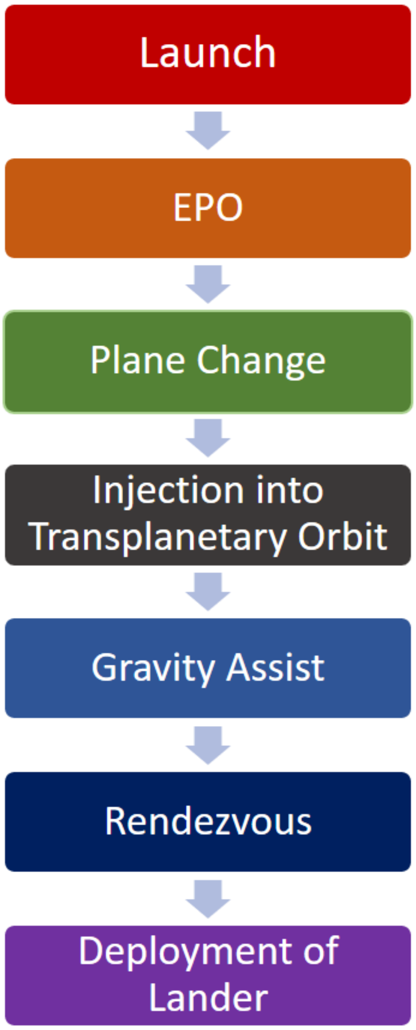


Fig. 3.1 Mission Overview

3.1 Optimum Launch Vehicle Selection

For selecting the optimum launch vehicle, the total velocity impulse required to launch the spacecraft into the interplanetary orbit is calculated and it is ensured that the difference between this value and the total velocity impulse capability of the spacecraft is minimized, with a certain weight assigned to it. Weights are also assigned to the payload fraction and the cost of launch and the vehicle that provides the least positive value for the product of the three quantities is chosen. These quantities are normalized and weighed by appropriate weights and the minimum of the product value of the velocity impulse difference, the reciprocal of the payload fraction and the cost (here, called as optimizer value) is considered. The weights assigned are as follows,

- $w_{cost} = 0.75$
- $w_{payload_fraction} = 0.05$
- $w_{vel_impulse_diff} = 0.20$

The weights assigned may be varied according to the designer's constraint. The flow chart explaining the code is as in fig.3.2.

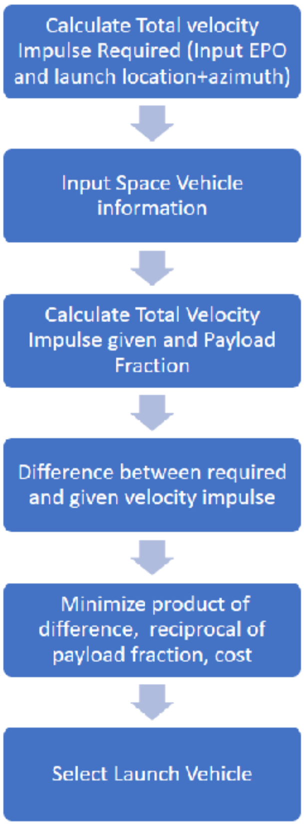


Fig. 3.2 Launch Vehicle Selection Code Flow Chart

It is to be noted that it is assumed that all the space vehicles considered in the code are capable of being transported and assembled at the launch location.

The launch vehicle specifications (please refer code, App.2) are obtained from various internet sources.

Spacecraft	Del_v1(km/s)	Del_v2 (km/s)	Del_v3 (km/s)	Del_v4 (km/s)	DelV_tot (km/s)	Cost (million US Dollars)	Payload Fraction	Optimizer Value
Atlas V	6.1978	7.04	0	0	13.2378	153	0.0058	0.0106
Delta IV Heavy	6.5459	7.4726	0	0	14.0185	400	0.0042	0.0214
GSLV mk II	3.2262	2.8309	5.3541	0	11.4111	47	0.0071	0.0075
GSLV mk III	4.8581	6.5365	0	0	11.3945	54	0.0047	0.0116
H-IIB	2.9656	5.6219	0	0	8.5876	112.5	0.0056	0
Long March 5	6.5863	4.9213	0	0	11.5076	70	0.0036	0.0159
Proton-M	2.6307	3.3304	3.0758	4.9404	13.9774	65	0.0043	0.0111

Fig. 3.3 Results of Launch Vehicle Selection Code

Based on the output given (Fig. 3.4) by the code, it is decided that GSLV Mk II is to be used as the launch vehicle for the mission.

Chapter 4

Propulsion Systems

4.1 Chemical Propulsion versus Electrical Propulsion Systems

The propulsion system aboard a spacecraft mainly serves two purposes - first, for velocity increments and propagation and second, for attitude control and trajectory correction. It is often one of the most crucial aspects that dictates the success of a mission. The system can either consist of chemical or electric propulsion. Chemical propulsion often consists of mono-propellant, bipropellant solid fuel rockets while the most commonly used electric propulsion systems are ion thrusters and Hall effect thrusters.

Efficiency of a propulsion is not gauged by its thrust capability but by the specific impulse it is capable of providing. Specific impulse is the thrust obtained per unit weight of the propellant. Conventional chemical propellant systems produce thrust by expelling finite amounts of fuel in finite amounts of time. On the other hand,in an electrical system acceleration continues throughout the flight by ejection of minute amounts of propellants. Chemical engines produce high thrusts but low specific impulses owing to the relatively large amounts of propellant ejected, impulsively. Electrical propulsion systems, on the other hand, produce low thrusts but high specific impulses due to the low mass but high velocities of ejected propellant. The low amounts of thrust can be compensated by operation for long durations of time, building larger total impulse.

While electric propulsion is the undoubtedly the obvious choice, it has quite a few complications associated with it. Electric systems can only be used in vacuum owing to their low thrust capacity. This might hamper the propulsion capability in our mission, in case of interaction with the comet coma. Electrical propulsion also requires power from either the solar arrays or if safety issues can be dealt with, a nuclear electric power system.

This increases the total power requirement as well as the complexity of design. In addition, the continuous acceleration of the spacecraft due to the continuous operation of such a system would make an already complicated trajectory, even more difficult to deal with from a monitoring and control point of view. Thus, to ensure simplicity of design and the overall mission, a chemical bi-propellant propulsion system has been chosen for spacecraft propulsion.

4.2 Using the Propulsion System

The launch vehicle injects the spacecraft into an earth parking orbit. The orbit of the comet is heliocentric and the velocity impulse required to rendezvous with the comet is enormous. This requires huge amounts of propellant. This in turn increases the mass of the spacecraft and makes the mission unfeasible. The solution to this problem is making use of gravity assists to provide necessary velocity increments to the spacecraft and use the spacecraft propellant system for further orbit correction and rendezvous maneuvers.

As in the Rosetta mission, the spacecraft will thus undergo gravity assists, trajectory correction maneuvers and rendezvous maneuvers. The velocity increment requirement for these maneuvers falls roughly around 2100 m/s. In addition, in case of deep space hibernation, the spacecraft is stabilized by spinning around an axis perpendicular to the solar panels, pointing towards the sun, in order to ensure continuous power supply. These spin-up and spin-down maneuvers shall require the propulsion system as well.

In addition, comet orbit insertion is not easy since ground based comet ephemerides are not very accurate. The main objectives are to reduce the relative velocity of the comet and the spacecraft to within 1 m/s and bend the trajectory progressively towards the comet. The onboard cameras provide active monitoring while attitude control and propulsion systems are used for orbit insertion.

4.3 System Specifications

The propulsion system primarily consists a central cylinder that accommodates two propellant tanks - Mono Methyl Hydrazine (MMH) as fuel and Nitrogen Tetroxide (NTO- MON1) oxidizer. A small amount of NO is added to the oxidizer to inhibit the corrosion of the welded Ti6AlV4 tanks. The total delta V capability should be around 2300 m/s and thus the propellant mass comes out to be nearly 1700 kg.

For purposes of system reliability, there is no single engine but 12 pairs of 10 N thrusters for velocity increment and attitude control requirements. These pairs consist of a prime and a redundant thruster. While 8 of these pairs located near the corners of the cuboidal spacecraft are used for attitude control, the remainder 4 pairs are used for velocity increment requirements. This design is inspired from the Rosetta Mission.

Taking into consideration, the duration of the mission and amount of propellant, it is necessary to pressurize the tanks to keep the pressure in the operational range of the thrusters. The thruters show lower specific impulse at lower pressures. In order to ensure efficient functioning, a He based pressurization system is required, equipped with sensors, valves and regulators, to ensure sufficient pressurization at all times.

The nominal mixture ratio expected is around 1.2. For low pressures and oxidizer rich mixtures, thrusters may enter unstable mode. At nominal pressure of 17 bar and the nominal mixture ratios, the thrusters have a capacity of delivering specific impulse of around 292 s. In order to ensure no time delay between the supply of fuel and oxidizer, a single regulator may be used.

Chapter 5

Orbiter

5.1 Payloads

5.1.1 ALICE (Ultraviolet Imaging Spectrometer)

ALICE analyses gases in the coma and tail and measures the comet's production rates of water and carbon monoxide/dioxide. It also provides information on the surface composition of the nucleus. ALICE will help us determine where comet C-G came from, what it is made of, and how its nucleus, coma, and tails interact. ALICE is also a tool for detecting other components in the comet that are critical to the development of molecules that nurture life on Earth. The far-UV, which ALICE excels at observing, is ideal for seeing water, carbon monoxide, and carbon dioxide, three key molecules formed in the coma by the comet's nucleus. ALICE will also measure the abundance of carbon, hydrogen, oxygen, nitrogen, and sulfur atoms in comet C-G's coma. In other studies of comets, scientists have seen significant amounts of the first four atoms. Along with sulfur, these atoms are important to lipids, carbohydrates, and proteins - complex organic molecules important for life on Earth. ALICE is the only remote-sensing instrument on orbiter that can detect these key atoms if they exist in the comet's atmosphere.

Passband	700-2050 Å
Spectral resolution	4-8 Å point source, 8-12 Å extended source
Mass/power	3.0 Kg / 4 W

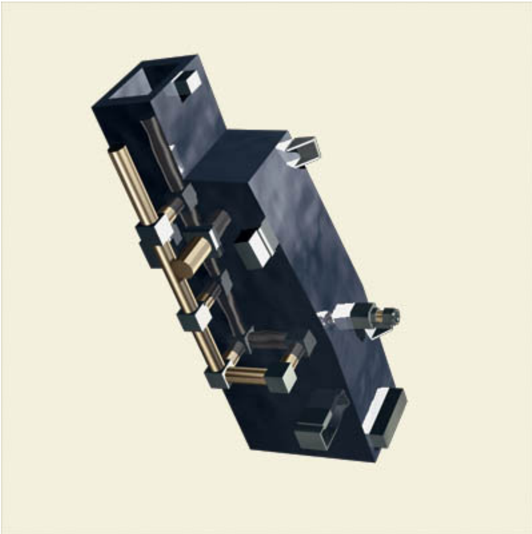


Fig. 5.1 ALICE

5.1.2 CONSERT(Comet Nucleus Sounding Experiment by Radiowave Transmission)

CONSERT probes the comet's interior by studying radio waves that are reflected and scattered by the nucleus. Its an In-situ instrument. Radio waves from the CONSERT experiment on the orbiter travel through the nucleus and are returned by a transponder on the lander.

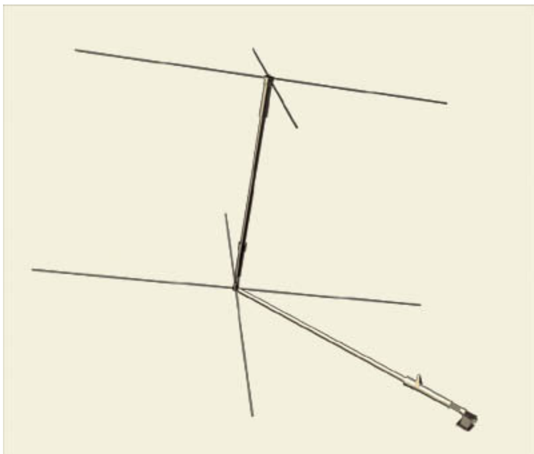


Fig. 5.2 CONSERT

5.1.3 RPC (Rosetta Plasma Consortium)

In this instrument, five sensors measure the physical properties of the nucleus, examine the structure of the inner coma,monitor cometary activity, and study the comet's interaction with the solar wind. The suite weighs only 7 kg, and consumes less than a quarter of the power of a light bulb. Scientists will use the RPC to characterize the electromagnetic forces that drive the high energy and complex environment of the comet's coma, which develops when the comet approaches the sun.
The Ion and Electron Sensor (IES) is one of five instruments that make up the Rosetta Plasma Consortium (RPC) suite. IES will observe the charged particles. IES will investigate the coma's developing layers or boundaries, focusing especially on the "inner shock" layer. In this layer, high-velocity particles spewing from the nucleus smash into lower-velocity particles from the coma's middle layer that have been slowed by the solar wind. IES will also monitor those ions and electrons in the solar wind that manage to find their way through the coma into the inner regions of the comet's atmosphere. The coma can grow to a million times the size of the comet's nucleus. IES cannot measure charged particles from afar. It is an in situ (in place) instrument, as is its companion, the magnometer; they must be where the particles can actually strike them.

Fig. 5.3 RPC (Rosetta Plasma Consortium)

Energy range	1 eV - 30 keV
Energy resolution	4 percent
Field of view	2.8 π steradians
Mass/power	1 Kg / 1.85 W

5.1.4 MIRO (Microwave Instrument for the Rosetta Orbiter)

MIRO is used to determine the abundances of major gases, the surface outgassing rate and the nucleus subsurface temperature. MIRO will be able to determine the surface and subsurface temperatures of comet C-G's nucleus. This will be particularly important as the comet begins its active phase and jets of gas open up on the nucleus surface. MIRO will measure that rate at which the comet jets gas from the nucleus. Combined with detection of the comet's surface-and-subsurface will these outgassing rates will help scientists understand the rate at which heat from the sun is conducted to the comet's interior, which will provide them with more clues about the internal structure of the comet's nucleus. Scientists will also be looking for evidence of pockets or surface features, such as valleys, in the nucleus that are more likely to produce jets of gas. MIRO will also track the speed of these gases using Doppler shift, which is like tracking the movement of a train by listening to the changing pitch of its whistle.

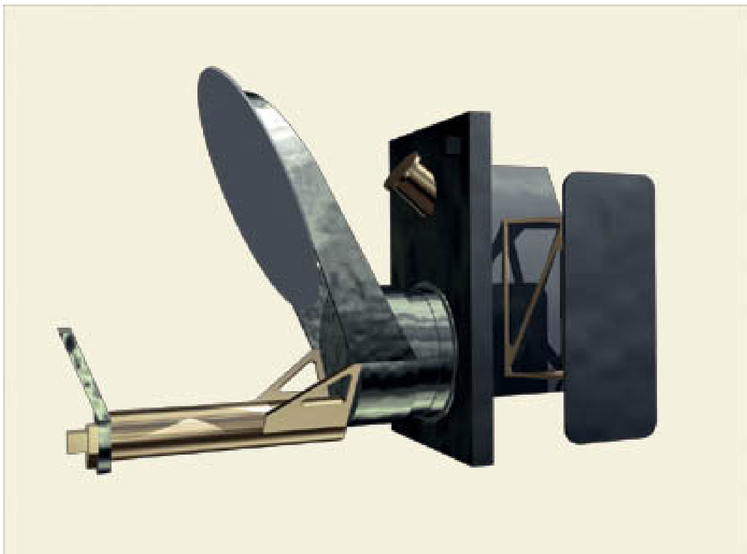


Fig. 5.4 MIRO

Passband	190 GHz, 1.6 mm (mm wavelengths)
	562 GHz, 0.5 mm (sub-mm wavelengths)
Spectral resolution	< 100 kHz (sub-mm)
Spatial resolution	75 m (mm); 25 m (sub-mm)
Field of view	< 22 arc minutes (mm); < 8 arc minutes (sub-mm)
Radiometric sensitivity	
1 K (continuum) Mass/power	19.9 Kg / 43 W

5.1.5 ROSINA (Rosetta Orbiter Spectrometer for Ion and Neutral Analysis)

ROSINA contains two sensors which will determine the composition of the comet's atmosphere and ionosphere, the velocities of electrified gas particles, and reactions in which they take part. It will also investigate possible asteroid outgassing. ROSINA has three separate subsystems; one is the DFMS, a Double Focusing Mass Spectrometer, for which NASA contributed a portion of the electronics

Fig. 5.5 ROSINA

5.1.6 VIRTIS (Visible and Infrared Thermal Imaging Spectrometer)

VIRTIS maps and studies the nature of the solids and the temperature on the surface of the nucleus. Also identifies comet gases, characterizes the physical conditions of the coma and helps to identify the best landing sites.

The primary scientific objectives of the VIRTIS :

- study of the cometary nucleus and its environment.
- determination of the nature of the solids in the nucleus surface.

Fig. 5.6 VIRTIS

- identification of gaseous species.
- characterization of physical conditions of the coma.
- evaluation of the nucleus surface temperature.

5.2 Satellite Bus or Spacecraft Bus

he bus is the infrastructure of a spacecraft, usually providing locations for the payload (typically space experiments or instruments). Bus-derived satellites are usually customized to customer requirements, for example with specialized sensors or transponders, in order to achieve a specific mission.

A bus typically consists of the following subsystems:

1. Command and Data Handling System
2. Communications system and antennas
3. Electrical Power System
4. Propulsion
5. Thermal control
6. Attitude Control System (ACS)

7. Guidance, navigation, and control (GNC) System
8. Structures and trusses
9. Life support (for crewed missions).

5.2.1 Propulsion Tanks

The propulsion tanks rest at or near the spacecraft's center of mass to avoid shifting of the center of mass as the propellant is used. Engines for translation control are aligned to thrust through the center of mass.

Propulsion lines and tanks must be protected from freezing, usually by thermostatically controlled guard heaters. Power for these heaters is included in the thermal subsystem. Electrically operated solenoid valves control propellant flow to the thrusters, but we account for their power in the ADC subsystem.

5.2.2 Attitude Control System

Engines for attitude control thrust tangentially and are mounted as far away from the center of mass as possible to increase the lever arm and thus increase the torque per unit thrust. Antenna must point toward the Earth, we need to control its attitude about 2 horizontal axes. Either spin stabilization or 3-axis control using sensors and torquers can be used to control the spacecraft's attitude. Possible sensors include Earth, Sun and star sensors, gyroscopes, magnetometers, and directional antennas. Torquers include gravity gradient, magnetic, thrusters, and wheels. Wheels include variable speed reaction wheels; momentum wheels, which have a nominal nonzero speed and therefore provide angular momentum to the spacecraft; and control moment gyros, which are fixed-speed gimballed wheels.

5.2.3 Communications system and antennas

The communications subsystem receives and demodulates uplink signals and modulates and transmits downlink signals. The subsystem also allows us to track spacecraft by retransmitting received range tones or by providing coherence between received and transmitted signals, so we can measure Doppler shift.

Communication access to a spacecraft requires a clear field of view for the spacecraft antenna. It also requires sufficient received power to detect the signal with acceptable error rate.

Consideration	Implication
Access	Ability to communicate with the spacecraft requires clear field of view to the receiving antenna and appropriate antenna gain
Frequency	Selection based on bands approved for spacecraft use by Intemational agreement Standard bands are S (2 GHz). X (8 GHz). and Ku (12 GHz)
Baseband Data	Data bandwidth and allowable error rate determine RF power level for communications

Table 5.1 System Considerations for Design of Communications Subsystems

5.2.4 Command and Data Handling System

The command and data handling subsystem, (CDH), receives and distributes commands and collects, formats, and delivers telemetry for standard spacecraft operations (housekeeping) and payload operations. kb/s to 500 Mb/s) and storage of payload data. The CDH subsystem may include encryptors, decryptors, a sequencer or timer, a computer for data processing, and equipment for data storage. It interfaces with the communications subsystem from which it receives commands and to which it sends the formatted telemetry stream. It also delivers commands to and receives telemetry from the other spacecraft subsystems and may have similar interfaces with the payload.

5.2.5 Thermal Control

The thermal design of a spacecraft involves identifying the sources of heat and designing paths for transporting and rejecting heat, so components will stay within required temperatures. The sources of heat include solar radiation, Earth-reflection and infrared radiation, and electrical energy dissipated in the electrical components. We can control the temperatures of compartments for conventional electronics by coating or insulating their outer surfaces. We select these coatings to strike a balance between the heat absorbed and the heat radiated to space. The coatings include various paints and tapes, and second surface glass mirrors. The weight of such coatings is almost independent of the quantity of heat dissipated and seldom exceeds 4 percent of the spacecraft dry weight. The thermal coatings, particularly insulation, can close out compartment openings and may also shield components from electromagnetic radiation. A typical medium-sized spacecraft (1,000 W) consumes 20 W in the thermal subsystem plus any power required for special thermal control. In most cases, heaters can operate from primary power.

5.2.6 Structure

Considering mass as major parameter aluminum is taken material for bus structure. For the present situation monocoque have been selected.

Axial frequency limit	25 Hz
Lateral frequency limit	10 Hz
Youngs modulus of 7075 Aluminum	$71 * 10\ ^{9}N/m^2$
Length of cylinder	2 m
Radius of the cylinder	1.5 m
Poisson's Ratio	0.33
Density	$2.8*10^3 kg/m^3$
Ultimate tensile Strength	$524*10^6 kg/m^2$
Yield Tensile Strength	$448*10^6 kg/m^2$
Axial Load factor	5
Lateral Load factor	3
Bending moment factor	3

Table 5.2 Parameters chosen for Structure thickness

Ultimate load	18191.25N
Required thickness	3.6835e-06m
Buckling Stess	$13865.5786N/m^2$
Critical Buckling Load	0.64446N
Accepted thickness	0.0026497m
Mass of structure	139.849kg

Table 5.3 Results obtained using information avaliable Space Mission Analysis and Design by J.Larson

The obtained mass for the structure is around 140kg which do not include support structure mass for solar panels and solar panels.
The method used for calculating the structural mass is programmed and shown below.

5.3 Orbiter and Lander Separation

Separation of lander from orbiter is a crucial thing is this mission. The separation method we chose should not have any shocks on orbiter and lander. This should impart low impulse and the control-ability on the the mechanism should be accurate. Any explosive cannot be used for this separation.

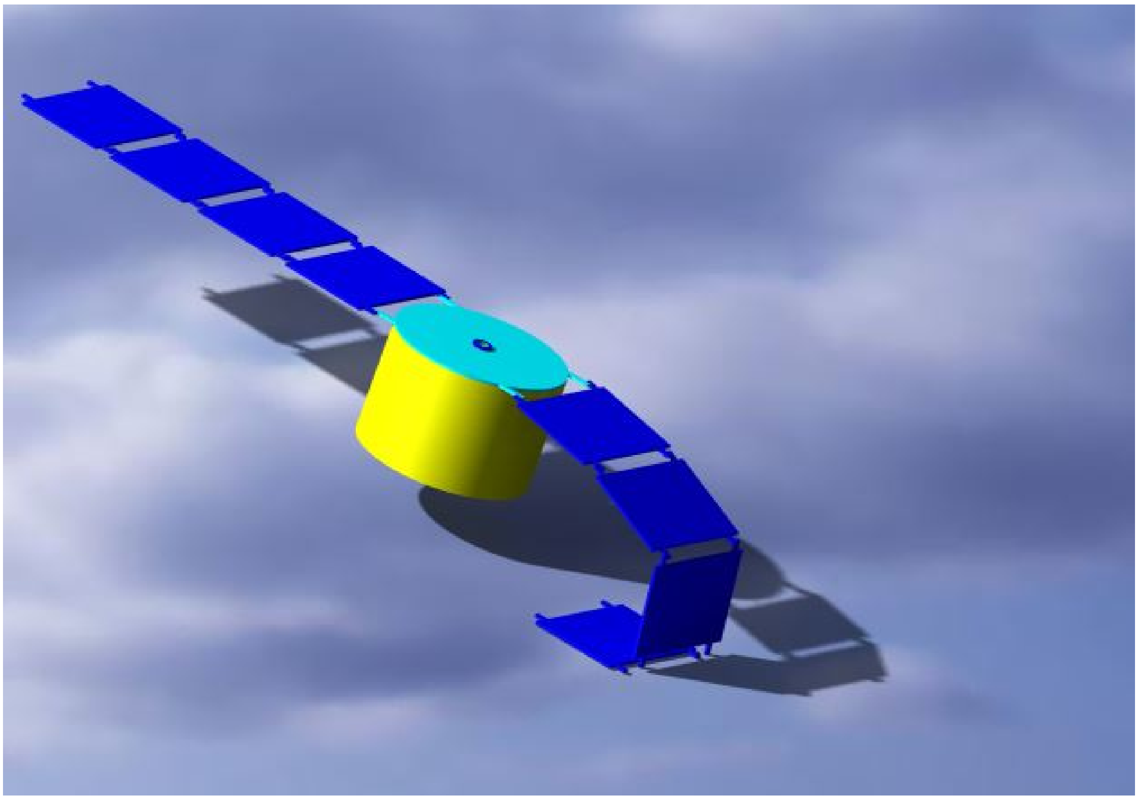


Fig. 5.7 Orbiter with Solar arrays

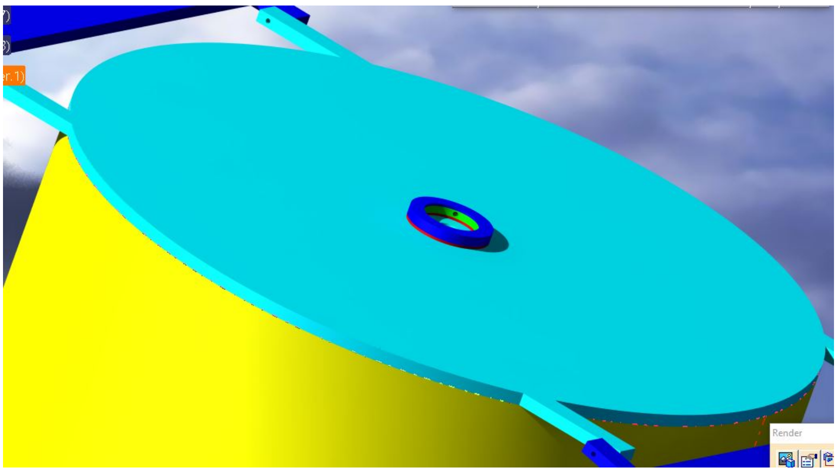


Fig. 5.8 Orbiter with separation mechanism on it

Many techniques have been used to provide the impulse to give a specified relative momentum, and hence separation velocity, between separating bodies. However, stage ignition, auxiliary rockets, thrust reversal, and springs have been the most frequently used techniques in vehicles made thus far. A separate device is therefore usually used to provide the impulse necessary for separation.

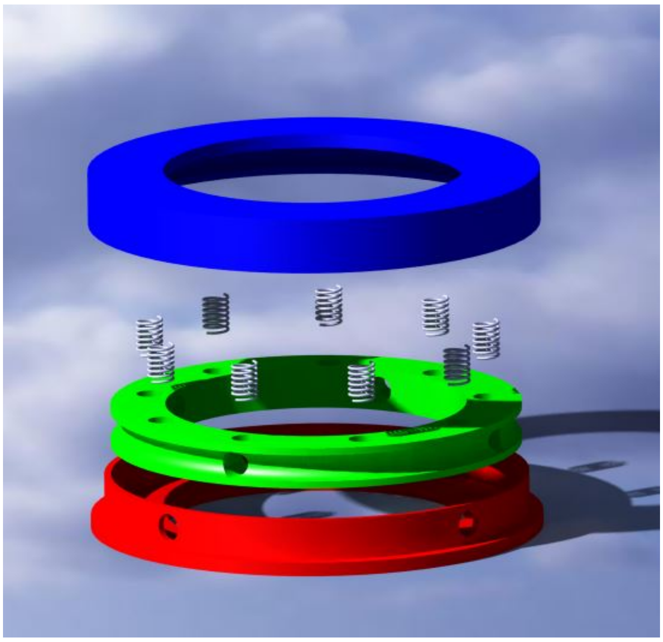
In terms of cost and reliability, springs are ideal separation-impulse devices when there are stringent tip-off requirements such as payload separation. Helical compression springs are the most common type of springs used for separation. A single spring is adequate for separation when the bodies are spinning, but three or more are used for nonspinning separations or when the allowable tip-off errors are small. As many as 30 springs have been used to reduce the out-of-tolerance effects of a given spring on tip-off rates and to lower the spring-stroke length necessary to induce the desired velocity change.

5.4 Ball Lock Separation

The ball lock separation mechanism is used in space vehicles for stage separation. Space vehicle includes artificial satellite and spacecraft having multistage rockets as their carriers. Parts of such space vehicle must be separated during flight to jettison stages and components that are no longer needed, to uncover equipment, or to deploy payloads. For a mission to be successful, the separations must occur at the correct times of flight and with minimum changes in the desired attitudes and rotational rates. There must be no impact between the separating bodies, no detrimental shock loads induced in the structure, and no excessive or harmful debris. Basically micro-satellite separation system based on "Ball Lock" release mechanism developed by ISRO for deploying micro-satellite up to 150 kg mass has been successfully used in PLSV. It functions by releasing a preloaded ball locked joint between two rings by rotating a ball retainer ring using pyro assisted thrusters.

Ball lock separation system is designed to separate out the nose cone fairing from payload module system. The ball lock system is basically works on a tongue and groove joint principle. The system can be actuated by hydraulically or pneumatically prior to pyro assembly as an acceptance test. The system characterized by good joint stiffness, light weight construction, tunable jettisoning velocity.

The ball lock separation mechanism is designed by considering the two basic criteria first one is it should withstand the structural loads for the given static and dynamic conditions without excessive deformation and mechanism should be dynamically functional. Separation mechanisms shall also be designed to separate structural segments only on command, without re-contact of the segments, and without causing damage or contamination, and without imparting excessive position errors to the continuing segment. Separation mechanisms shall be designed for reliability of performance proportionate with the specified overall system reliability of the vehicle.



The system consist of upper(blue) and lower stage(red) adapter ring held together held together by steel balls which is in turns are held by a retainer ring(green). The retainer ring is provided with escape tapered hole for the balls. In locked condition, the holes in retainer ring have given an angular offset. During actuation the retainer ring is rotated by pyro thrusters which makes angular offset to zero such that ball escape through holes present in the retainer ring. Now the compressed helical springs positioned between the flanges impart the required differential velocity. The upper adapter ring is connected to the lander and lower adapter ring is attached to orbiter and the retainer ring stays with lower adapter ring.

Ball lock separation mechanism when used along with helical springs can produce impulse of 0.06 to 1.8 meters per sec to the lander depending on the requirement. But the impulse cannot be adjusted on it is designed because it depends on the number of the springs and the characteristics mainly which are fixed once design is made.

5.5 Solar Panels and its Deployment

A solar panel works by allowing photons, or particles of light, to knock electrons free from atoms, generating a flow of electricity. Solar panels actually comprise many, smaller units called photo-voltaic cells. Photo-voltaic simply means they convert sunlight into electricity. Many cells linked together make up a solar panel.

Each photo-voltaic cell is basically a sandwich made up of two slices of semi-conducting material, usually silicon.

Solar panels power supply depends on the solar radiation on the panel that is directly proportional to the area exposed to the sun light. It is defined by the Solar constant. The solar constant, a measure of flux density, is the mean solar electromagnetic radiation (the solar irradiance) per unit area that would be incident on a plane perpendicular to the rays, at a distance of one astronomical unit (AU) from the Sun (roughly the mean distance from the Sun to the Earth). The solar constant includes all types of solar radiation, not just the visible light. It is 1.361 kilowatts per square meter. Solar constant is inversely proportional to the square of the distance from the sun. At 5.2 AU from the sun the solar constant

$$SC = 1.361 * 1^2/5.2^2 * 1000$$

$= 50.33W/m^2$.For 400 watts.

$$Area = 400/50.33 = 7.947m^2$$

If we assume that the solar panels had the 25 percent efficiency, then area of solar panels

$= 7.947/0.25 = 31.97m^2 = 32m^2$

The solar panels cannot be taken as a single sheet in to the launch vehicle. Launch vehicle has limited space so we have to fold the solar panels. Possible structural schemes are outlined below.

5.5.1 Single Frame, Double Concertina Fold of Membrane

The frame folds transversally and then longitudinally into a 600 mm by 800 mm stack; this requires several straight folds in the membrane. Note that the axes of the hinges in each side of the frame are perpendicular to that side and all hinges are coplanar; the corners of the frame are rigid, i.e. there are no hinges. All hinges would be self-locking hinges.

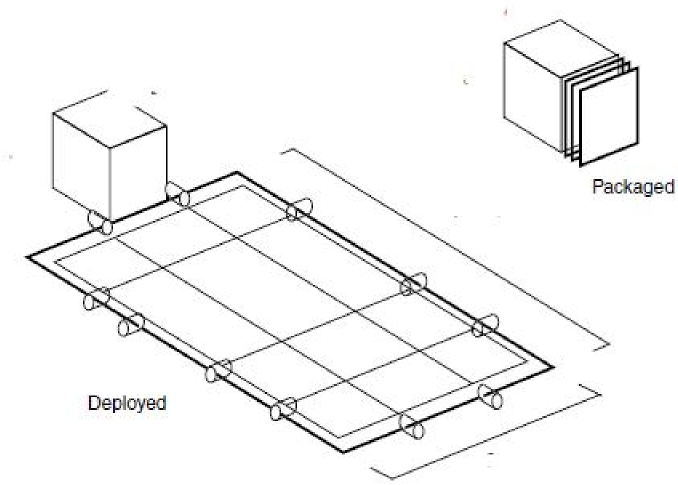


Fig. 5.9 Double concertina fold scheme

5.5.2 Fold-and-Roll-up Blanket with Deployable Boom

The blanket supporting the cells is tensioned by two end bars, each with two articulations. The blanket is folded over twice, together with the end bars, before being rolled over a roller. The deployable backbone of this structure is a tubular boom. Lack of sufficient torsional stiffness could be a problem.

5.5.3 Two Foldable Frames

The frame folds as a bundle of approximately parallel bars and the blanket is packaged between the bars. A frame with seven hinges will have mobility of one, and hence would exhibit well-controlled deployment behavior without the addition of any coupling devices. However, a much greater number of hinges is required to achieve the required packaging.

5.5.4 Split Solar Arrays

Arranging solar arrays on either side of the spacecraft. The area we have is 32 m^2 and if we chose each panel to be 2.5m x 1.6m we require 8 such panels. Four fold on each will

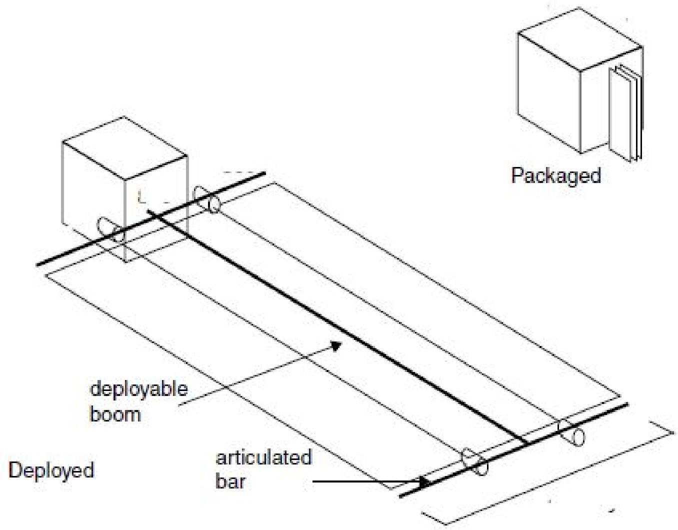


Fig. 5.10 Double fold and roll-up solar array.

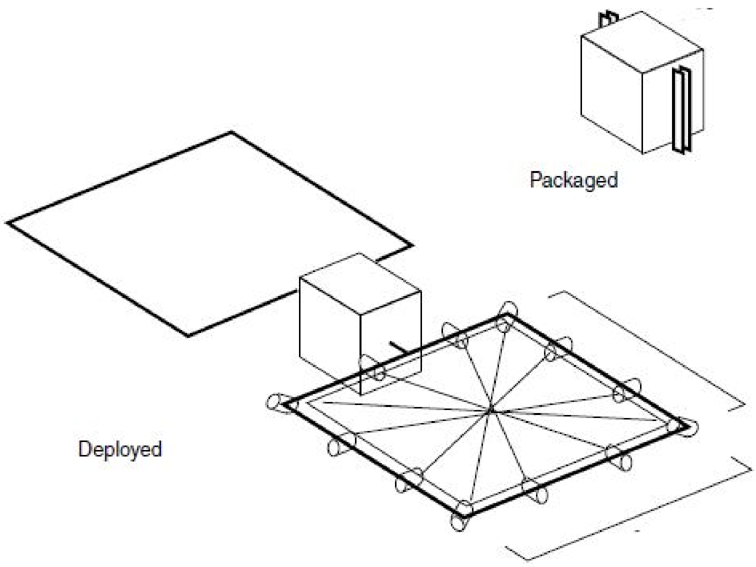


Fig. 5.11 Solar arrays that fold into two bundles of bars

be placed. By arranging the rectangular frames sideways the width of the structure can be increased. Three fold is shown as example in the below figure.

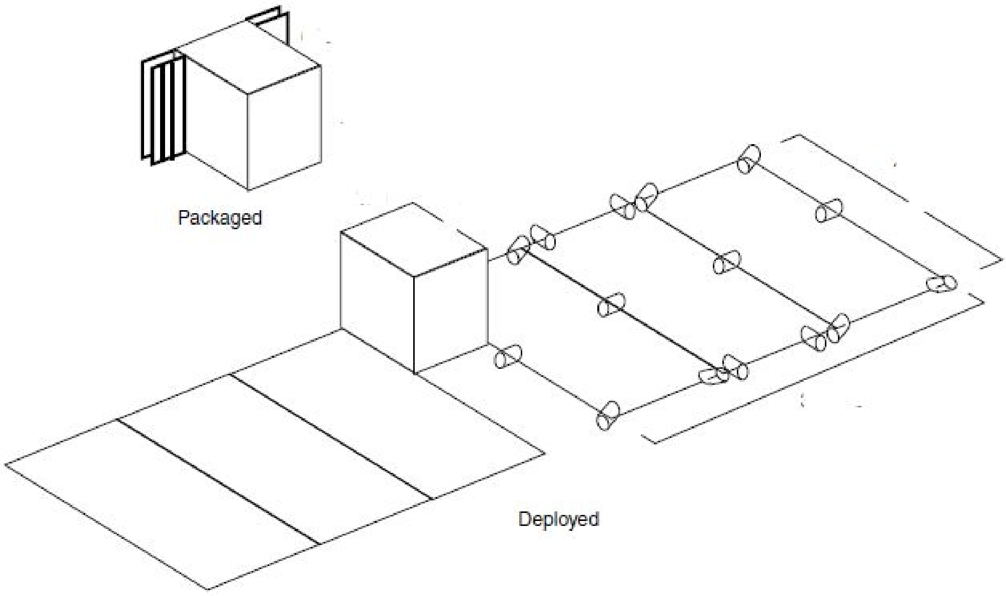


Fig. 5.12 Split Solar arrays three fold

Split four fold is used because easy to deploy and and manufacture the panels of such shape but volume will be constraint. But Ariane 5 had already carried Rosetta with 14 meters wing length with 5 folds. So it is possible to chose such deployment for Solar Arrays.

Chapter 6

Lander

Once orbiter is in position, the landing on comet 46P/Wirtanen will be attempted. The lander will be ejected from the orbiter using ball screw ejection mechanism and will achieve landing following a ballistic trajectory.

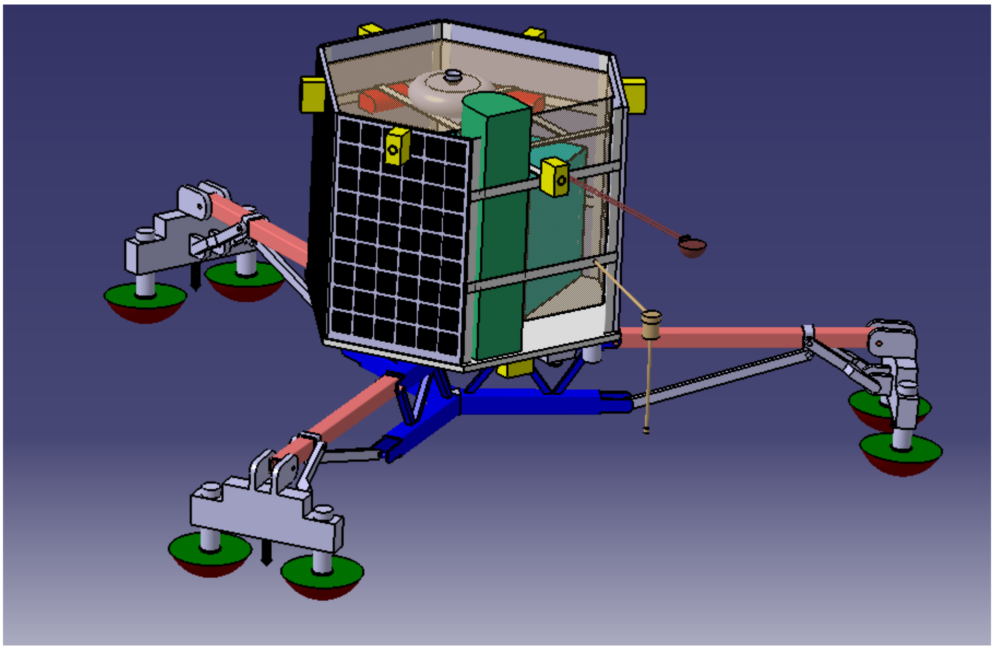


Fig. 6.1 Lander Model

6.1 Landing Strategy

The lander will be put on hibernation mode during the journey from earth to the comet. Once close enough approach is established with the comet in the orbit, the separation, descent and landing procedure for lander will start. Steps involved in SDL procedure,

- **Post hibernation check:** Once lander comes back from hibernation, all subsystems will be checked on orbiter and the status will be reported to ground station. The payloads, OBC and communication performance will be checked and compared with pre hibernation performance.
- **Mapping of comet:** Comet will be mapped using cameras and payloads available on orbiter.
  - OSIRIS(Optical, Spectroscopic, and Infrared Remote Imaging System): This imaging system will prepare a topographical map of the comet along with heat map using infrared.
  - VIRTIS(Visible and Infrared Thermal Imaging Spectrometer): The Visible and IR spectrometer is able to make pictures of the nucleus in the IR and also search for IR spectra of molecules in the coma.
  - MIRO(Microwave Instrument for the Rosetta Orbiter): The abundance and temperature of volatile substances like water, ammonia and carbon dioxide can be detected by MIRO via their microwave emissions.
  - other imaging payloads
- **Selection of landing site:** The mapping of comet will present us with the possible landing sites out of which one will be selected from the ground station. The criteria for ranking a landing site,
  - The angle between the sun direction and surface in a nominal limit for a nominal amount of time.
  - The impact velocities should be in nominal range.
  - The vertical axis should make a nominal angle with the surface.
  - The angle between velocity vector and lander z-axis should be within particular limit.
  - There is a limit on the velocity impulse ejection mechanism can provide.
- **Landing Scenario:** After the landing site selection, lander will be ejected from the orbiter and will descend ballistically, stabilised by flywheel. Soon after touchdown, two gas powered harpoons will fire towards the ground and cold air thruster will be

activated to negate the recoil from the harpoon firing. This will be done in order to prevent bouncing of lander after impact.

6.2 Payloads

- APXS(Alpha-Particle-X-Ray-Spectrometer): APX is a spectrometer used to analyse the elemental composition of material found directly on the comet's surface.
- CIVA (Comet Infrared and Visible Analyser): CIVA consists of the panorama camera system seen here, used to analyse the landing site, and microscopes to represent and provide spectroscopy of the material samples from the comet's surface acquired using the drill SD2.
- CONSERT (Comet Nucleus Sounding Experiment by Radio wave Transmission): CONSERT is a radio wave probe used for tomography of the comet core using interaction with the orbiter.
- COSAC (Cometary Sampling and Composition): COSAC is deployed to identify the elementary, isotopic and chemical composition of the frozen components on the comet's surface and down to a depth of around 30 centimetres. The instrument contains a mass spectrometer and a gas chromatograph and will analyse the organic components (for instance amino acids, if found) in particular.
- MUPUS (Multi-Purpose Sensors for Surface and Subsurface Science): MUPUS uses an array of sensors to measure the surface temperature and temperature profiles close to the surface, also the thermal conductivity of the surface material and the solidity of the comet's material. The most important scientific objectives with which MUPUS is charged are to determine the thermal and mechanical properties found in the outer layers of a comet and to identify the energy balance on the comet's surface and outer layers. The results acquired using this instrument should help achieve better understanding of cometary activity and may permit statements on the original nature of material. It also contributes to the cosmo-chemical experiments that the Philae landing craft will perform.
- PTOLEMY: PTOLEMY is a mass spectrometer with upstream gas chromatograph to analyse the isotopic composition of drilled specimens.
- ROLIS (Rosetta Lander Imaging System): ROLIS is one of two camera systems on board the Rosetta landing craft Philae. ROLIS will use a miniaturised CCD camera to

shoot high spatial resolution images of the landing site on the comet as Philae descends from the orbiter. After landing, ROLIS will focus on a distance of 30 centimetres to take images of the comet's surface beneath the landing craft. A lamp fitted with four monochromatic light emitting diodes in a spectral range of 470, 530, 600 and 4870 nanometres will permit multispectral images. The system will also shoot pictures of the point where the drilled samples are taken and the areas designated for analysis by the alpha X-Ray spectrometer (APXS). This will provide pictures of the immediate environment for comparison with measurements conducted in the in-situ analyser. The drill hole will be inspected once drilling is complete to study its morphology and to look for signs of different layers (stratification). The landing craft is able to rotate and to take stereo image pairs, hence facilitating cartography and the identification of surface structures.

- ROMAP (Rosetta Lander Magnetometer and Plasmamonitor): ROMAP determines the comet's magnetic field and its plasma environment.

- SD2 (Sample, Drill and Distribution): SD2 is a drilling mechanism to acquire samples from a depth of up to 30 centimetres.

- SESAME (Surface Electric Sounding and Acoustic Monitoring Experiment): SESAME is equipped with sensors to measure the acoustic and dielectric properties of the comet core and its structure close to the surface, along with a particle impact monitor. SESAME consists of the instruments CASSE, DIM and PP.

  The instrument CASSE (Cometary Acoustic Surface Sounding Experiment) will use acoustic methodologies to analyse the structure of the material found beneath the comet's surface. It offers two means of measurement: one passive, which, like a seismometer, listens in on the comet, and one active, which is similar to an echo sounder and probes the surface layers.

  The Dust Impact Monitor (DIM) registers the signals caused by the impact of cometary dust and ice particles on the sensor cube. The measurements can determine the number, mass and velocity of the particles.

  The Permittivity Probe (PP) instrument determines the water ice content in the cometary surface layers and its change over time. To do this PP uses quadrupole technology in which two electrodes connected with an alternating current generator induce variable current in the comet's soil.

6.3 Power and Mass Budget

Mass Budget:

Spacecraft component	**Mass**
Structure	15
Thermal	3.9
Power system	12.2 Kg
Active Descent System(ADS)	4.1 Kg
Reaction wheel	2.9 Kg
Landing Gear	12.5
Anchoring system	1.4 Kg
Central Data management system	2.9 Kg
Telecommunication system	9.8 Kg
Common electronic box	3.6 Kg
Payloads	26.7 Kg
Total	95

Spacecraft component	**Power**
Payloads	34.3 W
Communication	3W
OBC	5 W
Heater	15 W
Total	57.3

6.4 Landing Gears

The Landing Gears are designed in order to absorb the shock at the time of impact. The nominal impact velocity is expected to be very low (about 20-30 cm/s). So the emphasis is on damping to avoid bouncing back of the lander. At the moment of impact, least or no bouncing off is desired from the surface, for achieving this goal, present configuration uses,

- Ice screws - Each leg of the lander has an ice screw, which will be activated as soon as lander touches the surface.

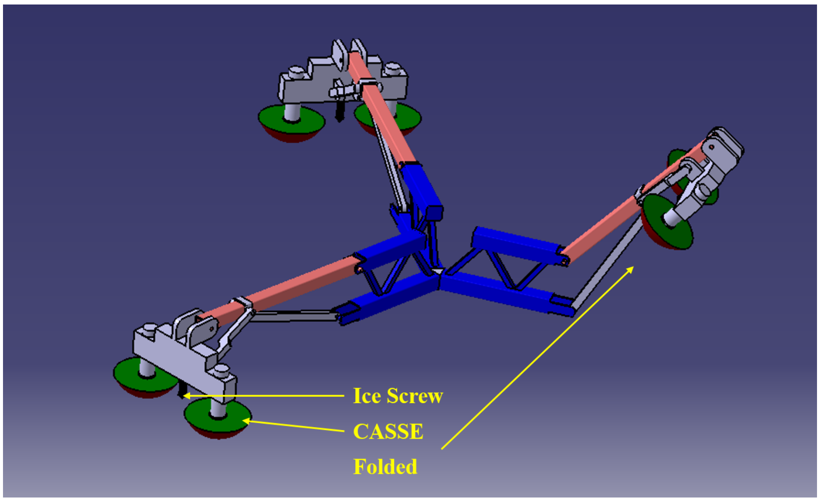


Fig. 6.2 Landing Gear

- Harpoons - Two standard gas powered harpoons will be fired for better gripping on the comet surface.
- Active Descent System - This system will be used to push the lander on the surface using cold gas thrusters, this will also be used to negate the recoil due to harpoons.

6.5 Spacecraft configuration

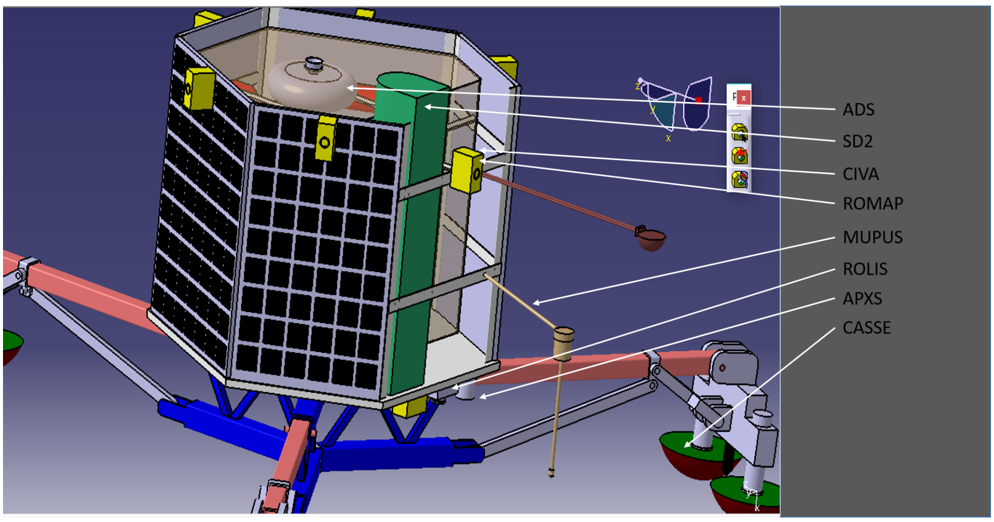


Fig. 6.3 Lander configuration

6.6 Structural analysis

6.6.1 Designing member to support ADS:

Material	Density (in kg/m3)	Thickness (in mm)	Mass (in Kg)	Maximum Deflection (in mm)
Aluminium 6061	2085	5	0.574	17.89
Titanium (Ti-6Al-4V)	4430	5	1.219	11.8
Steel (17-4PH H1150)	7860	5	2.162	6.8196
Steel (17-4PH H1150)	7860	6	2.595	3.94

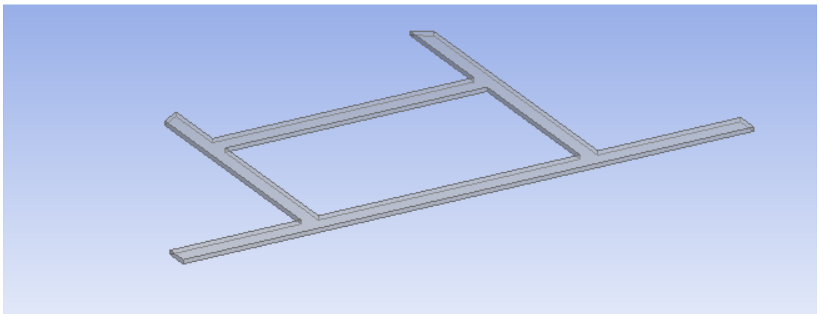


Fig. 6.4 Initial design

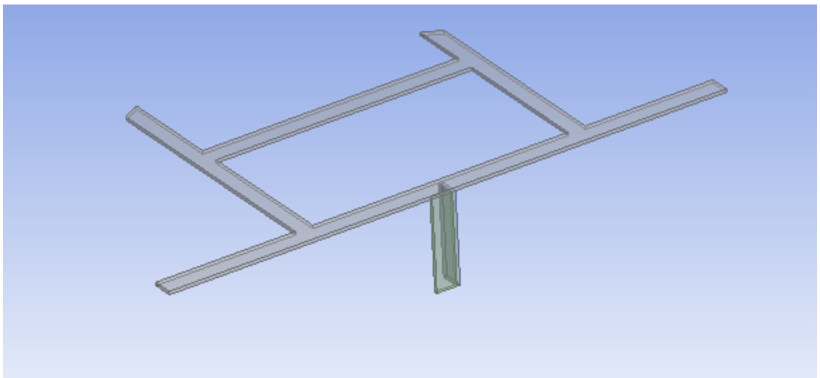


Fig. 6.5 Final design with a support underneath

None of the above, gives satisfactory results, so extra bar was added underneath this member with aluminium as material. This gave us 0.2mm maximum deflection with a mass of 0.657kg.

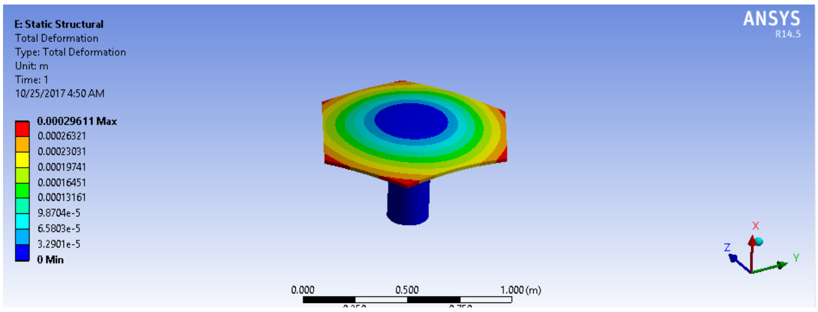


Fig. 6.6

6.6.2 Designing the base plate:

Material	Thickness (in mm)	Mass (in Kg)	Maximum Deflection (in mm)	Maximum Eq. Stress (in MPa)
Steel	20	227	0.1	7.46
Aluminium	20	60	0.3	7.46
Aluminium	18	50.8	0.4	9.26

In this case, we see that although, we are getting satisfactory structural performance, the mass of the plate is very high. So for better results we will use sandwich structure, with honeycomb core. This will give us good strength for lower mass.

6.6.3 Designing the supporting structure for solar panels

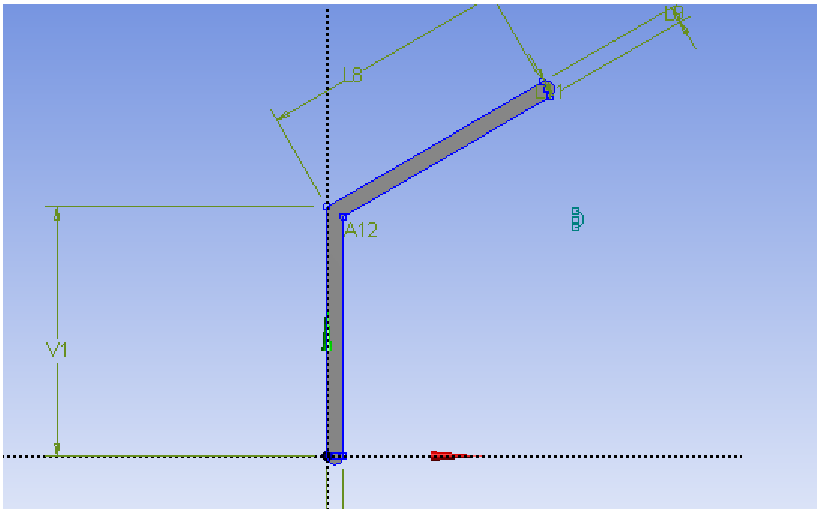


Fig. 6.7 Cross section of supporting beam($V1 = L8 = 30mm, A12 = 120°$)

The cross section was checked for failure under buckling load.

Material	Thickness	Load Multiplier	Mass
Aluminium	3mm	13.974	0.265 Kg
Aluminium	2mm	9.015	0.176 Kg

6.7 Thermal Analysis

Worst case, Hot: This will occur near earth where spacecraft is getting heat from solar radiation, earth's albedo, earth's blackbody radiation and internal heat generation. In this case we can use passive thermal controls such as coatings for external antenna and multi layer insulation for internal components.
Worst case, Cold: This will occur at comets aphelion when spacecraft is in shadow. In this case we will have to use heaters to keep the temperature in acceptable rates.
For calculation of heater power, we use

$$Q_{external} + Q_{internal} = Q_{radiator} + MLI \tag{6.1}$$

In this case, $Q_{external}$ is not present, and $Q_{internal}$ is the heat generated by heater. So,

$$Q_{heater} = \varepsilon\sigma AT^4 \tag{6.2}$$

Assuming vapour deposited aluminium coating,

$$Q_{heater} = 0.04 \times 5.67 \times 10^{-8} \times 0.96 \times 273^4 \tag{6.3}$$

$$Q_{heater} = 12W \tag{6.4}$$

Taking safe limit,

$$Q_{heater} = 15W \tag{6.5}$$

It was found from the literature, that the Philae Lander in Rosetta mission also used a 15W heater.

Chapter 7

Attitude Determination and Control System

A spacecraft has to be stabilized in orbit and often has to have a a particular orientation. This is achieved with the help of an attitude determination and control system. The system primarily consists of sensors that help in determining the orientation and position and actuators that help in re-orienting the spacecraft with sufficient forces and torques.

7.1 sensors

Any spacecraft requires a minimum of one sensor for attitude determination. Multiple sensors are often used on board as redundancy measures. The sensors used aboard our spacecraft are two pairs of coarse sun sensors, two star trackers and three inertial measurement units. Multiple nos. are used to increase reliability of measurement obtained. The readings obtained from all three types of sensors can be fed into a Kalman filter to obtain optimum attitude of the spacecraft.

7.1.1 Sun Sensors

The basic principle of the sun sensor is that it detects the change in intensity of light obtained from the sun and gives the position vector from the sensor to the sun. The position vector of the sun sensor with respect to the spacecraft body center is known. With the help of these two vectors, it is possible to determine the position vector of the space craft with respect to the Sun.

The sun sensor readings are used to orient the solar array panels towards the sun, autonomously. In case of an eclipse, the sun sensor readings can no longer be used and either algorithms deriving information from other sensors or a pre-fed model is used.

7.1.2 Star Trackers

The basic principle on which star trackers function is that they image the sky and detect the stars that are visible to the spacecraft. The in built algorithm then draws polygons joining a minimum of three stars. With the help of in built data it is subsequently able to recognize the part of the sky that is visible to the spacecraft, with the help of the polygon drawn.

One particular problem that arises with star trackers is that while in the coma of the comet, the star trackers are unable to distinguish between dust particles and stars and this significantly reduces their reliability.

7.1.3 Inertial Measurement Units

Inertial measurement units basically consist of three gyroscopes and three accelerometers, for x, y and z directions. The units require initialization in which attitude with respect to a central body is fed into the algorithm. This can happen at the moment of launch. The gyroscopes then start providing the angular acceleration readings while the accelerometers provide the accelerations in the three axes. These readings are then used to continuously update the real-time attitude with respect to the initial attitude.

7.1.4 NAVCAM

These cameras are primarily placed for imaging of the comet for landing site selection. Apart from that, the images taken can provide valuable insight into the nature of the comet.

7.2 Actuators

In addition to the 10 N thrusters present for attitude control, momentum wheels are used for accurate attitude control and to absorb environmental torques. A 1-DoF actuator allows the solar arrays to point towards the Sun at all times, with inputs from the sun sensor. A 2-DoF actuator system is further is used to point the antenna towards Earth.

Chapter 8

Power Systems

8.1 Spacecraft Power System

Various systems aboard the spacecraft have power requirements to keep them running. These include the payloads aboard the orbiter as well as the lander and other ADCS and communication equipment. Power is also required to keep the propellant tanks heated to prevent the propellant from freezing. The orbiter shall consist mainly of deployable solar arrays and lithium-ion batteries as back up source.

The deployable solar arrays are roughly of dimensions 2.2 m by 2.7 m and are 10 in nos. They are stowed on to the spacecraft until launch into orbit and are then deployed, where they fold out, with 5 arrays on each side. The total area of the arrays comes to around 60 m^2. They mounted on either side with a 1 dof actuator that enables rotation of the panels to face the sun at all times.

The photovoltaic layout of the arrays consists of silicon Hi-ETA solar cells that support Indium Tin Oxide coated conductive coverglass that is grounded to the structure. The maximum power capability of the solar array is around 7 kW at a distance of 1 AU and over 50 degrees Celsius. Most of the power is never utilized since the maximum power requirement of the components even in the worst case scenario just comes to about 900 W.

At the aphelion, the power produced by the solar arrays drastically decreases. This power is not sufficient to keep all the components of the spacecraft running. Hence most of the components are switched off and go into hibernation mode until more power is available. Only some crucial systems like the thermal system required to keep the propellants from freezing are kept on. Even for the minimal power produced, the solar arrays have to constantly face the sun. This is achieved by spinning the spacecraft around an axis pointing towards the sun and perpendicular to the solar panels. Inspite of the active attitude control being suspended during this period, the solar panels reliably face the Sun owing to conservation of angular momentum.

In addition to the solar arrays, the power system shall consist of lithium ion batteries to be mainly used during the post-launch period before the deployment of the solar arrays. In addition, these batteries also serve as back-up supply in case of solar power failure or if the power requirement of the components ever exceeds that produced by the solar arrays. As in the Rosetta mission, the battery contains three separate modules of 11 strings of 6 series-connected lithium-ion cells, with a total capacity of 1050 Wh.The mass of the battery comes to around 10 kg.

The orbiter also contains power control and distribution units to ensure constant power supply to the components. The input voltage to the control unit is highly variable depending on the distance from the sun. But the control unit regulates the power supply to provide a constant voltage output.

8.2 Solar Panels Sizing

A solar panel works by allowing photons, or particles of light, to knock electrons free from atoms, generating a flow of electricity. Solar panels actually comprise many, smaller units called photo-voltaic cells. Photo-voltaic simply means they convert sunlight into electricity. Many cells linked together make up a solar panel.

Each photo-voltaic cell is basically a sandwich made up of two slices of semi-conducting material, usually silicon.

Solar panels power supply depends on the solar radiation on the panel that is directly proportional to the area exposed to the sun light. It is defined by the Solar constant. The solar constant, a measure of flux density, is the mean solar electromagnetic radiation (the solar irradiance) per unit area that would be incident on a plane perpendicular to the rays, at a distance of one astronomical unit (AU) from the Sun (roughly the mean distance from the Sun to the Earth). The solar constant includes all types of solar radiation, not just the visible light. It is 1.361 kilowatts per square meter. Solar constant is inversely proportional to the square of the distance from the sun. At 5.2 AU from the sun the solar constant

$$SC = 1.361 * 1^2/5.2^2 * 1000 = 50.33W/m^2.$$

For 900 watts (worst case scenario).

$$Area = 900/50.33 = 17.882m^2$$

If we assume that the solar panels had the 30 percent efficiency, then area of solar panels

$= 7.947/0.30 = 59.606m^2 = 60m^2$

The solar panels cannot be taken as a single sheet in to the launch vehicle. Launch vehicle has limited space so we have to fold the solar panels.

8.3 Lander Power System

The lander power system is again solar power based. In addition, it consists of primary and secondary batteries to power the payloads aboard the lander. The primary batteries are not rechargable and are used only in the wake up sequence soon after landing on the comet, to ensure that all instruments are operated at least once. The lander solar power comes from a photovoltaic shroud that covers the lander.

Chapter 9

On-Board Computer and Communications

9.1 On-Board Computer

The brain of the satellite, the on-board computer is a multitasking component. It is desirable for it to have less mass and require less power. It is of utmost importance that this subsystem is robust and reliable with high durability, as all other subsystems depend on it for proper functioning. Some of its main tasks include:

1. Health monitoring: Collection of housekeeping data like the operational status of the subsystems, remaining power etc.
2. Maintaining flow of data between different subsystems and peripherals like power, payload, GPS etc
3. Maintaining flow of data from satellite to earth and vice versa.
4. Data collection from ADCS sensors, analysis and execution of steps for attitude correction.
5. In situ scientific data processing for landing site selection.
6. Decision taking for approach strategies, landing strategies etc.

The on-board computer system selected for these purposes is OBC695. It is a radiation tolerant flight computer with applications in LEO, MEO, GEO and interplanetary missions. It is designed for failure-critical applications or harsh environments. The unit can be used as a single board computer, or in dual redundant configuration. It is manufactured by Surrey Satellite Technology US LLC. Following are its specifications:

Fig. 9.1 OBC695

- Processor: ATMEL TSC695 Sparc V7
- Performance: 11 MIPS
- Memory: 4 or 8 MB (EDAC protected), 512 KB EEPROM, 64 KB PROM
- Mass: 1.5 kg
- Dimensions: 306 x 167 x 30 mm (single board)
- Power: 7 W
- Operating Temperature: -20ºC to +50ºC
- Radiation Total Dose: 50 kRad (Si) SEL 37MeV-cm²/mg
- 7.5 years design life

It has been previously used in the following missions:

1. GIOVE-A (2006)
2. Chandrayaan-1 (NASA variant) (2008)
3. Kanopus-1,2,3 (2 units each, 2010)

9.2 Communications

A communications suite in a spacecraft is necessary for transmission and reception of telemetry signals and mission data. Communication between the lander and ground station on Earth will take place through the orbiter, hence it is sufficient to install a low power antenna in lander. But the antennas in the orbiter need to be capable of deep space communication. Maximum separation from comet to Earth during mission duration is about 6 A.U. So maximum communication delay will be around 50 minutes.
Usually the uplink frequency is higher than the downlink frequency because the uplink signals have to cross the atmosphere, which presents large attenuation. Due to signal losses, the power has to be high, which the ground antennas can afford to provide. Satellites are low power sources and hence downlink frequency is usually lesser. Estimated rate of data transmission = 10000 - 22000 bits/s

- Uplink:
  - Frequency = 8 GHz
  - Bandwidth = X band
- Downlink:
  - Frequency = 2 GHz
  - Bandwidth = S band

Antenna design:

1. 2.2 m high-gain dish antenna
2. 0.8 m medium-gain antenna
3. Two omnidirectional low-gain antennas.

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Chapter 10

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