SPACE: THE FINAL FRONTIER

HANDBOOK

SNT COUNCIL SUMMER PROJECT'21

an introduction to rocket science

ASTRONOMY CLUB IITK

COSMOS

IS WITHIN US

Acknowledgement

We would like to express our thanks towards the mentors and will forever be thankful for your guidance and support. It was a great experience being a part of this project. We were really fortunate that we had this kind of association as well as supervision from you. Your exemplary guidance, constant encouragement and careful monitoring throughout the project were so great, that, even our most profound gratitude is not enough. Looking forward to more collaborations like this in the future.

Sunny K. Bhagat | sunny@iitk.ac.in

Mubashshir Uddin | muddin@iitk.ac.in

Varun Singh | varunsng@iitk.ac.in

Mentees

Ananya Upadhaya
Adit Jain
Aditya Kushwaha
Ananya Priyadarshi
Anubha Thakur
Anup Singh
Atharva Dehadraya
Bhawani Shankar
Ganesh Kumar
Kajal Deep
Nishithaarsh
Nallani Nithin
Ojsi Goel
Parth Bhatt
Prem Gujrathi
Ratan Lal Sahu
Renaissance Hansda
Saikrishna
Sameer Singh
Sarvagya Jain
Shashank Sinha
Shikha Singh
Shubhankari Rai
Yashika Malhotra

Contents

1 History of Rockets

1.1 Introduction

Todays rockets are remarkable collections of human ingenuity that have their roots in the science and technology of the past . It has been a fascinating subject for the scientific community. They are natural outgrowths of terribly thousands of years of experimentation and research. The data reporting the first use of of true rockets was in in 1232. At that time, the Chinese and Mongols were at the war. The fire arrows used were a simple form of solid propellant rocket.

1.2 Tsilokovsky's Contribution

In 1903, high school mathematics teacher Konstantin E. Tsiolkovsky (1857 - 1935), inspired by Verne and Cosmism, published *The Exploration of Cosmic Space by Means of Reaction Devices*. The Tsiolkovsky Equationthe principle that governs rocket propulsionis named in his honour . He also advocated the use of liquid hydrogen and oxygen for propellant, calculating their maximum exhaust velocity.

$$\Delta v = v_e ln \frac{m_0}{m_f} \tag{1.1}$$

Here, Δv is the change in speed of rocket, v_e is exhaust velocity, m_0 is the initial while m_f is the final rocket mass. He also published the a theory of multistage of rockets in 1929.

1.3 Goddards contribution

In 1912, Robert Goddard, concluded that conventional solid-fuel rockets needed to be improved in three ways.

- The fuel should be burned in a small combustion chamber, instead of building the entire propellant container to withstand the high pressures.

- The rockets could be arranged in stages which will make them lose unnecessary load in middle of flight.
- The exhaust speed could be greatly increased by using a DE naval nozzle.

He combined and put forward these concepts in 1914. Goddard worked on developing solid propellant rockets since 1914, and demonstrated a light battlefield rocket to the US Army Signal Corps only five days before the signing of the armistice that ended World War 1. He also developed gyroscope system for flight control of rockets. After his death NASA has named its first space flight complex as Goddard Space Flight Center in his honour.

1.4 Modernisation in Rocketry

Following developments by Goddard,rocketry was revolutionised but scientists had to still figure out to send the rockets to space. Rapid developments began when Germany realised the potential of the rockets as weapons in World War 2 through its V-2 rockets. A rocket was first used to send something in space on the Sputnik mission, which launched a Soviet satellite on Oct 4, 1957. Rocketry reached its pinnacle when it carried humans to the moon on July 16, 1969.

2 Design

2.1 Payload

Payload is the carrying capacity of an aircraft or launch vehicle, usually measured in terms of weight. Depending on the nature of the flight or mission, the payload of a vehicle may include cargo, passengers, flight crew, munitions, scientific instruments or experiments, or other equipment. Extra fuel, when optionally carried, is also considered part of the payload.

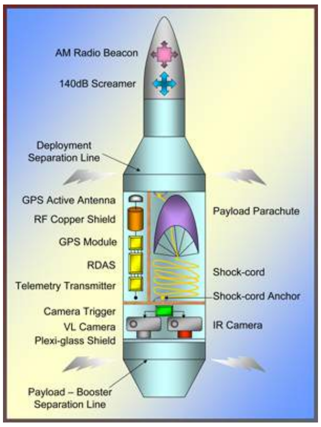


Figure 1: Payload

The payload system of a rocket depends on the rocket's mission. The earliest payloads on rockets were fireworks for celebrating holidays according to Chinese history (including the story of man strapped to a chair containing explosives XD). The payload of the German V2, shown in the figure, was several thousand pounds of explosives. Following World War II, many countries developed guided ballistic missiles armed with nuclear warheads for payloads. The same rockets were modified to launch satellites with a wide range of missions; communications, weather monitoring, spying, planetary exploration, and observatories, like the Hubble Space Telescope. Special rockets were developed to launch people into earth orbit and onto the surface of the Moon and advanced scientific rovers and crafts to the outer solar system and beyond..

2.2 Fairing

The spacecraft and upper stage are housed within the payload fairing, which usually has a diameter more than that of the second or third stage. During the early portion of the boost phase when the aerodynamic forces from the atmosphere could affect the rocket, faring plays a crucial role:

- Protection in lower Atmosphere
- Protection against biospheric contamination
- Protection against heat

Figure 2: Faring

Although faring may add extra weight to the rocket (which is negligible in comparison to the payload capacity), it's benefits outweigh the cons. Also as stated by the European Space Agency, “Once the launcher leaves the Earths atmosphere, approximately three minutes after liftoff, the fairing is jettisoned. This lightens the remaining launchers load as it loses approximately two tonnes of this no-longer required structure“

2.3 Body

The structural system, or body, of a rocket is similar to the fuselage of an airplane. As you can see on the figure, most of a full scale rocket is propulsion system surrounded by the frame:

2.3.1 Frame

The frame is made from very strong but light weight materials, like titanium or aluminum, and usually employs long "stringers" which run from the top to the bottom which are connected to "hoops" which run around around the circumference. The "skin" is then attached to the stringers and hoops to form the basic shape of the rocket. The skin may be coated with a thermal protection system to keep out the heat of air friction during flight and to keep in the cold temperatures needed for certain fuels and oxidizers:

Figure 3: Frame

2.3.2 Propellant & Fuel Tank(s)

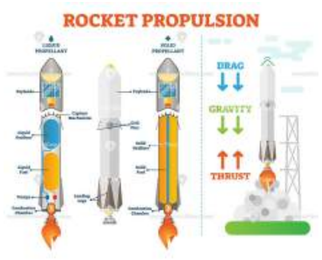


Figure 4: Rocket Propulsion

Rocket propellant is the reaction mass of a rocket. This reaction mass is ejected at the highest achievable velocity from a rocket engine to produce thrust. The energy required can either come from the propellants themselves, as with a chemical rocket, or from an external source, as with ion engines. This reaction mass is stored in the Fuel Tanks of the rocket, that account for more than 70% volume of a rocket (No kidding!).

Liquid Propellant is the most widely used fuel in a rocket. Liquid oxygen is the most common cryogenic liquid oxidizer propellant for spacecraft rocket applications, usually in combination with liquid hydrogen, kerosene or methane. Liquid-fueled rockets have higher specific impulse than solid rockets and are capable of being throttled, shut down, and restarted. Only the combustion chamber of a liquid-fueled rocket needs to withstand high combustion pressures and temperatures. This propellant is pushed into the rocket engines by means of tubes.

For the side-thrusters in heavy rockets that can carry immense payload to space, ***solid fuel*** is preferred as solid propellant rockets are much easier to store and handle than liquid propellant rockets. High propellant density makes for compact size as well. These features plus simplicity and low cost make solid propellant rockets ideal for military and space applications.

Ion thrusters ionize a neutral gas and create thrust by accelerating the ions (or the plasma) by electric and/or magnetic fields. Mainly, heavy elements such as Xe (Xenon) are used. Also, Ion thrusters can be operated and stored for a longer duration than either the solid or liquid propellants.

But of course, we need Engines to utilise the complete potential of these fuels:

2.3.3 Rocket Engine

This is the main component of the rocket. Yup, the one that is responsible for actually "lifting-off" the rocket from the ground to reach into space. There are two main classes of propulsion systems, liquid rocket engines and solid rocket engines. The V2 used a liquid rocket engine consisting of fuel and oxidizer (propellant) tanks, pumps, a combustion chamber with nozzle, and the associated plumbing.

Figure 5: Rocket engine

The Space Shuttle, Delta II, and Titan III all use solid rocket strap-ons. The engines may or may not use oxidiser (i.e. they may take oxidiser intake via air from the atmosphere).

2.4 Fins

The technical definition of a Fin is: *A surface used to give directional stability to any object moving through a fluid such as water or air.* In short, fins provide maneuverability and stability to the rocket in the upper atmosphere. The size of the fins, their shape, the number to use and their placement on rocket are all questions that can be answered only by experimentation. Fins and other aerodynamic properties are tested in wind tunnels at high wind speeds. More about the aerodynamic impact of fins later in the document.

3 Aerodynamics

3.1 Introduction

Rockets tend to break the sound barrier very fast and often when they haven't even left the lower atmosphere. The fins should be optimized for supersonic flight and ideally stay within the Mach Cone. There is always an optimum point and design style. In general, larger the fin, the more drag but the more stable. Thus drag and flow dynamics become extremely crucial in making of rockets. Following we discuss what are shock waves and how do they affect Rockets.

3.2 Definition

A shock wave is a propagating disturbance that moves faster than the local speed of sound in the medium.It carries energy and can propagate through a medium but is characterized by an abrupt, nearly discontinuous, change in pressure, temperature, and density of the medium.

3.3 Cause

Shock waves are formed when a pressure front moves at supersonic speeds(Speed more than that of sound) and pushes the air surrounding it. At that particular region, sound waves travelling against the flow reach a point where they are unable to travel any further upstream, and the pressure progressively develops at that particular region and a high-pressure shock wave rapidly forms.

3.4 Properties

Shock waves are not conventional sound waves; a shock wave takes the form of a very sharp change in the gas properties and is heard as a very loud noise. Over longer distances, it can change from a nonlinear wave into a linear wave, deteriorating into a conventional sound wave as the air is heated and the wave's energy is lost. A sound similar to that of created by a sonic boom is heard, which is usually produced by the supersonic flight of an aircraft.

3.5 Types of Shock Waves

- OBLIQUE SHOCK
- NORMAL SHOCK
- CROSSED SHOCK WAVES

3.5.1 Oblique Shock

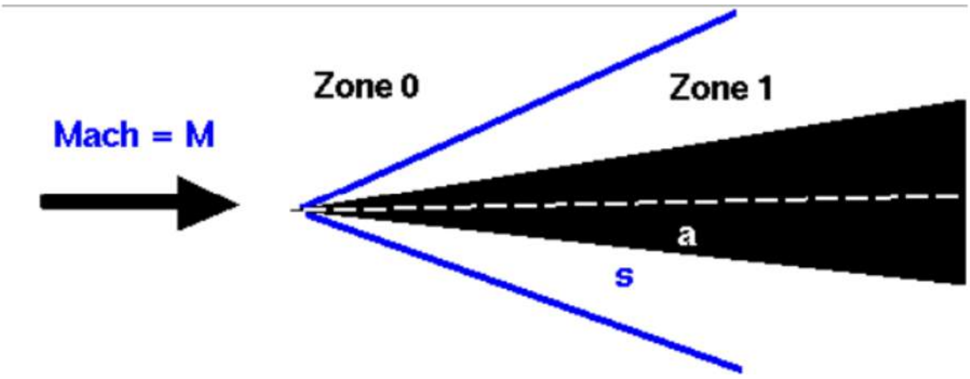


Figure 6: Oblique Shock

If the shock wave is inclined to the flow direction it is called an Oblique Shock.The flow changes and oblique shocks occur downstream of a nozzle if the expanded pressure is different from free stream conditions. For the Mach number change across an oblique shock there are two possible solutions; one supersonic and one subsonic.For normal conditions,supersonic solution is considered.

3.5.2 Normal Shock Waves

If the shock wave is perpendicular to the flow direction it is called a Normal Shock .It occurs in front of a supersonic object if the flow is turned by a large amount and the shock cannot remain attached to the body. The normal shock significantly increases the drag in a vehicle traveling at a supersonic speed. This property was utilized in the design of the return capsules during space missions such as the Apollo program, which needed a high amount of drag in order to slow down during atmospheric reentry.

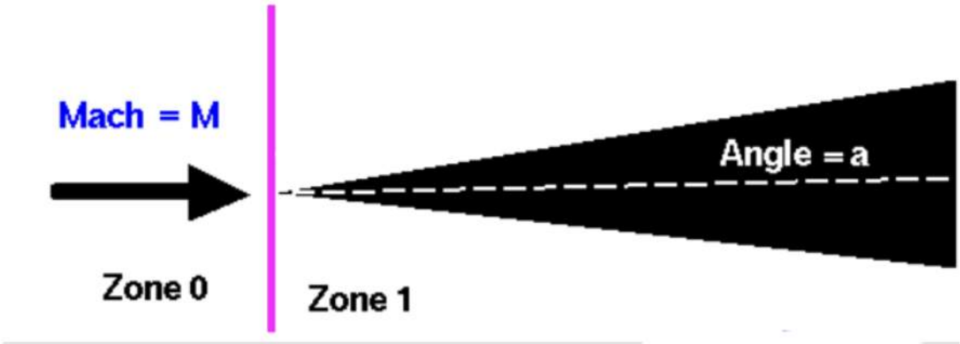


Figure 7: Normal Shock

3.5.3 Crossed Shock Waves

When two or more surfaces are there the shock waves generated due to it cross each other and the resultant is a third shock wave,called a Crossed Shock Wave. The shocks generated by the two wedges intersect at some point in the flow.In all of the shock reflections and intersections the Mach number of the flow is decreased

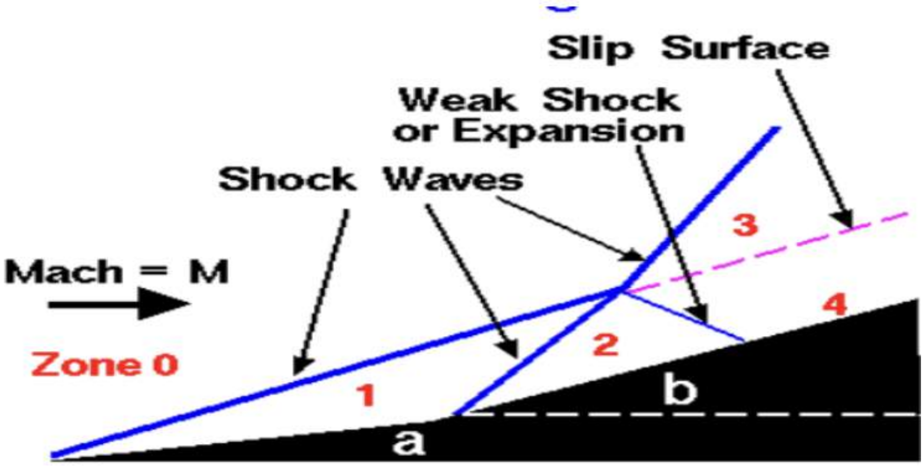


Figure 8: Crossed Shock

Stability

4

4.1 Introduction

Rockets stability is the key to control the whole path of the mission. Every mission has some specific goals and desire the rocket to fly in a defined and safe trajectory or path, therefore maintaining the stability of the rocket is a must to steer the rocket in its path.

Stability of rocket is broadly decided by the location of points of Center of gravity (C_g) and center of pressure (C_P). Centre of gravity (C_g) of a rocket is the point where the whole weight of the rocket can be averagely considered. And center of pressure (C_P) is the point where all the air pressure forces appear to concentrate.

4.2 Stability Criterion

Like any object, rockets can rotate about their center of gravity (C_g). During the flight, even a small gust of wind or thrust can cause instabilities and can cause the rocket to wobble, and can change the complete course of flight. Whenever the rocket is inclined to its actual path, lift forces combining with drag act through the center of pressure (C_P). These forces exert torques about the center of gravity (C_g) and in the direction so that it counteracts the disturbance caused by wind or other air pressures.

The most important point of concentration is that the rockets should be engineered in a way such that the constantly rising position of C_P should always be below C_g to maintain the counteracting nature of torques caused by lift and drag. Stability increases as the distance between C_P and C_g increases. The lift and the drag forces move the nose back towards the flight direction. Engineers call this as a restoring action/force because the forces restore the vehicle to its initial condition and nullify the action of disturbing forces to maintain the stability.

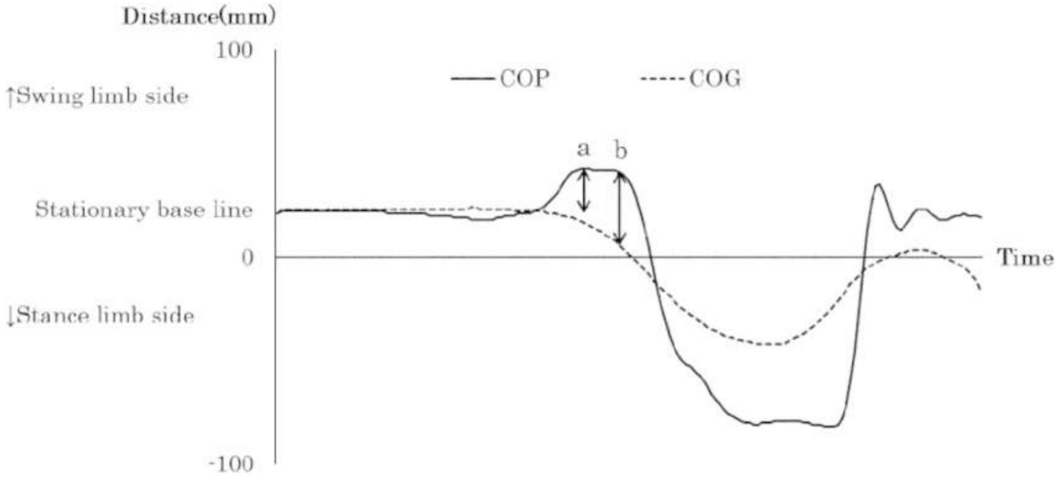


Figure 9: Centre of pressure and Centre of gravity versus time plot

As we can see int the figure 1, there certain strands of time where the C_P is above C_g which leads to instability in the flight which is counteracted and handled by engines.

A restoring force exists for the rocket because the center of pressure (C_P) is below the center of gravity(C_g). If C_P is above C_g at any time of flight, the lift and drag forces maintain their directions but the direction of the torque generated by the forces are reversed. This is called a de-stabilizing action/force. Any small disturbance in such condition can cause the rocket to completely loose its trajectory. One of the most trusted way to ensure stability is to add fins to the design so as to ensure that C_P remains below C_g throughout.

Re-Entry

5

As much as going to space fascinates people, re-entry of spacecraft is one of the most important part of a space mission. In early days, rockets were not that powerful and only could obtain suborbital hops. But our modern rockets are capable of obtaining higher earth orbits.

5.1 Modes of re-entry

- **Ballistic re-entry:** It is like normal projectile motion. The only forces involved in it are gravity and drag. While re-entering vehicle heats up rapidly and experiences extreme g-force. In ballistic re-entry the vehicle spends very less time in the atmosphere, so it needs high deceleration. This high deceleration and extreme g-force make this kind of re-entry very unsuitable for human flights.

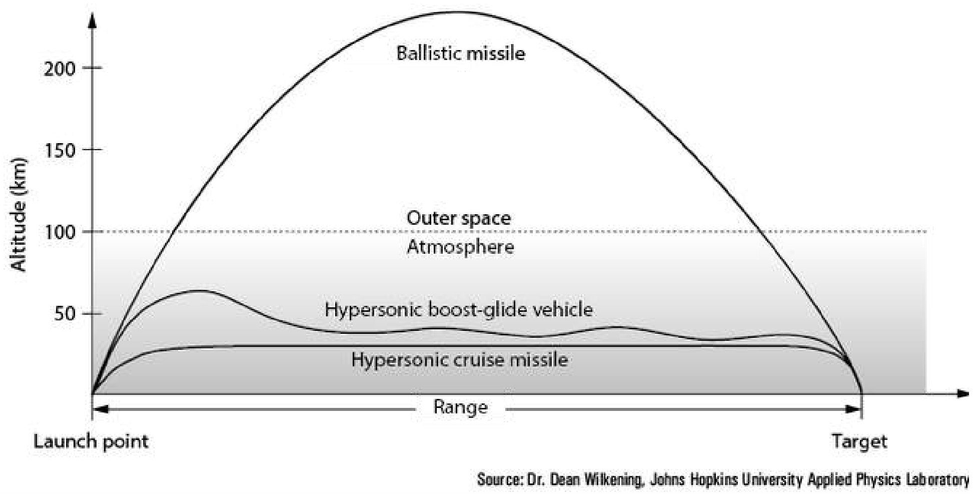


Figure 10: Ballistic Trajectory

- **Aerodynamic re-entry:** In addition to the gravity and drag force the spacecraft also introduces other forces while re-entering, to increase flight duration in atmosphere. During re-entry the spacecraft bleeds off most of its velocity in upper atmosphere and then plunges into the lower atmosphere with a lower velocity. Velocity, generated heat and experienced g-force during re-entry is lesser than that of ballistic re-entry which makes it more befitting for a crewed mission.

5.2 Factors during re-entry

- **Deceleration:** The structure and payload of spacecraft control its maximum deceleration. If it had a higher deceleration it will slow down rapidly and heat-up quickly and experience too much drag. But if a craft has a very low deceleration and it doesnt slow down enough then it may bounce off to the space.[**1**]

- **Heating:** During re-entry pods move with hyper-sonic speed which doesnt give time to the air in front of the pod to move aside and create friction. This air compression generates heat. The temperature can go up to 5000-6000. In fact, the compression generates shock waves, which causes heat to form plasma. This shock region is produced at a distance from the craft which is proportional to the curvature of the heating surface. So, temperature of the pod is not that high (1000-1200). Some air which escapes from the side, cause friction to the upper part of the pod.

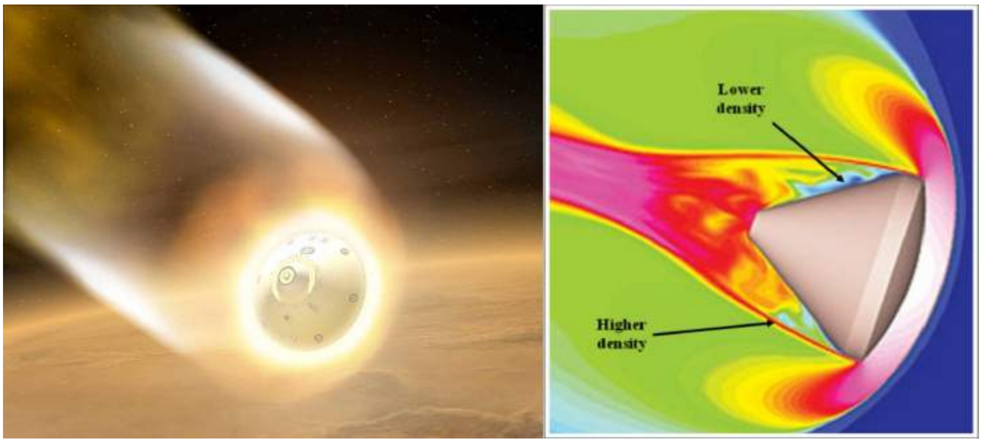


Figure 11: Heat during Re-Entry

5.3 Heat-shields: safety measurements

Heat-shields are designed to protect the spacecraft from excessive heat generated during re-entry. Heat-shields uses mainly two mechanisms: *thermal insulation* and *radiative cooling*. Some of the heat-shields used in modern rockets:

- **Insulation blankets:** The primary purpose of these are to protect the crew from engine noise and high temperature. These are lightweight, flexible, easier to maintain, replaceable and they firmly hold onto the body. They are particularly used for the fuselage wall cavities of the spacecraft, where the temperature is comparably below 649řC.

- **Insulation tiles:** These reusable insulation tiles are made from pure quartz sand. It prevents heat transfer to the underlying orbiter aluminium skin and structure. These tiles are very poor heat conductors and used where temperature is below 1260. These tiles are not mechanically fastened to the vehicle, but glued. So, they have a tendency to fall off which can be sometimes catastrophic.

Figure 12: Insulation blankets and insulation tiles

- **Reinforced carbon-carbon:** These are made from carbon fibres reinforced in matrix of graphite. It can withstand re-entry temperature up to 1510 and comparably heavy and stronger than the tiles. Usually used on wing leading edges and nose cap, where the craft experience very high temperature.

- **Ablative heat shields:** An ablative heat shield consists of pyrogenic material when heated up it produces certain gases. When the outer surface is heated to a gas, it burns off, which then carries the excess heat away. Some models used in space -missions:

  1. SLWA
  2. AVOCOAT
  3. PICA

Figure 13: RCC used on nose cone of a vehicle and PICA

- **Regenerative cooling:** This method is thought for cooling the combustion chamber. As, the propellants used in rockets are often cryogenic it can be passed through tubes, channels around the combustion chamber to cool the engines. But still its an experimental technology.

6 Rocket Equation

6.1 Ideal case

The two major considerations for ideal rocket equation taken are:-

- Conservation of mass
- Conservation of momentum

thus we get,

$$M.(\frac{dv}{dt}) = \frac{dM}{dt}.(-v_{ex}) \tag{6.1}$$

where M is the ***mass of the rocket*** *and v_{ex} is the velocity of the dm mass of the propellant i.e.* ***exhaust velocity****.*

On further solving equation (i),we get,

$$\Delta V = V_{ex}.ln(\frac{M_o}{M_f}) \tag{6.2}$$

[v_{ex} is somewhat constant at steady state arrival]

here, M_o**–initial mass of rocket**, M_f**-final mass of rocket** and v is the destination velocity i.e, orbital velocity where the rocket is supposed to land.

We can also get the **mass of the propellant**, ΔM required for the entire journey by the formula below:-

$$\frac{\Delta M}{M} = 1 - e^{(\frac{-\Delta v}{v_{exhaust}})} \tag{6.3}$$

6.2 Specific Impulse

In the language of ROCKET SCIENCE, specific impulse, I_{sp} can be defined as the efficient use of propellants in rockets/spacecrafts and fuels in jet engines. It is thus the measure of how proficiently a reaction mass creates thrust.

6.2.1 Formulation

$$I_{sp} = lim_{\Delta t \to 0} \frac{F_{thrust}.\Delta t}{M_p.g_o} \tag{6.4}$$

M_p is the mass of propellant and F_{thrust} is the thrust.

The above formula could be used to get the destination velocity V as well which is given by:-

$$\Delta V = I_{sp}.g_o.ln(\frac{M_o}{M_f}) \tag{6.5}$$

6.3 Non- Ideal Scenarios

6.3.1 Inclusion of the effect of gravity

Once the factor of gravity is incorporated, the rocket equations undergo a drastic change. Everything once again starts from the baseline i.e., Newtons Laws of Motions with the consideration of acceleration due to gravity, *g* along with the thrust produced by the propellants.

Thus by using $a = \frac{dv}{dt}$ and thrust = $-c.(\frac{dm}{dt})$, the velocity of the rocket can be easily calculated as:

$$v = c(ln\frac{1}{\mu} - \frac{1-\mu}{n}) \tag{6.6}$$

where constant μ is equal to **propellant mass fraction** and n is equal to $(\frac{F_{thrust}}{m_o g})$.The above expression could be integrated to get the displacement z of the rocket.

At the burn-out time (when all propellant is consumed), rocket will coast in free flight and would follow some trajectory. Hence, we can obtain the **energy per unit**

mass, E as a function of μ by satisfying the Keplers laws.

$$E = \frac{v^2}{2} + gz \tag{6.7}$$

Since 0<μ<1, 1-$\mu - ln(\frac{1}{\mu})$ is negative. Hence reducing n implies applying smaller thrust for a longer time, which gradually, deteriorates the final energy of the payload. To overcome this problem, we define an **Initial Ideal Impulsive Velocity (**V_o**)** and an **Equivalent Real Impulsive Velocity (**V_{eq}**)**, *where impulsive means at the ground level without much gravitational potential.*

$$(v_o)_{eq} = c\sqrt{(ln\frac{1}{\mu})^2 + \frac{1-\mu-ln(1/\mu)}{n}} \tag{6.8}$$

6.3.2 Gravity along with Drag

The effect of the drag force, D, is harder to quantify. It turns out that for many important applications drag effects are very small. The drag force is characterized in terms of a drag coefficient, C_D. Thus,

$$D = \frac{1}{2}\rho v^2 A C_D \tag{6.9}$$

where A is the cross-sectional area of the rocket. The air density changes with altitude z, and may be approximated by,

$$\rho = \rho_o e^{-z/H} \tag{6.10}$$

where $H \approx 8000$ m is the so-called scale height of the atmosphere, and ρ_o is the air density at sea level.
It is interesting to note that the effect of drag, losses contrary to what one might believe, is usually quite small, and it is often reasonable to ignore it in a first calculation. In order to see the importance of D versus the effect of gravity, we can estimate the value of the ratio $(\frac{D}{mg})$.

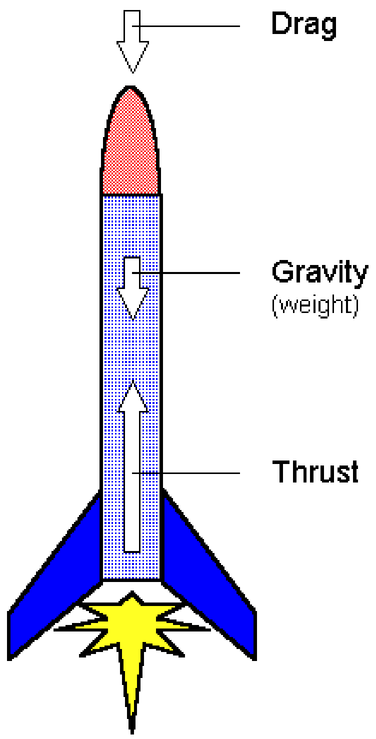


Figure 14: Forces acting on a rocket

Let's take an example: At conditions typical for maximum drag, $\rho \approx 0.25\ kg/m^3$ and $v = 700m/s$. Considering a rocket of 12,000 kg with a cross section of $A = 1m^2$ and $C_D = 0.2$, we have,

$$\frac{\rho A C_D v^2}{2mg} = 0.021 \tag{6.11}$$

which indicates that the drag force is **only 2%** of the gravity force.

SSTOs and DSTOs

7

7.1 SSTOs

Single-Stage-toOrbit commonly known as SSTO vehicle, was the first proposed launch vehicle to peep into outer space. It reaches Orbit From the surface of a body using only propellants and fluids without expending tanks, engines, or other major hardware. Notable single stage to orbit concepts include Skylon, the DC-X, the Lockheed-Martin X-33, and the Roton SSTO.

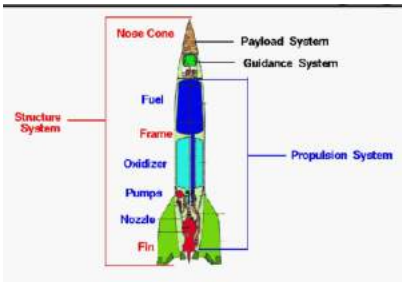


Figure 15: Structure of a SSTO vehicle

These vehicles are much easier to achieve on extraterrestrial bodies with weaker gravitational fields and lower atmospheric pressure than Earth, such as the Moon and Mars. They have been acquired from the Moon, the Apollo program's Lunar Module, by several robotic spacecraft of the Soviet Luna program.

7.2 Why Staging?

The weight of the propellants is the major contributor to the rocket's weight. As the propellants are burned off during powered ascent, a larger proportion of the vehicle's weight becomes the near-empty tankage and structure required when the vehicle was fully loaded. Thus, the concept of staging has been used to lighten the vehicle's weight to achieve orbital velocity and discard a portion of the vehicle in the process after the burnout.

7.3 DSTOs

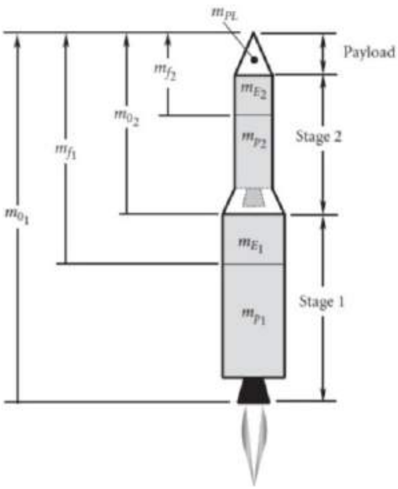


Figure 16: Structure of a DSTO vehicle

A double-stage-to-orbit launch vehicle is a spacecraft premised on the concept of a reusable launch system using two rocket stages, each containing its own engines and propellant - provide propulsion consecutively to achieve orbital velocity. The two stages are designed so that the first stage is reusable while the second is expendable. The first stage in the rocket helps in accelerating the vehicle, at liftoff.

At burnout of the fuel in first stage, the second stage steadily separates from it and continues to orbit using the fuel of its own. Nowadays, developments are taking place to make them somewhat reusable by retrieving the first stage components. One such example is Falcon 9, which achieved the first-stage reuse of an orbital vehicle.

7.4 Some Important Relations

m_e- structural part of the rocket stage(empty mass)

m_p - mass of the propellant

m_{pl} - mass of the payload

m_i - mass of the initial mass

m_f - final mass of rocket after burnout

$m_i = m_e + m_p + m_{pl}$

Certain dimensionless ratios have been defined for a meaningful comparison between the performance of the rocket and its stages.

1. Mass Ratio : the ratio between full initial mass of the stage of the rocket and final mass of the rocket stage, once at burnout - all of its fuel has been consumed. The equation for this ratio is:

$$\eta = \frac{m_e + m_p + m_{pl}}{m_e + m_{pl}} \tag{7.1}$$

2. Structural Coefficient : the ratio between the mass of structural part of the rocket stage, and the combined mass of the structural part of the rocket stage and propellant mass used in the stage, as shown in this equation:

$$\epsilon = \frac{m_e}{m_e + m_p} \tag{7.2}$$

3. Payload Ratio : the ratio between the payload mass and the combined mass of structural part of the rocket stage and the propellant, as shown in this equation :

$$\lambda = \frac{m_{pl}}{m_e + m_p} \tag{7.3}$$

4. Propellant Mass Fraction : the ratio between the propellant mass and full initial mass of the rocket, as shown in this equation :

$$\zeta = \frac{m_p}{m_i} = \frac{m_i - m_f}{m_i} = 1 - \frac{m_f}{m_i} \tag{7.4}$$

It can be easily observed that they are dependent on each other and, on further calculations following relation is obtained :

$$\eta = \frac{1 + \lambda}{\epsilon + \lambda} \tag{7.5}$$

These performance ratios can also be used as references for the efficiency of the rocket during performing optimizations and comparing configurations for analysis.

*In Restricted Staging, all stages are similar with each stage having the same specific impulse I_{sp},same structural coefficient ϵ , same payload ratio λ and hence same mass ratio η.

7.5 SSTO vs DSTO

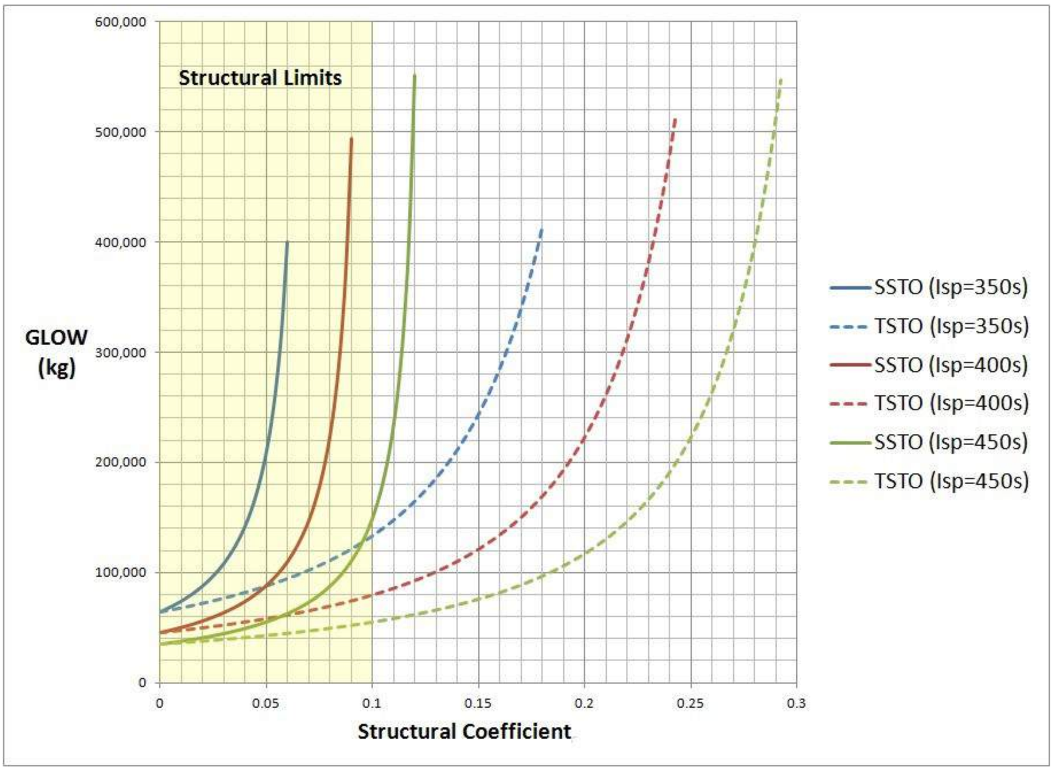


Figure 17: GLOW vs Structural Coefficient

The main advantage of SSTO's over DSTO's is that they can be made fully reusable, i.e., they can be sent to orbitals and can be reused many times on return, thus reducing the launch costs to a great extent. On the other hand, DSTOs have many advantages over SSTOs. As the first stage components get detached after the propellant is burnt, the total mass carried to the orbitals includes mainly the payload mass and no dead mass (the mass of the empty propellant tank as in SSTOs). Thus the high payload ratio of DSTO's indicates that we can send a large amount of payload with a small amount of propellant. Plotting a graph between the total initial mass of the rocket and structural coefficient shows that SSTO's require smaller structural coefficients than DSTOs when launched with the same payload mass and same propellant to achieve the same delta-v. However, currently, a structural coefficient of less than 0.1 is not attainable, limiting the structural efficiency of SSTOs.

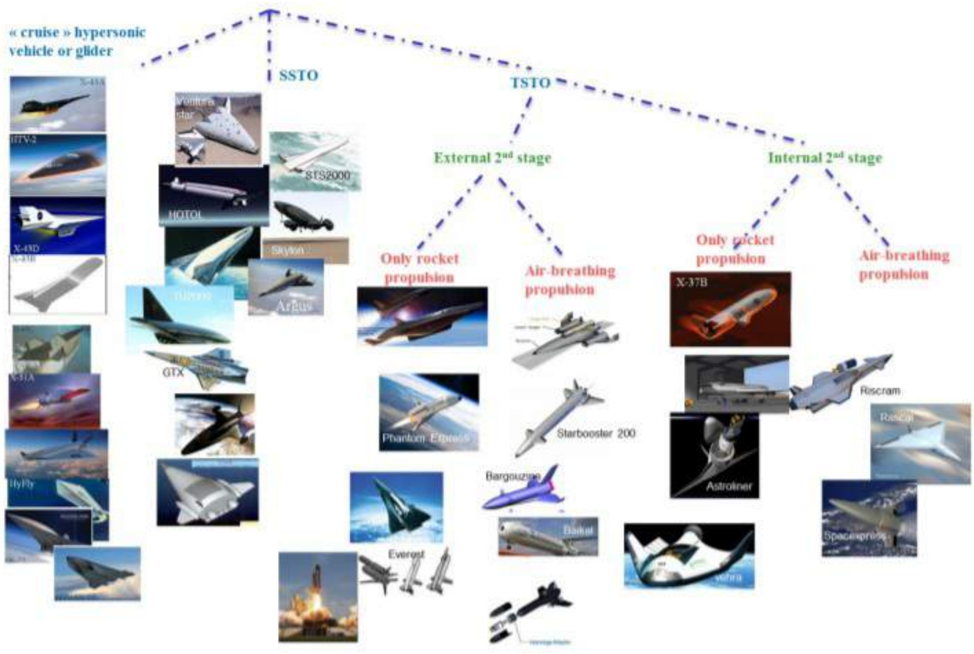


Figure 18: Complete review of SSTO and DSTO Rockets

Structural

8

8.1 Basic Structure

Structural system is one of the four major components of a full-scale rocket. The structural system of a rocket includes all of the parts which make up the frame of the rocket; the cylindrical body, the fairings, and any control fins. It transmits the load from the forces generated during the flight and provides low aerodynamic drag for the flight.

8.2 Dynamic Pressure

Dynamic pressure depends upon the velocity of the rocket and density of air.As the rocket rises, the velocity increases, thus the dynamic pressure increases to some maximum point called Max Q. It is given by q=(vš)/2, where q is the dynamic pressure is the air density and v is the velocity of the rocket. After this point the air density decreases due to which the dynamic pressure decreases.It is an important property that helps structural engineers to design rockets and to determine the minimum aerodynamic strength of the rocket so as to avoid buckling.

8.3 Buckling

Buckling is the sudden deformation in structural components due to excess load. It can cause complete collapse of that component. Structures can be internally pressurized to keep the walls from buckling. The net stretching force due to internal pressure is made is made greater than the compressional force due to flight loads so the walls experience no compression and buckling can be avoided. Sometimes thin skin panels are used to continue carrying load even in the buckled state.

8.4 Materials Used

1. The frame, is made up of strong, but lightweight materials such as aluminum or titanium as the performance of the rocket depends directly on the weight of the structure.

2. These are high density materials which can be stretched into thin sheets while maintaining their strength. They are also resilient to high temperatures.

3. The fins are attached to the bottom for stability during the flight.

4. It is also coated with a thermal protection system, to keep out heat of air friction during the flight.

8.5 Future Advancements

1. New advancements in making the frame of the rocket by using materials such as beryllium and composite materials using high strength filaments.

2. The best current high-temperature metals, e. g., nickel and ferrous alloys, may soon be replaced by molybdenum.

3. Ceramics such as carbides have very high melting points and show much promise for high-temperature use.

Relatively brief encounters with a hot environment can be survived by the protection method, as in the insulation of rocket nozzles and reentry nose cones.

Kerbal Space Program

9

9.1 What is Kerbal Space Program (KSP)?

It is a ***spaceflight simulation game*** based on a planet named **Kerbin**(Analogous to Earth) and other heavenly bodies in the solar system. It has a realistic **physics engine** and hence serves as a great medium to get a feel of space technology, orbital manoeuvres, Space Vehicles like rockets, aircraft, space planes, rovers etc.

Figure 19: Kerbal Space Program

The game is very well suited to plan a full-fledged **Space Mission** and is very detailed in terms of features in control systems, specifications and performance of parts used in making the vehicles. Even **NASA** embraced this game as the authentic simulations would help inspire a generation of interstellar explorers. This is why we have used it to simulate the concepts we learned during the lectures conducted.

9.2 Limitations of KSP:

Being a game to simulate real-life space exploration experience, it does have a few shortcomings in itself, which brings some deviation from reality:

- Kerbin, being quite smaller in radius than Earth(Nearly 600 km only), but nearly the same gravity as Earth, makes it denser than Earth.
- The specific impulse I_{sp} remains constant throughout, but in real life, it is dependent on mach number and varies constantly.
- The planetary distances are also lesser than the actual ones.

9.3 Mission Planning in KSP - Going to the Mun:

The mission given to us as a part of our project was to design a space mission that carried Kerbals to the mun, make a soft landing on the mun and bring the spaceship back safely to Kerbin. So we created a 3 stage rocket for this purpose. The recording for the mission can be accesed at - Recording
The full mission can be illustrated in the following steps:

- The pair of solid boosters provided the initial thrust, ensuring that we did not run out of fuel in the later stages. Around 2 minutes and 12 seconds after lift-off, the solid rocket boosters were separated as they ran out of fuel and will act as dead weight if we carry them any further.
- The rocket was tilted to achieve the required horizontal velocity to orbit around Kerbin. At around 3 minutes and 33 seconds, the rocket altitude was 70,000 m which is the height required for the low Kerbin orbit.
- The protective fairing was then separated safely as there is no more atmospheric drag that could damage the delicate instruments.
- When the rocket neared its apoapsis, a pro-grade burn was performed to achieve an orbit around Kerbin, and we finally reached the orbit at around 5 minutes and 41 seconds after liftoff.
- Then a trans-mun injection orbit manoeuvre was designed, which was later established. After the manoeuvres were demonstrated, we did a pro-grade burn at the required time and got it into a trans-mun injection orbit.
- After this, the rocket went from the Kerbins sphere of influence to the muns sphere of influence. ON reaching the periapsis of the mun, a retrograde burn was performed to slow down and get a stable orbit around the mun.
- Further retrograde burns were done to get a trajectory intersecting with its surface.
- Then the landing struts were extended to land safely on the muns surface.
- We waited for the altitude respective to mun to decrease and performed retrograde burns at the appropriate time to slow down the velocity and make a soft landing.
- After landing successfully, the ladders were deployed to get the Kerbals down safely on the surface of mun. After some time, they boarded back on the spacecraft, and it was time to return to Kerbin.
- And we lifted off from the mun and achieved an orbit around the mun.

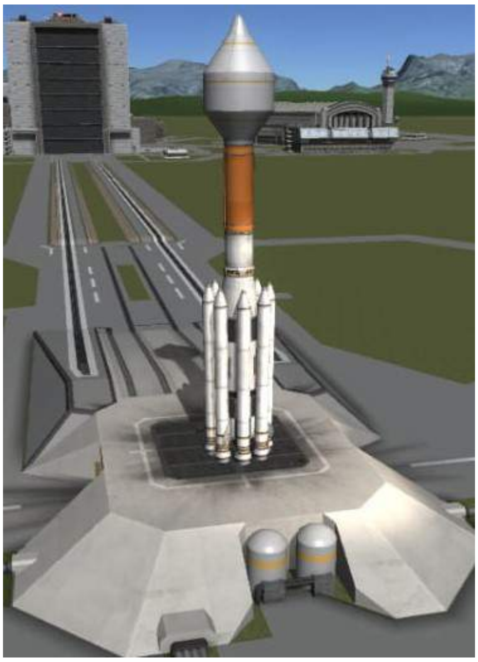


Figure 20: Before Launch

Figure 21(a): Stage Separation

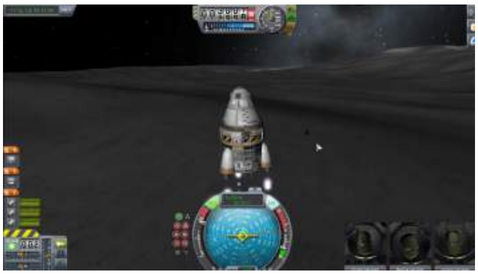


Figure 21(b): On MUN

9.4 Components of the rocket used:

Mk1-2 Command Pod, Mk16-XL Parachute, AE-FF2 Airstream Protective Shell (2.5 m), Heat Shield (2.5 m), Rockomax Brand Decoupler(x2), Advanced Reaction Wheel Module(Large), Z-4K Rechargeable Battery Bank, Rockomax X200-16 Fuel Tank, Rockomax X200-8 Fuel Tank, TR-XL Stack Separator, Mk-55 Thud Liquid Fuel Engine (x3), LT-2 Landing Strut (x3), OX-STAT Photovoltaic Panels (x3), RE-L10 Poodle Liquid Fuel Engine, Double-C Seismic Accelerometer, 2HOT Thermometer, Rockomax Jumbo-64 Fuel Tank, RE-M3 Mainsail Liquid Fuel Engine, Mk3 to 2.5 m Adapter, Mk3 Rocket Fuel Fuselage, Mk3 Rocket Fuel Fuselage Short, S3 KS-25x4 Mammoth Liquid Fuel Engine, Hydraulic Detachment Manifold (x8), S1 SRB-KD25k "Kickback" Solid Fuel Booster (x8).

9.5 Future Goals

Further, we would like to increase the efficiency of the missions we take up. Also, the next spot we will be aiming is to DUNA (Analogous to Mars).

10 Trajectory Optimization

10.1 Optimizing Trajectory

Rockets are defined by many variables and constraints, and ultimately deliver a payload to orbit at some cost. These characteristics provide the basis for an optimization problem. Maximize J_1 = Payload Mass (metric tons) and Minimize J_2 = Cost. The trajectory subsystem takes in several inputs and calculates the fuel usage and final altitude via the shooting method. It uses the ODE with a state vector composed of radial position, radial velocity, longitude, angular velocity, and mass. The model calculates the changes in velocity using the thrust, gravity, and drag applied at the correct angles. So, in overall define state vector being :

$$\mathbf{X} = [\ \mathbf{r}\ \theta\ v_r\ v_t\ \mathbf{m}\ ]$$

where, $\mathbf{r}$ = geocentric distance, θ = right ascension (Angular Displacement from launch pads initial position), $\mathbf{v}\ r$ and $\mathbf{v}\ t$ being radial and transverse velocity components. For position vector, we define **(ECI)** $Earth-centredinertial$ coordinate system while velocity vector being in **(LVLH)** $Local-Vertical-Local-Horizontal$ frame.

The resulting equations of motion $\dot{x} = f(x, u, t)$ are:

$$\frac{d}{dt}r = \dot{r} \tag{10.1}$$

$$\frac{d}{dt}\theta = \dot{\theta} \tag{10.2}$$

$$\frac{d}{dt}\dot{r} = -\frac{\mu}{r^2} + r\dot{\theta}^2 + \frac{T-D}{m}\cos(\alpha) \tag{10.3}$$

$$\frac{d}{dt}\dot{\theta} = \frac{T-D}{r*m}\sin(\alpha) \tag{10.4}$$

$$\frac{d}{dt}m = -\frac{T}{I_{sp}*g_0} \tag{10.5}$$

All these equations can be easily derived by force balance of the rocket in the radial and tangential directions.

Trajectory optimization, which uses gravity as the driving force to steer the rocket into a particular trajectory. During the gravity turn phase of the ascent trajectory the thrust direction is forced to be parallel to the relative velocity. In order to maintain the same equations of motion across all phases, the thrust magnitude, T is fictitiously split into two attributes, T_a and T_b. T_a represents the optimally controlled thrust contribution, while T_b is always parallel to the relative velocity. It can be noticed that T_a and T_b are alternatively null: during the zero-lift arcs, T_a is zero, while T_b is equal to the real thrust magnitude; conversely, T_a = T and $T_b = 0$ during the other propelled arcs. It offers two main advantages over a trajectory controlled solely through the vehicles own thrust:

1. The thrust is not used to change the spacecrafts direction, so more of it is used to accelerate the vehicle into orbit.

2. During the initial ascent phase the vehicle can maintain low or even zero angle of attack. This minimizes transverse aerodynamic stress on the launch vehicle, allowing for a lighter launch vehicle.

10.2 Plotting Curves

10.2.1 Thrust Force Spline Treatment

As the initial parameters, we are given thrust force generated by the Falcon 9 at 5 different altitudes: 0km, 50km, 100km, 200km, 400km as T_1, T_2, T_3, T_4 and T_5 respectively.

In between these altitudes the thrust is interpolated linearly. However, this approach could be adapted to use a spline instead of a simple linear interpolation. Initially the model used an exponentially decaying thrust, but this did not capture all of the characteristics of typical actual thrust profiles. Here is the spline treatment of the thrust curve of Falcon 9 as per available data:

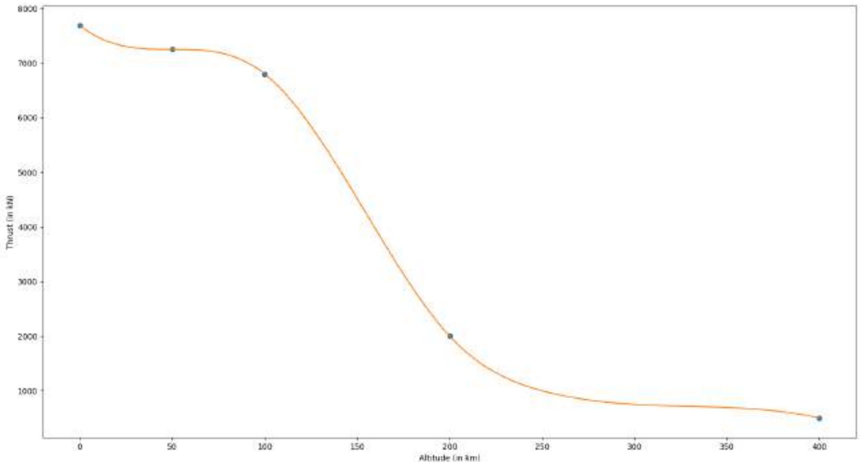


Figure 22: Thrust Force Spline Treatment

10.2.2 Thrust Angle

The thrust angle parameter variables α_1 and α_2 define the angle (with respect to a normal from the Earths surface) of the thrust vector over the course of the trajectory. α_1 is the altitude in km to start turning the rocket, while α_2 specifies the additional altitude over which to complete the turn.

If the altitude is less than α_1 then the angle is zero, and if it is greater than $\alpha_1 + \alpha_2$ then the angle is $\pi/2$. If it is in between then it is defined by:

$$Angle = [1 - cos(\pi * (\frac{A - \alpha_1}{\alpha_2}))] * \frac{\pi}{4} \tag{10.6}$$

No data was available for altitudes at which Falcon 9 started or ended the gravity turn maneuver, hence, data from the thrust force spline was used to estimate α_1 to be 100km and α_2 to be 250km.

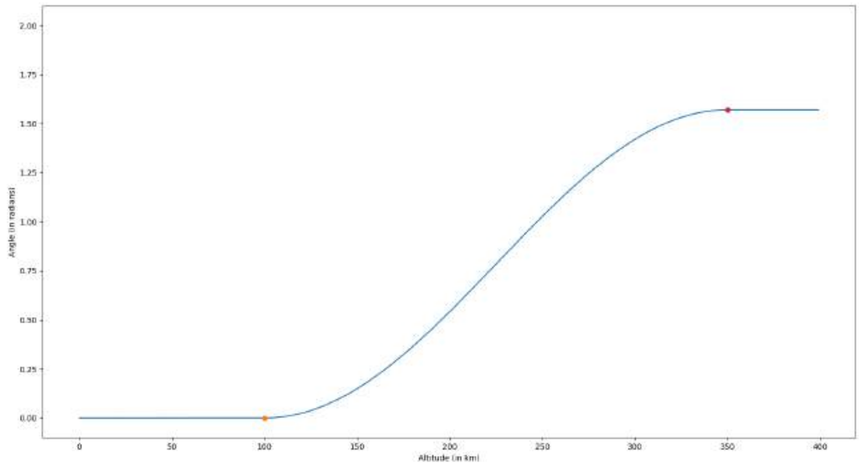


Figure 23: Angle variation with Altitude

10.2.3 Drag Force

The drag force on a rocket due to the atmosphere can be simply written as:

$$D = \frac{1}{2} C_d A \rho v_{rel}^2 \tag{10.7}$$

where..

C_d = Co-efficient of drag = $2 \sin^2(\theta_C)$
A = Reference surface
ρ = Atmospheric Density
v_{rel} = Velocity of Falcon 9 w.r.t the atmosphere

We consider the variation of ρ to be via isothermal exponential atmospheric model i.e:

$$\rho = \rho_0 \ \ exp(-\frac{r - r_0}{H}) \tag{10.8}$$

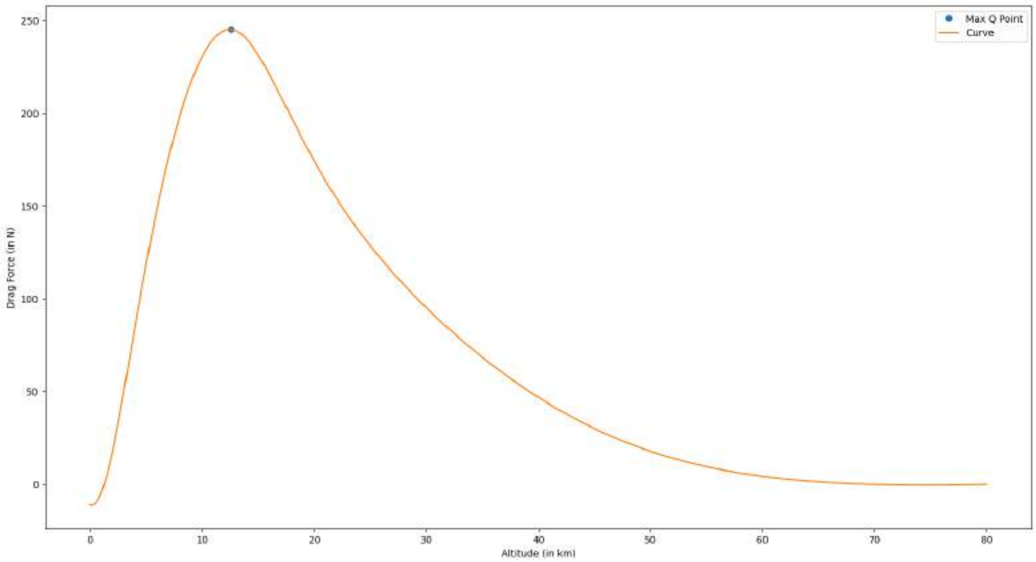


Figure 24: Drag force v/s Altitude

10.2.4 Mass and Cost Calculation

The propulsion subsystem inputs the mass of propellant from the trajectory subsystem, divides this mass up into oxidizer and fuel, and adds an ullage penalty. It also calculates the mass of the engine by scaling the Space Shuttle engine with max thrust according to the following equation:(Engine nozzle efficiency was not taken into account at different altitudes.)

$$m_{engine} = T_{max} * \frac{m_{ss}}{T_{ss}} \tag{10.9}$$

The cost subsystem calculates the cost for both materials and manufacturing. The material costs are based on material masses and engine mass. The engine is the largest of the material dry masses, and has the highest cost per kilogram, so it makes up the bulk of the material cost. The manufacturing cost is based on seam lengths. The cost parameters include cost per meter of seam and cost per kg of material. These parameters were taken from an external fuel tank model and have been scaled to produce numbers in the expected amounts.

Finally, the costs are summed and the payload mass is calculated according to following equation. Since the wet mass was an input, the mass that was not used up as fuel or taken up by structures is the available payload mass.

$$m_{payload} = m_{total} - m_{structural} - m_{oxidizer} - m_{fuel} \tag{10.10}$$

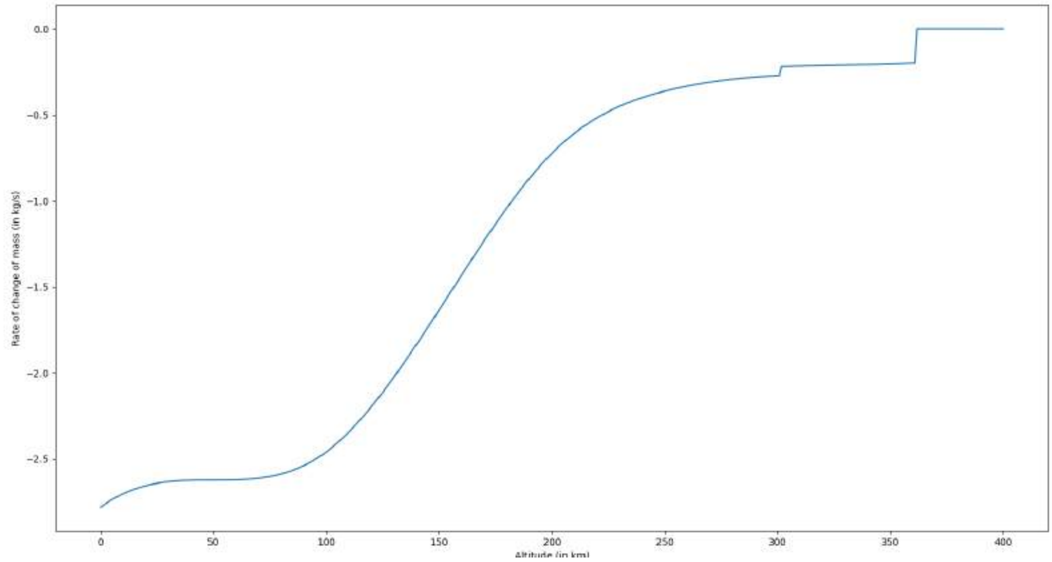


Figure 25: $\frac{dm}{dt}$ v/s Altitude

The abrupt change in rate of change of mass was expected since at an altitude of 300km, first stage separation was successfully completed and the second stage was fired. Hence, there was an abrupt change in I_{sp} value. At at altitude of 360km, we again observe a sudden jump. This is due to the fact that the second stage separation is also completed successfully. After that, we observe that there is virtually no change in mass. This is because the capsule resilience has now enough velocity to achieve an intersection of orbit with the ISS. Only minor fuel bursts are required for finer maneuvering, like docking with the ISS.

11 MOGA Modelling

Multi-Objective Genetic Algorithm, or MOGA, for short, is one of the most widely used algorithm for multiple variable optimization, in this case, maximizing payload capacity (J_1) while simultaneously minimizing the launch cost (J_2).

11.1 Algorithm:

1. **Choosing Design:** First, we choose multiple possible designs that could lead to a potentially optimized design. These designs are generated randomly and without any constraints.
2. **Populate:** We then proceed to populate the pareto front (the graph) using the data from these designs.
3. **Optimization:** Next, to optimize the data, we give a penalty to each design that dominated it by having a lower cost and higher payload capability. We also gave a penalty if the ending altitude was less than 400 km. The fitness was then squared to increase the gap between the more and less dominated designs. Finally, we give zero fitness for designs that were otherwise infeasible.
4. **Next Generation:** This fitness value was then used to decide which designs carried on to the next generation of the genetic algorithm. The fitness function for a feasible point is shown in the following slide:

11.2 Fitness Calculation

$$F = max(1.0 - 0.01 * n_{dom} - p(A_{final}), 0)^2 \tag{11.1}$$

A variable penalty shown below was used for the altitude constraint. The further the constraint was violated, the more severe the penalty applied. The penalty curve steepened with each generation. This is because a low curve would not penalize the infeasible designs enough, but a high curve would often cause the entire starting population to have zero fitness. By starting with a low curve and raising it, the MOGA was able to find the largest number of feasible designs. Example penalty curves from the 10th and 50th generations are shown here. If the penalty is greater than 1, then the fitness bottoms out at 0. This means that a penalty above the dotted line in the shown figure would lead to an infeasible design.

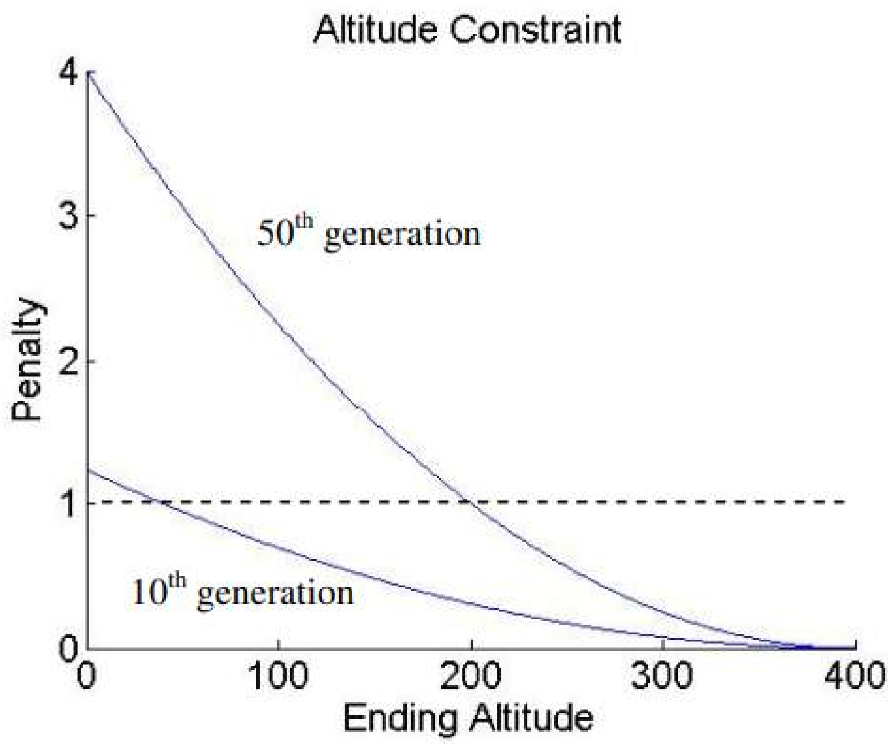


Figure 26: Penalty graph for 10^{th} and 50^{th} generation

$$p(A_{final}) = [(400 - A_{final})/max(1400 - 4 * generation)]^2 \tag{11.2}$$

where n_{dom} = no. of times an individual is dominated on the pareto front and A_{final} = Ending Altitude

11.2.1 Populating Pareto Fronts

The pareto front is populated by all the test cases that passed the fitness function. As we can see, a cluster is formed. Now, we need to choose the non-dominated individuals i.e. those points that have a better payload capacity and lower cost than their peers. (Refer the figure shown below for visual representation)

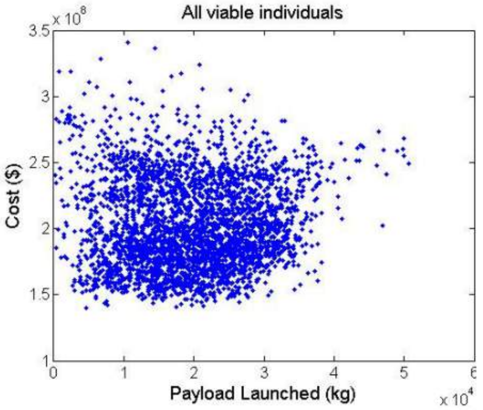


Figure 27(a): All Viable Individuals

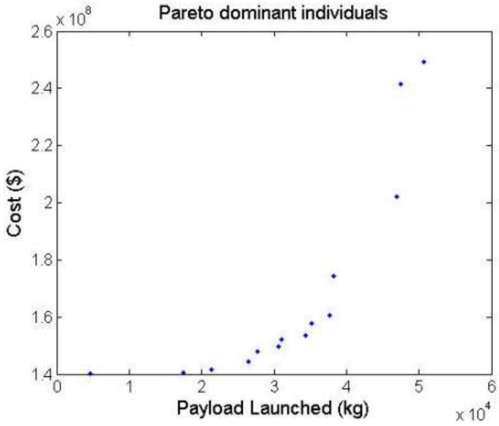


Figure 27(b): Pareto Dominant Individuals

The above process was repeated 10 times to get 10 different pareto fronts. Then all these scatter plots were merged and the dominated individuals were rejected due to the exact same reason and the dominant ones were kept/ selected. Below is the graphical depiction of the same:

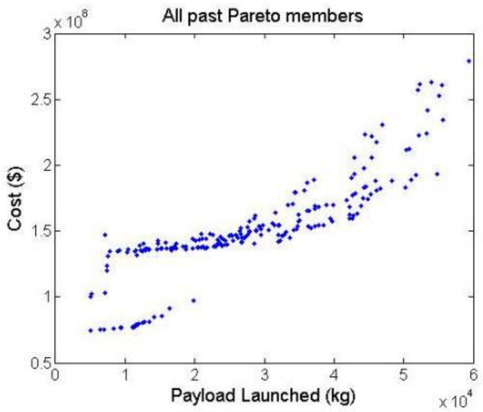


Figure 28(a): All past pareto members

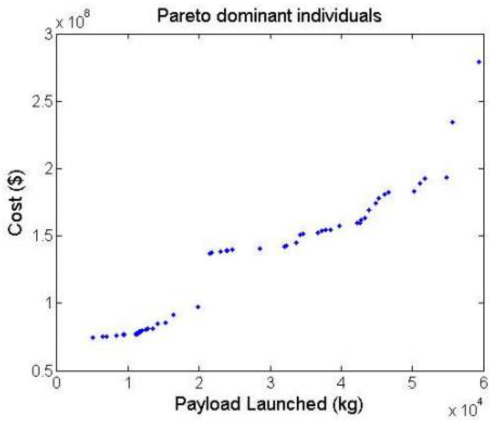


Figure 28(b): Pareto dominant Individuals

As we can clearly see, the latest graph is not continuous and this was expected, since we were dealing with random point generation. In order to get a smooth curve, more computation was required i.e. much more fronts were needed. So, instead of repeating the process 10 times, we did it 1640 times and obtained a pretty smooth curve of dominant individuals:

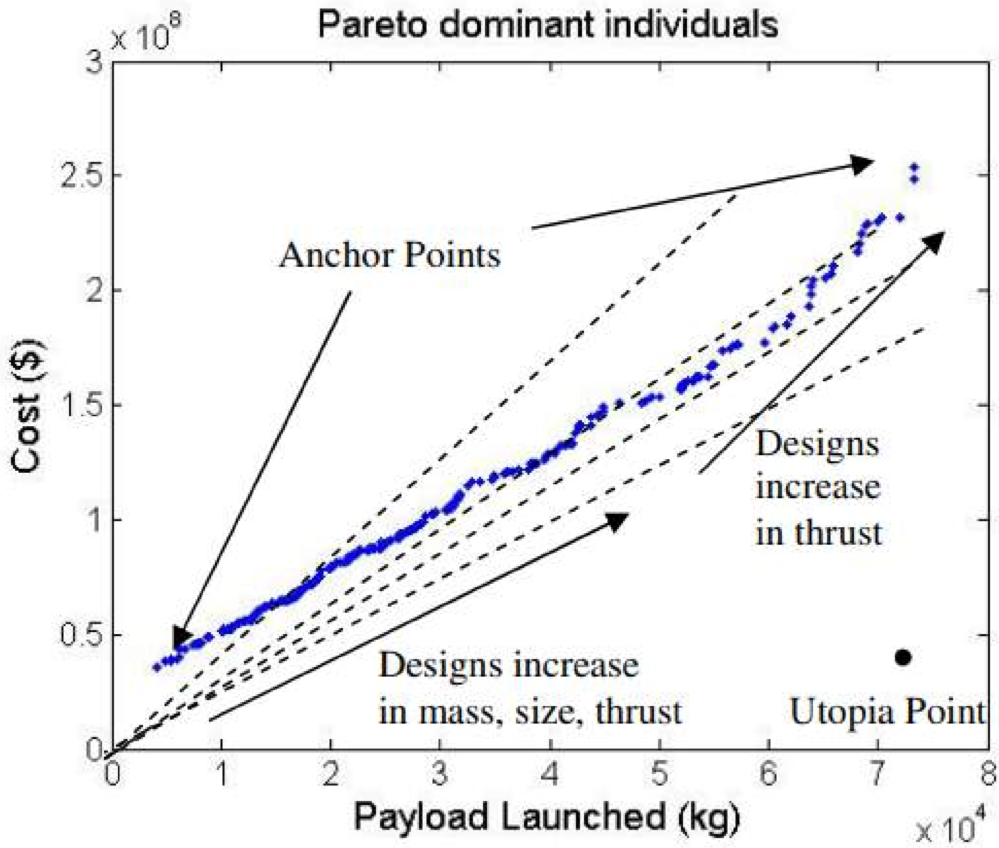


Figure 29: Pareto Dominant Individuals

The Pareto front is very linear, which is likely a result of the cost model being simple and heavily driven by engine size. We can also see the low cost and high payload mass anchor points. From those anchor points we can construct a normalized space with vertices (0,0), (1,0), (0,1), and (1,1). (0,0) is the low cost anchor point, (1,1) is the high payload anchor point, and (1,0) is the normalized utopia point. From this utopia point we can find the closest design, hereafter referred to as the best design.

11.3 Utopian Point

The following figure shows the normalized distance to the final utopia point as an increasing number of runs was performed. The Average line shows the progression of the average distance of each non-dominated point, while the Best line shows the distance of the closest point. The average distance tended to fall a little at a time, while the best distance tended to fall in jumps, which suggests a greater degree of randomness.

Figure 30: Optimizer Results

12 Engines and Propellants

12.1 Introduction

To leave earth and travel in space we need some force to overcome gravitational pull and change our velocity(speed+direction).Till now the best feasible way to apply force on an object that is going to far distances in vacuum(where there is no other significant energy source that can apply force in our desired direction) is expelling out mass carrying with us in the direction opposite to our desired direction of motion.So by Law of conservation of momentum our velocity will increase in desired direction.To apply more force we need to eject more mass as fast as possible with more velocity.

But in general our desired velocities are so huge that we need to carry a lot of mass with us to expel it out with some low velocity.So we can save money and resources by ejecting less amount of mass with very high velocity so that we can reach our desired velocity.

But how can we expel the mass with such huge velocities? Is there any such machine that can expel the mass with such high velocity? From where can we give energy to such a machine to work?

The best method to do this is to use energy from expelling mass itself to throw it with high velocity.So the expelling mass will be storing the energy to use while expelling it out.

So we are extracting the energy from the expelling mass by chemical reactions(forget about nuclear fission and fusion for now).

The mass that is using its own energy to expel out with higher velocity is called **FUEL**.Combustion is the most common chemical reaction that gives out huge amounts of energy.

But we need oxygen for combustion,but we don't have required concentration of oxygen even at sea level to carry out effective combustion to reach our thrust demands.So we will be carrying **OXYGEN** with us along with fuel to meet the thrust demand.

Here the machine that expels out the mass as fast as possible is called **ENGINE**.

12.2 Rocket Engine Hardware

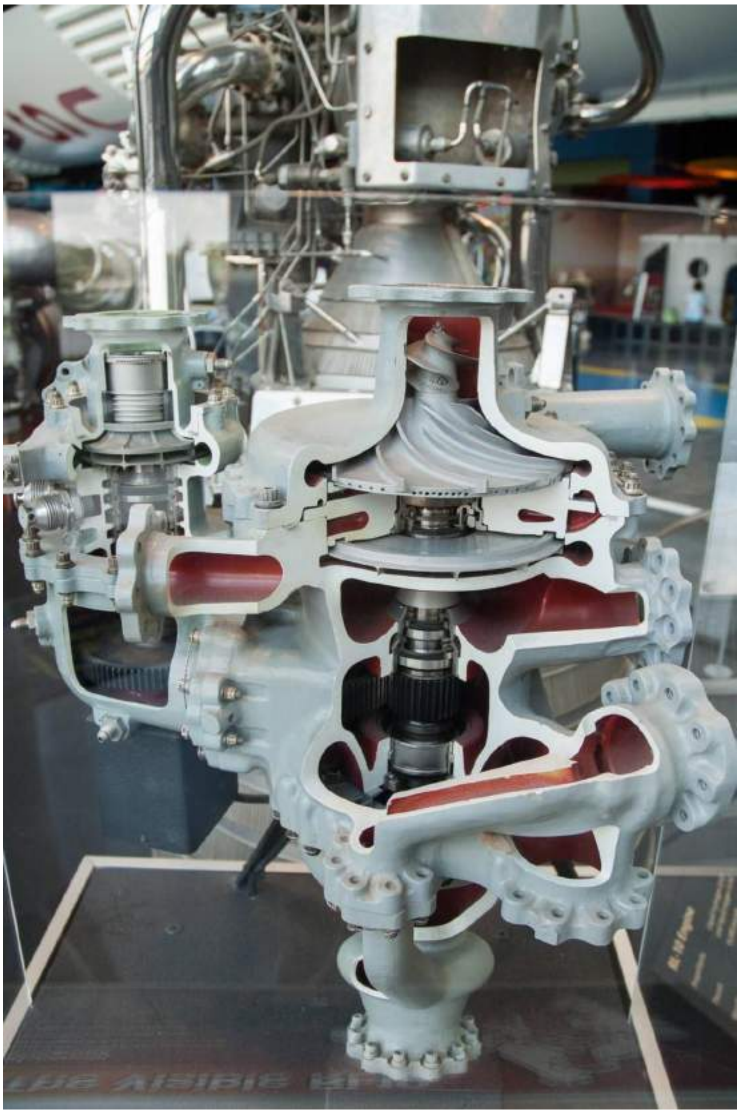


Figure 31: Turbo Pump

12.2.1 Turbo Pump

First of all fuel and oxidizer is stored in different tanks.These two undergo combustion in the combustion chamber so that the end products eject out with high velocities by absorbing the energy from combustion reaction.Thus we are able to eject the mass with higher velocities.

But the pressure and temperature is very high in the combustion chamber compared to pressure in fuel and oxidizer tanks.So propellant cannot move from high pressure to low pressure.So we want a higher pressure region in the propellants side than in the combustion chamber such that propellant moves from fuel tank to combustion chamber.

But if we increase the pressure in the tank such that it is greater than in the combustion chamber,we need to make tanks with very thick and costly material so it can withstand such pressures.And also the fuel tanks become very heavy so most of the fuel is consumed to accelerate them.So increasing pressure in the fuel tank is a bad idea.

Instead we will be using turbo pumps to increase the pressure such that it is greater than that in the combustion chamber.But we need to power the turbo pumps.As we already have a energy source from propellant,we will be using it to power the turbo pump instead of other sources.Some part of fuel and oxidizer mixture is diverted to the pre-burner where that fuel undergoes combustion and the products evolve out from the pre-burner with higher velocities,these gases are sent through the turbine,where they rotates the turbine and finally ejected out.The turbine is is connected to other two turbo pumps with same shaft thus powering them.

12.2.2 Pintle injector

Figure 32: Pintle Injector Spraying Water

Fuel and oxidizer should be mixed thoroughly in the combustion chamber for the combustion to be efficient.So the fuel and oxidizer are atomized(gaseous bubbles) so that they mix thoroughly.

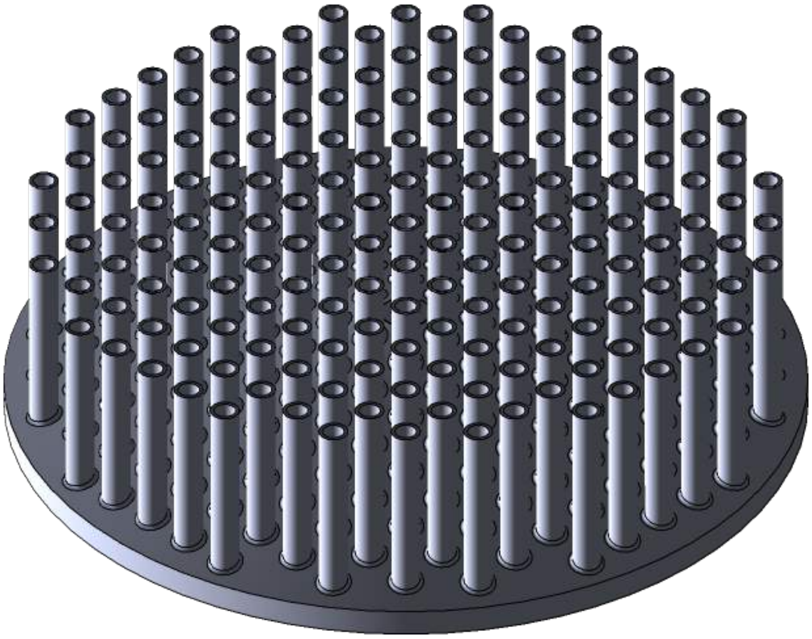


Figure 33: Pintle Injector CAD

12.2.3 Nozzle

Figure 34: Rocket Engine

After combustion of Fuel and oxidizer,end products are formed and they get kinetic energy by absorbing the energy released from combustion.But their velocities are in different directions and also small due to high pressure.We need to direct these gases in one direction and increase their velocity,so that we will be getting maximum thrust.

First of all we direct the gases in one direction by decreasing the area of cross section,thus velocity increases and pressure decreases.As the pressure decreases when we are moving from higher to lower cross section,we are able to direct the gases in one direction.As we move to lower cross-section the velocity keeps on increasing and finally we reach speed of sound at that point(at that particular temperature).

If we increase the velocity furthermore,then shock waves form and choking occurs.From now we will stop decreasing cross section area and start decreasing the pressure by increasing the cross section in a particular format as shown in the top figure.

As the pressure is decreasing the velocity keeps on increasing.But we need to stop expanding the nozzle exactly when the pressure equals the external pressure.If we dont do it then it will cause gas to diverge or converge leading to decrease in thrust.

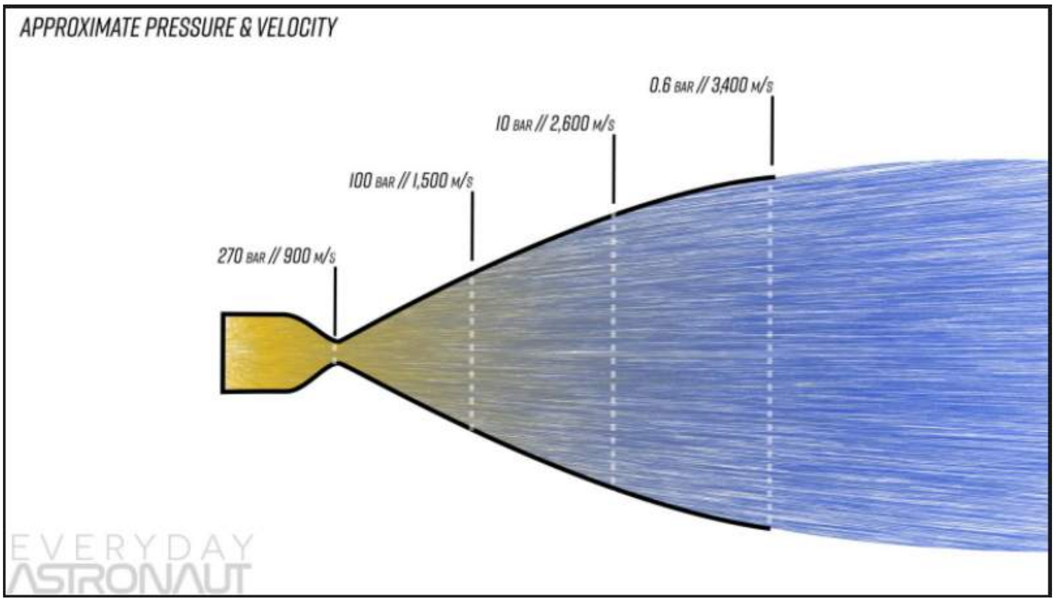


Figure 35: Nozzle

12.2.4 Regenerative Cooling

The Fuel is stored in very cool Temperature in The Fuel Tank.So We need to increase its temperature for making effective combustion.And also the temperatures in combustion chamber and in nozzle are very high such that no material can with stand such high temperatures.

So we will be passing this cool fuel around the nozzle and combustion chamber so that fuel absorbs that heat and cools them.This process is known as regenerative cooling.

Figure 36: Cold fuel passing through tubes inside the nozzle

12.3 Types of engines

12.3.1 Gas generator cycle

This is the most common and simple rocket engine type.Here Some part of fuel and oxidizer is sent to pre-burner to power the turbo pumps which eventually pumps Fuel into combustion chamber and also to the pre-burner.But the Propellant that is sent into the pre-burner is maintained fuel rich or to keep the temperature low to prevent turbine from melting.The remaining un-burnt fuel absorbs the heat and decreases the temperature.And finally these combustion products are exhausted out after passing through turbine(if passed into combustion chamber the soot will block the pintle injector holes leading to a blast).

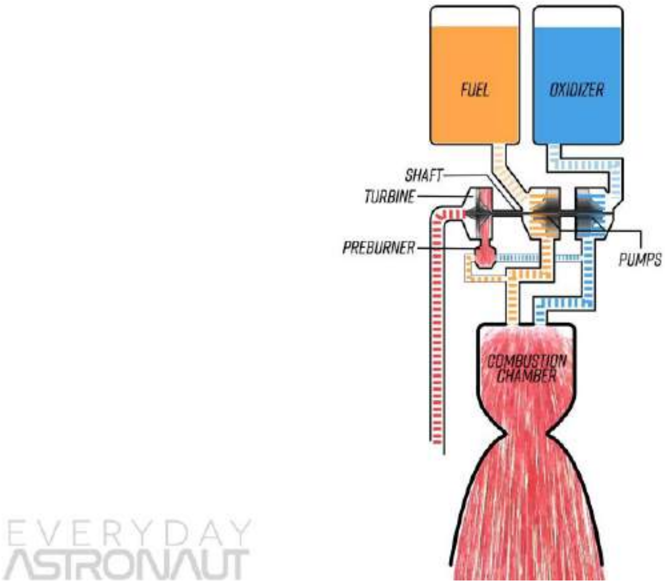


Figure 37: Gas generator cycle engine

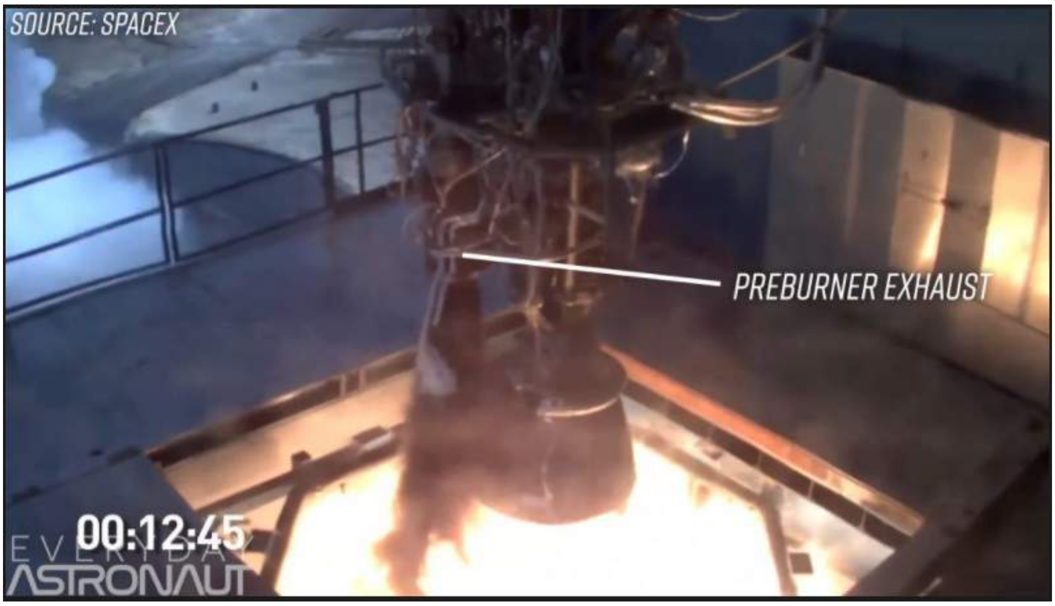


Figure 38: Pre-Burner Exhaust

12.3.2 Expander cycle

In Expander cycle engines the heat of nozzle and combustion chamber itself is used to power the turbine.First the fuel is passed through nozzle and combustion chamber.So it gets vaporized.And this vapour is used to power the turbine and finally enters the combustion chamber.

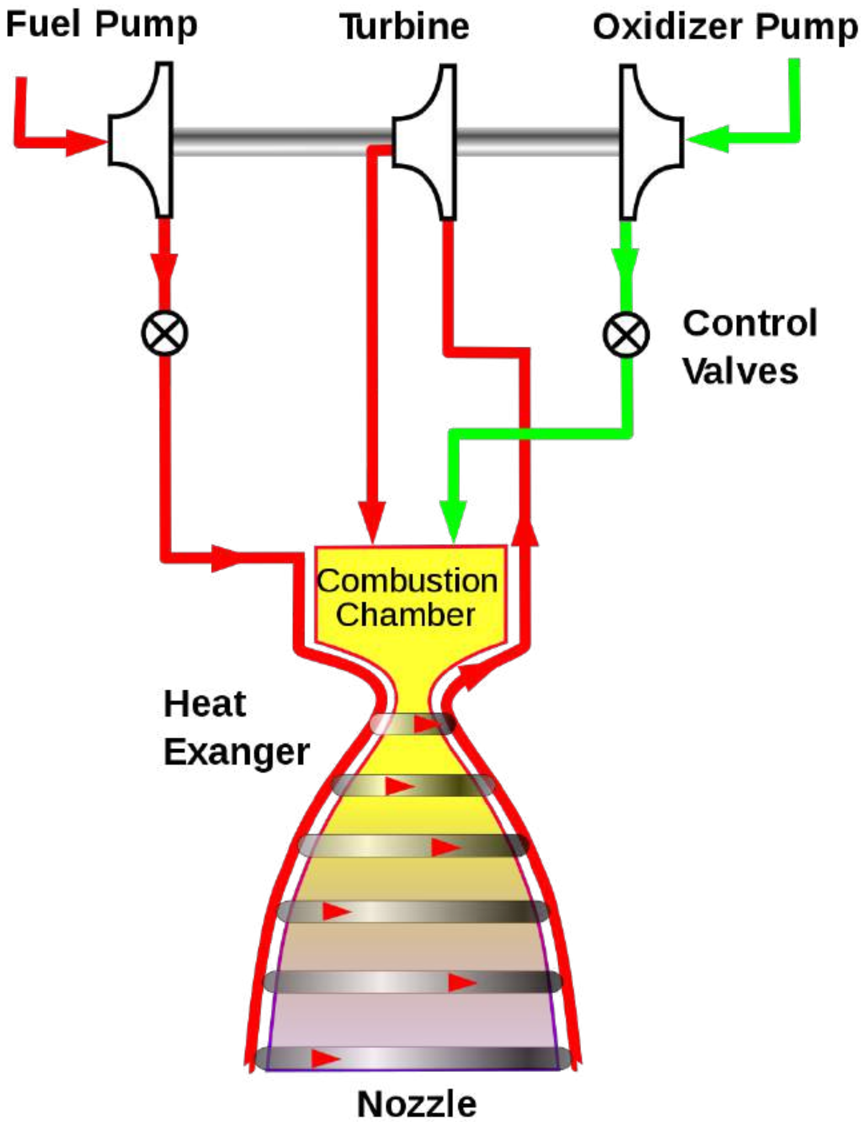


Figure 39: Expander Rocket Cycle

12.3.3 Staged combustion cycle

Here the complete Fuel rich or oxidizer rich propellant is passed into the pre-burner and the partially burnt exhaust powers the turbine and finally enters the combustion chamber.Here measures are taken to prevent choking due to fuel rich exhaust(soot).Thus the fuel is not wasted as it is in gas generator cycle engine.If it is oxidizer rich exhaust the measures are taken to prevent turbine material from corroding.

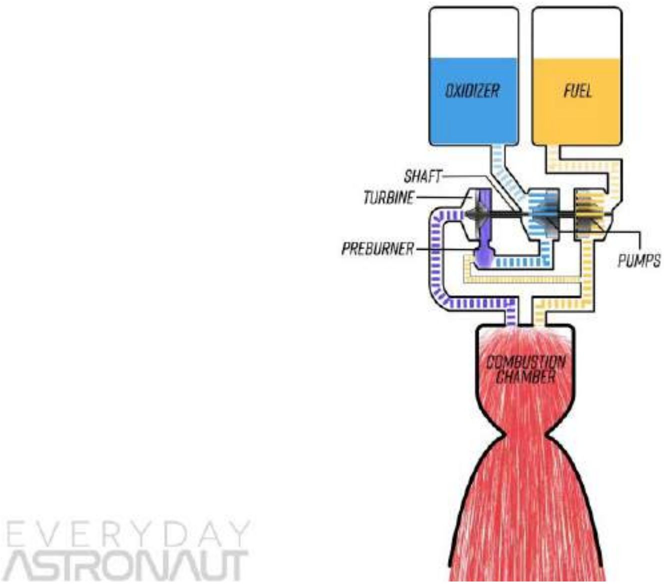


Figure 40: Oxygen rich stage combustion cycle

12.3.4 Full flow staged combustion cycle

Full flow stage combustion cycle engine is the combination of fuel rich and oxidizer rich stage combustion cycle engines.Here we will be having two separate pre-burners,turbines and turbo pumps.One is powered by fuel rich propellant which eventually powers the fuel pumping turbo pump.And other is powered by oxidizer rich propellant which eventually powers the oxidizer pumping turbo pump.This special setup is used in spite of staged cycle setup to prevent the blast caused by leakage of fuel and oxidizer through the shaft.This is a complex rocket engine,very tough to manufacture,but highly reliable.

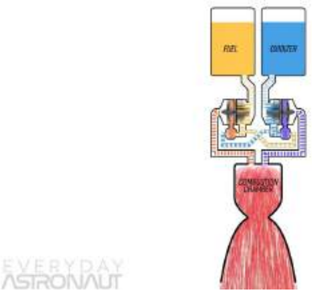


Figure 41: Full flow staged combustion cycle engine

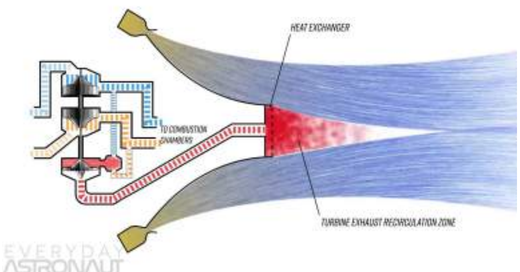


Figure 42: Aerospike Timeline

12.3.5 Aerospikes

These are special type of engines which are complementary to bell nozzle engines.These are very complex(tough to manufacture and keep it cool while running) and never flown till now.These engines can work at both above sea-level and vacuum.The exhaust from combustion chamber is directed in such a way that it always goes out straight irrespective of external atmospheric pressure.

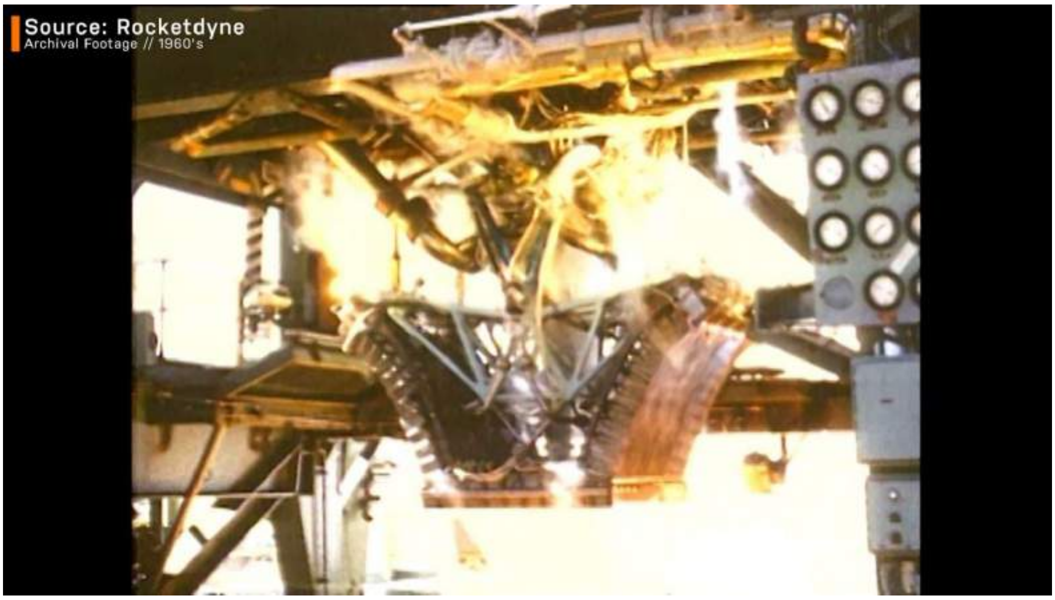


Figure 43: Linear Aerospike Engine

Figure 44: Toroidal Aerospike Engine

13 Orbital Dynamics

13.1 Kepler's Laws of Motion

This motion is described by the empirical laws of Kepler, which can be mathematically derived from Newton's laws. These can be formulated as follows:

1^{st} law: No edgings can be there on any planetary orbit. So possible smoothened orbit can be parabolic, hyperbolic, elliptic, circular(special).

From Newtonian law of Gravitation,

$$F = -\frac{GMm}{r^2} = -mA \tag{13.1}$$

$$A = -\frac{\mu}{r^2} \tag{13.2}$$

where,

A = acceleration of m caused by the gravitational force F of M

μ = GM = standard Gravitational Parameter

The equation of orbit under $\frac{1}{r^2}$ attractive force is given by

$$r\left(\theta\right) = \frac{p}{1 + ecos\theta} \tag{13.3}$$

$$p = a\left(1 - e^2\right) \tag{13.4}$$

$$e = \sqrt{1 + \frac{2EL^2}{m\mu^2}} \tag{13.5}$$

where,

a = semi-major axis

p = semi-latus rectum

e = eccentricity of conic-setion

E = total energy of moving planetary body

L = angular momentum of moving body

Thus, depending on the value of e, the orbit can be circular, elliptic, parabolic or hyperbolic.

	Energy	Orbit
e = 0	E < 0	Circular
0 < e < 1	E < 0	Elliptic
e = 1	E = 0	Parabolic
e > 1	E > 0	Hyperbolic

Figure 45: Planetary Orbit

Total energy of planets moving around the Sun is negative and thus planets move in an elliptic orbit around the Sun.

NOTE: During Mission-

From a circular orbit, thrust applied in a direction opposite to the satellite's motion changes orbit to elliptical; the satellite will descend and reach the lowest orbital point (the periapse) at 180 degrees away from the firing point; then it will ascend back. Thrust applied in the direction of the satellite's motion creates an elliptical orbit with its highest point (apoapse) 180 degrees away from the firing point.

2^{nd} law: The radius vector of a planet sweeps out area at a constant rate.

$$\frac{dA}{dt} = \frac{L}{2m} = constant \tag{13.6}$$

where,

A = area vector

3^{rd} law: Square of Time Period of body is moving directly proportional to third power of inter-body distances between them.

$$T^2 \propto a^3 \tag{13.7}$$

13.2 Escape Velocity

The specific energy (energy per unit mass) of any space vehicle is composed of two components, the specific potential energy and the specific kinetic energy. The specific potential energy associated with a planet of mass M is given by

$$PE(specific) = \frac{GM}{r} \tag{13.8}$$

while the specific kinetic energy of an object is given by

$$KE(specific) = \frac{v_2}{2} \tag{13.9}$$

and so the total specific orbital energy is

$$TE = KE + PE = \frac{v_2}{2} - \frac{GM}{r} \tag{13.10}$$

Since energy is conserved, TE cannot depend on the distance, r from the center of the central body to the space vehicle in question, i.e. v must vary with r to keep the specific orbital energy constant. Therefore, the object can reach infinite r only if this quantity is nonnegative, which implies:

$$v \geq \sqrt{\frac{2GM}{r}} \tag{13.11}$$

13.3 Hoffmann Transfer Orbit

In orbital mechanics, the Hohmann transfer orbit is an elliptical orbit used to transfer between two circular orbits of different radii around a central body in the same plane. The Hohmann transfer often uses the lowest possible amount of propellant in traveling between these orbits and thus is highly fuel efficient.

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):

$$E = \frac{mv^2}{2} - \frac{GMm}{r} = -\frac{GMm}{2a} \tag{13.12}$$

$$\Rightarrow v^2 = \mu\left(\frac{2}{r} - \frac{1}{a}\right) \tag{13.13}$$

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 Hohmann transfer can be computed as follows,

$$\Delta v_1 = \sqrt{\frac{\mu}{r_1}}\left(\sqrt{\frac{2r_2}{r_1 + r_2}} - 1\right) \tag{13.14}$$

$$\Delta v_2 = \sqrt{\frac{\mu}{r_2}}\left(1 - \sqrt{\frac{2r_1}{r_1 + r_2}}\right) \tag{13.15}$$

where,

$$a = \frac{r_1 + r_2}{2} \tag{13.16}$$

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 lower orbit

Δv_2 = Delta-v required to enter the higher Orbit from the Hohmann transfer orbit

Thus,

$$\Delta v_{total} = \Delta v_1 + \Delta v_2 \tag{13.17}$$

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

$$t = \frac{1}{2}\sqrt{\frac{4\pi^2 a^3}{\mu}} = \pi\sqrt{\frac{r_1 + r_2}{8\mu}} \tag{13.18}$$

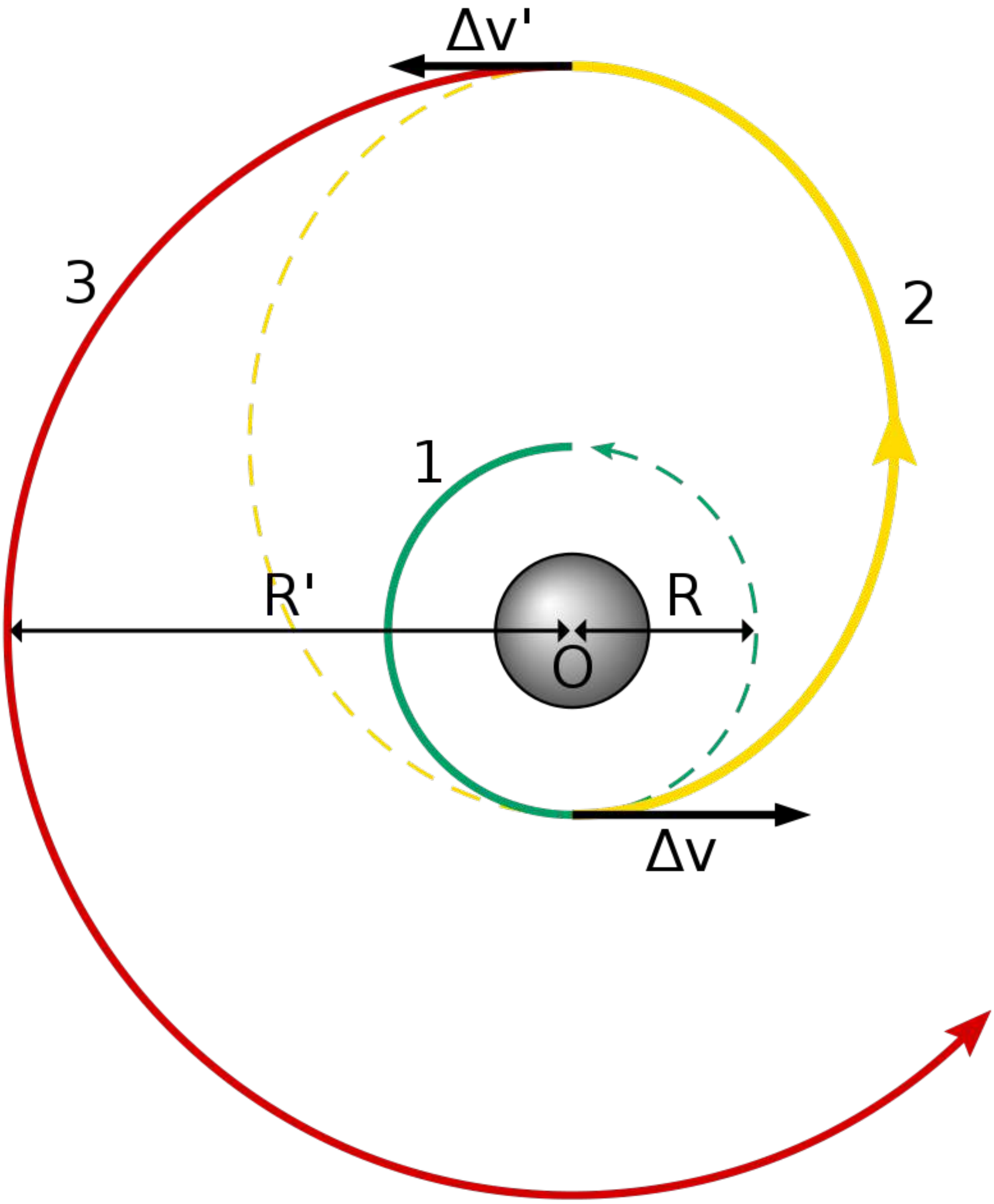


Figure 46: Hoffmann Transfer Orbit

Communication

14

14.1 Deep Space Communication

14.1.1 Introduction

The term **Deep Space** is generally used when referring to large distances from earth. NASA considers deep space to be any distance further than the moon which is roughly 384,000km. However,the European Space Agency (ESA) uses distances of greater than 2,000,000 km from earth. For the purpose of perspective,the NASA and ESA definitions equate the beginning of deep space to distances of 9.6 and 50 times the circumference of the earth respectively, significantly larger than any terrestrial communication links.

During each deep space mission,reliable communication with spacecraft,to send commands or software updates, track location and receive telemetry, images and scientific data is paramount to the success of the mission. This need for reliable deep space communication fuels a constant source of global interest for research and development in the area. In many cases,deep space research and development looks to develop existing technologies further for use in space, for example when pushing the boundaries of radio frequency communication.However, deep space communication,due to its unique challenges constantly push communication research and development to its limits.

Interplanetary missions have returned detailed images of our solar system, which are impossible to get from earth. For example the famous **Pale Blue Dot** image of earth from a distance of millions of km.These images,coupled with other scientific data have provided many thought inspiring discoveries, such as:

1,600 km/h winds on Neptune by the **Voyager** mission.

The discovery of Planet HD 106906 b and the relationship with its parent star which potentially challenges current gravitational theories.

The **Rosetta space probe** found organic carbon based compounds on a comet, which may provide clues to how life on earth began.

Figure 47: Earth from a distance of millions of km

14.1.2 Problems of Deep Space Communication

Imagine that you could not talk to your friends or family for six months. Now, imagine that you are also floating 250 miles above Earth. *What are some challenges you might face getting or receiving a message?*

Motion

Navigators must keep in mind when planning and executing a space mission that everything is moving. Not just the spacecraft, which may be traveling many thousands of kilometers per hour, but also the destination planet or moon. The Earth is rotating and moving around the Sun.

Distances

Navigators must account for the enormous distances between destinations. If Earth were the size of a softball, the International Space Station would be orbiting just above the seams, the Moon would be a marble about 2 meters (7 feet) away, and Mars would be 1.2 to 2.4 kilometers (.75 to 1.5 miles) away. The targets are small and moving.

Communication

Deep space missions are limited in the amount of power available for radio communication to and from Earth. Because the spacecraft travel so far from the Sun, they cannot generate as much power from solar panels as Earth satellites can. The radio signals they transmit are very weak and have to be picked out of background noise. The signals may take hours to reach the Earth. So a navigator cannot expect a quick response.

Gravity

The Sun's gravity determines the basic trajectory of an interplanetary spacecraft. But for deep space missions, a navigator also has to take into account gravitational forces from planets and moons and other forces that might affect the trajectory.

Data Transmission

So why is this so challenging?

First is the degradation of the signal. In transmitting data digitally, the **Bit Error Rate (BER)** is the amount of data that is altered or lost due to noise, interference, (or distortion). BER is a risk in all information sent digitally. When that data is sent across the stars, through the interstellar medium, the likelihood of errors increases exponentially. The errors occur as the transmissions encounter clouds of ionized gas: invisible pockets of distortion that are challenging to anticipate precisely.

Second is the ever-increasing energy demands of sending a signal over such massive distances. According to the inverse-square law, The power of any signal falls with the square of the distance traveled. Energy becomes the critical factor in data transmission. Any alteration to the message creates energy use: length of data transmission, sending to multiple sources at once, rate of data transmission. The delicate balance of maintaining enough energy to reach the destination versus keeping the exponentially rising energy needed to a manageable limit is a difficult process.

Oh, and one other thing: this is difficult if were talking stationary targets. If its planet to planet or star to star, we might be able to pull it off. But trying to communicate with a moving object is far more difficult. The challenge of communicating to a distant interstellar receiver is only compounded if that target is a ship traveling at the speeds necessary to cross that great gap.

14.1.3 Ideas to deal with the above problems

Communication at Interstellar Distances

How do we send data? How do we receive data? Using current methods, a simple approach is with a large ground-based or orbiting antenna dish hooked up to a nuclear power supply (remember the energy requirement is the square of the distance traveled). Assuming one bit per second, we can send roughly 40 megabytes (MB) of data every decade.

One idea held by the space research community posits that there are large portions of the microwave spectrum through which most of the material encountered in interstellar space would be transparent. If were sending signals, we know the frequency. But this only addresses the BER challenge and not the energy requirement.

Using the Sun as a Gravitational Lens

There is a unique idea that has come out of Italy that might address both of these issues: **turn the Sun into an antenna of sorts**. One highly-regarded astronomer, space scientist and mathematician has explored the use of the Suns gravitational lensing effect as a way to communicate at interstellar distances. Gravitational lensing is a discovery that is derived from the widely-accepted general theory of relativity. Weve even observed them, hundreds so far. Their use is of obvious benefit in the detection of **extrasolar planets**.

This researcher from Italy hopes that the Sun could be used to search for current extraterrestrial radio signals and then used to provide for sending and receiving data from other stars.

A **gravitational lens** works much like an optical lens except that its focal point is actually a focal line that is constructed of several focal points.

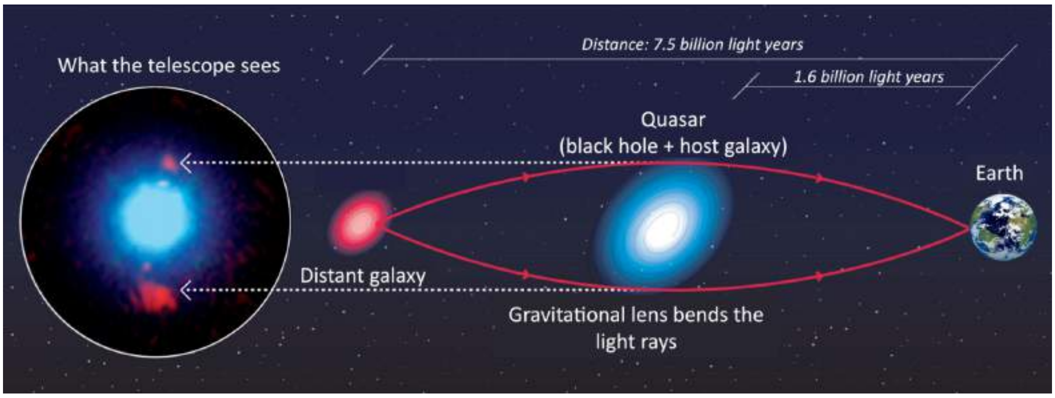


Figure 48: Gravitational Lensing

The same space scientist concludes (and proves) that the Bit Error Rate of transferring data will be unacceptable without gravitational lensing using any other method even to the short distance of Proxima Centauri. The ultimate dream and the end is the creation of a **radio bridge between two stars**. After digging through the various calculations, he determined that placing a probe behind Alpha Centauri A (the larger of the trinary star system) might require just one-tenth of a milliwatt to have perfect communication between the Sun and Alpha Centauri A through two 12-meter FOCAL spacecraft antennas.

However, all of this assumes technology that has yet to be invented. True, nothing exists yet that can actually make the gravitational lens real.

Interstellar Data Transmission

Another approach is an interstellar version of **Sneakernet**. *Whats Sneakernet, you might ask?* Its how the largest search engine in the world transmits massive amounts of data. In fact, if you want to transfer even a few hundred gigabytes of data, its generally faster to do overnight delivery of a hard drive than to send the files over the internet8. Until recent years, Cuba, largely cut off from the developing world, still had a massive proliferation of internet content through the physical distribution of media on USB drives hand-distributed one person to another.

120 terabytes of data, on a 100 megabit connection, would take nearly four months. Or it can be duplicated on a few hard drives and shipped overnight via ground transportation companies.

One space-based program working on interstellar space missions is proposing sending a micro-computer with a solar sail in the next decade or two to Alpha Centauri and the intent is to get that up to .20 lightspeed (which will still take it nearly 20 years to reach our nearest star).

40 years sounds like a long time and its one of the reasons that physical data packages are overlooked. Long ago, a well-known **nanotechnology** scientist put the work into debunking all of the reasons why probes wouldnt work. The conclusion was probes arent more expensive at the travel times were talking about in the interstellar medium. In the authors eyes, what does an extra decade or century mean when the data is guaranteed to arrive whole and all at once?

In one interesting approach, the founder of a global space-advocacy non-profit has suggested that rather than data drives, the physical delivery system should be microbes. He concluded, So, bacteria can be projected across interstellar space at essentially no power cost to the transmitting party, beyond that required to launch them to planetary escape velocity. *How do you load those packages up with important data?* The genetic material of common bacteria has potentially millions of bases bits of data encoded. This nanotechnology scientist concluded that bacteria can store data at a density of **900TB per gram**.

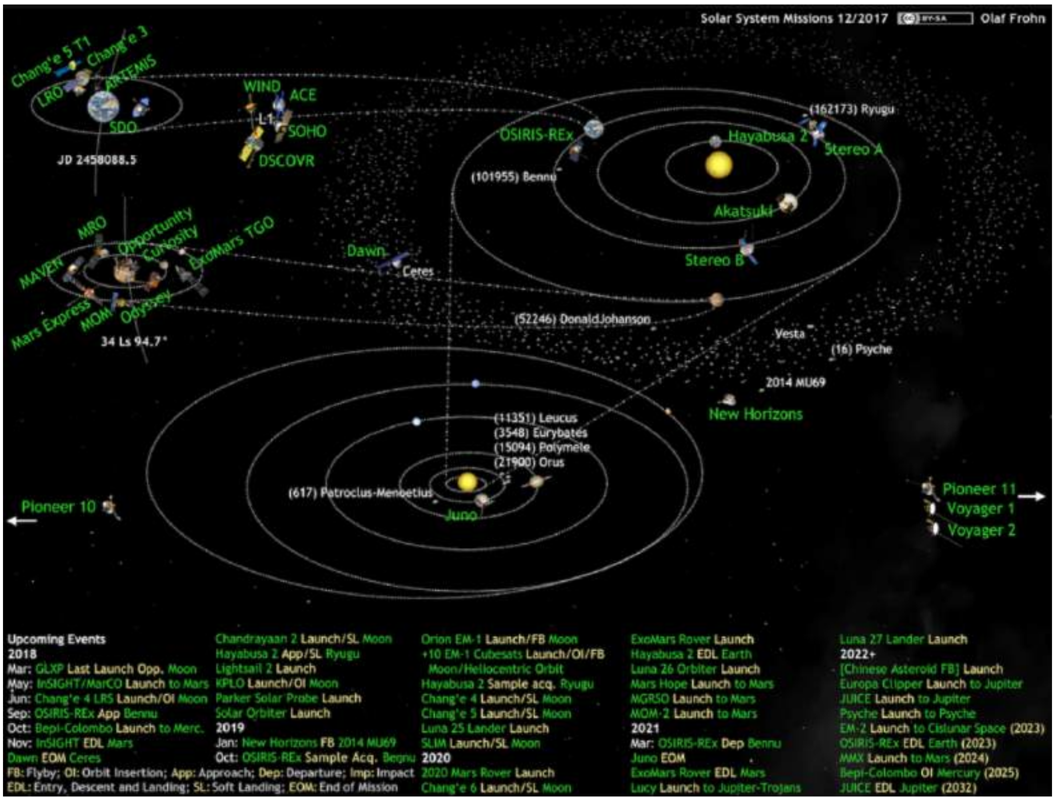


Figure 49: Upcoming Space Events

14.2 Deep Space Network

The Deep Space Network, or DSN, is much more than a collection of big antennas. It is a **powerful system for commanding, tracking and monitoring the health and safety of spacecraft at many distant planetary locales**. The DSN also enables powerful science investigations that probe the nature of asteroids and the interiors of planets and moons.

Deep space networks currently use **large parabolic dish antennas** to achieve two-way communication with spacecraft using RF (Radio Frequency).

RF communication is affected by interference from weather, solar radiation, industrial and household equipment, or other signals such as mobile phones and satellite television. Weather and solar radiation cannot be controlled, however interference from other sources can be minimized by selective placement of the ground stations.To do this, Space administrations and administrations have deep space network stations in areas far from dense industry or housing, in some cases areas that are protected by mountains.

Notable deep space networks:

NASA: NASAs DSN stations has one 70m-diameter antenna, one 34m and one 26m antenna.NASA has Deep Space Network (DSN) centres in California (USA), Canberra (Australia) and Madrid (Spain).

ISRO (Indian Space Research Organisation): Byalau space centre with one 32m antenna, one 18m antenna and one 11m antenna. ISRO also uses ship-borne terminals to track spacecraft in near earth orbit.

JAXA (Japan Aerospace Exploration Agency): Usuda deep space centre with one 64m antenna.

National Astronomical Observatories of China: Centres near Beijing with one 50m antenna, one 40m antenna near Yunnan and 18m antennas in Kashi and Quingdao. Also planned are 35m and 64m antennas in Kashgar and Jiamusi respectively.

Soviet/Russian Deep Space Network. Little up to date information is available. However the most recent information indicates Ussuriisk in Russia with 70m, 32m and 25m antennas and Yevpatria in Ukraine with one 70m and one 32m antenna, combined with an ADU-1000 array of 8,16m antennas.

14.2.1 DSN Functions

Telemetry

Telemetry data is made up of crucial science and engineering information transmitted to Earth via radio signals from spacecraft as they explore the far reaches of our solar system. The Deep Space Network, or DSN acquires, processes, decodes and distributes this data.

Spacecraft Command

Space mission operations teams use the DSN Command System to control the activities of their spacecraft. Commands are sent to robotic probes as coded computer files that the craft execute as a series of actions.

Tracking

The DSN Tracking System provides two-way communication between Earth-based equipment and a spacecraft, making measurements that allow flight controllers to determine the position and velocity of spacecraft with great precision.

Radio Science

DSN antennas are used by some space missions to perform science experiments using the radio signals sent between a spacecraft and Earth. Changes in radio signals between their transmission and receipt can provide lots of useful information about far off places in the solar system. Examples include probing the rings of Saturn, revealing the interior structure of planets and moons, and testing the theory of relativity.

Science

In addition to its vital role as the communications hub for deep space exploration, the DSN is also used as an advanced instrument for scientific research, including radio astronomy and radar mapping of passing asteroids.

14.2.2 Deep Space Network Antennas

All deep space network RF antennas operate in roughly the same way, utilizing a Cassegrain radiator design . Radio waves are received by the main parabolic reflector, and then reflected and focussed onto the receiving equipment. The receiving equipment then amplifies the signal, before sending onto a signal-processing centre as light down a fibre optic cable . The reflector is also used in reverse, to focus the energy into a narrow beam from transmitters when sending data .

The newer ground station antennas are **Beam Waveguide (BWG)** antennas. BWG antennas deviate from the traditional Cassegrain design by placing the sensitive receiving equipment underneath the dish in a belowground pedestal room, instead of centrally mounting above the reflector as in previous designs. This new placement makes repairs and upgrades much easier, allowing more equipment to be used and provides better thermal control.

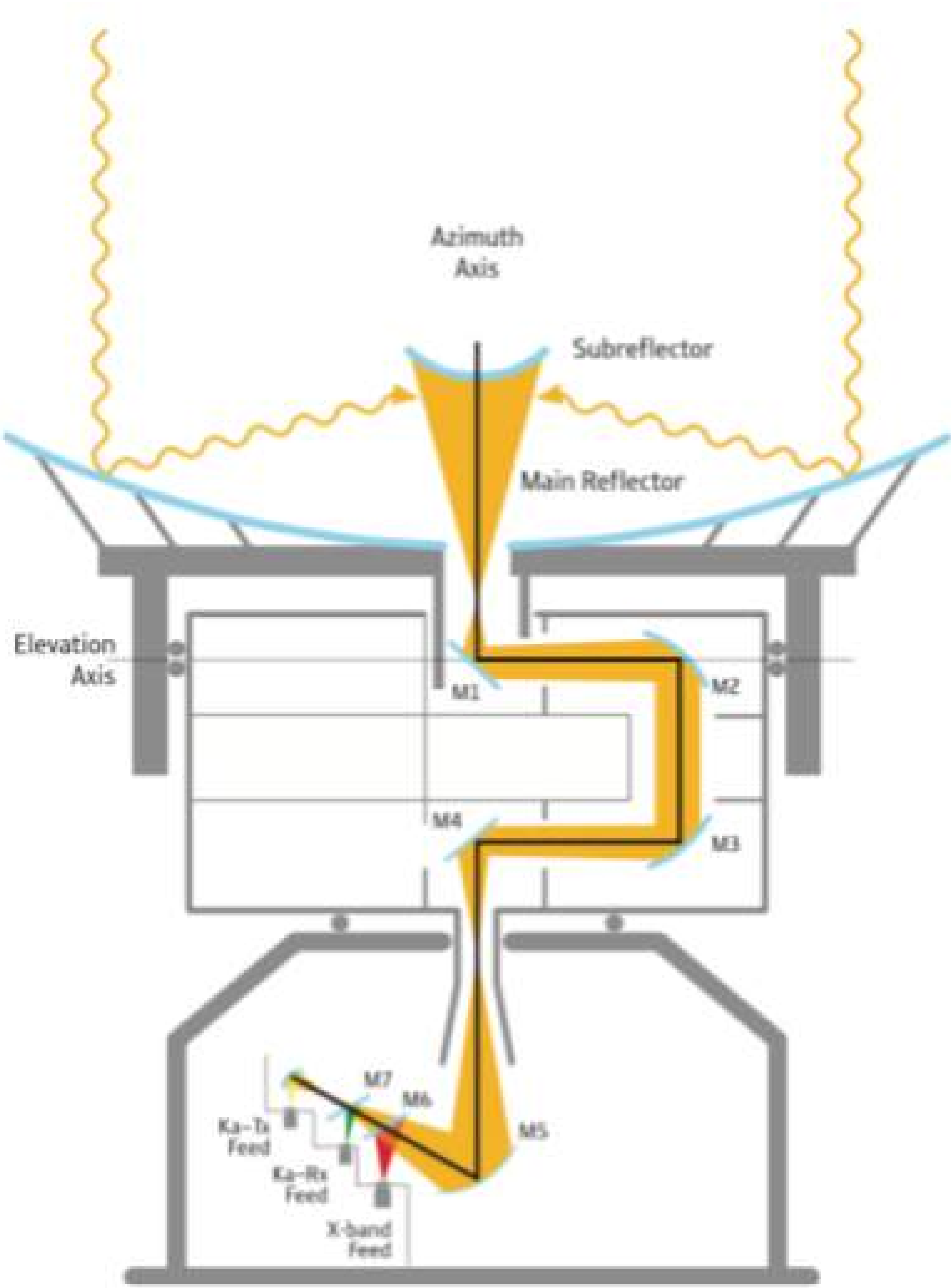


Figure 50: Beam WaveGuide (BWG) antennas

Each antenna generally supports more than one frequency band using several transmitters and receivers. The received frequencies are separated and directed to the appropriate receivers, using a series of dichroic mirrors.

Antennas built onto spacecraft follow the same Cassegrain design, but on a smaller scale. Spacecraft are normally fitted with multiple antennas, usually one 2-4m diameter High Gain Antenna (HGA) and one or more much smaller Low Gain Antennas.

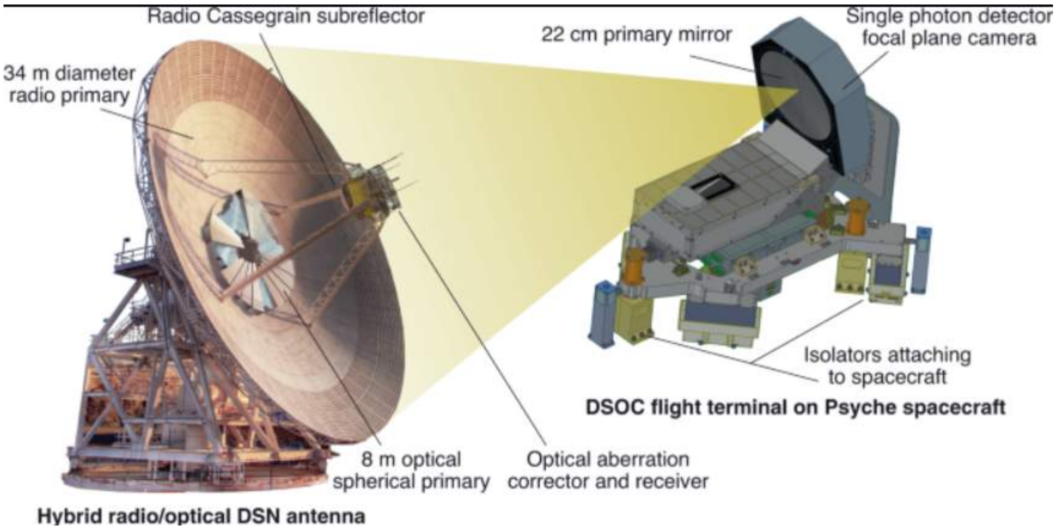


Figure 51: DSN Antenna

14.3 Radio Frequency Communication

The electromagnetic radio waves used in deep space RF communication, are part of the same spectrum as infrared, visible light, ultraviolet and X-ray. The radio waves travel in a straight line at around the speed of light, which is roughly 300,000 km per second in space.

The frequency ranges used for communication fit within a spectrum most commonly referred to as the microwave spectrum, which ranges from 1GHz to 300 GHz. This frequency range is broken down into a number of bands, and those typically used for deep space communication are shown below:

L-Band: 1.67 1.71 GHz

S-Band: 2.025 2.3 GHz

X-Band: 8-9 GHz

Ka-Band: 20-30 GHz

Higher frequencies offer higher data rates; however high frequencies are more affected by atmospheric interference, especially frequencies above 30 GHz. This is due in part, to shorter wavelengths being more easily affected by elements such as water droplets.

A radio waves power or intensity reduces with distance. The loss in intensity is roughly inversely proportional to the square of the distance, as shown in the formula below. Where I is the intensity (percentage) of the signal, p is the source point and d is the distance.

$$I = (p/d)^2 \tag{14.1}$$

This formula can be used to estimate, with all other things equal, how theoretical data rates reduce at the average distance of the planets in our universe, as shown below.

o Jupiter 5.2AU

o Saturn 10 AU 1/4 of bandwidth compared to Jupiter

o Uranus 19 AU 1/13 of bandwidth compared to Jupiter

o Neptune 30 AU 1/36 of bandwidth compared to Jupiter

An AU (Astronomical Unit) is the mean distance between the Earth and the Sun, defined by the International Astronomical Union as 1AU≈1.49*10^{14} m.

14.3.1 Polarisation and Modulation

Polarisation refers to the movement of the electric and magnetic forces within a radio wave. For example in linear polarization the radio wave moves up and down on a straight plane, like a typical sine wave. In circular polarisation, the radio wave moves in a circular motion, spinning right (Right-hand Circular Polarisation or RCP/RHCP) or left (Left-hand Circular Polarisation or LCP/LHCP). Circular polarisation is favoured over linear polarisation as it provide less loss . By using multiple transmitters, antennas can support simultaneous RCP and LCP to achieve higher data rates.

A single, steady wave cannot be used to transmit data as there is no way to separate the signal into the zeros and ones required to achieve digital communication. Modulation is the method used to encode zeros and ones within the radio communication. This is generally achieved by sending a carrier signal, which is steady and does not change, and one or more other signals, which differ from the carrier. Demodulation is simply the opposite, converting the received signal back to zeros and ones.

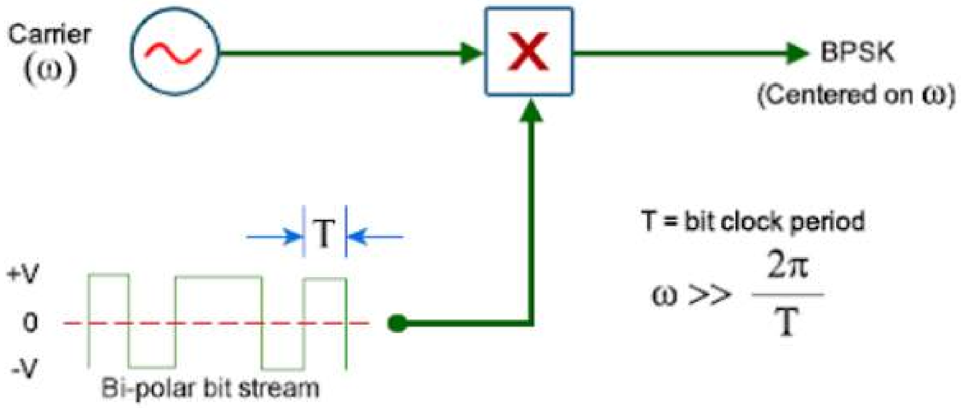


Figure 52: BPSK

The most common modulation method in use today by NASA and ESA is **Binary Phase Shift Keying (BPSK)**. BPSK sends signals, which are out of phase with each other by a predefined number of degrees, the degree offset is referred to as the index . Each detected shift is demodulated as a 1 (one) and is combined with the non-shifts as 0 (zeros) to create a bit stream.

14.3.2 Error Detection and Correction (EDAC)

In practice, the stream of zeros and ones received after demodulation will have a number of errors, due to noise picked up during the communication. For example, unwanted phase shifts introduced when the signal passes through the earths atmosphere. The errors will be in the form of zeros interpreted as ones and vice-verse known as bit-flips.

In order to achieve near error-free communication, a number of methods have been developed over the last few decades. The most common form of EDAC used in space missions is **Forward Error-Correction (FEC)**. This method sends additional bits (overhead), which can be used to check the consistency of the received data and then rebuild parts of the data stream if required.

NASA uses Reed-Solomon and Turbo Code for error detection and correction. Reed-Solomon was introduced as part of the Voyager mission, replacing the previous Golay method. This reduced the number of overhead bits by 20 percent and reduced the bit-error rate from 5 x 10^{-3} to 5 x 10^{-6} , the latter meaning only 5 bits out of one million are incorrect.

14.3.3 Data Rate and Bandwidth

Although data rate and bandwidth are often interchanged to mean the same thing, they are actually quite different. Data rate or throughput generally refers to the capacity of a network link and is measured in the number of bits per second that can be sent e.g. 2 Mbps (Megabits per second). Bandwidth generally refers to the range of frequencies available to transmit the data and is measured in Hz. In addition to the available bandwidth, the overall capacity of a network link is also governed by the quality of the received signal. This quality is most often referred to as the signal to noise ratio, S/N or SNR. The most notable piece of work describing the relationship between capacity, bandwidth and SNR is the Shannon-Hartley Theorem, typically represented by the formula below.

$$C = Wlog_2(1 + SNR) \tag{14.2}$$

Where *C* is the achievable capacity of a channel (in bits per second), *W* is the bandwidth of the channel (in Hertz), and *SNR* is the signal-to-noise ratio.SNR is usually expressed as decibels (dB) by the following formula.

$$10 * 10log_{10}(SNR) \tag{14.3}$$

SNR can, therefore, in conjunction with the number of available frequencies, have a significant effect on the data rates of any given communication. This formula is used to calculate the theoretical maximum capacity of a network link (known as the Shannon-Limit) using a value of zero for SNR. Network and communication engineers and scientists have been continually evolving communication methods and error correction algorithms for decades, attempting to develop communication methods, which are close to the Shannon-Limit.

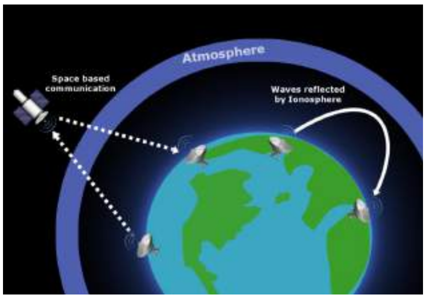


Figure 53: Influence of Atmosphere on Transmission

15 Habitability

Figure 54: Artistic impression of colonisizing other planets

If the human race is to survive in the long-run, we will probably have to look for life on other planets. Whether we make the Earth uninhabitable ourselves or it simply reaches the natural end of its ability to support life.We effectively already have one space colony, the International Space Station (ISS) and much of the technology developed for the ISS, such as radiation shielding, water and air recycling, solar power collection, is certainly transferable to future space settlements.The first requirement for a human settlement is a habitat. As the duration of spaceflight increases, the impact of habitability on mission success becomes significant..It is defined by the physical interface between the human user and the system/environment.it basically refers to :-

- Quality of life
- System usability

- Human-machine-environment mission interactions
- Physiological, psychological, social, and cultural factors.
- Performance, health, and well-being during duty and off-duty periods

An habitable environment should able to provide its inhabitants with a suitable biological conditions like :-

15.1 Proper atmospheric and climatic conditions and temperature ranges

15.1.1 Terraforming

Figure 55: Terraforming Mars

It is basically the process of transforming a planet that is not suitable for human life into a habitable one. There will be varying considerations depending upon the planet of interest like the atmosphere might be too toxic to breathe in or it may be too thin to be of any significant help in keeping the planet warm enough through the greenhouse effect (as in the case of Mars) or it may have an excess amount of greenhouse gases creating the exact opposite effect of too much greenhouse effect making the planet extremely hot (as in the case of Venus). These different conditions call for different approaches to make them habitable and are technologically far off into the future. So for now our mother earth is the only planet we have.

15.1.2 Generation ships

Figure 56: Concept Model

Now lets go beyond mere planets, to distant star systems in generation ships. It is a spacecraft on which a crew is living on-board for several decades or even centuries, such that it comprises multiple generations while travelling light years to other worlds. These should be entirely self-sustaining providing energy, food and water for everyone on board. It must also have extraordinarily reliable systems that could be maintained by the ship's inhabitants over long periods of time. This would require testing whether thousands of humans could survive on their own before sending them beyond the reach of help. Similar to make shift habitats space these will be constructed with the help of space mining.

15.1.3 Makeshift habitats

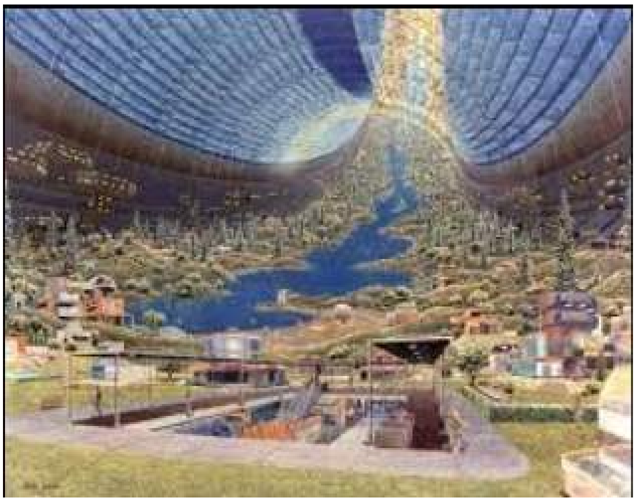


Figure 57: Inside a O'neill cylinder

A much more feasible alternative to terraforming other planets is to build our own habitats which could be located anywhere in the solar system, could be of any size that material science allows, and have different characteristics, such as temperature, climate, gravity, and even lengths of day. The materials required to build these humongous structures could be mined from space and the construction could be done in space itself. Unfortunately, we are still a very long way from building anything like a fully sized habitat. However, we are now one step closer to doing so with the release of a paper from a team at Texas A&M that describes a way to build an expandable space habitat of concentric cylinders that can house up to 8000 people.

15.2 Ceaseless supply of oxygen,water,food ,energy and other useful resources

15.2.1 Liquid lifelines

Once the habitat is built, the colony will need continuous supplies of water, oxygen, energy and food to sustain its inhabitants, presuming the colony wasnt built on an idyllic Earth-like planet with these resources in abundance.An initial settlement would need to carry a certain amount of water and recycle all waste liquids. But a colony would also likely try to extract water, possibly from underground supplies of liquid as may exist on Mars or ice, as can been found under the surface of certain asteroids. Water also provides as a source of oxygen via electrolysis.Research agencies is also working on developing techniques to regenerate oxygen from atmospheric byproducts, such as the carbon dioxide we exhale while breathing.

15.2.2 Agriculture in space

Figure 58: Artistic Render

Perhaps space will be the next suburb and before we start sending humans on an intergalactic bus ride, we must figure out new ways to accomplish everyday tasks in space, like growing food.Space farming requires greater understanding if humans are to survive in space without constant contact from Earth. Space farming simply refers to growing plants in space.But there are many challenges with farming or growing plants in environmental conditions of outer space and in planets and celestial objects like:- Less gravity :- Plants on earth depend very much on its gravity for transporting water in minerals throughout the body.So to encounter this loss of gravity in space, one has to produce artificial gravity with mechanical centrifuge.

Artificial lightning :-Most plants on Earth have access to loads of natural sunlight and grow toward that light, but researchers must fool plants growing in space to follow this same behavior. Therefore choice of lighting in the growth chambers is an important consideration.

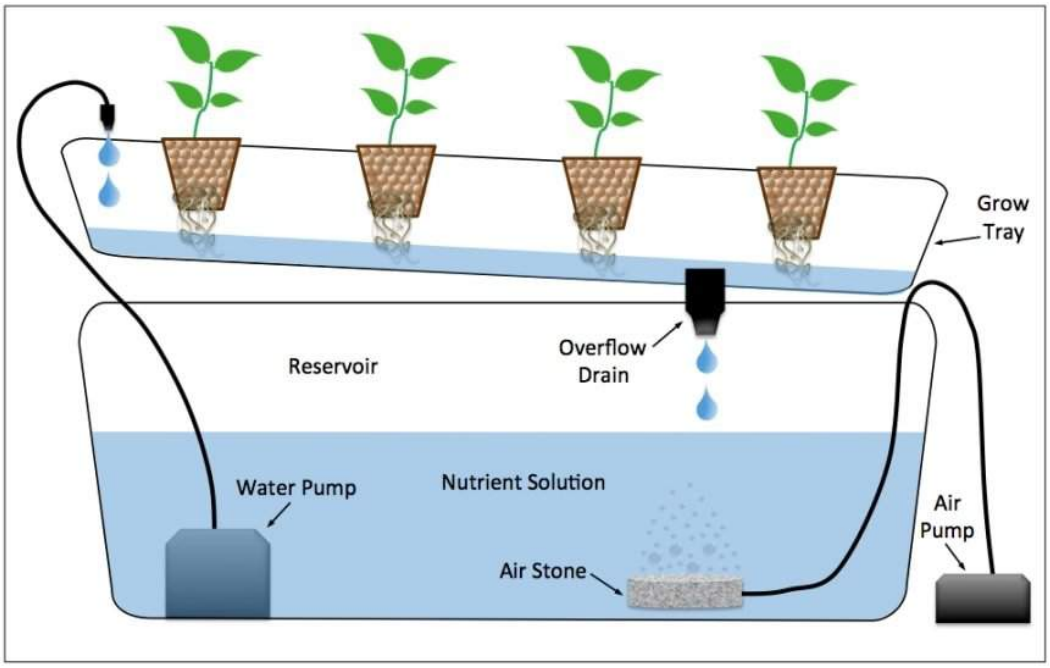


Figure 59: Hydroponics System

Rooting materials :- Hydroponics is the solution to the non availability of rooting material or soil in space. The nutrients required by the plant, in this water-based-agriculture method, is dissolved into the liquid in which the plant is sitting.The plants sit entirely in water without the presence of soil.

15.2.3 Space mining

Figure 60: How space mining would look like

Celestial bodies are potential sources for dozens of natural materials that, in the right time and place, are incredibly valuable. The so-called rare earth metals are potential targets of asteroid miners intending to service Earth markets. Consisting of 17 elements, including lanthanum, neodymium, and yttrium, these critical materials (most of which are mined at great environmental cost) are required for electronics.The Moon is a prime space mining target.It has several advantages.Its low gravity implies that relatively little energy expenditure will be needed to deliver mined resources to Earth orbit.

15.3 Other factors like radiation shielding and less or more gravity

15.3.1 Artificial gravity

our bones and muscles dont work as hard in a gravity-free environment. This weakens them. Without gravity, blood and other bodily fluids dont flow normally and can collect in the upper body.Plants also use gravity for transportation of water and other minerals to leaves and other parts from their roots. Artifcial gravity, due to its ability to mimic the behavior of gravity on the human body has been suggested as one of the most encompassing manners of combating the physical effects inherent with weightless environments.

15.3.2 Radiation Shielding

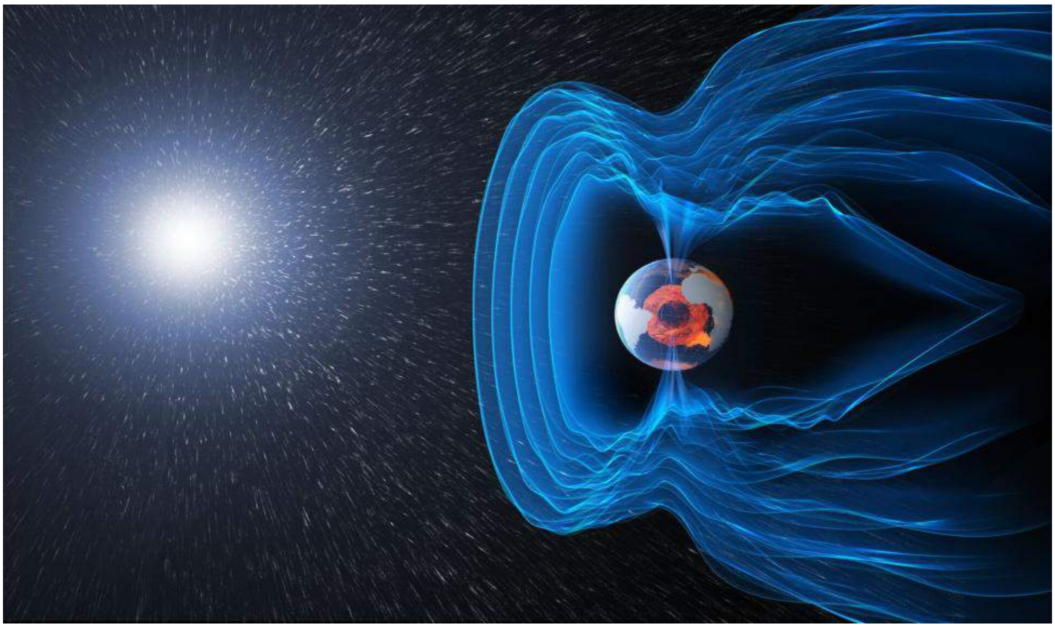


Figure 61: Magnetosphere shielding radiation

Radiation protection, also known as radiological protection,is referred to as The protection of people from harmful effects of exposure to ionizing radiation from sun and other bodies[1] Exposure can be from a source of radiation external to the human body or due to internal irradiation caused by the ingestion of radioactive contamination. Today, the Apollo-era flares serve as a reminder of the threat of radiation exposure to technology and astronauts in space. Understanding and predicting solar eruptions is crucial for safe space exploration. . The International Space Station cruises through low-Earth orbit, within Earths protection, and the stations hull helps shield crew members from radiation too.The more mass between the crew and radiation, the more likely that dangerous particles will deposit their energy before reaching the crew. Agencies are meeting the radiation challenge with creative solutions, developing technology such as wearable vests and devices that add mass, and electrically charged surfaces that deflect radiation.

16 Unconventional Methods of Propulsion

16.1 Near Future Methods

16.1.1 Explosion Powered Rockets

This type of rocket engine mainly uses controlled explosions to boost up into space. Its called a **Rotating Detonation Engine** (RDE), and it promises to make rockets lighter, faster, and simpler. RDEs are fundamentally the same as all other rocket engines, however, instead of using the pressurization systems to mix the fuel with the oxidizer, they use the shock waves from the detonation of Hydrogen bombs to provide the pressure for mixing up the above elements.

A famous example includes **PROJECT ORION**.

The Orion concept offers high thrust and high specific impulse, or propellant efficiency, at the same time. Its nuclear pulse drive combines a very high exhaust velocity, from 19 to 31 km/s in typical interplanetary designs, with mega-newtons of thrust.

Since weight is no limitation, an Orion craft can be extremely robust. An uncrewed craft can tolerate very large accelerations, perhaps 100g, thus easily attaining 1% of the speed of light.

It can reach the Alpha Centauri in just 44 years.

The unprecedented extreme power requirements for doing so would be met by nuclear explosions, of such power relative to the vehicle's mass as to be survived only by using external detonations without attempting to contain them in internal structures.

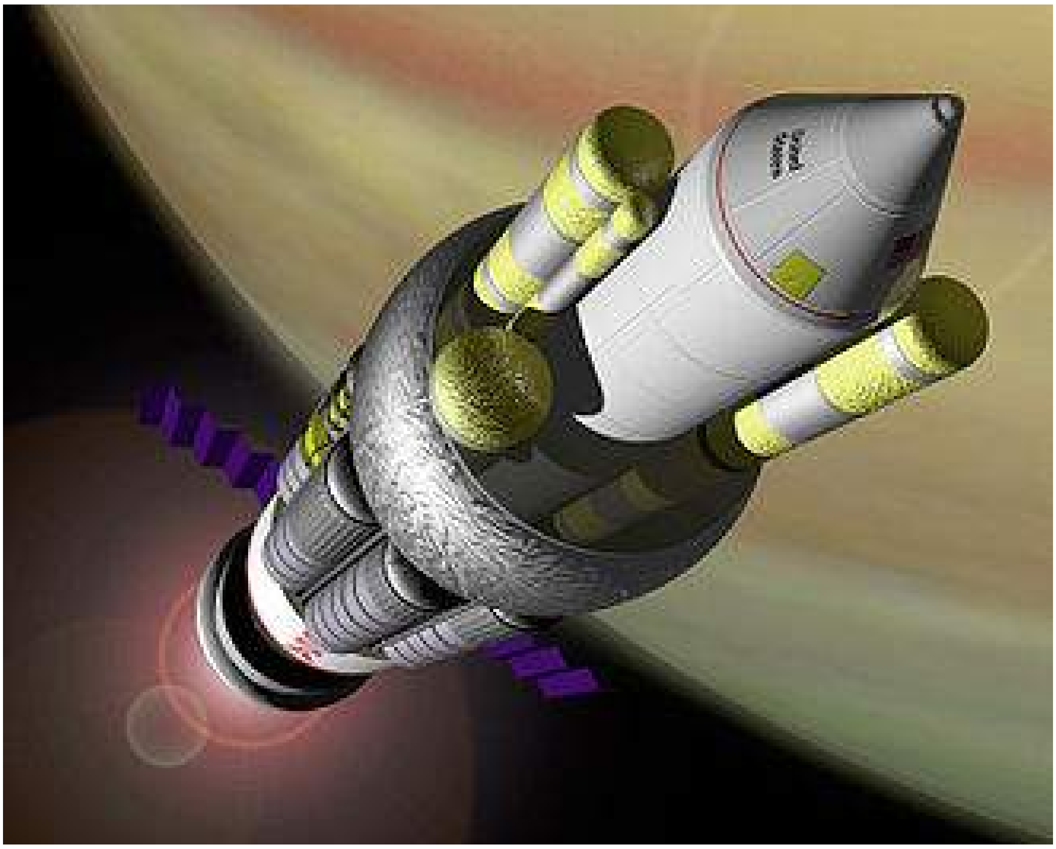


Figure 62: Orion

16.1.2 Fission Powered Rockets

A **nuclear thermal rocket (NTR)** is a type of thermal rocket where the heat from a nuclear reaction, often nuclear fission, replaces the chemical energy of the propellants in a chemical rocket.
In an NTR, a working fluid, usually liquid hydrogen, is heated to a high temperature in a nuclear reactor and then expands through a rocket nozzle to create thrust. The external nuclear heat source theoretically allows a higher effective exhaust velocity and is expected to double or triple payload capacity compared to chemical propellants that store energy internally.

NUCLEAR SALT WATER ROCKET (NSWR)

It is an incredibly efficient rocket with a high I_{sp} of around 10,000s. In place of traditional chemical propellant, this rocket would be fueled by salts of plutonium or 20 perce**enriched uranium**. The solution would be contained in a bundle of pipes coated in boron carbide (for its properties of neutron absorption).

Through a combination of the coating and space between the pipes, the contents would not reach critical mass until the solution is pumped into a reaction chamber, thus reaching a critical mass, and being expelled through a nozzle to generate thrust.

It has an exhaust velocity of about 66km/s.
Using this rocket, it is possible to accelerate a conventional spacecraft to 7% of the speed of light.

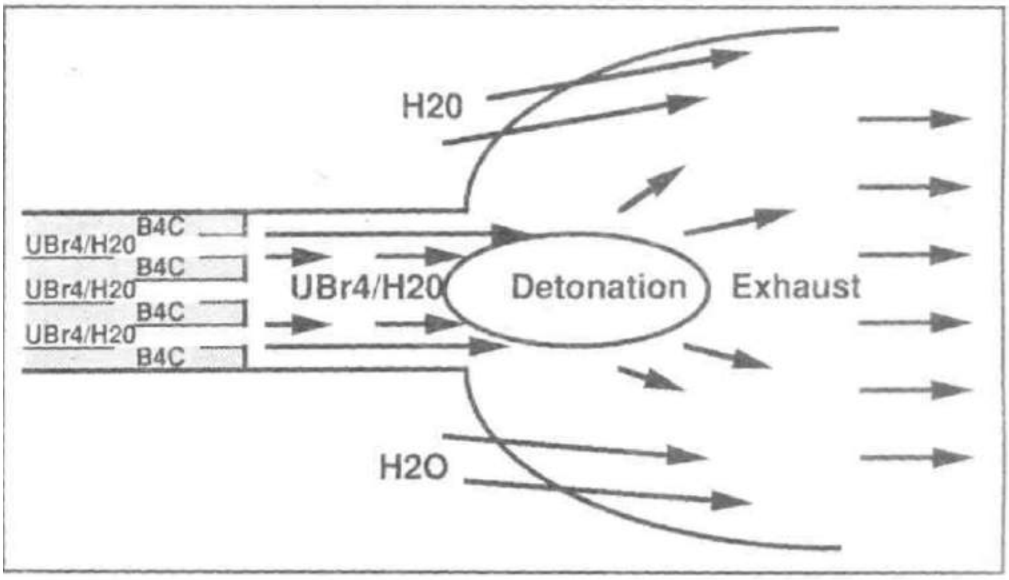


Fig. 1. The Nuclear Salt Water Rocket.

Figure 63: NSWR

NUCLEAR ENGINE FOR ROCKET VEHICLE APPLICATION (NERVA)

The Nuclear Engine for Rocket Vehicle Application (NERVA) was a nuclear thermal rocket engine development program that ran for roughly two decades. It is the only tested type of Nuclear Propulsion System.
In principle, the design of a nuclear thermal rocket engine is quite simple: a turbo pump would force hydrogen through a nuclear reactor that would heat it to very high temperatures.

For fuel, Uranium-235 was considered. To control the reactor, the core was surrounded by control drums coated with graphite or beryllium (a neutron moderator) on one side and boron (a neutron poison) on the other. It has the capacity to reach Mars in just 3 months at an acceleration of $0.5g$ along with a 220 ton spacecraft.

However, this project was eventually canceled due to some gruesome political issues.

Figure 64: NERVA

16.2 Far Future Methods

16.2.1 Fusion Powered engines

A fusion rocket is a theoretical design for a rocket driven by fusion propulsion that could provide efficient and sustained acceleration in space without the need to carry a large fuel supply. Fusion's main advantage is its very high specific impulse, while its main disadvantage is the (likely) large mass of the reactor. A fusion rocket may produce less radiation than a fission rocket, reducing the shielding mass needed.

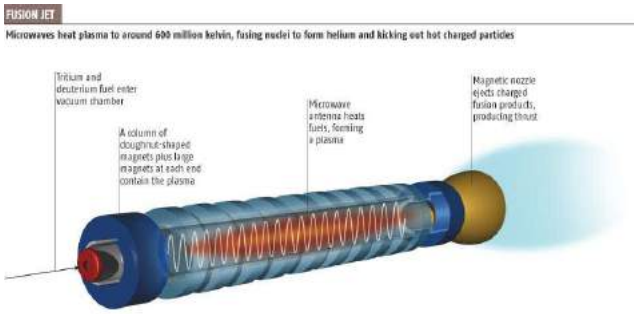


Figure 65: Fusion Powered Engines

They are broadly of two types:-

a. Magnetic Confinement Fusion

To sustain a fusion reaction, the plasma must be confined. The most widely studied configuration for terrestrial fusion is the tokamak, a form of magnetic confinement fusion.

Currently tokamaks weigh a great deal, so the thrust to weight ratio would seem unacceptable, but still as a method to be used in future, one can consider it.

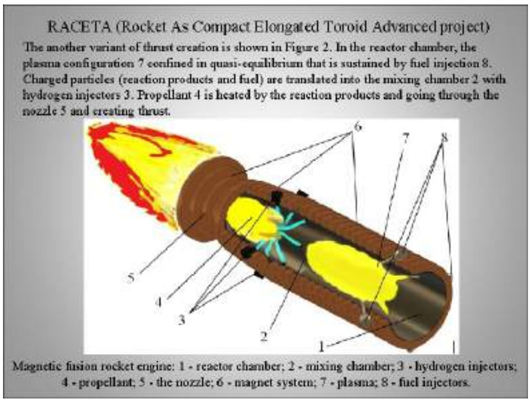


Figure 66: Magnetic Confinement Fusion Rockets

NASA's Glenn Research Center proposed a small aspect ratio spherical torus reactor for its "Discovery II" conceptual vehicle design. "Discovery II" could deliver a crewed 172 000-kilogram payload to Jupiter in 118 days (or 212 days to Saturn) using 861 metric tons of hydrogen propellant, plus 11 metric tons of Helium-3-Deuterium fusion fuel. The hydrogen is heated by the fusion plasma debris to increase thrust, at a cost of reduced exhaust velocity (348463 km/s) and hence increased propellant mass.

b. Inertial Confinement Fusion

Project Daedalus.

A small pellet of fusion fuel (with a diameter of a couple of millimeters) would be ignited by an electron beam or a laser. To produce direct thrust, a magnetic field forms the pusher plate. In principle, the Helium-3-Deuterium reaction or an aneutronic fusion reaction could be used to maximize the energy in charged particles and to minimize radiation, but it is highly questionable whether using these reactions is technically feasible.

Both the detailed design studies in the 1970s, the Orion drive and Project Daedalus, used inertial confinement. In the 1980s, Lawrence Livermore National Laboratory and NASA studied an ICF-powered "Vehicle for Interplanetary Transport Applications" (VISTA). The conical VISTA spacecraft could deliver a 100-tonne payload to Mars orbit and return to Earth in 130 days, or to Jupiter orbit and back in 403 days. 41 tonnes of deuterium/tritium (D-T) fusion fuel would be required, plus 4,124 tonnes of hydrogen expellant. [3] The exhaust velocity would be 157 km/s.

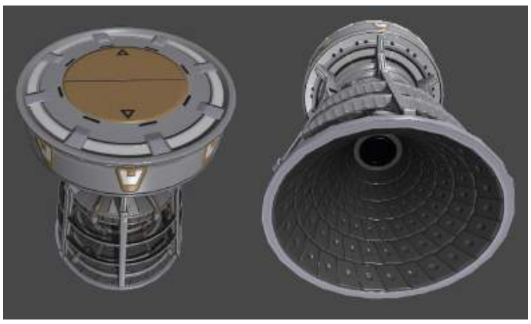


Figure 67: Inertial Confinement Fusion Rockets

16.2.2 Anti-Matter Powered Rocket Engine

An antimatter rocket is a proposed class of rockets that use antimatter as their power source. There are several designs that attempt to accomplish this goal. The advantage to this class of rocket is that a large fraction of the rest mass of a matter/antimatter mixture may be converted to energy, allowing antimatter rockets to have a far higher energy density and specific impulse than any other proposed class of rocket.

Anti-Matter rockets can be divided into 3 types:-

a. Anti-matter Drive

Anti-Matter Drive utilizes antimatter to kick start a fission reaction which subsequently induces a fusion reaction. Antimatter is only required in small quantities; a maximum of 100 g for intra-system travel up to Pluto. In addition, minimal fission is required to start the fusion reaction which reduces radioactive waste, and fusion which is difficult to sustain only has to be maintained for a short time.

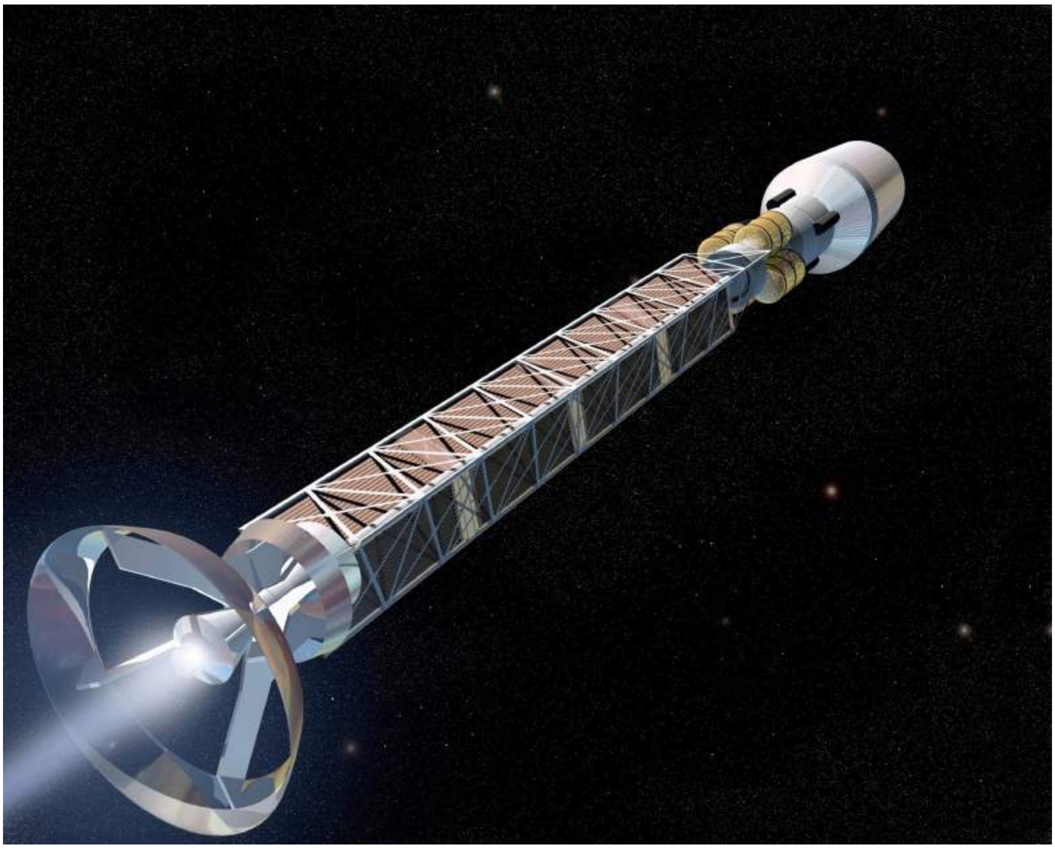


Figure 68: Anti-Matter Rocket

The propellant will come in the form of a pellet of deuterium, tritium and uranium-238 (9 parts D-T to 1 part U-238). It is firstly injected into a reaction chamber where it will undergo compression by ion particle beams. Subsequently, the propellant will be irradiated with a burst of antiprotons.

The antiprotons will annihilate some of the pellet, releasing sufficient energy to initiate fission of U-238. Following which,

Figure 69: Anti-Matter Drive

the fission reaction causes fusion to commence in the D-T core. The products of the entire process are in the form of radiation and hot plasma which is ejected to produce the thrust for the spacecraft.

b.Thermal Antimatter Rocket

This type of antimatter rocket is termed a thermal antimatter rocket as the energy or heat from the annihilation is harnessed to create an exhaust from non-exotic material or propellant.

The **solid core** concept uses antiprotons to heat a solid, high-atomic weight (Z), refractory metal core. Propellant is pumped into the hot core and expanded through a nozzle to generate thrust. The performance of this concept is roughly equivalent to that of the nuclear thermal rocket (I_{sp} 103 sec) due to temperature limitations of the solid.

Several methods for the **liquid-propellant** thermal antimatter engine using the gamma rays produced by antiproton or positron annihilation have been proposed. These methods resemble those proposed for nuclear thermal rockets. One proposed method is to use positron annihilation gamma rays to heat the solid engine core.

The **gaseous core** system substitutes the low-melting point solid with a high temperature gas (i.e. tungsten gas/plasma), thus permitting higher operational temperatures and performance (I_{sp} 2 Œ 10^3 sec). However, the longer mean free path for thermalization and absorption results in much lower energy conversion efficiencies.

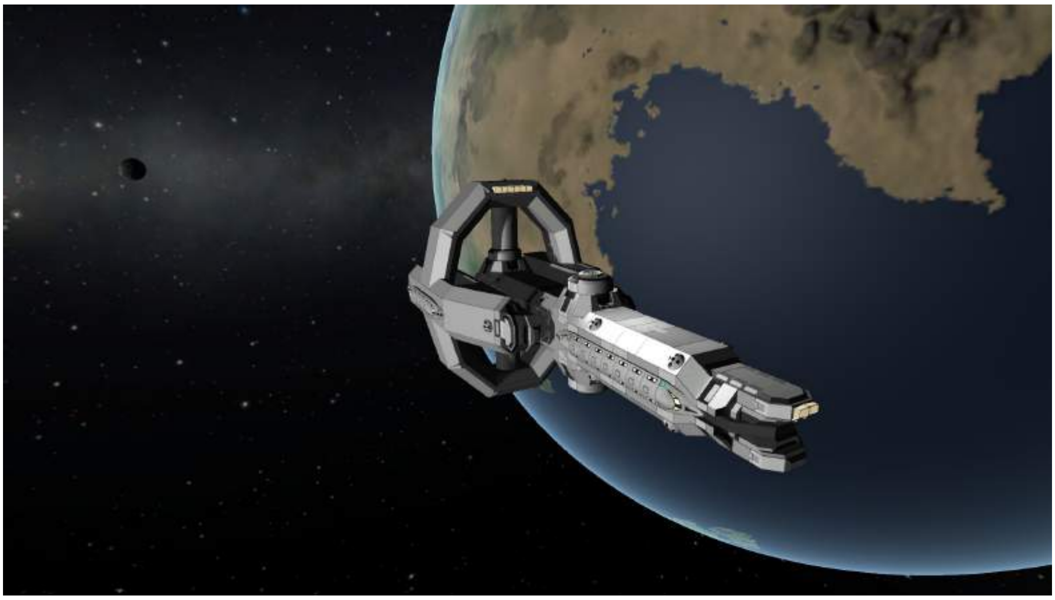


Figure 70: Thermal Anti-Matter Rocket

The **plasma core** allows the gas to ionize and operate at even higher effective temperatures. Heat loss is suppressed by magnetic confinement in the reaction chamber and nozzle. Although performance is extremely high (I_{sp} 104-105 sec), the long mean free path results in very low energy utilization.

c.Antimatter catalysed Fission/Fusion Engine

This is a hybrid approach in which antiprotons are used to catalyze a fission/-fusion reaction or to "spike" the propulsion of a fusion rocket or any similar applications.

The antiproton-driven Inertial confinement fusion (ICF) Rocket concept uses pellets for the D-T reaction.

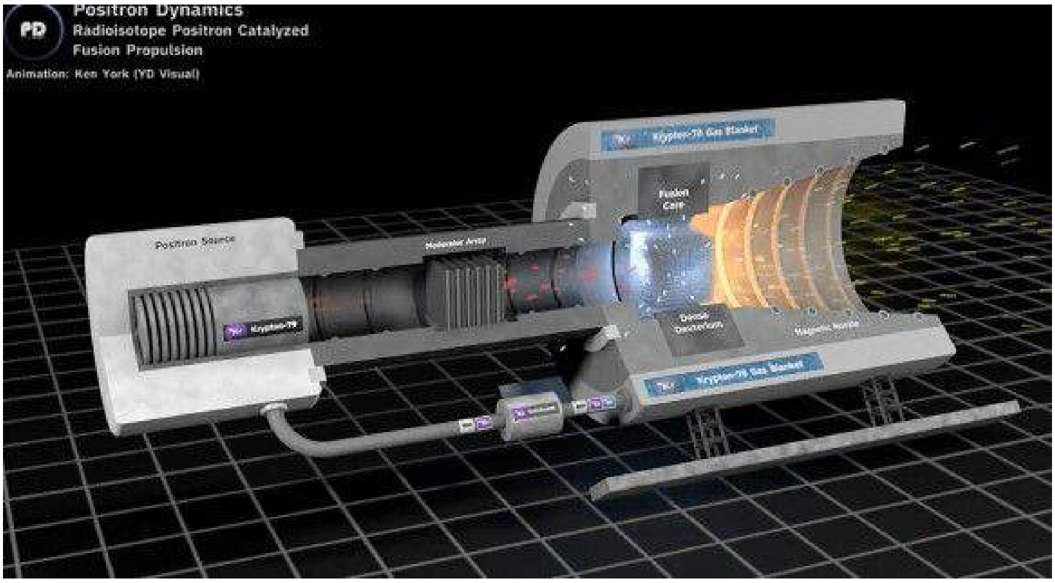


Figure 71: Anti-Matter Catalysed Fission/Fusion Rocket

The pellet consists of a hemisphere of fissionable material such as U235 with a hole through which a pulse of antiprotons and positrons is injected. It is surrounded by a hemisphere of fusion fuel, for example deuterium-tritium, or lithium deuteride. Antiproton annihilation occurs at the surface of the hemisphere, which ionizes the fuel. These ions heat the core of the pellet to fusion temperatures.

16.2.3 Solar Sails

Solar sails (also called light sails or photon sails) are a method of spacecraft propulsion using radiation pressure exerted by sunlight on large mirrors.

Solar sail craft offers the possibility of low-cost operations combined with long operating lifetimes. Since they have few moving parts and use no propellant, they can potentially be used numerous times for delivery of payloads.

Solar pressure affects all spacecraft, whether in interplanetary space or in orbit around a planet or small body. It also affects the orientation of a spacecraft.

The total force exerted on an 800 by 800 metre solar sail, for example, is about 5 newtons making it a low-thrust propulsion system. They require infrastructure and technology that is still beyond the reach of mankind.

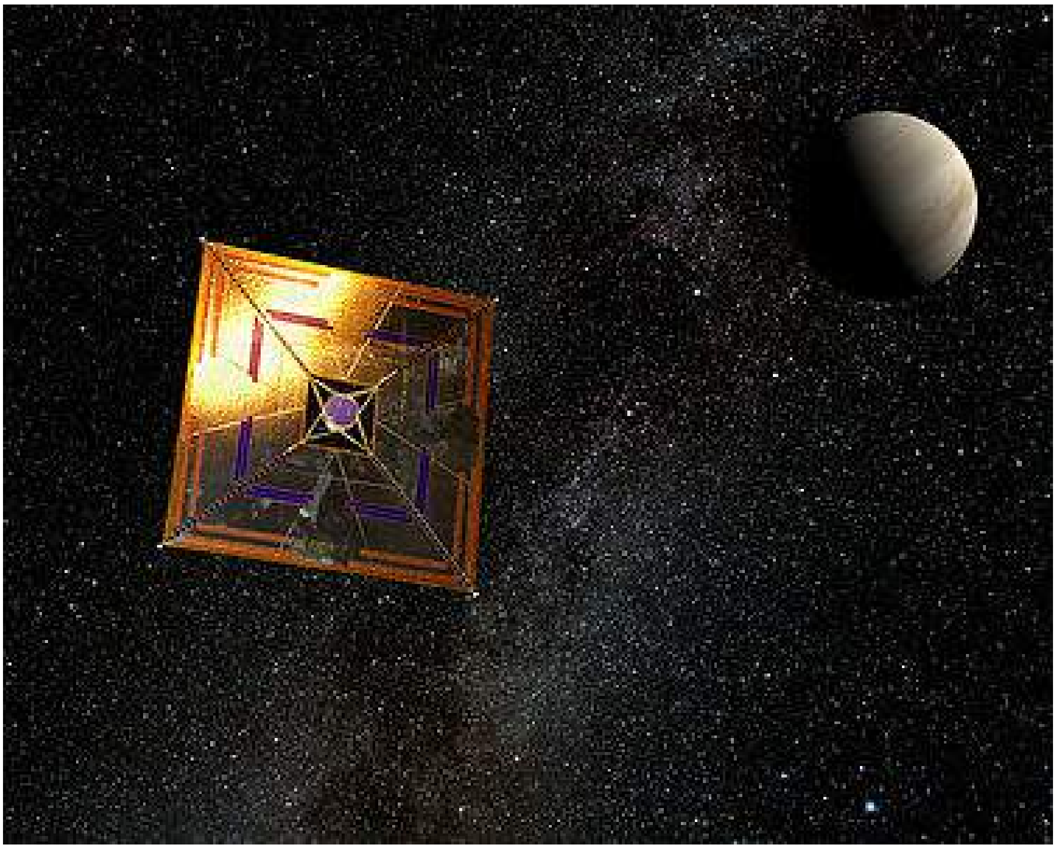


Figure 72: Solar Sails

16.2.4 Beam Powered Propulsion

Beam-powered propulsion, also known as directed energy propulsion, is a class of aircraft or spacecraft propulsion that uses energy beamed to the spacecraft from a remote power plant to provide energy. The beam is typically either a microwave or a laser beam and it is either pulsed or continuous.

A continuous beam lends itself to thermal rockets, photonic thrusters and light sails, whereas a pulsed beam lends itself to ablative thrusters and pulse detonation engines.

Since efficient laser technology has not arrived yet, this project is still near impossible. Also, it requires assistance from earth hence it is not feasible for interstellar space voyages.

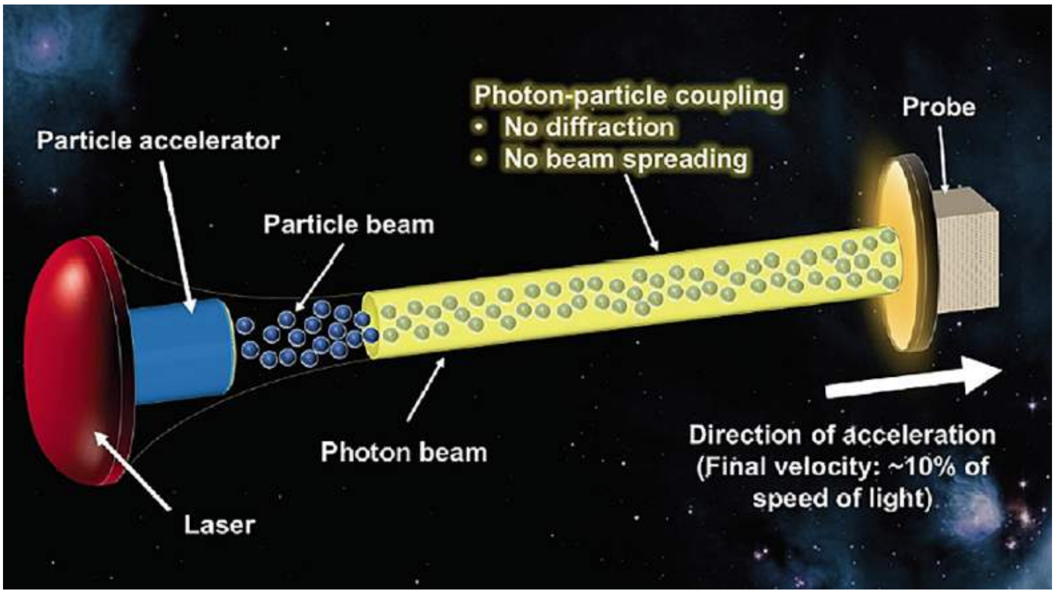


Figure 73: Beam Powered Propulsion System

16.3 Exotic Propulsion Methods

These methods discuss about some projects that are impossible with the current laws of physics available to us as these systems defy them. They are discussed yet none have been made/experimented successfully till the present day.

16.3.1 Alcubierre Drive

This is a speculative warp drive idea based on a solution of Einstein's field equations in general relativity as proposed by theoretical physicist Miguel Alcubierre by which a spacecraft could achieve apparent faster-than-light travel if a configurable energy-density field lower than that of the vacuum (that is, negative mass) could be created.

Rather than exceeding the speed of light within a local reference frame, a spacecraft would traverse distances by contracting space in front of it and expanding space behind it, resulting in effective faster-than-light travel. Objects cannot accelerate to the speed of light within normal spacetime; instead, the Alcubierre drive shifts space around an object so that the object would arrive at its destination more quickly than light would in normal space without breaking any physical laws.

However, the construction of such a drive is nearly impossible.

The proposed mechanism of the Alcubierre drive implies a negative energy density and therefore requires exotic matter or **manipulation of dark energy**. If exotic matter with the correct properties cannot exist, then the drive cannot be constructed. At the close of his original article, however, Alcubierre argued (following an argument developed by physicists analyzing traversable wormholes) that the Casimir vacuum between parallel plates could fulfill the negative-energy requirement for the Alcubierre drive. Still this model is an impossible task to achieve.

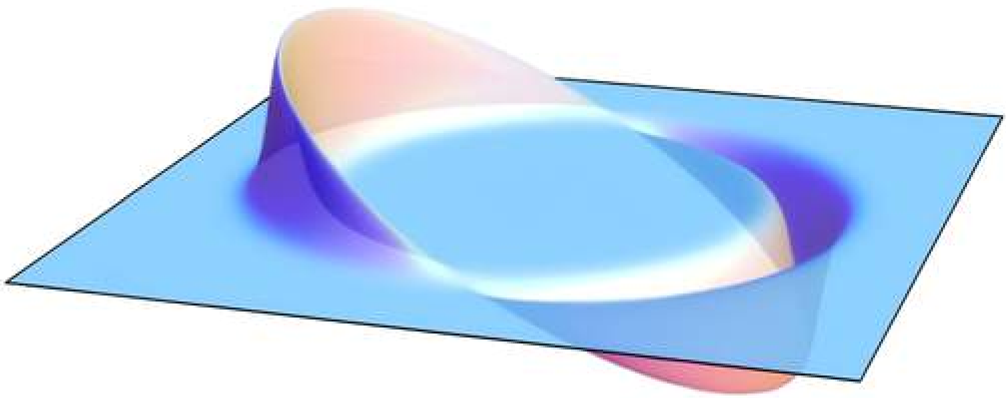


Figure 74: Alcubierre Drive Idea

16.3.2 EM drive

EMDrive is a concept for a radio frequency (RF) resonant cavity thruster that is claimed to have potential applications as a spacecraft thruster. It is purported to generate thrust by reflecting microwaves internally in the device, in violation of the law of conservation of momentum and other laws of physics. The device has been often referred to by the media as the Impossible Drive. It was introduced in 2001 by Roger Shawyer.

There exists no official design for this device, and neither of the people who claim to have invented it have committed to an explanation for how it could operate as a thruster or what elements define it, making it difficult to tell whether a given object is an example of such a device. However, several prototypes based on its public descriptions have been constructed and tested. In 2016, the Advanced Propulsion Physics Laboratory at NASA reported observing a small apparent thrust from one such test, a result not since replicated, and subsequent studies have indicated that the thrust observed was measurement error caused by interactions with the Earth's magnetic field or by thermal gradients.

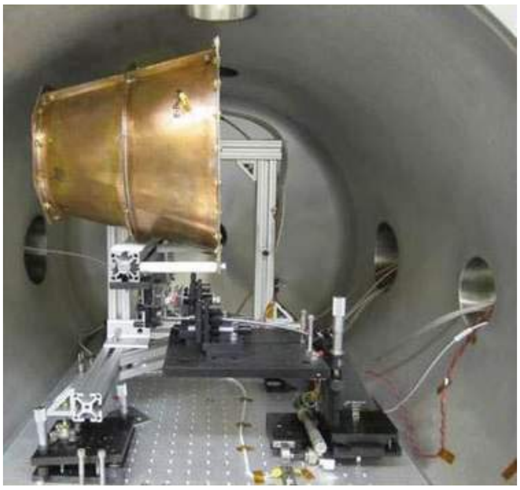


Figure 75: EM Drive Prototype

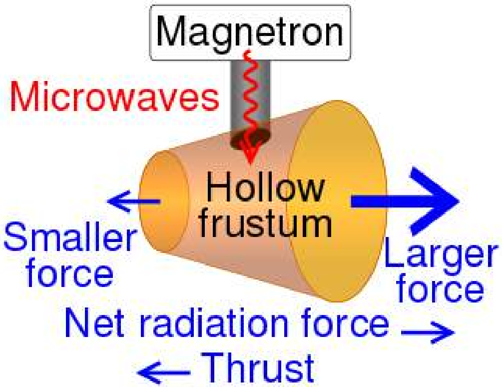


Figure 76: Idea for EM Drive

17

Sample Mission Reports

The mentees were divided into groups in which they prepared **Sample Mission Reports** of a previously successful or an upcoming space mission. All the SMRs have been included below...

ARTEMIS

ASTRO IIT KANPUR

Humanity's Return to the Moon

Summer Project
Space Mission Report

Space Mission: Artemis Program

Group Mentor

Varun Singh

Mentees

Aditya Kushwaha

Atharva Dehadraya

Ganesh Kumar

Nishithaarsh

Ojsi Goel

Sarvagya Jain

July, 2021

Contents

1. Why ARTEMIS

1.1 Introduction

It has been almost 50 years since astronauts last walked on the lunar surface during the Apollo Program. Since then, the robotic exploration of deep space has seen decades of technological advancement and scientific discoveries. For the last 20 years, humans have continuously lived and worked aboard the International Space Station 250 miles above Earth, preparing for the day we move farther into the solar system. NASA's Artemis Program, built on a half-century of experience and preparation to establish a robust human-robotic presence on and around the Moon, will lead humanity forward and prepare us for the next giant leap, the exploration of Mars.

1.2 Mission Aim/Vision

The Moon plan is twofold: it's focused on achieving the goal of an initial human landing by 2024 with acceptable technical risks while simultaneously working toward sustainable lunar exploration in the mid-to-late 2020s. With the confidence and experience gained through moon missions and explorations, will seed our way to the exploration of our nearest neighbour, Mars. NASA has the vision of making a sustainable lunar economy, in partnership with U.S. commercial companies and international partners, one where they benefit from and build on what we learn. NASA also established the Commercial Lunar Payload Services or CLPS initiative in 2018, encouraging the U.S. commercial space industry to introduce new lander technologies to deliver NASA and commercial payloads to the surface of the Moon. It has already selected more than two dozen instruments to study the Moon and test new technologies for these early CLPS flights, including VIPER. It also aims to land the first woman on the Moon by 2024, in the process achieving many other firsts.

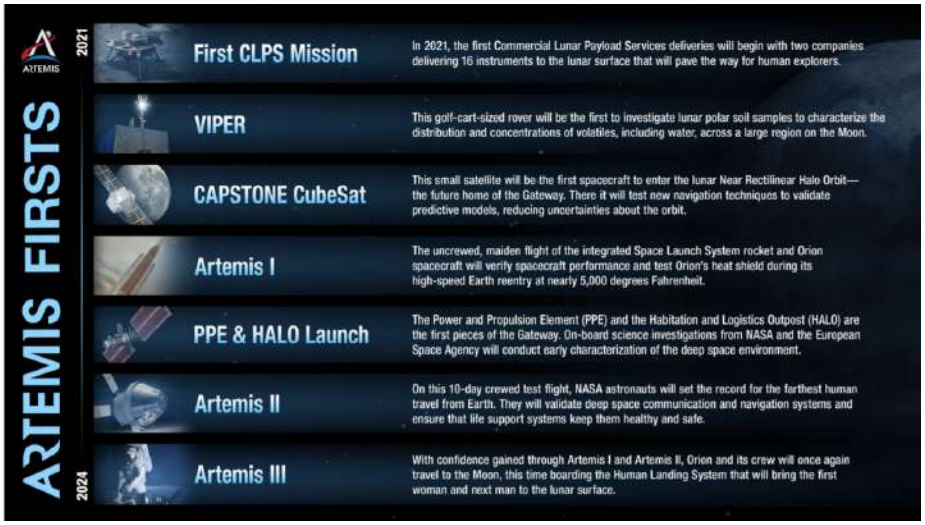


Figure 1.1: Firsts aimed for Artemis

2. Launch Vehicle: SLS

In order to design the rocket for our mission we must keep these checkpoints- know what your rocket needs to do, establish mission parameters, call in experts, start drawing, whittle down the possibilities and pick the best design. For the Artemis Mission Program, the SLS (Space Launching System), is the launch vehicle used to launch the Artemis I and henceforth Artemis flights.

Basic description of SLS Rocket as follows:

SLS is the most powerful rocket ever built, the backbone for a permanent human presence in deep space. It offers more payload mass, volume capability, and energy to speed missions through space than any current launch vehicle. SLS is designed to be flexible and evolvable and will open new possibilities for payloads, including robotic scientific missions to places like the Moon, Mars, Saturn, and Jupiter. In order to make the SLS spacecraft as light as possible, it is constructed with lightweight, strong materials, such as aluminum alloys and composites. It has a ABORT vehicle system which can deploy easily even faster than its speed when there is an error or critical circumstances which basically provides a proper safety to our astronomers. The engineering design challenge focuses on the thrust structure, which attaches the four liquid fuel engines to the body of the rocket. The thrust structure is an essential part of the spacecraft, which must be kept lightweight.

SLS Rocket

2.1 Basic components of SLS rocket

- **ICPS:** The Interim Cryogenic Propulsion Stage (ICPS) for SLS Block 1 is the initial configuration that can deliver 27 metric tons of payload to the moon. Based on the proven Delta Cryogenic Second Stage and powered by one Aerojet Rocketdyne RL10 engine, ICPS will propel an uncrewed Orion spacecraft to fly beyond the moon and back on the Artemis I mission.

ICPS

- **LVSA:** The Launch Vehicle Stage Adapter (LVSA) connects the Block 1 core stage to the upper stage while providing structural, electrical and communication paths. It separates the core stage from the second stage that includes astronauts in the Orion crew vehicle.

LVSA

- **Forward Skirt:** It houses flight computers, cameras and avionics — the routers, processors, power, other boxes and software that control stage functions and communications. Along with the liquid oxygen tank and the intertank, it makes up the top half of the core stage.

Forward Skirt

- **LOX tank:** The liquid oxygen (LOX) tank holds 196,000 gallons (742,000 liters) of liquid oxygen cooled to minus 297 degrees Fahrenheit. Its thermal foam coating protects it from extreme temperatures — the cold of the propellants and the heat of friction.


LOX Tank

- **LH2 Tank:** The liquid hydrogen (LH2) tank comprises two-thirds of the core stage, weighs 150,000 pounds (68,000 kilograms) and holds 537,000 gallons (2 million liters) of liquid hydrogen cooled to minus 423 degrees Fahrenheit.


LH2 Tank

- **Solid Rocket Boosters:** The largest human-rated solid rocket boosters ever built for flight, the SLS twin boosters stand 17 stories tall and burn about six tons of propellant every second. Each booster generates more thrust than 14 four-engine jumbo commercial airliners. Together, the SLS twin boosters provide more than 75% of the total thrust at launch.

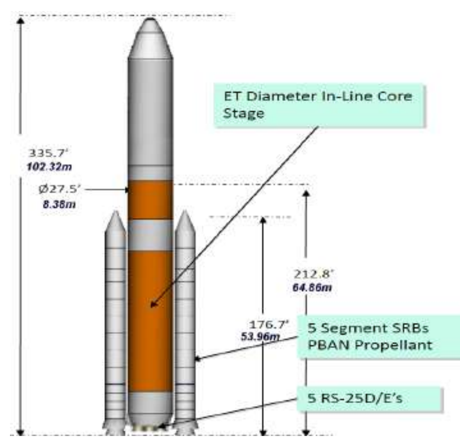

Solid Rocket Boosters

- **Engine Section:** The engine section is a crucial attachment point for the four RS-25 engines that work with two solid rocket boosters to produce a combined 8.8 million pounds of thrust at liftoff. Four RS-25 engines will deliver more than 2 million pounds of thrust at altitude. Combined with two five-segment solid rocket boosters, the propulsion system will give SLS about 8.8 million pounds of thrust at launch — more lift than any current rocket and 15% more than the Saturn V.


RS-25 Engine

2.2 Dimension and Properties

The design is 212 Feet tall, 27.6 Feet in diameter and has 2.3 pounds with propellant, rated safe for humans. It reaches MACH 23, faster than 17000 mph in just 8.5 minutes. The model initially in 2017-21 was capable of 70 metric tons; serves as primary transportation for Orion and exploration missions and also provides back-up capability for crew/cargo to ISS. Post 2021, the model can be evolved to be capable of 105 tons and 130 tons, thus offering large volume for science missions and payloads; it is modular and flexible, just right-sized for mission requirements. With maximum use of common elements and existing assets, infrastructure and workforce, make the model affordable.



Dimensions marked for SLS Rocket

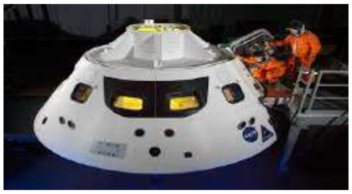
3. SpaceCraft

3.1 ORION Capsule

Compilation of Orion Spacecraft

Orion (officially **Orion Multi-Purpose Crew Vehicle** or **Orion MPCV**) is a class of partially reusable space capsules to be used in NASA's human spaceflight programs.Capable of supporting a crew of six beyond low Earth orbit, Orion can last up to 21 days undocked and up to six months docked. Orion is primarily designed to launch atop a Space Launch System (SLS) rocket, with a tower launch escape system. The spacecraft consists of a crew module, service module and a launch abort system.

3.1.1 Crew Module



Orion Crew Module

Orion Crew Module, manufactured by Lockheed Martin Corporation, is a reusable transportation capsule that provides a habitat for the crew, provides storage for consumables and research instruments and contains the docking port for crew transfers. Orion's CM uses the latest technologies, including glass cockpit, autodock feature, improved waste-management facilities, a nitrogen/oxygen atmosphere at sea level and reduced pressures, etc.

3.1.2 Service Module

Orion Service Module

European Service Module, ESM based on ESA's Automated Transfer Vehicle (ATV), is the service module component of Orion Spacecraft, serving as its primary power and propulsion component until discarded at the end of each mission. Manufactured by Airbus Defence and Space (in Bremen), there are two **spacecraft adapters**, connecting the service module to the crew module and to the upper stage of the SLS, are ultimately discarded during staging; the three fairing panels are jettisoned after protecting the service module during launch and ascent.

3.1.3 Launch Abort System

The Launch Abort System, or LAS, is positioned atop the Orion crew module,designed to protect astronauts if a problem arises during launch by pulling the spacecraft away from a failing rocket. Weighing approximately 16,000 pounds, the LAS can activate within milliseconds to pull the vehicle to safety and position the module for a safe landing. It consists of the following parts :

- **Jettison Motor**:The jettison motor will pull the crew module,allowing Orion's Parachutes to deploy and the spacecraft to safely land on the ocean.
- **Altitude Control Motor**:The Motor can exert up to 7000 pounds of steering force in any direction upon command from the Orion crew module.
- **ABORT Motor**:Capable of producing about 400000 pounds of thrust to quickly pull the crew module away from danger if problems develop on the launch pad or during the ascent.
- **Fairing Assembly**:It is a lightweight composite structure that protects the capsule from the environment around it-whether it's heat,wind or acoustics.

With the incorporation of an abort system in the module, the programmatic cost is reduced and the crew safety is ensured in support of NASA's explorations.

Orion Crew Module

3.2 Gateway

The Gateway, a vital component of NASA's Artemis program, will serve as a multi-purpose outpost orbiting the Moon that provides essential support for sustainable, long-term human return to the lunar surface and serve as a staging point for deep space exploration. The Gateway is planned to be deployed in a highly elliptical seven-day near-rectilinear halo orbit (NRHO) around the Moon. NASA is working with commercial and international partners like SpaceX to establish the gateway orbitor. NASA has selected SpaceX's Falcon Heavy to deliver the first two segments of the moon-orbiter gateway space station of its Artemis program somewhere in 2024. The pair of modules SpaceX will ferry into space are the power and propulsion element (PPE) and the habitation and logistics outpost (HALO).The PPE will provide the Gateway with power, enabling communications as well as helping the station move to various lunar orbits, while HALO will give astronauts a place to stay on their way to the moon and HALO will also provide docking support for space vehicles. Once deposited in lunar orbit, the Gateway will serve as an outpost for astronauts and equipment heading to the moon as part of NASA's Artemis program. Roughly one-sixth the size of the International Space Station, the Gateway will support research investigations, crew, and expeditions to the lunar surface. The outpost will serve as a docking station for visiting spacecraft, such as NASA's Orion spacecraft and will orbit the moon, tens of thousands of miles away.

Figure 3.1: Gateway Demonstration

Figure 3.2: Dragon XL Spacecraft

3.3 Dragon XL

The Dragon XL resupply spacecraft has been designed to carry pressurized and unpressurized cargo, experiments and other supplies to NASA's planned Gateway under a NASA Gateway Logistics Services (GLS) contract. The equipment delivered by Dragon XL missions could include sample collection materials, spacesuits and other items astronauts may need on the Gateway and the surface of the Moon, according to NASA. Its payload capacity is expected to be more than 5,000 kg (11,000 lb) to lunar orbit.It will launch on SpaceX's Falcon Heavy launch vehicle from pad LC-39A at the Kennedy Space Center in Florida.

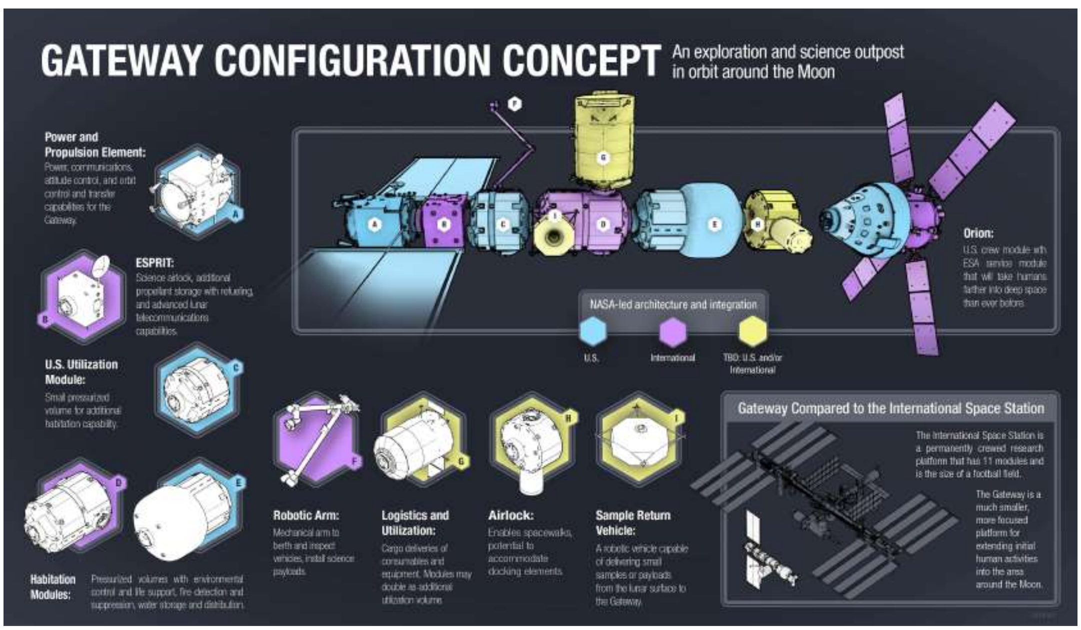


Figure 3.3: Gateway Configuration

4. Lander

4.1 Human Landing System (HLS)

Starship HLS, a lunar lander variant of the Starship spacecraft being developed by SpaceX under a contract with NASA; intends to dock in lunar orbit with either the NASA Orion spacecraft or NASA lunar Gateway space station, in order to take on passengers before descending to the lunar surface. The design of the HLS being optimized to operate exclusively in the vacuum of space, has no heat shield nor airbrakes, both of which are an integral component of the main Starship design. Further equipped with a complement of the high-thrust oxygen- and methane-fueled thrusters located mid-body on the lander, to assist in the lunar descent and liftoff from the lunar surface. The HLS is supplied with electrical power by a band of solar panels around the circumference of the vehicle. It would be launched using the Super Heavy booster and then serve as its own second stage to complete the ascent to low-Earth orbit (LEO). On orbit, it would be refueled before climbing out to lunar orbit to meet the Gateway and Orion crew capsule. It has been chosen to transport two NASA Artemis 3 astronauts along with cargo to and from the lunar surface, each time it lands on the moon.

4.2 HERACLES

HERACLES (Human-Enhanced Robotic Architecture and Capability for Lunar Exploration and Science) is a planned robotic transport system to and from the moon by Europe (ESA), Japan (JAXA) and Canada (CSA) that will feature a lander called the European Large Logistic Lander (or EL3), further constituted by Lunar Descent Element (LDE), which will be provided by Japan's JAXA,the ESA-built Interface Element that will house the rover, and the European Lunar Ascent Element (LAE) that will return the samples to the Lunar Gateway. The lander can be configured for different operations such as up to 1.5 tons of cargo delivery, sample-returns, or prospecting resources found on the Moon. The system is planned to support the Artemis program and perform lunar exploration using the Lunar Gateway space station as a staging point.

Figure 4.1: Starship HLS by SpaceX

Figure 4.2: HERACLES(Spacecraft)

5. Staging

Every payload needs an extra kick of thrust to overcome the weight of the rocket and payload to get it into space. It's a tricky balancing act. It's not just the payload that has mass, there's also a need to cancel out with thrust against the downward force of gravity. The rocket's body has mass as well, as does the fuel on board. The payload is typically the smallest portion of mass on a launch. The propellants — the fuel and oxidizer — weigh the most. Staging is a way of getting rid of dead weight so the energy of the burning engine is transferred to the payload so it gets into orbit. SLS rocket, used for launch in the Artemis program has been doubly staged with a Core Stage and Upper Stage.

5.1 Core Stage

SLS core stage is the world's tallest and most powerful rocket stage. Towering 212 feet with a diameter of 27.6 feet, it stores cryogenic liquid hydrogen and liquid oxygen and all the systems that will feed the stage's four RS-25 engines. The core stage is designed to operate for approximately 500 seconds, reaching nearly Mach 23 and more than 530,000 feet in altitude before it separates from the upper stage and Orion spacecraft.

5.2 Upper Stage

The main components of the stage are ICPS and Orion Spacecraft. The Exploration Upper Stage (EUS) is being developed as a large second stage for Block 1B of the Space Launch System(SLS), succeeding Block 1's Interim Cryogenic Propulsion Stage(ICPS). The EUS is to complete the SLS's ascent phase and then re-ignite to send its payload to destinations beyond low Earth orbit. This is a similar function to the S-IVB stage of the old Saturn V rocket. The design of the EUS, allowing Boeing to proceed with development of the stage, including hardware fabrication.

Figure 5.1: Demonstartion of seperation of stages. For more details on its parts, refer Figure 7.4

6. Aerodynamics

Aerodynamic support for the Artemis (SLS) requires the use of both wind-tunnel tests and computational simulations to develop aerodynamic databases across the flight mission profile, seen in Fig. below. These data are generated for a range of flight regimes including launch, liftoff, ascent, and booster separation. Flight conditions for the SLS vary from low-speed conditions on or near the launchpad to supersonic speeds during ascent. Because of this wide range of flight conditions, numerous tools are required to accurately capture the properties of the complex flowfields that evolve over time. In order to accurately capture the massively-separated flowfields that arise during launch of the vehicles, an unsteady CFD(Computational Fluid Dynamics) solver is utilized. These unsteady IDDES (improved delayed detached eddy simulation) CFD calculations yield a wealth of information that can be interrogated to provide visualizations of the evolving flowfield. As an example, a solution animation in which isosurfaces of constant **Q criterion** are colored by the magnitude of vorticity is shown in figures below:

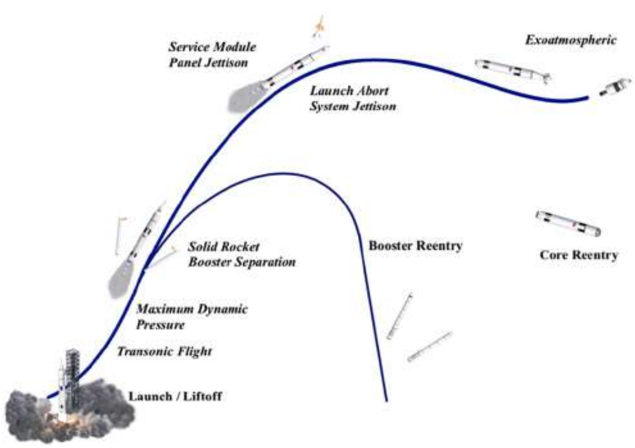


Figure 6.1: Aerodynamic Databases.

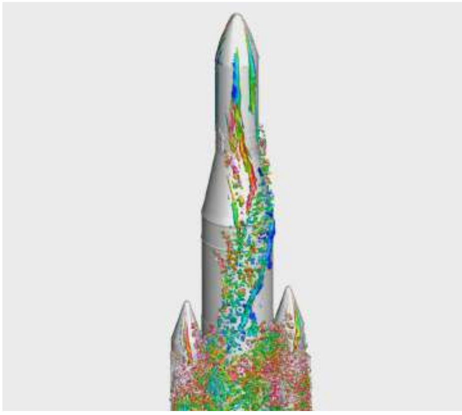


Figure 6.2: Isosurfaces of constant Q criterion.

Ascent Aerodynamics Run Matrix: Mach 0.5 to 5.0

From roughly sea level to very high dynamic pressure to near vacuum simulate out to $\alpha = \pm 8°$, even though flight is mostly close to 0. 6.3 is the graph of dynamic pressure variation with respect to mach number and 6.4 is the graph of α with respect to β.

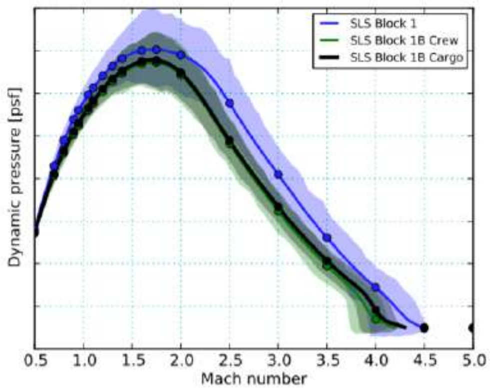


Figure 6.3: Dynamic Pressure Vs Mach Number.

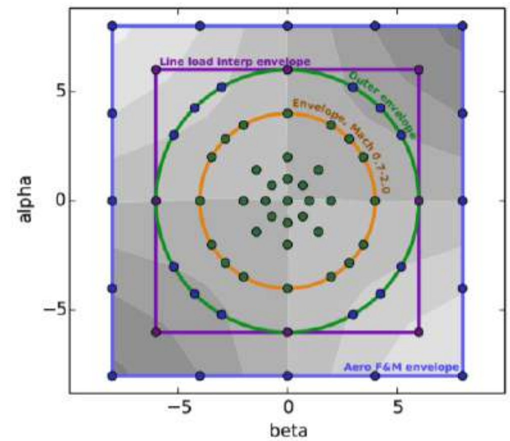


Figure 6.4: alpha Vs beta

Shock waves are formed when a pressure front moves at supersonic speeds(Speed more than that of sound) and pushes the air surrounding it. At that particular region, sound waves travelling against the flow reach a point where they are unable to travel any further upstream.6.5, 6.6 and 6.7 are the shock wave profiles at different Mach numbers ranging from 0.5 to 5.0. The proposed shock waves profiles are:

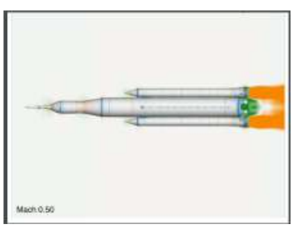


Figure 6.5: Shock Wave Profile at 0.5 mach.

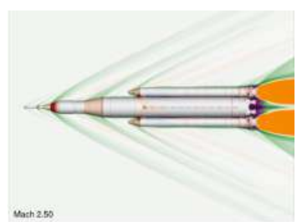


Figure 6.6: Shock Wave Profile at 2.0 mach

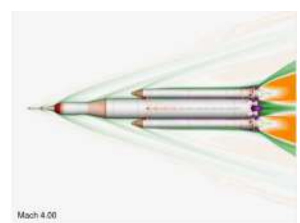


Figure 6.7: Shock Wave Profile at 4.0 mach

7. Engine

Preparations for Artemis I are in full swing. The production is already complete for the SLS engines-which is going to comprise four RS-25 liquid rocket engines(core stage) and RL10 engines(upper stage), two solid rocket boosters, a massive core stage, and an interim cryogenic propulsion stage that would provide Orion final push toward the Moon.

7.1 RS-25 Engine

The RS-25 engine has one of the most storied histories in spaceflight. As the space shuttle main engine, it has a proven record of launching 135 missions over 3 decades, including building the ISS and deploying the Hubble Space Telescope. Hence, the RS-25 engine offered an opportunity to forgo the costs of developing a new engine and bringing superb capabilities and experience to the table at the same time.

RS-25 burns cryogenic liquid hydrogen and liquid oxygen propellants, with each engine producing 1,859 kN (418,000 lbf) of thrust at liftoff. Thanks to this, during liftoff, the Block 1 configuration of SLS will produce 8.8 million pounds of thrust — 15% more than the Saturn V rockets that launched astronauts on journeys to the Moon.

Several Components of the RS-25 Engine are:

1. **Turbopumps:**
   (a) **Oxidizer System**:The low-pressure oxidizer turbopump (LPOTP) is an axial-flow pump that operates at approximately 5,150 rpm driven by a six-stage turbine powered by high-pressure liquid oxygen from the high-pressure oxidizer turbopump (HPOTP).
   (b) **Fuel System**: The low-pressure fuel turbopump (LPFTP) is an axial-flow pump driven by a two-stage turbine powered by gaseous hydrogen.
2. **Powerhead:**
   (a) **Pre Burners**:The oxidizer and fuel pre-burners are welded to the hot-gas manifold. The fuel and oxidizer enter the pre-burners and are mixed so that efficient combustion can occur.
   (b) **Main Combustion Chamber**: It uses Staged Combustion Mechanism. The engine's main combustion chamber (MCC) receives fuel-rich hot gas from a hot-gas manifold cooling circuit. The gaseous hydrogen and liquid oxygen enter the chamber at the injector, which mixes the propellants. The mixture is ignited by the "Augmented Spark Igniter", an H2/O2 flame at the centre of the injector head.
3. **Nozzle**: The nozzle is a bell-shaped extension bolted to the main combustion chamber, referred to as a de Laval nozzle. The RS-25 nozzle has an unusually large expansion ratio (about 69:1) for the chamber pressure.
4. **Controller**: Each engine is equipped with a main engine controller (MEC), an integrated computer that controls all of the engine's functions (through the use of valves) and monitors its performance.

5. **Main Valves**: To control the engine's output, the MEC operates five hydraulically actuated propellant valves on each engine; the oxidizer pre-burner oxidizer, fuel pre-burner oxidizer, main oxidizer, main fuel, and chamber coolant valves.

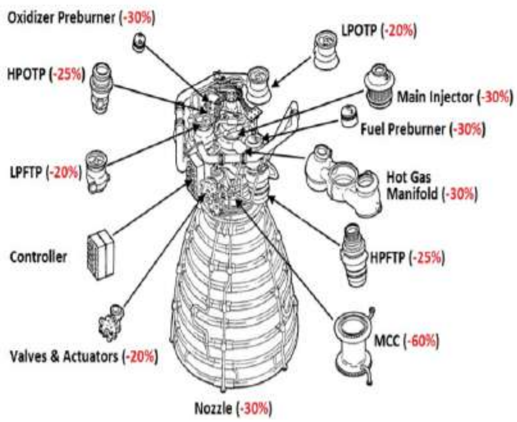


Figure 7.1: RS-25 components and cost reduction objectives

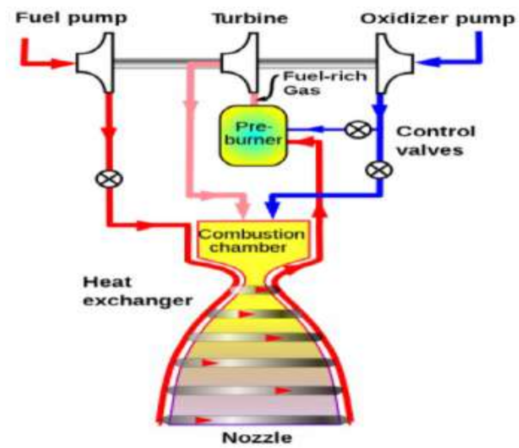


Figure 7.2: Staged Combustion

6. **Gimbal**: Each engine is installed with a gimbal bearing, a universal ball and socket joint which is bolted to the launch vehicle by its upper flange and to the engine by its lower flange.
7. **Helium System**: The launch vehicle's main propulsion system is also equipped with a helium system consisting of ten storage tanks. The system is used in-flight to purge the engine and provides pressure for actuating engine valves within the propellant management system and during emergency shutdowns.

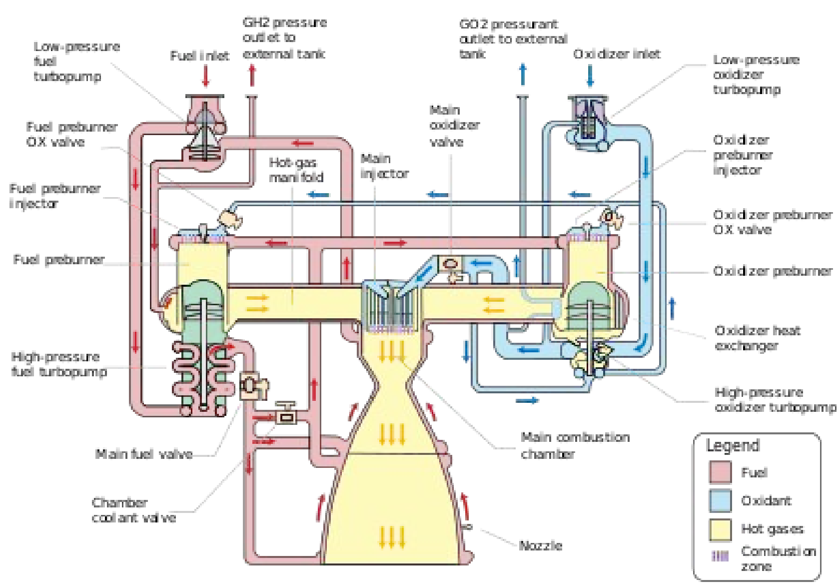


Figure 7.3: Working Mechanism of RS-25

7.2 RL 10 B-2 Engine

The RL10 B-2 is a liquid-fuel cryogenic rocket engine built in the United States by Aerojet Rocketdyne that burns cryogenic liquid hydrogen and liquid oxygen propellants. Modern versions produce up to 110 kN (24,729 lbf) of thrust per engine in vacuum. The expander cycle that the engine uses drives the turbopump with waste heat absorbed by the engine combustion chamber, throat, and nozzle.

RS-25 Engine Facts

Thrust	512,300 lbs. (vacuum) 416,300 lbs. (sea level)
Size	4.2m x 2.4m (14 ft x 8 ft)
Weight	3.49 t (7,800 lbs.)
Operational Thrust	109 percent
Operational Time	approximately 8 minutes
Operational Temp Range	-423 to +6000 degrees F

RL-10 B2 Engine Facts

Thrust	24,750 lbf (110.1 kN)
Height	4.14m (163.5 in)
Diameter	2.21m (84.5 in)
Weight	301.2kg (664 lb)
Specific Impulse	465.5 s (4565 km/s)
Expansion Ratio	280 to 1

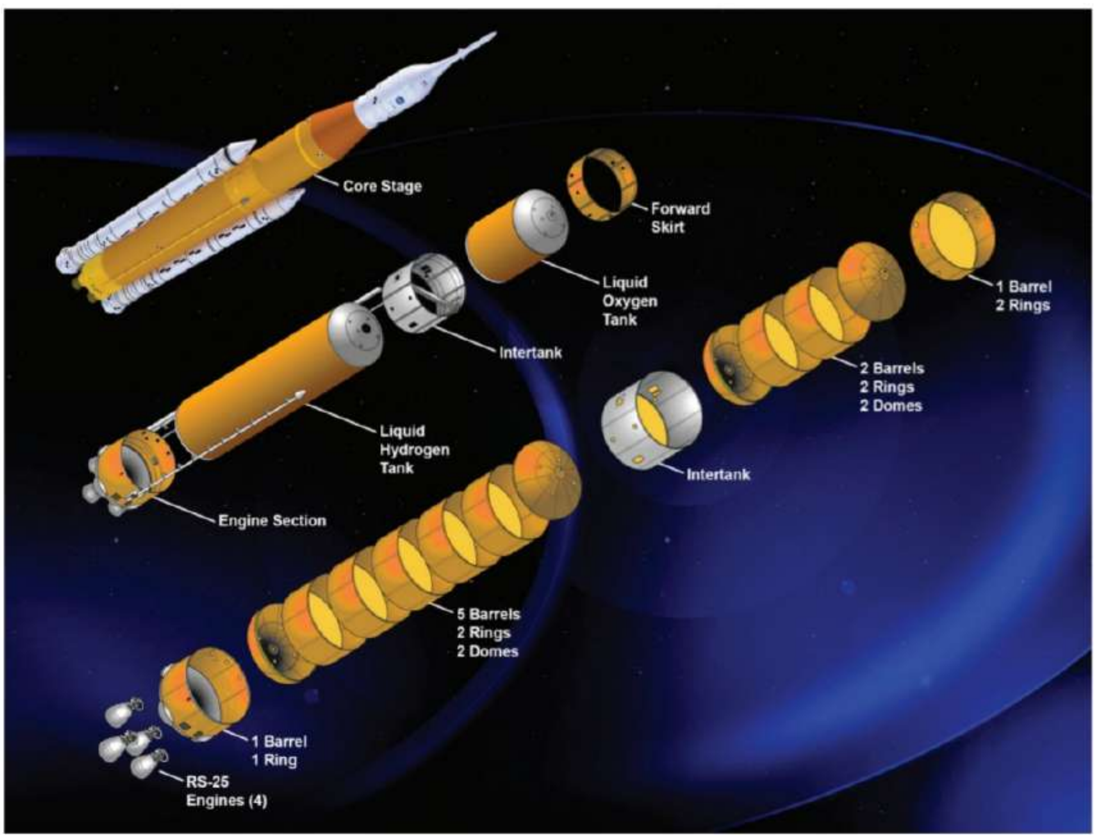


Figure 7.4: SLS Stages

8. Propellant

Rocket propellant is the reaction mass of a rocket. This reaction mass is ejected at the highest achievable velocity from a rocket engine to produce thrust. The energy required can either come from the propellants themselves, as with a chemical rocket, or from an external source, as with ion engines.The Reaction mass is stored in the Fuel Tanks of the rocket,which account for more than 70% volume of a rocket. In our mission we are dealing with both SOLID and LIQUID Propellants.

8.1 Solid Propellant

In **Boosters** solid propellant will be used - ammonium perchlorate composite propellant **(APCP)** (which uses a mixture of 70% granular ammonium perchlorate as an oxidiser, with **20%** aluminium powder as a fuel), bound together using 10% polybutadiene acrylonitrile **(PBAN)**. PBAN, Polybutadiene acrylonitrile copolymer, also noted as polybutadiene—acrylic acid—acrylonitrile terpolymer is a copolymer compound used most frequently as a rocket propellant fuel mixed with ammonium perchlorate oxidizer. APCP,Ammonium perchlorate composite propellant (APCP) is a modern fuel used in solid-propellant rocket vehicles It differs from many traditional solid rocket propellants such as black powder or **zinc-sulfur**. It is cast into shape, as opposed to powder pressing as with black powder.

8.2 Liquid Propellant

In **Core** stage **ISPC** and **Upper** Stage exploration Liquid Hydrogen and Liquid Oxygen will be used as propellants. The combination of **LOX and LH2** is mostly used for the upper stages that propel a vehicle into orbit. The lower density of the liquid hydrogen requires higher expansion ratios (gas pressure – atmospheric pressure) and therefore works more efficiently at higher altitudes. **Liquid hydrogen** fuelled rockets generally produce the lightest design and are therefore used on those parts of the spacecraft that actually need to be propelled into orbit or escape Earth's gravity to venture into deep space.

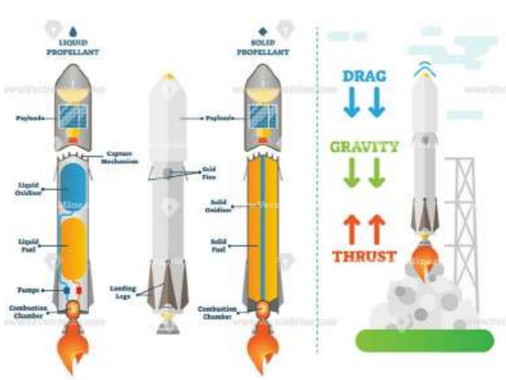


Figure 8.1: Demonstrating location of propellants in the rocket

9. Orbital Dynamics, Trajectory Optimisation

For Artemis I, an uncrewed Orion will fly a round-trip mission to a lunar Distant Retrograde Orbit(DRO) on a trajectory that includes two lunar flyby maneuvers, several burns and earth entry from Earth Entry Interface(EI) . The Artemis I mission is optimized with NASA's **Copernicus spacecraft trajectory design and optimization application** using the SNOPT optimization method. Copernicus makes use of a multiple-shooting approach, and the mission is designed using numerous coast and burn trajectory segments that are numerically integrated both backwards and forwards in time.In Copernicus, all mission phases are integrated explicitly using the DDEABM integration method (a variable-step size variable-order Adams-Bashforth-Moulton implementation) with a 10-12 error tolerance .For this trajectory design and optimization, only the CM and SM of Orion are included.The major burns in trajectory are : Upper Stage Separation (USS), Outbound Trajectory Correction (OTC)-1, Outbound Trajectory Correction (OTC)-2, Outbound Powered Flyby (OPF), DRO Insertion (DRI), DRO Plane Change (DPC), DRO Departure (DRD) and Return Powered Flyby (RPF), as demonstrated:

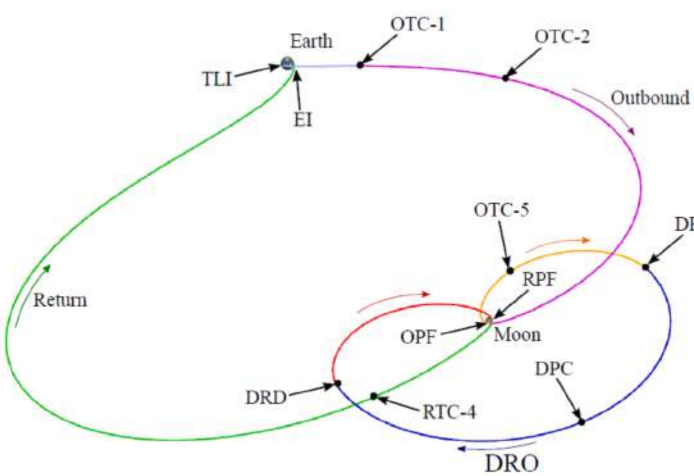


Figure 9.1: An overview of nominal mission trajectory

9.1 Tracing the Trajectory

In the following figures from 9.2 to 9.5, we trace the trajectory, demonstrating the orbit maneuvers and significant burns:

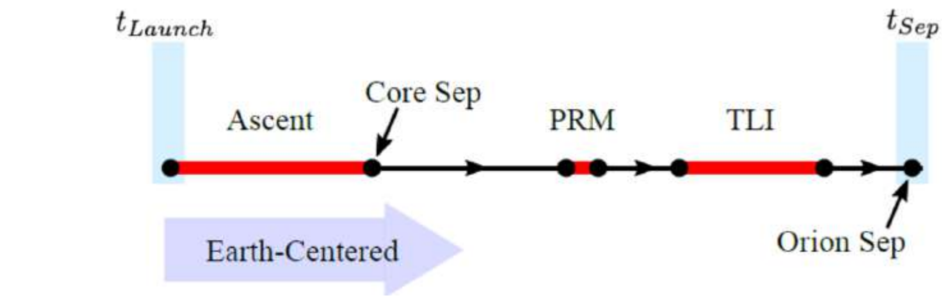


Figure 9.2: Launch to Orion Spring Separation. Ascent is performed by the SLS and the Perigee Raise Maneuver (PRM) and Trans Lunar Injection (TLI) are performed by the ICPS.

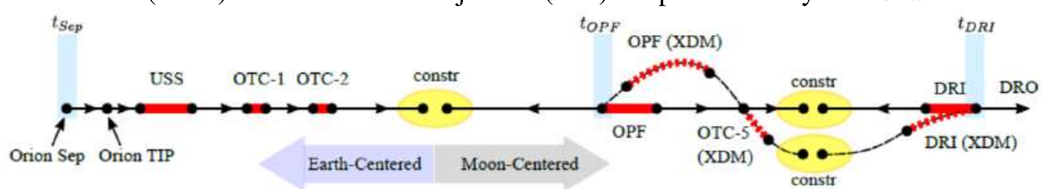


Figure 9.3: Orion Spring Separation to DRO Insertion. The outbound part of the mission includes two major burns: OPF and DRI.

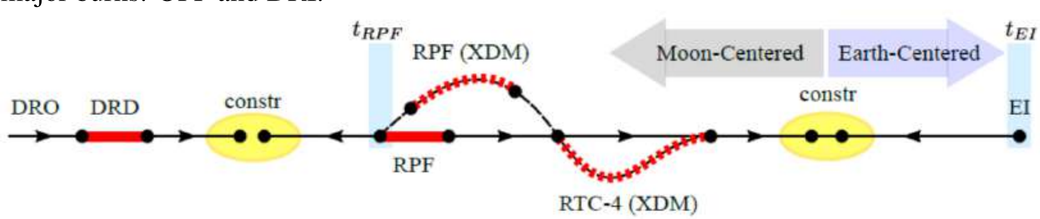


Figure 9.4: DRO Stay Time and Mission Duration for Different Mission Classes. Extended missions can be used to achieve desired landing lighting conditions not otherwise possible with the nominal mission of approximately 26 days.

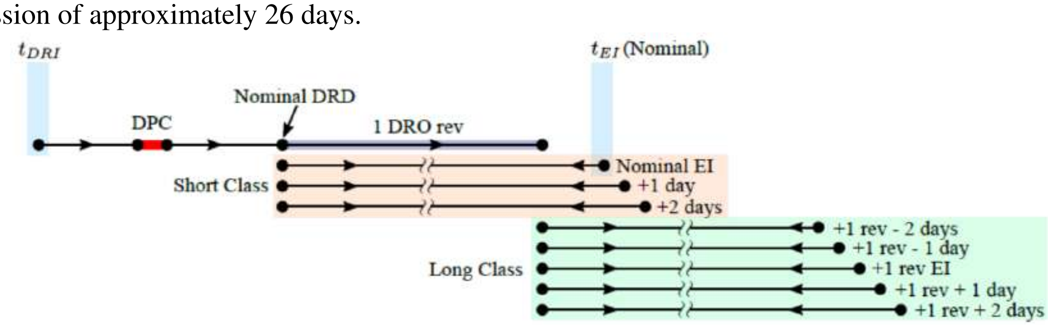


Figure 9.5: DRO Departure to Earth EI. The return part of the mission includes two major burns: DRD and RPF.

9.2 ORION on orbit trajectory

Orion in-space portion of the trajectory begins at the Space Launch System(SLS) core separation and ends at the Orion service module Earth Entry Interface(EI) point. In the above diagrams, the arrows indicate the direction of propagation of the mission phases in the Copernicus multiple-shooting transcription. The segments from the mission timeline integrated forward from one time(ex- tOPF) and backward from later time(ex-tDRI), with the required constraints imposed to make a continuous trajectory.

Once the ascent to Orion separation mission phase is completed, the objective function for the on-orbit trajectory comes out to be the combined ICPS(Interim Cryogenic Propulsion Stage) and Orion total (delta)v required, which is to be minimized.

The thrust directions for all burns are modeled by the spherical angles- α(right ascension) and β (declination), as functions of time:

$$\alpha(t) = \alpha_0 + \dot{\alpha}_0(t - t_0) \tag{9.1}$$
$$\beta(t) = \beta_0 + \dot{\beta}_0(t - t_0) \tag{9.2}$$

where t_0 is the burn start time, and the direction of thrust vector $\hat{u}$ at time t is given by:

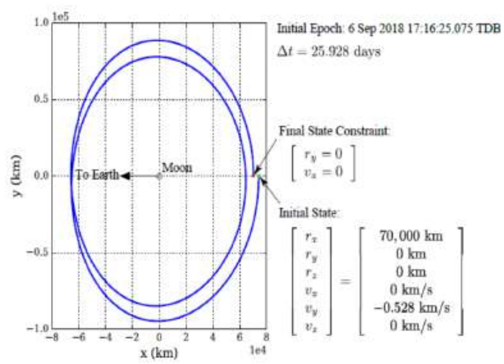
$$\hat{u}(t) = [\cos\alpha(t).\cos\beta(t)]\hat{e_1} + [\sin\alpha(t).\cos\beta(t)]\hat{e_2} + [\sin\beta(t)]\hat{e_3} \tag{9.3}$$

where the basis vectors $[\hat{e_1}\ \hat{e_2}\ \hat{e_3}]$ can be:
- the *IJK* frame($\hat{e_1}$ = **i** $\hat{e_2}$ = **j** $\hat{e_3}$ = **k**)
- the Copernicus *VUW* controls frame ($\hat{e_1}$ = **v/||v||** $\hat{e_3}$ = **h/||h||** $\hat{e_2}$ = $\hat{e_3} x \hat{e_1}$)
- the Copernicus *VNC* controls frame ($\hat{e_1}$ = **v/||v||** $\hat{e_2}$ = **h/||h||** $\hat{e_3}$ = $\hat{e_1}$x$\hat{e_2}$)

Note that these frames can be defined in either an inertial or rotating reference frame.

9.3 Launch Window Trajectories

For missions targeting a very specific orbit, the concept of launch window trajectories are used, which are programmed into launch vehicles prior to launch. The destination orbit for all Artemis missions is lunar DRO. The DRO is computed along with the nominal mission as part of the Orion on orbit trajectory optimization . The initial state of the DRO is propagated backwards in time to serve as the target for the DRO Insertion (DRI) burn. Note that the nominal DRO is planar (i.e., in the Earth-Moon plane), but during the eclipse mitigation process, it may be inclined.



Representation of Distant Retrograde Orbit(DRO)

9.3.1 Optimisation Variables

The major optimization variables are:
- The finite burn control law parameters α_0, β_0, for each of the optimized maneuvers.
- The burn time Δt for each of the optimized maneuvers.
- The various intermediate flight times.
- The launch epoch.
- The OPF and RPF flyby epochs.
- The OPF and RPF flyby state parameters (periapsis radius, eccentricity, inclination, ascending node, argument of periapsis, and true anomaly).
- The EI epoch.
- The EI longitude, latitude, velocity, azimuth, and flight path angle.

9.3.2 Constraints

- The various time and state continuity conditions along the trajectory.
- There is a minimum 5.15 day outbound (launch to OPF) flight time constraint in order to maximize flight time and ensure consistency for the secondary payloads on board ICPS.
- The EI longitude, azimuth, and flight path angle are constrained to be on the EI target line.
- The total propellant mass used by the XDM OPF through DRI segments is constrained to be equal to the propellant mass used by the nominal OPF through DRI segment burns.
- The total propellant used by the XDM RPF through RTC-4 segments is constrained to be equal to the propellant mass used by the nominal RPF segment burn.
- Maximum and minimum bounds for distance traveled between EI and splashdown site.
- Sunrise and sunset time constraints for landing.
- Maximum eclipse durations throughout the trajectory.
- Non-viable latitude locations for EI - between 14 and 21.9 latitude, due to the SM disposal requirements near the Hawaiian islands.

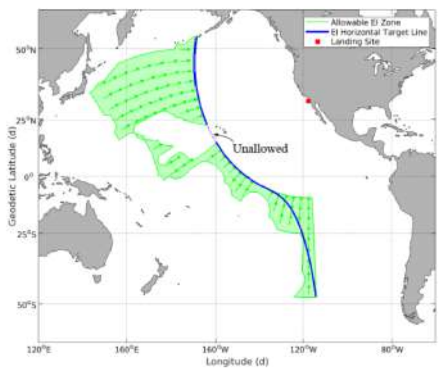


Figure 9.6: EI Target Line with Allowable Zone. The blue target line approximates the green region in the optimization problem.

9.3.3 Optimisation Algorithm

To generate launch window trajectories, the "optimal point" is found first, which is the optimal launch azimuth point per day corresponding to the minimum ICPS plus Orion total (delta)v. An appropriate mission class and length is selected based on Orion constraints and any additional constraints are added in order to mitigate violations. The final optimized trajectory is used as the origin to compute the launch window. The optimization problem is then proceeded as follows :

- The launch epoch is fixed.
- The secondary payload outbound transit time with a minimum bound of 5.15 day from launch to OPF is removed.
- The OPF epoch tOPF is fixed to ensure a variable launch azimuth mission design approach is achieved. This helps to ensure more consistent outbound transit times.
- OTC-2 is enabled as an optimized burn.
- The DRO insertion state and time (tDRI) is locked down. This helps to ensure the same DRO is achieved during the entire window.

Note that, since the launch window scan is done using the TLI database (beginning at the Orion Sep point), only the Orion v is being minimized in the objective function. The launch scan continues (forward and backwards) until any of the following conditions are violated:

- The problem does not converge (i.e., the mission is infeasible).
- Orion exhausts its available propellant for major burns.
- The launch to OPF duration is < 4 days.

A continuation method is used to perform epoch scans of the basic mission, mission classes, and computation of the launch window. There are three kinds of scans: epoch scan, mission class scan, and launch window scan. During the scan, the lunar geometry is used in various ways to update the initial guess to provide robust convergence. In order to make Earth- Moon geometry similar, for epoch steps, rather than using integer days, a better time step is computed for each epoch using a simple lunar ephemeris.

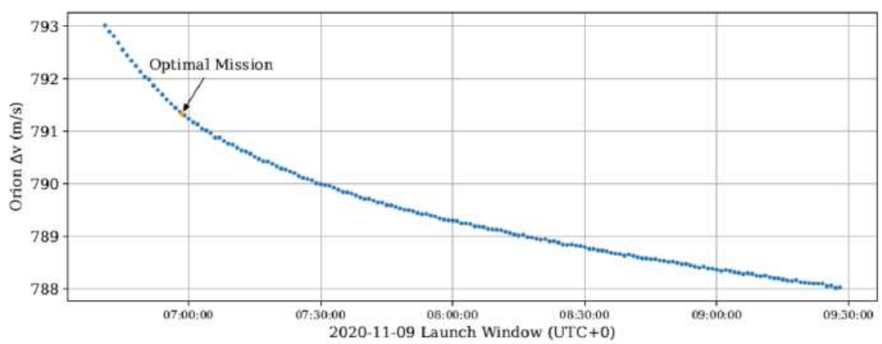


Figure 9.7: Graph obtained after Launch Window Optimization

9.4 Off-Normal Trajectories

For every nominal mission trajectory, the Artemis I mission design team will create a set of off nominal missions. They can be broken into three categories depending on the goal of the off-nominal trajectory:

- An alternate mission is a response to a failure that occurs prior to the Orion-ICPS separation. In this scenario, the baseline mission is infeasible and is replaced with a scenario within vehicle constraints.
- A return to the nominal trajectory is a response to a relatively small failure, such as a missed or partial Orion burn. In this scenario, a burn did not perform as expected due to a failure or operational constraint. A recovery burn is used to return Orion to the baseline Artemis I trajectory and ensures all intended mission objectives are met.
- An abort trajectory is a response to a significant failure. It is used in the event the vehicle cannot complete the nominal mission and must abort to ensure vehicle recovery.

10. MOGA Modelling

Invoking the Trajectory Optimisation equations from the data obtained from Altitude Testing of the Space Launch System would have been the basic pre-requisite to proceed with the application of the Multi-Objective Genetic Algorithm(MOGA). But due to the unavailability of that data publicly at present, we can only obtain rough estimates about the Thrust and Angle profiles based on the initial parameters we know. The plots obtained by the spline interpolation of tentative Thrust and Angle data in MATLAB come out to be:

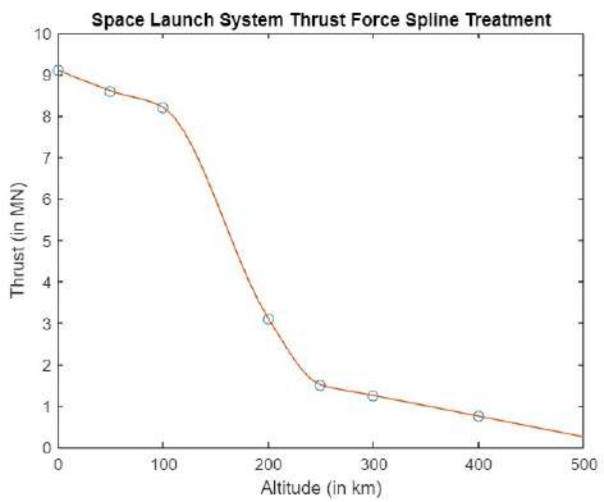


Figure 10.1: Spline Interpolation in MATLAB

Due to lack of data, it is also not possible to plot Pareto Fronts to obtain the high-payload and low-cost optimised individuals. But, there is another technique via which several other parameters can be taken into consideration and the GA can be applied to the functions generated to get the optimised results and the Pareto fronts.

This technique is very well explained in this research paper by Mehran Mirshams, Hasan Naseh, H.R. Fazeley.

For this mission, a new multi-objective technique using Holistic Concurrent Design(HCD) and Multi-Objective Genetic Algorithm(MOGA) is applied to optimize the multidisciplinary design of Space Launch System(SLS).

This method reduces the multi-objective constrained optimization problem to a single-objective unconstrained optimization. The design problem is established using the fuzzy rule set based on the designer's expert knowledge with a holistic approach. The independent design variables in this model are nozzle exit pressure, combustion chamber pressure, oxidizer to fuel mass flow rate(O/F), stringer thickness, ring thickness, shell thickness. To handle the mentioned problems, a fuzzy–MOGA optimization methodology is developed based on the Pareto optimal set.

Taking Propulsion and Structure as base, objective functions are formed with certain range constraints on few parameters. This is followed by plotting of Pareto Fronts by the GA for several generations. And thus, ultimately getting the optimum design solutions, which are depicted in the Table below:

Design variables						
	P_{CC} (bar)	O/F	P_{ex} (bar)	t_{st} (mm)	t_r (mm)	t (mm)
Initial	69.64	2.25	0.7716	5.57	5.68	5.20
Final	70.75	2.27	0.7255	5.45	5.5	5.17
Attitude parameters			Wish design attributes			
	$[p, q, \alpha]$		W_{Tank} (ton)	W_{eng} (ton)	I_{SP} (s)	
Initial	[9.0, 1.6, 0.6]		33.5	8.245	245.0	
Final	[10.77, 1.32, 0.58]		32.8	8.440	263.0	
Overall must satisfaction ($\mu_M^{(p)}$)			Overall satisfaction ($\mu^{(p,q,\alpha)}$)			
Initial	0.356		0.255			
Final	0.580		0.555			

Figure 10.2: Initial and Final design solutions

Also, the Pareto Fronts obtained for the objective functions have been plotted in the research paper as well:

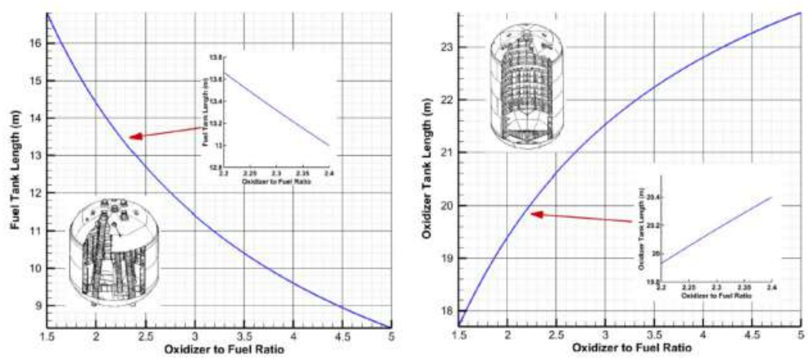


Figure 10.3: Fuel Tank weight and Oxidizer Tank length vs O/F

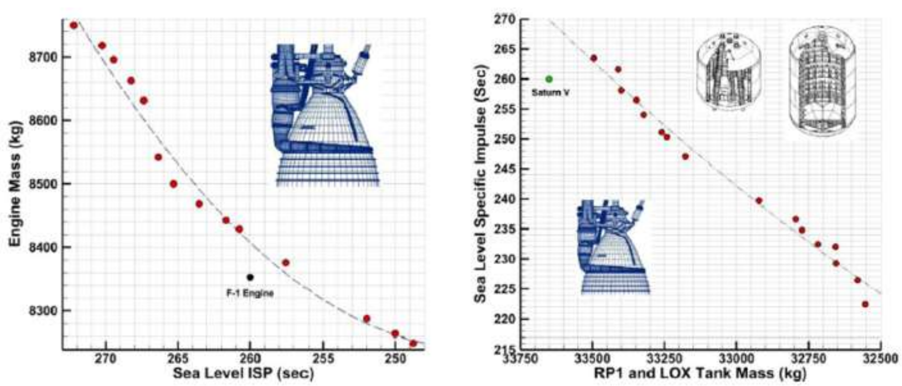


Figure 10.4: Pareto Frontier for objective functions (Specific Impulse, Engine Weight, Tank Weight)

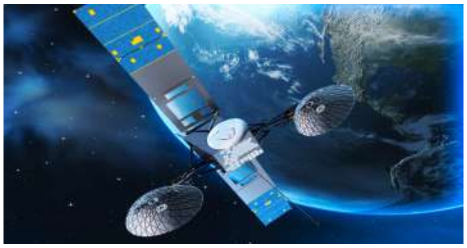
So, finally, the optimum values for the considered parameters are obtained. Hence, this methodology provides an interesting decision-making approach to design multi-product batch plants under conflicting goals.

11. Communication

Artemis I will demonstrate NASA's networks' comprehensive services for journeys to lunar orbit. The mission requires all of NASA's space communications and navigation networks to work in tandem, providing different communications and tracking service levels as Orion leaves Earth, orbits the Moon, and returns safely home. Communications services allow flight controllers in mission control centers to send commands to the spacecraft and receive data from Orion and SLS systems. Tracking, or navigation, services enable the flight controllers to see where the spacecraft are along their trajectory through space.

11.1 ARTEMIS I Navigation

Navigation services enable flight controllers to track where spacecraft are along their trajectory through space. On the journey to the Moon and in orbit around the Moon, the Deep Space Network's large ground antennas will provide primary tracking data. Near Space Network ground stations in Chile and South Africa will supplement this tracking data.



Navigation Sattelite.

11.2 Network Support for ARTEMIS I

11.2.1 Near Space Network(NSN)

NASA's Near Space Network provides a comprehensive suite of communications and navigation services through commercial and government-owned, contractor-operated network infrastructure. It's mainly divided into two components:

1. **NSN DTE**- Near Space Network DTE services are provided by a worldwide network of ground stations
2. **NSN TDRS**- The Near Space Network's TDRS constellation can provide near-continuous communications services to spacecraft near Earth.

11.2.2 Deep Space Network(DSN)

The Deep Space Network will handle Artemis I communications beyond Near Space Network coverage, en route to and in orbit around the Moon. Additionally, the network will facilitate communications during the deployment of CubeSat payloads that will provide additional research opportunities for Artemis I.

Figure 11.1: NSN Equipment.

Figure 11.2: DSN Ground Equipment.

11.3 Search And Rescue (SAR)

The Orion spacecraft is equipped with an emergency beacon designed by NASA's Search and Rescue office. Using Cospas-Sarsat, the international satellite-aided search and rescue network, this beacon will help NASA to quickly locate Orion upon activation of the beacon during splashdown or in the unlikely event of an abort scenario.

Future Internet Like Search and Rescue Network.

11.4 ARTEMIS Mission Support

NASA's Space Communications and Navigation (SCaN) program provides strategic oversight and funding to NASA's networks and to the development of new communications and navigation technologies. SCaN will support all Artemis missions while providing astronauts with revolutionary communications capabilities. SCaN is also developing LunaNet, a exible lunar communications and navigation architecture that will play a key role in NASA's ambitious exploration initiatives under the Artemis program. LunaNet will allow NASA to extend internet-like service to the Moon

ANGEL:Developed By Nasa.

12. Re-entry

When NASA's Orion spacecraft is nearing its return to Earth after its Artemis I mission to the Moon, it will attempt the first skip entry for a human spacecraft – a maneuver designed to pinpoint its landing spot in the Pacific Ocean.During this skip entry, Orion will dip into the upper part of Earth's atmosphere and use that atmosphere, along with the lift of the capsule, to skip back out of the atmosphere, then re-enter for final descent under parachutes and splashdown. It's a little like skipping a rock across the water in a river or lake.This skip entry has various benefits over the classical entry method used. It will divide the impact thus reducing the G force experienced by astronauts Eventually making the ride more safer and smoother. Dividing the impact will affect the friction heating caused. It will be reduced. Also skip entry will make the landing more accurate thus reducing the recovery coast, as now the navy won't have to deploy various ships in the target sea/ocean.

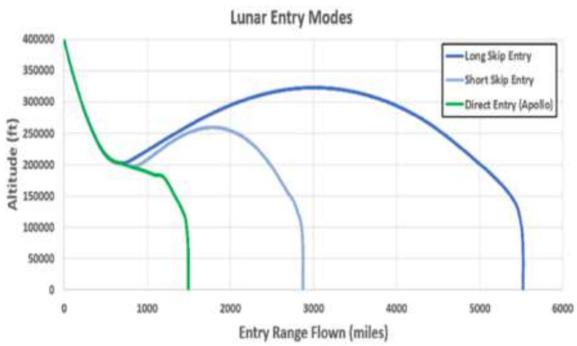


Figure 12.1: Entry Range Vs Altitude Graph.

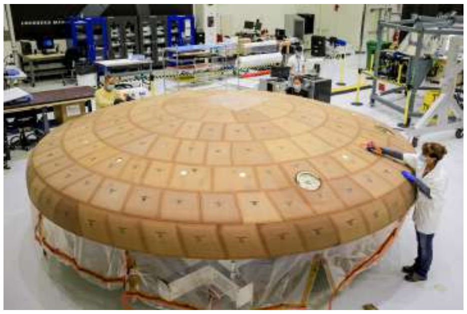


Figure 12.2: Orion Heat Shield.

12.1 Heat Shield

Orion returns on a high-speed Earth reentry at Mach 32, or 24,500 miles per hour, thus heating the module to nearly 5,000 degrees Fahrenheit before splashing down in the Pacific Ocean for retrieval and post-flight engineering assessment.To protect the Crew Module during Earth re-entry, the dish shaped AVCOAT heat shield ablator system, in a honeycomb structure was selected as heat shields. Licensed by Textron, AVCOAT material is produced at New Orleans's Michoud Assembly Facility by Lockheed Martin. This heat shield will be installed at the base of the crew module to provide a controlled erosion moving heat away from the crew module into the atmosphere.Its honeycomb structure prevents the module from ablation. It was after rigorous testing and study of its structure, that NASA approved of them to be installed on the crew module.

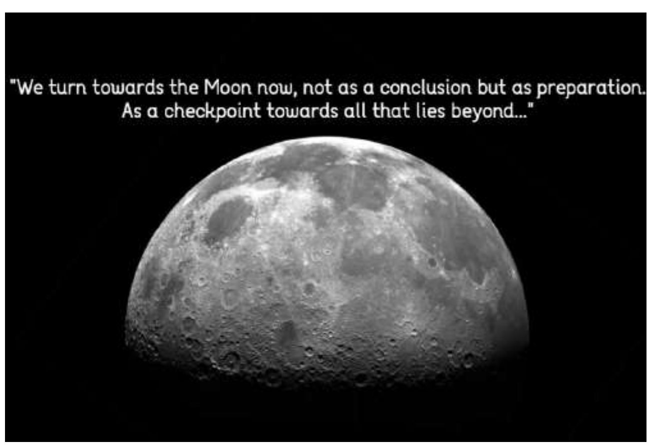
13. Interstellar Voyage and Conclusion

With confidence based on the Artemis I mission and the thousands of hours put into the prior flight and ground testing, NASA plans to send the Artemis II crew who will board Orion atop the SLS for an approximate 10-day mission. It will be the first crewed flight of SLS and Orion that will send four astronauts to the lunar environment for the first time in more than 50 years. They will set a record for the farthest human travel beyond the far side of the Moon in a hybrid free return trajectory.

Further, by 2024 NASA plans to send Artemis III, culmination of the rigorous testing and more than two million miles accumulated in space on NASA's deep space transportation systems during Artemis I and II. This time, Orion and its crew of four will travel to the Moon again, but to make history with the first woman and next man to walk on its surface.

Gateway is a crucial part of Artemis program, which will seed our future deep space explorations. Gaining new experiences on and around the Moon will prepare NASA to send the first humans to Mars in the coming years, and the Gateway will play a vital role for all the landers and spacecraft enroute to destinations beyond Moon. The Gateway-to-surface operational system is also analogous to how a human Mars mission may work—with the ability for the crew to remain in orbit and deploy to the surface. It is crucial to gain operational confidence in this system at the Moon before the first human missions to Mars.

This incremental build-up of capabilities and robotic presence on and around the Moon is essential to establishing long-term exploration of Earth's nearest neighbor Mars. With our arrowhead pointed to Mars, we aim to explore deep into space, and bring back the information and knowledge collected back to earth. By partnering with several native and international partners, NASA aims to build a sustainable space economy, alongwith making space accessible for humans to probe for knowledge and explorations. This mission, seed from minds with a vision, backed by perspicacity and technology, will open paths to deep space, thus help uncover space mysteries and paths to trace back our existence in the Universe.



14. References

1. The Artemis Plan: NASA's Lunar Exploration Program Overview, Link to the Document
2. Boeing's Page for SLS Description, Link to the page
3. NASA's Page on SLS, Link to the page
4. Amelia L. Batcha, Jacob Williams, Timothy F. Dawn, Jeffrey P. Gutkowski, Maxon V. Widner, Sarah L. Smallwoodk, Brian J. Killeen, Elizabeth C. Williams, and Robert E. Harpold. *Artemis I Trajectory Design and Optimization,* Link to the Paper
5. Mehran Mirshams, Hasan Naseh, H.R. Fazeley. *Multi-objective multidisciplinary design of Space Launch System using Holistic Concurrent Design*. Aerospace Science and Technology, Volume 33, Issue 1. Link to the Paper
6. Derek J. Dalle. *Launch Vehicle Aerodynamics Database Development for SLS,* Link to the Paper
7. NASA Guide on Spacecraft Structures, Link to the Guide
8. eoPortal Directory on Artemis I, Link to the Directory
9. Artemis Communication Systems and Networks, Link to the page
10. New Re-entry techniques for Orion, Link to the page
11. NASAfacts: Orion Launch Abort System, Link to the Fact Sheet
12. William J. Koenig, Carlos Garcia, Lisa D. Akers, Richard F. Harris. *Artemis Innovative Assembly and Integration Operations of the Launch Abort System at KSC,* Link to the Paper

SPACE

THE FINAL FRONTIER

SAMPLE MISSION REPORT

CREW DRAGON - RESILIENCE

MENTOR

Sunny K. Bhagat | sunny@iitk.ac.in

MENTEES

ADIT JAIN | aditj20@iitk.ac.in

ANUBHA THAKUR | anubhat20@iitk.ac.in

NALLANI NITHIN | nallani20@iitk.ac.in

PARTH BHATT | parthb20@iitk.ac.in

SHASHANK SINHA | shashanks20@iitk.ac.in

SHIKHA SINGH | sshikha20@iitk.ac.in

MISSION REPORT

SpaceX Crew Dragon: RESILIENCE

SPACEX Crew Dragon - Resilience

Sample Mission Report, Group 3

1. INTRODUCTION

The Space Crew-Dragon 1 was launched on November 16th on Falcon 9 to the orbit from NASA's Kennedy Space Center, Florida. The mission was a part of NASA's Commercial Crew Program.

The crew included Mike Hopkins (Commander), Victor Glover (Pilot), Shanon Walker and JAXA astronaut Soichi Noguchi (Mission Specialists). Duration of the mission was approximately 6 months. The crew returned back on earth on May 2nd 2021. The spacecraft *Resilience* splashed down in the Atlantic ocean off the coast of Florida.

Falcon 9 is a reusable double stage rocket. A double-stage-to-orbit launch vehicle is a spacecraft premised on the concept of a reusable launch system using two rocket stages, each containing its own engines and propellant - provide propulsion consecutively to achieve orbital velocity. The two stages are designed so that the first stage is reusable while the second is expendable.

The objective of the mission was to reduce the cost of production, transport of 500 pounds of cargo and crew to the ISS and for further scientific research like food physiology of the crew, effect on gene and how it affects the brain functionality of the crew while they are in space, how microgravity affects the the structure and functionality of organs, growth of plants in space in various types of soils and varying light and temperature, to improve the design of spacesuit, and many more.

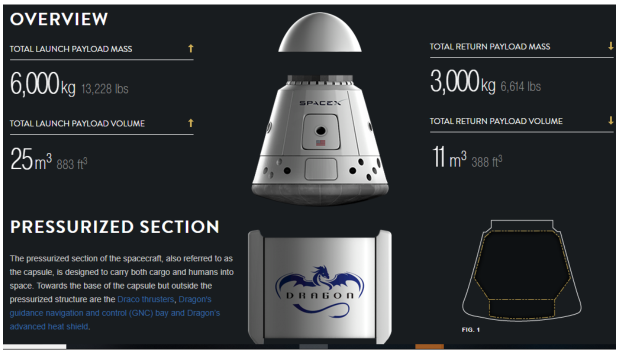


Fig-1: Overview

2. Payload

The payload system of a rocket depends on the rocket's mission. The earliest payloads on rockets were fireworks for celebrating holidays. The payload of the German V2, was several thousand pounds of explosives. Following World War II, many countries developed guided ballistic missiles armed with nuclear warheads for payloads. The same rockets were modified to launch satellites with a wide range of missions; communications, weather monitoring, spying, planetary exploration, and observatories, like the Hubble Space Telescope. Special rockets were developed to launch people into earth orbit and onto the surface of the Moon.

For this mission, the *Resilience* was able to carry a payload of 6,000 kilograms (13,000lb) to the orbit, which can be all pressurized, all depressurized or anywhere between. It can return to Earth 3,000 kilograms (6,600lb) of return pressurized cargo and 800kg (1,800lb) disposed cargo. The spacecraft also delivered over 500 pounds of cargo, science hardware and experiments to the ISS.

Falcon 9 can lift payloads of up to 22,800 kilograms (50,300 lb) to low Earth orbit (LEO), 8,300 kg (18,300 lb) to geostationary transfer orbit (GTO) when expended, and 5,500 kg (12,100 lb) to GTO when the first stage is recovered, in a cargo shroud offering 145 cubic meters of volume.

3. Fairing

The fairing is a protective cover that surrounds the payload on the launch vehicle as it ascends through Earth's atmosphere on its way to space.A payload fairing is a nose cone used to protect a spacecraft payload against the impact of dynamic pressure, biospheric contamination and aerodynamic heating during launch through an atmosphere. An additional function on some flights is to maintain the cleanroom environment for precision instruments. Once outside the atmosphere the fairing is released, exposing the payload to outer space.

Payload fairings usually burns up in the atmosphere or destroyed upon impacting the ocean. SpaceX has managed to successfully catch the fairings – catching both halves of the fairing used on one of its Falcon 9 rocket launches. It attempts to reduce the cost of its launches by building In as much reusability as it can. SpaceX estimates that it can save as much as $6 million per launch by recovering and reusing the fairing halves.

The fairing halves don't have any propellant systems to control their landing like the Falcon 9 first stages do – instead, they're slowed via parachutes, meaning there's a bigger reliance on the ships to actually be positioned correctly to anticipate their fall, since it's not specifically programmed.

Falcon 9 fairing measures as 13 m (43 ft) long, 5.2 m (17 ft) in diameter, weighs approximately 1900 kg, and is constructed of carbon fiber skin overlaid on an aluminum honeycomb core. The Falcon 9 fairing can accommodate a combination of up to three access doors or radio frequency (RF) windows in the cylindrical portion. The standard payload fairing door is a maximum of 24 inches (61 cm) in size. Combinations of acoustic surfaces are used inside the payload fairing to help achieve the acoustic environment.

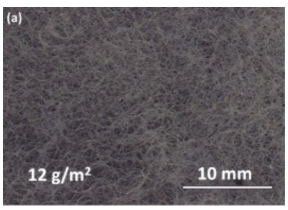


Fig-2a: Material Texture

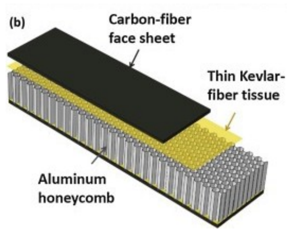


Fig-2b: Faring Material

The Falcon 9 was described as capable of launching approximately 9,500 kilograms (20,900 lb) to low Earth orbit, and was projected to be priced at US $27 million per flight with a 3.7 metres (12 ft) payload fairing and US $35 million with a 5.2 metres (17 ft) fairing.

4. Design

SpaceX's Dragon space capsule design is a gumdrop-shaped spacecraft built for spaceflights into and from low-Earth orbit. The spacecraft was initially designed as an unmanned spacecraft, but is being scaled up to launch astronauts into space as well.

Crew Dragon was developed in collaboration with NASA's Commercial Crew Program. In 2014, NASA awarded Commercial Crew Transportation Capability (CCtCap) contracts to Boeing and SpaceX to each safely and cost-effectively transport astronauts to the International Space Station from the United States. Crew Dragon is capable of carrying up to seven passengers but will carry up to four astronauts for NASA missions, and is designed for water landings. Crew Dragon's displays will provide real-time information on the state of the spacecraft's capabilities—anything from the spacecraft's position in space, to possible destinations, to the environment on board. Crew Dragon is a fully autonomous spacecraft that can be monitored and controlled by onboard astronauts and SpaceX mission control in Hawthorne, California.

1. Dragon is composed of two main elements: the capsule, which is designed to carry crew and critical, pressurized cargo, and the trunk, which is an unpressurized service module. The capsule is subdivided into the pressurized section, the service section and the nose cone, which is opened once on orbit and stowed prior to re-entry.Above the seats, there is a three-screen control panel, a toilet (with privacy curtain), and the docking hatch. . Near the base of the capsule, but outside the pressurized structure, are the Draco thrusters, which allow for orbital maneuvering. Additional Draco thrusters are housed under the nose cone, along with Dragon's Guidance Navigation and Control (GNC) sensors . Ocean landings are accomplished with four main parachutes in both variants. The parachute system was fully redesigned from the one used in the prior Dragon capsule, due to the need to deploy the parachutes under a variety of launch abort scenarios.
2. Crew Dragon has an Environmental Control and Life Support System (ECLSS) that provides a comfortable and safe environment for crew members. During their trip, astronauts on board can set the spacecraft's interior temperature to between 65 and 80 degrees Fahrenheit.
3. Crew Dragon has eight side-mounted SuperDraco engines, clustered in redundant pairs in four engine pods, with each engine able to produce 71 kN (16,000 lbf) of thrust to be used for launch aborts. Each pod also contains four Draco thrusters that can be used for attitude control and orbital maneuvers. The SuperDraco engine combustion chamber is printed of Inconel, an alloy of nickel and iron, using a process of direct metal laser sintering. Engines are contained in a protective nacelle to prevent fault propagation if an engine fails *(to be discussed further in engines and propellant section)*.
4. Once in orbit, Dragon 2 is able to autonomously dock to the ISS *(to be discussed further in rocket equations section)*. Dragon 1 was berthed using the Canadarm2 robotic arm, requiring substantially more involvement from ISS crew. Pilots of Crew Dragon retain the ability to dock the spacecraft using manual controls interfaced with a static tablet-like computer. The spacecraft can be operated in full vacuum, and "the crew will wear SpaceX-designed space suits to protect them from a rapid cabin depressurization emergency event". Also, the spacecraft will be able to return safely if a leak occurs "of up to an equivalent orifice of 6.35 mm [0.25 in] in diameter".
5. Propellant and helium pressurant for both launch aborts and on-orbit maneuvering is contained in composite-carbon-overwrap titanium spherical tanks *(to be discussed further in engines and propellant section)*. A PICA-X heat shield protects the capsule during re-entry, while a movable ballast sled allows more precise attitude control of the spacecraft during the atmospheric entry phase of the return to Earth and more accurate control of the landing ellipse location. A reusable nose cone "protects the vessel and the docking adaptor during ascent and re-entry", pivoting on a hinge to enable in-space docking and returning to the covered position for re-entry and future launched *(to be discussed further in re-entry section)*.
6. Dragon's trunk provides the mating interface for the capsule to Falcon 9 on its ascent to space. On orbit, half of the trunk contains a solar array, which powers Dragon, and the other half contains a radiator, which rejects heat. Both the radiator and solar array are mounted to the exterior of the trunk, which remains attached to Dragon until shortly before re-entry when the trunk is jettisoned.
7. The technical definition of a Fin is: A surface used to give directional stability to any object moving through a fluid such as water or air. In short, fins provide maneuverability and stability to the rocket in the upper atmosphere. Falcon 9 is equipped with four hypersonic grid fins positioned at the base of the interstage. They orient the rocket during reentry by moving the center of pressure.

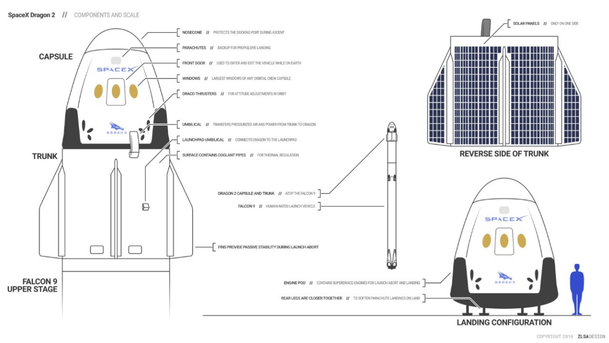


Fig-3: RESILIENCE Information

4.1. System Overview of Falcon 9

1. Height: 70 meters or 229.6 feet
2. Mass: 549,054 kilograms or 1,207,920 pounds
3. Diameter: 3.7 meters or 12 feet

5. Boosters

A booster rocket is either the first stage of a multistage launch vehicle, or else a shorter-burning rocket used in parallel with longer-burning sustainer rockets to augment the space vehicle's takeoff thrust and payload capability.

A Falcon 9 first-stage booster is a reusable rocket booster used on the Falcon 9 and Falcon Heavy orbital launch vehicles manufactured by SpaceX. The manufacture of first-stage booster constitutes about 60% of the launch price of a single Falcon 9 and three of them over 80% of the launch price of a Falcon Heavy, which led SpaceX to develop a program dedicated to recovery and reuse of these boosters for a significant decrease in launch costs.

The booster used in this mission is Booster 1061. Falcon 9 B1061 first launched Crew-1 to the ISS in November 2020, the first operational flight of Crew Dragon. Following landing on drone ship following the Crew-1 flight, this first stage has completed three flights by June 2021. B1061 is a Falcon 9 Block 5 core *(to be discussed further in aerodynamics section)*.

6. Abort System

A launch escape system (LES) or launch abort system (LAS) is a crew-safety system connected to a space capsule that can be used to quickly separate the capsule from its launch vehicle in case of an emergency requiring the abort of the launch, such as an impending explosion.

The abort system of the spacecraft can be activated by three ways:

1. The crew can pull a handle inside the spacecraft.
2. Mission control can send a remote command to the spacecraft.
3. Or the craft itself can automatically start the sequence if it detects a problem in the rocket.

This will cause the eight small SuperDraco rocket engines on the capsule to fire and lift it away from the rocket.

During the abort, the astronauts experience extreme forces, stronger than gravity, ascending about approximately a kilometer and a half before the capsule splashes down in the Atlantic Ocean under parachute. It's an extreme maneuver for extreme emergencies.

There are four compartments (Quads) that integrate a total of eight SuperDraco engines (four pairs, two per Quad) with a thrust of 73 kN per unit. SuperDraco, high-performance hypergolic propellant engines, serve the dual role of abort thrusters on emergency launch and braking retro rockets on descent. Each of the eight SuperDraco engine generates 15,000 pounds of thrust and burns about six seconds. The test began at 9 a.m. After the engines shut down, the Dragon spacecraft's trunk, will separate when it reaches peak altitude.

The most dicey part of the launch occurs in the second abort stage. This is the point of peak aerodynamic stress known as "max q," which occurs about a minute and a half after launch. The rocket is moving at about 1,500 mph and all the aerodynamic pressure experienced by the capsule during max q makes it the worst possible time to abort. But it's also the period during a launch when things are most likely to go wrong.

During the Crew Dragon launch abort test as the rocket entered max q, SpaceX mission control killed its engines. The capsule automatically registered that something was wrong, fired its SuperDraco engines, and pulled away from the Falcon 9 rocket as it exploded in the air. The capsule kept coasting into the stratosphere before beginning its descent to Earth and splashing down in the Atlantic Ocean under parachute. When it comes to human spaceflight, the best abort scenario is the one that never happens *(to be discussed further in engine and propellant section)*.

7. Engine and Propellants

Here our goal is to take astronauts to ISS(international space station) which is revolving at 7.66 km/s around earth.so we need to keep the astronauts in the orbit with nearly same velocity so they can reach ISS without any major impulse which kills them.So we need to apply force(human bearable acceleration<9g) which does work against gravity to reach required altitude and accelerates the astronauts to the required velocity.

Till now the best feasible way to apply force on an object that is going to far distances in vacuum(where there is no other significant energy source that can apply force in our desired direction) is expelling out mass carrying with us in the direction opposite to our desired direction of motion.So by Law of conservation of momentum our velocity will increase in desired direction.To apply more force we need to eject more mass as fast as possible with more velocity.

But our desired velocity is so huge that we need to carry a lot of mass with us to expel it out with some low velocity.So we can save money and resources by ejecting less amount of mass with very high velocity so that we can reach our desired velocity.' But how can we expel the mass with such huge velocities? Is there any such machine that can expel the mass with such high velocity? From where can we give energy to such a machine to work? The best method to do this is to use energy from expelling mass itself to throw it with high velocity.So the expelling mass will be storing the energy to use while expelling it out. So we are extracting the energy from the expelling mass by chemical reactions.(forget about nuclear fission and fusion for now,take it for granted that they are not feasible in this mission)

The mass that is using its own chemical energy to expel out with higher velocity is called FUEL.Combustion is the most common chemical reaction that gives out huge amounts of energy. But we need oxygen for combustion,but in most of the points of trajectory of our rocket we don't have required concentration of oxygen to carry out combustion.So we will be carrying OXYGEN with us along with fuel to meet the oxygen demand. Here the machine that expels out the mass as fast as possible is called ENGINE.

8. Falcon 9 Specifications

8.1. Engine

Merlin - gas generator powered open cycle engine(both above sea level and vacuum optimized engines exist)

8.2. Propellant

Keralox(RP-1 + liquid oxygen), where RP-1 is the rocket grade kerosene

9. Gas Generator Powered Open Cycle Engine

This is the simplest combustion engine.First of all fuel and oxidizer is stored in different tanks.These two undergo combustion in the combustion chamber so that the end products eject out with high velocities by absorbing the energy from combustion reaction.Thus we are able to eject the mass with higher velocities. But the pressure and temperature is very high in the combustion chamber compared to pressure in fuel and oxidizer tanks.So propellant cannot move from high pressure to low pressure.So we want a higher pressure region in the propellant's side than in the combustion chamber such that propellent moves from fuel tank to combustion chamber. But if we increase the pressure in the tank such that it is greater than in the combustion chamber,we need to make tanks with very thick and costly material so it can withstand such pressures.And also the fuel tanks become very heavy so most of the fuel is consumed to accelerate them.So increasing pressure in the fuel tank is a bad idea.

9.1. Turbo Pump

Instead we will be using turbo pumps to increase the pressure such that it is greater than that in the combustion chamber.But we need to power the turbo pumps.As we already have a energy source from propellant,we will be using it to power the turbo pump instead of other sources.Some part of fuel and oxidizer mixture is diverted to the pre-burner where that fuel undergoes combustion and the products evolve out from the preburner with higher velocities,these gases are sent through the turbine,where they rotates the turbine and finally ejected out.The turbine is is connected to other two turbo pumps with same shaft thus powering them.

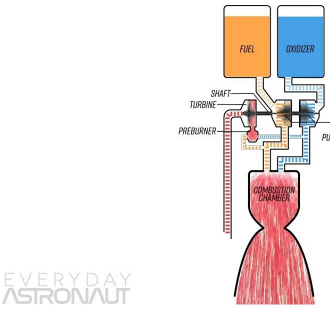


Fig-4: Turbo Pump

9.2. Fuel Rich Combustion

But there is a problem here.The combustion temperature is very high(3000k) such that the pre-burner and combustion chamber are made with high melting point materials and also very thick.But the turbine is needed to be made with very thin and lightweight material(which generally will not have high melting points) so it will be rotated fast by gases.Thus the turbine melts off with such high temperatures,so we need to decrease the temperature.So we are making an inefficient combustion by keeping incorrect fuel oxidizer mixture(fuel rich or oxidizer rich) in the pre-burner,so that partial combustion occurs and temperature stays low.

In the Merlin engine we are using a fuel rich preburner,where all the fuel is not involved in combustion,so remaining fuel absorbs that excess heat and decreases the temperature.Thus the partially burnt propellant comes out as soot (black colour).

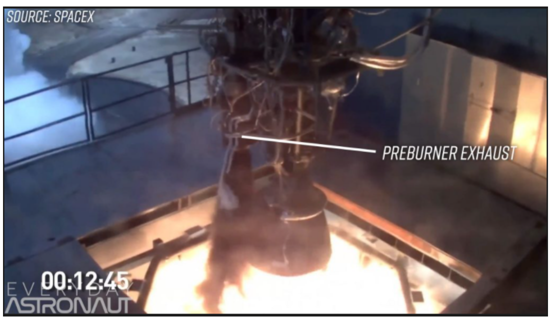


Fig-5: Exhaust

9.3. Pintle Injector

Fuel and oxidizer should be mixed thoroughly in the combustion chamber for the combustion to be efficient.So the fuel and oxidizer are atomized(gaseous bubbles) so that they mix thoroughly.

Fig-6: Pintile Injector atomizing Water

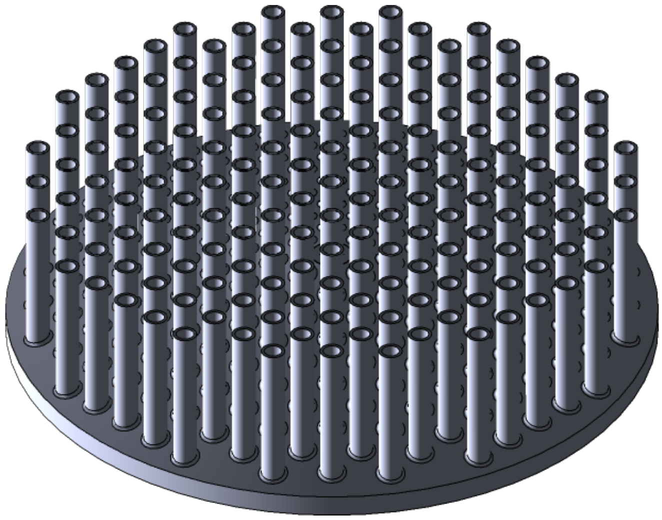


Fig-7: Pintile Injector Visualization

9.4. Nozzle

After combustion of RP-1 and oxygen,carbon dioxide(CO_2) and water vapour(H_2O) are formed and they get kinetic energy by absorbing the energy released from combustion.But their velocities are in different directions and also small due to high pressure.We need to direct these gases in one direction and increase their velocity,so that we will be getting maximum thrust.

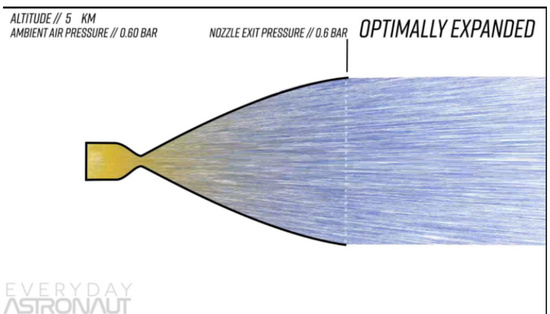


Fig-8: Optimally Expaned thrust

First of all we direct the gases in one direction by decreasing the area of cross section,thus velocity increases and pressure decreases.As the pressure decreases when we are moving from higher to lower cross section,we are able to direct the gases in one direction.As we move to lower cross-section the velocity keeps on increasing and finally we reach speed of sound at that point(at that particular temperature). If we increase the velocity furthermore,then shock waves form and choking occurs.From now we will stop decreasing cross section area and start decreasing the pressure by increasing the cross section in a particular format as shown in the top figure. As the pressure is decreasing the velocity keeps on increasing.But we need to stop expanding the nozzle exactly when the pressure equals the external pressure.If we don't do it then it will cause gas to diverge or converge leading to decrease in thrust.

10. Merlin Engine

10.1. Max Thrust

Sea level engine: 854 KN
Vacuum optimized engine: 981 KN

There are 9 engines per booster and there are 3 boosters(27 engines) in the 1 st stage. There is only one upper stage vacuum optimized engine with burn time of 397 s. At the time of liftoff engines are throttled down such that the thrust at liftoff is 7686 KN

10.2. Thrust to Weight Ratio

198 m/sec^2
The Merlin engine has the highest thrust to weight ratio of all other rocket engines.

10.3. Chamber Pressure

97 bar

10.4. I_{sp}

Above sea level(ASL): 282 s Vacuum(VAC): 346 s

11. RP-1 Fuel

1. Density: 813 g/l
2. Boiling point: 490K
3. Boiling point: 490K
4. Boiling point: 490K
5. Boiling point: 490K

12. Other Engine Information

1. Price of one engine: \$ 1M
2. Price of one engine: \$ 1M
3. cost/Thrust: 1170 \$KN
4. Cost per KN per Flight: 117 \$KN
5. Success rate of engine(for 71 flights till now): 99.9%

13. Rocket Equation

Lets now derive the rocket equation and apply it in this case. Just to make things simpler, we will be considering the ideal case in which the drag force due to air and the force of gravity are neglected. In doing so we will be saving a lot of unnecessary calculations and easily discuss the main concepts.

The two major considerations for ideal rocket equation taken are:-

- Conservation of mass
- Conservation of momentum

thus we get,

$$M.(\frac{dv}{dt}) = \frac{dM}{dt}.(-v_{ex}) \tag{1}$$

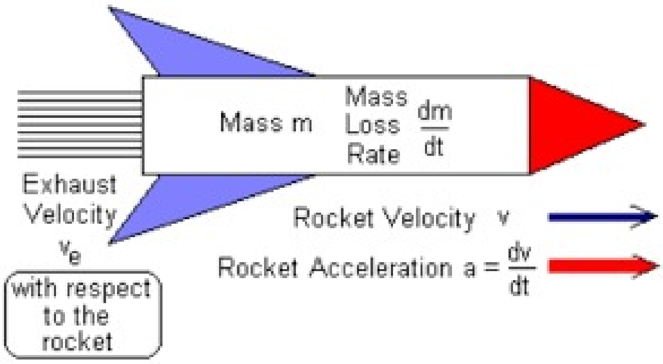


Fig-9: Diagramatic Representation

*where M is the **mass of the rocket** and v_{ex} is the velocity of the dm mass of the propellant i.e. **exhaust velocity**.*

On further solving equation (i),we get,

$$\Delta V = V_{ex}.ln(\frac{M_o}{M_f}) \tag{2}$$

[v_{ex} is somewhat constant at steady state arrival]

here, M_o**–initial mass of rocket**, M_f**-final mass of rocket** and v is the destination velocity i.e, orbital velocity where the rocket is supposed to land.

We can also get the **mass of the propellant**, ΔM required for the entire journey by the formula below:-

$$\frac{\Delta M}{M} = 1 - e^{(\frac{-\Delta v}{v_{exhaust}})} \tag{3}$$

13.1. Specific impulse

In the language of ROCKET SCIENCE, specific impulse, I_{sp} can be defined as "the efficient use of propellants in rockets/spacecrafts and fuels in jet engines". It is thus the measure of how proficiently a reaction mass creates thrust.

$$I_{sp} = lim_{\Delta t \to 0} \frac{F_{thrust}.\Delta t}{M_p.g_o} \tag{4}$$

M_p is the mass of propellant and F_{thrust} is the thrust.

The above formula could be used to get the destination velocity V as well which is given by:-

$$\Delta V = I_{sp}.g_o.ln(\frac{M_o}{M_f}) \tag{5}$$

14. Trajectory to the ISS

The International Space Station is orbiting the earth at an altitude of approximately 408 km with an orbital velocity of 7660 m/s. We need to reach the same orbit with our mission, so that the crew dragon capsule is almost stationary with respect to the ISS for easy docking.

You might think based on what has been discussed in the previous section, just by knowing the ΔV required to reach a specific orbit we can just launch our rocket and give it that ΔV and we are done! And you are partially correct to assume that as the rocket has to reach that specific ΔV but it does not happen in one go. There are various burn stages that the rocket goes through:-

- After the launch of the rocket the stage 1 gets separated at an altitude of about about 90 km. The first stage flies off, following a ballistic trajectory and actually crosses the karman line at its apoapsis,then goes on to land safely on a floating pad in the ocean,which had been placed based on precise calculations of its trajectory.
- Then the rocket goes into a stable orbit of about 200 km, where the rocket is brought in phase with the ISS. The frequency of the inner orbit is higher than that of the outer one so there is scope for some correction(if required).
- Now the final orbit transfer burn occurs from 200 km to 408 km through the Hohmann transfer orbit(it is an elliptical orbit used to transfer between two circular orbits of different radii around a central body in the same plane. The Hohmann transfer often uses the lowest possible amount of propellant in traveling between these orbits)

REFERENCE: Figure 10 represented below, gives us a detailed view of orbital dynamics implementation in the Crew Dragon - Resilience mission

15. Trajectory Optimization

Rockets are defined by many variables and constraints, and ultimately deliver a payload to orbit at some cost. These characteristics provide the basis for an optimization problem. Maximize J_1 = Payload Mass (metric tons) and Minimize J_2 = Cost. The trajectory subsystem takes in several inputs and calculates the fuel usage and final altitude via the shooting method. It uses the ODE with a state vector composed of radial position, radial velocity, longitude, angular velocity, and mass. The model calculates the changes in velocity using the thrust, gravity, and drag applied at the correct angles. So, in overall define state vector being :

$$\mathbf{X} = [\ \mathbf{r}\ \theta\ v_r\ v_t\ \mathbf{m}\ ]$$

where, $\mathbf{r}$ = geocentric distance, θ = right ascension (Angular Displacement from launch pads initial position), $\mathbf{v}\ r$ and $\mathbf{v}\ t$ being radial and transverse velocity components. For position vector, we define (**ECI**) $Earth-centredinertial$ coordinate system while velocity vector being in (**LVLH**) $Local-Vertical-Local-Horizontal$ frame.

The resulting equations of motion $\dot{x} = f(x, u, t)$ are:

$$\frac{d}{dt} r = \dot{r} \tag{6}$$

$$\frac{d}{dt} \theta = \dot{\theta} \tag{7}$$

$$\frac{d}{dt} \dot{r} = -\frac{\mu}{r^2} + r\dot{\theta}^2 + \frac{T-D}{m} \cos(\alpha) \tag{8}$$

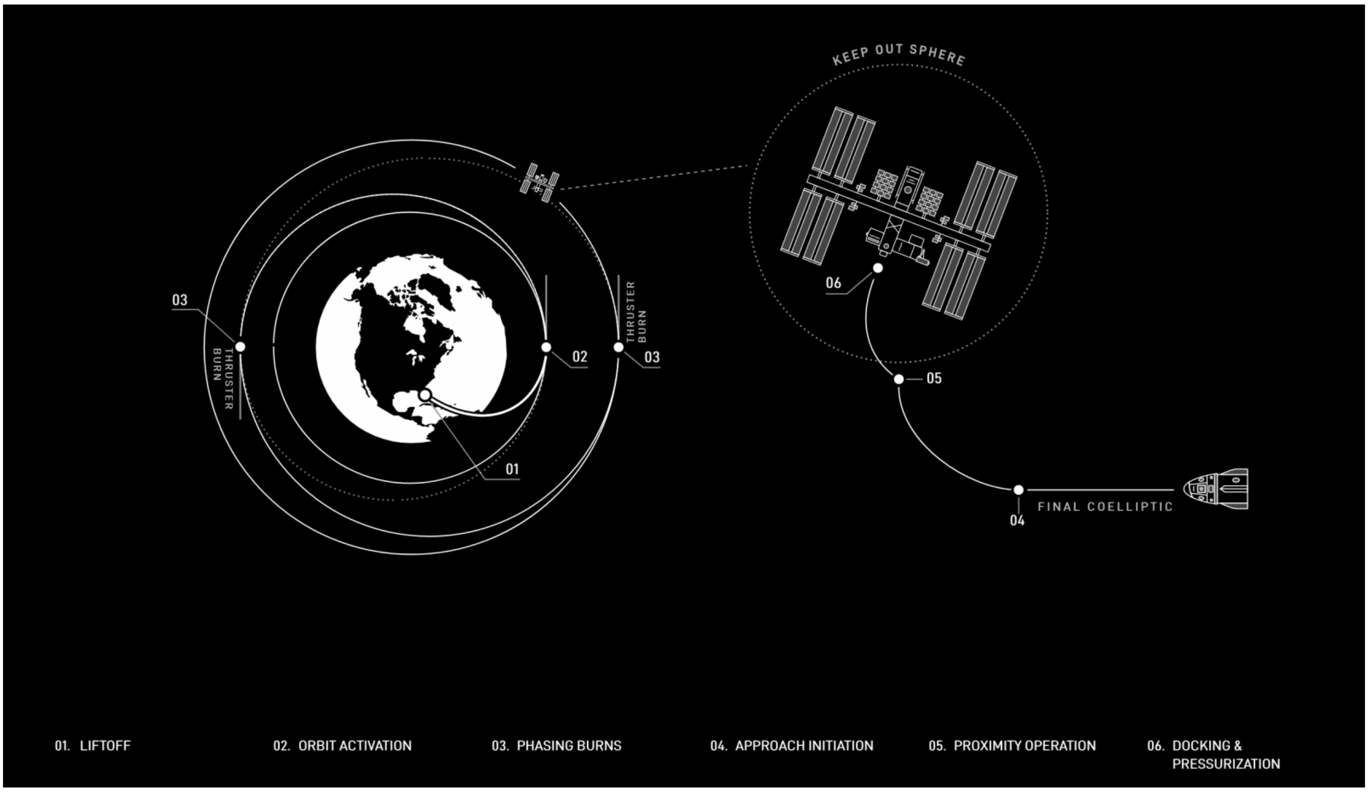


Fig-10: Orbital Dynamics Representation

$$\frac{d}{dt}\dot{\theta} = \frac{T - D}{r * m}\sin(\alpha) \tag{9}$$

$$\frac{d}{dt}m = -\frac{T}{I_{sp} * g_0} \tag{10}$$

All these equations can be easily derived by force balance of the rocket in the radial and tangential directions.

Trajectory optimization, which uses gravity as the driving force to steer the rocket into a particular trajectory. During the gravity turn phase of the ascent trajectory the thrust direction is forced to be parallel to the relative velocity. In order to maintain the same equations of motion across all phases, the thrust magnitude, T is fictitiously split into two attributes, T_a and T_b. T_a represents the optimally controlled thrust contribution, while T_b is always parallel to the relative velocity. It can be noticed that T_a and T_b are alternatively null: during the zero-lift arcs, T_a is zero, while T_b is equal to the real thrust magnitude; conversely, T_a = T and $T_b = 0$ during the other propelled arcs. It offers two main advantages over a trajectory controlled solely through the vehicle's own thrust:

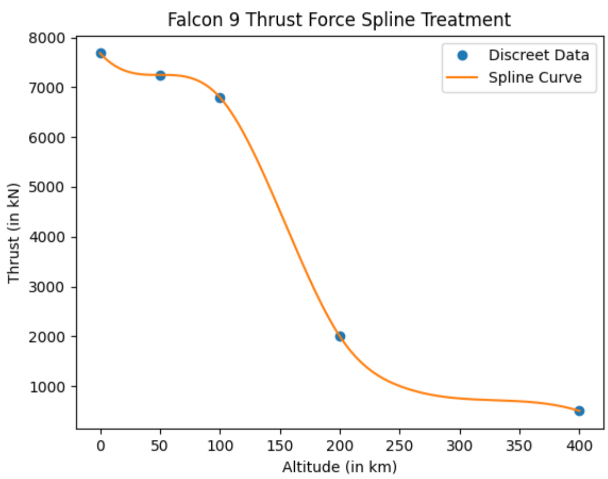
1. The thrust is not used to change the spacecraft's direction, so more of it is used to accelerate the vehicle into orbit.
2. During the initial ascent phase the vehicle can maintain low or even zero angle of attack. This minimizes transverse aerodynamic stress on the launch vehicle, allowing for a lighter launch vehicle.

16. Plotting Curves

16.1. Thrust Force Spline Treatment

As the initial parameters, we are given thrust force generated by the Falcon 9 at 5 different altitudes: 0km, 50km, 100km, 200km, 400km as T_1, T_2, T_3, T_4 and T_5 respectively.

In between these altitudes the thrust is interpolated linearly. However, this approach could be adapted to use a spline instead of a simple linear interpolation. Initially the model used an exponentially decaying thrust, but this did not capture all of the characteristics of typical actual thrust profiles. Here is the spline treatment of the thrust curve of Falcon 9 as per available data:



Graph-1

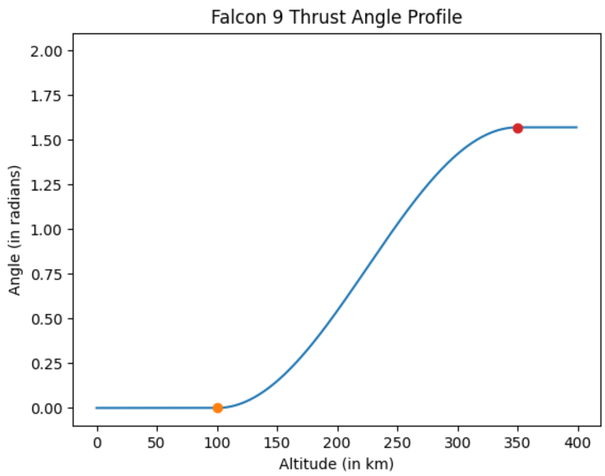
16.2. Thrust Angle

The thrust angle parameter variables α_1 and α_2 define the angle (with respect to a normal from the Earth's surface) of the thrust vector over the course of the trajectory. α_1 is the altitude in km to start turning the rocket, while α_2 specifies the additional altitude over which to complete the turn.

If the altitude is less than α_1 then the angle is zero, and if it is greater than $\alpha_1 + \alpha_2$ then the angle is $\pi/2$. If it is in between then it is defined by:

$$Angle = [1 - cos(\pi * (\frac{A - \alpha_1}{\alpha_2}))] * \frac{\pi}{4} \tag{11}$$

No data was available for altitudes at which Falcon 9 started or ended the gravity turn maneuver, hence, data from the thrust force spline was used to estimate α_1 to be 100km and α_2 to be 250km.



Graph-2

16.3. Drag Force

The drag force on a rocket due to the atmosphere can be simply written as:

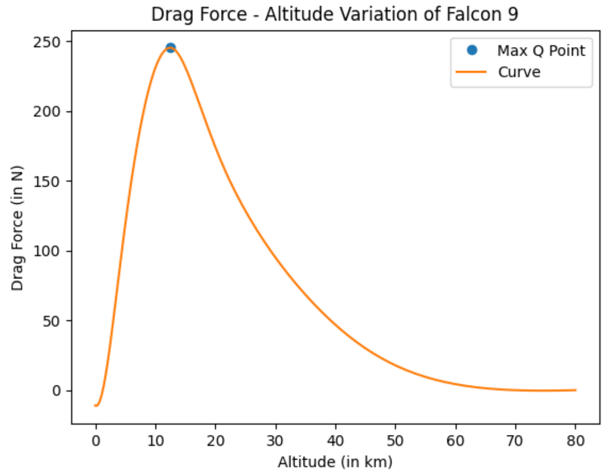
$$D = \frac{1}{2}C_d A \rho v_{rel}^2 \tag{12}$$

where..

C_d	=	Co-efficient of drag = $2\sin^2(\theta_C)$
A	=	Reference surface
ρ	=	Atmospheric Density
v_{rel}	=	Velocity of Falcon 9 w.r.t the atmosphere

We consider the variation of ρ to be via isothermal exponential atmospheric model i.e:

$$\rho = \rho_0 \ exp(-\frac{r - r_0}{H}) \tag{13}$$



Graph-3

16.4. Mass and Cost Calculation

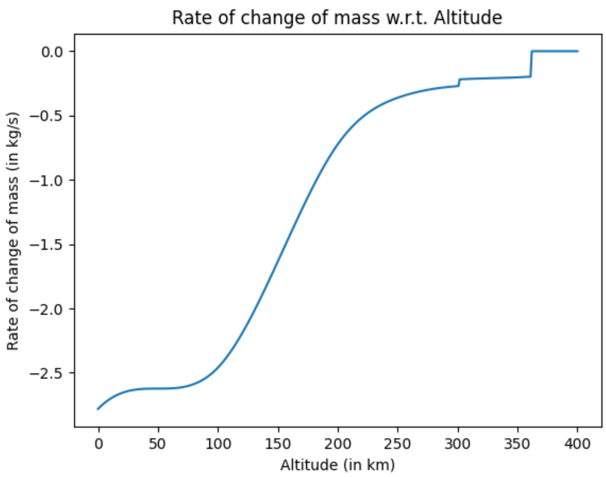
The propulsion subsystem inputs the mass of propellant from the trajectory subsystem, divides this mass up into oxidizer and fuel, and adds an ullage penalty. It also calculates the mass of the engine by scaling the Space Shuttle engine with max thrust according to the following equation:(Engine nozzle efficiency was not taken into account at different altitudes.)

$$m_{engine} = T_{max} * \frac{m_{ss}}{T_{ss}} \tag{14}$$

The cost subsystem calculates the cost for both materials and manufacturing. The material costs are based on material masses and engine mass. The engine is the largest of the material dry masses, and has the highest cost per kilogram, so it makes up the bulk of the material cost. The manufacturing cost is based on seam lengths. The cost parameters include cost per meter of seam and cost per kg of material. These parameters were taken from an external fuel tank model and have been scaled to produce numbers in the expected amounts.

Finally, the costs are summed and the payload mass is calculated according to following equation. Since the wet mass was an input, the mass that was not used up as fuel or taken up by structures is the available payload mass.

$$m_{payload} = m_{total} - m_{structural} - m_{oxidizer} - m_{fuel} \tag{15}$$



Graph-4

The abrupt change in rate of change of mass was expected since at an altitude of 300km, first stage separation was successfully completed and the second stage was fired. Hence, there was an abrupt change in I_{sp} value. At at altitude of 360km, we again observe a sudden jump. This is due to the fact that the second stage separation is also completed successfully. After that, we observe that there is virtually no change in mass. This is because the capsule resilience has now enough velocity to achieve an intersection of orbit with the ISS. Only minor fuel bursts are required for finer maneuvering, like docking with the ISS.

17. MOGA Modelling

Multi-Objective Genetic Algorithm, or MOGA, for short, is one of the most widely used algorithm for multiple variable optimization, in this case, maximizing payload capacity (J_1) while simultaneously minimizing the launch cost (J_2).

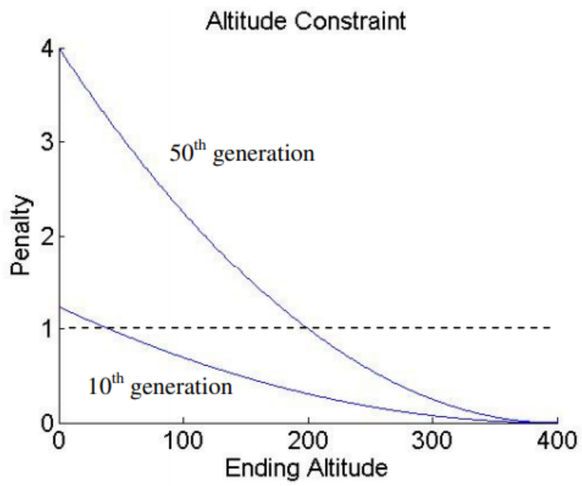
17.1. Algorithm:

1. **Choosing Design:** First, we choose multiple possible designs that could lead to a potentially optimized design. These designs are generated randomly and without any constraints.
2. **Populate:** We then proceed to populate the pareto front (the graph) using the data from these designs.
3. **Optimization:** Next, to optimize the data, we give a penalty to each design that dominated it by having a lower cost and higher payload capability. We also gave a penalty if the ending altitude was less than 400 km. The fitness was then squared to increase the gap between the more and less dominated designs. Finally, we give zero fitness for designs that were otherwise infeasible.
4. **Next Generation:** This fitness value was then used to decide which designs carried on to the next generation of the genetic algorithm. The fitness function for a feasible point is shown in the following slide:

17.2. Fitness Calculation

$$F = max(1.0 - 0.01 * n_{dom} - p(A_{final}), 0)^2 \tag{16}$$

A variable penalty shown below was used for the altitude constraint. The further the constraint was violated, the more severe the penalty applied. The penalty curve steepened with each generation. This is because a low curve would not penalize the infeasible designs enough, but a high curve would often cause the entire starting population to have zero fitness. By starting with a low curve and raising it, the MOGA was able to find the largest number of feasible designs. Example penalty curves from the 10th and 50th generations are shown here. If the penalty is greater than 1, then the fitness bottoms out at 0. This means that a penalty above the dotted line in the shown figure would lead to an infeasible design.



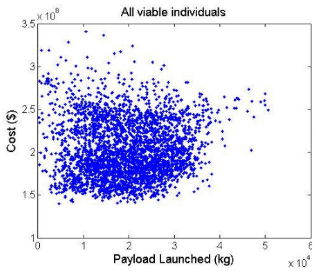
Graph-5

$$p(A_{final}) = [(400 - A_{final})/max(1400 - 4 * generation)]^2 \quad (17)$$

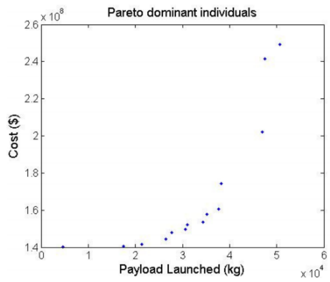
where n_{dom} = no. of times an individual is dominated on the pareto front and A_{final} = Ending Altitude

17.3. Populating Pareto Fronts

The pareto front is populated by all the test cases that passed the fitness function. As we can see, a cluster is formed. Now, we need to choose the non-dominated individuals i.e. those points that have a better payload capacity and lower cost than their peers. (Refer the figure shown below for visual representation)

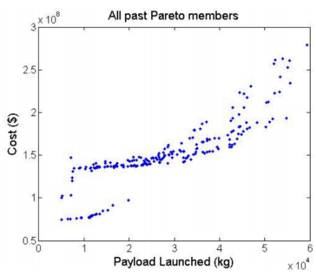


Graph-6

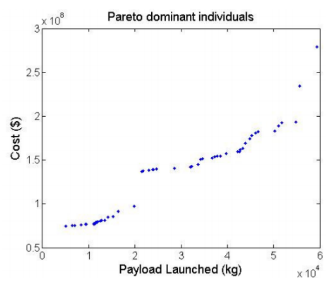


Graph-7

The above process was repeated 10 times to get 10 different pareto fronts. Then all these scatter plots were merged and the dominated individuals were rejected due to the exact same reason and the dominant ones were kept/ selected. Below is the graphical depiction of the same:

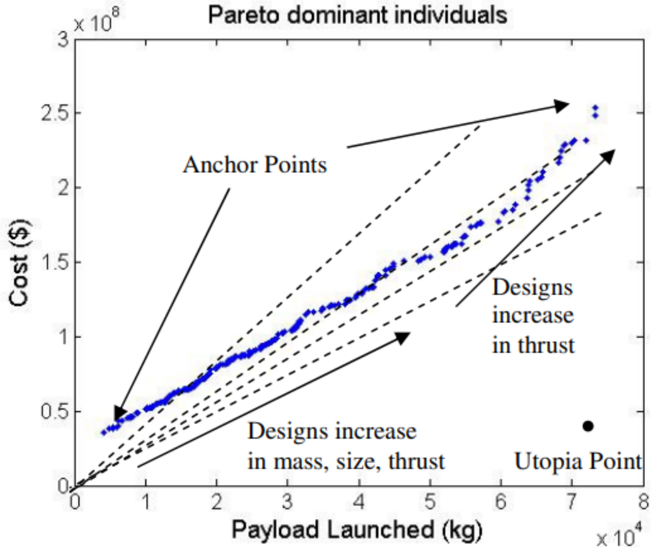


Graph-8



Graph-9

As we can clearly see, the latest graph is not continuous and this was expected, since we were dealing with random point generation. In order to get a smooth curve, more computation was required i.e. much more fronts were needed. So, instead of repeating the process 10 times, we did it 1640 times and obtained a pretty smooth curve of dominant individuals:

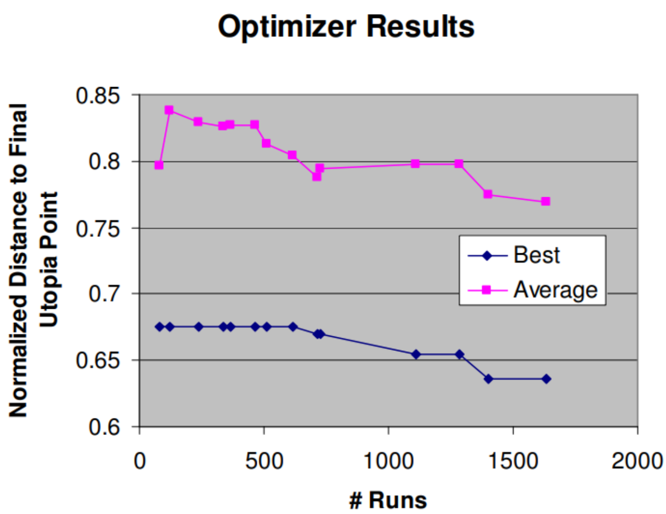


Graph-10

The Pareto front is very linear, which is likely a result of the cost model being simple and heavily driven by engine size. We can also see the low cost and high payload mass anchor points. From those anchor points we can construct a normalized space with vertices (0,0), (1,0), (0,1), and (1,1). (0,0) is the low cost anchor point, (1,1) is the high payload anchor point, and (1,0) is the normalized utopia point. From this utopia point we can find the closest design, hereafter referred to as the "best" design.

17.4. Utopian Point

The following figure shows the normalized distance to the final utopia point as an increasing number of runs was performed. The Average line shows the progression of the average distance of each non-dominated point, while the Best line shows the distance of the closest point. The average distance tended to fall a little at a time, while the best distance tended to fall in jumps, which suggests a greater degree of randomness.



Graph-11

17.5. Comparision with actual rocket

Here we have summarized the rocket model that we generated from our optimization process, it's comparison to the actual Falcon 9 rocket and the error margin. As you can see, MOGA Modelling generated a pretty low error percentage which is in the acceptable range.

Data	Calculated	Actual	Error (%)
Mass (in kg)	6144.7	6000	2.41
Cost (in million $)	64.59	62	4.31
Mass/ Cost	95.13	96.77	1.69

18. Aerodynamics

Shock wave is a propagating disturbance which is formed when a pressure front moves at supersonic speeds (Speed more than that of sound) and pushes the air surrounding it and is characterized by an abrupt, nearly discontinuous, change in pressure,temperature, and density of the medium. Shock waves are mainly comprised of three types-

- Oblique shock waves
- Normal shock waves
- Crossed shock waves

19. Mach Number

The Mach number is the ratio of the speed of the rocket to the speed of sound .

$$M = \frac{u}{c} \tag{18}$$

..where u is the speed of rocket and c is the speed of sound

For M<1,the flow is said to be subsonic.
For M=1,the flow is said to be transonic
For 1 < M < 5,the flow is said to be supersonic
For M>5,the flow is called hypersonic.

20. Max - Q

Max-Q is a point when an aerospace vehicle's atmospheric flight reaches maximum dynamic pressure. This is a significant factor in the design of such vehicles because the aerodynamic structural load on them is proportional to dynamic pressure. Dynamic pressure, q, is defined mathematically as

$$q = \frac{1}{2}\rho v^2 \tag{19}$$

..where ρ is the local air density, and v is the vehicle's velocity

It can be thought off as equivalent to kinetic energy density of air with respect to the vehicle. For a launch of a rocket from the ground into space, dynamic pressure is:

- Zero at lift-off, when the air density ρ is high but the vehicle's speed v = 0
- Zero outside the atmosphere, where the speed v is high, but the air density $\rho = 0$
- Always non-negative, given the quantities involved

During the launch, the rocket speed increases but the air density decreases as the rocket rises. Comprehending by Rolles theorem,there is a point where the dynamic pressure is maximum.

Before reaching max q, the dynamic pressure change because of the increasing velocity is greater than that happening due to the decreasing air density so that the dynamic pressure which opposes kinetic energy acting on the craft increases. After passing the value of max q, the converse happens. The dynamic pressure acting against the craft decreases as the air density decreases, eventually reaching zero when the air density becomes zero.

The LEO mission had it's max q reported at 67 seconds after the liftoff and at an altitude of 12.57 kilometres. For the graph of drag vs altitude,refer Trajectory optimisation..

SpaceX has designed Falcon 9 in a such a way that it's horizontal manufacturing, processing and integration reduce work at height during various integration, processing and manufacturing procedures, and eliminates numerous overhead operations. The side-boosters restraint and release and the separation systems use pneumatic devices thereby providing low-shock release and positive force separation over a comparatively long stroke. Four events happening during flight result in loads that are characterized as shock loads are as follows:

1. Release of the launch vehicle hold-down at liftoff.
2. Stage separation.
3. Fairing deployment.
4. Spacecraft separation.

Events 1 and 2 are negligible for the payload relative to fairing deployment and spacecraft separation because the shocks will travel and dissipate because of the large distance and number of joints. The maximum shock environment predicted at the 1575-mm interface for fairing deployment is enveloped by the shock environment from typical spacecraft separation. As a result of which, the shock environment is a function of the spacecraft adapter and separation system selected for the mission. However, the actual shock environments experienced by the payload at the top of the mission-unique payload adapter are determined after the selection of a specific payload adapter and separation system.

21. Re-Entry

Re-entry of a spacecraft is the most important part of any mission especially when it is returning back from a mission with the crew members. Similarly in the case of Crew Dragon C207 it splashed down in the Gulf of Mexico, off the coast of Panama City, Florida, at 2:57 AM EDT (06:57 UTC) on Sunday May 2, marking the end of the first of six contracted, long duration, operational missions for SpaceX as a part of NASA's Commercial Crew Program. Resilience undocked from the International Space Station (ISS) at 8:30 PM on Saturday May 1 (00:30 UTC on May 2) to begin the journey home. Its return marks the end of Expedition 64 and the start of Expedition 65, with JAXA Astronaut Akihito Hoshide becoming the commander of the ISS. He's the second Japanese astronaut to command the station, the first being Koichi Wakata during Expedition 39.Previously scheduled for Wednesday, April 28, and Friday, April 30, the Crew-1 undocking and splashdown was postponed due to unfavorable weather in the splashdown zones off the coast of Florida. The new schedule for Sunday morning is the first night time splashdown of a crewed American spacecraft since Apollo 8 in 1968.

The last time NASA and SpaceX returned astronauts from the ISS was for the historic Demo-2 mission. Since this is the first return and recovery of a fully operational crewed mission, there have been several lessons learned from the Demo-2 test flight which were implemented for Crew-1. After post-flight inspections of Crew Dragon Endeavour, teams noticed greater than expected erosion of Dragon's heat shield at four points where the capsule bolts to the trunk of the vehicle using tension ties. SpaceX stated that the erosion was likely caused by airflow phenomena that were not expected to occur.

The heat shield design was changed to include more erosion-resistant materials at the ties. The heat shield was reinforced, and tested both on the ground and in-flight during the Cargo Dragon CRS-21 flight.

In addition, the drogue parachutes on board the Endeavour spacecraft deployed lower than expected, although it was still within the allowable range. A new instrument — which measures barometric pressure — was added to determine the altitude for parachute deployment and resolve this issue. "We made changes to the design and part of the heat shield and drogue chute deploy algorithm," said Nicole Jordan, NASA Mission Manager for Crew-1 in an interview with NASASpaceflight. "Fortunately, those changes were made on CRS-21 Cargo Dragon first, so we've actually not only tested it on the ground but also validated those changes in CRS-21. They've both worked as intended, but that is something we'll see for the first time with the crew onboard on Crew-1 landing."

NASA and SpaceX teams have designated seven splashdown zones for Crew Dragon. This includes four sites at the Gulf of Mexico, namely Pensacola, Panama City, Tallahassee, and Tampa, and three sites in the Atlantic Ocean: Jacksonville, Daytona, and Cape North.

Two weeks prior to the return, teams select the primary and the alternate landing sites, pending weather conditions. For the Crew-1 mission, the selected primary splashdown site was Pensacola, with Panama city being the alternate location, both located in the Gulf of Mexico, off the coast of Florida. The Panama City landing zone was selected for splashdown on Sunday morning.

Additionally, a backup unsupported landing site (outside of the seven sites mentioned earlier) with suitable weather conditions is also identified to mitigate the risk of weather changes and ensure a minimum of two landing sites are identified at all times. In the unlikely scenario this site is used, the rescue operations will be conducted by the U.S. Department of Defense. The images below are of spacex crew dragon reilience before undocking from the ISS and after it lands safely on earth.

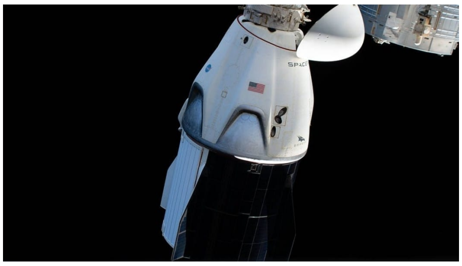


Fig-11: Docked with ISS

Fig-12: Splashdown!

22. Heat Shields

Heat shields are a cool or rather, hot piece of technology. They are a relatively simple solution for any extreme problem. The problem being when a spacecraft reenters the atmosphere 10 times faster than a bullet, it must cope with temperatures half as hot as the surface of the sun.

NASA and SPACEX collectively played an important role in making of heat shields for crew dragon resilience. The features of crew dragon were improved so as to include 4 member in the mission and also remain in the space for 210 days. The spacecraft also features an improved backshell that will increase the wind limits for reentry. Most heat shields, the Dragon Capsule included, use an ablative material. These work by ablating, or flaking away, as they heat up. Thereby taking some of the heat away with them. Meaning, you could even use wood as a heat shield per se.

Both the old and new Dragons use PICA-X, which is SpaceX's variant of the NASA designed Phenolic Impregnated Carbon Ablator. The material, engineered in the '90s, has been used on Mars missions and the Orion spacecraft. The heat shield, in conjunction with the thrusters, can precisely steer the spacecraft through re-entry.

Dragon is engineered such that the centre of mass is offset, allowing for the heat shield to produce lift. Through careful control, the lift generated can be exploited to guide the vehicle. By rotating the spacecraft, the lifting vector changes. Therefore, if turned in one direction, Dragon pulls up from the normal vector. Conversely, if rotated 180 degrees, it will pull down deeper into the atmosphere. The image given below is of heat shield used in crew dragon

Fig-13: Heat Shield

23. Parachutes

The final, and perhaps most important, piece to the mission is the parachutes. After all, without chutes, an otherwise successful mission would certainly lead to the loss of craft and crew members.

Parachutes, although they seem a simple technology, have proven to be one of the most challenging components. You'd think such an old field would be completely explored by now. NASA and its commercial partners, Boeing and SpaceX, through rigorous testing and certification for the Commercial Crew Program, found new errors and failure modes.

As NASA administrator Jim Bridenstine mentioned in a 2019 interview a reporter had with him: "We were taking risks with other parachutes that we did not know we were taking… because we have done testing, we now know what we didn't know before."


Fig-14: Parachutes

It took SpaceX about 100 failed tests and three generations of chutes to certify them completely. The most recent, inventively designated Mk3, managed to pass the stringent safety requirement for the mission.The Mark3 design itself went through 27 drop tests before it was considered safe for human use. As a result of the huge testing campaign, SpaceX now has one of the safest and most reliable parachutes ever made in the world. The biggest challenge was that parachutes had very narrow window for operation. If you're going too fast and try to deploy a chute, the airstream can destroy the canopy, rip the lines or their attachment points to the canopy.

SpaceX started using Zylon, a unique polymer, in place of the nylon previously used in other parachute. The risers, which attach the canopy lines to the capsule, had to be bolstered when a load modelling inaccuracy proved that they did not meet the human safety factor . After all, as the chutes initially deploy and reef open, there can be very big shocks. The trick is developing a chute that deploys slowly and smoothly, doesn't tangle, and most importantly, doesn't fall apart.

24. Landing

At an altitude of 2 km, Crew Dragon deploys its four main parachutes. The vehicle has undergone "parachute out" testing, meaning it can safely splashdown under only three chutes. This is quite the change from the original Dragon capsule, which only had three chutes to begin with. But all these systems have to work in unison for a successful mission. If the shape of the capsule were designed wrong, it wouldn't slow down enough in the upper atmosphere, and its terminal velocity could be much higher, resulting in harsher landing conditions for the parachutes or less time for them to deploy fully.

It's the entire system that makes re-entry safe, not just any one particular component. The depth of thought that has to go into each and every aspect of the vehicle to safely return from orbit is amazing.

25. Communication

25.1. Ground Communication

A ground station is a terrestrial radio station designed for extraplanetary telecommunication with the spacecraft or reception of radio waves from astronomical radio sources. Ground stations may be located either on the surface of the Earth, or in the atmosphere. Earth stations communicate with spacecraft by transmitting and receiving radio waves in the super high frequency (SHF) or extremely high frequency (EHF) bands (e.g. microwaves). When a ground station successfully transmits radio waves to a spacecraft (or vice versa), it establishes a telecommunications link. A principal telecommunications device of the ground station is the antenna.

Ground stations may have either have a fixed position or it may change it's position according to the convenience. Article 1 of the International Telecommunication Union (ITU) Radio Regulations describes various types of stationary and mobile ground stations, and their interrelationships.

Specialized satellite Earth stations are used to telecommunicate with satellites — chiefly communications satellites. Other ground stations communicate with crewed space stations or uncrewed space craft. A ground station that primarily receives telemetry data, or that follows space missions, or satellites not in geostationary orbit, is called a ground tracking station, or space tracking station.

When a spacecraft or satellite is within the ground station's line of sight, the station is said to have a view of the spacecraft . A spacecraft can communicate with more than one ground station at a time. A pair of ground stations are said to have a spacecraft in mutual view when the stations simultaneously share line-of-sight contact with the spacecraft.

Dragon supports communications via satellites such as NASA's Tracking and Data Relay Satellite System, but it is also capable of communicating via Ground Stations on Earth. Data Rates are 300kbps for Command Uplink and 300Mbps or more for telemetry and data downlink. Payloads on the Vehicle can be integrated via standard communication interfaces like Ethernet or RS-422 and 1553 standards. Redundant telemetry and video transmitters in S-Band helps accomplish vehicle communication. Dragon is equipped with on-board compression as well as encryption/decryption systems.

25.2. Communication with space station

As Dragon chases the station, the spacecraft will establish UHF communication using its COTS Ultra-high-frequency Communication Unit (CUCU). Also, using the crew command panel (CCP) on board the station, the space station crew will interact with Dragon to monitor the approach. This ability for the crew to send commands to Dragon will be important during the rendezvous and departure phases of the mission.

During final approach to the station, a go/no-go is performed by Mission Control in Houston and the SpaceX team in Hawthorne to allow Dragon to perform another engine burn that will take it 250m (820 feet) away from space station. At this distance, Dragon will starts using its close-range guidance systems, composed of LIDAR (light radar) and thermal imagers. Then these systems will confirm that Dragon's position and velocity are accurate by comparing the LIDAR image that Dragon receives against Dragon's thermal imagers. Using the Crew Command Panel, the ISS crew, monitored by the Dragon flight control team in Hawthorne and the NASA flight control team at the Johnson Space Center's International Space Station Flight Control Room, will command the spacecraft to approach the station from its hold position.

After another go/no-go is performed by the Houston and Hawthorne teams, Dragon is permitted to enter the Keep-Out Sphere (KOS), which is an imaginary sphere drawn 200 meters (656 feet) around the station within which the Dragon approach is monitored very carefully so as to minimize the risk of collision. Dragon will proceed to a position 30 meters (98 feet) from the station and will automatically hold. Another go/no-go is completed. Then Dragon will proceed to the 10-meter (32 feet) position that is the capture point. A final go/no-go is performed, and the Mission Control Houston team will notify the crew they are go to capture Dragon.

- Communications between Dragon and the ISS are provided by the COTS UHF communications unit (CUCU) which was delivered to the space station on STS-129.
- Crew command panel (CCP) was used for SS crew command dragon.

– Dragon can also communicate on S-band via either tracking data relay system (TDRSS) or ground stations.

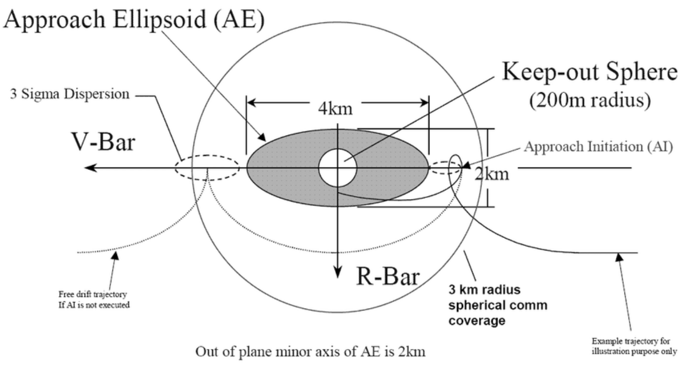


Fig-15: Keep Out Sphere

References

[1] SpaceX page for Dragon Capsule: https://www.spacex.com/vehicles/dragon/
[2] Falcon 9 Wikipedia: https://en.wikipedia.org/wiki/Falcon_9
[3] Altitude and Drag Force variationof Falcon 9 w.r.t. Time: https://forum.nasaspaceflight.com/index.php?action=dlattach;topic=40983.0; attach=1668216;image
[4] MIT Paper on MOGA Modelling: http://strategic.mit.edu/spacelogistics/pdf/AIAA-2006-1720-231.pdf
[5] E. Seedhouse, SpaceX: Making Commercial Spaceflight a Reality, DOl 10.1007/978-1-4614-5514-1_3, © Springer Science+Business Media New York 2013
[6] "Space Exploration Technologies". Published by the American Institute of Aeronautics and Astronautics, Inc., with permission. (2008)
[7] Braun, R. D., Powell, R. W., Lepsch, R. A., Stanley, D. O., and Kroo, I. M., “Comparison of Two Multidisciplinary Optimization Strategies for Launch Vehicle Design,”Journal of Spacecraft and Rockets, Vol. 32, No.3, May-June 1995, pp. 404-410.
[8] https://www.nasa.gov/image-feature/the-spacex-crew-dragon-resilience-before-undocking
[9] Tsuchiya, T., and Mori, T., “Optimal Conceptual Design of Two-Stage Reusable Rocket Vehicles Including Trajectory Optimization,” Journal of Spacecraft and Rockets, Vol. 41, No.5, September-October 2004, pp. 770-778
[10] Schuman, T., “Integration of System-Level Optimization With Concurrent Engineering Using Parametric Subsystem Modeling,” Master’s Thesis, Aeronautics and Astronautics Dept., Massachusetts Institute of Technology, Cambridge, MA, 2004.6.
[11] Ahuja, R. K., Orlin, J. B., Tiwari, A., “A Greedy Genetic Algorithm for the Quadratic Assignment Problem,” Computers and Operations Research, Vol. 27, 2000, pp. 917-934.
[12] U.S. Standard Atmosphere, 1962, U.S. Government Printing Office, Washington, D.C., 1962.
[13] Isakowitz, S. J., Hopkins, J. B., and Hopkins, J. P., Jr., International Reference Guide to Space Launch Systems, 4th ed., AIAA, Virginia, 2004, pp. 109-146
[14] Taylor, C., “Single Stage to Orbit Rocket Model,” Massachusetts Institute of Technology, Cambridge, MA, 2005.
[15] Schoonover, P. L., Crossley, W. A., and Heister, S. D., “Application of a Genetic Algorithm to the Optimization of Hybrid Rockets,” Journal of Spacecraft and Rockets, Vol. 37, No.5, September-October 2000, pp. 622-629.
[16] obieszczanski-Sobieski, J., “External Fuel Tank Model,” AIAA Short Course on Multidisciplinary Design Optimization, NASA Langley Research Center, Hapton, VA, 2002.

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

Mentor: Mubashshir Uddin
Participants:
Sameer Singh
Ananya Priyadarshi
Kajal Deep
Prem Milind Gujrathi
Renaissance Hansda
Yashika Malhotra

July 27, 2021

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.

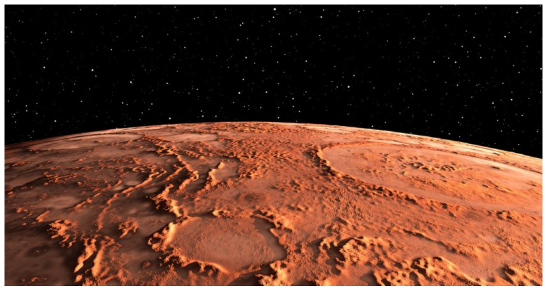


Figure 2: Surface of Mars

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

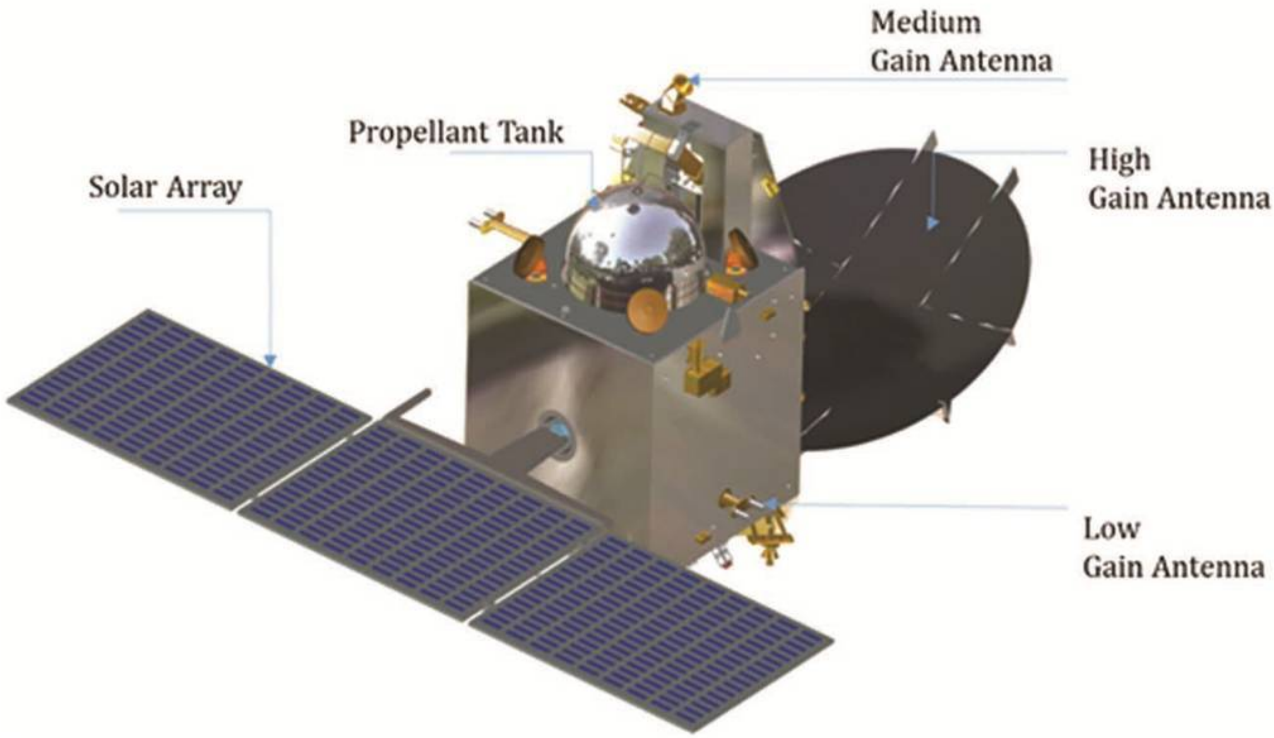


Figure 4: Mars Orbiter Spacecraft

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.

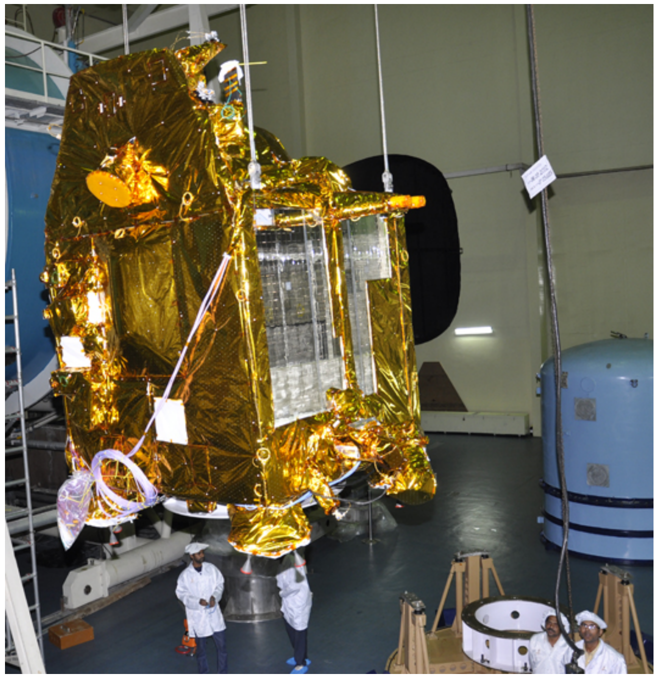


Figure 6: MLI

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.

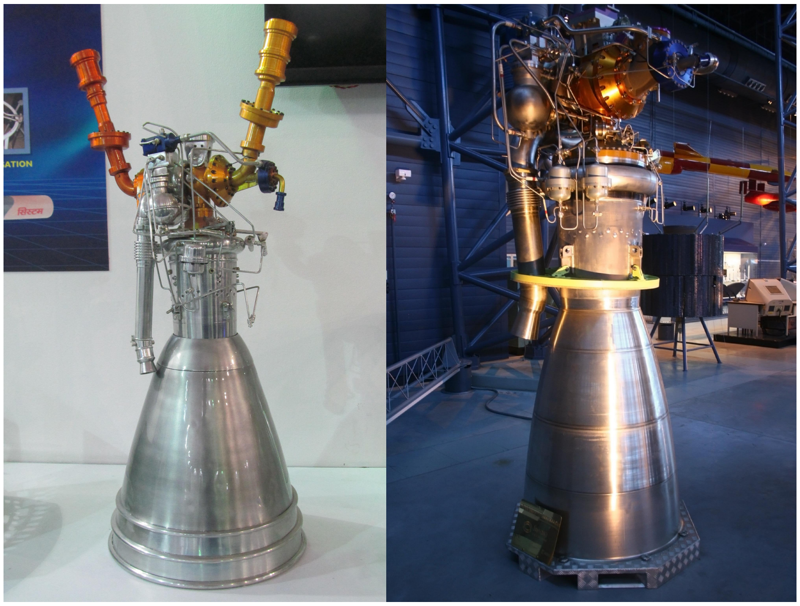


Figure 7: VIKAS engine used in MOM

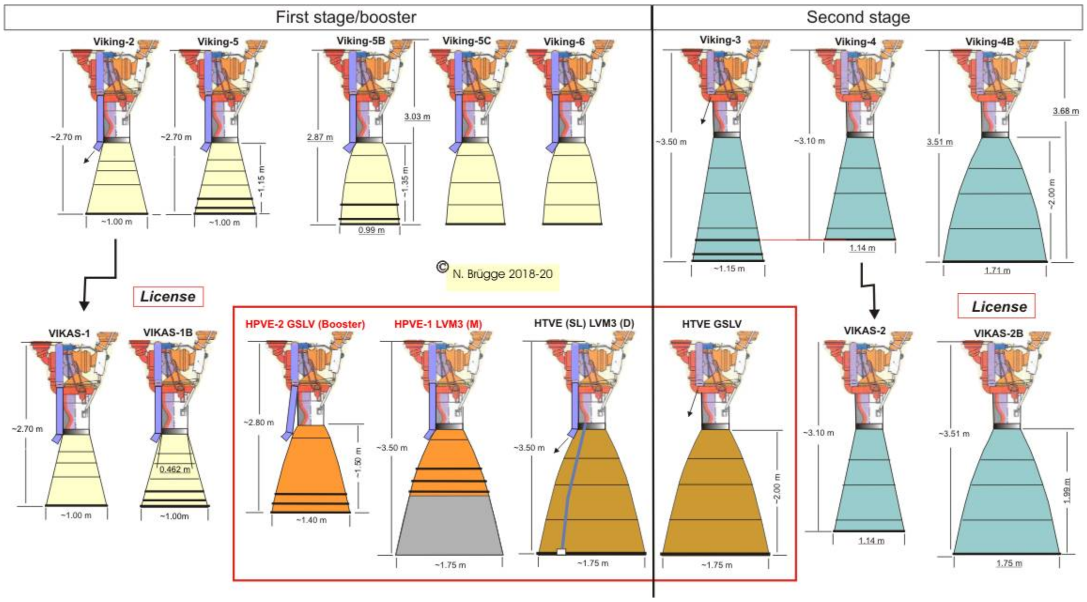


Figure 8: Evolution of the VIKAS engine

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)

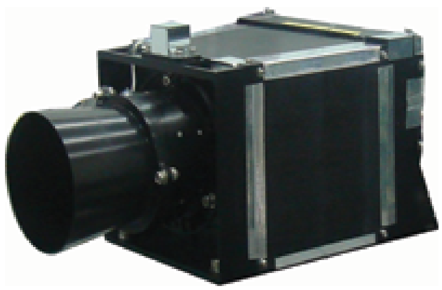


Figure 10: Mars Color Camera.

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)

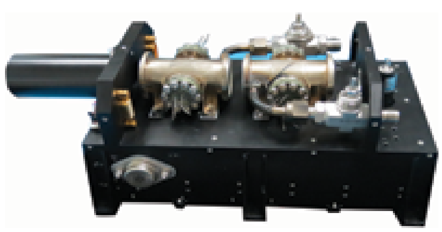


Figure 11: Lyman Alpha Photometer.

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)

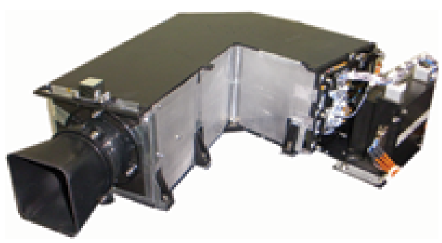


Figure 13: Thermal Infrared Imaging Spectrometer.

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.

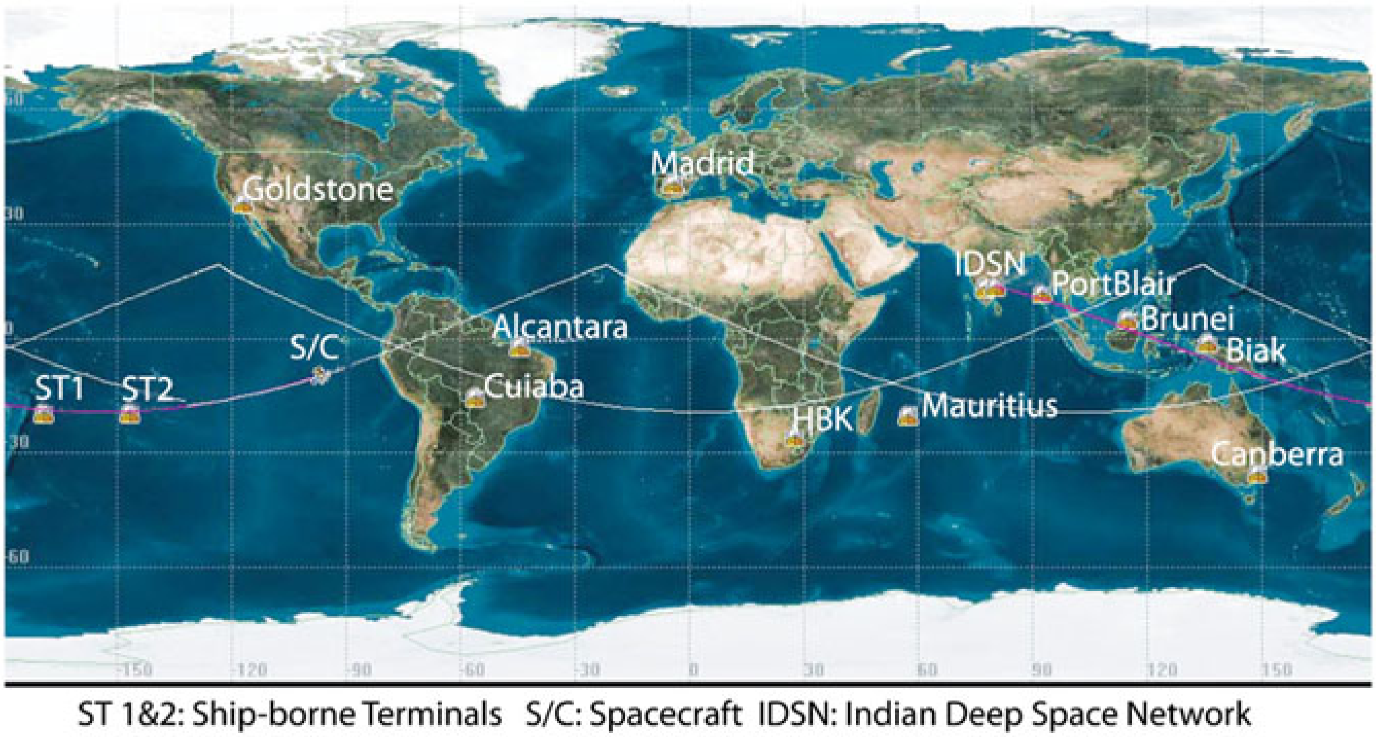


Figure 14:

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.

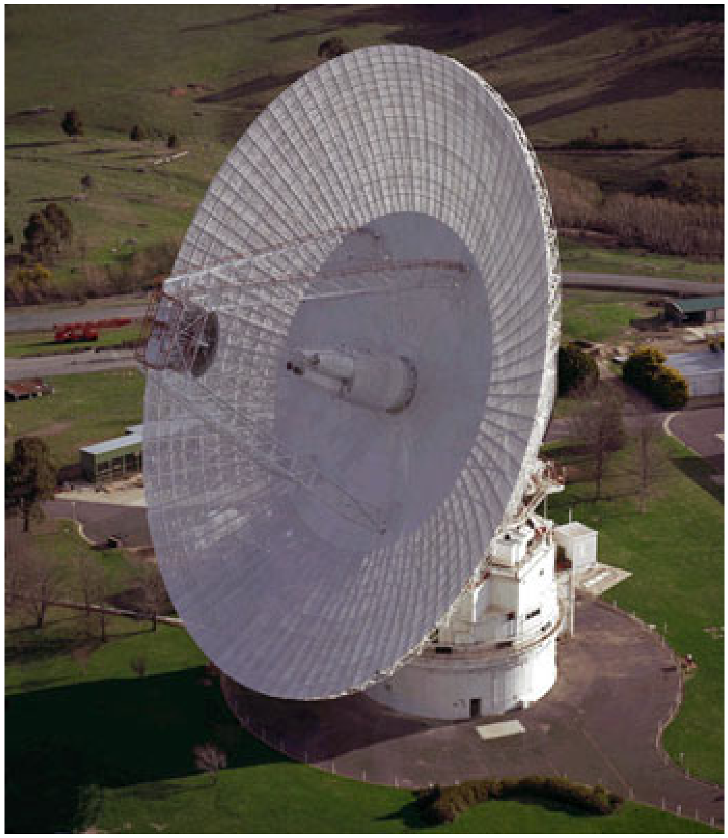


Figure 15:

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)

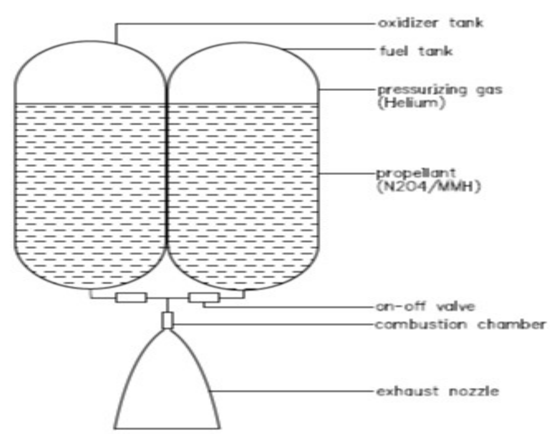


Figure 17: Gas pressurized N_2O_4/MMH liquid propellant rocket engine. [1]

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.

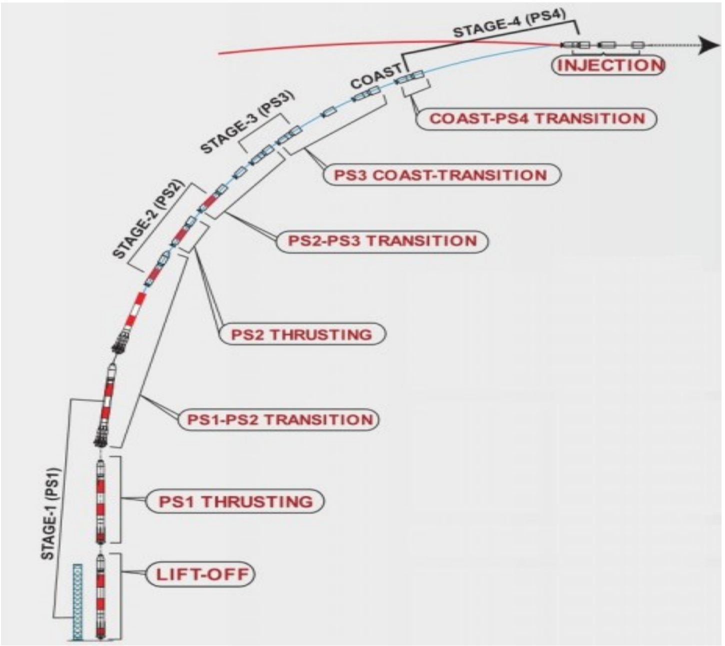


Figure 18: Mission Profile

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.

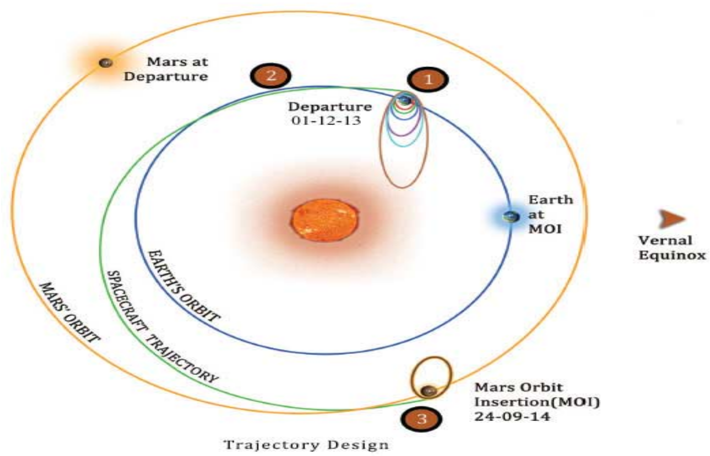


Figure 19: Mission Profile

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.

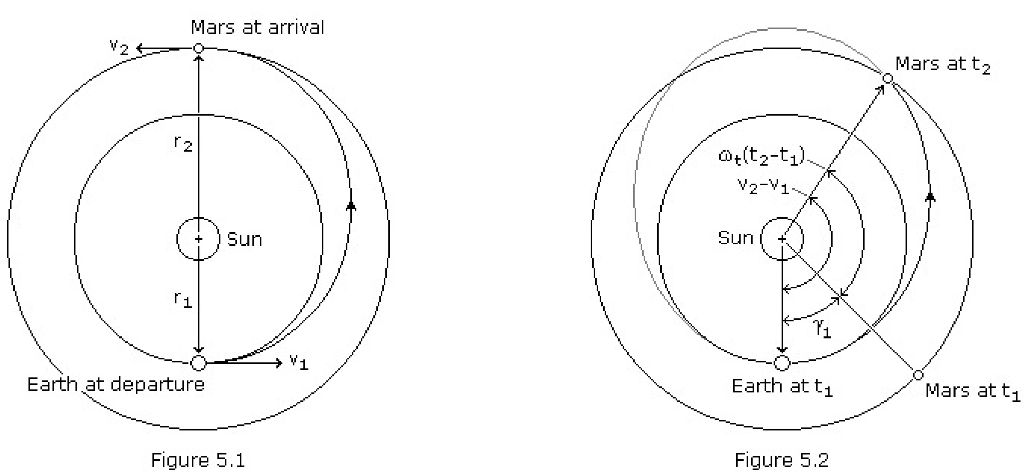


Figure 21: Heliocentric Phase

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.

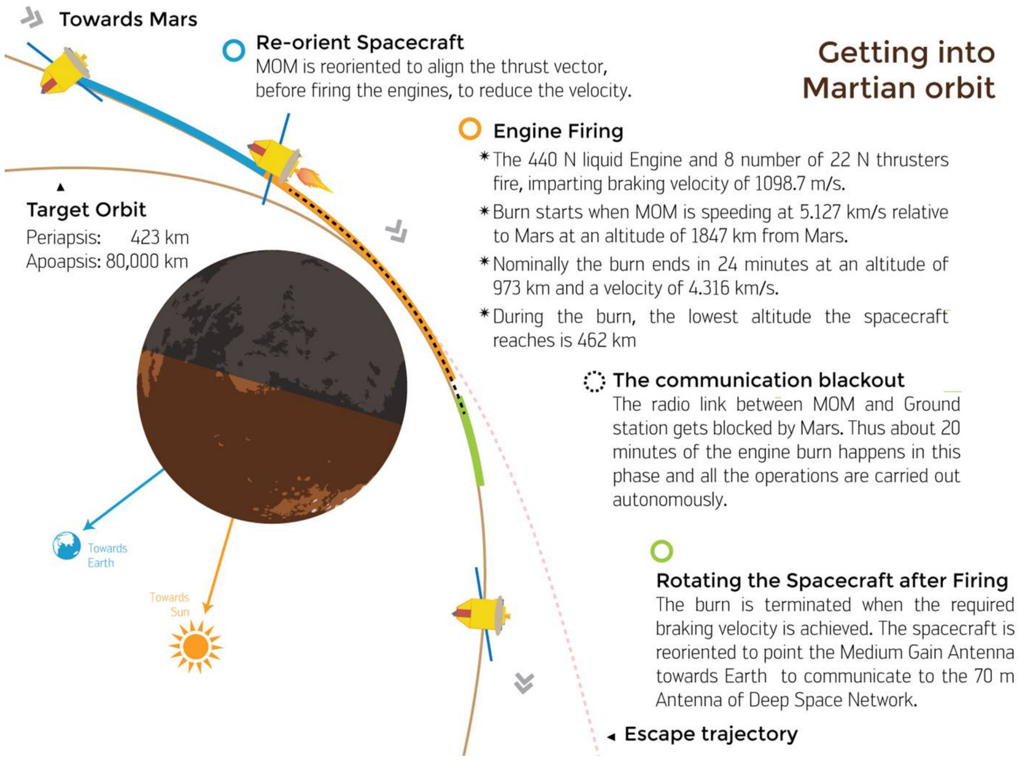


Figure 22: Areocentric Phase

References

[1] Staging: `https://spaceflight101.com/mom/mars-orbiter-mission/`

[2] PSLV-C25: `https://www.isro.gov.in/launcher/pslv-c25`

[3] Mission Profile: `https://www.isro.gov.in/pslv-c25-mars-orbiter-mission/mars-orbiter-mission-pro`

[4] Apogee-raising maneuver: `https://spaceflight101.com/mom/mars-orbiter-mission/`

[5] Areocentric Phase `https://www.isro.gov.in/pslv-c25-mars-orbiter-mission/mars-orbiter-mission-p`

[6] `https://en.wikipedia.org`

[7] `https://www.isro.gov.in/launchers/pslv`

[8] `https://www.sciencedirect.com/`

[9] `https://www.zmescience.com/`

[10] `https://economictimes.indiatimes.com/`

[11] Noor Muhammad Feizal B Muhalim and Subramaniam Krishnan. “DESIGN OF NITROGEN-TETROXIDE/MONOMETHYL-HYDRAZINE THRUSTER FOR UPPER STAGE APPLICATION”.

Space: The Final Frontier

Sample Mission Report (SMR)
Group-1

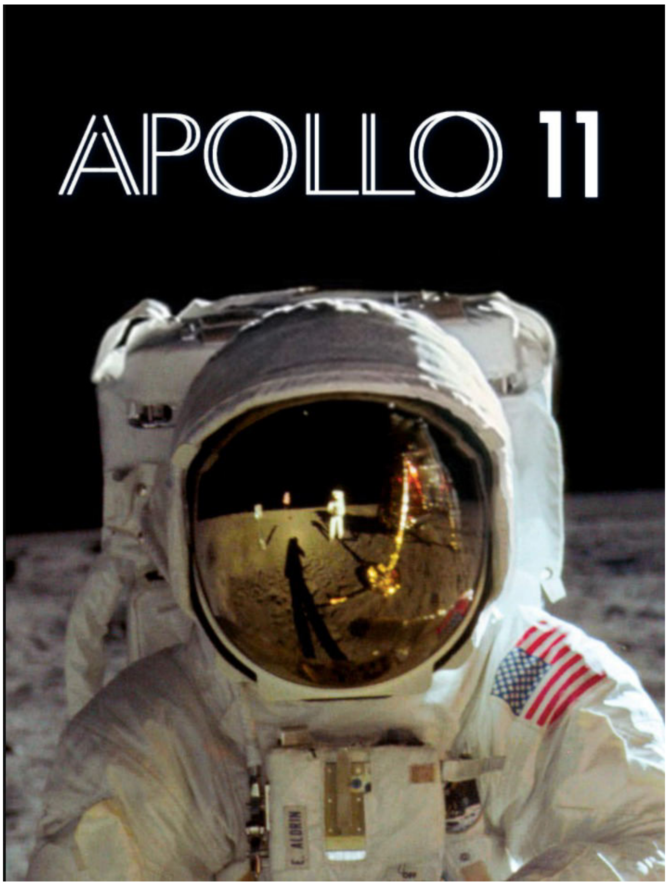
Apollo-11

Anup Singh
Bhavani shankar
Saikrishna
Ratanlal Sahu
Ananya Upadhyaya
Shubhankari Rai

Mubashshir Uddin
Mentor

Keywords: *Apollo Rocket, Moon Missions, Saturn V, Rocket Engines, Lunar Transfer*

INTRODUCTION



Apollo 11 was launched from **Cape Kennedy** on **July 16, 1969**, carrying **Commander Neil Armstrong, Command Module Pilot Michael Collins and Lunar Module Pilot Edwin "Buzz" Aldrin** into an initial Earth-orbit of 114 by 116 miles. An estimated 650 million people watched Armstrong's televised image and heard his voice describe the event as he took "...one small step for a man, one giant leap for mankind" on July 20, 1969.
The primary objective of Apollo 11 was to complete a national goal set by President John F. Kennedy on May 25, 1961: perform a crewed lunar landing and return to Earth.

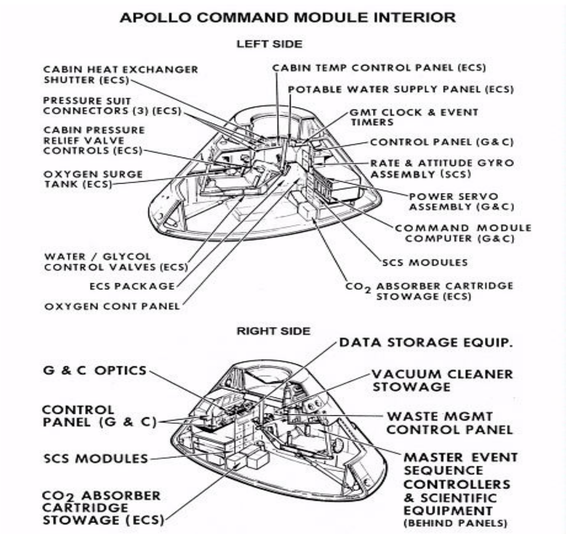
1. ROCKET DESIGN

The Apollo 11 mission primarily had 3 spacecraft:

1.1 Command Module - Columbia

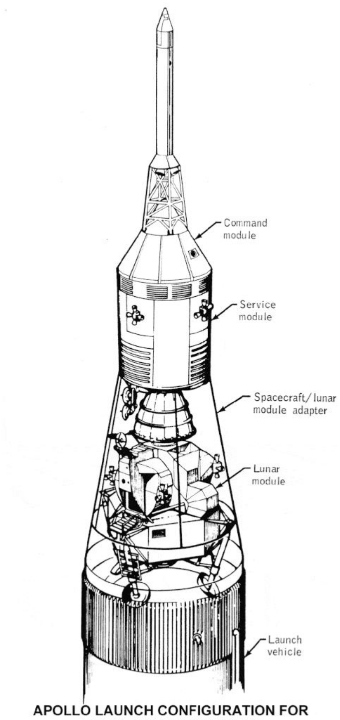
The Apollo 11 Command Module Columbia carried astronauts Neil Armstrong, Edwin Aldrin, and Michael Collins on their historic voyage to the Moon and back on July 16-24, 1969.

During the journey to and from the Moon, Columbia, having an interior space just around greater than an automobile, served as main quarters for the astronauts, a place for working and living. The blunt-end design for the Command Module was chosen to build upon experience gained with the similarly shaped Mercury and Gemini spacecraft.



Command Module Specifications :

- Height: 3.2 m (10 ft 7 in)
- Maximum Diameter: 3.9 m (12 ft 10 in)
- Weight: 5,900 kg (13,000 lb)
- Manufacturer: North American Rockwell for NASA
- Launch Vehicle: Saturn V



APOLLO LAUNCH CONFIGURATION FOR LUNAR LANDING MISSION

1.2 Service Module

A service module (also known as an equipment module or instrument compartment) is a component of a crewed space capsule containing a variety of support systems used for spacecraft operations. Usually located in the uninhabited area of the spacecraft, the service module serves a storehouse of critical subsystems and supplies for the mission such as electrical systems, environmental control, and propellant tanks.
The service module is jettisoned upon the completion of the mission and usually burns up during atmospheric re-entry.

1.3 Lunar Module - Eagle

Because lunar modules were designed to fly only in the vacuum of space, they did not have to be streamlined like an aircraft or carry a heat shield for protection during re-entry. Once a lunar module was launched into space, it could not return to Earth.

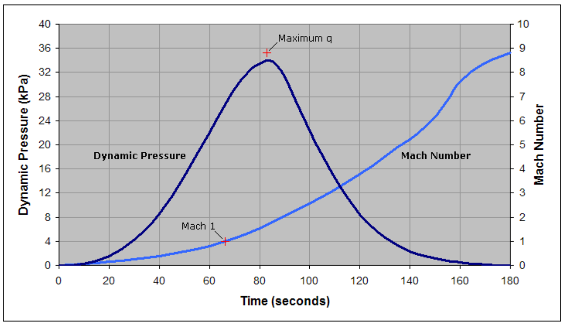
Lunar Module Specifications:

- Weight (empty): 3920 kg (8650 lb)
- Weight (with Crew & Propellant): 14,700 kg (32,500 lb)
- Height: 7.0 m (22 ft 11 in)
- Width: 9.4 m (31 ft 00 in)
- Descent Engine Thrust: 44,316 Newtons (9870 lb) maximum, 4710 Newtons (1050 lb) minimum
- Ascent Engine Thrust: 15,700 Newtons (3500 lb)
- Fuel: 50-50 mix of Unsymmetrical Dimethyl Hydrazine (UDMH) & Hydrazine
- Oxidizer: Nitrogen Tetroxide
- Prime Contractor: Grumman Aerospace Corporation

2. AERODYNAMICS

To be successful in the mission, the rocket must be able to withstand the value of maximum dynamic pressure or Max Q that the rocket experiences during its journey. If it is not able to do so, it results in buckling and the rocket collapsing, which causes a catastrophe.
The dynamic pressure experienced by the rocket is given by:

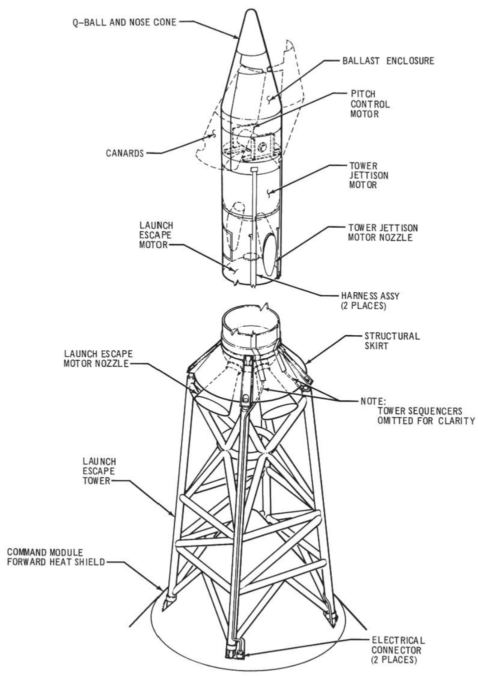
$$q = \frac{1}{2}\rho v^2$$



Suitable materials were used in order to avoid buckling like aerospace grade aluminum and titanium. The nose cone and fins of a rocket were designed to minimize drag (air resistance) and to provide stability and control (keep it pointing in the right direction without wobbling). Heat-resistant nickel and steel alloys were used in order to absorb heat when exposed to the sun and radiate to the blackness of deep space.

3. STRUCTURAL SYSTEM

The launch vehicle, AS-506 (Apollo Saturn), was used and was the fourth human-crewed Saturn V vehicle. The structural system consists of the fairings, cylindrical body, and control fins.
NASA made few modifications to the Apollo 11 space vehicle from the preceding configuration like the Lunar module (LM).



LAUNCH ESCAPE SYSTEM.

4. ABORT SYSTEM AND HEAT SHIELD

The **Apollo Launch abort system (LAS) or Launch escape system (LES)** was designed for the group to escape if a few failures occurred during the preliminary phase of launching.
It contained three solid-propellant rocket motors:

- Launch escape motor
- Pitch control motor.
- Tower jettison motor

The **Launch escape motor** (main motor) in the Abort system had a thrust of 155,000 pounds.
It detaches the command module away from the path of the remaining portions of the launch vehicle if any malfunctioning occurs during launch.
The **Pitch control motor** was essential for establishing a safe trajectory for the launch abort system.

The **Tower jettison motor** was used to separate the launch abort system from the command module before the deployment of the parachutes.
To protect the Apollo Command Module (Columbia) *from the extreme heat throughout re-entry*, NASA selected an **Ablative heat shield AVCOAT** composed of a set metallic honeycomb substructure, a fiberglass honeycomb shell stuffed with phenolic epoxy glue. The material was designed such that it vaporized because of atmospheric friction, which prevents the heat from coming into the group compartment.
The **Lunar Module (LM)** was used for dropping to the lunar surface and served as a base camp while the astronauts were on the Moon for explorations. Several materials cover the spacecraft to defend its internal
structure against temperature and small meteoroids. The sheets soak up the heat when exposed to the Sun and radiate to space.
The Lunar module had two stages:

1. **Ascent stage**, containing the group's compartment that managed the entire spacecraft.
2. **Descent stage**, almost like the ascent stage, containing the rocket engine and tanks for fuel and oxidizer.

The descent (lower) stage contained a rocket motor to slow down the velocity of descent to the lunar surface.
It contained some scientific exploration equipment and remained on the Moon when the astronauts left for further research on the Moon.

Lunar Module Ascent Stage (Control)

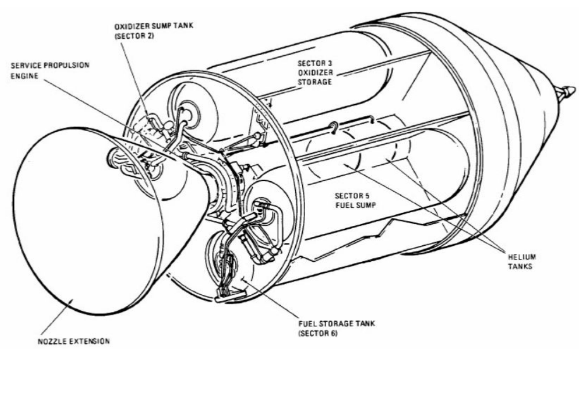
The crew compartment in the ascent stage included some essential things like the life support system, communication and navigation system for astronauts. The Controls and displays for the main engine allowed the crew member to fly the craft.
It contained its own Ascent Propulsion System (APS) engine for return to lunar orbit and a Reaction Control System (RCS) for altitude and translation control.
For rejoining with the Command Module (CM), the astronauts started the ascent-stage engine and lifted off from the Moon. Then the ascent stage docked with the CM in lunar orbit. The ascent stage engine then crashed into the Moon.

5. PROPULSION

Propulsion is the action or process of pushing or pulling to drive an object forward. A propulsion system consists of a source of mechanical power and a *propulsor*(means of converting this power into propulsive force).In case of Apollo-11, Liquid-Propulsion was used, and the service module propulsion system was reignited.

Spacecraft and Subsystems

The Apollo 11 Command and Service Module contains mass of 28,801 kg was the launch mass including propellants and expendables, of which the Command Module (CM 107) had a mass of 5557 kg and the Service Module (SM 107) 23,244 kg.



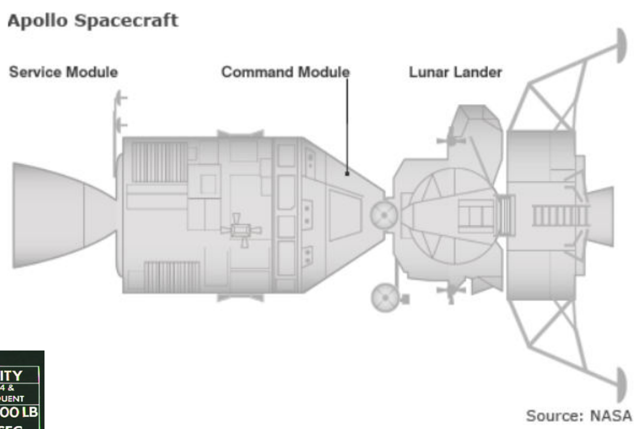
Service Propulsion System (Service Module)

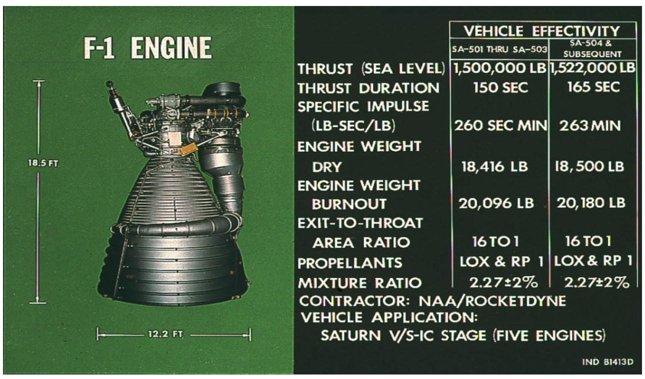
The SM was a cylinder 3.9 meters in diameter and 7.6 m long, which was attached to the back of the CM. The outer skin of the SM was formed of 2.5 cm thick aluminum honeycomb panels. The interior was divided by milled aluminum radial beams into six sections around a central cylinder. At the back of the SM mounted in the central cylinder was a gimbal mounted re-startable hypergolic liquid propellant 91,000 N engine and cone-shaped engine nozzle. Attitude control was provided by four identical banks of four 450 N reaction control thrusters, each spaced 90 degrees apart around the forward part of the SM. The six sections of the SM held three 31-cell hydrogen-oxygen fuel cells, which provided 28 volts, two cryogenic oxygen and two cryogenic hydrogen tanks, four tanks for the main propulsion engine, two for fuel and two for oxidizer, and the subsystems the main propulsion unit. Two helium tanks were mounted in the central cylinder.

6. ENGINE DESIGN

F-1 Engine

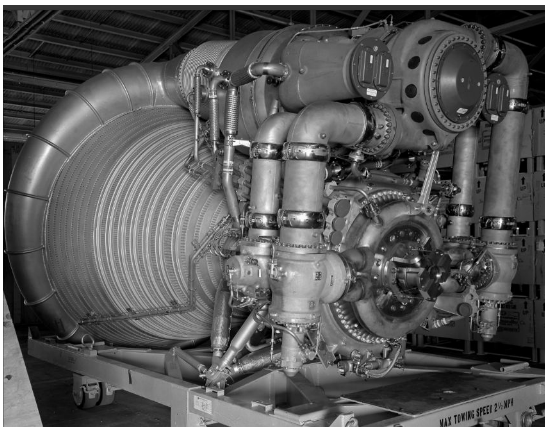
Five F-1 engines were used in the S-IC first stage of each Saturn V, which served as the main launch vehicle of the Apollo-11 mission. The F-1 remains the most powerful single combustion-chamber liquid-propellant rocket engine ever developed. It uses Gas-Generator cycle, which is an open cycle engine, easy to build and operate, but it loses efficiency due to discarded propellants. The F-1 engine is the most powerful single-nozzle liquid-fueled rocket engine ever flown. The F-1 burned RP-1 (rocket grade kerosene) as the fuel and used liquid oxygen (LOX) as the oxidizer.





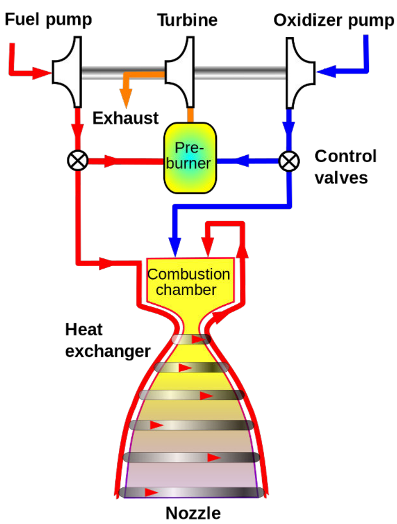
A turbo-pump was used to inject fuel and oxygen into the combustion chamber. Also, the RD-170 produces more thrust but has four

nozzles. One notable challenge in the construction of the F-1 was regenerative cooling of the thrust chamber. There was an issue known as 'starvation' due to hydrodynamic and thermodynamic characteristics of the F-1 which occurs when an imbalance of static pressure leads to 'hot spots in the manifolds.
The heart of the engine was the thrust chamber, which mixed and burned the fuel and oxidizer to produce thrust. A domed chamber at the top of the engine served as a manifold supplying liquid oxygen to the injectors, and also served as a mount for the gimbal bearing which transmitted the thrust to the body of the rocket. Below this dome was the injectors, which directed fuel and oxidizer into the thrust chamber in a way designed to promote mixing and combustion. Fuel was supplied to the injectors from a separate manifold; some of the fuel first traveled in 178 tubes down the length of the thrust chamber — which formed approximately the upper half of the exhaust nozzle — and back in order to cool the nozzle. A gas generator was used to drive a turbine which drove separate fuel and oxygen pumps, each feeding the thrust chamber assembly. Structurally, fuel was used to lubricate and cool the turbine bearings.



F-1 engine

Five huge F-1 engines used in Apollo-11



Gas-Generator Cycle

Diagram of a Saturn V launch vehicle

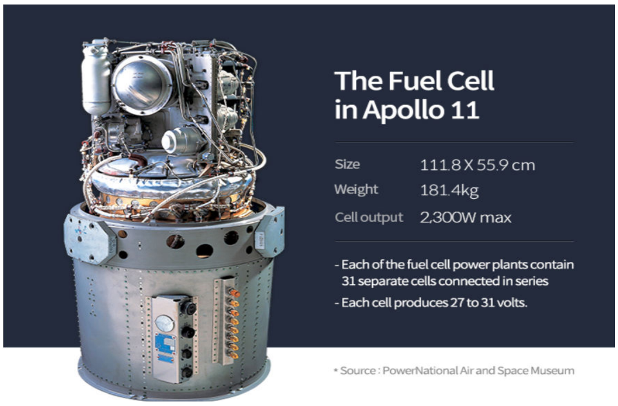
Saturn V Launch Vehicle

Saturn V launch vehicles and flights were designated with an AS-500 series number, "AS" indicating "Apollo Saturn" and the "5" indicating Saturn V.

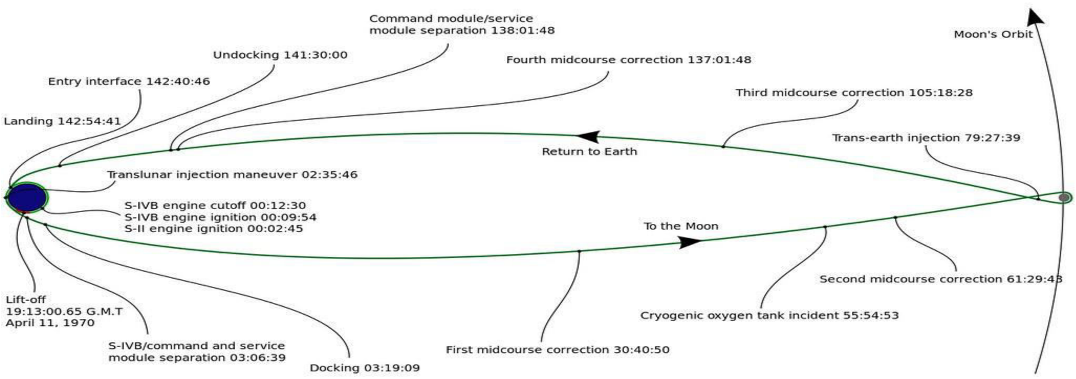
The three-stage Saturn V was designed to send a fully fueled CSM and LM to the Moon. The S-IC first stage burned RP-1/LOX for a rated thrust of 7,500,000 pounds-force (33,400 kN), which was upgraded to 7,610,000 pounds-force (33,900 kN). The second and third stages burned liquid hydrogen;the third stage was a modified version of the S-IVB, with thrust increased to 230,000 pounds-force (1,020 kN) and the capability to restart the engine for translunar injection after reaching a parking orbit.

Fuel Cells

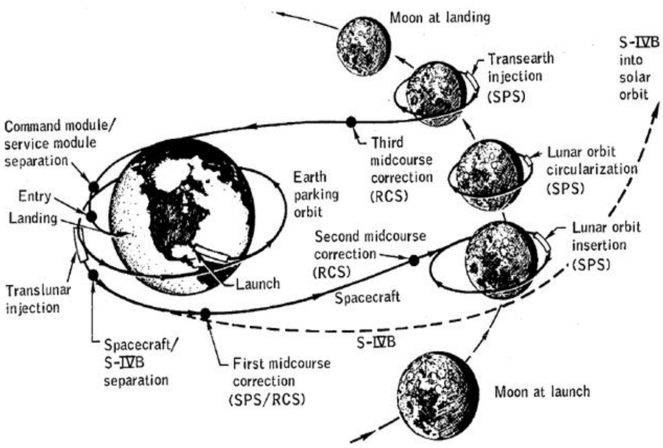
Both Gemini and Apollo spacecraft obtained electrical power from hydrogen-oxygen fuel cells. A fuel cell is like a battery. It converts energy released in a chemical reaction directly to electrical power. Unlike a storage battery, a fuel cell continues to supply current as long as chemical reactants are available or replenished (even while the cell is operating).



7. ORBITAL DYNAMICS



- Apollo 11 reached orbit around earth at around t=11 min 49 sec.
- Then in order to set a trajectory to the moon, a trans-lunar insertion orbit was done.
- **Atrans-lunar injection** (**TLI**) is a propulsive maneuver used to set a spacecraft on a trajectory that will cause it to arrive at the Moon.
- This maneuver happened at approximately t=2 hr 50 min 13 sec. For the Apollo lunar missions, TLI was performed by the restartable J-2engine in the S-IVB third stage of the Saturn V rocket.
- This particular TLI burn lasted approximately 350 seconds, providing 3.05 to 3.25 km/s of change in velocity, at which point the spacecraft was traveling at approximately 10.4 km/s relative to the Earth.
- The TLI placed Apollo on a "free-return trajectory" - often illustrated as a figure of eight shape, which would enable a return to Earth with no engine firing, providing a ready abort of the mission at any time prior to lunar orbit insertion.
- The command and service module, or CSM, Columbia separated from the stage, which included the spacecraft-lunar module adapter, or SLA, containing the lunar module, or LM, Eagle.
- The CSM docked with the LM. On July 19, after Apollo 11 had flown behind the moon out of contact with Earth, came the first lunar orbit insertion maneuver.
- At about 75 hours, 50 minutes into the flight, a retrograde firing for 357.5 seconds placed the spacecraft into an initial, elliptical-lunar orbit of 69 by 190 miles.
- Later, a second burn for 17 seconds placed the docked vehicles into a lunar orbit of 62 by 70.5 miles.
- At 100 hours, 12 minutes into the flight, the Eagle undocked and separated from Columbia for visual inspection.
- At 101 hours, 36 minutes, the LM descent engine fired for 30 seconds to provide retrograde thrust and commence descent orbit insertion, changing to an orbit of 9 by 67 miles.
- At 102 hours, 33 minutes, after Columbia and Eagle had reappeared from behind the



  moon and when the LM was about 300 miles uprange, powered descent initiation was performed with the descent engine firing for 756.3 seconds.

- After eight minutes, the LM was at "high gate" about 26,000 feet above the surface and about five miles from the landing site. The descent engine continued to provide braking thrust until about 102 hours, 45 minutes into the mission.
- Partially piloted manually by Armstrong, the Eagle landed in the Sea of Tranquility.
- During liftoff from the moon, the ascent stage engine fired at 124 hours, 22 minutes.
- It was shut down 435 seconds later when the Eagle reached an initial orbit of 11 by 55 miles above the moon.
- As the ascent stage reached apolune at 125 hours, 19 minutes, the reaction control system, or RCS, fired so as to nearly circularize the Eagle orbit at about 56 miles, some 13 miles below and slightly behind Columbia.
- Subsequent firings of the LM RCS changed the orbit to 57 by 72 miles.
- Docking with Columbia occurred at 128 hours 3 minutes into the mission.
- Trans-Earth injection of the CSM began July 21 as the SPS fired for two-and-a-half minutes.
- Re-entry procedures were initiated July 24, 44 hours after leaving lunar orbit. The SM separated from the CM, which was re-oriented to a heat-shield-forward position.
- Parachute deployment occurred at 195 hours, 13 minutes.
- After a flight of 195 hours, 18 minutes, 35 seconds. Apollo 11 splashed down in the Pacific Ocean, 13 miles from the recovery ship USS Hornet.

Radiation Protection methods

Van Allen Belts:

Although the Apollo missions have placed men outside the protective geomagnetic shielding and have subjected them to types of ionizing radiation seldom encountered in earth environments, radiation doses to Apollo crewmen have been minimal.
Spacecraft transfer from low earth orbit to translunar coast necessitates traverse of the regions of geomagnetically trapped electrons and protons known as the Van Allen belts.
When beyond these belts, the spacecraft and crewmen are continuously subjected to high-energy cosmic rays and to varying probabilities of particle bursts from the sun.
The problem of protection against the natural radiations of the Van Allen belts was recognized before the advent of manned space flight.
The small amount of time spent in earth orbit and the rapid traverse of the radiation belts during Apollo missions have minimized astronaut radiation dose.

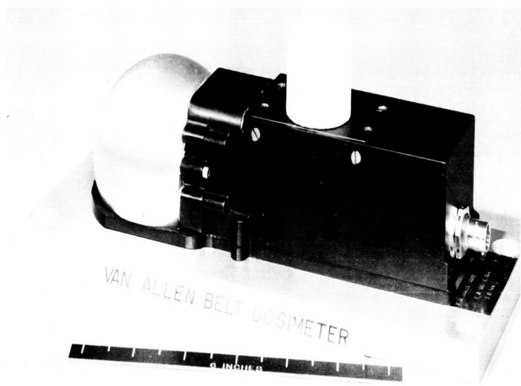


Figure 3. - Van Allen belt dosimeter.

The Van Allen belt dosimeter (VABD) (fig.3) was designed specifically for Apollo dosimetry within these radiation belts and has proved satisfactory because dose values derived from its greater than 180 degree radiation acceptance angle have correlated well with doses indicated by postflight analyses of passive dosimeters worn by the crewmen.

Solar-Particle Events:

- No major solar-particle events have occurred during an Apollo mission.
- Although much effort has been expended in the field of solar -event forecasting, individual eruptions from the solar surface are impossible to forecast. The best that could be provided at that time was an estimate of particle dose, given radio frequency (RF) confirmation that an eruption has occurred.

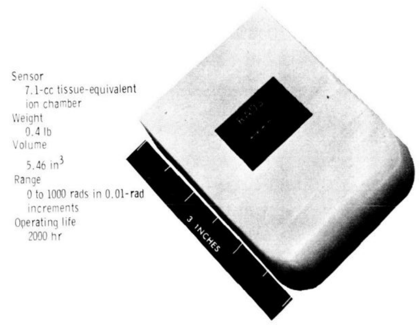


Figure 5. - Personal radiation dosimeter.

Cosmic Rays:

- One particular effect possibly resulting from cosmic rays has been the light flash phenomenon reported on the Apollo 11 and subsequent missions.

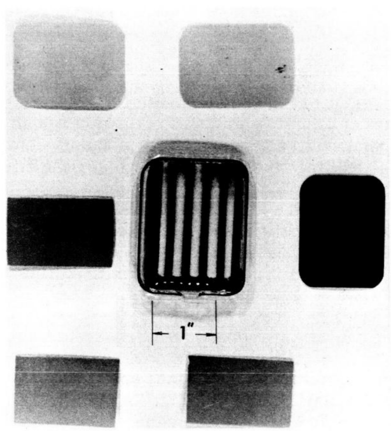


Figure 6. - Passive dosimeter with component parts.

- Although ionizing radiation can produce visual phosphenes (subjective sensations best described as flashes of light) of the types reported, a definite correlation has not been established between cosmic rays and the observation of flashes.
- The light flashes had been described as starlike flashes or streaks of light that apparently occur within the eye.
- The flashes were observed only when the spacecraft cabin was dark or when blindfolds were provided and the crewmen were concentrating on the detection of the flashes.

- There was a possibility that visual flashes may indicate the occurrence of damage to the brain or eye; however, no damage had been observed among crewmen who had experienced the light flash phenomenon.

Neutrons:

- Neutrons created by cosmic rays in collision with lunar materials were postulated to be a potential hazard to Apollo crewmen.
- Whole-body counting and neutron-resonant foil techniques had been initiated on the Apollo 11 mission.
- The results of these analyses indicated that neutron doses were significantly lower than had been anticipated.

Overall Radiation Spectrum:

- During a complete Apollo mission, astronauts are exposed to widely varying fractions of radiations from the Van Allen belts, cosmic rays, neutrons, and other subatomic particles created in high energy collisions of primary particles with spacecraft materials. In addition, the individual responsibilities of the crewmen differ, and, therefore, radiation exposure may differ.
- To allow accurate determination of radiation exposure of the crewman, each carried a personal radiation dosimeter (PRD) (fig. 5) and three passive dosimeters (fig. 6).
- The PRD provides a visual read-out of accumulated radiation dose to each crewman as the mission progresses.

Conclusion:

- Apollo missions had not undergone any major space radiation contingency.
- However, the development of spacecraft dosimetry systems, the use of a space radiation surveillance network, and the availability of individuals with a thorough knowledge of the space radiation environment have assured that any contingency would be recognized immediately and would be coped with in a manner most expedient for both crew member safety and mission objectives.
- It had been shown on the Apollo missions that the spacecraft and its crewmen had successfully avoided the large radiation doses that, before the Apollo missions, had been cited as a possible deterrent to manned space flight.

8. COMMUNICATIONS AND SYSTEM

The Apollo missions were incredibly complex, with multiple space vehicles performing intricate maneuvers in deep space, which required accurate tracking at extreme distances.
An S-Band Transponder was the only mode or link of communications between Apollo 11 Astronauts of NASA's control mission and millions of people watching on Earth, which was built by General Dynamics explicitly. The Unified S-band (USB) operated in the S-Band portion of the microwave spectrum, unifying voice communications, telemetry, command, television, tracking and ranging into a single system to save size and weight and simplify operations. This ground

communication network was managed by Goddard Space Flight Center (GSFC). In this, Collin Radio, Blaw-Knox, Motorola and Energy systems were the commercial contractors.

DEVELOPMENT OF THE S-BAND TRANSPONDER FOR APOLLO

The equipment had to be designed to withstand the extreme cold, heat and radiation they would experience, and for Apollo 11, they also needed to transmit more data than previous NASA missions, including television and video.
Hundreds of employees in Scottsdale, AZ began developing the Unified S-Band Transponder in 1962, a new system that would accurately track the Apollo spacecraft, transmit and receive telemetry signals, communicate between ground stations and the spacecraft, and provide the link for the historic broadcast from the surface of the moon. The formal contract was awarded in 1963 to Motorola's Government Electronics Division, a legacy company of General Dynamics.
The concept was presented by Lincoln Laboratory in an initial report on July 16, 1962, titled Interim Report on Development of an Internal On-Board RF Communications System for the Apollo Spacecraft. In thatreport,

it was shown that many on-board electronic functions could be performed very effectively by a single system that was a suitable adaptation of the transponder developed by Jet Propulsion Laboratory for use with the DSIF tracking stations. This was the origin of the Goal System for Apollo, later called the Integrated (or Integral) RF system, then later known as the Unified Carrier System. The idea behind the unified S-Band communications system was to reduce the number of systems previously used in the Mercury space program, which provided a multiplicity of electromagnetic transmitting and receiving equipment. In early flights, these operated at seven discrete frequencies within five widely separated frequency bands. Largely because of expediency, the following separate units were employed:

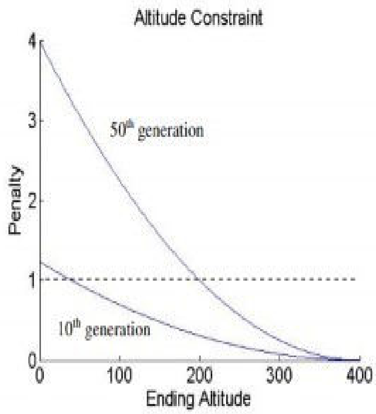
- HF voice transmitter and receiver
- UHF voice transmitter and receiver
- Command receiver
- Telemetry transmitter No. 1
- Telemetry transmitter No. 2
- C-band transponder beacon
- S-band transponder beacon

JOURNEY TO THE MOON

The components produced by Scottsdale employees equipped the Apollo spacecraft with the fundamental communications capabilities to remain in contact with mission control throughout the journey.Once the spacecraft reached a distance of 30,000 miles from Earth, the astronauts completely relied on the Unified S-Band Transponder to stay connected. The Transponder was their only link to mission control and transmitted all voice and video communications, spacecraft status, mission data, distance, the astronauts' biomedical data and emergency communications.

9. MOGA Modelling

In order to populate a Pareto front, a multi-objective genetic algorithm (MOGA) was used. This heuristic technique was chosen because discrete design variables such as material type and propellant type were used, and genetic algorithms handle discrete variables well. The two objectives were to maximize J1 (payload mass) and minimize J2 (cost). The fitness function initialized at a maximum of 1 for each individual. Then it gave a small penalty of 0.01 to each design for each other design that dominated it by having a lower cost and higher payload capability. The fitness was then squared to increase the gap between the more and less dominated designs. Finally, it gave zero fitness for designs that were otherwise infeasible. This fitness value was then used to decide which designs carried on to the next generation of the genetic algorithm. The fitness function for a feasible point is given below:



$$F = max\{1.0 - 0.01 * n_{dom} + p(A_{final}), 0\}^2 \tag{1}$$

The penalty curve steepened with each generation. This is because a low curve would not penalize the infeasible designs enough, but a high curve would often cause the entire starting population to have zero fitness. By starting with a low curve and raising it, the MOGA was able to find the largest number of feasible designs.

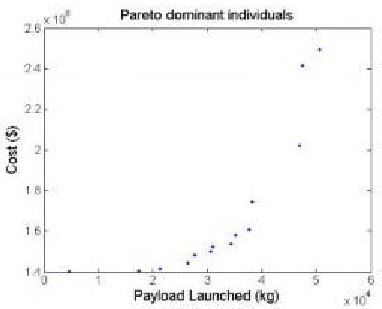


Figure 6. Non-dominated individuals

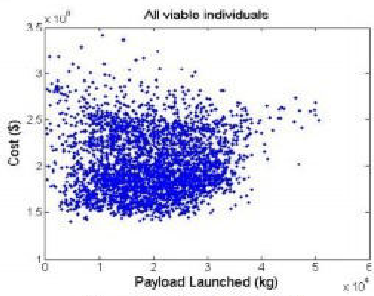


Figure 5. All feasible individuals from a MOGA run.

Notice that the Pareto front made is not very well distributed and has a couple steps. Running the MOGA multiple times would lead to covering different ranges and help fill out the Pareto front.

After many runs, the MOGA was able to form a well-populated and fairly smooth Pareto front. There is no guarantee that these points are truly optimal, and a look at the sensitivity analysis suggests that they could be improved, at least slightly. Since the final designs from the MOGA could be improved slightly by simply adjusting the design variables slightly, it would be interesting to examine the benefits of using a gradient-based optimizer as a final step. This could improve the designs slightly by guiding the Pareto points to local maxima.

TRAJECTORY OPTIMIZATION EQUATION

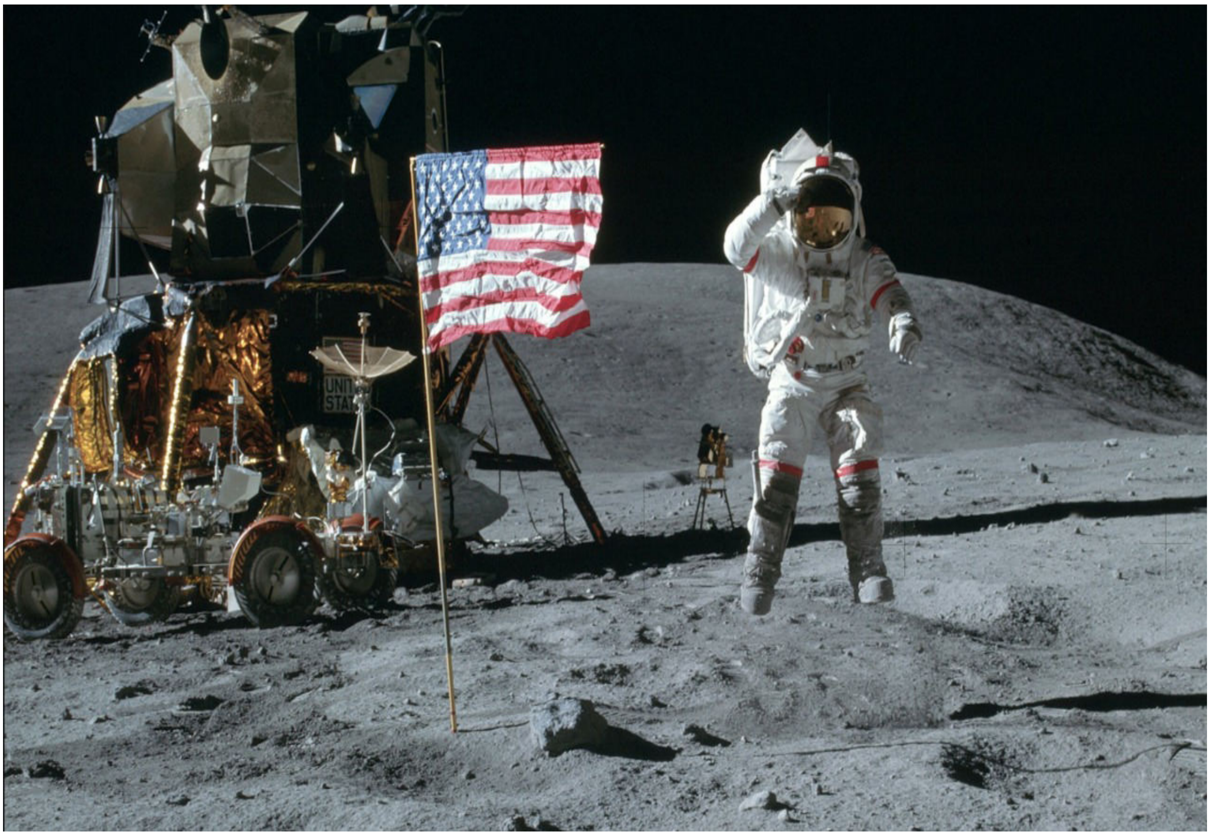
$$\frac{\mathrm{d}}{\mathrm{d}t}(r) = r' \tag{2}$$

$$\frac{\mathrm{d}}{\mathrm{d}t}(r) = -10^9\mu/r^2 + r*\theta^2 + (T[r, T_1....T_5] - D[r, v, \theta_c, R_r])*\cos(\alpha[r, \alpha_1...\alpha_5])/m \tag{3}$$

$$\frac{\mathrm{d}}{\mathrm{d}t}(\theta) = \theta' \tag{4}$$

$$\frac{\mathrm{d}}{\mathrm{d}t}(\theta') = (T[r, T_1....T_5] - D[r, v, \theta_c, R_r])*\sin(\alpha[r, \alpha_1...\alpha_5])/(r*m) \tag{5}$$

$$\frac{\mathrm{d}}{\mathrm{d}t}(m) = -T[r, T_1....T_5]/(I_{sp}*g_0) \tag{6}$$



References:

1. https://www.nasa.gov/mission_pages/apollo/missions/apollo11.html
2. https://en.wikipedia.org/wiki/Trans-lunar_injection
3. https://history.nasa.gov/SP-4029/Apollo_11i_Timeline.htm
4. https://airandspace.si.edu/exhibitions/apollo-to-the-moon/online/apollo-11/about-the-spacecraft.cfm
5. https://en.wikipedia.org/wiki/Apollo_11
6. https://en.wikipedia.org/wiki/Rocketdyne_F-1
7. https://www.nasa.gov/topics/history/features/f1_engine.html
8. https://history.nasa.gov/SP-4206/ch4.htm
9. https://drive.google.com/file/d/1XjQFrNWfiZlM8yAT24AR7D3eQeBu9Bwe/view?usp=sharing

18 References

1. Brian K. Bairstow, Olivier de Weck, Jaroslaw Sobieszczanski-Sobieski. Multiobjective Optimization of Two-Stage Rockets for Earth-To-Orbit Launch. Link to Paper

2. Isakowitz, S. J., Hopkins, J. B., and Hopkins, J. P., Jr., International Reference Guide to Space Launch Systems, 4th ed., AIAA, Virginia, 2004, pp. 109-146

3. Taylor, C., Single Stage to Orbit Rocket Model, Massachusetts Institute of Technology, Cambridge, MA, 2005.

4. Schoonover, P. L., Crossley, W. A., and Heister, S. D., Application of a Genetic Algorithm to the Optimization of Hybrid Rockets, Journal of Spacecraft and Rockets, Vol. 37, No.5, September-October 2000, pp. 622-629.

5. Suchiya, T., and Mori, T., Optimal Conceptual Design of Two-Stage Reusable Rocket Vehicles Including Trajectory Optimization, Journal of Spacecraft and Rockets, Vol. 41, No.5, September-October 2004, pp. 770-778

6. Sobieszczanski-Sobieski, J., External Fuel Tank Model, AIAA Short Course on Multidisciplinary Design Optimization, NASA Langley Research Center, Hapton, VA, 2002.

7. Schuman, T., Integration of System-Level Optimization With Concurrent Engineering Using Parametric Subsystem Modeling, Masters Thesis, Aeronautics and Astronautics Dept., Massachusetts Institute of Technology, Cambridge, MA, 2004.6.

8. U.S. Standard Atmosphere, 1962, U.S. Government Printing Office, Washington, D.C., 1962.