# Interplatary Mission to Mars

*TEAM REPORT*

## TEAM EVOLON

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Internship And Project Division

Society for space education Research and Devolopment Bangalore

(SSERD)

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# Declaration

We, Mr **Md Aitesam Ahmed**, of MVJ College of Engineering, Bangalore, Mr **Hrishikesh S Nair**, of MVJ College of Engineering, Bangalore, Mr **M Greeshmanth**, of Sastra University, Mr **Keyur Devganiya**, of Sardar Vallabhbhai Patel Institute of Technology (SVIT), Vasad, Mr **Kiran Kandel** of Chandigarh University, Ms **Mayuri M. Ghadigaonkar**, of MCT's Rajiv Gandhi Institute of Technology, Mr **Arunachalam E**, of Jeppiaar Engineering College, Mr **Allwyn Prince.A**, of Jeppiaar Engineering College, Mr **Girish Kumbargoudar**, of KLS GIT Belagavi, Mr. **Jagadish Kare** of KLS GIT, Belagavi, Mr. **Chetan Tanangi**, of KLS GIT, Belagavi, and Mr. **Dileep Kumar**, of Indian Institute of Technology, Chennai, hereby declare that:

This research internship work entitled "Interplanetary Mission To Mars" has been carried out by us under the guidance of Mr **Rakeshh Mohanarangan**, Propulsion Engineer and Ms **Rashika SN** Space Engineering student at Politecnico Di Milano and Logistics Officer of the Science team at Mission Asclepios.

# Acknowledgements

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This opportunity allowed us to work with students from different geographical locations and help us not only share knowledge but also increase our professional network. We also express our gratitude to Mr. Rakeshh Mohanarangan and Ms. Rashika SN for guiding and mentoring us in our journey. We would also like to thank our co-ordinator Mr. Thejas KV for being our source of networking. The sessions held by Mr. Pavan in order to expose us to professional methodology cannot be thanked enough.

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# Abstract

Humans have always been fascinated by Space. Since ancient times, we have wondered about a world beyond ours. With Mars, that search, it seems, have come to a conclusion. A planet which is very similar yet distinguish from Earth is human's new obsession. Many missions were done to study the Red Planet. The result was the discovery of possibility of water and life in a distant past. The possibility that there can be two planets supporting life increases the human's curiosity.

With development in Rocket Science and growing Technology, seems very likely that in near future there will be a human base on the fourth planet. With so much knowledge about our neighbor, one would assume that there is very less to do now. Yet, many mysteries of the planet are unknown that would play a major role in Human Settlement.

This project is to design a propulsion system that will carry a certain payload to the Mars orbit from LEO. Our aim is to develop a efficient system that can achieve the task of carrying the scientific instruments as payload.

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# 1

# Introduction

Mars Missions, although recent, are not a new field for humans. Mars has always been an interest of Humans to explore. Designing a propulsion system for these missions requires pre-requisite knowledge of Rocket Propulsion. The Newton's Second and Third laws of motion forms the base for any propulsion system.

## 1.1 | Motivation

On 5th November, 2013, ISRO launched their Mars Orbiter Mission which was the cheapest mission to Mars. With Space exploration growing with each passing day, it becomes important to have different propulsion systems for different missions having their own respective objectives. The propulsion plays an important role in taking the satellite or the rockets to their destined location.

## 1.2 | Aims and Objectives

The prime objective of this mission is to design a propulsion system which can carry a payload of about 20kgs from Lower Earth Orbit (LEO) to an altitude of 200km above the Martian Surface.

# 2

# Literature Review

•**George P. Sutton [1]** has introduced us with the basics of Rockets and its propulsion system in his book “Rocket Propulsion Element”. It describes the fundamentals necessary for the design of a propulsion system. The Ideal Rocket Equation, calculation of various other parameters were done based on the equations and data provided in the book.

•**Dieter K. Huzel and David H. Huang [2]** in their book “Modern Engineering for Design of Liquid-Propellant Rocket Engines” has describe the math necessary for the design of tank for cryogenic propellants. This was used to calculate the tank for the bi-propellants Lox and liq. $CH_4$.

•**R.J.G. Hermsen [3]**, has provided with the equations needed for the calculation of pressurized gas Helium from his book "Cryogenic propellant tank pressurization".

•**Federal Aviation Administration [4]**, in their “Advance Aerospace Medicine” in “Section-III” has the necessary information for obtaining the value of Del V during a Hohmann Transfer Orbit.

•**Braeuing [5]**, was used to get the necessary equations for the Nozzle design.

•**Krzycki [6]** aided us with the equations for combustion chamber dimensions via his book "How to Design, Build, and Test Small Liquid-Fuel Rocket Engines"

# 3

# Methodology

## 3.1 | Payload Selection

The Payloads that were selected for the mission are Lymann Photometer and Terrain Mapping Camera.

• **Lyamnn Alpha Photometer** was recently sent in the Mars Orbiter Mission with an objective to measure D/H (Deuterium to Hydrogen Abundance Ratio) in the Martian atmosphere which allows us to understand especially the lost process of water from the planet.

• **Terrain Mapping Camera** was used in Chandrayaan I for the mapping of the Terrain of Lunar surface. The same objective has been now shifted to this mission.

| Lymann Alpha Photometer | |
|---|---|
| Operational Range | 3000 km |
| Weight | 1.97 kg |
| Power | 7.2 W |
| Altitude | 200-400 km |

Table 3.1: **Specifications of Lymann Alpha Photometer**

| Terrain Mapping Camera | |
|---|---|
| Operational Range | 20km |
| Weight | 6 kg |
| Power | 13 W |
| Altitude | 200-400km |

Table 3.2: **Specifications of Terrain Mapping Camera**

## 3.2 | Transfer Orbit

The transfer orbit is the locus of positions that a spacecraft goes through when getting transferred from one circular orbit to another. This is explained by Hohmann Transfer Orbit (Fig 1). For this internship, the orbits of the planets (Earth and Mars) are assumed to be circular. The figure below shows the transfer orbit.

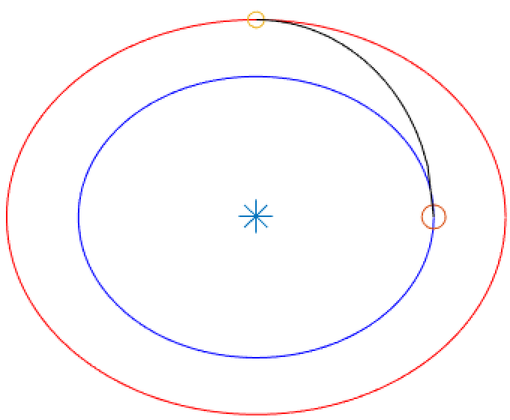

Figure 3.1: Hohmann Transfer Orbit

## 3.3 | Delta V Calculation

With the concept of Transfer Orbit clear, the task was to calculate the Delta V needed for the mission. The calculation was divided into 2 parts.

- **Leaving LEO and Entering the Heliocentric Orbit**
- **Entering the Mars Orbit**

The suitable equations were used to calculate the Del V for each part. For Leaving LEO and Entering the Heliocentric Orbit, the Del V ($\Delta V_1$) was found to be 3510 m/sec. For the second part, i.e., Entering Mars Orbit, the Del V ($\Delta V_2$) was calculated to be 2010 m/sec. The final Del V ($\Delta V$) was the sum of the $\Delta V_1$ and $\Delta V_2$. The Delta V for our mission is, therefore, 5520 m/sec.

| Delta V Calculation | |
|---|---|
| From LEO to Heliocentric Orbit ($\Delta V_1$) | 3510 m/sec |
| Entering Mars Orbit ($\Delta V_2$) | 2010 m/sec |
| Final ($\Delta V$) | 5520 m/sec |

Table 3.3: **Calculation of Delta V**

| **Properties** | **Fuel ($CH_4$)** | **Oxidizer (LoX)** |
|---|---|---|
| **Density** | 424 kg/m$^3$ | 1140 kg/m$^3$ |
| **Freezing Point** | -219$^o$C | -184$^o$C |
| **Boiling Point** | -183$^o$C | -162$^o$C |
| **$I_{sp}$ = 380 sec** | | |

Table 3.4: **Properties of the Propellant**

## 3.4 | Propellant Selection

The propellant selection is an important task for a mission as it is the prime source of thrust generation. The type of propellant selected for this is liquid propellant, bi-propellants to be specific, namely LoX and $CH_4$. The propellants were selected, though being cryogenic for this long mission based on the recent ZBO development for cryogenic propellants. The $I_{sp}$ of this bi-propellant is a staggering 380 sec which allows for a very efficient thrust generation.

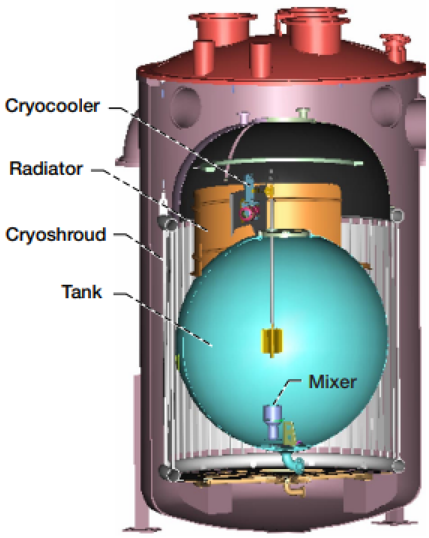


Figure 3.2: ZBO Development (a)

(1)

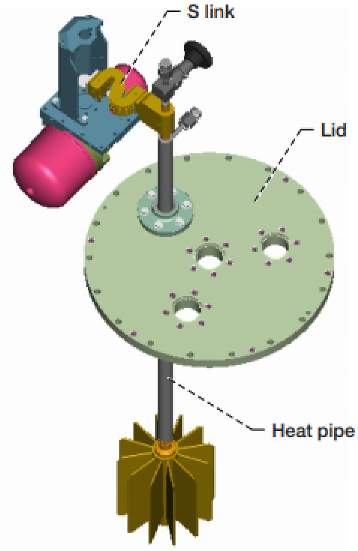


Figure 3.3: ZBO Development (b)

## 3.5 | Staging and Mass Calculation

The Rocket equation, shown below, was used to calculate the Initial mass. With various calculation, the mission was divided in two stages.

$$\Delta V = I_{sp} g_o \ln(\frac{m_o}{m_f})$$

Where $\Delta V$ is Delta V, $I_{sp}$ is the specific Impulse, go is the standard gravity, $m_o$ is the initial mass and $m_f$ is the final mass. The amount of mass decreased from single stage mission to 2 stage mission is far more than the mass decrease from 2 stages mission to 3 stages mission. Hence, the mission had two stages.

With 2 stages, payload mass of 20kgs and assumed dry mass of 600kgs, the Rocket Equation was used to calculate the initial mass, which turned out to be 2035.83 kg. The table below shows the progression. The propellant mass required for this mission was calculated to be 1415kg.

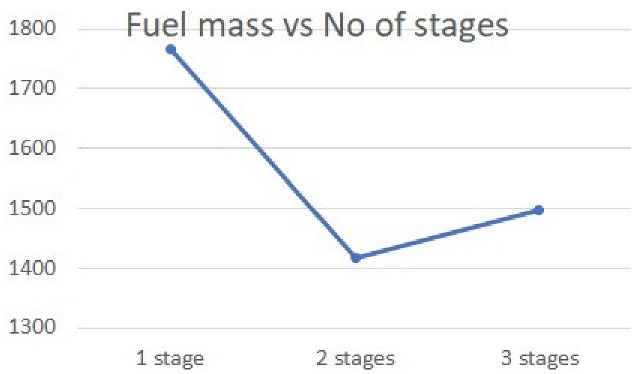


Figure 3.4: Fuel Mass v/s Number of Stages

| **STAGE 2** | |
|---|---|
| **$I_{sp}$=380 sec** | |
| Final mass (Payload mass + Dry Mass) | 320 kg |
| Delta V ($\Delta$V) | 2760 m/sec |
| Initial Mass | 670.95 kg |
| Propellant Mass | 351 kg |

Table 3.5: **Mass Calculation- Stage 2**

| **STAGE 1** | |
|---|---|
| **$I_{sp}$=380 sec** | |
| Final mass (Payload mass + Dry Mass) | 970.95 kg |
| Delta V ($\Delta$V) | 2760 m/sec |
| Initial Mass | 2035.83 kg |
| Propellant Mass | 1065 kg |

Table 3.6: **Mass Calculation- Stage 1**

# 3.6 | Major Subsystems

For the further designing of the system, there were four main domains that needed to be covered. They are; **System Engineering, Engine Design, Tank Design and Feed System Design.**

## 3.6.1 | Objectives of Subsystems

The objective of each of the sub domains are:

1. **System Engineering:**

- Thrust Calculation
- Make a Feed System Architecture
- Mass Budgeting

**2. Engine Design:**

- Designing the Nozzle
- Designing the Combustion Chamber

**3. Tank Design:**

- Material Selection for the propellant tanks
- Shape of the Tank
- Dimensions of the Tank

**4. Feed System Design:**

- Type of Feed System
- Sizing
- Pressure Drop Study

# 4

# Subsystems

## 4.1 | System Engineering

### 4.1.1 | Thrust Calculation

•For Calculation of Thrust, the thrust-to-weight ratio of a previous Mars mission was taken as reference. For this mission, Mars Reconnaissance Orbiter was selected. This is due to the similarity between the initial mass of our mission with that of MRO.

•The following table shows the comparison of our mission with them.

| MRO | | Our Mission | |
|---|---|---|---|
| Weight | 2189 Kg | Weight | 2035.83 Kg |
| T/W | 0.532 | T/W | 0.54 |
| Thrust | 1159.2 N | Thrust | 1100 N |

Table 4.1: **Comparison between MRO and Our Mission**

•The Thrust is further divided into the main Thrust from the Engine to be 700N and 50N thrust from 8 small thrusters for altitude correction.

| | |
|---|---|
| Thrust (main engine) | 700 N |
| Thrust (Thruster) | 50 N |
| Thrust (8 Thrusters) | 400 N |
| Total Thrust | 1100 N |

Table 4.2: **Thrust Values**

## 4.1.2 | Feed System Architecture

•Feed System Architecture involves the making of the blue print for the feeding of propellants to the engine and thrusters.

• Since the mission has relatively low thrust of 1100N, the gas pressurized feed system was selected.

• Also, the gas system is a blow down system rather than a regulated one.

• This is because, in blow down systems, the mass required of the pressurized gas is reduced significantly

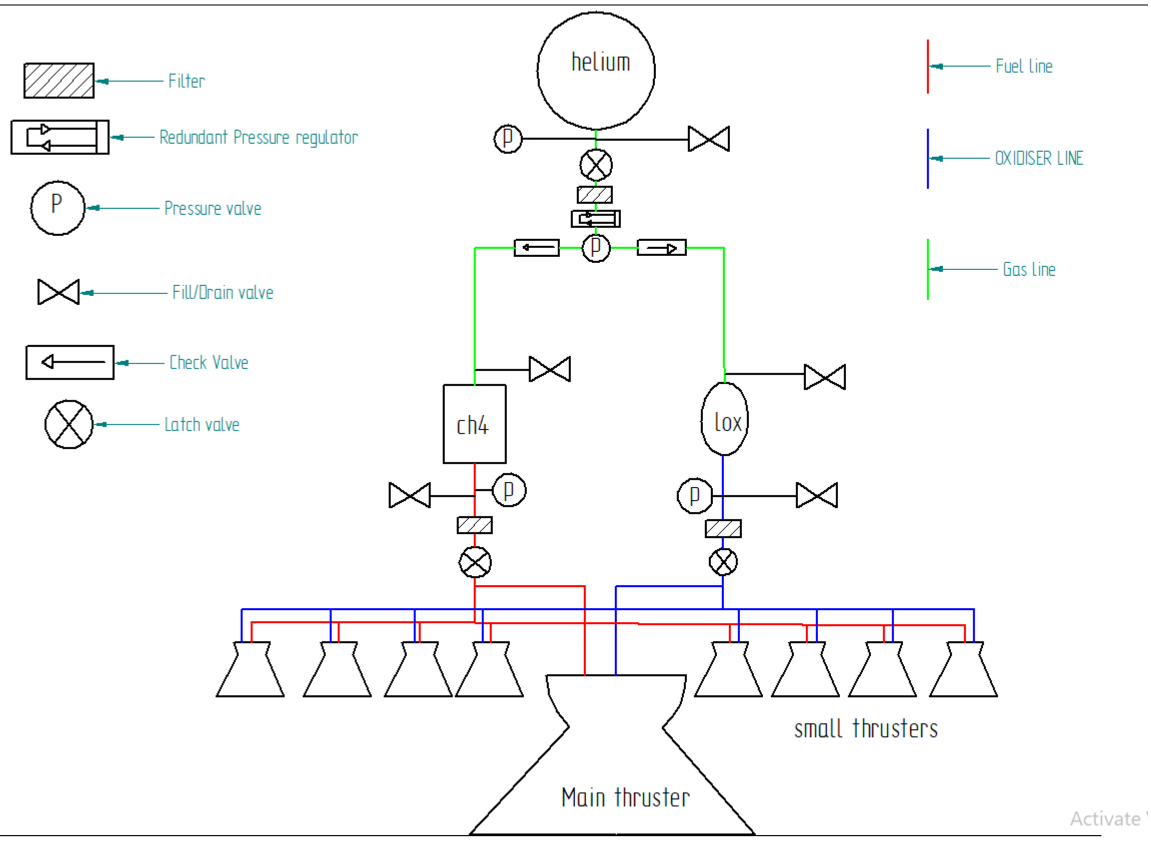


Figure 4.1: Feed System Architecture

### 4.1.3 | Mass Budgeting

•The calculation of mass, initially was based on the ideal $I_{sp}$ of the propellant. In practice, the bi-propellant $I_{sp}$ depends on the efficiency of the engine to utilize the propellant.

•After the design of Engine(in the upcoming sections) the $I_{sp}$ value of the propellant was found to be 352 sec. Since this new $I_{sp}$ is almost 95% of the ideal $I_{sp}$, a mass budgeting was required. The updated mass of the mission was calculated to be 2241 kg.

•The table below shows the updated mass budget.

| **STAGE 2** | |
|---|---|
| **$I_{sp}$=352 sec** | |
| Final mass (Payload mass + Dry Mass) | 320 kg |
| Delta V ($\Delta$V) | 2760 m/sec |
| Initial Mass | 710 kg |
| Propellant Mass | 390 kg |

Table 4.3: **Updated Mass Calculation- Stage 2**

| **STAGE 1** | |
|---|---|
| **$I_{sp}$=352 sec** | |
| Final mass (Payload mass + Dry Mass) | 1010 kg |
| Delta V ($\Delta$V) | 2760 m/sec |
| Initial Mass | 2241 kg |
| Propellant Mass | 1231 kg |

Table 4.4: **Updated Mass Calculation- Stage 1**

## 4.2 | Engine Design

### 4.2.1 | Nozzle Design

•The thrust of the mission was found to be 1100N for a thrust-to-weight (T/W) ratio of 0.54. The overall thrust was divided into, 700N thrust from the main engine and 50N thrust from 8 small thrust each.

•The nozzles were designed for 700N and 50N separately. For obvious reasons, a Convergence-Divergence (C-D nozzle) was designed. To start off, some values were necessary to assume. The assumed values were, $P_c$ of 1.2 MPa, Pe of 1 KPa. With these values, the Oxidizer-to-Fuel (O/F) ratio was found to be 2.65.

•Building up on it, the Flame Temperature was 3170 K, the Molecular Weight of gas was 19.05 and ratio of specific heats ($C_p/C_v$) or ($\gamma$) to be 1.216. The following graphs show the plotting of each of these properties against chamber pressure and exit nozzle pressure.

•With these inputs, two nozzles were designed separately. One for main engine for Thrust 700 N and other for thrusters for Thrust 50 N.

| Input Values | |
|---|---|
| Thrust (Main Engine) | 700 N |
| Thrust (Each Thruster) | 50 N |
| Chamber Pressure ($P_c$) | 1.2 MPa |
| Exit Pressure ($P_e$) | 1KPa |
| Mixture Ratio (O/F) | 2.65 |
| Specific Heat Ratio ($\gamma$) | 1.2 |
| Chamber Temperature ($T_c$) | 3170 K |

Table 4.5: **Input Value for Nozzle Design**

#### 4.2.1.1 | Main Engine

The thrust produced by the main engine is 700 N. With input values available from the graphs, a bunch of equations were used to calculate. The following shows the step wise calculation of the values.

•Exhaust Velocity ($V_e$) is the velocity at which gases leave the nozzle. It is calculated to be 3340.056 m/sec.

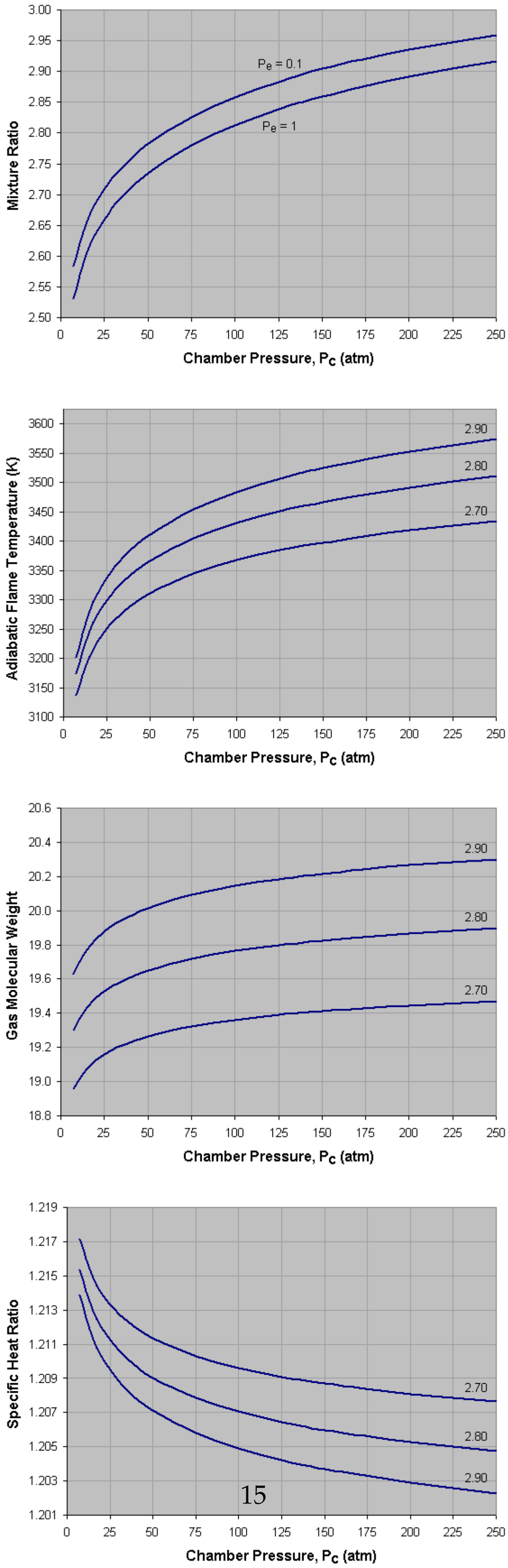




Figure 4.2: Chamber Pressure v/s Properties

•Pressure at throat ($P_t$) and Temperature at throat ($T_t$) are found to be 0.674 MPa and 2861.011 K.

•The temperature at the nozzle exit($T_e$) is calculated by using the Adiabatic relation between Pressure and Temperature and found to be 899.7 K

•The Mach number at throat in a C-D nozzle is 1. To calculate the exit Mach number ($M_e$), the process was assumed to be adiabatic. It was found to be 4.8

•The area ratio ($A_e/A_t$) plays a vital role in nozzle design. It was 78.15 for our mission.

•The thrust co-efficient ($C_f$) is defined as the thrust per unit frontal area per unit dynamic pressure (incompressable). It was calculated was to be 1.92.

•Now with almost all the necessary values known, the throat area, and by extension exit area are calculated to be 3.03 cm2 and 237.47 $cm^2$ respectively.

•With Area value known, it was easy to know the throat and exit diameter of the nozzle, which were 1.96 cm and 17.4 cm respectively.

•Also, the mass flow rate is calculated using the Thrust Equation and the value obtained was 0.202 kg/sec.

•The practical $I_{sp}$ for the mission is then calculated to 352 sec.

•With the help of O/F ratio, the individual mass flow rate can also be calculated.

**Nozzle Length**

The length of the nozzle is calculated by doing literature review and obtaining the radius of throat.

•A bell shaped nozzle is taken and the values/assumptions are shown in table below. The figure 7 shows the nozzle for main engine.

| Properties | Equation Used | Value |
|---|---|---|
| Exhaust Velocity | $V_e = \sqrt{(\frac{2\gamma}{\gamma-1})(\frac{R^* T_c}{M})(1-(\frac{P_e}{P_c})^{\frac{\gamma-1}{\gamma}}}$ | 3340.05 m/sec |
| Throat Pressure | $P_t = P_c\ (1+\frac{\gamma-1}{2})^{\frac{-\gamma}{\gamma-1}}$ | 0.674 MPa |
| Throat Temperature | $T_t = (\frac{T_c}{1-\frac{\gamma-1}{2}})$ | 2861.011 K |
| Exit Temperature | $\frac{T_e}{T_t} = (\frac{P_e}{P_t})^{\frac{\gamma-1}{\gamma}}$ | 899.7 K |
| Exit Mach Number | $V_e = M_e\ \sqrt{\gamma R T_e}$ | 4.8 |
| Area Ratio | $\frac{A_e}{A_t} = (\frac{\gamma-1}{2})^{\frac{-(\gamma+1)}{2(\gamma-1)}}\ \frac{(1+(\frac{\gamma-1}{2})M_e^2)^{\frac{\gamma+1}{2(\gamma-1)}}}{M_e}$ | 78.15 |
| Thrust Co- efficient | $C_f = \sqrt{(\frac{2\gamma^2}{\gamma-1})(\frac{2}{\gamma-1})(\frac{\gamma+1}{\gamma-1})} + \frac{(P_e-P_0)A_e}{P_o A_t}$ | 1.92 |
| Throat Area and Exit Area | $T = C_f P_c A_t$ | 3.03 $cm^2$ and 237.47$cm^2$ |
| Throat Diameter and Exit Diameter | $A = \pi r^2$ | 1.96 cm and 17.4 cm |
| Mass Flow Rate | $\dot{m}\frac{P_c A_t}{C^*}$ | 0.202kg/sec |
| $I_{sp}$ = 352 sec and Efficiency = 92.6% | | |

Table 4.6: **Nozzle Design Parameters for Main Engine**

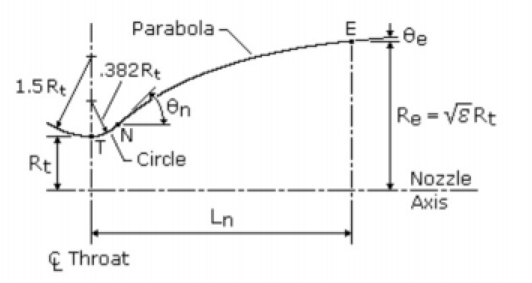


Figure 4.3: Nozzle Length Parameters

| Type | Bell Nozzle | $L_n$ = 80% $L_f$ |
|---|---|---|
| Thrust | T | 700 N |
| | $\theta_n$ | 42.87 $^o$ |
| | $\theta_e$ | 6.80 $^o$ |
| Fractional length | $L_f$ | 0.167 m |
| Nozzle length | $L_n$ | 0.138 m |

Table 4.7: **Nozzle Length (Main Engine)**

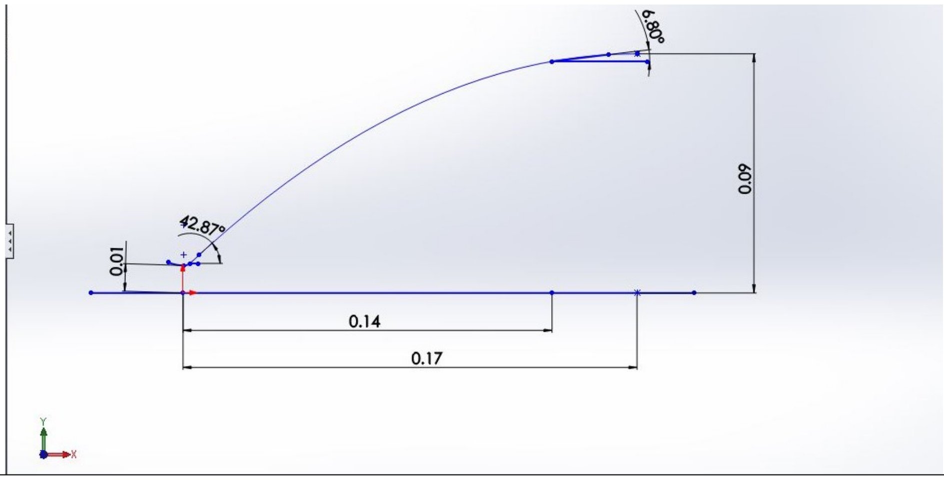


Figure 4.4: Nozzle for Main Engine

### 4.2.1.2 | Thrusters

•The input values from Table 4.5 were used to calculate the parameters for the nozzle that is being designed for thrusters providing 50 N of thrust.

•There are 8 of them to be designed. The steps of calculation remained similar to the main engine nozzle design.

•**Nozzle Length:**
1.Since the nozzle length of main engine is known, a similar approach resulted in nozzle length of thrusters as well.
2.The table below and Figure shows the same.

## 4.2.2 | Combustion Chamber

| Properties | Equation Used | Value |
|---|---|---|
| Exhaust Velocity | $V_e = \sqrt{(\frac{2\gamma}{\gamma-1})(\frac{R^*T_c}{M})(1-(\frac{P_e}{P_c})^{\frac{\gamma-1}{\gamma}}}$ | 3340.05 m/sec |
| Throat Pressure | $P_t = P_c\ (1+\frac{\gamma-1}{2})^{\frac{-\gamma}{\gamma-1}}$ | 0.674 MPa |
| Throat Temperature | $T_t = (\frac{T_c}{1-\frac{\gamma-1}{2}})$ | 2861.011 K |
| Exit Temperature | $\frac{T_e}{T_t} = (\frac{P_e}{P_t})^{\frac{\gamma-1}{\gamma}}$ | 899.7 K |
| Exit Mach Number | $V_e = M_e\sqrt{\gamma RT_e}$ | 4.8 |
| Area Ratio | $\frac{A_e}{A_t} = (\frac{\gamma-1}{2})^{\frac{-(\gamma+1)}{2(\gamma-1)}}\ \frac{(1+(\frac{\gamma-1}{2})M_e^2)^{\frac{\gamma+1}{2(\gamma-1)}}}{M_e}$ | 78.15 |
| Thrust Co-efficient | $C_f = \sqrt{(\frac{2\gamma^2}{\gamma-1})(\frac{2}{\gamma-1})(^{\frac{\gamma+1}{\gamma-1}})} + \frac{(P_e-P_0)A_e}{P_oA_t}$ | 1.92 |
| Throat Area and Exit Area | $T = C_fP_cA_t$ | 0.217cm$^2$ and 16.96cm$^2$ |
| Throat Diameter and Exit Diameter | $A = \pi r^2$ | 0.53cm and 4.65cm |
| Mass Flow Rate | $\dot{m}\frac{P_cA_t}{C^*}$ | 0.014kg/sec |
| **$I_{sp}$ = 352 sec and Efficiency = 92.6%** | | |

Table 4.8: **Nozzle Design Parameters for Thrusters**

| Type | Bell Nozzle | $L_n$ = 80% $L_f$ |
|---|---|---|
| Thrust | T | 50 N |
| | $\theta_n$ | 43.62 $^o$ |
| | $\theta_e$ | 9.81 $^o$ |
| Fractional length | $L_f$ | 0.043 m |
| Nozzle length | $L_n$ | 0.035 m |

Table 4.9: **Nozzle Length (Thrusters)**

### 4.2.2.1 | Main Engine

•The combustion chamber is cylindrical in shape with a characteristic length ($L^*$) of 1m.
•From section 4.2.1.1 the throat area (At) value is 3.03 cm$^2$ and the volume of the chamber is calculated ($V_c$).
•The combustion chamber, a certain length and diameter ratio is needed. Hence a bunch of (L/d) values are taken to calculate the curved surface area (CSA) of the chamber. The (L/d) with least CSA is selected.
•The following table shows the L/d v/s curved surface area of certain values and Figure 6 shows the points plotted in the graph.

•Once Length and diameter are known for the complete chamber, (cylindrical part and cone); 0. 046m and 0.092m respectively, the cross section area of the chamber was calculated and found to be 66.2 cm2.

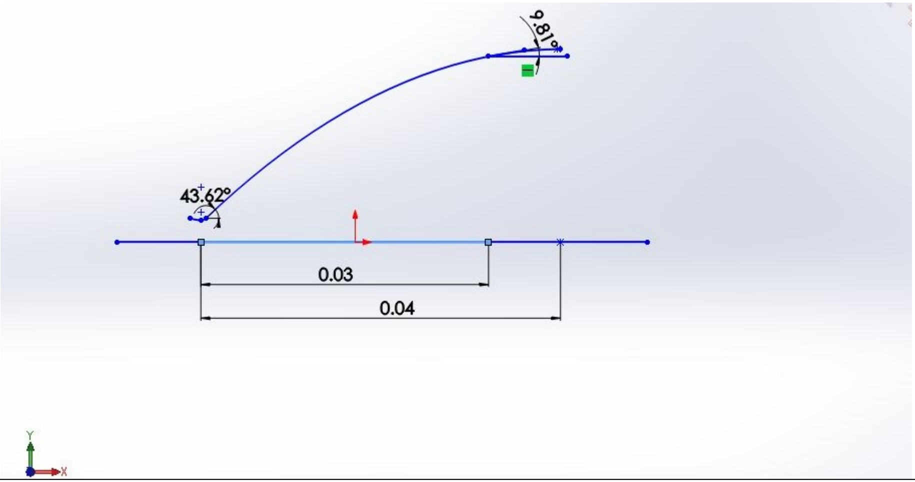


Figure 4.5: Nozzle for Thrusters

| L/d | r (m) | d (m) | L (m) | CSA ($m^2$) |
|---|---|---|---|---|
| 0.5 | 0.046 | 0.092 | 0.046 | 0.013 |
| 1 | 0.036 | 0.072 | 0.072 | 0.017 |
| 1.5 | 0.032 | 0.064 | 0.096 | 0.019 |
| 2 | 0.029 | 0.058 | 0.116 | 0.021 |
| 2.5 | 0.027 | 0.054 | 0.135 | 0.022 |
| 3 | 0.025 | 0.050 | 0.15 | 0.024 |

Table 4.10: **(L/d) v/s CSA for Main Engine**

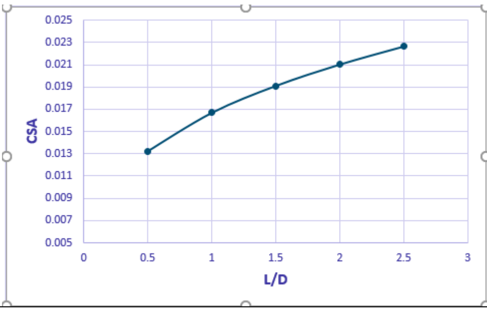


Figure 4.6: (L/d) v/s CSA for Main Engine

•The length of the cylindrical section of the chamber (Lc) is then calculated by using the following equation. It was found to be 0.028m.

$$L_c = \frac{1}{\epsilon}\left[\frac{V_c}{A_t} - \frac{1}{3}\ \frac{(\epsilon-1)}{tan\theta}\ \sqrt{\frac{A_t}{\pi}}\right]$$

Where,$\theta$ is assumed to be $10^o$ and is the ratio of cross section area of chamber to throat

area i.e., = ($A_c/A_t$).
•The length of the cone ($L_{cone}$) is the difference of total length to length of cylindrical part.

$$L_{cone} = L\text{-}L_c$$

•The table below shows the parameters of the combustion chamber of the main engine.

| L/d | r (cm) | d (cm) | L (cm) | CSA ($cm^2$) | $V_c$ | $A_c$ | $A_c/A_t$ | $L_c$ | $L_{cone}$ | $\theta$ |
|---|---|---|---|---|---|---|---|---|---|---|
| 0.5 | 4.6 | 9.2 | 4.6 | 130 | 303.85 | 66.2 | 21.78 | 2.82 | 1.77 | $10^o$ |

Table 4.11: **Parameters of Combustion Chamber for Main Engine**

## 4.2.2.2 | Thrusters

•With Main engine combustion chamber being design, we took a similar approach to design the chamber for small thrusters.
The characteristic length was again assumed 1m and further development was similar to main engine calculation. The throat area value was taken from section 4.2.1.2 as 0.217 $cm^2$

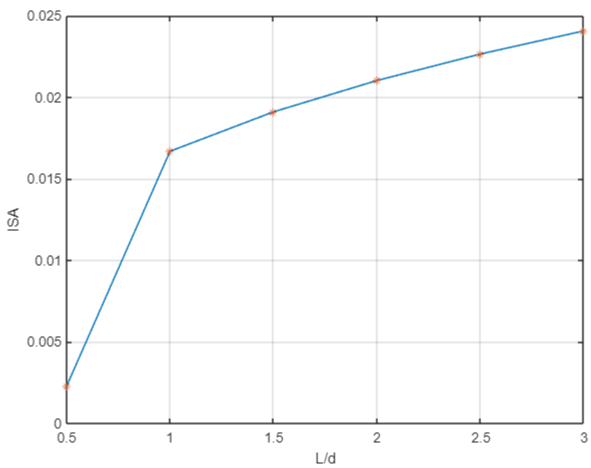


Figure 4.7: (L/d) v/s CSA for Thrusters

| L/d | r (m) | d (m) | L (m) | CSA ($m^2$) |
|---|---|---|---|---|
| 0.5 | 0.019 | 0.038 | 0.038 | 0.0022 |
| 1 | 0.015 | 0.030 | 0.030 | 0.0028 |
| 1.5 | 0.013 | 0.026 | 0.039 | 0.0032 |
| 2 | 0.012 | 0.024 | 0.048 | 0.0036 |
| 2.5 | 0.011 | 0.022 | 0.055 | 0.0039 |
| 3 | 0.010 | 0.020 | 0.60 | 0.0041 |

Table 4.12: **(L/d) v/s CSA for Thrusters**

•The corresponding output values were calculated using the same equations from section 4.2.2.1.

The table below shows the dimensions of the thruster combustor chamber.

| L/d | r (cm) | d (cm) | L (cm) | CSA ($cm^2$) | $V_c$ | $A_c$ | $A_c/A_t$ | $L_c$ | $L_{cone}$ | $\theta$ |
|---|---|---|---|---|---|---|---|---|---|---|
| 0.5 | 1.9 | 3.8 | 1.9 | 22 | 21.75 | 11.4 | 52.142 | 1.42 | 0.48 | $10^o$ |

Table 4.13: **Parameters of Combustion Chamber for Thrusters**

### 4.2.2.3 | Thickness of the Chamber

•To calculate the thickness of a chamber, the first step is to decide the material for the chamber.

•Material for Combustion Chamber is Inconel 625.

•Inconel 625 is a high-performance nickel-chromium-molybdenum alloy known for its high level of strength, temperature resistance, and corrosion resistance.

| Ni | Cr | Mo | Fe | Nb+Ta | Co | Mn | Al |
|---|---|---|---|---|---|---|---|
| 58-71% | 21-23% | 8-10% | 5% | 3.2-3.8% | 1% max | 0.5% max | 0.4% max |
| Working Stress (S) = 827.375 MPa | | | | | | | |

Table 4.14: **Chemical Composition of Inconnel 625**

• Thickness of the wall can now be calculated from the equation:

$$t_w = \frac{PD}{2S}$$

•The thickness of the combustion chamber is found to be **6.66** $\cdot 10^{-5}$ **m** for **700 N** and **2.77** $\cdot 10^{-5}$ **m** for **50 N**.

## 4.3 | Tank Design

### 4.3.1 | Material Selection

•We choose Titanium 6AL-4V as the material for the tank because of its high tensile strength to weight ratio and low thermal conductivity at low temperature.
•Availability and the knowledge about the alloy made us choose this material.
•Few other composites are also considered but not selected because of not enough research is done on them.
•The table below will have all the information about the material required for the design.

| **Properties of Titanium 6AL-4V** | |
|---|---|
| Material | Titanium 6AL-4V |
| Young's Modulus(E) | 113.8 GPa |
| Ultimate Strength($F_u$) | 950 MPa |
| Yield Strength($F_y$) | 880 MPa |
| Poisson's Ratio($\nu$) | 0.342 |
| Weld Efficiency($E_w$) | 1 |
| Density ($\rho$) | 4430kg/$m^3$ |

Table 4.15: **Properties of Titanium 6AL-4V**

### 4.3.2 | Shape

We choose spherical shape for the following reasons

•Low surface area for a given volume.
•Higher structural tolerance than other shapes.

Low weight and low heat transfer are direct consequences of low surface area.

### 4.3.3 | Dimensions

•The O/F ratio and other values obtained in the previous segments were used as the inputs in calculating the dimensions.
•Table showcases the inputs taken.

| Propellants | Mass for the mission(Kg) | Density(Kg/m$^3$) | Volume of Propellant(m$^3$) | Ullage% | Total Volume(m$^3$) |
|---|---|---|---|---|---|
| Lox(Oxidizer) | 1027.32 | 1140 | 0.901 | 3 | 0.928 |
| $CH_4$(Fuel) | 387.67 | 424 | 0.914 | 3 | 0.942 |

Table 4.16: **Inputs for Propellant Tank Design**

•From the literature review, we obtained equations to calculate the dimensions of the tank.

•Table below shows the dimensions of both fuel and oxidizer tank.

| Tank Properties | Equation used | Oxidixer |
|---|---|---|
| Radius(R) | $V = \frac{4}{3}\pi r^2$ | 0.6051 m |
| Surace Area(S) | $A = 4\pi r^2$ | 4.6m$^2$ |
| Chamber Pressure($P_c$) | From Engine Design | 1.2 MPa |
| Maintained Pressure Tank($P_s$) | Chamber pressure(1.2MPa) plus possible pressure drop(0.8 MPa) | 2 MPa |
| Maximun Pressure in tank($P_m$) | **$P_s$ x F.o.S(2.5)** | 5 MPa |
| Maximun Stress Allowed($S_a$) | Min of ($F_y$/1.10) and ($F_u$/1.25) | 760 MPa |
| Wall Thickness | $t_s = \frac{P_m R}{2 S_a E_w}$ | 1.99 mm |
| Tank Weight($W_t$) | $W_t = \rho S t_s$ | 40.55 Kg |
| Critical Pressure($P_{cr}$) | $P_{cr} = \frac{2Et_s^2}{R^2}\sqrt{3(1-\nu^2)}$ | 4.048 KPa |

Table 4.17: **Dimensions of Oxidizer Tank**

| Tank Properties | Equation used | Oxidixer |
|---|---|---|
| Radius(R) | V = $\frac{4}{3}$ $\pi$ r$^2$ | 0.6081 m |
| Surace Area(S) | A = 4$\pi$r$^2$ | 4.647m$^2$ |
| Chamber Pressure(P$_c$) | From Engine Design | 1.2 MPa |
| Maintained Pressure Tank(P$_s$) | Chamber pressure(1.2MPa) plus possible pressure drop(0.8 MPa) | 2 MPa |
| Maximun Pressure in tank(P$_m$) | P$_s$ x F.o.S(2.5) | 5 MPa |
| Maximun Stress Allowed(S$_a$) | Min of (F$_y$/1.10) and (F$_u$/1.25) | 760 MPa |
| Wall Thickness | $t_s = \frac{P_m R}{2 S_a E_w}$ | 2 mm |
| Tank Weight(W$_t$) | $W_t = \rho S t_s$ | 41.17 Kg |
| Critical Pressure(P$_{cr}$) | $P_{cr} = \frac{2Et_s^2}{R^2}\sqrt{3(1-\nu^2)}$ | 4.048 KPa |

Table 4.18: **Dimensions of Fuel Tank**

# 4.4 | Feed System Design

## 4.4.1 | Type of Feed System

•The Feed Systems are of two kinds; Gas Pressurized System and Turbopump System. Since the mission is a low thrust mission, gas pressurized system is preferred.

•In a gas pressurized system, the pressure gas is an inert gas. Hence, Helium is selected considering its low weight and inertness.

•The architecture of this mission is already been revealed earlier (Section 4.1.2).

## 4.4.2 | Sizing of Pressure Tank

•Similar to propellant tanks, a tank for pressurized gas is needed.

•The spherical tank is selected as it provides the least surface area for a certain amount of volume.

•From literature review, Ti-6AL-4V is selected. It's Properties are already discussed in Section 4.3.1.

•To get the values, we used the relation between Pressure and Volume.

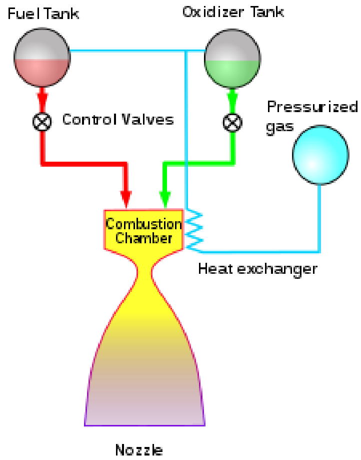


Figure 4.8: Gas pressure feed system

$$P_o V_o + P_f V_f = P_g V_g$$

Where,

1. $P_o$, $V_o$, $P_f$, $V_f$ are pressure and volume of oxidizer and fuel respectively

2. $P_g$ and $V_g$ are pressure and volume of Helium.

•To calculate further, certain volume of 200 L is assumed and Helium pressure is calculated.

•Once, pressure is known, we used the ideal gas equation to calculate the mass of the Helium at 85 K.

•The following table shows the input for Helium tank design:

| Helium Tank Inputs | | | |
|---|---|---|---|
| | **Mass** | **Density** | **Volume** |
| Helium | 5.26 Kg | 145 Kg/$m^3$ | 0.2 $m^3$ |

Table 4.19: **Helium Tank Inputs**

•With similar approach as propellant tanks, the other dimensions of the Helium tank were calculated.

•The following table shows the dimensions of the Helium Tank.

| Tank Properties | Equation used | Helium |
|---|---|---|
| Radius(R) | $V = \frac{4}{3}\ \pi\ r^2$ | 0.362 m |
| Surace Area(S) | $A = 4\pi r^2$ | $1.646 m^2$ |
| Maintained Pressure Tank($P_s$) | $P_o V_o + P_f V_f = P_g V_g$ | 18.6 MPa |
| Maximun Pressure in tank($P_m$) | $P_s$ x F.o.S(3) | 55.8 MPa |
| Maximun Stress Allowed($S_a$) | Min of ($F_y$/1.10) and ($F_u$/1.25) | 760 MPa |
| Wall Thickness | $t_s = \frac{P_m R}{2 S_a E_w}$ | 4.43 mm |
| Tank Weight($W_t$) | $W_t = \rho S t_s$ | 32.3 Kg |

Table 4.20: **Dimensions of Helium Tank**

## 4.4.3 | Pressure Drop Study

•In any system, the losses play a vital role in determining its efficiency. Similar is the case with feed systems.

•The pressurized gas and propellants are kept at a certain pressure but there exists various points where pressure drop can happen.

•The pressure drop mainly occurs in the following regions:

1.Storage pressure drop

2.Dynamic pressure loss

3.Line Losses

(i) Frictional losses in pipe

(ii) Pressure drop in flow restrictions.

•Major of the losses occurs due to line losses.

#### 4.4.3.1 | Line Losses

*** Line Losses (Piping):**

•The friction is neglected in almost every ideal condition, but in practice, is always present. The presence of friction in pipes contributes to the pressure drop.

•The pressure drop is given why the mathematical equation:

$$\Delta p_f = \rho g_o H_f$$

Where $H_f$ is head loss and is given by,

$$H_f = f \frac{Lu^2}{2dg_o}$$

Where f is the friction factor, d is the pipe diameter, L is pipe length and u is the flow velocity.

•Therefore, the substituting Hf in pressure drop equation, we have,

$$\Delta p_f = f\rho \frac{Lu^2}{2d}$$

*** Line Losses (Restrictions):**

•The pipe is added with equipments like orifices and valves to perform their respective functions.

•Though the presence of orifices and valves is necessary for the proper working of feed system, they often end up contributing to the pressure drop.

•The pressure drop across an orifice is calculated by using the coefficient of discharge ($C_d$). The pressure is hence given by,

$$\Delta p = \frac{(\frac{\dot{m}}{AC_d})^2}{2\rho}$$

•The pressure drops are tried to be kept as low as possible.

5

# Summary and Conclusion

## 5.1 | Summary

**•Payloads selected are Lymann Alpha Photometer and Terrain Mapping Camera.**

**•The basics of Hohmann Transfer Orbit are understood and Delta V is calculated.**

**•The propellant selected for the mission is LoX + CH because of its high $I_{sp}$.**

**•Mass calculation and number of stages are selected based on Mass v/s Stages graph observation.**

**•The four main domains for the designing of the propulsion system are; System Engineering, Engine Design, Tank Design and Feed System Design.**

**•System Engineering involved thrust calculation, feed system architecture and mass budgeting.**

**•Engine Design involved the calculating the parameters of Nozzle and Combustion Chamber.**

**•Tank Design involved the material selection, shape and designing the dimensions of propellant tanks.**

**•Feed System Design involved the selection of Feed System for the mission, designing the tank for the pressurized gas.**

## 5.2 | Conclusion

•The propulsion system is designed to carry a payload mass of about 20 Kg from LEO to 200km above the Martian surface. The key factor of the system is the propellant selected. Hence, the feasibility of the system is promising in the near future with further development in ZBO technology.

# 6

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