Analysis and Design of a Propulsion System for an Interplanetary Mission to Venus

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Abstract: Interplanetary missions have always been a field of interest for humans. We have always wanted to explore why life exists the way it does on our planet and not on other neighboring planets. We ponder upon questions like: "Didthe other planets have life earlier?" or "Will we able to thrive on other planets?" To try to answer these questions, we must be able to study the planet's conditions.

Currently, there has been a great demand for space missions and also a need tomake these missions affordable, efficient and safe. Venus, our sister planet, hasmany mysteries that need to be unveiled. By understanding the conditions of planets like Venus, we could figure out what makes Earth a haven for human life. In this paper, we have designed a propulsion system, for a spacecraft, thatwould take a payload from Earth's orbit (400km from surface) to a desired orbit around Venus, to unveil these mysteries.

Keywords: propellant, chemical rocket propulsion, CubeSat, numerical analysis, simulations, specific impulse etc.

1. Introduction

The endeavor to launch a spacecraft to Venus is to increase the understanding on our sister planet, Venus. Venus's mysteries have amused us all including the recent discovery of phosphine gas in the planet's upper atmosphere that could turn scientists' gaze to a planet long overlooked in the search for extraterrestrial life.

Today, the second planet from the sun has an atmosphere stifled by carbon dioxide gas, and surface temperatures that average more than 800 degrees Fahrenheit. The dense atmosphere of Venus exerts a pressure of more than 1,300 pounds per square inch on anything at the surface. That is more than 90 times the 14.7 pounds per square inch at sea level on Earth, or the equivalent to being 3,000 feet underwater in the ocean. High in the toxic atmosphere of the planet Venus, astronomers on Earth have discovered signs of what might belife. Often called Earth's twin, Venus is roughly the same mass as Earth. Many scientists think that Venus was once covered in water and possessed an atmosphere where life as we know it could have flourished.

With this mission, certain mysteries of our sister planet will be unveiled just like the previous Venera missions, Vega missions and Venus Express mission. The main objective of this project is to design a propulsion system for an inter- planetary mission to Venus which carries a CubeSat for studying the unknown UV absorbers present in the Venusian atmosphere. This is done to determine the nature, concentration and distribution of UV absorbers and to understand the overall radiative and thermal balance of the planet along with the atmospheric dynamics and the chemistry of the upper clouds.

2. Methodology

a. | Payload Selection

1. CubeSat UV Experiment

Recent study has shown that an unknown absorber is Venus' clouds absorbs in the UV 50 percent of the incoming solar radiation. The absorbed energy is theprimary atmospheric engine of Venus. This mission payload will be a

CubeSatwith a mass of 180kgs which will be released in a polar orbit at an altitude of 75kms its purpose would be to Characterize Venus' unknown UV absorbers [6].A trade off study was conducted on this payload option:–

a. Scientific Importance: This mission will help determine the nature, concentration and distribution of UV absorbers to understand the overall radiative and thermal balance of the planet, atmospheric dynamics and the chemistry of the upper clouds [6]. Hence the CubeSat will provide high resolution UV spectrum of Venus

b. Feasibility: The CubeSat is an compact yet effective piece of equipment and its use in previous missions makes it a highly feasible payload. It's also generally employed as a secondary payload.

c. Technology Readiness Level: The instruments for this mission are readily avail- able and have also been used by NASA in previous missions.By assessing the above factors, we decided to score each of the tradeoff pointsout of 10.

	Scientific importance	TRL	Feasibility	Total
CubeSat	8	10	9	27
Crash Landing Probe	9	6	7	22
Topology Study	8	10	8	26

Table 0.1: Trade-off study

Hence, we decided to go ahead with the CubeSat UV Experiment whichis focused at studying the dense clouds in the Venusian atmosphere which ab- sorbs the UV radiation emitted by the Sun.

b. | Mission Profile

i. | Trajectory design

Our spacecraft has to reach from earth's orbit (400km above surface) to an orbit around Venus (6126.8km X 50,000km – from the center of Venus).A Hohmann transfer orbit is used to transfer between two circular orbits from basic orbital mechanics [7]. A Hohmann transfer has been considered from earth to Venus because in gen- eral (not always) it takes the least amount of fuel. This basically involves starting from a circular orbit around earth, then per- forming a prograde burn to get into an elliptical orbit around the sun while having escaped earth. This point will be the apoapsis around the sun (furthest distance from sun). When the spacecraft reaches Venus, it will be at some distance from its surface. Here, another burn (retrograde) is performed to make the spacecraft get cap- tured into an orbit around Venus. This point around which the burn will be performed will be the periapsis of the elliptical transfer orbit around the sun. This point will also be the periapsis of an elliptical orbit around Venus.

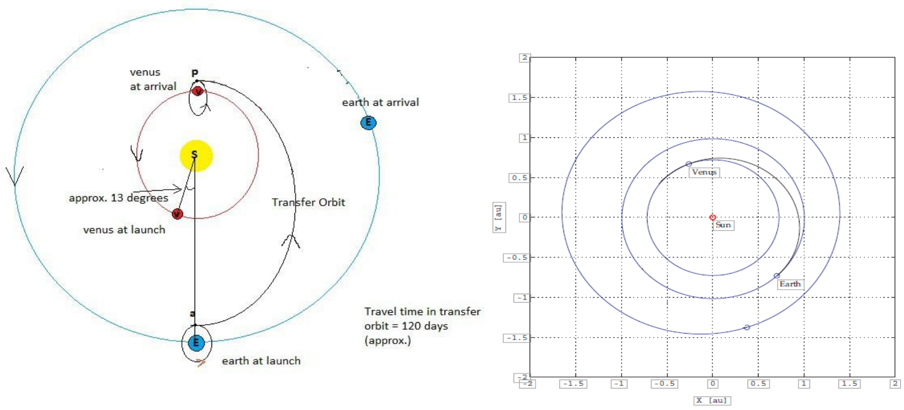

Figure 0.1: Trajectory Design

ii. | Δv calculation

We know that,

Velocity of earth w.r.t sun = 29.78km/s Velocity of Venus w.r.t sun = 35.10km/sAlso,

$\mu_s = 1.327 \times 10^{11} km^3/s^2$

$\mu_e = 3.986 \times 10^5 km^3/s^2$

$\mu_v = 3.249 \times 10^5 km^3/s^2$

Initial orbit around Earth:

The orbit is circular - 400km from surface:

$$v = \sqrt{\frac{u_e}{R+400}}$$

We perform a prograde burn in this orbit at point 'a' that will get us into our desired Hohmann transfer orbit.(elliptical)

Hohmann Transfer Orbit:

Ellipse with a sun at one focus, point 'a' at apoapsis and point 'p' at periapsis.Speed at 'a' and 'p' in transfer orbit is given by[7]

$$V = \sqrt{2u_s\left(\frac{1}{r} - \frac{1}{2a}\right)} \quad (2)$$

where,

r = distance from sun 2a = 2(semi-major axis)

2a = $r_e + r_v = 225.46 \times 10^6 km$

r_e, r_v are the distances of earth and Venus from the sun respectively.Using the equation 1 , we get,

speed at a, $V_a = 27.25 km/s$speed at p, $V_p = 37.85 km/s$

Now, our escape trajectory from earth will be Hyperbolic w.r.t sunHyperbolic excess velocity, $v_\infty = 27.25 - 29.78 = -2.53 km/s$

The negative sign shows that the spacecraft will be slower than earth w.r.t sun.

Using energy equation for hyperbolic trajectories [7],

$$\frac{v_\infty^2}{2} = \frac{v^2}{2} - \frac{\mu_e}{r} \quad (3)$$

$$r = R + 400km$$

$$v = \sqrt{V_\infty^2 + \frac{2u_e}{r}} = 11.4\ km/s$$

Therefore, our spacecraft should increase the velocity from 7.67 km/s to 11.14 km/s (from equation 1)

$\Delta v_1 = 11.14 - 7.67 = 3.47 km/s$

$\Delta v_1 = 3.47 km/s$

Desired orbit around Venus (6,126.8 x 40,000 km):

We know that speed at periapsis of orbit around Venus is 9.59 km/s (using vis- viva equation).

When we reach periapsis 'p' of transfer orbit, it will be an Hyperbolic escape trajectory w.r.t to Venus. So, we have to perform a retrograde burn to slow it down.

$$v = \sqrt{V_\infty^2 + \frac{2u_e}{r}}$$

$v\infty = vp - v_{V/S}$

$(v_\infty = 37.85 - 35.10)$

$v = 10.66 km/s$

Therefore, we must decrease the velocity from 10.66 km/s to 9.59 km/s,

$\Delta v_2 = 10.66 - 9.59 = 1.07 km/s$

Thus, the total Δv for the mission with buffer for trajectory correction maneuvers is,

$\Delta v = \Delta v_1 + \Delta v_2$

$\Delta v = 4.6 km/s$

The time of flight in transfer orbit is 120 days. And to ensure that our spacecraft intersects Venus, Venus must be approximately 13 degrees behind earth whilelaunching.

c. | Mass Budget and Staging

Mass Budgeting is the calculation of various mass related parameters for a spacecraft. This includes the dry mass, propellant mass and the payload mass, for each stage. Thus, an overview on the total mass of the spacecraft is obtained using this study, which can be kept into account while designing the various subsystems of the spacecraft. Also, the study gives a prior insight into the mass distribution on each stage of the spacecraft. As far as our problem statement is concerned, the dry mass for each stage was defined and the mass of payload for our mission was literature reviewed. Thus, the study gave the values of the propellant needed for each stage. Therefore, mass of each stage can be simply obtained by summing the propellant mass,the dry mass and the payload mass, of that stage. Staging, in simpler terms, is stacking up the various sections of a rocket in a defined or sequential manner. The advantage of staging is that it becomes eas- ier for the rocket to get to a desired orbital speed without carrying any excess mass (such as empty propellant tanks and early-stage rockets) with it. So, the fuel/oxidizer of each stage is completely used up and the stage is jettisoned. Thus, staging reduces the propellant requirement of the rocket as comparedto a single-stage-to-orbit spacecraft (SSTO). But there is a certain limit up-to which stages in a rocket can be used to optimize fuel consumption. This limit is calculated in staging study. Thus, there has to be a optimum staging study forit [9][4][14][5]. For our study, the Mass Budget of stages were calculated using Microsoft Excel. Using this, the propellant mass of each stage and the total propellant requiredwas found out. This was simply done using the Ideal Rocket equation.

$$\Delta v = \mathrm{I_{sp}} \text{ x } \mathrm{g_0} \text{ x } \ln\frac{\mathrm{m_i}}{\mathrm{m_f}} \qquad (4)$$

$$m_i = m_f \text{ x } \frac{\Delta v}{e^{Is_P \times g_0}} \qquad (5)$$

Where,

Δv = change in velocity of the stage (m/s).

I_{sp} = Specific impulse (s).

m_i = Initial Mass of the stage (Kg).

m_f = Final mass of the stage (Kg).

A MATLAB program was also generated for the study which gave the mass parameters and the optimum staging condition for the required conditions. A graph of Total propellant consumed v/s the total no. of stages was one of the outputs of the program. The input parameters are namely the total no. of stages, the dry mass condition, total ΔV requirement and payload mass. The best part of using MATLAB program is that, it can be used for 'n' stages with any user defined input conditions. The same thing done in Excel can be- come tedious for greater number of stages, thus the program helps the user in it. We know that the payload mass for our mission is 180kg. Also, the dry mass of spacecraft is defined as 10 times the payload mass. The delta-v for the mission is 4.6km/s and the I_{sp} is 310 s. Thus, applying it in Rocket equation,we get the Propellant requirement of 5309.4kg.

Delta V2 = 2300m/s, Isp = 310s				
Stage 2				
Md(stage 2)	Payload	Mf	Mi	Mp
1000	180	1180	2514.539	1334.5394

Delta V1 = 2300m/s, Isp = 310s				
Stage 1				
Md(stage 1)	Payload	Mf	Mi	Mp
1000	2514.539	3514.539	7489.363	3974.823

Total Propellant Mass:	5309.363	kg
Total Mass of Spacecraft :	7489.363	kg

Figure 0.2: Mass tabulation for 2 stages

Now, applying the same conditions using the MATLAB program, we get values of 5309.4kg for 2 stages and 7011.2kg for 1 stage (see table 0.2 for reference). Since we have a function, we'll calculate the value for a series of stages. For now, a total of 5 stages has been considered for validation.

	1 stage	2 stage	3 stage	4 stage	5 stage
Mass of propellant(kg)	7011.2	5309.4	5388.7	6069.3	7143.3

Table 0.2: Mass of propellant required for different stages

From the graph 0.3, we can see that the propellant requirements increases after the third stage. Thus, for the design of our spacecraft, 2 stages would require the least amount of propellant for the mission. Thus, it can be validated from

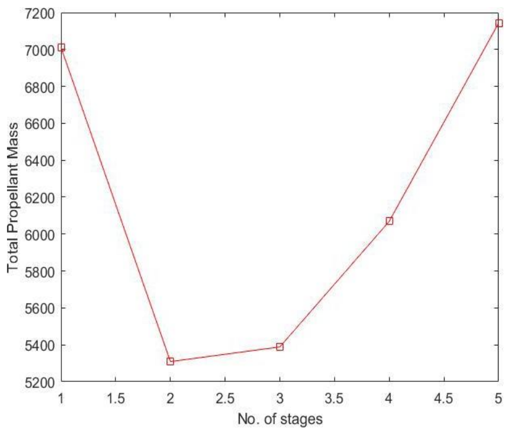

Figure 0.3: Plot of total propellant mass v/s no. of stages

The graph that two stages require least amount of propellant for our mission.

d. | Propellant Selection

LOX + LH2:

This liquid fuel has been extensively used by various space research organizations and has very high Isp values. The fuel is also highly cryogenic and haslow density values which will account for a greater fuel mass and a complicated system design due to which we decided not to use this propellant [11].

UDMH +N_2O_4:

This bipropellant liquid rocket fuel turned out to be the ideal fuel for our mission. It has a 300+ Isp value as well as a high density which would in turn reduce the mass of propellant leading to an overall reduction in propellant mass [17].

Characteristics of liquid propellant:

1. The chosen propellant was bipropellant liquid rocket. This propellant generally uses a liquid fuel and liquid oxidizer.

2. Liquid-propellant rockets can be throttled and have control of mixture ratio; they can also be shut down, and, with a suitable ignition system or self-igniting propellant, restarted. All liquid rocket engines have tankage and pipes to store and transfer propellant, an injector system, a combustion chamber which is very typically cylindrical, and one (sometimes two or more) rocket nozzles. Liquid systems enable higher specific impulse than solids and hybrid rocket motors and can provide very high tankage efficiency.

3. After a literature survey and analyses, we finalized the propellant as UDMH (Unsymmetrical Di-Methyl Hydrazine) as liquid fuel and N_2O_4 as liquid oxidizer having properties as mentioned below was selected as it had all the useful parameters readily available, required for further calculations, comparatively.

Performance and combustion properties[17]:

1. Density – 1180 kg/m^3

2. Specific impulse – 310s

3. Temperature of combustion – 3415 C

4. Ratio of specific heat – 1.25

5. Universal gas constant - 375 J/kg-K

6. Thrust to weight ratio – 10^{-1} – 10^3

7. Specific power – 10^{-2} – 10^2 kW/kg

8. Molar product mass – 22.16 g/mol

9. O/F ratio – 2.6

e. | Systems Engineering

i. | Thrust determination

Thrust for the mission is estimated from the previous Venus express mission[13]. The thrust is divided for 8 thrusters and one main engine.

Total Launch Mass	1245 kg
Propellants	570 kg

Payload	94 kg
Propulsion	414 N , 317 s
Thrust to weight ratio	0.4

Table 0.3: Systems engineeringThrust required = 0.4 × $TotalMass$

Thrust required = 0.4 × 6849Thrust required = 2760N

Main Engine Thrust = 1960N Thruster's total thrust = 8 × 100N

ii. | System Architecture

The gas pressure feed system is designed in such a way that it requires minimum number of components to attain minimum weight. The system is de-signed to fulfil the safety, reliability, control and re-usability requirements with fewer components. Isolation valves shut off a section of the system in case of leak and also pro-vides a path for fluid to flow during filling and venting. Check valves prevent back pressure flow of the fluid. Pyrotechnic valves are one time cut off valve actuated by a small electric pulse [1]. They provide safety to the system by cut- ting off the fluid during back pressurization and prevent leakage. The valve is added before the manifold so that the risk of fuel and oxidizer to come in contact before combustion is eliminated. The filter is required to remove dust particles or any other debris before it enters the regulator and the combustion chambers. Pressure transducer and thermocouple are used to measure and monitor the condition of the spacecraft through computers. Fuel and oxidizer for the thrusters are branched out from the main tanks. Thepyrotechnic valve is used as an emergency cut off valve and also to control theflow of the fluid for maneuvering.

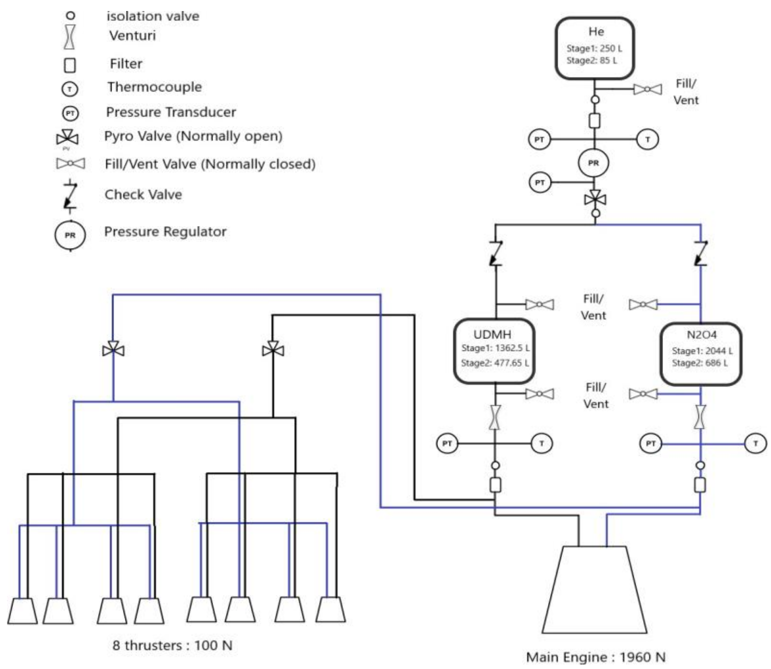

Figure 0.4: Systems architecture of the propulsion system

Component	Number of components
Gas Tank (He)	1
Propellant tanks	2
Ball valves	5
Check valves	2
Pyrotechnic valves	3
Isolation valve	4

Venturi	2
Pressure transducer	3
Thermocouple	3
Filter	3
Pressure Regulator	1

Table 0.4: List of components in system architecture

f. | Engine Design

i. | Main Engine Design

The fuel and oxidizer are mixed in the combustion chamber after they are atomized by passing through the injectors. This combustion produces hot exhaust which is passed through a nozzle to accelerate the flow and produce thrust. Thus, design of engine plays a very important role.

Engine design consists of calculating dimensions, specific impulse otherparameters like exit velocity of the engine based on given thrust requirement. Here, we have looked into the combustion chamber, throat nozzle. The injectorhas not been discussed in this report. But a good choice for our requirementwould be a coaxial swirl injector.

Moving forward, we will now discuss the input parameters we have and then look into the outputs we have to calculate for one main engine (1960N-thrust) and a small thruster (100N-thrust).

The propellant that has been selected is UDMH+N_2O_4.

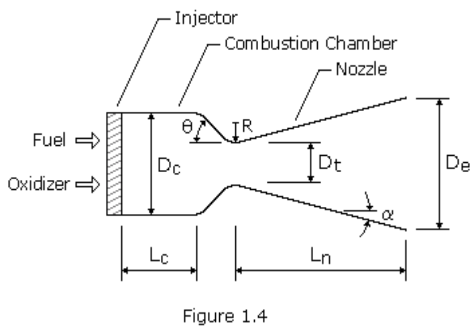

Figure 0.5: Rocket Nozzle

We have some input parameters based on propellant data thrust value asgiven earlier in section 2.5.1.

INPUT PARAMETERS:

- Thrust
- Chamber Pressure
- Exit Pressure
- Oxidizer Fuel Ratio
- Ratio of specific heats
- Combustion temperature
- Characteristic velocity.

All the above-mentioned parameters have to be computed first in order to perform the further calculations for the output parameters.

THRUST

Thrust for the main engine is 1960N and 100N for the thruster as mentioned in section 2.5.1.

CHAMBER PRESSURE AND EXIT PRESSURE

Pressure that is maintained inside the combustion chamber for a rocket engineis fairly high, that is from 10 to 200 bar and the pressure with which the exhaust gases leave the nozzle is the exit pressure Pe. From the propellant data[17][2], the optimum chamber pressure Pc is 15 bar. The exit pressure Pe is 1000 Pascals from the reference data from the previous rocket engines.

COMBUSTION TEMPERATURE

The combustion temperature Tc is the lowest temperature in which the fuel and oxidizer spontaneously ignites external sources of ignition inside the combustion chamber. The combustion temperature for UDMH/N_2O_4 mixtureis 3415 K.

OXIDIZER TO FUEL RATIO

A certain ratio of oxidizer weight to fuel weight in a combustion chamber will usually yield a maximum performance value. This ratio is defined as the optimum mixture ratio. But in practical applications the optimum mixture ratio is slightly higher than the stoichiometric mixture ratio. This is because a gas which is slightly rich in fuel tends to have a lower molecular weight. This results in a higher overall engine system performance.

By using this stoichiometric combustion equation

$$C_2H_8N_2 + 2N_2O_4 \longrightarrow 2CO_2 + 4H_2O + 3N_2$$

The stoichiometric oxidizer to fuel ratio is 3.06

For UDMH/N_2O_4 combination of fuel and oxidizer the optimum O/F ra- tio is 2.6 from the propellant data[17].

The input parameters values are listed in the Table 0.5

CHAMBER PRESSURE (P_c)	$1.5x10^6$ Pascals
EXIT PRESSURE (P_e)	1000 Pascals
O/F RATIO	2.6
Gamma (γ)	1.25
COMBUSTION TEMPERATURE (T_c)	3415 K
CHARACTERISTIC VELOCITY (C^*)	1720 m/s
THRUST	1960N

Table 0.5: Main engine Inputs

OUTPUT PARAMETERS:

- Exit velocity
- Exit Mach Number
- Total Mass Flow Rate
- Specific Impulse
- Coefficient of Thrust

- Area Ratio
- Thermodynamic values at throat and exit

GAS CONSTANT CALCULATION:

From the propellant data[17], we know that the value of C^*.Using the formula,

$$C^* = \sqrt{\frac{RT_C}{\gamma}} \times [\frac{Y+1}{2}]^{\frac{Y+1}{2(Y-1)}} \quad (6)$$

Thus, we get the value of R as 375 J/kg K.

Using the value of R, we get molecular weight as 22.161 grams.

EXIT VELOCITY

Exit velocity is the Velocity with which the exhaust gasses leave the nozzle.

$$V_e = \sqrt{(\frac{2yRT_c}{y-1}) \text{ x } (1-\frac{P_e}{P_c})^{\frac{y-1}{y}}}$$

$$V_e = 3137.57 m/s$$

EXIT MACH NUMBER

Exit Mach number is the Mach number with which the exhaust gasses leave

the nozzle and is calculated using the equation 8.

$$M_e = \frac{v_e}{\sqrt{YRT_e}}$$

$$\frac{T_e}{T_C} = \left(\frac{P_e}{P_C}\right)^{\frac{1-1}{Y}}$$

$$T_e = 790.83K$$

$$M_e = \frac{v_e}{\sqrt{YRTe}}$$

$$M_e = 5.15$$

AREA RATIO

The ratio of the exit area to throat area of a nozzle is termed as the area ratio. As area ratio (AR) increases, the specific impulse increases, due to higher expansion of hot gas which generates higher velocity at nozzle exit.

$$\frac{A_t}{A_e} = \frac{r+1}{2}^{\frac{-u+1}{2(y-1)}} \text{ x } (\frac{1+\frac{(y-1) \text{ X } M^2}{2}}{M}^{\frac{Y+1}{2(Y-1)}})$$

TOTAL MASS FLOW RATE

For calculating the mass flow rate, three equations are considered.

$$F = (\dot{m}_{total} \times V_e) + P_{ee} \quad (10)$$

$$\frac{A_e}{A_t} = 82.48$$

$$\dot{m}_{total} = \frac{A_t \text{ x } P_c}{C^*}$$

Solving these three equations which has three unknowns namely A_e, A_t and

$\dot{m}_{total}$

we get,

A_e = 609.1148cm^2

A_t = 7.385cm^2

$\dot{m}_{total}$ = 0.644 kg/s

The mass flow rate for fuel and oxidiser has to be computed separately fordetermining the amount of each required in the combustion process.

Thus,

$\dot{m}_{fuel}$ = 0.3577 kg/s

$\dot{m}_{oxidizer}$ = 0.2862 kg/s

SPECIFIC IMPULSE

Mathematically, the Isp is a ratio of the thrust produced to the weight flow of the propellants. The word "specific" just means "divided by weight". The specific impulse Isp is given by:

$$I_{sp} = \frac{F}{\dot{m} \times g_0} \quad (13)$$

$$I_{sp} = \frac{1960}{0.644 \times 9.81}$$

$$I_{sp} = 310s$$

COEFFICIENT OF THRUST

The thrust force of a jet-propulsion engine per unit of frontal area per unit ofin-compressible dynamic pressure is the thrust coefficient C_f.

$$C_f = \frac{I_{sp} \times g}{C^*} \quad (14)$$

$$C_f = \frac{310 \times 9.81}{1720}$$

$$C_f = 1.768$$

THERMODYNAMIC VALUES AT THROAT AND EXIT

$$T_t = T_c \left(\frac{2}{1+\gamma}\right) \tag{15}$$

$T_t = 3035.55K$

$$P_t = P_c \times \left(1 + \frac{\gamma - 1}{2}\right)^{\frac{-\gamma}{\gamma+1}} \tag{16}$$

CHAMBER CHARACTERISTICS

OUTPUT PARAMETERS

- Chamber Volume
- Chamber Length
- Chamber Area

CHAMBER VOLUME AND CHARACTERISTIC LENGTH

For sufficient time to ensure complete mixing, atomization, vaporization and combustion. This time is termed as stay time and has the predominant effect on combustion efficiency. The total combustion process, starting from injection of propellants to the completion of chemical reaction and conversion of productsin to hot gas, requires finite time and volume. The rate of combustion and hence the stay time depends on propellant injection conditions, combustion chamber geometry and injector design. The combustion volume has a definite effect on combustion efficiency and is a function of mass flow rate of propellants.

Characteristic length (L^*) can be used to specify the propellant stay time in the chamber. The Characteristic length is defined as the ratio of chamber volumeto the nozzle throat area. This can be given as

$$L^* = \frac{V_C}{A_t} \tag{17}$$

While designing the combustion chamber, proper value of L^* is to be consid- ered because an increase in L^* beyond a certain point results in

- Higher thrust chamber volume and weight
- Creates more surface area and hence more cooling requirements
- Increased frictional losses at the chamber walls reducing nozzle stagnationpressure and hence the resultant

$L^* = 80cm$

$$\frac{80}{7.385}\ L =$$

$V_c = 590.8cm^3$

CHAMBER LENGTH:

The graph for Throat diameter - chamber length is plotted as shown in 0.6and the chamber length is considered to be 13cm after interpolating from the graph[3]. $L_c = 13cm$

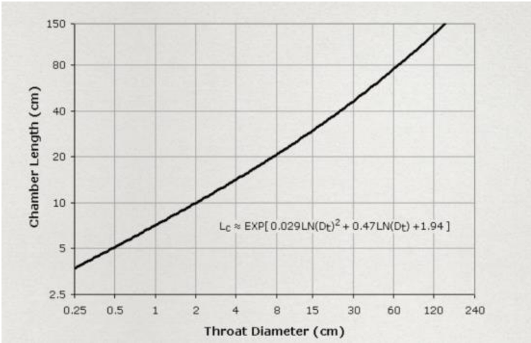

Figure 0.6: Plot of Throat diameter v/s chamber length

CHAMBER AREA:

$A_c = \frac{VC}{L_c}$

$A_c = 45.4461 cm^2$

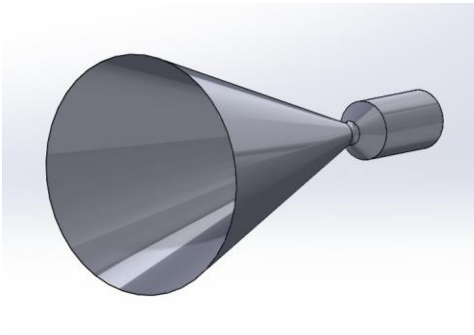

Figure 0.7: 3D model of main engine

ii. | Thruster Design

There are 8 small thrusters for altitude control and trajectory correction for our spacecraft, each with a thrust of 100N. Same methodology is applied for the design of the small thrusters. The output parameters are obtained using the same formulae as used earlier and is tabulated as show in table 0.6.

Mass flow rate	0.03187 kg/s
Area at the exit (A_e)	30.1052 cm^2
Area at the throat (A_t)	0.365 cm^2
Isp	320.5
Thrust coefficient (C_f)	1.828
Volume of the chamber (V_c)	29.2 cm^3
Diameter of the chamber (D_c)	2.5091 cm
Length of the nozzle (L_n)	11.752 cm
Length of the chamber (L_c)	5.905 cm
Mass of the chamber (M_c)	10.816 g
Mass of the nozzle (M_n)	40 g
Total mass of small engines	406.528 g

Table 0.6: Output parameters of thruster

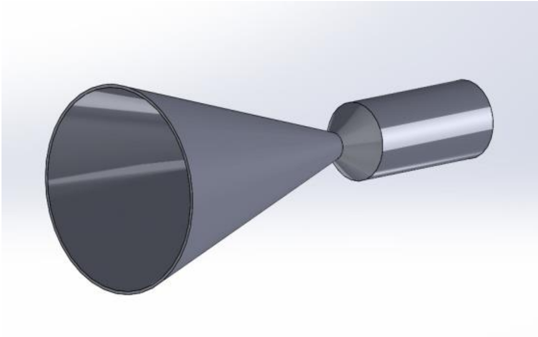

Figure 0.8: 3D model of thruster

g. | Feed Systems Design

Feed system in a propulsion system has the function of increasing the Enthalpyof the propellants in the tanks by raising the pressure and supplying them tothe combustion chamber at a required mass flow rate. Energy required for such operations either come from tanks with highly pressurized gas in it or with the help of centrifugal pumps.

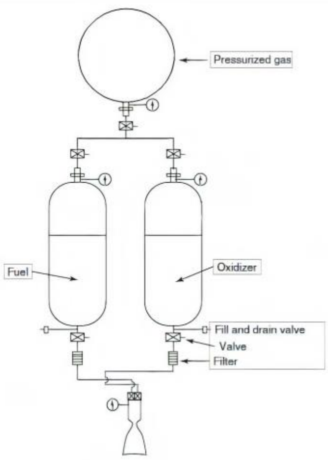

Figure 0.12: Pressure Feed System

i. | Calculation for feed system parameters

From thermodynamics, we can say that, the work done in expansion of highly pressurized gas in the gas tank, when the valve is opened, is equal to the work done by the gas in pushing the propellants to the combustion chamber.

Thus, Mathematically it can be written as[15]:-

$$P_g V_g = P_p(V_g + (V_f + V_o)) \quad (20)$$

where,

P_g = Pressure of the gas in the gas tank,

V_g = Volume of the gas in the gas tank,

P_p = Pressure required to push the propellants,

V_f = Volume of fuel,

V_o = Volume of oxidizer.

Calculation for Stage 1

From Mass Budgeting, we know that the propellant required for stage 1 is 3462.6kg.

Thus, we can say that,

$$m_f + m_o = 3974.82 \quad (21)$$

Here, m_f = Mass of fuel and m_o = Mass of oxidizer.

Also, we know that the ratio in which the fuel and oxidiser burn is 2.6

$$m_o = 2.6 \times m_f \quad (22)$$

Solving equation 21 and 22, we get m_f = 1104.1 kg and m_o = 2870.7 kgNow, $\rho_f = 793kg/m^3$ and $\rho_o = 1440kg/m^3$

Thus, $V_f = 1.39m^3$ and $V_o = 1.99m^3$

Also, we know that P_p = 23bar, $T_g = 288K$ Substituting the values in equation 20, we get,

$$\frac{333}{23} = \frac{V_g + 3.38}{V_g}; V_g = 250L$$

Note: The Pressure value 333 bar was selected by doing iterations on different pressure values. Pressure of 333 bar is selected as the volume of gas tank needs to be less and is inversely proportional to the Pressure.

Now, the mass of the gas required can be easily found out using the Ideal gas equation.

$$P_g V_g = m_g R T_g \quad (23)$$

For Helium, R = $\frac{8.314}{0.004} = 2078.5J/Kg.K$

Substituting the values in the equation 23 , we get, $m_g = 14kg$

Similarly, the same approach can be applied to the calculations of stage 2parameters,

$$m_f + m_o = 1334.54 \quad (24)$$

Thus, solving the equations 22 and 24,

$m_f = 370.7kg$ and $m_o = 963.84kg$

Thus, the volumes can be found out as same propellants are used.

$V_f = 0.467m^3$ and $V_o = 0.67m^3$

Substituting values in equation 20,

$$\frac{330}{23} = \frac{V_g + 1.137}{V_g}$$

$V_g = 85.3L$

By substituting the data in eq 23,

$m_g = 4.7kg$

Thus, the parameters of the gas, for the feed system, are tabulated as follows:-

	Stage 1	**Stage 2**
Gas	Helium	Helium
Pressure(bar)	333	330
Volume(m^3)	0.25	0.085
Temperature(K)	288	288
mass(kg)	14	4.7

Table 0.9: Feed system parameters

2.8.2 | Design of Helium tanks

The design of Helium tanks should be such that it should withstand the highly pressurized gas containing in it. Thus, the thickness of the tank and the material selection for the tank plays a vital role in it.

The thickness of the spherical tank is calculated for various materials. The material with best strength and giving least mass for the tank is considered for the design.

From the calculations earlier, it was inferred that the volume of the tank for stage 1 should be 250L and 85.3L for stage 2.

As we are using spherical tanks for simplicity,

$$V = \frac{4}{3}\pi r_i^3$$

Now, substituting the values for volume in the equation,we get,

$r_i = 0.39m$ (Inner radius for stage 1)

$r_i = 0.27m$ (Inner radius for stage 2)

Now, that we have the internal radius of tank, the thickness of the tank

would be the difference of the outer radius and inner radius.

$thickness = r_o - r_i$ The thickness of spherical tank as a function of Pressure,radius and Yield strength is given as [10]

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where,

P = Max. Pressure in the tank.

$$t = \frac{1.5 \times P \times r_i}{2 \times T} \quad (25)$$

r_i = Inner or mean radius of the tank.T = Yield strength of material.

Calculations were done using the equation 25 for different materials hav-ing good strength-to-weight ratio and good tensile strength.

Material	**T**(MPa)	**Pressure**(MPa)	r_i(m)	**thickness**(mm)	r_f(m)	**Mass**(kg)
Al5052	230	37.5	0.39	50.20	0.44	291.7
Al3003	130	37.5	0.39	88.81	0.48	577.1
Ti 6Al-4V	880	37.5	0.39	12.46	0.40	109

Table 0.10: Tank parameters for different materials (stage 1)

Since Ti 6Al-4V has the least mass and has good strength, we select the materialfor the stage 1 helium tank.

Similarly we apply the same approach to the design of the stage 2 tanks.

Material	**T**(MPa)	**Pressure**(MPa)	r_i(m)	**thickness**(mm)	r_f(m)	**Mass**(kg)
Al5052	230	37.5	0.273	35.14	0.308	100.05
Al3003	130	37.5	0.273	62.17	0.335	197.93
Ti 6Al-4V	880	37.5	0.273	8.72	0.281	37.37

Table 0.11: Tank parameters for different materials (stage 2)

As we can see from the table 0.11, Ti 6Al-4V gives least mass among the others.Thus, we select it for stage 2 tanks as well.

Thus, the tanks finalised has the following parameters:

Stage 1

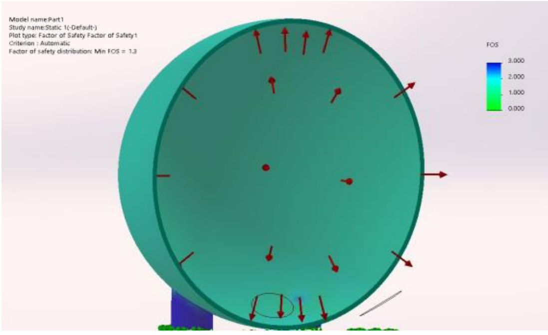
Material	**Mass of tank(kg)**	**Volume of tank(L)**	**Pressure(bar)**	**Min. F.O.S**
Ti 6Al-4V	109	250	375	1.3

Table 0.12: Stage-1 Helium tank

Stage 2

Material	**Mass of tank(kg)**	**Volume of tank(L)**	**Pressure(bar)**	**Min. F.O.S**
Ti 6Al-4V	38	85	375	1.4

Table 0.13: Stage-2 Helium tank



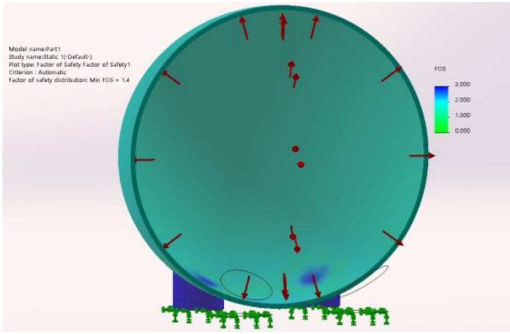


Figure 0.13: Static analysis of stage-1 helium tank(left) Figure 0.14: Static analysis of stage-2 helium tank(right)

3. Conclusion

A Propulsion system for a mission to Venus was designed along with the choice of payload. The mission profile including the trajectory and delta-v requirements were worked upon. The mass for each stage was calculated and based on the stage optimization study, the spacecraft was designed to be two staged due to its less fuel requirement. Different propellants were studied for the spacecraft and based on the trade-off study, $UDMH + N_2O_4$ were considered. The thrust required for the mission was reviewed and a thrust of approximately 2760N was calculated. The feed architecture for the spacecraft was designed with least number of components and maximum safety. The main engine and thrusters for the spacecraft were designed and were found to be 93 percent efficient. The propellants tanks for both the stages of the spacecraft were designed with pis-ton type of PMD. Gas pressure type of feed system was designed with highly pressurized helium gas tanks. Thus, the spacecraft can easily take a payload of180kg from earth's orbit (around 400km from earth's surface) to a desired orbit around Venus (6126.8 x 40,000 km).

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