KOÇ UNIVERSITY

MECH 427

ROCKET PROPULSION TERM PROJECT

Group 2

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1.0 Mission design

Our task is to land our rocket, which is initially rotating around a Circular Lunar Orbit (500 km).

Given Requirements and Constraints

- Land to a site at 0 degrees latitude
- Payload that will be landed: 200 kg
- Landing velocity needs to be less than 1 m/s
- Must have 10% reserve fuel
- Maximum acceleration should be 3 g

Moon Conditions:

- There is no atmosphere so the there is no drag force
- $R_{moon} = 1737.1\ km$ which is the %27.25 of the Earth radius
- $g_{moon} = 1.62\ m/s^2$ which is the %16.5 of the Earth gravity
- $\mu_{\ moon} = 4900\ km^3/s^2$ which is the %1.23 of the Earth gravitational parameter
- Temperature range of the Moon is -178°C to +117°C

For the mission requirement DeltaV we have the following calculation:

$$\Delta V_{req} = \Delta V_{co} + \Delta V_{poten} + \Delta V_{grav} + \Delta V_{drag} - \Delta V_{rot}$$

Since there is no drag force in moon:

$$\Delta V_{req} = \Delta V_{co} + \Delta V_{poten} + \Delta V_{grav} + \cancel{\Delta V_{drag}} - \Delta V_{rot}$$

In order to analyze the mission requirements and conditions more accurately, the trajectory equations should be integrated.

$$\frac{dV}{dt} = \frac{T\cos(\alpha_{TVC})}{M} - \frac{D}{M} - \frac{\mu\ \sin(\gamma)}{r^2}$$

$$\frac{d\gamma}{dt} = \frac{V\cos(\gamma)}{r} + \frac{T\sin(\alpha_{TVC})}{V\,M} + \frac{L}{V\,M} - \frac{\mu\cos(\gamma)}{V\,r^2}$$

$$\frac{dy}{dt} = V\sin(\gamma) \qquad \frac{dx}{dt} = V\cos(\gamma)$$

$$\frac{dM}{dt} = -\frac{T}{I_{sp}g_o}$$

$R_{moon} = 1737.1\ km \quad = 4900\ \frac{km^3}{s^2}$

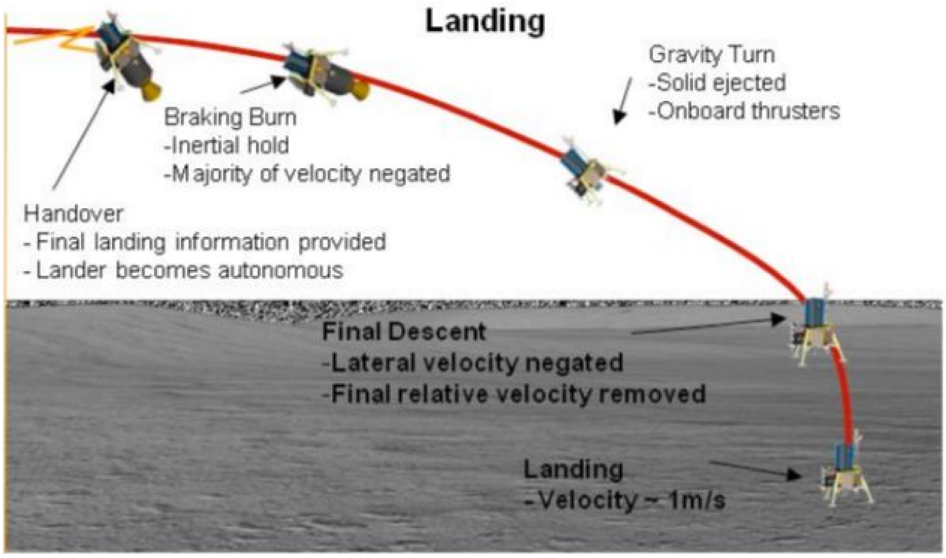


Figure 1. Landing Scenario

To calculate the required ΔV, we need to calculate the escape velocity.

$V_{es} = 2093\ m/s \quad V_f = 1\ m/s \quad \gamma_i = 0^\circ \quad \gamma_f = 90^\circ$

$\Delta V_{co} = 2092\ m/s \qquad \Delta V_{poten} = \sqrt{\frac{\mu}{R_o+500}} = 370\ m/s \qquad \Delta V_G = 750\ m/s$

$$\Delta V_{total} = 3213\ m/s$$

2.0 Propulsion system design and optimization

Hybrid-fueled rocket is a type of rocket that uses rocket fuels in different phases in the rocket engine. One of these fuels is in solid form and the other is in gas or liquid form. In figure 1, a simple hybrid rocket schematic is illustrated. In contrast to this approach, solid oxidizer and liquid fuel integration can be seen as a reverse hybrid rocket propulsion system. However, the first approach for hybrid rocket systems is generally popular. Furthermore, fuel burns along downstream where the oxidizer to fuel ratio (O/F) constantly decreases in the hybrid systems, whereas this incident is completely different in solid and liquid propulsion systems.

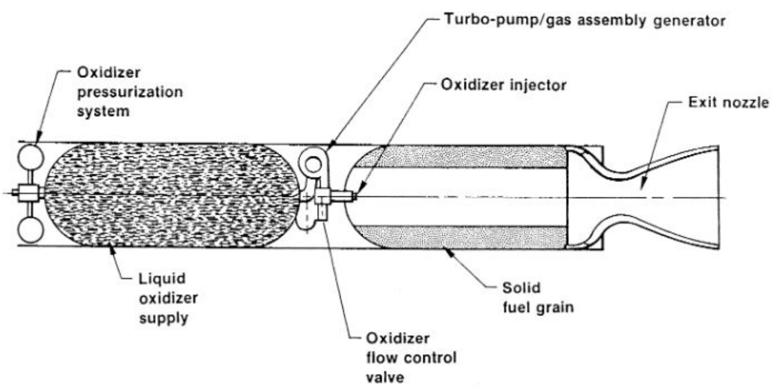


Figure 2. Hybrid Rocket Schematic

2.1 Why Hybrid Rocket?

Hybrid rockets have advantages over solid and liquid rocket systems. These positive properties are listed below. In addition to the listed parameters, there are some properties that make hybrid systems efficient and superior. Cost is the one effective parameter. Hybrid rockets have low operational costs. The other important feature is the robustness of grains. In contrast to solid rockets, fuel grain cracks are not a crucial effect.

	Advantages over	
Feature	**Liquids**	**Solids**
System	• Mechanically simpler • Less liquids – simpler injection, feed and control systems	• Chemically simpler (including fuel preparation process) • Restartable, throttle able
Safety	• Reduced fire hazard • Less prone to hard starts	• Reduced explosion hazard • Zero TNT equivalent • Able to stop
Performance	• Higher propellant density • Possible to improve performance by the addition of metals	• Higher performance
Environment	• Comparable with RP-1/LOX	• Does not need any toxic and harmful propellant

Table 1. Advantages of Hybrid Rockets

Hybrid rocket was chosen for this project in order to avoid the dangerous situations arising during mass transportation in solid fuel rockets and the mechanical complexity of liquid fuel rockets.

2.2 Fuel Selection

Fuels are dense repositories of energy that react with an oxidizer to release the energy for rocket propulsion systems. In order to maximize available energy in the fuels, substances need to have strong bonds, and elements should be light. It will also optimize the I_{sp} and c*. In hybrid systems generally, fuels are solid and oxidizers are liquid phase. For solid fuel, hydrocarbons are very abundant and efficient. Light hydrocarbon mixture, e.g. petrol is normally at liquid phase. However, it could be frozen and used as a solid fuel grain in a hybrid rocket motor. It makes it possible to reach higher than usual fuel regression rates. Considering hydrocarbons it is also possible to use paraffin as the hybrid fuel [1]. From the data that we got from NASA CEA, paraffin and LOX is a good combination for hybrid rocket propellant.

Oxidiser	Fuel	O/F	Specific Impulse m/s
NO_2	Paraffin	2	2624.9
LOX	RP-1	2	3288.6
LOX	Paraffin	2	3254.8 m/s

Table 2. Propellant Compasion

The reasons we chose paraffin as a solid fuel;

- Much cheaper compared to fuels used in this field
- No explosion hazard in a stationary state
- Much safer in protection and transportation
- High regression rate compared with HTPB.
- Paraffin Wax burns 5-5.5 times faster than the HDPE polymer.

2.3 Oxidizer Selection

Oxidizers are chemical compounds that are used to burn fuels in any rocket propulsion system. In the earth's atmosphere, most of the rocket propulsion systems use oxygen since it is abundant in the vicinity. In contrast, rocket systems must take oxidizer tanks with them in space [1]. In hybrid rocket systems, liquid oxidizers are used to burn selected solid propellant. Different types of liquid oxidizers could be selected in terms of their oxidizing capability. In the periodic table, potent oxidizers are positioned in the upper right part of it, however specific oxidizers show better performance among them.In hybrid rocket systems, liquid or gaseous oxygen, hydrogen peroxide, and nitrous oxide are commonly used. Different types of parameters should be considered for using oxidizer properly. One of the most essential ones is the convenient vaporization of oxidizers for performing the rocket effectively. Unless inconvenient vaporization takes place, an excessive number of regression rate differences may occur [2]. From all this given information, it is obvious that the selection of the oxidizer is a fairly important process.

In this project, we decided to utilize liquid oxygen (LOX) as an oxidizer because of its performance characteristics compared to other oxidizers. Moreover, LOX has been used as a cryogenic liquid oxidizer prevalently since its first appearance in the first liquid fueled rocket. The advantages of LOX are listed below.

- Its performance characteristics are high enough to meet the requirements of rocket propulsion because other inert elements are not added in LOX for diluting and LOX is convenient with different types of fuels.
- It also exhibits no toxicity characteristics.
- In terms of I_{sp} performance, it is a reasonable choice to use.
- It is abundant in the world.
- It has a low boiling point. (90K Cryogenic)
- It is affordable.

The NASA CEA results for the oxidizer and fuel we selected are given below.

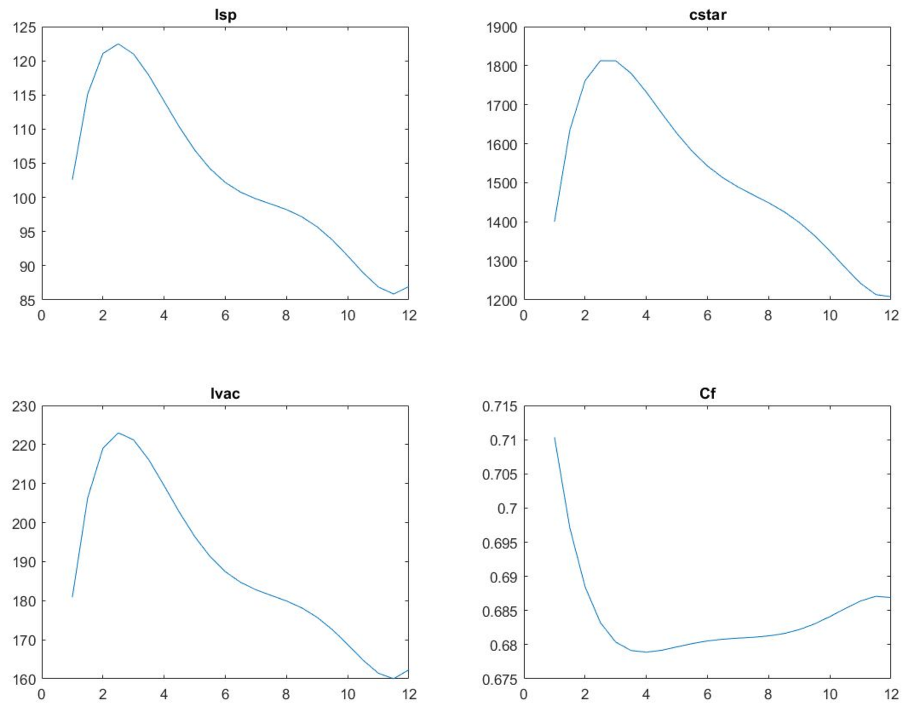


Figure 3. NASA CEA Results

As seen in the figure 3 above, at the O/F ratio 2 we got the maximum I_{sp} and I_{vac} but our calculation was for the Ae/At at 1 then we found the optimum area ratio as 70 then recalculating from NASA CEA we found that:

Ae/At	1.0000	70.000
CSTAR, *m/s*	1762.9	1762.9
CF	0.6678	1.8463
I_{vac}, *m/s*	2181.8	3370.3
I_{sp}, *m/s*	1177.2	3254.8

2.3 Toxicity and Explosiveness of Propellants:

Our propellant selection process includes safety as one of the first criteria. Safety refers to the toxicity and the explosiveness of our propellants. We are using Liquid Oxygen as oxidizer and Paraffin Wax as solid fuel. Paraffin Wax is almost perfectly safe because it is just regular candle wax. It is inert, which means it is not explosive. Paraffin Wax is easy to work with. Probably it won't burn with 5 seconds of direct exposure to lighter flame. It is also not toxic. When it burns, it may release some carbon monoxide which is toxic but there will be no one close enough to get affected by it during combustion. Also, we will land on the moon, where no living creatures exist. And a side note, automobiles and cigarettes constantly produce carbon monoxide everywhere.

Liquid Oxygen is also a non-toxic propellant. But the problem is it is cryogenic. Generally, we have liquid oxygen at -190°C. It is very hard to store liquid oxygen on earth but it is okay to have these temperatures in space. Even though oxygen itself is not explosive, it may explode its tank due to the high expansion rate during vaporizing. But when handled carefully, it is rarely an issue. Moreover, it delivers high performance, therefore it is one of the best oxidizers available on the market, with small issues on safety (still much better than most of them).

Hybrid rockets are much safer than solid and liquid rocket motors. Solid motors are very dangerous. They are in an active form and mixed. When they start burning they do not stop. This can easily cause explosions. Liquid motors are also dangerous. During a crash, propellants mix homogeneously and explode with massive power. Also, hard starts are an issue with liquid propellants, which can blow up our motor. On the other hand, hybrid rocket motors are less prone to explosion because solid fuel is in inert form and liquid oxidizer does not mix with the fuel homogeneously.

2.4 Nozzle area ratio

In order to find the Nozzle area ratio from the Nozzle Aretio/Cf diagram, firstly Cf value and the gamma (γ) should calculate.

$$I_{sp} = c^* \times CF/g_0$$

from the cea which explained in the upper we have I_{sp} =343.52 m/s, c*=1762.9 m/s then:

$$3331,784\ m/s = 1762.9\ m/s * CF/9.81 \quad CF = 1.85$$

which is parallel what we found from the directly CEA which are:

$$I_{vac} \times g_0 = 3370.3\ m/s \quad c^* = 1762.9\ m/s \text{ and the } \quad CF = 1.8463$$

Now the gamma (γ) should found:

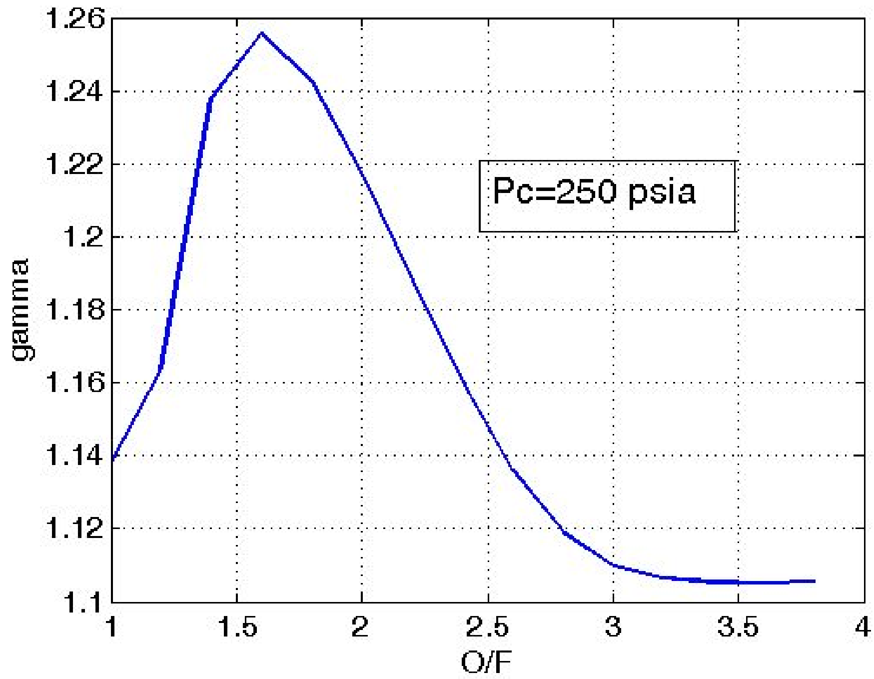


Figure 4. O/F Ratio vs Gamma

As explained in upper we initially select the O/F ratio as the 2 corresponding gamma value for our O/F 2 ratio is 1.22 for the given figure. However, the figure for

250Psia which is the around 18bar, but our chamber pressure is 40 bar and corresponding famma value slightly bigger than the 1.22 so, for our 40bar chamber pressure our gama should near to 1.24

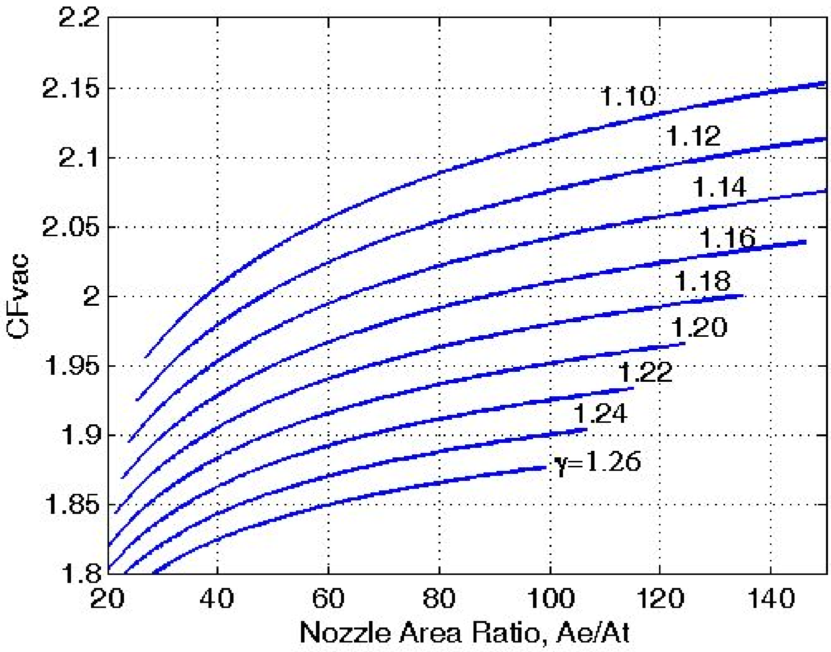


Figure 5. Nozzle Area Ratio vs CF

From the upper figure using the corresponding gamma value=1.24 and CFvac=1.86 we found that our optimal area ratio is around 70.

2.5. Chamber pressure

On the CEA at O/F ratio 2 we get the results for I_{vac} at the pressure values between 5 and 70 bars, with an increment of 5 bar. There were not too many differences on the Ivac due to the pressure difference. We know also the pressure has no significant effect on the regression rate, so mainly thinking of the safety of the combustion chamber, we research the typical combustion chamber pressure for the hybrid rocket motor is around the 35-45 bar, so we choose the 40 bar chamber pressure to stay on the conventional designs. For deeper calculation for the combustion chamber we couldn't find any literature.

2.6. Oxidizer to fuel ratio

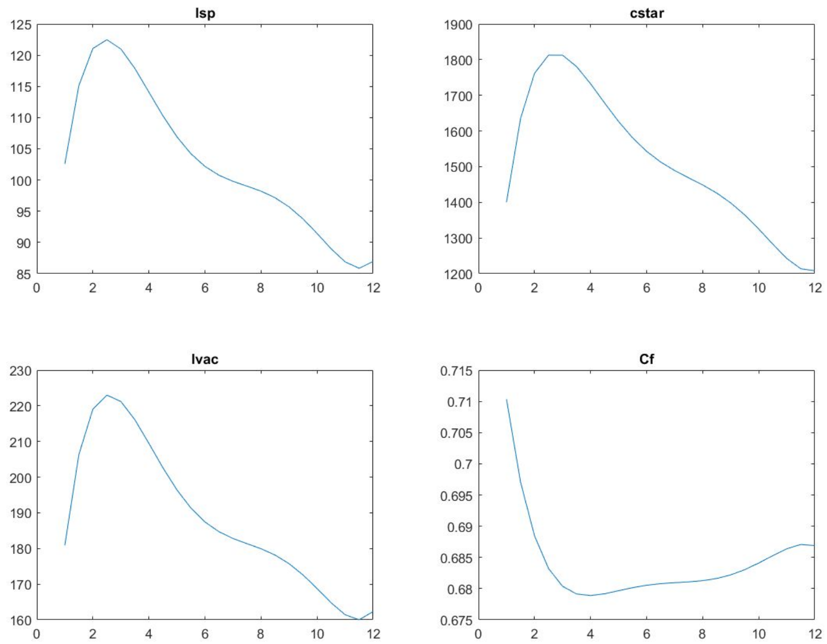


Figure 6. NASA CEA Results

By analyzing different O/F ratios at NASA CEA, we found that the most optimal result was 2. for the Lox/ParaffinWax case at the combustion pressure 40 bar and the subsonic area ratio both for 1 and 70.

2.7. The geometrical properties of the propulsion system

$$c^* = \frac{P_c * A_t}{mdot} \quad 1692.384 = \frac{40\times10^5\times A_t}{12} \qquad A_t = 0.005\ m^2 \quad D_t = 0.08\ m$$

$$\frac{A_e}{A_t} = 70 \qquad A_e = 0.35\ m^2 \quad D_e = \ 0.667\ m$$

Effective divergence angle is chosen 20°. $\tan(20^\circ) = \frac{D_e - D_t}{L} = \frac{0.667-0.08}{L}$ L=0.8 m

We assume that regression rate= $11.7 * 10^{-5} \times G_{ox}^{0.62}$

$$a = 11.7 \times 10^{-5} \qquad n = 0.62$$

$$D_f = [(\frac{(2n+1)\times 2^{\prime n+1}\times a}{\pi^n}) \times (\frac{M_{ox}^2 \times t_b^{1-n}}{1-(\frac{D_i}{D_{pf}})})]^{1/(2n+1)}$$

$$D_f = 0.5\ m$$

$$\text{Since we assume } \frac{D_{pf}}{D_{pi}} = 2, \quad D_{pi} = 0.25\ m$$

$$L_{fuel\ grain} = \frac{4\times M_{ox}}{\pi\times\rho_f\times(D_{pf}^2 - D_{pi}^2)}$$

$$L_{fuel\ grain} = 2.29\ m$$

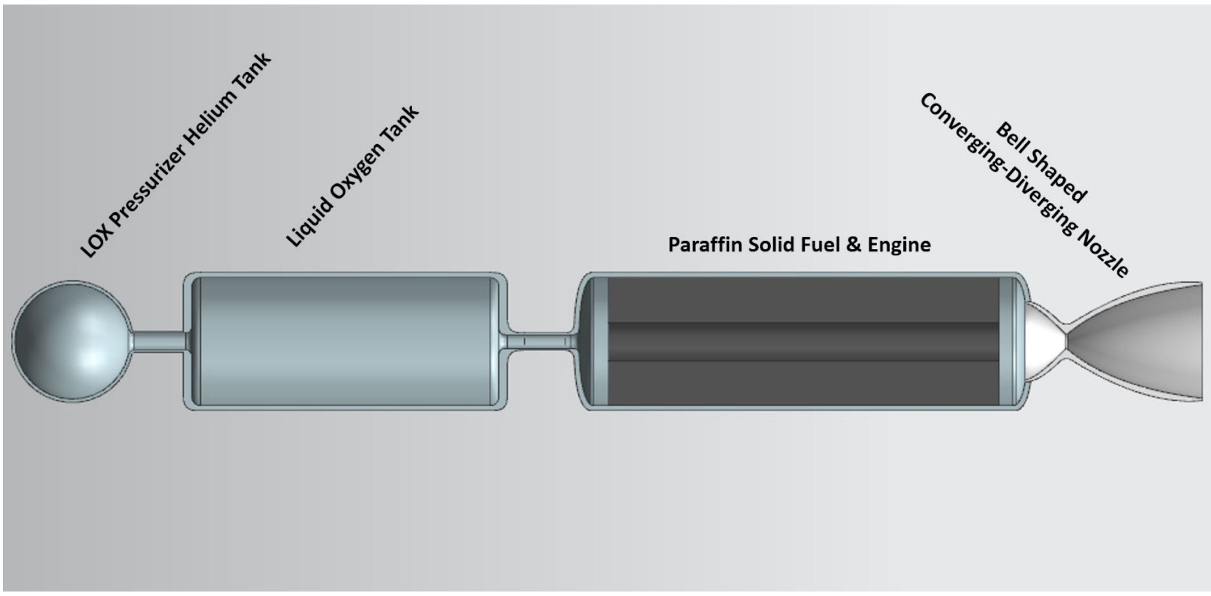
We determined the outer layer as 3 cm in order to withstand high temperatures and 40 bar.

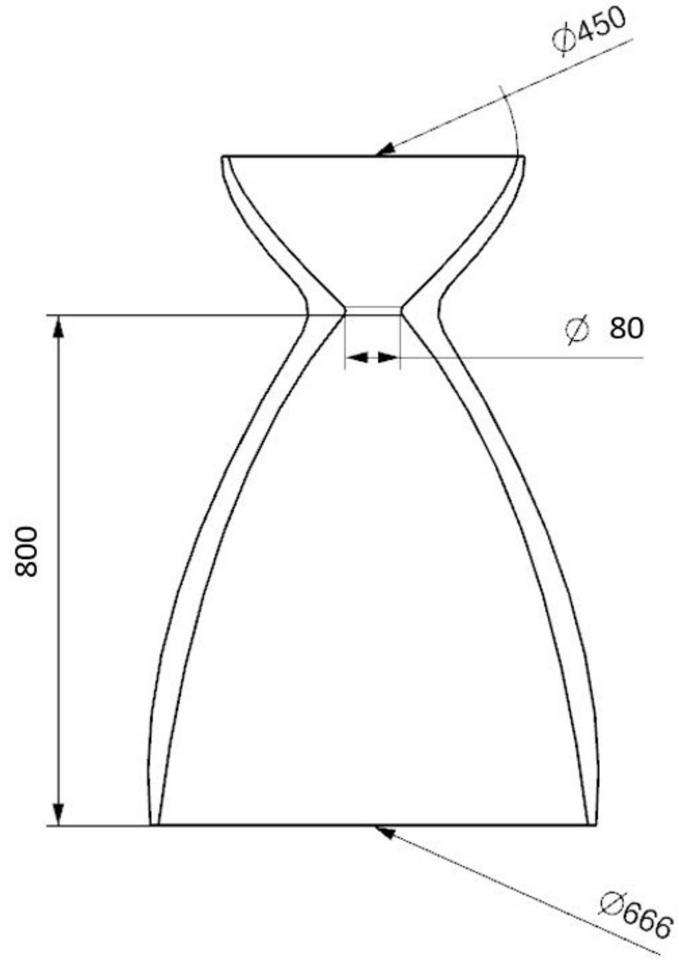
$t_{wall} = 0.03$

$D_{chamber} = D_{grain} + 2 \times t_{wall}$

$D_{chamber} = 0.56\ m$

3.0 Structural design





4.0 Weight estimation

From mission design, we computed $\Delta V_{req} = 3213\ m/s$

$$\Delta V = I_{sp} \times g_0 \times ln(\frac{M_i}{M_b}) = 342.52 \times 9.81 \times ln(\frac{M_i}{M_b})$$

$$ln(\frac{M_i}{M_b}) = 0.95 \quad \frac{M_i}{M_b} = 2.6 \qquad \frac{M_d + M_p + M_s}{M_d + M_s + 0.1 * M_p} = 2.6 \quad M_s = 235.3\ kg \quad M_p = 941.2\ kg$$

$$M_{total} = 1376.47$$

From the restriction, a=3g ,

$$a = \frac{I_{vac} \times mdot}{M_{total}} \qquad mdot = 12\ kg/s$$

We choose our O/F=2

$$M_{fuel} = 31.37\ kg\ M_{\ ox} = 62.746\ kg$$

$$mdot_{ox} = 8\ kg/s\ \ mdot_{\ fuel} = \ 4\ kg/s$$

$$\rho_{Lox} = 1141\ kg/m^3\ \ \rho_{paraffin\ Wax} = 930\ kg/m^3$$

For the system structure, we used the composite and for the nozzle, we used Graphene. Actually, at the nozzle, we could have used Granite since it is more durable against the high temperature however shaping it is difficult, and considering the price, we chose Graphane.

$$\rho_{composite\ fiberglass\ density} = 1800\ kg/m^3\ \ \rho_{graphene} = 2260\ kg/m^3$$

As you can see, the overall system dry mass depends on the mass of the mold of the rocket which we are using the composite fiberglass since we cannot consider the mass of the communication systems ignitor and the injector system additions we increase the height of the rocket from the necessary volume which will increase the total mass with a sensible amount of it. We prepared two system mass calculation one of them for the at nozzle area ratio1 which I_{vac}=222.42m/s and the other one for the nozzle area ratio 70 which's I_{vac}=343.52m/s

- **Consider the survivability of the propulsion system and its support hardware during extended operation (months) in space:**

Of course for the months of operation, we should add a battery system to continue communication with the rocket. For these systems best-using electricity resources of course will be the sun rather than using our fuels the get somehow electricity power for these purposes we should add the solar panels the system and the lithium-ion batteries. The reason for choosing a Lithium-ion battery in addition to its efficiency is it has less risk of blowing up compared to the lithium battery and from a solar panel we will not get a perfectly regular current and it can affect the lifespan of the lithium polymer. However, depending on the time spent on the space lithium polymer batteries can also be chosen because lithium polymer batteries can easily producible with the desired system. The other risk of the system in space rather than the feeding up of the communication system getting protected from the asteroids. To easily escape from the asteroid rains we can add the system an ion propulsion system. All these additional requirements needed a recalculation of the rocket equations. It even might be necessary to change the number of stages. For example, if we were to stay more in space, we might use 3 stages. The first stage to leave the earth's orbit, the second stage for navigating in space using an ion propulsion system, and the third stage can be used for the final purpose: to land a planet or a moon. So, unfortunately, we are not at a technological level to be able to produce a general purpose rocket system. We can only develop a rocket system for certain restricted conditions and purposes.

5.0 Concept of operations & Executive summary

At the beginning of the project, we computed the required ΔV. Then we determined our rocket type as a hybrid rocket. Because the performance of hybrid rockets is much higher than solid propellant rockets. In addition, hybrid rockets are much less complex than liquid propellant rockets. After selecting the hybrid rocket, we search for articles about Hybrid Rocket Propulsion. According to data about the fuels and oxidizers from articles, we used the NASA CEA program to choose our fuel and oxidizer. Here we tried different combinations of fuels and oxidizers. We have seen that the highest efficiency is in LOX and Paraffin Wax. Afterward, we made our analysis according to different O/F ratios. Especially by looking at the c* values, we achieved the highest performance when O/F = 2. Likewise, we determined the nozzle area ratios and chamber pressure through NASA CEA.

We specified that $A_t/A_e = 70$ O/F=2 $P_{chamber} = 40\ bar$

After the calculations in NASA CEA, we determined orbital velocity at the given place, and calculated delta V for desired final landing velocity. Also, we compute the required potential velocity. However, we assumed that the required velocity for gravitational loss is more than the potential velocity. Then, we calculated the total delta V for this mission by summing these velocities.

From the rocket equation, we computed the mass ratio by considering the stage structural mass fraction is 0.2. Moreover, total propellant mass and its flow rate was found from the related mass ratio. By considering the desired reserve propellant mass, burn time was calculated.

In order to find the geometries of nozzle, we computed throat area by considering chamber pressure, c* and mass flow rate. After finding the throat area, it was simple to find the exit area by using area ratio. For calculating the length of the nozzle, effective divergence angle was assumed as 20 degree. In addition, it is important to determine the dimensions of the fuel grain size for the hybrid rocket. Regression rate is one parameter that affects the dimensions of it. For the given system including paraffin-Lox, regression rate constants are determined and put into the fuel grain dimension equation. After the computations of fuel grain sizes, we assumed the material thickness for chamber pressure is 3 cm in terms of convenience and real life applications. Furthermore, the length of the chamber pressure was considered to be longer than the length of the fuel grain due to the pre-combustion and post-combustion.

References

1. Surmacz, Paweł & Rarata, Grzegorz. (2009). Hybrid Rocket Propulsion Development and Application. Progress in Astronautics.
2. Wikipedia contributors. (2020, December 26). *Hybrid-propellant rocket*. Wikipedia.
3. https://en.wikipedia.org/wiki/Hybrid-propellant_rocket#:%7E:text=Oxidizer-,Common%20oxidizer%20choices,the%20rocket%20to%20perform%20efficiently.
4. Arif Karabeyoğlu Lectures