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Sustainable Design and Manufacturing

Master in Aerospace Engineering

Sustainable Approach in LRMs: Fuel and Manufacturing

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Abstract

The purpose of this project is to compare five different configurations of propellants and tanks' materials regarding performance, environmental impacts and costs associated. The five different configurations subjected to the analysis are: Rocket Propellant 1 (RP-01) with Liquid Oxygen (LO_2) as oxidizer, Liquid Hydrogen (LH_2) with the same oxidizer, and lastly Unsymmetrical Dimethylhydrazine (UDMH) with Dinitrogen Tetroxide (NTO). All propellants were combined with Aluminium tanks and with Carbon Fiber tanks, except for UDMH which was solely paired with an Aluminium tank. The project was divided in three interdependent analyses focused on performance, environmental impacts and economic analysis. The findings of this project confirmed the initial hypothesis that, due to the size of the fuel tanks, Carbon Fiber is not a viable choice neither when assessing its environmental effects, nor in terms of costs associated. Regarding the propellant choice, when total costs were evaluated, which accounted for not only fuel and tank materials but also labour and manufacturing costs for instance, RP-01 significantly outperformed the other two propellants. While the difference in environmental impacts was not as predominant, the analysis through a Life Cycle Assessment revealed that LH_2 is overall the better choice for global warming influence and other environmental consequences.

Contents

1 Introduction

The present project focuses on Liquid Rocket Motors and how sustainable it is to use a certain propellant coupled with a specific tank's material. The Proton-M Russian launcher was taken as the base comparison and as the base design, mostly due to its use of Hydrazine as a propellant; indeed such fuel is extremely toxic both for the environment and for human health, therefore it is important to evaluate how other fuel options would perform, in order to see if it is possible to slowly substitute Hydrazine in the future iterations of rocket launchers. The objective of the project is then established as follows:

To reach the same height of the Proton-M launcher, between 170 and 230 km, maintaining the same external rocket structure (case), using 5 different combinations of fuel and tank's material.

The fuels assessed with their respective oxidizers are:

- UDMH: a derivative of Hydrazine, coupled with Nitrogen Tetroxide (NTO). Despite its high efficiency and storability, UDMH is very toxic and pollutant.
- RP-1: a specific type of refined kerosene for rocket motors, coupled with Liquid Oxygen (LOX). This combination is very common among rocket propellants due to its high density, low production cost and the fact that can be stored at room temperature.
- LH2: Liquid Hydrogen, coupled with Liquid Oxygen (LOX). This propellant is one of the most well-known alternatives to conventional fuels, not only due to its high energy density, but because the only product from its combustion is water vapor.

The RP-1 was chosen due to its wide usage, since it boasts numerous advantages, such as: good efficiency, reliability, cheap, easy to produce and handle, easily accessible on the market and many others. LH2 is instead a good option due its high efficiency as a fuel and has overall a smaller environmental impact thanks to its nature. Unfortunately, it is not as easy to handle as RP-1 and is also more expensive, for reasons discussed later, which is why it is still a relatively less used fuel compared to the latter.
The materials for the tanks that will be assessed are:

- Aluminum Alloy AL2219.
- Carbon Fiber Filament.

Every propellant is then paired with each tank's material, except for UDMH which is considered only with Aluminum since it's the Proton-M original combination. In order to verify if the objective is fulfilled, the fuel and tank masses must be calculated. This can be achieved through a trial and error process, by simulating the launcher's behaviour through the combination of 3 softwares: `Matlab`, `Rasaero II` and `Cpropep`. The values obtained are then utilized in the next steps: the **Life Cycle Assessment** with the aid of `SimaPro`, done for each combination, considering a Well-to-Wheel approach for the fuels and Cradle-to-Grave for the tanks and, finally, the **Cost Analysis**. In the end, the results are compared to determine which is the most valuable solution environmentally and economically.

2 Launcher Design

The characteristics of the launcher are as aforementioned based on the Proton-M model. This translates, as stated in the objective, into the same overall geometry, thus same dimensions regarding the case length and diameter. This is going to be the case for every combination, which cannot be initially stated without any calculation. Indeed, such verification is made by checking the amount of propellant needed to reach the correct height and therefore the tanks size and mass necessary to hold it, making sure the tanks themselves do not exceed the case size. This is later shown in the Masses Calculation section.

Obviously, there are also assumptions to be made: first of all, the rocket is treated as a single stage launcher instead of 3 stages. This is required in order to run the simulation through `Matlab` with a preexisting code, otherwise it would need to be heavily changed and consequentially it may not work as intended. Moreover, the rocket's components are just the fairing, case, fins, nozzle, tanks, hydraulic system and the combustion chamber. They were toned down to simplify the problem but the main ones were effectively kept to correctly simulate the rocket ascent.

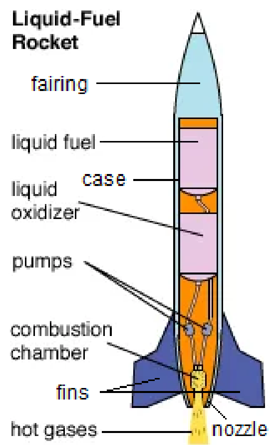

Figure 1: Cross section of a typical single stage LRM.

2.1 Structure and Aerodynamics

The Proton-M has a length of $L = 53m$ and a diameter $D = 7.4m$. The length is comprised not only of the case but also of the fairing and the nozzle, whereas the diameter solely corresponds to that of the case. Given the general size, the other main parts are designed as following:

- Fairing: It has the same diameter of the case, whereas the length is $7.61m$. The type of fairing used is a Von Karman ogive.
- Case: As mentioned previously, it has a diameter of $D = 7.4m$, the length is instead $45.38m$. The thickness is not relevant for the simulation.
- Fins: There are 3 fins aft of the case, they are biconvex and $2.54m$ high, root is $3.81m$ long and the tip is $1.52m$.
- Nozzle: nozzle *Throat Area* and *Exit Area* are dependant on the fuel type, specifically on the parameter $C*$ which is the characteristic velocity and it expresses the efficiency of the propellant, *Mass Flow Rate*

$\dot{m}$ and *Combustion Chamber Pressure* p_0 according to this formula:

$$Ag = \frac{C^*\dot{m}}{p_0}$$

which means they change depending on the propellant involved.

- Tanks: there should be two, one for the fuel and one for the oxidizer, but for simulation purposes, it's considered to be only one containing both, with a diameter $d = 7m$, smaller than that of the case, in order to account for its thickness, the presence of fluidics and other components, which even though they are not explicitly considered, they are still involved in the overall *Inert Mass*, a value that represents everything that is not *Propellant Mass* or *Payload Mass*.
- Hydraulic system: Although its presence is necessary regarding pressurization and liquid distribution, the dimensions of the pipes and pumps are not necessary for simulation purposes.
- Combustion Chamber: The diameter is the same of the tanks, $d = 7m$ whereas the length depends on the fuel. For the purpose of this project choosing the correct length mostly depends on how much space is left inside the rocket after knowing the tank size rather on performance. This obviously is not true in a real scenario, but due to the complexity of the problem it was an assumption made to easen the computational work.

The simplified model of the rocket is as follows:

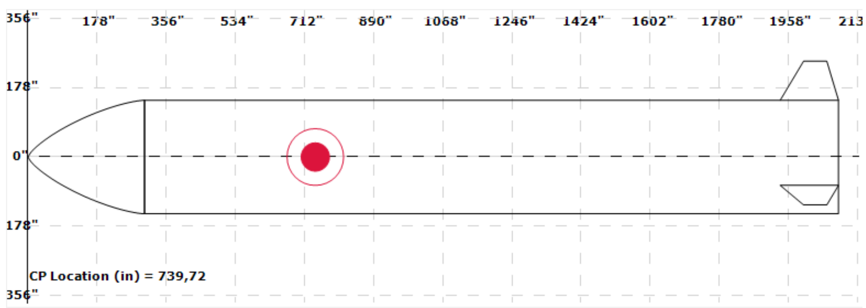

Figure 2: Rocket drawing on Rasaero II, measurement in inches.

With CP being the Centre of Pressure of the launcher, useful for aerodynamic purposes. Indeed, with Rasaero II the Drag Coefficient or CD can be easily computed:

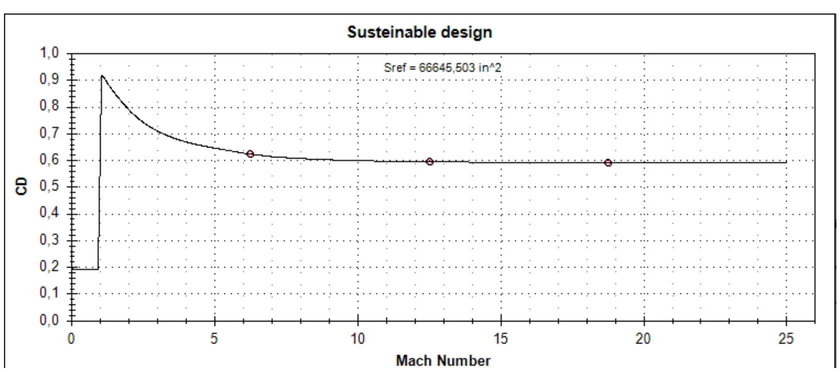

Figure 3: CD - Mach plot of the launcher.

This is especially important due to its role in the simulation. In fact, it is required to know how the drag changes with the velocity during the ascent, so that the correct altitude can be calculated. If the dimensions are the same for every combination, like mentioned, than the drag is always the same.

2.2 Combustion

The combustion was simulated through the help of Cpropep. Basically, the approach consisted of choosing two typical values of *Combustion Chamber Pressure* p_0 and O/F, which is the oxidizer over fuel mass ratio, and then tweak those values to find a valid solution. In order to check if this approach is correct, the propellant of the Proton-M was first tested, so that it was possible to compare the results with the values present online for the launcher. The parameter considered is the *Specific Impulse* Isp, which is a measurement of the efficiency of a rocket motor. It is defined as follows:

$$Isp = \frac{F}{\dot{m}g} = \frac{v_e\dot{m}}{\dot{m}g} = \frac{v_e}{g} = \frac{C^* C_f}{g}$$

Where F is the thrust of the rocket, g is the gravity acceleration, v_e is the effective exhaust velocity, which expresses the velocity of the gasses coming out of the nozzle, and C_f is the thrust coefficient, that is the efficiency of the nozzle itself.

For the Proton-M, after choosing $O/F = 2.87$ and $p_0 = 25bar$ these are the results:

```
Propellant composition
Code  Name                               mol       Mass (g)   Composition
963   UNSYM-DIMETHYLHYDRAZINE (UDMH)     1663.9378 100000.0000   2C  8H  2N
639   NITROGEN TETROXIDE (LIQ.)          3119.1896 287000.0000   2N  4O
Density :  1.179 g/cm^3
4 different elements
C  H  N  O
Total mass:  387000.000000 g
Enthalpy  :  214.21 kJ/kg

148 possible gazeous species
3 possible condensed species

                         CHAMBER     THROAT        EXIT
Pressure (atm)    :       24.673     13.817       0.010
Temperature (K)   :     3342.706   3009.494     670.390
H (kJ/kg)         :      214.209   -423.770   -4477.074
U (kJ/kg)         :     -944.831  -1467.272   -4709.523
G (kJ/kg)         :   -37230.591 -34135.944  -11986.739
S (kJ/(kg)(K)     :       11.202     11.202      11.202
M (g/mol)         :       23.979     23.979      23.979
(dLnV/dLnP)t      :     -1.00000   -1.00000    -1.00000
(dLnV/dLnT)p      :      1.00000    1.00000     1.00000
Cp (kJ/(kg)(K))   :      1.92531    1.90322     1.42779
Cv (kJ/(kg)(K))   :      1.57858    1.55649     1.08105
Cp/Cv             :      1.21965    1.22277     1.32074
Gamma             :      1.21965    1.22277     1.32074
Vson (m/s)        :   1145.19096 1129.58494   532.45325

Ae/At             :                 1.00000   115.00947
A/dotm (m/s/atm)  :                66.85738  7689.23191
C* (m/s)          :              1649.57760  1649.57760
Cf                :                 0.68477     1.85690
Ivac (m/s)        :              2053.37753  3138.98425
Isp (m/s)         :              1129.58494  3063.09744
Isp/g (s)         :               115.18561   312.34901
```

Figure 4: Propellant data for the UDMH + NTO

The values inside the red rectangles are the ones useful for the simulation, alongside the aforementioned O/F. The values chosen let us have the $Isp = 312.35$, which is extremely close to the average Isp_M of the Proton-M, in fact:

$$Isp_M = \frac{285 + 327 + 325}{3} = 312.33$$

With the three values at the numerator being the values of each stage of the rocket. Furthermore, the C^* was compared to the values found online for a similar value of O/F:

	Online	Cpropep
O/F	2.61	2.87
C^*	1720 m/s	1650 m/s

Since $O/F = 2.61$ is the optimum value for the UDMH + NTO propellant, having a different one leads to a smaller C^*.

For the other fuels, the same approach was undertaken, with the only difference being that p_0 is considered fixed and equal to $p_0 = 25bar$ like the UDMH, thus simplifying the process now that there is only one degree of freedom. The values for LH2 and RP-1 are:

	RP-1	LH2
O/F	2.52	4
C^*	1671.5 m/s	2387 m/s

```
Propellant composition
Code   Name                                        mol      Mass (g)  Composition
797    RP-1                                        71.2932 1000.0000    2H  1C
686    OXYGEN (LIQUID)                             78.7530 2520.0000    2O
Density :  1.017 g/cm^3
3 different elements
H  C  O
Total mass:  3520.000000 g
Enthalpy  : -1884.58 kJ/kg

114 possible gazeous species
3 possible condensed species

                          CHAMBER      THROAT        EXIT
Pressure (atm)      :      24.673      13.870       0.010
Temperature (K)     :    3341.298    3023.105     720.620
H (kJ/kg)           :   -1884.583   -2532.753   -6788.078
U (kJ/kg)           :   -3067.017   -3602.584   -7043.095
G (kJ/kg)           :  -40127.511  -37133.805  -15035.954
S (kJ/(kg)(K)       :      11.446      11.446      11.446
M (g/mol)           :      23.495      23.495      23.495
(dLnV/dLnP)t        :    -1.00000    -1.00000    -1.00000
(dLnV/dLnT)p        :     1.00000     1.00000     1.00000
Cp (kJ/(kg)(K))     :     2.04811     2.02528     1.53100
Cv (kJ/(kg)(K))     :     1.69422     1.67139     1.17712
Cp/Cv               :     1.20888     1.21173     1.30064
Gamma               :     1.20888     1.21173     1.30064
Vson (m/s)          :  1158.52603  1138.57185   555.23268

Ae/At               :                 1.00000   121.79554
A/dotm (m/s/atm)    :                67.74623  8251.18907
C* (m/s)            :              1671.50839  1671.50839
Cf                  :                 0.68116     1.87352
Ivac (m/s)          :              2078.19670  3213.04431
Isp (m/s)           :              1138.57185  3131.61141
Isp/g (s)           :               116.10202   319.33549
```

```
Propellant composition
Code   Name                                        mol      Mass (g)  Composition
457    HYDROGEN (CRYOGENIC)                        49606.1247 100000.0000    2H
686    OXYGEN (LIQUID)                             12500.4687 400000.0000    2O
Density :  0.287 g/cm^3
2 different elements
H  O
Total mass:  500000.000000 g
Enthalpy  : -1219.20 kJ/kg

9 possible gazeous species
2 possible condensed species

                          CHAMBER      THROAT        EXIT
Pressure (atm)      :      24.673      13.808       0.010
Temperature (K)     :    2910.362    2619.455     529.691
H (kJ/kg)           :   -1219.197   -2556.857  -10840.102
U (kJ/kg)           :   -3646.777   -4741.787  -11281.926
G (kJ/kg)           :  -66159.878  -61006.362  -22659.420
S (kJ/(kg)(K)       :      22.314      22.314      22.314
M (g/mol)           :       9.968       9.968       9.968
(dLnV/dLnP)t        :    -1.00000    -1.00000    -1.00000
(dLnV/dLnT)p        :     1.00000     1.00000     1.00000
Cp (kJ/(kg)(K))     :     4.64344     4.55023     3.23234
Cv (kJ/(kg)(K))     :     3.80932     3.71611     2.39822
Cp/Cv               :     1.21897     1.22446     1.34781
Gamma               :     1.21897     1.22446     1.34781
Vson (m/s)          :  1691.85109  1635.65235   733.87248

Ae/At               :                 1.00000   105.49489
A/dotm (m/s/atm)    :                96.74120 10205.70256
C* (m/s)            :              2386.90356  2386.90356
Cf                  :                 0.68526     1.83776
Ivac (m/s)          :              2971.46808  4487.27124
Isp (m/s)           :              1635.65235  4386.54879
Isp/g (s)           :               166.79012   447.30349
```

Figure 5: Propellant data for the RP1 + LOX on the left and LH2 + LOX on the right.

2.3 Altitude Goal

The Proton-M typical mission's altitude varies between $170km$ and $230km$. Knowing the parameters of the rocket regarding Structure and Aerodynamics, Combustion and the various masses (*Inert, Propellant, Payload*), that are going to be discussed in detail in the next section Masses Calculation, it is possible to estimate through `Matlab` the actual height reached, which is equal to $H = 223.36km$, inside the range given. This is therefore going to be the target also for the other combinations, but extreme accuracy is not required ($\pm 5km$).

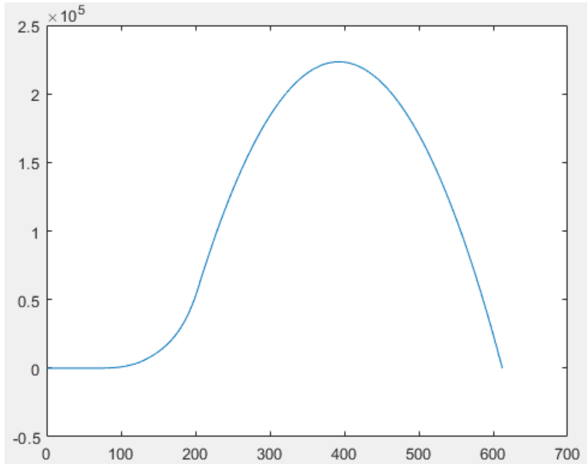

Figure 6: Altitude (m) - Time (s) graph of the Proton-M

To see the efficiency of the propellant, the mass of the fuel can be changed gradually so that the altitude reached varies. The result is as follows:

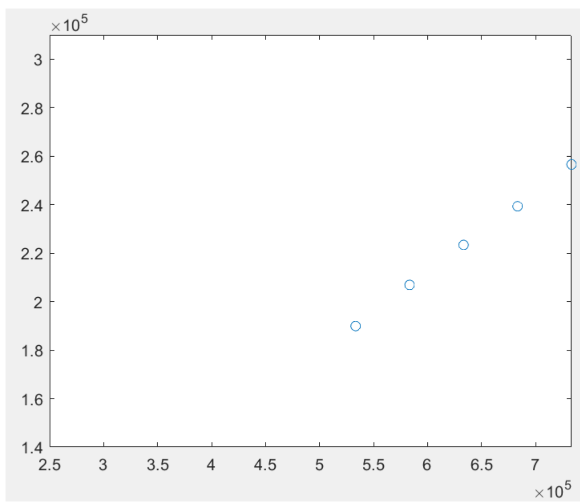

Figure 7: Altitude (m) - Propellant Mass (kg) graph of the Proton-M

The third value is the one corresponding to the correct amount of fuel and oxidizer shown in Figure 6. This is later going to be compared with the other fuels to determine visually the difference in efficiency between the three choices.

2.4 Masses Calculation

First of all, the rocket is comprised of three main masses: *Payload Mass*, the *Inert Mass* and the *Propellant Mass*. The *Payload Mass* is fixed and equal to $m_{pl} = 23000kg$, a typical value of a Proton-M mission. The *Propellant Mass* m_{prop} is instead variable, because it is dependant on the fuel type, since different efficiencies determine different heights reached. The *Inert Mass* m_{in} changes due to the *Tank Mass*, that is also dependant on the fuel type, but in this specific project, due to the relative change in mass with different materials being small compared to the overall weight of the rocket, it does not make a big difference in the performance, therefore it can be considered fixed and equal to $m_{in} = 48626.6kg$, that is the original value of the Proton-M. In smaller sized rockets the change in *Inert Mass* is extremely important due to its "snowball effect", which can be better understood from the Tsiolkovsky equation:

$$\Delta V = Isp\, g_0\, ln(\frac{m_0}{m_f})$$

Where ΔV is the total change of velocity of the rocket, g_0 is the standard gravity, m_0 is the wet mass, which is the total initial mass of the rocket, and m_f is the dry mass, the final mass of the rocket without propellant.

As it is possible to see, if the m_f is increased thanks to the *Inert Mass*, to get the same ΔV it is needed more propellant with the same Isp and *Payload Mass*. But more propellant means more *Inert Mass* since the tanks must be increased to allow for more space and also other components might need to be upsized, for example, the pipes and pumps of the hydraulic system.

Starting with the *Propellant Masses*, the value for the UDMH fuel and NTO oxidizer are directly given by combining the O/F in the Combustion section and the total propellant mass of the Proton-M found online. For the other two, the masses are instead calculated through a trial and error process by simulating multiple times, again with the correct O/F previously found, until the launcher reaches the required height, $223.36km \pm 5km$, with the same external structure, as stated by the objective. The results are the following:

	UDMH + NTO	RP-1 + LOX	LH2 + LOX
Fuel	163661 kg	179941 kg	84000 kg
Oxidizer	469712 kg	453442 kg	336000 kg

It is possible to now compare the efficiencies of every propellant graphically, by showing how the *Height Reached* changes with the *Propellant Mass*, like already done in Figure 7 for the UDMH + NTO:

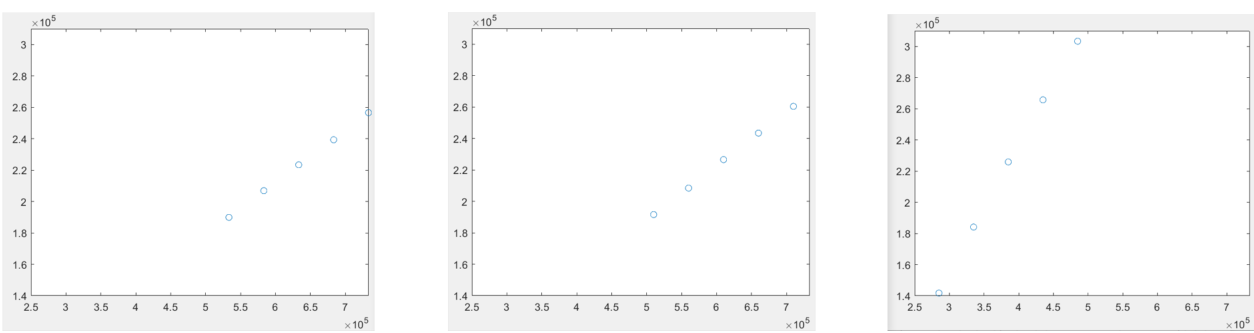

Figure 8: Comparison between the 3 propellants, higher inclination means higher efficiency. From left to right: UDMH + NTO / RP-1 + LOX / LH2 + LOX

In a real case scenario the trend would not be linear of course, since the *Inert Mass* would not be fixed. Indeed here the tank was oversized in other to be able to hold the supplementary propellant, going as far as to make it as long and wide as the case itself, but the mass was kept the same. This is obviously oversimplified because huge differences in *Propellant Mass* require as much of a change in *Inert Mass*, but the purpose of this graphs is to just show visually how the efficiency of the propellant can impact very highly the height reached.

The next step is calculating the tank masses and sizes. As stated at the beginning of this section, the *Tank Mass* does not affect the *Inert Mass*, but the value is still important for the **Cost Analysis** and the **LCA**.

On the other hand, the size is extremely significant since if it is too large it can exceed the case size and the rocket must be upsized. Now that the *Propellant Masses* are available, it is possible to check for each one of them the tank dimensions and mass, through these calculations:

```
%characteristic of the propellant (LH2+LOX+Aluminum)
rho_F = 70.8; %density of the fuel
rho_O = 1429; %density of the oxidizer
OF=4; %optimal O/F
m_F = m_prop/(OF+1); %mass of the fuel
m_O = m_prop-m_F; %mass of the oxidizer
V_F = m_F/rho_F; %volume of the fuel
V_O = m_O/rho_O; %volume of the oxidizer
V_prop = V_F+V_O; %volume of the whole propellant
m_prop= 420000; %mass of the propellant
rho_prop = m_prop/V_prop; %density of the whole propellant

%tank dimensions, at the beginning assumed as long as the case
d=7; %diameter of the tank
L=45.38; %length of the case
Ftu=0.413*10^9;  %ultimate tensile strength of aluminum
rho_mat=2840; %density of aluminum
p0=25*10^5; %combustion chamber pressure
fs = 1.25; %safety factor for p0
pb = fs*pcc; %burst pressure
t_tank = pb*d*0.5/Ftu; %thickness of the case
Vcs = pi*L*(((d*0.5+t_tank)^2)-(d*0.5)^2); %volume of the tank, assumed cylindric with an emisphere at each end

%true tank dimensions
ni_imp=V_prop/Vcs; %packing factor, to see how much space the propellant actually occupies
L_new=L*ni_imp; %true length of the tank
V_tank = pi*L_new*(((d*0.5+t_tank)^2)-(d*0.5)^2); %true volume of the tank
m_tank = rho_mat*V_tank; %mass of the tank
```

Figure 9: Calculation of the aluminum tank mass for the LH2 fuel

This procedure is applied for every combination and as previously mentioned, the dimensions fit every time inside the case since the "packing factor" is always $ni_{imp} < 1$. The results for the masses are:

	UDMH + NTO	RP-1 + LOX	LH2 + LOX
Aluminum	30450 kg	30627 kg	68799 kg
Carbon Fiber	-	12068 kg	27108 kg

Now that the *Propellant Masses* and the *Tank Masses* are known, it is possible to proceed with the **LCA** and **Cost Analysis**.

3 Life Cycle Assessment

3.1 Goal and Scope

Although at the present date rocket launches do not have a significant impact on the global environment, the quick expansion of the sector in the last few years has given rise to an increase concern for the effect on ozone layer depletion and global warming. As an attempt to quantify the influence on the environment of each of the five configurations studied, a Life Cycle Assessment was performed for each configuration.

The goal of this LCA is to assess and compare the environmental impacts of both fuel change and fuel tank material change on all five configurations already stated, studied across different stages of manufacturing and use.

The life cycle of the fuels and respective oxidizers will be considered from Well to Wheel so from production of the compound to the respective combustion. The direct emissions from the launch are assumed to be released at ground level, as done by Schabedoth on the first LCA of the entire life cycle of rocket propellants, to facilitate the analysis without compromising the accuracy of the study. For the tanks, a similar approach to the fuels will be taken, where the impacts from raw materials to direct use of the tanks will be studied (Cradle to Grave approach). This study will only focus on the national portion of transportation (in this case United States of America) thus shipping by air and sea are not considered.

The functional unit chosen for this LCA as a mean to compare directly the results is the mission previously established.

The LCA calculations were performed using the software SimaPro whose extended database allowed the complete simulation of all configurations

3.2 Life Cycle Inventory Analysis

In order to evaluate the environmental impacts of the five configurations, some key values are necessary, namely the mass of fuels and respective oxidizers, mass of Aluminium and Carbon Fibre Filament necessary for building the respective tanks, and the products of the fuel combustion. Furthermore, because SimaPro did not include the manufacturing processes related to the Carbon Fibre Tanks, these needed to be modelled so the energy involved was also necessary to estimate.

From the previous phase of this project, the masses of each fuel and material for the tanks are already known. The proportion of oxidizers used for each fuel was taken from online sources related to the aerospace industry.

The combustion products were not as easy to compute as was previously expected. The confidentiality characteristic of aerospace and aeronautical technology hindered the obtainment of the combustion products, thus the only solution found was to estimate the amount of combustion products through direct analyses of the chemical equation of this process in its simplest form without taking into consideration variations in pressure and temperature. Given the reactants mass it is possible to get simple equations that allow to compute the mass of products from the combustion.

Although the chemical formula for RP-01 was not possible to know for certain, since it is a mixture of various hydrocarbons ($CH_{1.97}$), the approach taken was similar to the ones in the literature where a model of a generic hydrocarbon is used. For this study, cyclododecane ($C_{12}H_{24}$) was the chosen compound.

The following equations represent the combustion reactions for each fuel

$$\text{UDMH: } C_2H_8N_2 + 1.87N_2O_4 \longrightarrow 1.48CO_2 + 4H_2O + 2.87N_2 + 0.52CO$$

$$\text{RP-01: } C_{12}H_{24} + 13.23O_2 \longrightarrow 2.46CO_2 + 12H_2O + 9.54CO$$

$$LH_2 : 4H_2 + O_2 \longrightarrow 2H_2O + 2H_2$$

It is possible to verify that both the combustion of RP-01 and Hydrazine are done with a fuel-rich mixture whereas the liquid hydrogen is oxidized in a fuel-lean mixture.

For the estimation of the energy related to the manufacturing of the Carbon Fibre tanks it was necessary to find the power of the machinery responsible for each process. Given the power, it was just a matter of estimating the total time or velocity of the process and then compute the total energy necessary. This information is present on the table 3, which can be found on the costs analysis' section of this project.

3.3 Life Cycle Impact Assessment Analysis

Using Simapro several assemblies were created. As stated previously, the functional unit of this study was a space mission capable of delivering a specific payload to around 230 km of altitude. Thus, the assemblies constructed in Simapro were all created accounting for the necessities of each different configuration of fuel and fuel tank in order to ensure mission success.

Fuels

As stated previously, in this study a well-to-wheel approach was used for the fuels. In order to perform the LCA on each fuel, all fuels were assumed to be bought at market in Simapro. Nonetheless, all fuels were transported to the launch site and all fuels had their respective emissions accounted for as "Emissions to Air". In figures 10 and 11, we can see the results of the LCA in terms of the fuel manufacture, shipping and combustion.

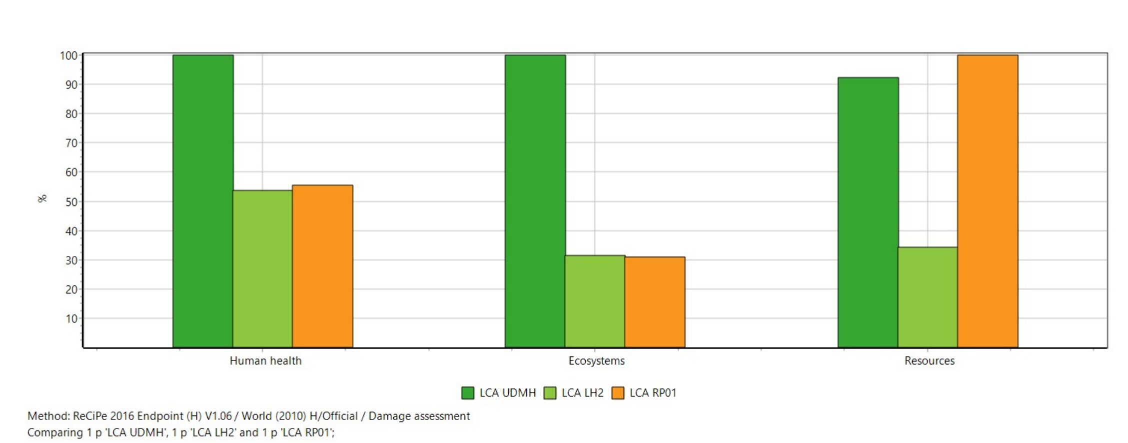

Figure 10: Life Cycle Assessment for fuels in terms of major categories.

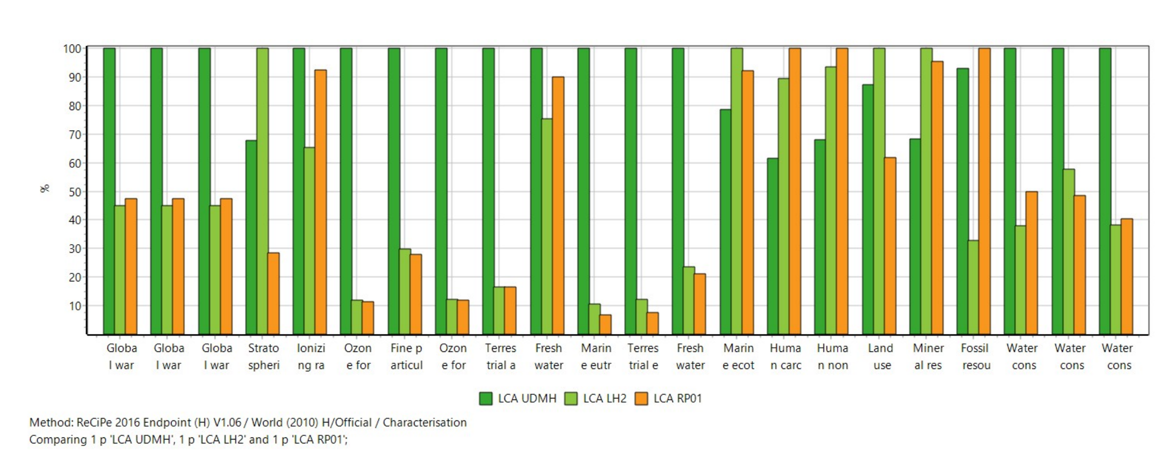

Figure 11: Life Cycle Assessment for fuels in terms of minor categories.

From Figures 10 and 11, it is possible to see that the liquid hydrogen fuel (LH2) yields the best results overall, which was expected given the cleanliness of this fuel and its easiness to produce via water electrolysis.

Fuel Tanks

The fuel tanks utilized in this study served only one purpose which was to deliver the payload to the desired altitude, these fuel tanks were expected to burn and disintegrate upon reentry in the atmosphere. Therefore their usage should only be done a single time for each mission. Five different fuel tanks were constructed using an assembly in SimaPro, each for every configuration. For the aluminium fuel tanks, sheets of market aluminium underwent a succession of processes such as rolling and TIG (Tungsten Inert Gas) welding in order to manufacture the tanks. For the carbon fibre fuel tanks, filament of market carbon fibre underwent a process which accounted for the energy necessary to manufacture the tanks. This was performed due to the fact that Simapro did not have the exact processes needed to manufacture these tanks. After manufacturing, the tanks were dispatched to the launch site. In Figures 12 and 13 we can see the results of the LCA comparison between the different fuel tanks for each fuel.

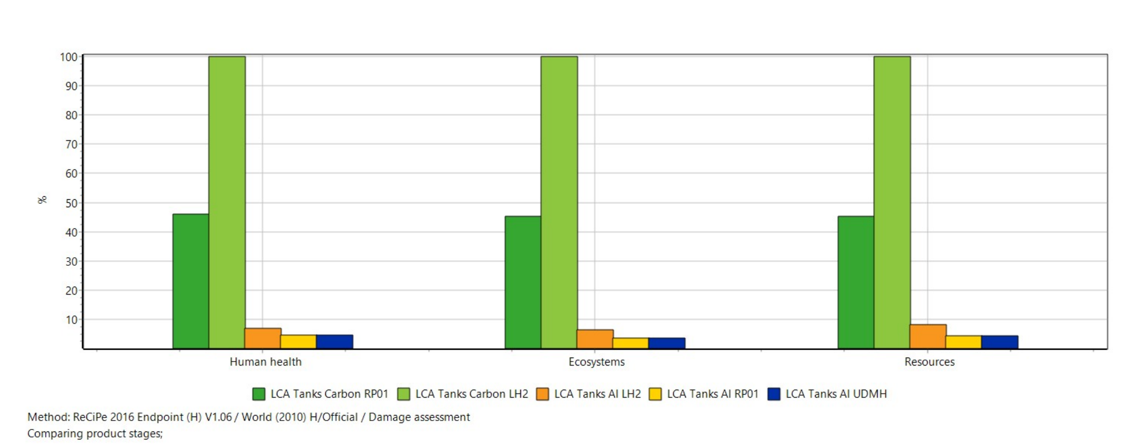

Figure 12: Life Cycle Assessment for the fuel tanks in terms of major categories.

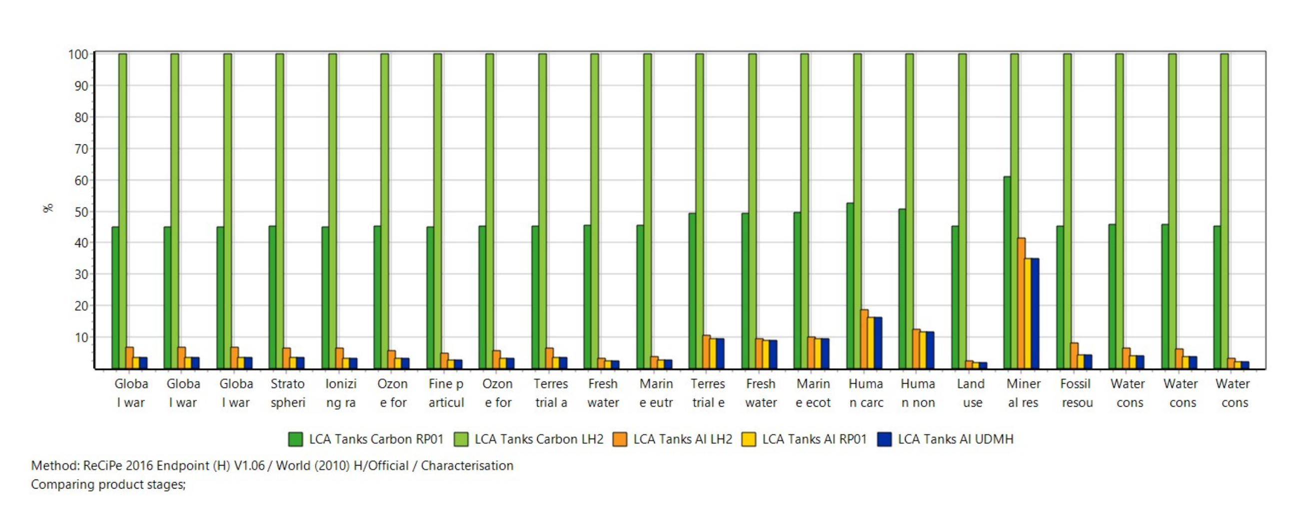

Figure 13: Life Cycle Assessment for the fuel tanks in terms of minor categories.

From Figures 12 and 13, one can see that the fuel tanks that yielded the best results are the Aluminium tanks, this is due to the fact that carbon fibre production and the fuel tanks manufacture are still extremely unsustainable processes. However, within the Aluminium fuel tanks, the one that had the best overall results was the Aluminium fuel tank for UDMH this is because it needed the lowest mass of Aluminium to be manufactured. Nonetheless, even though overall the Aluminium LH2 fuel tank is less sustainable than the Aluminium UDMH fuel tank, the difference lies below % 5, which is a negligible difference in sustainability.

All Configurations Comparison

In order to compare side-by-side each of the studied configurations an LCA comparison of the assemblies comprising each fuel and each fuel tank was performed. In Figures 14 and 15, we can see the results of these comparisons. From the previously shown results, it was expected that the configurations that yielded the best results were the Aluminium tanks for both RP01 and LH2. This arises from the fact that both RP-01 and LH2 fuels are more sustainable to manufacture than UDMH and from the fact that Aluminium fuel tanks are more sustainable to manufacture than carbon fibre fuel tanks.

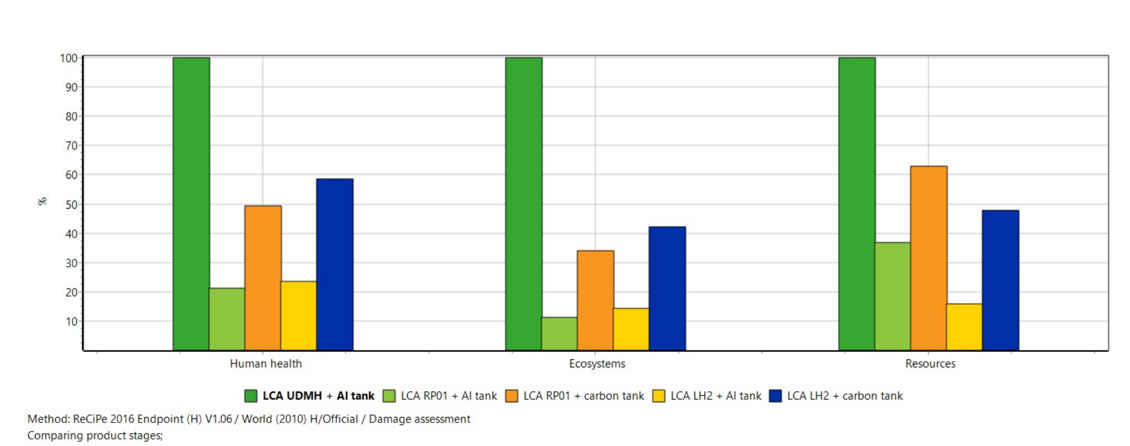

Figure 14: Life Cycle Assessment for the 5 configurations in terms of major categories.

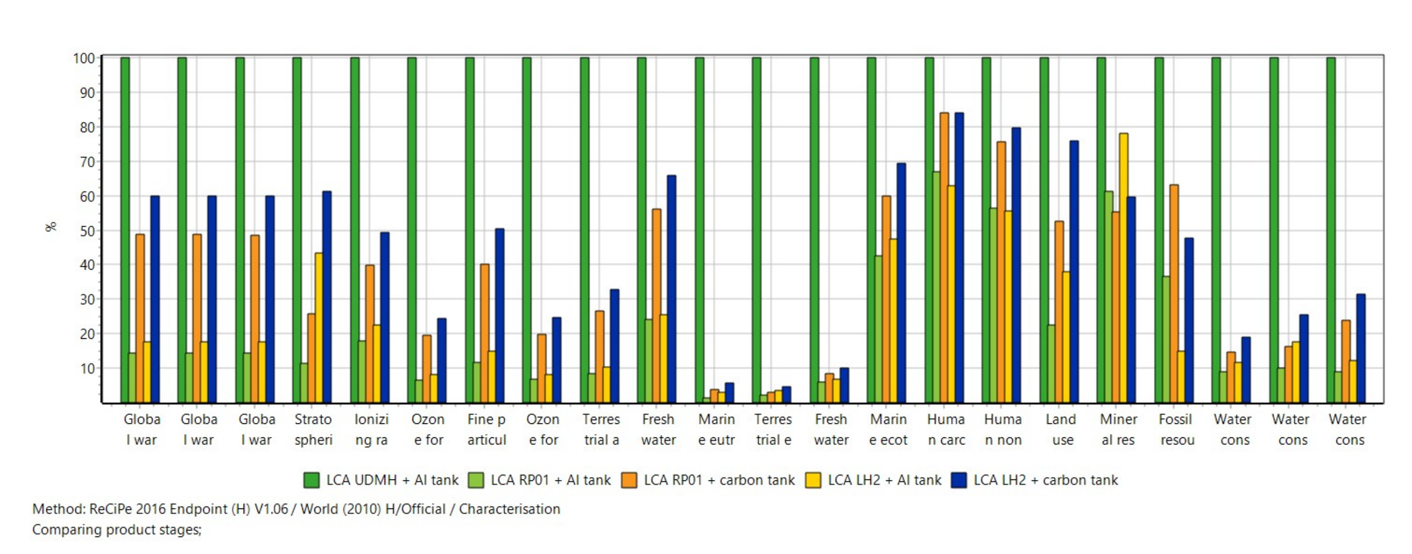

Figure 15: Life Cycle Assessment for the 5 configurations in terms of minor categories.

In fact, from Figures 14 and 15, it is possible to conclude that the best configuration is either LH2 fuel with an Aluminium fuel tank or RP-01 fuel with an Aluminium fuel tank. However, there is a clear advantage in sustainability for the former.

Best Configuration

The best configuration according to the previously shown results is the LH2 fuel with Aluminium fuel tank. Therefore, it is important to show the LCA of that specific configuration. In figures 16 and 17, we can see the LCA discriminated in its most dominant parcels.

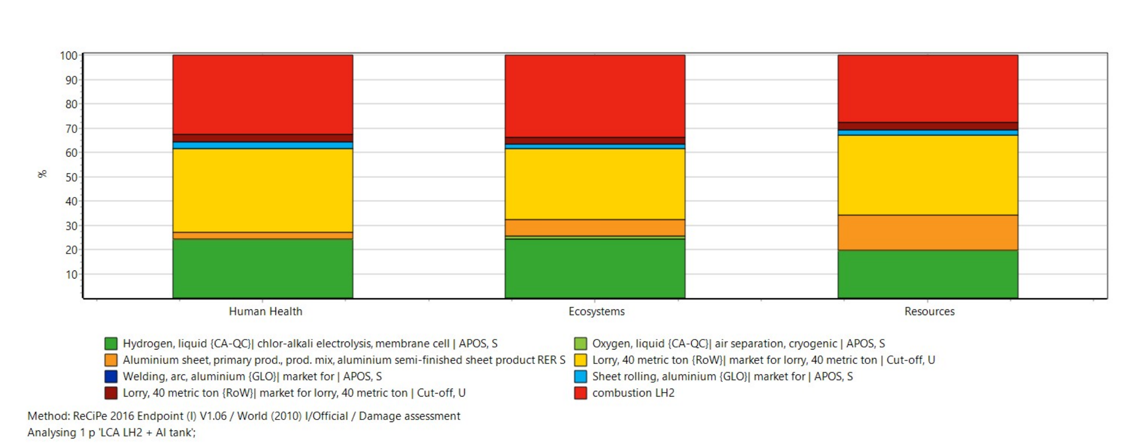

Figure 16: Life Cycle Assessment for the configuration with LH2 and Aluminium tank in terms of major categories.

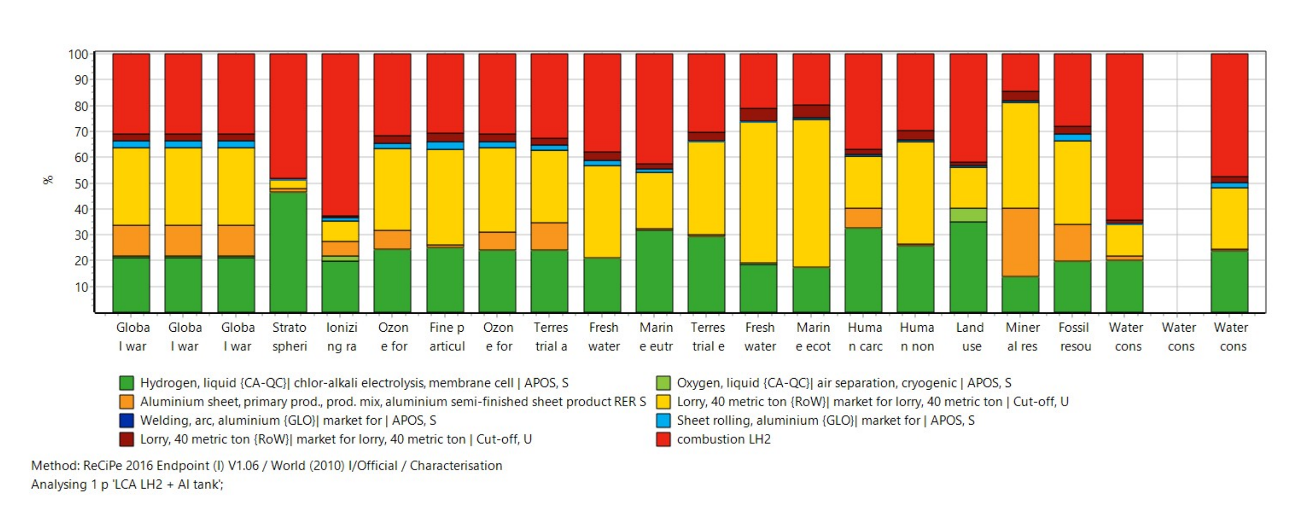

Figure 17: Life Cycle Assessment for the configuration with LH2 and Aluminium tank in terms of minor categories

From the above Figures, one can see that the majority of the impacts in this configuration comes from liquid hydrogen production, its transportation and from hydrogen combustion. The result that hydrogen combustion represents a high parcel of the impacts might lie on the fact that the GWP (Global Warming Potential) of water vapor, although very low, is not zero, and therefore, high emissions of water vapour to the atmosphere still contribute to the green house effect.

4 Costs Analysis

The costs analysis' section of this project will consider the following main factors: fuel costs, materials costs, manufacturing costs, and insulation costs. These points will be developed more thoroughly in the next subsections. The values gathered were collected from various online sources, such as, catalogs from manufacturers, online forums and others.

4.1 Fuel Costs

To determine the fuel costs for the rocket designs, the first step was to evaluate the cost per kilogram of fuel and then calculate the overall expense for each individual rocket configuration. By implementing this approach, it was possible to estimate the total fuel costs.

	Fuel / Ox	**Mass (kg)**	**\$/kg**	**Total cost (\$)**	**Total cost (\$)**
Fuels/Oxidizers	RP-01	179941	\$1,15	\$206 932	\$274 948
	LOX - RP-01	453442	\$0,15	\$68 016	
	UDMH	163661	\$3,50	\$572 813	\$1 902 098
	NTO	469712	\$2,83	\$1 329 284	
	LH2	84000	\$4,50	\$378 000	\$428 400
	LOX - H	336000	\$0,15	\$50 400	

Table 1: Fuel Costs

As the above table shows, the most expensive fuel is the UDMH with a total cost of almost \$2M. This value is expected, since both the NTO and the UDMH have the highest and second highest prices, respectively, of all compounds. The lowest total cost is related to the configuration with RP-01 and LOX.

4.2 Materials Costs

Another vital aspect of rocket costs is the cost of the tanks. As previously stated, the two types of materials for the tanks are aluminium and carbon fiber filament. The cost of aluminum is \$7, while the cost of carbon fiber filament is approximately \$82 (values per kg of material). This leads to a significant disparity in price, despite the fact that the masses required for each material are vastly different. This information is crucial in determining the most cost-effective material to use on the different designs, and will play a key role in the budgeting and financial planning for the project.

Fuel	Material	Mass (kg)	$/kg	Total cost ($)
RP-1	AL 2219	30627	$7,00	$214 389
	CF Filament	12068	$81,57	$984 386
UDMH	AL 2219	30450	$7,00	$213 150
LH2	AL 2219	68799	$7,00	$481 593
	CF Filament	27108	$81,57	$2 211 199

Table 2: Materials Costs

4.3 Manufacturing Costs

In addition to the materials expenses, it is also necessary to factor in the costs related to the manufacturing processes for each tank. In order to do this, labour and energy expenses are taking into consideration. Additionally, the main manufacturing processes for each tank type are evaluated.

As the Table 3 indicates, these values will not have a significant impact on the final costs due to their small magnitude compare to the other factors already assessed.

Material	Aluminium Alloy		Carbon Fiber Filament	
Manufacturing Process	**Roll Bending**	**TIG Welding**	**Shape Winding**	**Furnace Curing**
Power (kW)	37	10	18,5	75
RP-1 Time (h)	0,03	12,1	6	15
LH2 Time (h)	0,03	26,9	13,55	20
Labour Cost ($/h)	$19,00		$45,00	
Energy Cost ($/kWh)	$0,15			
RP-1 Total Labour Cost ($)	$2,85	$1 149	$1 350	$3 375
LH2 Total Labour Cost ($)	$2,85	$2 555	$3 048	$4 500
RP-1 Total Energy cost ($)	$0,17	$18,15	$16,65	$168,75
LH2 Total Energy cost ($)	$0,17	$40,35	$37,60	$225,00

Table 3: Manufacturing Costs

4.4 Insultation Cost

When working with liquid hydrogen and liquid oxygen, the rocket's tanks need cryogenic insulation to maintain the fuel in a liquid state. This significantly increases the cost of insulation, as the table shows that cryogenic insulation is 50 times more expensive than traditional insulation. As can be seen in Table 4, the liquid hydrogen tanks are by far the most expensive regarding insulation. This is caused by the low density of the liquid hydrogen, which leads to the largest tank of all configurations, and to the fact that for this configuration both fuel and oxidizer need to be kept liquid.

	RP-1	LOX	LH2	LOX	UDMH	NTO
Diameter (m)	7		7		7	
Length m()	18,447		41,4388		18,34	
Type of insulation	Storable	Cryogenic	Cryogenic	Cryogenic	Storable	Storable
Insulation cost($/sqm)	$16,84	$861	$861	$861	$16,84	$16,84
Total Cost ($)	$355 933		$1 568 440		$13 576	

Table 4: Insulation Costs

4.5 Total Cost

In conclusion of this analysis, the total values for each main factor and each configuration are summarized in Table 5 and the overall costs are calculated.

	RP-01		LH2		UDMH
Material	Aluminium	CF Filament	Aluminium	CF Filament	Aluminium
Fuel Costs ($)	$274 948	$274 948	$428 400	$428 400,00	$1 902 098
Material Costs ($)	$214 389	$984 386	$481 593	$2 211 199	$213 150
Insulation ($)	$355 933	$355 933	$1 568 440	$1 568 440	$13 576
Energy Manufacturing ($)	$18	$185	$40	$262	$18,10
Labour Costs ($)	$1 152	$4 725	$2 558	$7 548	$1 148
Total ($)	**$846 441**	**$1 620 179**	**$2 481 032**	**$4 215 851**	**$2 129 991**

Table 5: Total Costs

The table clearly illustrates that the fuels, materials and insulation have the most significant impact on the final cost. From an economic standpoint, the most cost-effective option is using RP-01 fuel with aluminum tanks.

5 Conclusion

After comparing five different configurations of fuel and tank materials according to their performance, environmental impacts and costs associated, the results have shown to be matching what was expected.

Summarizing the steps, firstly, in order to evaluate the capacity of each fuel to achieve the mission established, a performance analysis was carried out for each different propellant where structural design, aerodynamics and combustion parameters were all taken into consideration. By doing this, the masses of each propellant were computed, showing that it was possible for every combination to achieve the goal set at the beginning. On the next chapter, a Life Cycle Assessment was implemented with the goal of evaluating the environmental impact of each configuration. This way, it was possible to conclude that carbon fiber tanks were not viable environmentally and that LH_2 was the better alternative regarding fuel choice, while UDMH revealed to be, as expected, the worst of the three. Finally, a cost analysis was performed with the goal of estimating the total costs associated with the different configurations, namely with propellant, tank material, manufacturing and insulation costs. Once again the Carbon Fiber tanks were outperformed by the ones in Aluminium.

Considering the configurations themselves, table 6 shows the ranking of all of them in terms of LCA comparison and Cost Analysis comparison. Additionally, to find the best solution an approximate formula was employed, which takes into account both the analyses done for each combination.

Table 6: Configuration Ranking

Configuration	Cost Analysis Ranking	LCA Ranking	Result (Cost × LCA)	Overall Ranking
RP-1 + Al	#1	#2	2	#1
RP-1 + C	#2	#3	6	#3
LH2 + Al	#4	#1	4	#2
LH2 + C	#5	#4	20	#5
UDMH + Al	#3	#5	15	#4

As one can see, the best overall configuration is the RP-1 with an Aluminum tank, mostly due to financial reasons. However, the sustainability of the LH2 with Aluminum tank is the highest one so one can argue that the latter could be the best choice. Since sustainability is the main concern, the best overall configuration accounting for our established trade-off between cost and sustainability is the **LH2 powered rocket with an Aluminium fuel tank**.

In closing, it is possible to conclude that RP-1 and LH_2 with Aluminium tanks are the most viable choices. In any case, performance is always the main goal when constructing rocket launchers. That means that Carbon Fiber might be viable in certain cases due to its reduced weight. Furthermore, Hydrazine is also used due to some of its peculiarities as a fuel which were not discussed in this project since they are mostly unrelated. For all this reasons and many more, even though the result on the analysis done still stands, a more thorough approach would be recommended in order to find the optimal choice for each specific application.

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