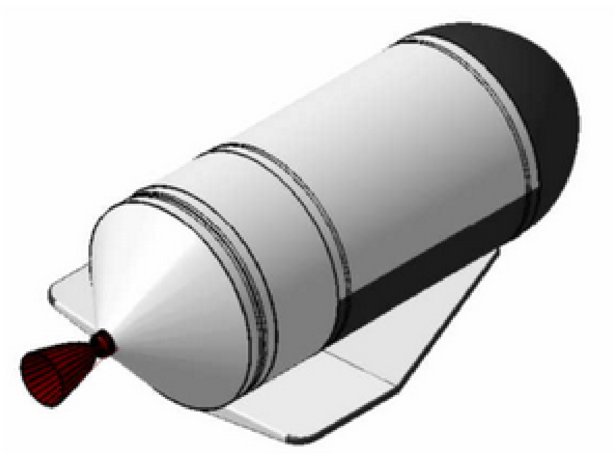
# GREDER
# Green RE-usable DEbris Remover



*Authors :*
Tim ***Lewis***
Alina ***Trifunovic***
Lukas ***Krause***
Alexis ***Rolin***
Julien ***Huynh***

*Supervising professor :*
Prof. Dr.-Ing. Uwe *Apel*

Version 0.7
February 23, 2020

# Contents

# Chapter 1

# Introduction

Ever since the launch of Sputnik 1, the first artificial satellite in October 1957, the number of satellites launched has sharply risen and as of 2020, thousands of satellites are orbiting around the Earth. However, each of every one of those spacecrafts will eventually see their mission being stopped, usually due to a lack of resources from the satellite itself as it reaches the end of its life.

Those who are at the end of their lives will turn into a simple uncontrolled object that keeps orbiting around the Earth and should be avoided as another functioning satellite could take their slot as some of the most important orbits around our planet are starting to get overcrowded and the demand keeps rising.

Moreover, such uncontrolled objects in space can become dangerous as collisions could potentially happen and at such high velocity, those can heavily damage other spacecrafts and create even more debris.

As public awareness grows towards the space debris problem, our mission, Green Debris Remover (GREDER), is looking to contribute to a solution to this problem in a particular orbit, the geo stationary orbit, which is particularly overcrowded due to the many different kinds of satellite operating there.

# Chapter 2

# Schedule

At the start of the project a dedicated group meeting was performed in order to agree on a common project sequence, tasks and challenges as well as work distribution. This group meeting was deemed essential to structure the work packages and to achieve a consolidated baseline for the whole project including time line.

The result is a complex MS project Gantt diagram, which can be found in Annex 1.

## 2.1 Initial Schedule

The first version of the schedule starts with a project Kick-Off in October which is afterwards followed by a short planning phase. In this planning phase issues as scheduling, work distribution and scope of the project were addressed.

Subsequently, the definition phase started. Within this phase, the vehicle requirements were defined and the mission was planned, calculated and visualized in MATLAB. The outcomes of the definition phase are the boundary requirements which are set to provide a frame for both: the vehicle itself as well as the propulsion system. The requirements were defined at the beginning of the project and were verified after project completion.

Upon definition of the boundary requirements, the specification phase started. Within this phase, different propellant combinations were identified, discussed and compared. Additionally a first mass budget was calculated. The result of this phase is the system specification.

The sequence of the project includes several presentations. The first one was performed in October for a quick overview on the project planning. The second one after the boundary requirements and system specification was set. After this presentation, the vehicle and sub system design phase started. This phase included major parts of the work packages including propulsion system design with all sub-assemblies as RACS/ACS, propellant tanks, feeding and pressurization system, turbo pumps, catalyzer, engine, injector and nozzle. The outcome of this phase is a preliminary vehicle design and sub system design which was presented in the mid-term presentation.

As a last major work package the simulation phase started. The whole system was simulated including all sub systems and additionally the complex H2O2 decomposition regulation. The final presentation was performed after all tasks were completed and the simulation was finalized.

An overview the compressed initial schedule is shown in Figure 2.1. The detailed Gantt schedule can be found in Annex 1.

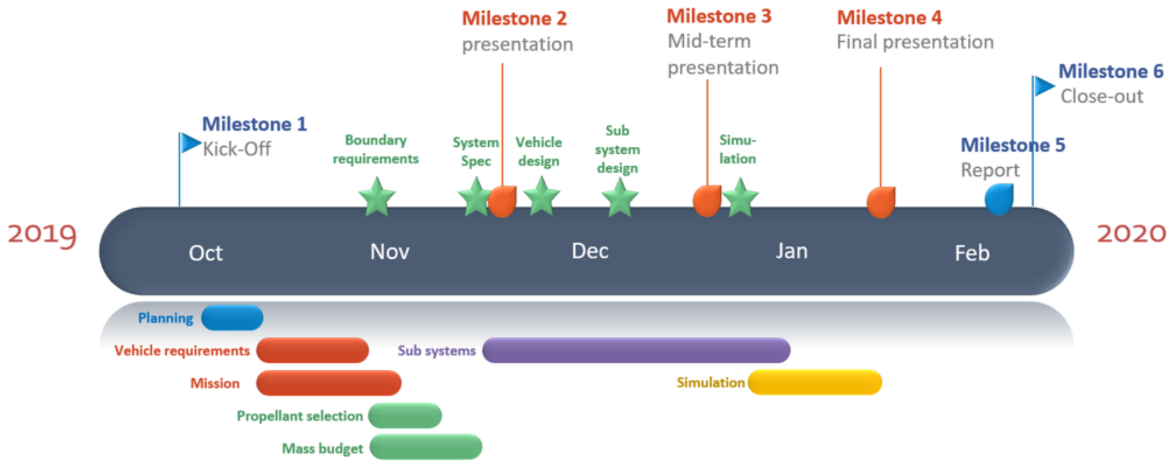



Figure 2.1: Initial schedule

## 2.2 Final schedule – comparison as planned and as achieved

As usual in project management and project work, not all milestones were achieved in time. As it is shown in the compressed final schedule in Figure 2.2, the finalized vehicle design, the finalized sub system design and the corresponding simulation shifted within the project schedule (Figure 2.2, shown in red). Nevertheless all work packages have been successfully completed until Milestone 4, the final presentation.

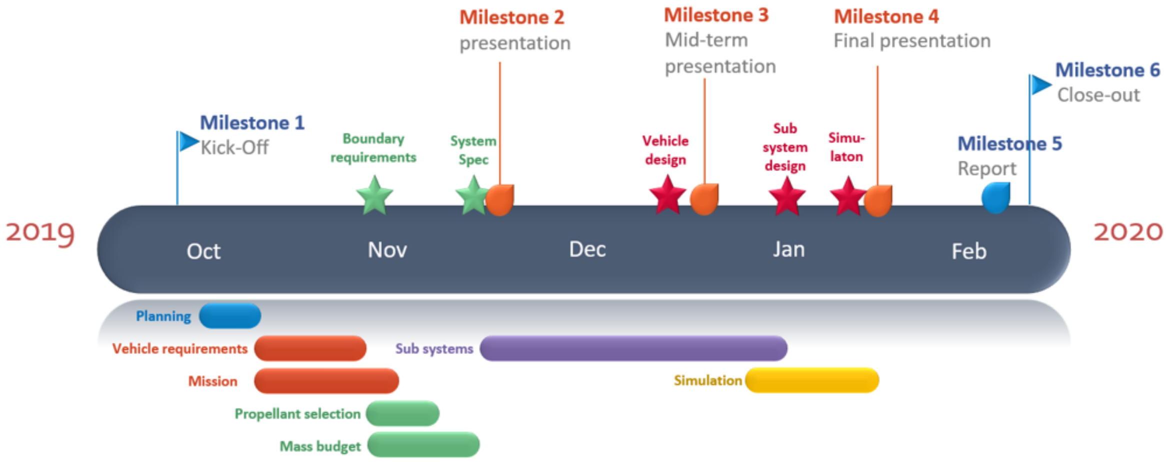



Figure 2.2: Final schedule - comparison as planned and as achieved

# Chapter 3

# Requirements

The top level requirements for the vehicle and propulsion system are divided in three categories: operation, environment and vehicle. The operation requirements are related to the propulsion system and its applicability for the planned missions. The environmental requirements are defined to ensure that both the vehicle and propulsion system is capable of operating and sustaining during launch, mission and in space environment. The scope of the vehicle requirements is to cover the major parts of the planned mission as refuel-ability, aero braking or accuracy.

| **Operation** |
|---|
| TL-1 Provide sufficient thrust for completion of the mission profile including a safety margin |
| TL-2 Re-ignitable at least 1000 times |
| TL-3 Service life time of at least 100 missions or 25 years in orbit |
| TL-4 Ignition and functional reliability shall be higher than 99,5% |
| **Environment** |
| TL-5 Withstand the launch phase |
| TL-6 Operate in vacuum |
| TL-7 Operate in an ambient temperature range of 1K to 5K |
| TL-8 Withstand the temperature gradients resulting from areas turned towards or away |
| TL-9 Sustain space-related radiation throughout it's complete life time |
| TL-10 Withstand debris impact of under 1cm diameter with a max relative speed of 15 km/s |
| **Vehicle** |
| TL-11 The engine shall be the main propulsion system of a GEO satellite recovery vehicle |
| TL-12 Refuelable between missions |
| TL-13 Perform aerobrake maneuvers in Earth's atmosphere. |
| TL-14 Control flight path in Earth's atmosphere using non-propulsive flight control systems<br>TL-15 Remain within the ARIANE 6/Falcon 9 payload launch capabilities to LEO.<br>TL-16 Remain on it's guided trajectory with less than 0.1% deviation. |

Table 3.1: Requirements for the vehicle

# Chapter 4

# Specification

## 4.1 Satellite catching process

### 4.1.1 Choice of the process

In order to catch and de-orbit a satellite in GEO, we considered the following usable tools :

- Net
- Harpoon
- Claw
- Magnet

Other solutions such as towing the satellite or simply removing them from GEO were not considered as they either did not fit our program or would create too much strain on our spacecraft.

We then took a closer look at the feasibility of each solution and compared the advantages and disadvantages :

| Solution | Advantages | Drawbacks | Feasibility |
|---|---|---|---|
| **Net** | Cheap, simple, low mass | Slow, hard to handle | Yes (JAXA, ESA) |
| **Harpoon** | Fairly cheap | Can create more debris | Yes (ESA) |
| **Claw** | Safer | Mechanical, moving parts | WIP (CleanSpace One) |
| **Magnet** | Adjustable, no moving parts | Higher mass, needs power | Research State |

As the main focus of our mission is reliability and re-usability, we made the choice of using magnets to catch and hold the satellite we would like to de-orbit.

### 4.1.2 Magnetic solution

Even though we decided that we would use magnets, we needed to make sure it was feasible and to lower the drawbacks related to this solution as much as possible. The first precision we need to make is that we will be using electromagnets in order to regulate the intensity of the current in the coil of it, thus, modulating the attraction force so the contact between our spacecraft and the satellite will not be made at high velocity, avoiding damages and space debris creation.

Even though it is still at the state of research, we believe that using electromagnets as our catching solution is realistic as both ESA (with ISAE SupAero) and the NASA have been considering and studying this solution since 2017.

However, as a matter of complexity, we will have to make assumptions in order to simplify the problem. The objective in this part is to prove that, with assumptions, this solution can be applied to our mission and to find the required energy to both catch and hold the satellite until its release.

**Assumptions**

In our calculations, we assumed that :

1. We can consider the magnetic circuit between GREDER and the nozzle of the satellite is a closed one (no air gaps)
2. About 0.1% of the satellite's mass is magnetically operable
3. Mutual attraction is relatively low compared to the magnetic force
4. Residuals in the alloy of the magnets' cores are neglectable

### 4.1.3 Sketching and Calculations

Using the first assumption, we can use the formula for the magnetic force in a magnetic circuit with no air gap :

$$F = \frac{(\mu N I)^2 A}{2\mu_0 L^2} \tag{4.1}$$

With :

- $\mu$ the magnetic permeability of the core of our magnet (determined by the alloy)
- $N$ the number of turns of the coil around the core
- $I$ the current running through the coil

- $A$ the cross section area of the core
- $\mu_0$ the magnetic constant
- $L$ the length of the mean magnetic circuit

In order to have a good balance between thermal properties and magnetic properties we decided to use an alloy made of 90% of iron and 10% of cobalt for our core. We decided to use this alloy as iron has the best magnetic properties (high relative magnetic permeability) and added cobalt as it has a higher Curie Temperature than iron but has a lower relative magnetic permeability.

In terms of magnet design, we chose to use a squared cross section of $5cm \times 5cm$ for the magnet core and with a length of $15cm$, made of an iron-cobalt alloy and a copper coil around it. We decided to have 135 turns of the coil around the core with a coil diameter of $1mm$ in order to not have the wire revolutions stuck to each other. We also want to run a current of $10A$ through the coil.

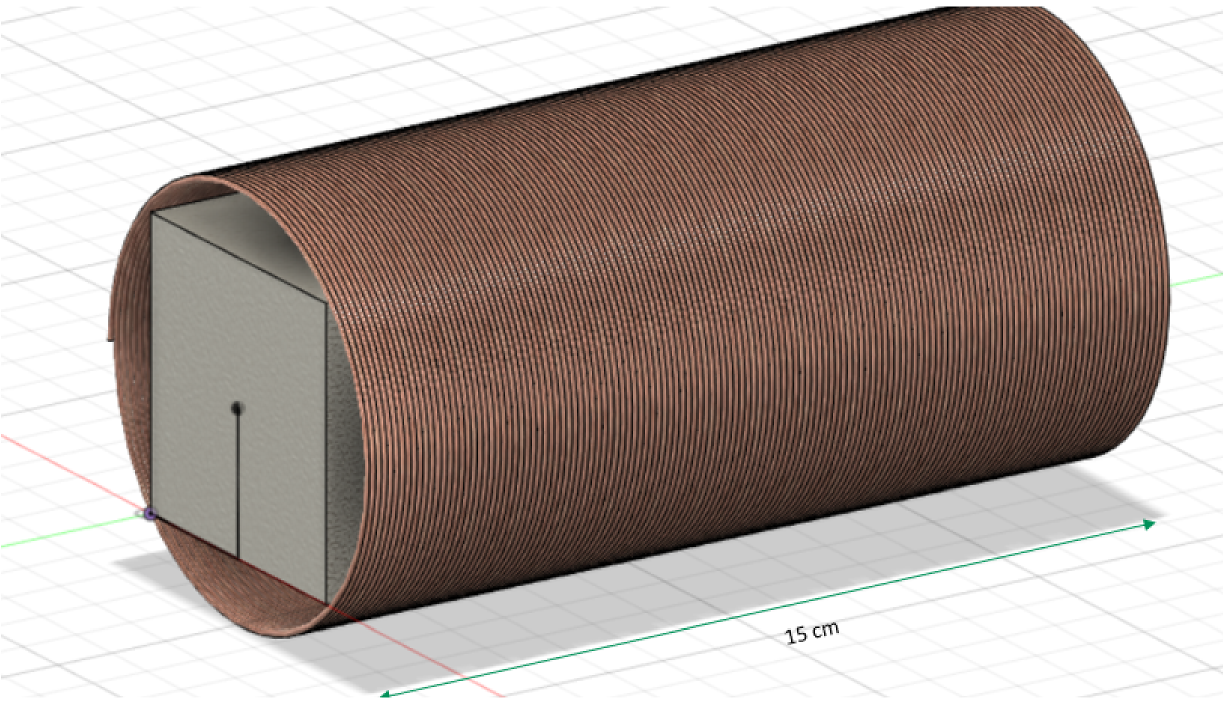



Figure 4.1: CAD of a catching magnet

With those design choices we get the following parameters while taking into consideration that there should not be any kind of residuals in the core alloy :

- $\mu_0 = 4\pi \times 10^{-7}\ H/m$

- $\mu = \left(\frac{\mu_{iron}+\mu_{cobalt}}{2}\right) \times \mu_0 = 5.868 \times 10^{-3}\ H/m$
- $N = 135$ turns
- $I = 10\ A$
- $A = 25cm^2 = 2.5 \times 10^{-3}\ m^2$
- $\rho_{core} = \rho_{iron} \times 0.9 + \rho_{cobalt} \times 0.1 = 7.9726\ g/cm^3$
- $\rho_{coil} = \rho_{copper} = 8.96\ g/cm^3$

We then need to find the length $L$ of the mean magnetic circuit. In order to do so, we decided to start the catching sequence at 10 $m$ from the satellite and that the target has an exploitable nozzle of 40 $cm$ of diameters which is realistic for spacecrafts in the mass range of 3 500 $kg$. We could then sketch the catching sequence as so (not to scale) :

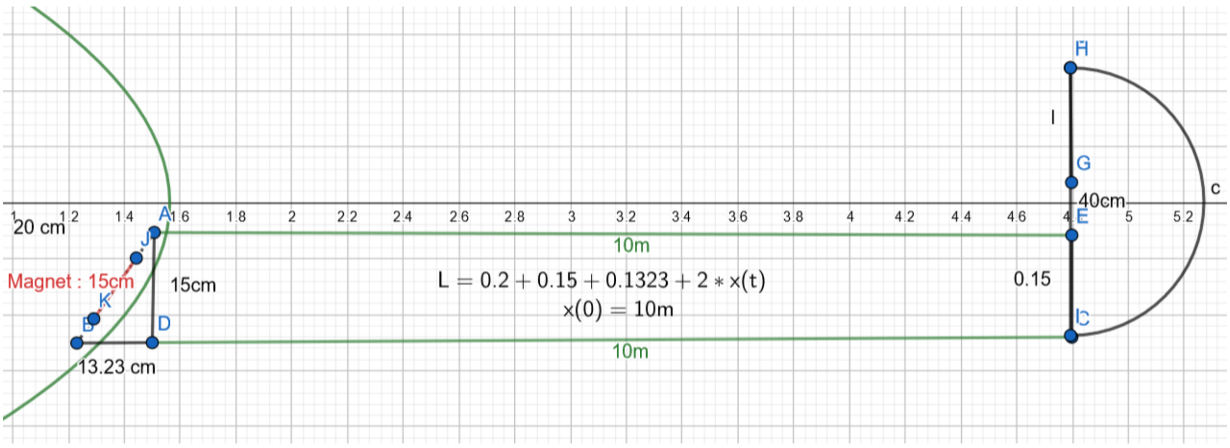



Figure 4.2: Catching sequence (not to scale)

We can then have the length $L$ as a function of the distance between the tip of our spacecraft and the nozzle of our target. As a result we can proceed to find the feasibility of our solution with this magnet design by finding the time it would require at this state to attract the target. However, in this case, the current modulation when the target is close has not been modeled due to its complexity.

Considering that the force will be on one axis only and $m = 0.001 \times m_{target} = 3.5\ kg$ :

$$\vec{F} = m\vec{a} \tag{4.2}$$

$$\frac{(\mu NI)^2 A}{2\mu_0 L(x)^2} = m \times \ddot{x} \tag{4.3}$$

$$\frac{(\mu NI)^2 A}{2\mu_0 [0.2 + 0.15 + 0.1323 + 2x(t)]^2} = m\ddot{x}(t) \tag{4.4}$$

$$\frac{(\mu NI)^2 A}{2\mu_0 [0.4823 + 2x(t)]^2} = m\ddot{x}(t) \tag{4.5}$$

The catching time can then be found using *ode*45 on Matlab :

```
clearvars; clc;
catchtime = 1;
x0 = 10;
while 1
    [t,x] = ode45(@f3,[0:1:catchtime], [x0; 0; 0; 0]);
        if (x(catchtime ,1) >= 2 * x0)
            break
        else
            catchtime = catchtime +1;
        end
end
```

And the function used for the *ode* solver :

```
function  [Xdot] = f3(t, X)
mu = 5.686e-3; mu0 = 4 * pi * 10 ^(-7);
N = 135; I = 10; A = 0.05 ^ 2; m = 3500 ;
x = X(1); y = X(2); vx = X(3); vy = X(4);
Fmag = (mu * N * I) ^2 * A / (2 * mu0 *(2 * norm(X(1:2)) + 0.4823) ^ 2 );
Xdot = [vx; vy;  0.001 * Fmag / m; 0];
end
```

We then get a catching time of 812 seconds. As a result, we can determine the energy required to operate the magnets aswell as their masses and volumes. We are considering a holding time for approximately half an hour and we also need to verify that the magnets will be able to hold the target while we are de-orbiting.

## 4.2 Propellant selection

## 4.3 Mass analysis

## 4.4 Mass Budget - First Iteration

Before actually going into our mass budget, we wanted to get a reference idea for the propellant mass so that we would be sure to be able to achieve our $\Delta v$. In order to get this, we decided to find a relation between the usable propellant mass and the mass of the rest as a ratio. This is then fixed and will also allow us to know roughly how much propellant we need depending on the dry mass. Let $m_{UP}$ be the mass of usable propellant. Moreover, we would be aiming for a total initial mass of roughly 20 to $25t$ on our last iteration. This first iteration was done with another magnet design, presented in November which consisted in two large discs of 600 $kg$ each and have then been abandoned for the second iteration.

### 4.4.1 Coefficients & Masses after steps

Considering that $ISP = 295s$ and annotating $\frac{m_{UP_i}}{m_{total_i}} = K_i$ with $i$ the burn number :

| Step | Required $\Delta v$ in $m/s$ | $K_i$ | Mass after step |
|---|---|---|---|
| 1 | 2802.4 | 0.620 | 0.38 $m_{initial}$ |
| 2 | 1342.2 | 0.371 | $0.239m_{initial}$ |
| 3 | 522.9 | 0.165 | $0.200m_{initial}$ |
| Satellite caught | NA | NA | $0.2m_{initial} + 3500$ |
| 4 | 1487.8 | 0.402 | $0.1196m_{initial} + 2093$ |
| Satellite release | NA | NA | $0.1196m_{initial}$ - 1407 |
| 5 | 5.3 | 0.002 | $0.1194m_{initial}$ - 1404.186 |
| 6 | 72.4 | 0.0247 | |

### 4.4.2 Global equation between $m_{UP}$ and $m_{initial}$

| Step | $\frac{m_{UP}}{m_{initial}}$ | Bias due to debris |
|---|---|---|
| 1 | 0.620 | 0 |
| 2 | 0.141 | 0 |
| 3 | 0.039 | 0 |
| 4 | 0.0804 | 1407 |
| 5 | 0.00024 | -2.814 |
| 6 | 0.00295 | -36.68 |
| TOTAL | 0.88359 | +1369.506 |

We then get our general relation between the usable propellant mass and the initial mass

$$m_{prop} = 0.88359 m_{init} + 1369.506$$

And as $m_{initial} = m_{UP} + m_{rest}$ :

$$m_{prop} = \frac{1}{0.11641}\left[0.88359 m_{rest} + 1369.506\right]$$

$m_{rest}$ includes the dry mass and the propellant required for the ACS.

### 4.4.3 First iteration of mass budget

**Sub systems**

| Contributor | Mass in kg |
|---|---|
| **EPS** | - |
| Fuel cells | 165.6727 |
| H2 for fuel cell (tank included) | 10 |
| Cables | 20 |
| GNC | 5 |
| Batteries | 61.3333 |
| Actuators (for flaps) | 10 |
| Servos | 1 |
| **On board computer** | 5 |
| **Telecommunications** | 10 |
| **Thermal control** | 10 |
| **ACS/RCS** | - |
| Reaction wheels | 106 |
| ACS (without propellant) | 36.16 |
| ***Total*** | 440.166 |

**Payload**

| Contributor | Mass in kg |
|---|---|
| Magnet | 1200 |

**Structure**

| Contributor | Mass in kg |
|---|---|
| Hull | 509 |

| Contributor | Mass in kg |
|---|---|
| Wing | 54 |
| Engine | 60 |
| Engine frame | 51 |
| Connectors | 25 |
| Tanks | 350 |
| Heat shield | 472 |
| ***Total*** | 1521 |

### Others

| Contributor | Mass in kg |
|---|---|
| Catalyzer | 10 |
| Lines | 25 |
| ACS including Propellant | 672 |
| Non usable propellant (Residuals, transient, etc.) | 200 |
| Helium (including tank) | 30 |
| ***Total*** | 937 |

We then get

$$m_{rest} = m_{Subsystems} + m_{Payload} + m_{Structure} + m_{Others} = 4098.166kg$$

Which, with the previously obtained equation :

$$m_{UP} = 42\ 870.926kg$$

As the mixture ratio is $MR = 7.07$ and $m_{UP} = m_{UF} + m_{UOP}$

$$m_{UsableFuel} = \frac{m_{UP}}{1 + MR} = 5\ 312kg$$
$$m_{UsableOxidizer} = MR \times m_{UsableFuel} = 37\ 559kg$$

### Results

We can sum this first iteration up with the following table :

| Contributor | Mass (kg) |
|---|---|
| **Structure** | 1 521 |
| **Magnets** | 1 200 |
| **Sub Systems** | 440.166 |
| **Tank Pressurization** | 30 |
| **Engine** | 60 |
| **Catalyzer** | 10 |
| **Lines** | 25 |
| **Dry mass** | 3 286.166 |
| **Non usable propellant** | 200 |
| **ACS/RCS Propellant** | 142.12 |
| **Usable propellant** | 42 870.926 |
| **Total initial mass** | 46 969.092 |

This first initial mass is way over what we are targeting and there are many parameters to be refined during the next iteration.

## 4.5 Mass Budget - Second iteration

After refining multiple parameters and fixing others to get more accurate values, we went into the second iteration of our mass budget. Having our $I_{SP}$ changed also required another iteration in our calculation formula between the usable propellant mass the the rest of the mass.

### 4.5.1 Coefficients & Masses after steps

Considering that $ISP = 315s$ and annotating $\frac{m_{UP_i}}{m_{total_i}} = K_i$ with $i$ the burn number :

| Step | Required $\Delta v$ in $m/s$ | $K_i$ | Mass after step |
|---|---|---|---|
| 1 | 2802.4 | 0.596 | 0.404 $m_{initial}$ |
| 2 | 1342.2 | 0.352 | $0.261792m_{initial}$ |
| 3 | 522.9 | 0.156 | $0.221m_{initial}$ |
| Satellite caught | NA | NA | $0.221m_{initial}$ + 3500 |
| 4 | 1487.8 | 0.382 | $0.137m_{initial}$ + 2163 |
| Satellite release | NA | NA | $0.137m_{initial}$ -1337 |
| 5 | 5.3 | 0.0017 | $0.1368m_{initial}$ - 1334.73 |
| 6 | 72.4 | 0.023 | |

### 4.5.2 Global equation between $m_{UP}$ and $m_{initial}$

| Step | $\frac{m_{UP}}{m_{initial}}$ | Bias due to debris |
|---|---|---|
| 1 | 0.596 | 0 |
| 2 | 0.142 | 0 |
| 3 | 0.041 | 0 |
| 4 | 0.084 | 1337 |
| 5 | 0.0002 | -2.273 |
| 6 | 0.0032 | -30.699 |
| TOTAL | 0.8664 | +1304.028 |

This time our equation between those two masses is given by

$$m_{UsableProp} = \frac{1}{0.1336}\left[0.8664m_{rest} + 1304.028\right] \tag{4.6}$$

### 4.5.3 Second iteration of mass budget

As our way of presenting our first iteration of the mass budget didn't seems clear enough to us, we decided to present it in another, more logical way : **Structure**

| Contributor | Mass (kg) |
|---|---|
| Hull | 192 |
| Tanks (including non usable propellant) | 700 |
| Wings | 136 |
| Lines | 60 |
| Connectors | 16 |
| $H_2$ tank | 12 |
| **Structure** | 1 116 |

**Electrical related contributors**

| Contributor | Mass (kg) |
|---|---|
| Batteries | 241 |
| Fuel cells | 202 |
| On Board Computer | 5 |
| Cables | 20 |
| $H_2$ for fuel cells | 5 |
| Wing actuators | 10 |
| Telecommunications | 10 |
| GNC | 5 |
| Thermal Control | 10 |
| Magnets (Payload) | 25.65 |
| **Electrical related contributors** | 533.65 |

**ACS and RCS**

| Contributor | Mass (kg) |
|---|---|
| Thrusters | 36 |
| $H_2O_2$ | 90 |
| Reaction wheels | 106 |
| **ACS & RCS** | 232 |

**Propulsion**

| Contributor | Mass (kg) |
|---|---|
| Engine | 93 |
| Turbopumps | 25 |
| Pressurization ($He$) | 1.3 |
| Catalyzer | 30 |
| **Propulsion** | 149.3 |

With those tables, we can deduce $m_{rest}$ :

$$m_{rest} = m_{Structure} + m_{Elec} + m_{ACS\&RCS} + m_{Propulsion} \tag{4.7}$$
$$m_{rest} = 2\ 030.95kg \tag{4.8}$$

Thus,

$$m_{UsableProp} = \frac{1}{0.1336}\left[0.8664 m_{rest} + 1304.028\right] \tag{4.9}$$
$$m_{UsableProp} = 22\ 931kg \tag{4.10}$$
$$m_{Fuel} = \frac{m_{UsableProp}}{MR + 1} \tag{4.11}$$
$$m_{Fuel} = 2841.6kg \tag{4.12}$$
$$m_{Ox} = m_{UsableProp} - m_{Fuel} \tag{4.13}$$
$$m_{Ox} = 20\ 089.89kg \tag{4.14}$$
$$m_0 = 24\ 962.41kg \tag{4.15}$$

In this second iteration with a better $I_{sp}$ and refined values for all of the contributors, we have a large improvement as our initial mass decreased drastically.

## 4.6 Frozen information

After our second iteration of the mass budget, we decided to make a list of the fixed values that we will work around in our further design.

### 4.6.1 Frozen points

- We will do 20 aerobrakes
- We will have a separate tank design
- $H_2O_2$ will be pressurized by its decomposition
- The decomposition control will be managed by rotation of the spacecraft
- ACS/RCS Layout similar to the Space Shuttle
- $H_2O_2$ catalyzers separate
- $H_2O_2/O_2$ separation via thermodynamic properties
- $H_2/O_2$ will be used in fuel cells to produce energy

| Data | Value | Unit |
|---|---|---|
| Empty raw mass | 2 031 | kg |
| Usable propellant | 22 931 | kg |
| Total mass | 24 962 | kg |
| Flowrate | 10 | kg/s |
| Rocket diameter | 2 | $m$ |
| $I_{sp_{vacuum}}$ | 335 | s |
| Thrust $F = \dot{m} I_{sp} g_0$ | 32863.5 | N |
| Mixture Ratio | 7.07 | - |
| Wall thickness | $TBA$ | m |
| $H_2O_2$ internal pressure | 1.35 | bar |

## 4.7 Mass Budget - Final iteration

As the fixed $I_{sp}$ has been refined as well as other parameters, we went into our final iteration of the mass budget with the same process as the two previous ones.

### 4.7.1 Coefficients & Masses after steps

Considering that $ISP = 335s$ and annotating $\frac{m_{UP_i}}{m_{total_i}} = K_i$ with $i$ the burn number :

| Step | Required $\Delta v$ in $m/s$ | $K_i$ | Mass after step |
|---|---|---|---|
| 1 | 2802.4 | 0.574 | 0.426 $m_{initial}$ |
| 2 | 1342.2 | 0.335 | $0.283m_{initial}$ |
| 3 | 522.9 | 0.148 | $0.241m_{initial}$ |
| Satellite caught | NA | NA | $0.221m_{initial}$ + 3500 |
| 4 | 1487.8 | 0.364 | $0.153m_{initial}$ + 2226 |
| Satellite release | NA | NA | $0.153m_{initial}$ -1274 |
| 5 | 5.3 | 0.0016 | $0.1528m_{initial}$ - 1271.96 |
| 6 | 72.4 | 0.0218 | |

### 4.7.2 Global equation between $m_{UP}$ and $m_{initial}$

| Step | $\frac{m_{UP}}{m_{initial}}$ | Bias due to debris |
|---|---|---|
| 1 | 0.574 | 0 |
| 2 | 0.143 | 0 |
| 3 | 0.042 | 0 |
| 4 | 0.088 | 1274 |
| 5 | 0.0002 | -2.038 |
| 6 | 0.0033 | -27.73 |
| TOTAL | 0.8505 | +1244.232 |

Thus,

$$m_{UsablePropellant} = \frac{1}{0.1495}[0.8505 m_{rest} + 1244.232]$$

### 4.7.3 Detailed contributors

**Structure**

| Contributor | Mass (kg) |
|---|---|
| Hull | 192 |
| Tanks | 700 |
| Wings | 136 |
| Lines | 70 |
| Connectors | 16 |
| Brackets | 35 |
| $H_2$ Tanks | 12 |
| **Structure** | 1161 |

**Electrical systems**

| Contributor | Mass (kg) |
|---|---|
| Batteries | 369.03 |
| Fuel cell | 202 |
| OBC | 10 |
| Cables | 57 |
| $H_2$ for fuel cells | 5 |
| Wing actuators | 10 |
| Data transmission | 20 |
| GNC | 10 |
| Thermal control | 20 |
| Magnets | 25.65 |
| **Electrical systems** | 728.68 |

**Attitude control**

| Contributor | Mass (kg) |
|---|---|
| Thrusters | 36 |
| $H_2O_2$ for the ACS | 90 |
| Reaction wheels | 106 |
| **Attitude control** | 232 |

**Propulsion**

| Contributor | Mass (kg) |
|---|---|
| Engine | 93 |
| Turbopumps + Electric motors | 180 |
| $He$ for pressurization | 1.3 |
| Catalyzer | 40.86 |
| **Propulsion** | 315.16 |

### 4.7.4 Final mass budget

| Contributor | Mass (kg) |
|---|---|
| Structure | 1161 |
| Electrical systems | 728.68 |
| Attitude control | 232 |
| Heat shield | 360 |
| Propulsion | 315.16 |
| $m_{rest}$ | 2796.84 |
| $m_{UP}$ with a 5% performance window | 25 508 |
| $m_{Fuel}$ | 3160 |
| $m_{Ox}$ | 22 347 |
| **Initial wet mass** | 28 304 |

## 4.8 Simulation concept

# Chapter 5

# Design of propulsion systems

## 5.1 System conceptualization

## 5.2 Subsystem design

### 5.2.1 Regenerative cooling

### 5.2.2 Catalyzer

Besides the advantages of $H_2O_2$ there is one major drawback that we have take into account. This drawback is that $H_2O_2$ needs to be decomposed in $O_2$ and $H_2O$ in order to react with RP-1.

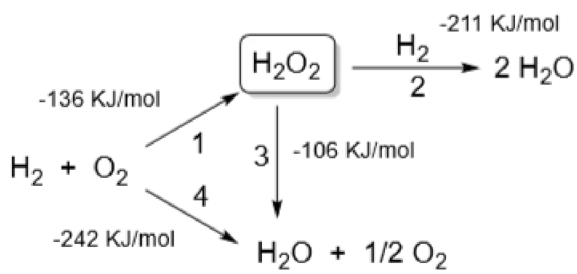



Figure 5.1: $H_2O_2$ chemical decomposition process

This decomposition is natural but at a low rate whereas we need a very high decomposition rate in order to feed the combustion chamber and sustain a proper flame. This decomposition is an exothermic decomposition. That's why we need a catalyser.

This catalyser needs to be placed between the turbo pump and the injectors. It allows to decompose the $H_2O_2$ at the last time.

The way a catalyser works is pretty simple; the $H_2O_2$ goes through a catalyst bed of silver pellets, reacts and generates heat. Why silver ? We chose silver because is the mostly used catalyser for $H_2O_2$. However, a lot of other different material exists, like Platinum, Manganese or even Gold but these materials are rarely used due to their cost and also the fact that they need to be made in complex alloy in order to optimize the reaction.

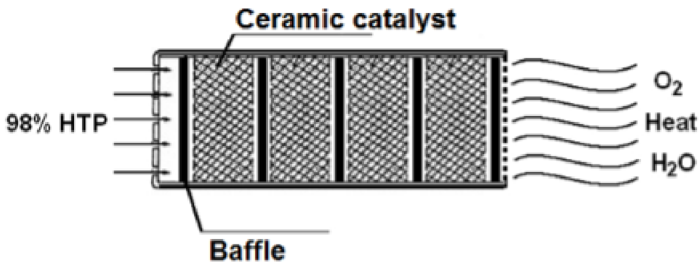



Figure 5.2: Example of catalyst bed

The figure above shows basically how a catalyser works. To have an idea of the shape of ours we just have to swap the ceramic catalyst by silver catalyst. So, it will be a steel cylinder filled with small spherical silver pellet separated by some baffles (silver grid mesh).

An important characteristic of the catalyser is the pressure drop it creates. This pressure drop influences the whole feeding system, the turbo pump sizing and even the injector design. that's why we need to characterize the pressure drop created by the catalyser. In order to do so, we are going to use the Ergun equation for packed bed reactor:

$$\frac{\Delta p}{L} = 151.2\frac{\mu}{d^2}\frac{(1-\epsilon)^2}{\epsilon^2}u + 1.8\frac{\rho}{d}\frac{1-\epsilon}{\epsilon^3}u^2$$

With $\mu$ the dynamic viscosity, $\epsilon$ the porosity, $d$ the pellet diameter, $\rho$ the density, $L$ the length of the bed and $u$ the velocity.

In this equation we need some important component such as $\epsilon$. The porosity is complicated to compute and need to be model. According to a recent research we determined a porosity of 0.3802 with a pellet diameter of 5mm.

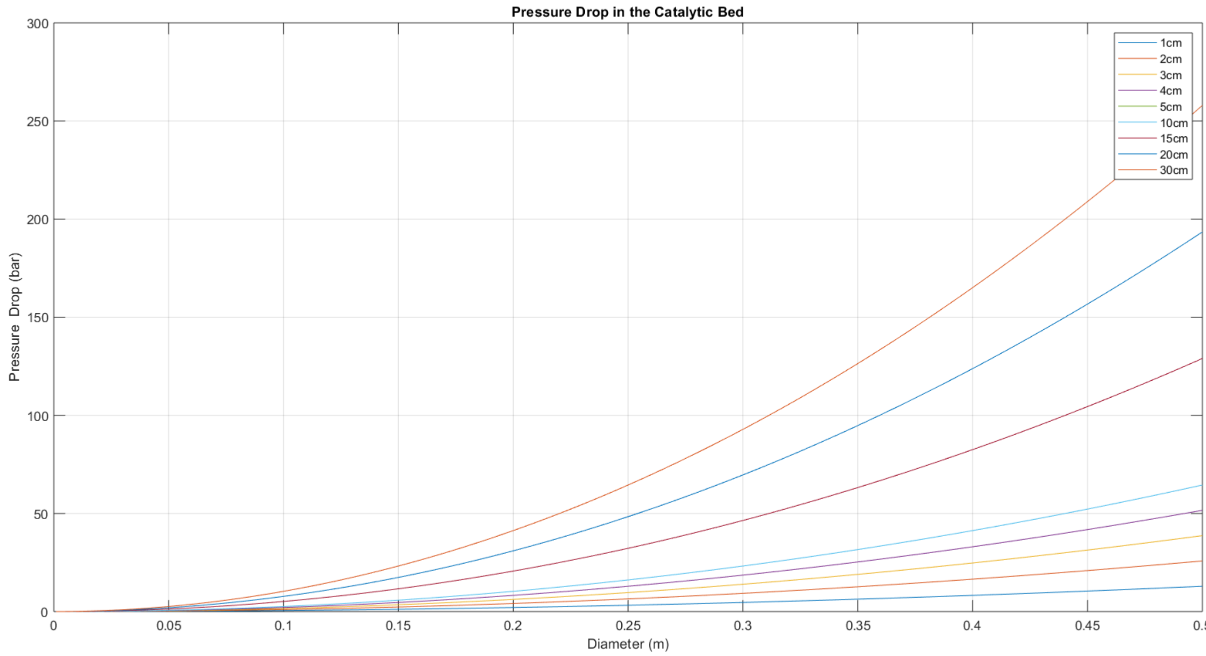



Figure 5.3: Pressure drop depending on the size of the catalyst bed

We see on the graph that the pressure drop rise quickly with the size of the bed. So we need to limit the size of our catalyser in order to not oversize the turbo pump. To do so, we chose to limit our pressure drop to around 30 bars. Finally we obtain a cylinder of 20 cm of diameter and 20 cm of length.

This geometry allows the catalyser to provide a sufficient decomposition rate, a "contained" pressure drop and size. It also generates a great heat as high as $1000K$ at the exit of the catalyst bed.

### 5.2.3 Injectors

### 5.2.4 Feeding system

After having designed most of our propulsion system. We need to carefully link them by designing our feeding system. The biggest challenge is to create a system that will both fit in our spacecraft and deliver the right amount of propellant from the tanks to the engine through our different, required other subsystems.

The general pressure loss in a system is given by :

$$\Delta P = K\frac{\rho}{2}w^2$$

With $K$ depending on the type of change in system geometry as follow :

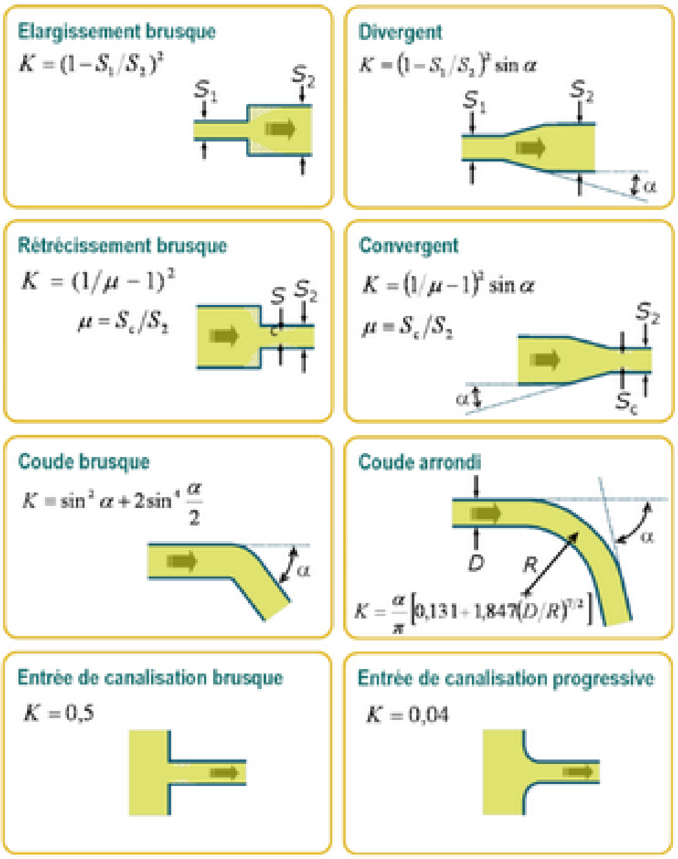



Figure 5.4: $K$ values for geometry changes

In our case, we will use the progressive line entering loss $K = 0.04$ and the bends $K(\alpha) = \sin^2(\alpha) + 2\sin^4\left(\frac{\alpha}{2}\right)$. Another $K$ will also be used for the entrance of the injector, which will be specified later on.

For manufacturing costs and simplicity purposes, we choose to only use 45° bends which will result in $K_{bends} = 0.5429$.

With that and the length measurements in mind, we designed the following feeding system layout for which we will then calculate the pressure variations along it :

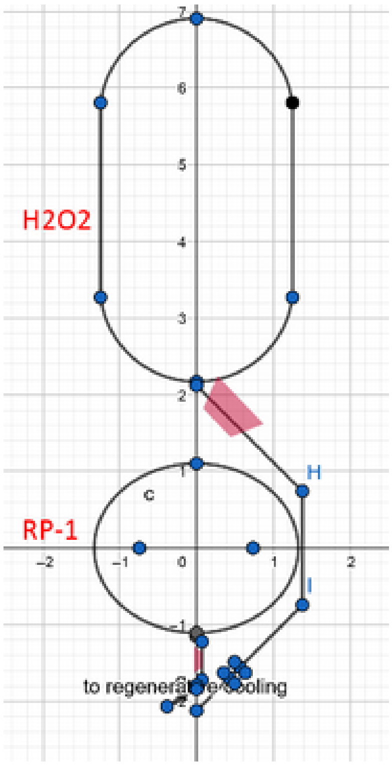



Figure 5.5: Feeding system layout (To scale)

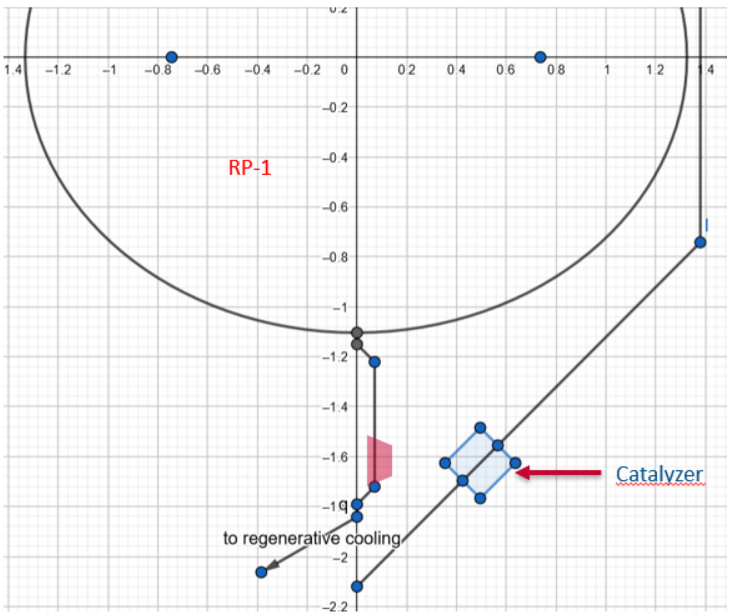



Figure 5.6: Feeding system layout - Zoomed (To scale)

### Line diameters

In order to choose our line diameters, we can first use our volume flow and then get the line area from it and then at the end, the line diameter.
Fuel

$$\dot{V}_f = \frac{\dot{m}_f}{\rho_f} = 0.0015298 m^3/s \tag{5.1}$$

$$A_{line_f} = \frac{\dot{V}_f}{w_f} = 5.21 \times 10^{-5} mm^2 \tag{5.2}$$

$$d_{line_f} = 2\sqrt{\frac{A_{line_f}}{\pi}} = 8.1447 mm \tag{5.3}$$

As we are trying to insure a high injection velocity and to insure a certain margin in pressure and velocity, we will choose a line diameter of 7 mm for the fuel feeding system.
Oxidizer

$$\dot{V}_o = \frac{\dot{m}_o}{\rho_o} = 0.006042 m^3/s \tag{5.4}$$

$$A_{line_o} = \frac{\dot{V}_o}{w_o} = 0.000275 mm^2 \tag{5.5}$$

$$d_{line_o} = 2\sqrt{\frac{A_{line_o}}{\pi}} = 18 mm \tag{5.6}$$

In this case, we will choose a line diameter of 15 mm for the oxidizer.

### Fuel feeding system

The following items in our fuel feeding system will cause pressure drops :

- Tank exit : $K = 0.04$
- $4 \times 45°$ bends : $K = 0.5429$ for each
- Straight line losses : $\Delta P = \frac{\rho}{2} w^2 f \frac{L}{D}$
- Friction coefficient : $f = 0.02$
- Regenerative cooling : $\Delta P = 0.25$ bar
- Fuel injection : $\Delta P = 9.3843$ bars

With our current layout, we have 5 straight lines which will cause pressure losses on the fuel side. Three of them are before the turbopump which is placed at the end of the third straight section, right before the third bend. We consider a velocity of 8 m/s before the turbopumps and of $v_{inj} = 29.363$ m/s after. The line loss for each section is given by :

$$\Delta P = \frac{810}{2} w^2 \times 0.02 \times \frac{L_{section}}{0.007}$$

1. First section ($L_{section} = 0.05m$) : $\Delta P = 0.037029$ bar
2. Second section ($L_{section} = 0.1m$) : $\Delta P = 0.074057$ bar
3. Third section ($L_{section} = 0.5m$) : $\Delta P = 0.37029$ bar
4. Fourth section ($L_{section} = 0.1m$) : $\Delta P = 0.997699$ bar
5. Fifth section ($L_{section} = 0.05m$) : $\Delta P = 0.49884$ bar

There are also two different values for the bend losses depending on the position of the bend (before or after the turbopump), we have 2 of each :

- $\Delta P_{before} = 0.14072$ bar
- $\Delta P_{after} = 1.8957$ bar

The tank exit loss is :

$$\Delta P_{exit} = K_{exit} \times \frac{\rho_F}{2} \times 8^2 = 0.010368 \text{ bar}$$

**Oxidizer feeding system**

The following items in our oxidizer feeding system will cause pressure drops :

- Tank exit : $K = 0.04$
- 4 × 45° bends : $K = 0.5429$ for each
- Straight line losses : $\Delta P = \frac{\rho}{2} w^2 f \frac{L}{D}$
- Friction coefficient : $f = 0.02$
- Catalyzer : $\Delta P =$ bars
- Oxidizer injection : $\Delta P = 4.9599$ bars

On this part of the feeding system, we also have 5 sections and 4 bends of 45° each. We also consider a velocity of 8 m/s before the turbopump and 21.971 m/s after. However, due to the larger distances (due to our tank layout), the turbopump's position in the feeding system is different and is now positioned after the first bend, right at the beginning of the second straight line. This results in 3 bends being at high velocity and 1 at relatively slower velocity.

Here, each straight line loss section is given by :

$$\Delta P = \frac{1450}{2} w^2 \times 0.02 \times \frac{L_{section}}{0.015}$$

With :

1. First section ($L_{section} = 0.05m$) : $\Delta P = 0.030933$ bar
2. Second section ($L_{section} = 1.95m$) : $\Delta P = 9.0992$ bars
3. Third section ($L_{section} = 1.4836m$) : $\Delta P = 6.9228$ bars
4. Fourth section ($L_{section} = 1.15m$) : $\Delta P = 5.3662$ bars
5. Fifth section ($L_{section} = 0.6m$) : $\Delta P = 2.7997$ bars

For the bends, we have :

- $\Delta P_{before} = 0.2519$ bar (1 of them)
- $\Delta P_{after} = 1.0614$ bar (3 of them)

The tank exit loss is :

$$\Delta P_{exit} = K_{exit} \times \frac{\rho_o}{2} \times 8^2 = 0.01856 \text{ bar}$$

### 5.2.5 Turbo pumps

As most of our subsystems have a defined pressure drop due to their specific design, we have made the choice to use this feeding system design with all losses included to then design our turbopumps to have a pressure rise in accordance with our pressure requirements. We chose to go with electrically driven turbo pumps as we have a good amount of electrical power since we use fuel cells in our spacecraft.

Our respective turbopump required created pressures are :

- Fuel side : $\Delta P_{T_f} = P_{Chamber} + \Delta P_{feeding_f} + \Delta P_{inj_f} + \Delta P_{Regenerative\ cooling} - P_{Tank_f}$
- Oxidizer side : $\Delta P_{T_o} = P_{Chamber} + \Delta P_{feeding_o} + \Delta P_{inj_o} + \Delta P_{Catalyzer} - P_{Tank_o}$

Thus,

$$\Delta P_{T_f} = 54.395 \text{ bars} \quad (5.7)$$
$$\Delta P_{T_o} = 102.28 \text{ bars} \quad (5.8)$$

### 5.2.6 Pressure evolution summary

#### Fuel side

| Contributor | Pressure Drop (bars) | Pressure at the end of this part (bars) |
|---|---|---|
| Tank | NA | 1.3 |
| Tank exit | 0.010368 | 1.29 |
| First section | 0.037 | 1.253 |
| First bend | 0.14 | 1.113 |
| Second section | 0.074 | 1.039 |
| Second bend | 0.14 | 0.899 |
| Third section | 0.37 | 0.529 |
| Turbo pump | 54.395 (Rise) | 54.924 |
| Third bend | 1.8957 | 53.0283 |
| Fourth section | 0.997 | 52.0313 |
| Fourth bend | 1.8957 | 50.1356 |
| Fifth section | 0.499 | 49.6366 |
| Cooling | 0.25 | 49.3866 |
| Injection | 9.38 | 40.0066 |
| Combustion chamber | NA | 40.0066 |

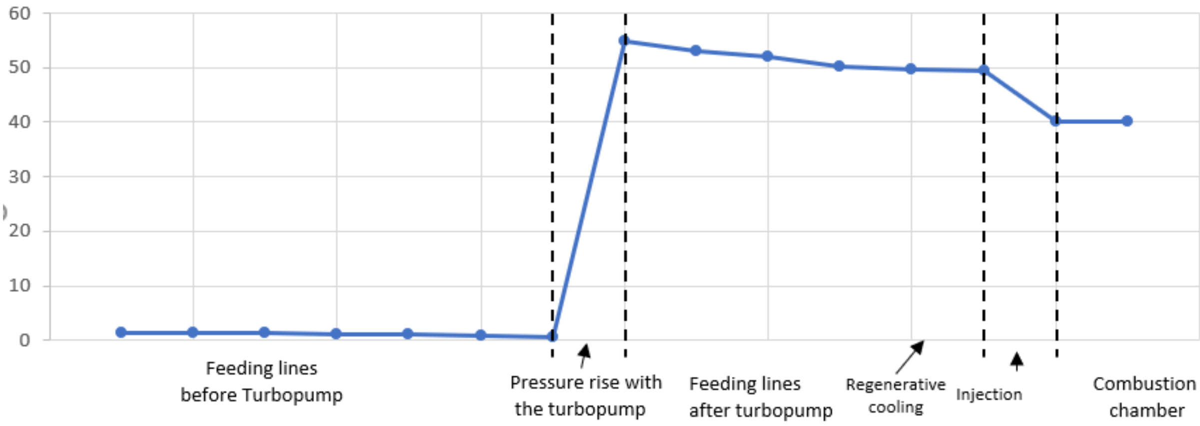



Figure 5.7: Pressure evolution on fuel side (bars)

**Oxidizer side**

| Contributor | Pressure Drop (bars) | Pressure at the end of this part (bars) |
|---|---|---|
| Tank | NA | 1.3 |
| Tank exit | 0.010368 | 1.29 |
| First section | 0.031 | 1.259 |
| First bend | 0.25 | 1.009 |
| Turbo pump | 102.28 (Rise) | 103.289 |
| Second section | 9.09 | 94.199 |
| Second bend | 1.06 | 93.139 |
| Third section | 6.9228 | 86.2162 |
| Third bend | 1.06 | 85.1562 |
| Fourth section | 5.366 | 79.7902 |
| Fourth bend | 1.06 | 78.7302 |
| Fifth section | 2.7997 | 75.9305 |
| Catalyzer | 30.95 | 44.9805 |
| Injection | 4.96 | 40.0205 |
| Combustion chamber | NA | 40.0205 |

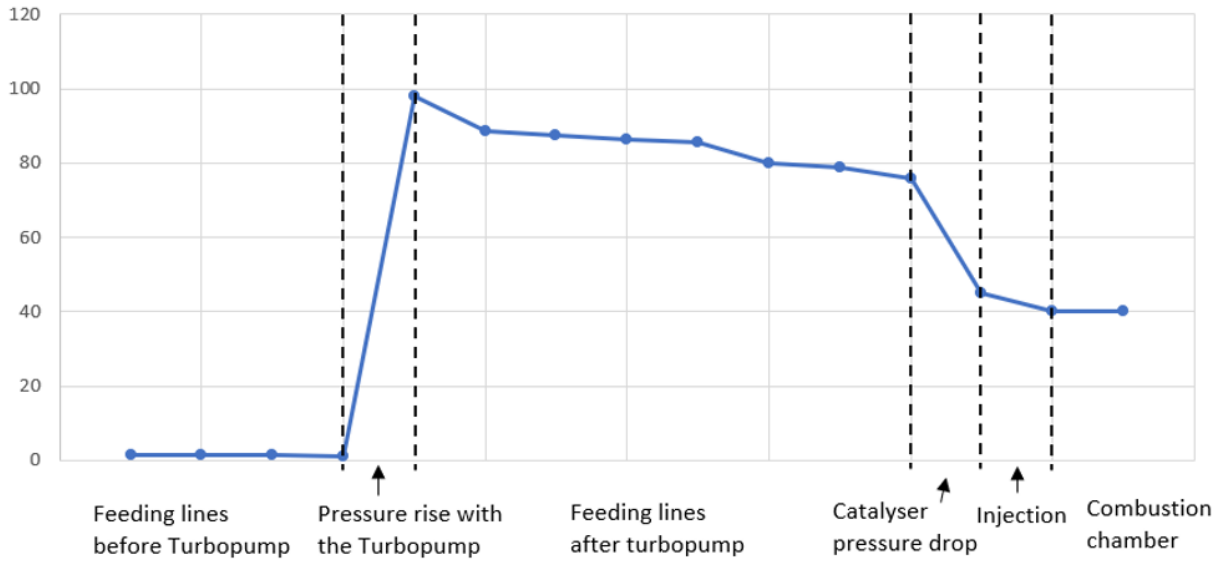



Figure 5.8: Pressure evolution on oxidizer side (bars)

## 5.3 Design review

# Chapter 6

# Simulation

## 6.1 Subsystem simulation

## 6.2 System simulation

## 6.3 Simulation preparation and execution

## 6.4 Simulation review

# Conclusion

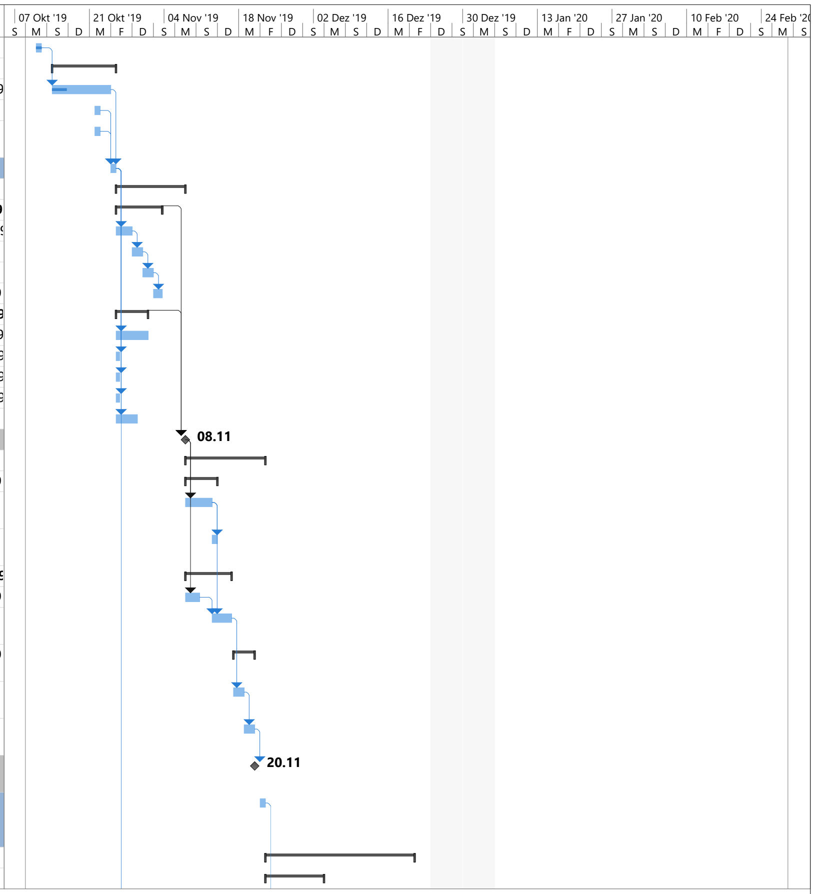
# List of Figures

# List of Tables

# Annexes

## Annex I - Gantt Diagramm

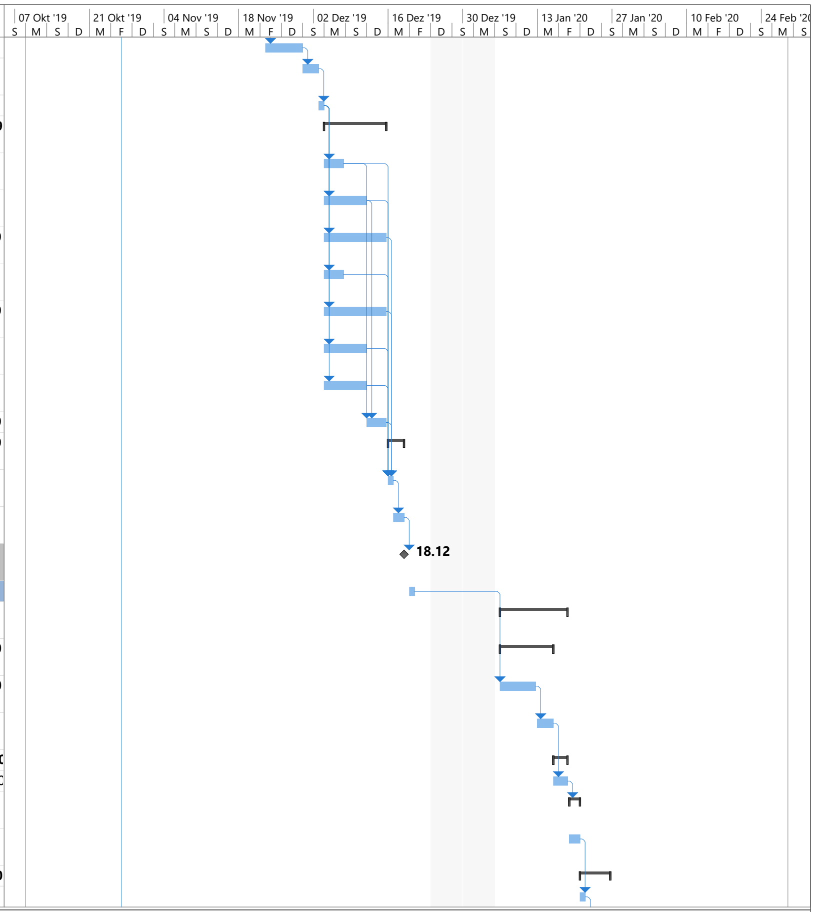
| Nr. | | Vorgangsı | Vorgangsname | Dauer | Anfang | Ende |
|---|---|---|---|---|---|---|
| 1 | | | Kick-off | 1 Tag | Fre 11.10.19 | Fre 11.10.19 |
| 2 | | | **Planning** | **12 Tage** | **Mon 14.10.1** | **Fre 25.10.19** |
| 3 | | | Schedule | 11 Tage | Mon 14.10.1 | Don 24.10.19 |
| 4 | | | Clarification of responsibilities | 1 Tag | Die 22.10.19 | Die 22.10.19 |
| 5 | | | Definition of group meeting slots for each week | 1 Tag | Die 22.10.19 | Die 22.10.19 |
| 6 | | | Presentation: project planning | 1 Tag | Fre 25.10.19 | Fre 25.10.19 |
| 7 | | | **Definition** | **13 Tage** | **Sam 26.10.1** | **Fre 08.11.19** |
| 8 | | | **Vehicle requirements** | **9 Tage** | **Sam 26.10.1** | **Son 03.11.19** |
| 9 | | | Interfaces to spacecraft structure | 3 Tage | Sam 26.10.19 | Mon 28.10.1 |
| 10 | | | Required Sub-Systems | 2 Tage | Die 29.10.19 | Mit 30.10.19 |
| 11 | | | Launcher constraints | 2 Tage | Don 31.10.19 | Fre 01.11.19 |
| 12 | | | Vehicle architecture | 2 Tage | Sam 02.11.19 | Son 03.11.19 |
| 13 | | | **Mission requirements** | **6 Tage** | **Sam 26.10.1** | **Don 31.10.19** |
| 14 | | | Delta-V analysis | 6 Tage | Sam 26.10.19 | Don 31.10.19 |
| 15 | | | Re-Ignitability | 1 Tag | Sam 26.10.19 | Sam 26.10.19 |
| 16 | | | Life Time Requirements | 1 Tag | Sam 26.10.19 | Sam 26.10.19 |
| 17 | | | Reliability | 1 Tag | Sam 26.10.19 | Sam 26.10.19 |
| 18 | | | Environment | 4 Tage | Sam 26.10.19 | Die 29.10.19 |
| 19 | | | Boundary requirements | 0 Tage | Fre 08.11.19 | Fre 08.11.19 |
| 20 | | | **Specification** | **15 Tage** | **Fre 08.11.19** | **Fre 22.11.19** |
| 21 | | | **Propellant selection** | **6 Tage** | **Fre 08.11.19** | **Mit 13.11.19** |
| 22 | | | Selection of appropriate propellant combination based on requirements | 5 Tage | Fre 08.11.19 | Die 12.11.19 |
| 23 | | | Analysis and optimization of mixture ratio | 1 Tag | Mit 13.11.19 | Mit 13.11.19 |
| 24 | | | **Mass analysis** | **9 Tage** | **Fre 08.11.19** | **Sam 16.11.1** |
| 25 | | | Sub-System mass contributors | 3 Tage | Fre 08.11.19 | Son 10.11.19 |
| 26 | | | Mass budget calculation | 4 Tage | Mit 13.11.19 | Sam 16.11.19 |
| 27 | | | **Simulation concept** | **4 Tage** | **Son 17.11.19** | **Mit 20.11.19** |
| 28 | | | Simulation goals | 2 Tage | Son 17.11.19 | Mon 18.11.19 |
| 29 | | | Accuracy / Simplicity Trade-Off | 2 Tage | Die 19.11.19 | Mit 20.11.19 |
| 30 | | | System specification | 0 Tage | Mit 20.11.19 | Mit 20.11.19 |
| 31 | | | Presentation: vehicle design concept, preliminary mass model and simulation concept | 1 Tag | Fre 22.11.19 | Fre 22.11.19 |
| 32 | | | **Design of propulsion system** | **28 Tage** | **Sam 23.11.1** | **Fre 20.12.19** |
| 33 | | | **System conceptualization** | **11 Tage** | **Sam 23.11.1** | **Die 03.12.19** |




Projekt: Schedule_DMSP_Grou
Datum: Fre 21.02.20

| | | | | |
|---|---|---|---|---|
| Vorgang | Projektsammelvorgang | Manueller Vorgang | Nur Anfang | Stichtag |
| Unterbrechung | Inaktiver Vorgang | Nur Dauer | Nur Ende | In Arbeit |
| Meilenstein | Inaktiver Meilenstein | Manueller Sammelrollup | Externe Vorgänge | Manueller Fortschritt |
| Sammelvorgang | Inaktiver Sammelvorgang | Manueller Sammelvorgang | Externer Meilenstein | |

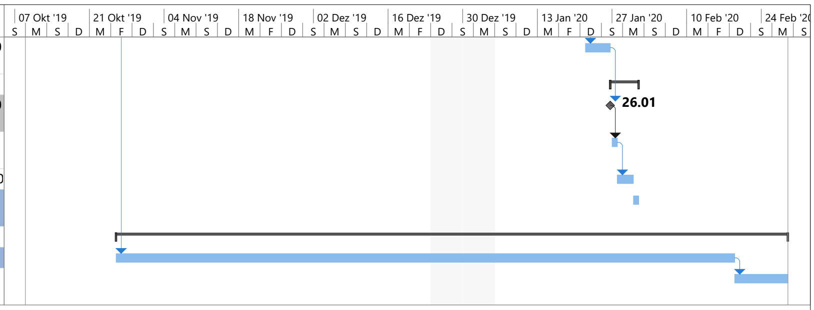
| Nr. | | Vorgangsmodus | Vorgangsname | Dauer | Anfang | Ende |
|---|---|---|---|---|---|---|
| 34 | | | Selection of appropriate cycle | 7 Tage | Sam 23.11.19 | Fre 29.11.19 |
| 35 | | | Requirement breakdown to sub-system level | 3 Tage | Sam 30.11.19 | Mon 02.12.19 |
| 36 | | | Distribution of sub-system resonsibilities | 1 Tag | Die 03.12.19 | Die 03.12.19 |
| 37 | | | **Sub-system design** | **12 Tage** | **Mit 04.12.19** | **Son 15.12.19** |
| 38 | | | Propellant tank dimensioning | 4 Tage | Mit 04.12.19 | Sam 07.12.19 |
| 39 | | | Feeding system design | 8 Tage | Mit 04.12.19 | Mit 11.12.19 |
| 40 | | | Injection system design | 12 Tage | Mit 04.12.19 | Son 15.12.19 |
| 41 | | | Ignition system design | 4 Tage | Mit 04.12.19 | Sam 07.12.19 |
| 42 | | | Thrust chamber design | 12 Tage | Mit 04.12.19 | Son 15.12.19 |
| 43 | | | Nozzle design | 8 Tage | Mit 04.12.19 | Mit 11.12.19 |
| 44 | | | Cooling system design | 8 Tage | Mit 04.12.19 | Mit 11.12.19 |
| 45 | | | Structure design | 4 Tage | Don 12.12.19 | Son 15.12.19 |
| 46 | | | **Design review** | **3 Tage** | **Mon 16.12.19** | **Mit 18.12.19** |
| 47 | | | Sub-system review of the propulsion system | 1 Tag | Mon 16.12.19 | Mon 16.12.19 |
| 48 | | | Change implementation | 2 Tage | Die 17.12.19 | Mit 18.12.19 |
| 49 | | | Propulsion system design review | 0 Tage | Mit 18.12.19 | Mit 18.12.19 |
| 50 | | | Presentation: propulsion system design | 1 Tag | Fre 20.12.19 | Fre 20.12.19 |
| 51 | | | **Simulation** | **13 Tage** | **Mon 06.01.20** | **Sam 18.01.20** |
| 52 | | | **Sub-System simulation** | **10 Tage** | **Mon 06.01.20** | **Mit 15.01.20** |
| 53 | | | Implementation of sub-systems into simulation environment | 7 Tage | Mon 06.01.20 | Son 12.01.20 |
| 54 | | | Validation of sub-system simulations | 3 Tage | Mon 13.01.20 | Mit 15.01.20 |
| 55 | | | **System simulation** | **3 Tage** | **Don 16.01.20** | **Sam 18.01.20** |
| 56 | | | Assembly of sub-systems | 3 Tage | Don 16.01.20 | Sam 18.01.20 |
| 57 | | | **Simulation preparation and execution** | **2 Tage** | **Son 19.01.20** | **Mon 20.01.20** |
| 58 | | | Validation of system simulation | 2 Tage | Son 19.01.20 | Mon 20.01.20 |
| 59 | | | **Simulation review** | **6 Tage** | **Die 21.01.20** | **Son 26.01.20** |
| 60 | | | Evaluation of simulation results | 1 Tag | Die 21.01.20 | Die 21.01.20 |




Projekt: Schedule_DMSP_Grou
Datum: Fre 21.02.20

| | | | |
|---|---|---|---|
| Vorgang | Projektsammelvorgang | Manueller Vorgang | Nur Anfang |
| Unterbrechung | Inaktiver Vorgang | Nur Dauer | Nur Ende |
| Meilenstein | Inaktiver Meilenstein | Manueller Sammelrollup | Externe Vorgänge |
| Sammelvorgang | Inaktiver Sammelvorgang | Manueller Sammelvorgang | Externer Meilenstein |
| Stichtag | In Arbeit | Manueller Fortschritt | |

| Nr. | | Vorgangs | Vorgangsname | Dauer | Anfang | Ende |
|---|---|---|---|---|---|---|
| 61 | | | Design optimization | 5 Tage | Mit 22.01.20 | Son 26.01.20 |
| 62 | | | **Close out** | **5 Tage** | **Son 26.01.20** | **Fre 31.01.20** |
| 63 | | | Final preliminary design of the propulsion system | 0 Tage | Son 26.01.20 | Son 26.01.20 |
| 64 | | | Lessons learned and way forward | 1 Tag | Mon 27.01.20 | Mon 27.01.20 |
| 65 | | | Buffer | 3 Tage | Die 28.01.20 | Don 30.01.20 |
| 66 | | | Presentation: integrated design and modeling | 1 Tag | Fre 31.01.20 | Fre 31.01.20 |
| 67 | | | **Reporting** | **110 Tage** | **Sam 26.10.1** | **Fre 28.02.20** |
| 68 | | | Reporting | 100 Tage | Sam 26.10.1 | Die 18.02.20 |
| 69 | | | Reporting buffer | 10 Tage | Mit 19.02.20 | Fre 28.02.20 |




Projekt: Schedule_DMSP_Grou
Datum: Fre 21.02.20

| | | | |
|---|---|---|---|
| Vorgang | Projektsammelvorgang | Manueller Vorgang | Nur Anfang | Stichtag |
| Unterbrechung | Inaktiver Vorgang | Nur Dauer | Nur Ende | In Arbeit |
| Meilenstein | Inaktiver Meilenstein | Manueller Sammelrollup | Externe Vorgänge | Manueller Fortschritt |
| Sammelvorgang | Inaktiver Sammelvorgang | Manueller Sammelvorgang | Externer Meilenstein | |