Design and Modeling of Space Propulsion Systems

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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

Definition

2.1 Vehicle requirements

- TL-10 The engine shall be the main propulsion system of a GEO satellite recovery vehicle.
- TL-11 The vehicle shall be refuelable between missions.
- TL-12 The vehicle shall be able to perform aerobreak maneauvers in earth's atmosphere.
- TL-13 The vehicle shall be able to manipulate it's flight path in earth's atmosphere using non-propulsive flight control systems.
- TL-14 The vehicle shall remain within the ARIANE 5 payload launch capabilities to LEO.
- TL-15 The vehicle shall be able to withstand debris impact of objects under 1cm of diameter with a maximum relative speed of 15km/s
- TL-16 The vehicle shall be able to remain on it's guided trajectory with less than 0.1 % deviation.

2.2 Operationnal requirements

- TL-1 The engine shall be able to provide sufficient thrust for completion of the mission profile including a safety margin.
- TL-2 The engine shall be re-ignitable at least 1000 times.
- TL-3 The engine shall have a service life time of at least 100 missions or 25 years in orbit.

- TL-4 The engine's ignition and functional reliability shall be higher than 99,5%.

2.2.1 Environment

- TL-5 The engine shall be able to withstand the launch phase.
- TL-6 The engine shall be able to operate in vacuum.
- TL-7 The engine shall be able to operate in an ambient temperature range of 1K to 5K.
- TL-8 The engine shall be able to withstand the temperature gradients resulting from areas turned towards or away from the sun.
- TL-9 The engine shall be able to sustain space-related radiation throughout it's complete life time.

Overall, our mission will take our spacecraft between the higher layers of the Earth's atmosphere to the geostationary orbit. Thus, we will often be in a void environment which has particular effects that we need to take into account.

Thermal effects

Among those effects is the thermal problem. In space, there are only two ways of exchanging heat :

- Conduction
- Radiation

This has an impact on our thermal control system as we need to keep our temperature stable with a low ambient temperature but a temperature close to Earth's surface inside.

Chapter 3

Specification

3.1 Satellite catching process

3.1.1 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.

3.1.2 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

3.1.3 Sketching and Calculations

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

$$F = \frac{(\mu N I)^2 A}{2\mu_0 L^2} \tag{3.1}$$

With :

- μ 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
- μ_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.

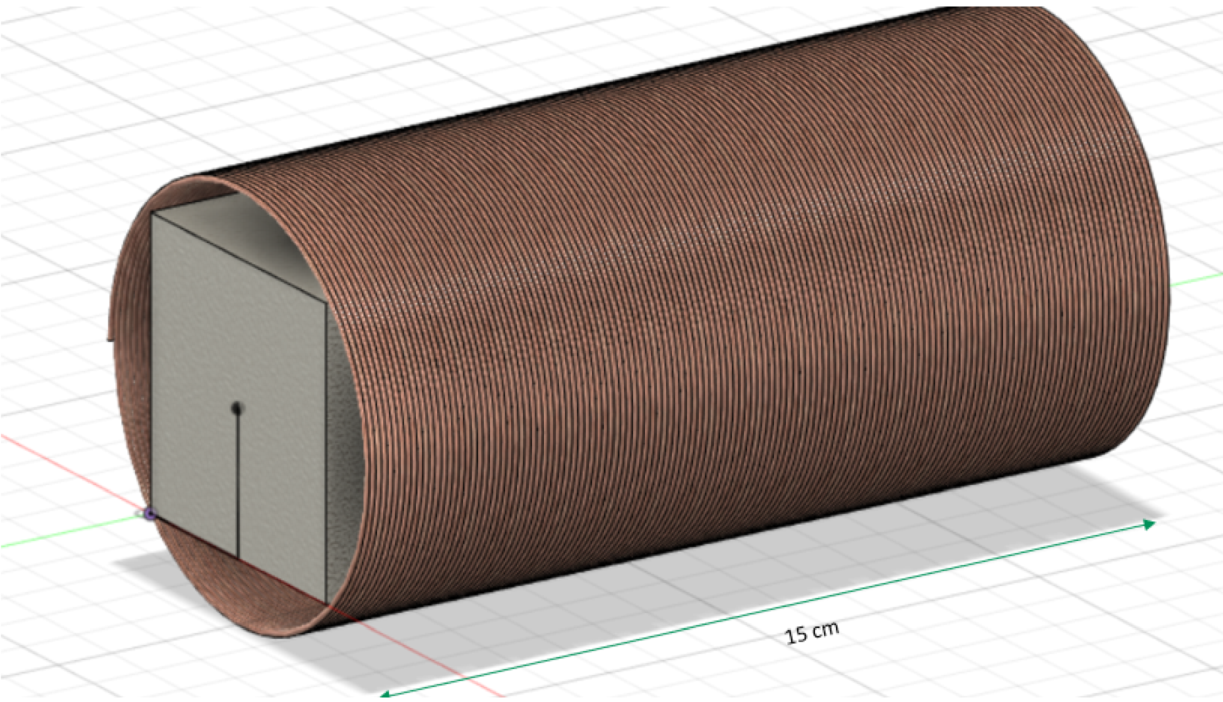

Figure 3.1: CAD of a catching magnet

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) :

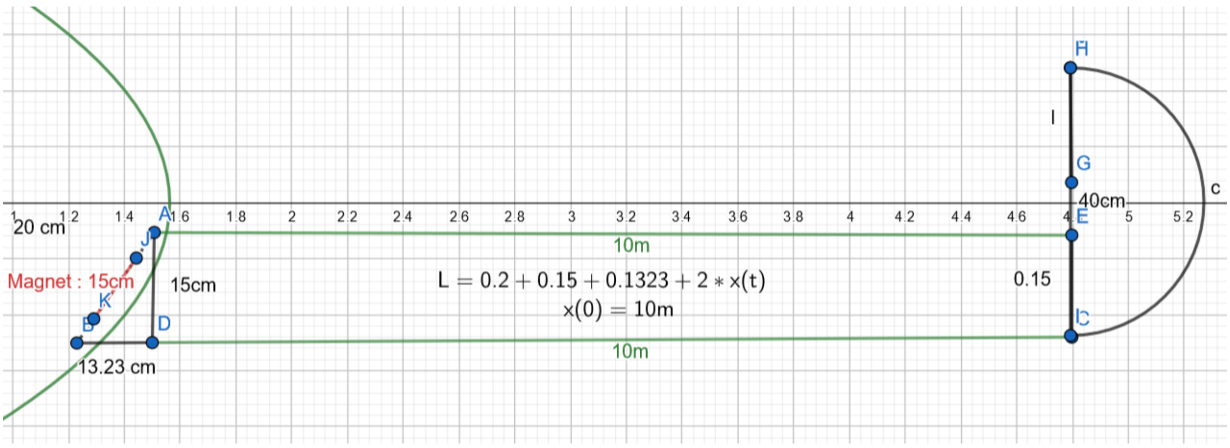

Figure 3.2: Catching sequence (not to scale)

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 $m = 0.001 \times m_{target} = 3.5\ kg$:

$$\vec{F} = m\vec{a} \tag{3.2}$$

$$\frac{(\mu NI)^2 A}{2\mu_0 L(x)^2} = m \times \ddot{x} \tag{3.3}$$

$$\frac{(\mu NI)^2 A}{2\mu_0[0.2 + 0.15 + 0.1323 + 2x(t)]^2} = m\ddot{x}(t) \tag{3.4}$$

$$\frac{(\mu NI)^2 A}{2\mu_0[0.4823 + 2x(t)]^2} = m\ddot{x}(t) \tag{3.5}$$

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.

3.2 Propellant selection

3.3 Mass analysis

3.4 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 Δv. 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.

3.4.1 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 Δv 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	

3.4.2 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.

3.4.3 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.

3.5 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.

3.5.1 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 Δv 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	

3.5.2 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

$$m_{UsableProp} = \frac{1}{0.1336}\left[0.8664m_{rest} + 1304.028\right] \tag{3.6}$$

3.5.3 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} :

$$m_{rest} = m_{Structure} + m_{Elec} + m_{ACS\&RCS} + m_{Propulsion} \tag{3.7}$$
$$m_{rest} = 2\ 030.95kg \tag{3.8}$$

Thus,

$$m_{UsableProp} = \frac{1}{0.1336}\left[0.8664m_{rest} + 1304.028\right] \tag{3.9}$$
$$m_{UsableProp} = 22\ 931kg \tag{3.10}$$
$$m_{Fuel} = \frac{m_{UsableProp}}{MR+1} \tag{3.11}$$
$$m_{Fuel} = 2841.6kg \tag{3.12}$$
$$m_{Ox} = m_{UsableProp} - m_{Fuel} \tag{3.13}$$
$$m_{Ox} = 20\ 089.89kg \tag{3.14}$$
$$m_0 = 24\ 962.41kg \tag{3.15}$$

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.

3.6 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.

3.6.1 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

3.7 Mass Budget - Third iteration

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

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

3.7.2 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	4.186
6	0.0033	-36.68
TOTAL	0.8664	+1304.028

3.8 Simulation concept

Chapter 4

Design of propulsion systems

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Chapter 5

Simulation

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