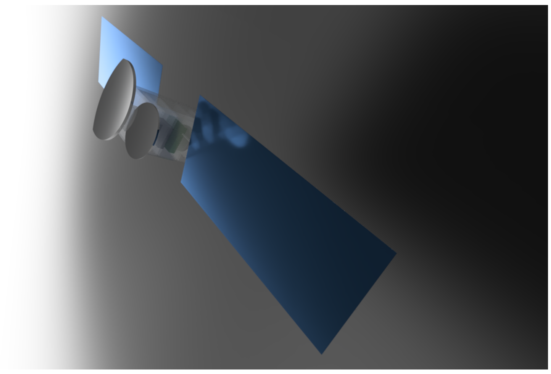
Spacecraft Design
Lunar INterconnection Keepers

ALBERIC DE LAJARTE



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1 Introduction

With the recent launch of the Artemis program, a new interest for Lunar mission has risen, being from private or governmental association. To support this program, many projects will be developed to help crewed mission settle on the lunar pole. One of them will be the lunar Gateway, a small space station placed in orbit around the Moon to interface between the lunar base and the arriving spacecraft. However, if the lunar Gateway will be enough to support the first mission based near the Moon's pole, it will be necessary to develop more complex network of satellite to ensure the same support for the whole surface of the Moon. In this project, we propose a first simple design of a satellite network able to relay information between all the Moon and the Earth by placing two additional satellites in a complementary orbit to that of the Gateway. The design of each satellite will be described in two separate documents. Here, we will discuss the design of a satellite placed in a halo orbit to guarantee an uninterrupted communication link with the Earth.

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2 Mission Overview

As this project aims to support the human activity around the Moon, it will be part of the Artemis program. However due to the international nature of this program, it will be developed and financed by the European Space agency as a medium class mission, with a budget bounded to 470 millions euros.

2.1 Mission definition

The goal of this mission is twofold: first by placing the spacecraft in an orbit near the Moon, it will be able to relay data from other missions or satellite to the Earth. By working with other spacecrafts, a complete network can be created that ensure proper communication between the Earth and the whole Moon. In this network, our spacecraft is responsible of ensuring a permanent link between the far side of the Moon and the Earth, which could be useful either to relay data from various satellite of a constellation, or simply to transmit direct communication from a lunar base to Earth. The second part of this mission is to demonstrate the ability of a spacecraft to use only electric propulsion to reach a Moon orbit orbit from a medium earth orbit using only its solar panel, effectively reducing the overall cost of the mission. The use of electric engine will greatly increase the duration of the mission, as the transfer from a medium-Earth orbit to a Moon orbit can take several month. However it will also allow to extend the lifetime of the Moon-orbit by making the necessary maneuvers corrections with very little propellant.

The summary of the top level requirements are summarized in the table:

ID	Description
`L1_MS_01`	The spacecraft shall use only electric propulsion to reach its final orbit from a MEO
`L1_MS_02`	The spacecraft shall maintain a stable orbit near the Moon for at least [5] year
`L1_MS_03`	The spacecraft shall always maintain radio communication with the Earth
`L1_MS_03`	The spacecraft shall always maintain radio communication with the far side of the Moon

These requirements will be the drivers requirements of this project. However, we can also use references to other similar missions in order to help us on the different design choices. Two particular mission are interesting for us:

SMART-1 is an ESA mission launched in 2003. Its goal was to demonstrate the ability of ion engines to reach a NEO at low cost (110 millions euros). It successfully reached the Moon one year after its launch, then continued its scientific mission during two years before being deorbited and crashed on the Moon surface in 2006.

Queqiao is a chinese satellite orbiting around the L2 Langrangian point of the Moon in order to relay radio communication of the lunar Chang'e 4 rover. It started its relay mission at the end of 2018, when Chang'e 4 landed on the far side of the Moon.

2.2 Mission design

2.2.1 Initial Orbit

The choice of initial orbit is of major importance, as to reduce the number of maneuvers, it is preferable to have orbital elements closer to those of the final orbit. In our case, we will target the orbital elements of the Moon, which for a specific date will constrain the inclination and RAAN of the initial orbit. The ideal next time would be in March 2034 when the inclination of the Moon is the lowest, but as this date is too far in the future we will simply launch when the spacecraft is finished, assuming around five years of development, which lead us around September 2024.

Then another major constraint for our mission is the Van Allen belt that will damage our electronic and solar panels during the long transfer time of the mission. To avoid that, a first solution would be to launch at a high inclination, as the limit of the Van Allen belt is around 60°. In the end this idea was abandoned because the cost of the plane change was too expensive. Another solution is to target an initial altitude above the first Van Allen belt as it is the most energetic of the two. This solution also has the advantage of reducing eclipse time and travel duration, and was thus chosen.

The last constraint is the eclipse period, which due to our low inclination of 5.14 °with respect to the ecliptic is inevitable. We can however choose to have the eclipse at a time when less power is needed. As our strategy will be to thrust near the perigee, it was found using GMAT that a argument of perigee of 230 °would ensure that no eclipse would happen near the perigee for the whole transfer time. All the orbital elements are summarized in Table 1.

Semi major axis	21371 km
Eccentricity	0
Inclination	28.7 °
RAAN	0 °
Argument of perigee	230 °
True anomaly	270 °
Period	8:38 h

Table 1: Orbital elements of the initial orbit

2.2.2 Transfer

In this section, we will describe the process to go from a circular medium earth orbit to the desired L2 halo orbit.

As we want to demonstrate the performance of electric propulsion systems, the entirety of the transfer will be performed with the selected electric engine. This means that we cannot use simple Hohmann transfer that rely on short, impulsive burn, to get to the Moon. Instead we can use low energy transfer that are more adapted to low-thrust engine. These type of transfer use the fact that stable manifold exist between the Earth and the Moon that can be used to guide any spacecraft from an Earth orbit to a Moon orbit, which allow to greatly reduce the delta-V required, at the cost of longer transfer time.

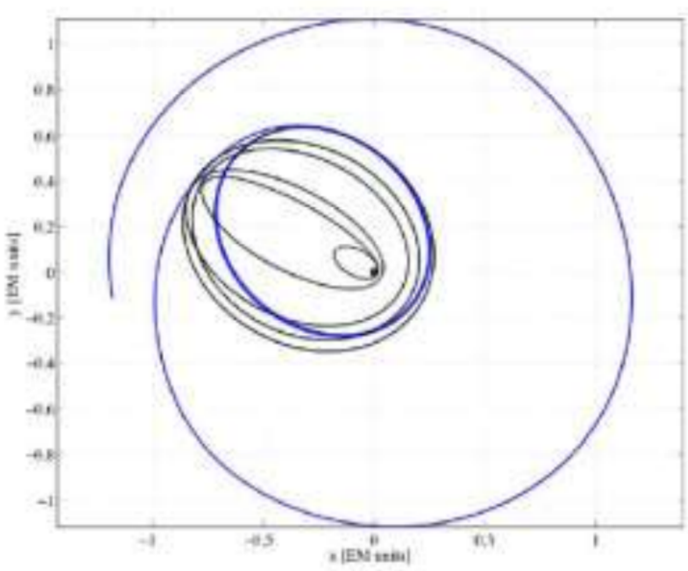

Figure 1: Interior low energy transfer trajectory. The blue line correspond to the Cruising phase and the black line to the Manifold insertion. (3)

The complete trajectory is detailed as follow:

- **MEO insertion**: Our spacecraft is launched with a Soyuz rocket from Earth and inserted into a 15000km of altitude, circular orbit.
- **Manifold insertion**: By thrusting only during the periapsis ($\theta \in [-90°; 90°]$), the apoapsis is slowly increased to around 347'000 km of altitude. At this point the spacecraft enter the stable manifold region.
- **Cruising**: The spacecraft is guided on the manifold trajectory up to the L2 Lagragian point. This phase necessitate very little propellant, only to correct the small perturbations.
- **Halo orbit insertion**: Once the spacecraft has reached the L2 point, a last maneuver is necessary to insert the spacecraft into an Halo orbit with an amplitude of 3500km. This last maneuver also uses very little propellant
- **Station keeping**: As the Halo orbit are not stable, it will be necessary to do regular correction burn every 7.32 days (1) of about 2 m/s each.

- **End of life**: the instability of the orbit means that a simple delta-V of less than 50 m/s is enough to crash the satellite into the Moon after its 5 years of service

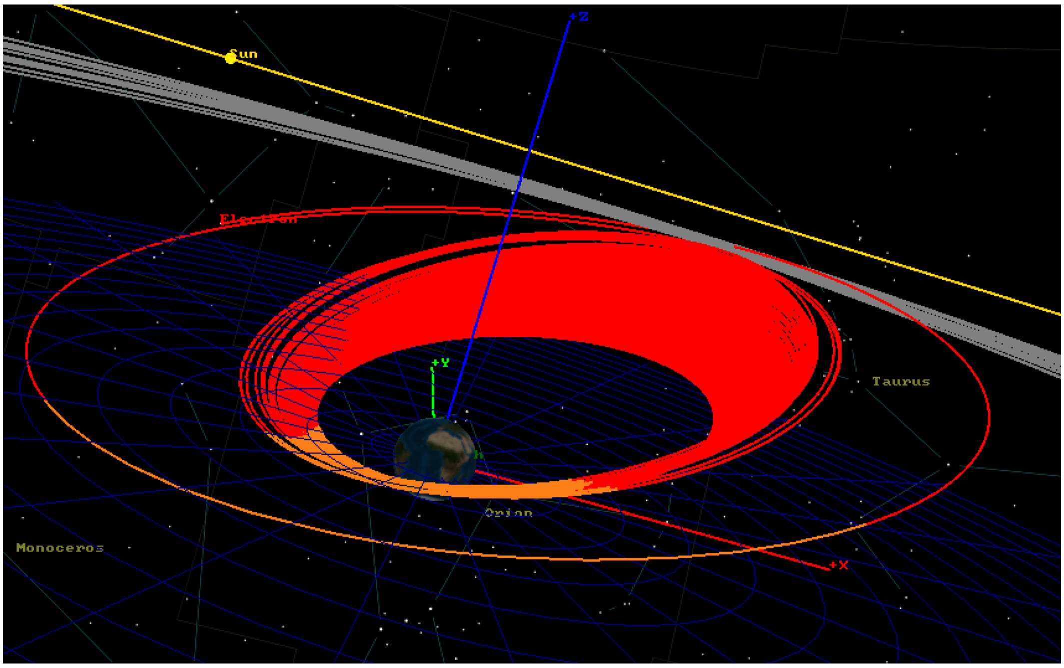

Figure 2: GMAT simulation of the manifold insertion phase. In red is the coast phase, and in orange the thrust phase. The path of the Sun is in yellow and the path of the Moon in grey

Phase	Duration	Delta-V [m/s]
Manifold insertion	4 years	1207
Cruising	60 days	50
Station keeping	5 years	518
End of life	7 days	125
Total	1525 days (+5 years)	1900

Table 2: Different phases of the MEO to Moon transfer. The first line was estimated using GMAT simulation, and the last one come from two articles (3) (1). Note that the MEO insertion and Halo orbit insertion phases are not present in this table because their cost and duration are negligible as a first approximation

This delta-V budget allow us to estimate the mass of propellant needed:

$$m_p = m_f * (\exp\frac{\Delta v}{v_e} - 1) = 2303 * (\exp\frac{1825}{41104} - 1) = 104kg$$

During the Manifold insertion phase, the ion thrusters will be powered on and consume a lot of energy. To reduce the size of the solar panels, the idea is thus to only thrust near the perigee with the power from the battery and the solar panels combined, then to cut of the engine during a coast phase where the battery can be charged again. For this strategy to work, it is important to avoid the eclipse phase near the peripasis, so the argument of perigee was set to 230 °to have the eclipse phase at the limit of the thrust phase, and the manifold insertion duration was limited to:

$$t = \frac{365}{2} - \frac{365}{360} * \arctan\frac{R_e}{SMA} * 2 = 148.8days$$

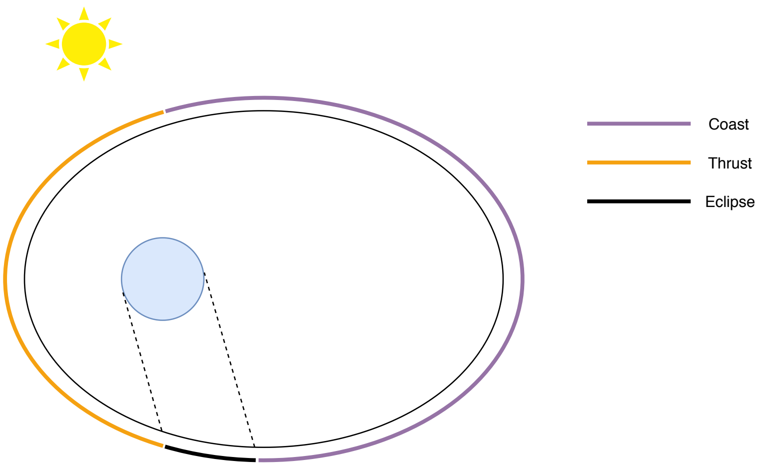

Figure 3: ConOps of the Manifold insertion phase. At this time of the year, the Eclipse phase is at the limit of the Thrust phase

However, due to the large mass of the spacecraft, this requirements is far for being respected, as this phase last approximately 4 years based on GMAT simulation. To mitigate that, the spacecraft may have to reduce the ion thrusters power during the thrust phase, which would lengthen the mission even more. This scenario has not been implemented in GMAT.

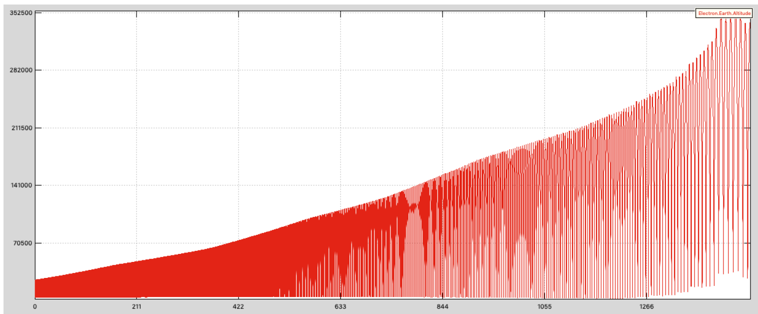

Figure 4: Altitude of the spacecraft during the Manifold insertion phase

2.2.3 Final orbit

The final orbit is an Halo orbit around the Moon-Earth L2 Lagragian point. Its orbital period is around 14.6 days and the radius of the orbit was chosen to be 3500 km as it is the minimal radius that allow the spacecraft to never be hidden by the Moon. Choosing a small circular orbit reduces the cost of the station keeping maneuvers (1), and the beamwidth of the lunar based antenna. Indeed, if we take into account the lunar libration of $\pm$ $3.87°$, with a spacecraft orbiting at 64500 km of the Moon in a circular orbit of 3500 km in radius, we can use a fixed lunar antenna with a minimal half-beamwidth of:

$$\theta = \arctan(\frac{3500}{64500}) * \frac{180}{\pi} + \frac{3.87}{2} \approx 5°$$

to guarantee that our spacecraft will always be in view of the lunar station without needing for antenna tracking. Once the spacecraft is on a stable halo orbit it will start its relay mission, which is twofold:

- **Far side relay**: Because the rotation of the Moon around its axis is synchronized with its rotation around the Earth, our spacecraft will always be visible from the Far side of the Moon, which will allow a lunar base based here to have a constant communication link with the Earth.

- **Lunar poles relay**: To help the pole communicate with the Earth and the rest of the Moon, our spacecraft will work with another satellite orbiting the Moon. The idea is that the Moon satellite will gather data from the poles and send them to the Earth either directly when orbiting above the near side, or via our L2 spacecraft when orbiting above the far side

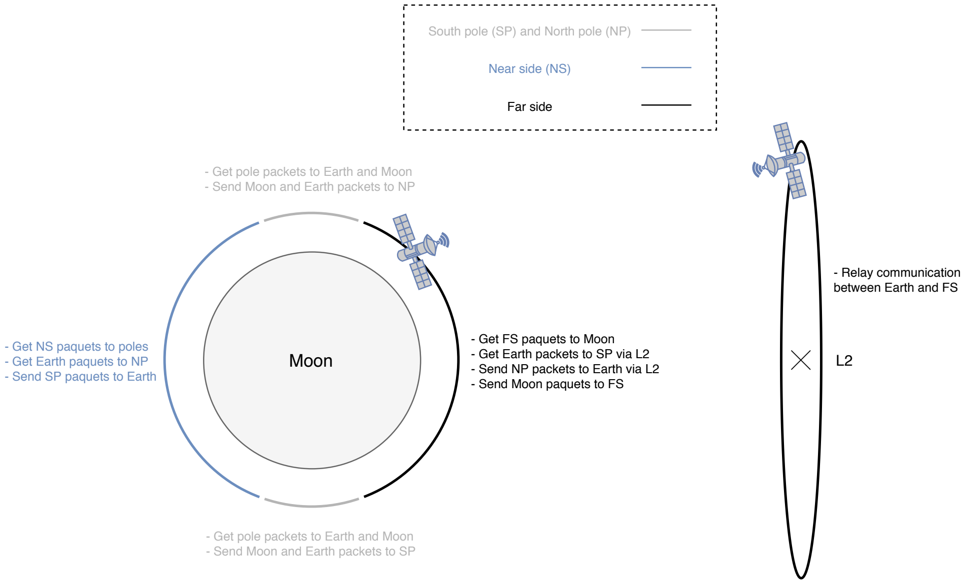

Figure 5: ConOps of the final orbit. In addition to the two missions described above, this network also allow lunar bases from the near side and far side to communicate with the poles

Due to the maximum eclipse duration of 55.3h, the spacecraft will have to enter a low power mode some part of the year, when the line of nodes of the Moon's orbit is close to being colinear with the Eart-Sun direction. When in low power mode, the spacecraft will have to stop its relay mission.

3 System overview

In this section we will go more in depth about the different systems needed to achieve this mission.

3.1 System engineering

3.1.1 System description

The first part of the project is to define more formally the project. Starting with the functional analysis, we can list the different functions that our spacecraft will need to accomplish its mission.

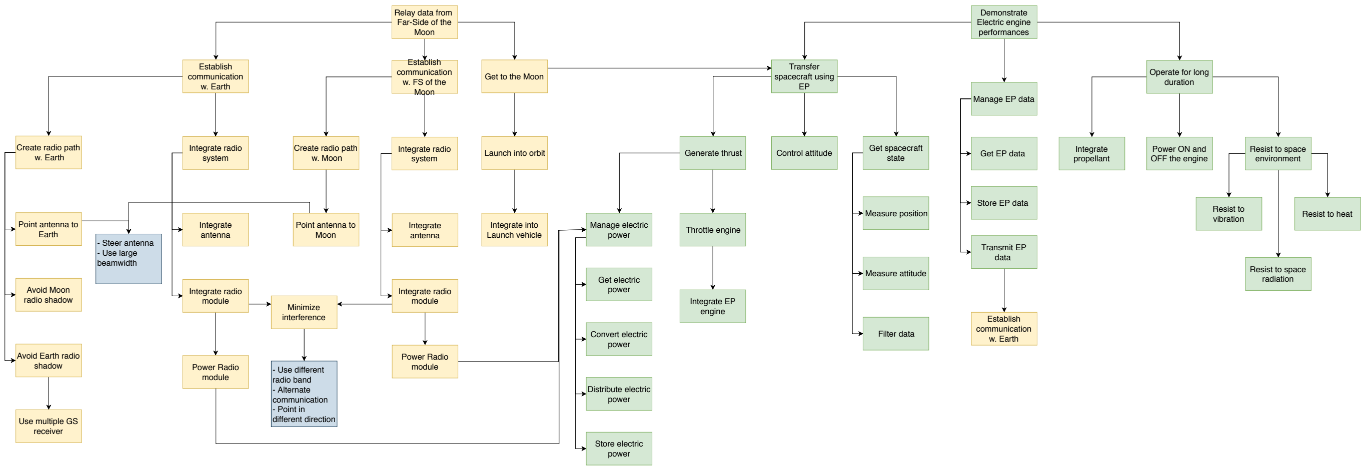

Figure 6: Functional diagram of the project. The communication part of the mission is in yellow while the propulsion is in green

We see here that the mission is clearly divided into two parts. In yellow we find all the functions that describes how the satellite should relay the information from the far side of the Moon to the Earth. One of the top level function is actually to get to the Moon, which is related to the second part of the diagram, describing the testing process of our electric engine. Because one of the main focus of the mission is to test our propulsion, no additional science equipment will be used except those serving the overall mission. (e.g attitude and position sensors).

From this diagram, we can define the different systems and subsystems needed to implement each functionality. This process is summarized in figure 7.

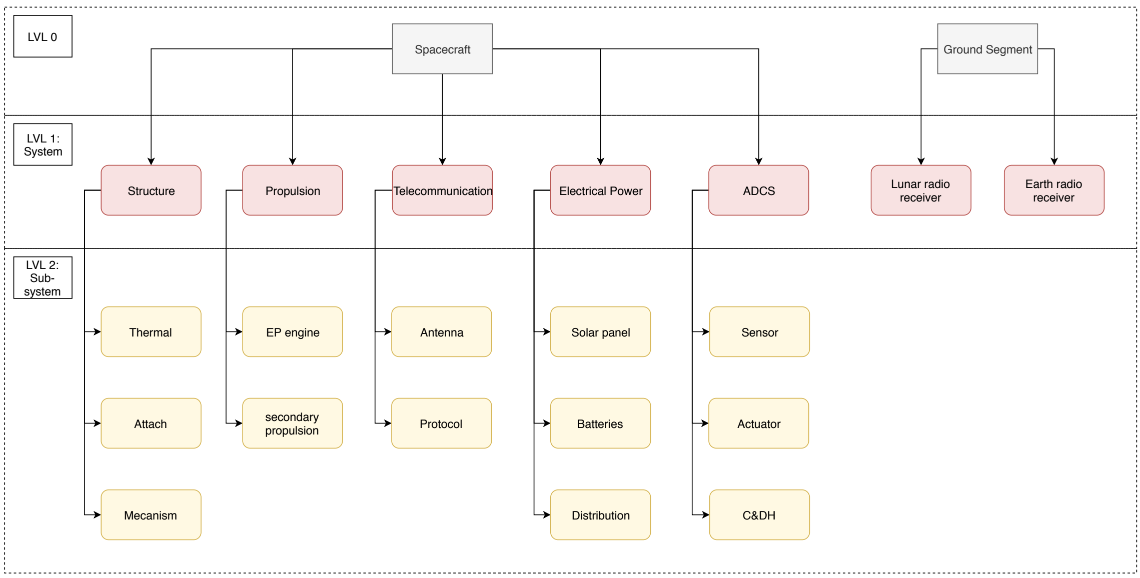

Figure 7: System decomposition

Each of these systems will be discussed more in depth in a dedicated sections. For the ground segments systems, they will be directly described in the Telecommunication part, as the goal of this project is not to design the ground segment.

3.1.2 Timeline

As it was mentioned above, the development phase of the project will last five years, as no additional equipment will have to be designed for the mission. Indeed, most of the work here will be to choose and interface the different components necessary for each subsystem.

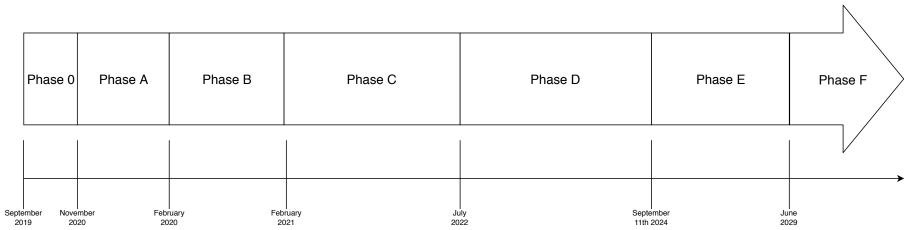

Figure 8: Complete planning of the mission. The longest phase is phase E, during which our spacecraft act as communication relay for lunar operations

3.1.3 Budget

To help design the different systems, it is important to keep track of various budgets.

We can find the power budget in Figure 9. This budget will be useful for the design of the electrical power system, as it give us a first estimation of the size of the battery and solar panel.

Mission phase		MEO insertion	Manifold insertion			Cruising			Station keeping			
sub-phase			Coast	Thrust	Eclipse	Coast	Thrust	Eclipse	Coast	Thrust	Eclipse	Lunar Eclipse
Spacecraft mode		low power	Charge	Full power	low power	Charge	Full power	low power	Relay	Full power	low power	low power
Power OUT [W]		**294,48**	**971,28**	**21534,48**	**294,48**	**971,28**	**21534,48**	**294,48**	**9611,28**	**21534,48**	**280,08**	**4600,08**
Structure	Spacecraft cooling	100	100	200	100	100	200	100	100	200	100	100
	Solar panel orientation	7	7	7	7	7	7	7	7	7	7	7
	Total (20% cont.)	128,4	128,4	248,4	128,4	128,4	248,4	128,4	128,4	248,4	128,4	128,4
Propulsion	EP engines	0	0	13800	0	0	13800	0	0	13800	0	0
	Gimbals	19,50	19,50	19,50	19,50	19,50	19,50	19,50	19,50	19,50	19,50	19,50
	Total (20% cont.)	23,4	23,4	16583,4	23,4	23,4	16583,4	23,4	23,4	16583,4	23,4	23,4
Telecommunication	Primary system	0	0	0	0	0	0	0	6000	0	0	3000
	Secondary system	18	18	18	18	18	18	18	18	18	18	18
	Total (20% cont.)	21,6	21,6	21,6	21,6	21,6	21,6	21,6	7221,6	21,6	21,6	3621,6
Electrical Power	Distribution	10	480	860	10	480	860	10	480	860		
	Total (20% cont.)	12	576	1032	12	576	1032	12	576	1032	0	0
ADCS	Sensors	20	20	20	20	20	20	20	20	20	20	20
	CMG	25	25	25	25	25	25	25	25	25	25	25
	C&DH	5	5	5	5	5	5	5	5	5	5	5
	Total (20% cont.)	60	60	60	60	60	60	60	60	60	60	60
Power IN [W]		0	13519,1	13519,1	0,0	13519,1	13519,1	0,0	13519,1	13519,1	0,0	0,0
Electrical Power	Solar Panel	0	13519,1	13519,1	0,0	13519,1	13519,1	0,0	13519,1	13519,1	0,0	0,0
Total Power [W]		**-294,48**	**12547,8**	**-8015,4**	**-294,5**	**12547,8**	**-8015,4**	**-294,5**	**3907,8**	**-8015,4**	**-280,1**	**-4600,1**
Duration [h]		3,83	3,5	4,3	0,9	233,6	1,0	2,4	350,4	0,0	55,3	6,0
Battery level [%]		97,39	100,0	20,2	19,6	100,0	81,5	79,9	100,0	100,0	64,2	36,2
Margin [Wh]		**42162,14**		**8493,4**			**34567,9**			**27800,9**		**15689,5**

Figure 9: Power budget of the different systems and subsystems

Then the link budget is presented in figure 10. This budget was used for the design of the antenna and the choice of ground stations.

Link		L2/Earth		L2/Moon	
	Emitter	**L2**	**Earth**	**L2**	**Moon**
Emitter	Tx Power	27,0	39	27,0	27,0
	Tx circuit loss	-1,0	0	-1,0	-1
	Tx antenna gain	36,4	89	40,6	3,03E+01
	Tx pointing loss	-3,0	0	-1,0	-1
	Total	**59,4**	**128,0**	**65,6**	**55,3**
Path	Space loss	-222,6	-222,6	-207,1	-207,1
	Athmospheric loss	-1,0	-1,0	0,0	0,0
	Polarization loss	-1,0	-1,0	-1,0	-1,0
	Total	**-224,6**	**-224,6**	**-208,1**	**-208,1**
Receiver	Rx pointing loss	0,0	-3	-1,0	-1
	Rx antenna gain	50,8	3,64E+01	40,6	40,6
	Rx circuit loss	0,0	-1	-1,0	-1
	Total	**50,8**	**32,4**	**38,6**	**38,6**
SNR	Boltzmann	228,6	228,6	228,6	228,6
	Thermal noise	-24,6	-24,6	-26,0	-26,0
	Data rate	-69,0	-69,0	-80,0	-80,0
	E/N	-4,4	-4,4	-4,4	-4,4
	total	**130,5**	**130,5**	**118,2**	**118,2**
	margin	**16,1**	**66,4**	**14,2**	**3,9**

Figure 10: Link budget of the primary wireless communication between the Earth or Moon and our spacecraft. All the units are in dB.

Link		L2/Earth	
	Emitter	**L2**	**Earth**
Emitter	Tx Power	5,0	38
	Tx circuit loss	-1,0	0
	Tx antenna gain	10,0	79
	Tx pointing loss	0,0	0
	Total	**14,0**	**117,0**
Path	Space loss	-211,0	-211,0
	Athmospheric loss	-0,5	-1,0
	Polarization loss	-1,0	-1,0
	Total	**-212,5**	**-213,0**
Receiver	Rx pointing loss	0,0	-3
	Rx antenna gain	37,5	10
	Rx circuit loss	0,0	-1
	Total	**37,5**	**6,0**
SNR	Boltzmann	228,6	228,6
	Thermal noise	-24,6	-24,6
	Data rate	-35,5	-35,5
	E/N	-4,4	-4,4
	total	**164,1**	**164,1**
	margin	**3,1**	**74,1**

Figure 11: Link budget of the secondary wireless communication between the Earth and our spacecraft. All the units are in dB.

		Mass	Quantity	Total with cont.
Structure	Primary structure	197,4	1	256,6
	Secondary structure	94,6	1	123,0
	Steering mechanism	3,6	2	9,4
	Deployment mechanism	1,5	2	3,9
	Total (20% cont.)			**471,5**
Propulsion	EP engines	56,8	2	147,6
	Gimbals	6,0	2	15,6
	Tank	18,1	1	23,6
	Propellant	132,0	1	171,6
	Total (20% cont.)			**430,0**
Telecommunication	X band transceiver	58,0	2	150,8
	S band transceiver	1,3	1	1,6
	Earth antenna	83,1	1	108,0
	Moon antenna	27,1	1	35,3
	RF cables	20,0	1	26,0
	Total (20% cont.)			**386,0**
Electrical Power	Solar panel	35,3	2	91,8
	Battery	535,8	1	696,5
	PCU	51,0	1	66,3
	Cables	30,0	1	39,0
	Total (20% cont.)			**1072,3**
ADCS	Sensors	30,0	1	39,0
	Data storage	10,0	1	13,0
	Cables	20,0	1	26,0
	Total (20% cont.)			**93,6**
	Total mass Spacecraft [kg]			2453,5
	Adapter [kg]			135
	Max mass at 15000 km [kg]			3400
	Margin [kg]			**811,5**

Figure 12: Mass budget of the spacecraft and the final margin from the Soyuz performance

Structure	Thermal probe	4	16	5	0,31
	Solar panel position sensor	2	32	10	0,63
	Total with 20% contingency				**1,13**
Propulsion	Gimbal position sensor	4	32	10	1,25
	Power control	6	16	50	4,69
	Pressure sensor	6	16	50	4,69
	Temperature sensor	7	16	50	5,47
	Flowmeter	4	16	50	3,13
	Total with 20% contingency				**23,06**
Telecommunication	RSSI	3	16	5	0,23
	Data rate	3	16	5	0,23
	Total with 20% contingency				**0,56**
Electrical Power	Battery level	23	16	5	1,80
	Battery health	23	8	5	0,90
	Power meter	7	16	5	0,55
	Solar panel health	2	16	1	0,03
	Total with 20% contingency				**3,93**
ADCS	Sun sensor	2	32	50	3,13
	Star tracker	2	32	50	3,13
	Magnetometer	2	16	50	1,56
	GPS	2	112	5	1,09
	Accelerometer	6	16	100	9,38
	Gyroscope	6	16	100	9,38
	Control moment gyroscope feedback	4	16	10	0,63
	Total with 20% contingency				**24,56**
Total with 30% contingency					**69,21**

Figure 13: Data budget of the spacecraft

3.2 Telecommunication

To reduce the power consumption of the spacecraft, the telecommunication system is composed of two sub-systems: the first one will be used once in orbit to relay data between the Moon and the Earth, while the other one will be only used to transmit internal data of our spacecraft during the whole mission.

3.2.1 Primary system

To ensure a high data rate between the Earth and the Moon, each element had to be designed thoughtfully. The first important decision was the choice of antenna: because the spacecraft is far from the Moon and the Earth, we can use high gain parabolic antenna with a small beamwidth. With the help of some simple geometry, we find that from our orbit, the field of view of the Moon and Earth combined is 5.6°, which is bigger than the sum of their field of view (3.1°and 1.9°respectively) because these two field of view don't intersect. It is thus far more interesting to use two separate antennas for the Moon and the Earth, as the gain of a parabolic antenna is:

$$G = 10 * \log(\frac{27000}{\theta^2})$$

with θ the half-beamwidth of the antenna, set to half the field of view of the Earth or Moon from our spacecraft. However, as the diameter of a parabolic antenna is:

$$D = \frac{70 * \lambda}{\theta}$$

we find that the diameter of these two antennas would be 1.6m and 2.6m for X band radio, which when combined is more than the diameter of the Soyuz fairing. One solution would be to use higher carrier frequency to reduce the size of the antenna, but due to a lack of transceivers and ground stations operating in the Ka and Ku radio band, and the additional path loss that it would generate, it was decided to find a solution to integrate these antennas anyway: we can either store the antennas on the side and deploy them when the relay mission start, or use parabolic antennas with part of the edge cutoff. This solution doesn't change the property of beamwidth and frequency response, but has an impact on their gain, which will have to be accounted for on the link budget. In the end it was decided to use this solution in order to avoid the risk of a failure of the deployment mechanism which would be mission critical.

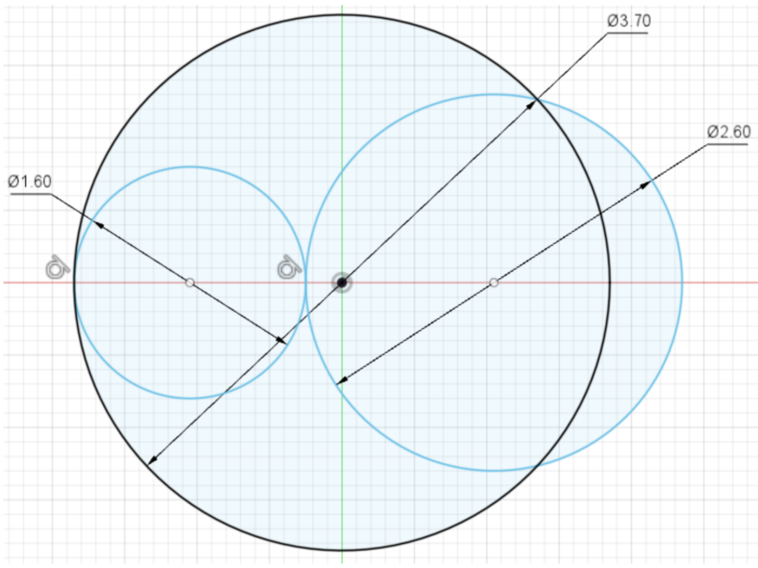

Figure 14: 2D view of the two antennas with the maximum size of the fairing. Only the larger antenna is cutoff, which make it gain go from 44.7dB to 36.4dB

Then to power the antenna, two X-band transceivers are used from Advantech wireless (11) at a frequency of 8.4GHz for the transmission and 7.25GHz for the reception. As they don't integrate a diplexer, we will need to add one before each antenna in order to separate the transmission and reception frequencies. With a power input of 3000W and a output gain of 27dB, they can only be used once placed in halo orbit, when the ion thrusters are powered off. Using two separate antennas and transceiver also allow to deliver the maximum data rate, as the spacecraft doesn't have to share the resources between the link with the Earth and the link with the Moon.

Name	Localization	EIRP [dB]	X band G/T [dB]	Data rate [Mbps]
Cerebros-1	Spain	128	50.8	8
Kiruna 1	Sweden	N/A	36.9	100
Kiruna 2	Sweden	N/A	35.6	100
Kourou 1	French Guyana	112.8	41	2
Maspalomas 1	Maspalomas (Spain)	112.8	37.5	40
New Norcia 1	Australia	127	50.1	2
Perth 1	Australia	112.8	37.5	2
Santa Maria 1	Acores(Portugal)	N/A	30	1
Santiago	Chile	N/A	40	2
Malindi 1	Kenya	N/A	31.8	40
Svalbard 3	Svalbard (Norway)	N/A	32	300

Table 3: List of the ground station from the ESTRACK network that uses X band. (10)

Then to receive the signal, the ESTRACK European radio network will be used, as most of its ground station operates in the X band. The different parameters of the ground station are presented in table 3. We see that there is a potential receiver in almost all continent, which guarantee us a constant link between the Earth and the Moon. For the data rate, the goal is to reach 100 Mbps, however, as only 3 ground station can support this data rate and that they will be used also by other missions, we cannot guarantee it constantly.

To be compatible with this network, we will use BPSK with QPSK plus modulation, and a R-1/2 Viterbi decoding, which guarantee us a maximum data rate of $D = 0.5 * 7.25 = 3.63Gbps$, at the cost of $\frac{E_b}{N_0} = 4.4dB$ for a BER of 10^{-5}. Finally, for the lunar based station, we assume that a similar transceiver is used with a fixed parabola antenna of 5 °3dB beamwidth, which ensure a constant link with our L2 spacecraft and a gain of 30.3 dB. With all these parameters, we can have a precise estimataion of the link budget, presented in figure 10. For the noise temperature, we simply assume a mean temperature of 290K for the Earth and a maximum temperature of 400K for the Moon, as we expect each of the antenna to be precisely pointed at its target.

3.2.2 Secondary system

The goal of this sub-system is to ensure a constant link between the Earth and the spacecraft in order for the Earth to monitor the status of the spacecraft. Compared to the primary system, its requirements are too have a much lower consumption which allow to be used in low power mode, an omidirectional radiation pattern which guarantee a radio link regardless of the orientation of the spacecraft, and a smaller data rate to transmit only the internal state of the spacecraft.

From the data budget, we estimate to 70,87 kbps the data rate needed by the spacecraft, however as this budget takes into account the maximum frequency of the sensors and their redundancy, we can choose to send only 5% of these data, so 3.54 kbps. This lower data rate allow us to use the S band which is less sensitive to atmospheric perturbation. We can thus use the STC-MS03 transceiver (12), which transmit at 2.2 GHz and receive at 2.025 GHz (with integrated diplexer) with a power consumption of 18W. This transceiver will be connected to two monopole antennas situated to opposite faces of the spacecraft via a power divider. This configuration ensure a permanent connection to the ground station as each antenna cover a full hemisphere. To receive the signal on Earth, we will also use the ESTRACK network as it is compatible with the S band. However its lower sensitivity in the S band region constrain the maximum data rate (see figure 11), which could be unsufficient to characterize the state of the spacecraft. To mitigate that, we can send the full 70.87 kbps via the primary system and consider the secondary system as a backup when the spacecraft enter low poer mode during eclipse or if a problem occurs.

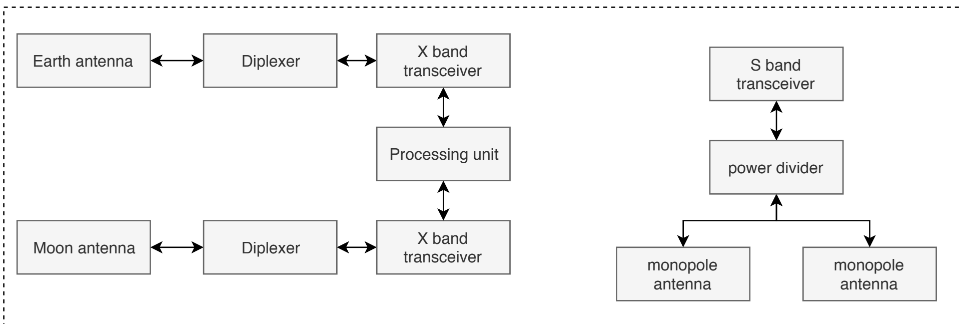

Figure 15: Hardware diagram of the telecommunication system

3.3 Propulsion

3.3.1 Electric propulsion engine

There is a lot of different type of electric propulsion technologies, but the two most important one are:

- **Hall thruster**: These engines use the combination of an electric and a magnetic field to accelerate ions. They allow for the highest thrust among the different types of electric propulsion

- **Ion thruster**: These engines generate a high voltage between two grids in order to accelerate the ions. They have a lower thrust than Hall effect thruster, but have a higher efficiency and specific impulse.

We compare the performances of a few engines in Table 4

Engine	Type	Isp[s]	Thrust[mN]	Power[W]	Efficiency[%]
NSTAR	Ion	3100	92	2300	60
NEXT	Ion	4190	236	6900	70
PPS-1350	Hall	1660	90	1500	55
SPT-100D	Hall	2200	112	2500	50

Table 4: Comparison of different EP engines

As the goal of this mission is to assess the performances of an EP engine, it was decided to use the NEXT engine, which is described as the successor of the NSTAR engine, used during the DAWN mission. The NEXT engine has never been used outside test benches, so it is a great opportunity to assess its performances in a real space environment.

The main parts of this engine are:

- **Thruster**: This is the chamber where the xenons atoms are ionised and accelerated before exiting the spacecraft

- **Propellant management system (PMS)**: is used to control the flow of Xenon to the thruster. It is divided into a High Pressure Assembly which manage the Xenon tank and a Low Pressure Assembly (LPA) which receive the flow of Xenon from the HPA and divide it into the main, cathode and neutralizer flow.

- **Power Processing Unit (PPU)**: it is responsible to deliver the high power and voltage required to drive the engine. From an unregulated 80 to 160 V it can output 275 to 1800 V to the main channel. It also has five other channel to power other functions of the engine

- **Digital Control Interface Unit (DCIU)**: this is the electronics controlling the engine. It translate the high level commands from the spacecraft into a sequence of commands suitable for the engine

- **Gimbal**: this mechanism allow to change the orientation of the thrust to accommodate for the displacement of the center of mass and for special maneuvers. It has two axes with 17°range each

- **Tank**: This is the storage for the Xenon propellant. It can withstand 9MPa and carry up to 425 kg

Due to the high mass of the spacecraft, it was chosen to use two NEXT engine, each placed at the back of the spacecraft. However, as the tank has enough capacity for the mission, only one will be used for the two ion thrusters.

Part	Mass [kg]	Size [cm]	Volume (dm^3)	Power [W]
Thruster	13.5	⊘ 58x44	116.2	6860
PMS	5	33x15x6.4 + 38x30.5x6.4	10.6	83
PPU	34.5	41.9x52.1x14	30.5	28
DCIU	3.75	18x24x8	3.5	100
Gimbal	6	62.2x71.8x46.7	47.7	21.5
Tank	18.14	-	267.9	-
Total	80.89	62.2x71.8x46.7 + tank	476.4	7092.5

Table 5: Physical characteristics of the NEXT engine. As the gimbal also act as the structure for the thruster, PMS, PPU and DCIU, the total volume is only the sum of the gimbal and tank volume

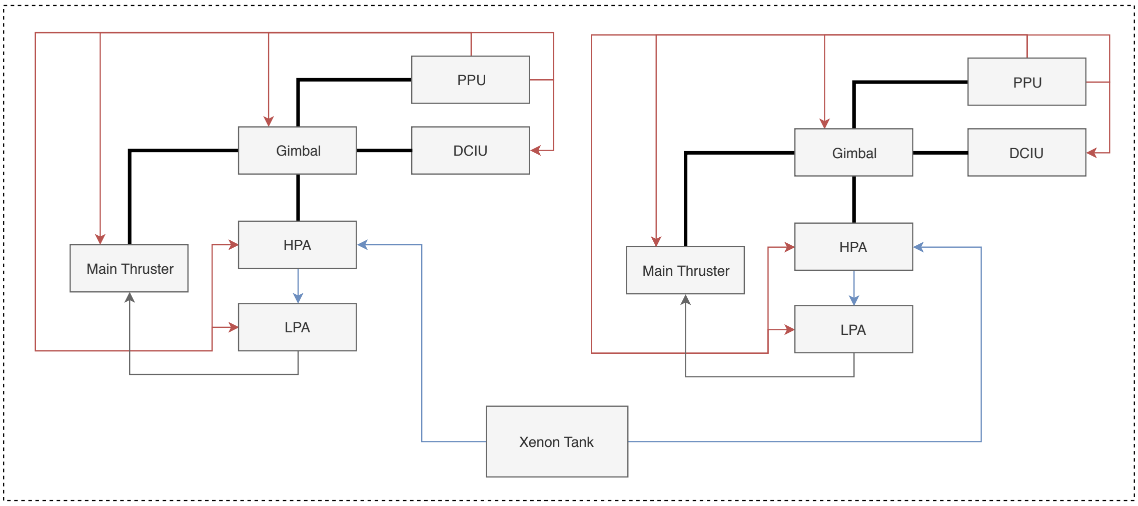

Figure 16: Harware diagram of the propulsion system, with the main engine and the attitude control engines

3.3.2 Attitude control

To control the orientation of the spacecraft, the ion engines will be used in conjunction with control Momentum gyroscope, as using two engines with each a two degree of freedom gimbals allow to control the three axes of rotation. However for pure rotation, it is preferable to use only the CMG, described more in depth in the ADCS section.

3.4 Electrical power

3.4.1 Solar Panel

To accommodate for the large power of the NEXT engine, we will need large solar panels. As it was explained in the transfer part of the Mission overview, the idea is to use two ion engines, one powered by the battery and the other directly from the solar panel. From the power budget, we find that power needed from the solar panel is 13'519W. If we take triple junction GaInP2/GaAs/Ge solar panels from Spectrolab (15), this means that the total surface of the solar panel should be 36.9 m^2. From the spacecraft CAD, we can estimate that the maximum width of the solar panels should be 2.9m in order to fit inside the 3.7m diameter fairing of Soyuz. Thus the length of each solar panel will be:

$$L = \frac{S}{2*w} = \frac{36.9}{2*2.9} = 6.4m$$

With these dimension, the spacecraft can use both ion engine during the Manifold insertion phase, and then the high power transceiver during the relay phase. As the most demanding phase is the transfer which is at the beginning of the mission, we can accept some degradation of the solar panel without needing for larger solar panel, as we can see in the power budget.

3.4.2 Batteries

The role of the batteries is to store the additional energy gathered by the solar panel during daylight and use it to power the systems during eclipse and support the solar panel during the thrust phase. From the power budget, we can estimate the size of the battery to have 20% margin during the Manifold insertion phase to be 43'290Wh, which correspond to

$$n = \frac{43'290}{3.7*75*0.8} = 195cells$$

from the LVP65 serie (7) with a depth of discharge of 0.8.

Battery type	Energy [Wh]	Mass [kg]	Volume [Wh/l]	Power output [kW]
LVP65	240.5	2.75	1.18	1.2
Battery pack	54112	535.7	233.2	234

Table 6: Specification of the LVP65 battery for one cell and for the complete battery pack

Due to the long duration of the eclipses during the relay phase (6h and 55.3h for lunar and standard eclipse respectively), this large battery pack will allow us to still run in a lower power mode without the need of the solar panels.

3.4.3 Distribution

In this last part, we will discuss the choice of a power conditioning and distribution system. The role of this component is to convert and distribute power between the solar array, the batteries and the other systems. Here, the main requirement is the power capacity of the PCU, which has to handle the 13.5kW of power generated by the solar panel during daylight while charging the battery pack.

A possible candidate would be the PCU Next generation from Thales (8), which allow up to 21.6 kW of power. It is also specifically made to handle Li-ion charge and discharge from solar panel, for a total mass of 51 kg.

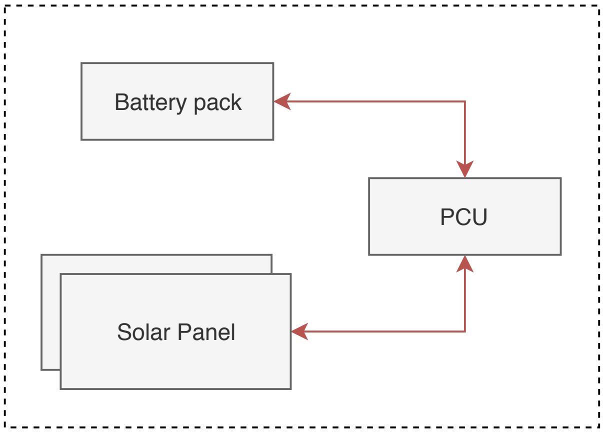

Figure 17: Hardware diagram of the electrical power system

3.5 ADCS

3.5.1 Sensor

In order to accurately determine the position and attitude of the spacecraft at all time, several sensor will be used:

- **Accelerometer:** is used to measure the acceleration in the three direction of the spacecraft. When integrated, it gives the position in three dimension.
- **GPS:** gives an absolute measure of the spacecraft's position to correct the integration error of the accelerometer
- **Gyroscope:** measures the angular rotation of the spacecraft to estimate its orientation by integration
- **Star tracker:** from the position of the stars, measure the attitude of the spacecraft to be compared with the gyroscope
- **Sun sensor:** gives an estimation of the Sun's position, used for attitude determination and solar panel orientation
- **Magnetometer:** measure the magnetic field of the Earth, used for attitude determination near the Earth and avoid Van Allen belt high radiation zone

These sensors will be integrated in a sensor fusion algorithm to give a precise estimation of the spacecraft state that can be used for trajectory maneuvers or antenna pointing.

3.5.2 Actuator

To control the attitude of the spacecraft we will be using three control Momentum Gyroscopes (CMG) from Airbus (14) that will provide the fine attitude control necessary to have a good radio communication. For redundancy purposes, a fourth one is also integrated, as the electronic box from Airbus can control up to four CMG. To de-saturate the CMG, we can use the ion thrusters, as each thruster as two degree of liberty, allowing to control the three rotation of the spacecraft with thrust vectoring and thrust differential. However these maneuvers will have to be combined with transfer or station keeping maneuvers, because we cannot achieve pure rotation with the ion thrusters, as they will always produce a net force whe powered on.

3.5.3 Command and Data Handling

To manage the large quantity of data received by the spacecraft as a relay satellite, we have to use a dedicated processing unit that will automatically transfer the data from the emitter to the receiver. Then to manage the internal data generated by the spacecraft's sensor, a separate system will be used based on a federated bus architecture with the different sensor and actuators of each systems connected to the same CAN bus. This bus was chosen due to its reliability and its baud rate of maximum 1 Mbps, ensuring that our data budget will be respected.

Finally to save the data generated by the spacecraft, 2 redundant flash memory of 10 Go will be used to each save one day of internal data. The rest of the data are then saved on Earth.

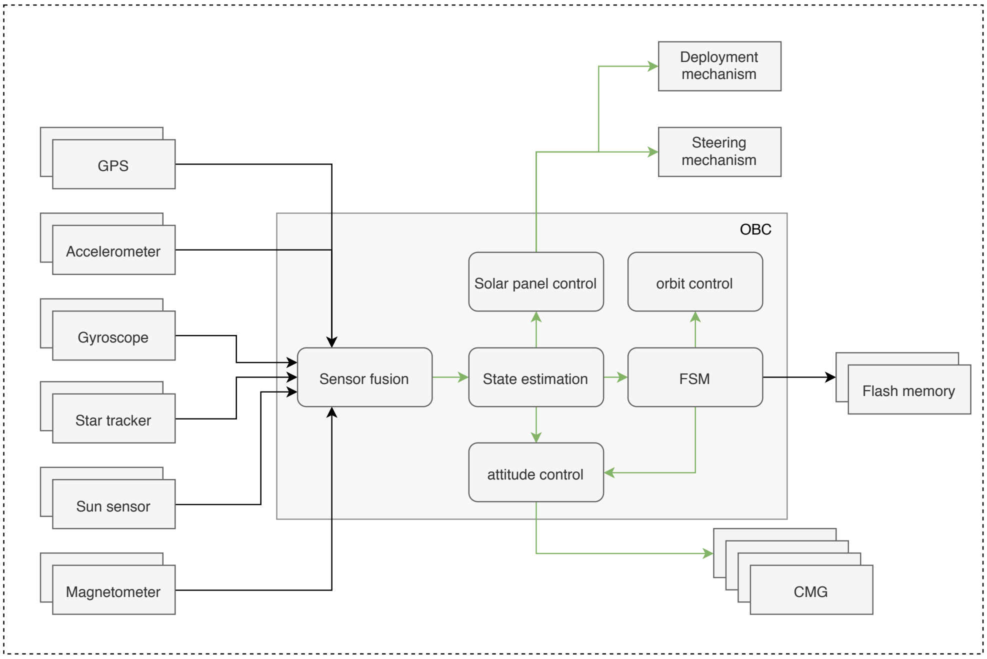

Figure 18: Hardware diagram of the ADCS system

3.6 Structure

3.6.1 Attachment

The primary and secondary structure of the spacecraft are both made of aluminium honeycomb panel from HexWeb. The primary structure is a simple rectangular box of 1.4m in larger and width, and 3.1m in length. Inside this box is attached 4 others 1.4m square panels that act as a secondary structure on which the internal components can be fixed, and also reinforce the overall spacecraft.

The secondary structure also create different compartments which separate the systems from one another, preventing one component failure to affect the other systems. Finally, to hold the solar panel, we mount them on the same aluminium panel but with a thickness of 6.35mm

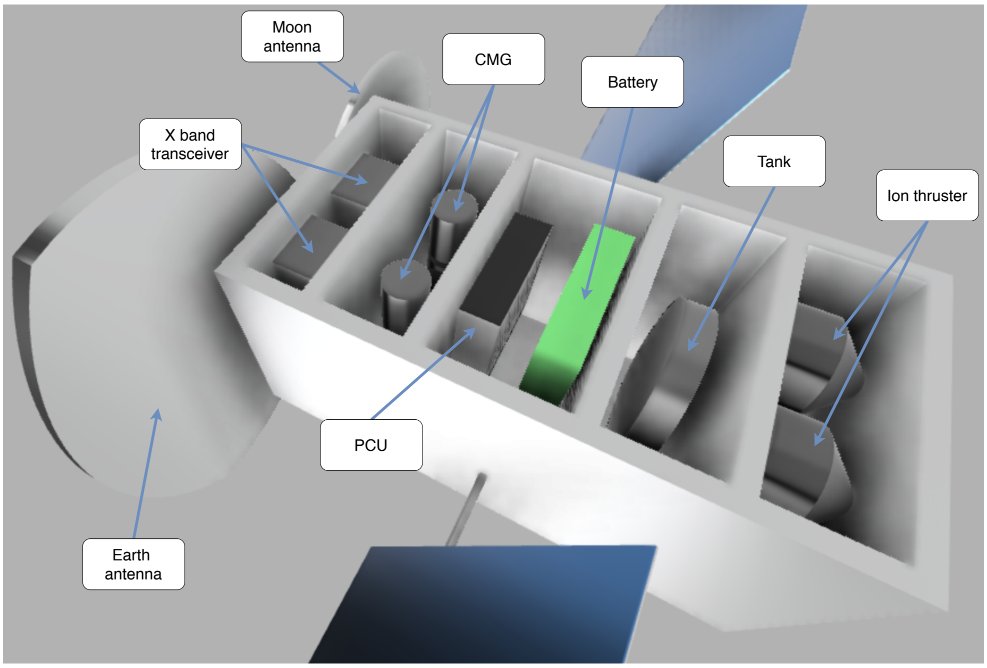

Figure 19: Architecture of the spacecraft. The secondary structure composed of honeycomb panel allow to fix the component inside

From Soyuz's user manual, we find a maximum acceleration of 4.3g, which combined with the 2403 kg of our spacecraft create a maximum force of $F = 4.3 * 9.81 * 2403 = 101.4kN$, distributed between the four lateral panels. From the honeycomb datasheet (13), we see that a 76.2 mm thick panel is enough to support this load, as the pressure will be:

$$P = \frac{F}{4 * 1,4 * 76,2} = 237,6kN/m^2$$

which is less than the 434.4 kN/m^2 supported by the panel with a safety factor of 0.82.

3.6.2 Mechanism

In order to use properly the solar panel, we nee two additional mechanisms. The first one is used to liberate the solar panel after that the spacecraft is put into orbit. For this we can use a deployment system from RUAG (17), which is specifically made to deploy solar panel and incorporate a damping mechanism to reduce the shock at the opening. The second mechanism needed is for the orientation of the solar panels, as during the transfer phase and the halo orbit, the position of the Sun with respect to the spacecraft will change. In order to always have the best efficiency at all time, we can again use a simple mechanism from RUAG (16), which consist of a motor, a slip ring, and potentiometers for feedback. However the sun orientation is not incorporated in the mechanism, and will have to be determined via a Sun sensor from the ADCS system.

3.6.3 Thermal

Just after our spacecraft is put in orbit, it start to experience serious thermal issues, as the conjunction of the Sun illumination, Earth proximity and heat produced by the ion thruster can elevate the temperature of the spacecraft well above its operating temperature. With this equation:

$$T = (\frac{A_p * J_p}{\sigma * A_{rad}} + \frac{Q_{internal}}{\sigma * A_{rad} * \epsilon} + \frac{A_s * J_s + A_a * J_a * \alpha}{\sigma * A_{rad} * \epsilon})^{\frac{1}{4}}$$

we can calculate the equilibrium temperature of our spacecraft during the eclipse phase of the orbit and the illumination phase.

The result are present in figure 20.

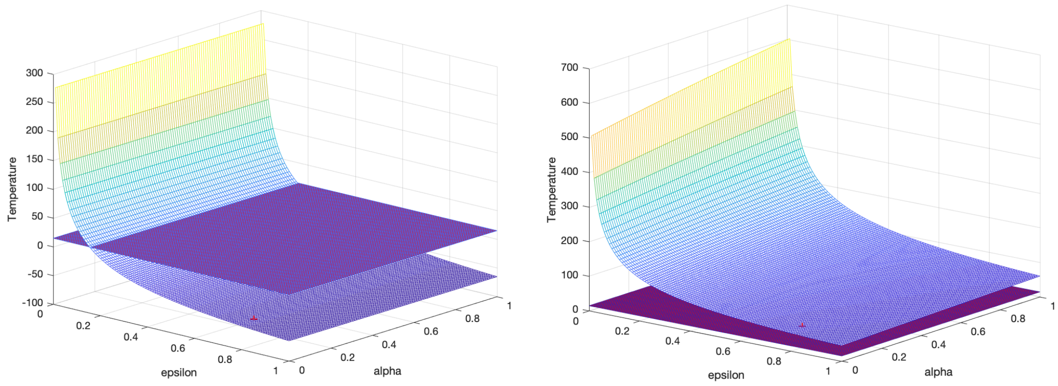

Figure 20: Equilibrium temperature for different values of ϵ and θ. The red cross correspond to a silver coated FEP surface. The plane correspond to 15°.At the left is during eclipse and at the right during illumination

We see that during eclipse, the equilibrium temperature is around -58.2°C, while during illumination, it is around 53.5 °C. This mean that with enough multilayer insulation (MLI), the resulting temperature will be between these two extreme temperature, near an operating temperature suitable for the components. However, as the spacecraft gain altitude, the resulting temperature will increase as well, as the spacecraft spend more time in the illuminated zone. To mitigate that, a cooler near the sensible components like the battery and the sensors has been accounted for in the power budget, but not properly sized.

4 Conclusion

To conclude this project, we can see that the main objectives of the mission have been respected and all the major elements of the spacecraft designed. One minor problem that happened is the long duration of the transfer from Earth to the Moon which is due to a too heavy spacecraft. This problem could probably have been avoided if a better mass budget had been made earlier.

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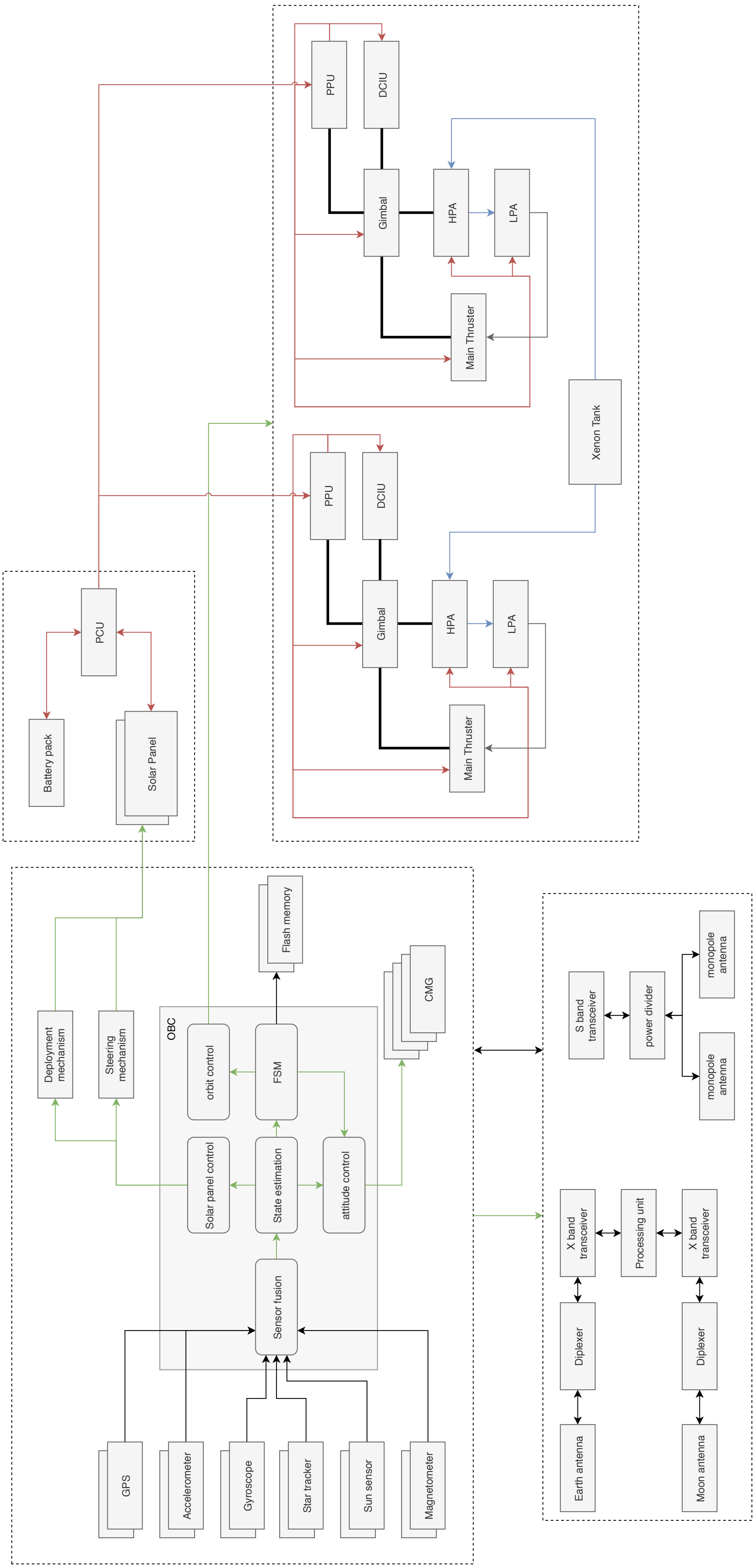
A Terms and abbreviations

CMG	Control Momentum Gyroscope
EP	Electric propulsion
LEO	Low Earth Orbit
LV	Launch Vehicle
MS	Mission Requirements
MEO	Medium Earth Orbit
NEO	Near Earth Object
PCU	Power Conditioning Unit

B Hardware diagram



PPU
DCIU
Gimbal
HPA
LPA
Main Thruster
Xenon Tank
PPU
DCIU
Gimbal
HPA
LPA
Main Thruster
PCU
Battery pack
Solar Panel
Deployment mechanism
Steering mechanism
OBC
orbit control
FSM
Flash memory
CMG
Solar panel control
State estimation
attitude control
Sensor fusion
GPS
Accelerometer
Gyroscope
Star tracker
Sun sensor
Magnetometer
S band transceiver
power divider
monopole antenna
monopole antenna
X band transceiver
Processing unit
X band transceiver
Diplexer
Diplexer
Earth antenna
Moon antenna