Spacecraft Design

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Very good work but mass budget, concept of operation and a few elements in mis

1 Introduction

As we are celebrating the 50 years of the Apollo 11 mission, we see new interest for the Moon emerging, being from private or governmental association. However, the far side of the Moon is still getting very little attention as few missions have tried to explore it . In this project, we will be designing a mission that will help future lunar exploration by relaying data from the Moon to the Earth over long period of time.

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You actually orbit the Eart-Moon L2 if I understand correctly. This is important because you then will only be able to relay from missions on the far side of the Moon.

2 Mission Overview

2.1 Mission definition

The goal of the mission is to demonstrate the ability of a spacecraft to use an electric propulsion engine to reach a Moon orbit orbit from a low earth orbit using only its solar panel, effectively reducing the overall cost of the mission. Once the satellite has reached its final orbit, it will be used to relay data transmission of other Moon mission to the Earth. The use of electric engine will greatly increase the duration of the mission, as the transfer from a low-Earth orbit to a Moon orbit can take more than a year. 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 LEO orbit
`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 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:

Having for main objectiv

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

Nobody expects a satellite to place itself in orbit...

As electric propulsion engine have a thrust to weight ratio well below one, they cannot be used to place our spacecraft into orbit. For this we will be using a commercial launch vehicle using standard chemical propulsion. Because electric propulsion engine have a high specific impulse, less propellant will be needed for the mission, reducing the initial mass of the spacecraft. This combined to the low altitude orbit targeted means that we can choose a small-lift launch

Rocket	Max payload to LEO [kg]	Inclination [°]	Price [M$]
Minotaur 1	580	28.5	29
Pegasus	470	11	40
Pegasus	450	28.5	40
Epsilon	1200	30	38 (19)
Vega	1963	5.4	32 (16)

Table 1: List of different LV suitable for the mission. Prices in parentheses are an estimation of the real cost due to sharing of the launch vehicle with another payload

Source for these cos

vehicle to get to LEO at a fair price, without having to share the launch vehicle with another payload. A comparison of various launch vehicle is listed below in Table 1

An important parameter to consider here is the inclination of the LEO. As the moon declination changes over time with a maximum of 28.5 °, launching our spacecraft at a too high inclination will result in additional maneuvers to reach the declination of the Moon. For example, the Minotaur 1's lowest inclination is 28.5 °, which correspond to the maximum inclination of the Moon happening every 18.6 years. To reduce the complexity of the transfer, only Pegasus and Vega can be used as they have relatively low inclination allowing to reach directly the Moon without extra maneuvers.

We see that Vega is cheaper than Pegasus, even if we use the whole 1963 kg capacity, so using this LV is a great option as it will give us a lot of flexibility in terms of mass. Also, because Vega inject us into a 5.4 degree orbit, we see on figure 1 that we have a window every 13.66 days when the Moon reaches the same inclination as us, which also give us some flexibility on the launch date.
If we assume five years of development, a possible arrival date would be in November 13th, 2024. As we see in Section 2.2.2, the typical transfer duration is 181 days, so the corresponding launch date would be in May 16th, 2024 However it is important to note that for LEO, Vega insert us into an elliptical orbit with an apogee at 1500km and a perigee at 200km of altitude.

All the orbital elements are summarized in Table 2. While the three first parameters are constraint by our choice of launch vehicle, we can choose the last two in order to maximize Sun illumination during the first phase of the transfer, when the altitude is the lowest and thus the eclipse time is the highest. By deciding to launch in May 16th, choosing a RAAN of 287 °and an argument of perigee of 90 °will minimize the eclipse duration approximately 15 days after the launch, which correspond to a third of the first transfer phase (see Table 3)

Your insertion orbit could also depend on if you are a secondary payload. In all cases, where does the 200x1500km orbit comes from? Your should be able to launch to a quasi-circular orbit as well and not suffer problems with your perigee at the beginning of mission.

For your choice of Raan, should you not match the moon's raan instead if you try to minimize discrepancy between the two orbital planes?

2.2.2 Transfer

In this section, we will describe the process to go from a low-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.

Semi major axis	7221 km
Eccentricity	0.09
Inclination	5.4 °
RAAN	287 °
Argument of perigee	90 °
Period	1.69 h

Table 2: Orbital elements of the initial orbit

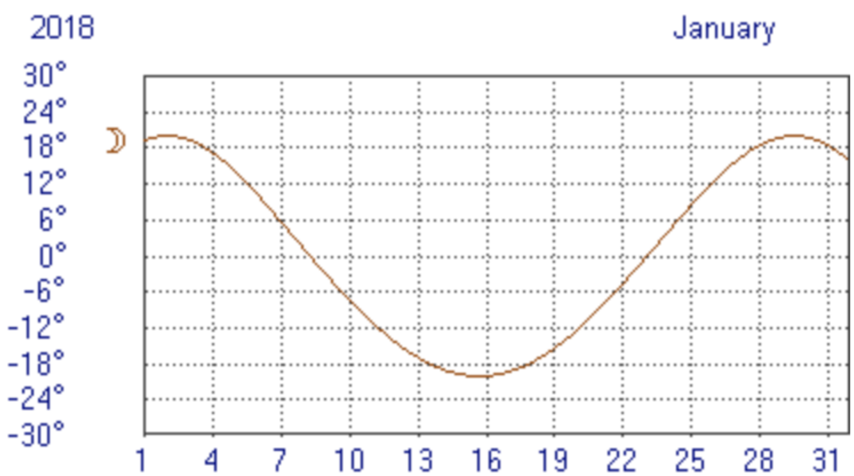


Figure 1: Moon declination over one Month. At this period, the maximum of 28.5 degree is never reached

Phase	Duration [days]	Delta-V [m/s]
Van-Hallen escape	89	3138
Manifold insertion	92	3542
Cruising	0	50
Station keeping	5 years	518
Total	181 (+5 years)	7198

Cruising duration?

Table 3: Different phases of the LEO to Moon transfer. The two first lines were estimated using Matlab simulation, and the two last comes from two articles (3) (1). Note that the LEO insertion and Halo orbit insertion phases are not present in this table because their cost and duration are negligible as a first approximation

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.
The complete trajectory is detailed as follow:

- **LEO insertion**: Our spacecraft is launched from Earth and inserted into a 200km-1500km of altitude, elliptical orbit.
- **Van-Hallen escape**: The radius of the orbit is raised to 12'000 km by continuously thrusting along the velocity vector. This maneuvers allows to escape the first Van Allen belt which can cause major damage to the solar arrays.
- **Manifold insertion**: By thrusting only during the periapsis, the apoapsis is slowly increased to around 300'000 km of altitude, then circularised to raise the periapsis at the same altitude. At this point, the spacecraft enter the stable manifold trajectory.
- **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.

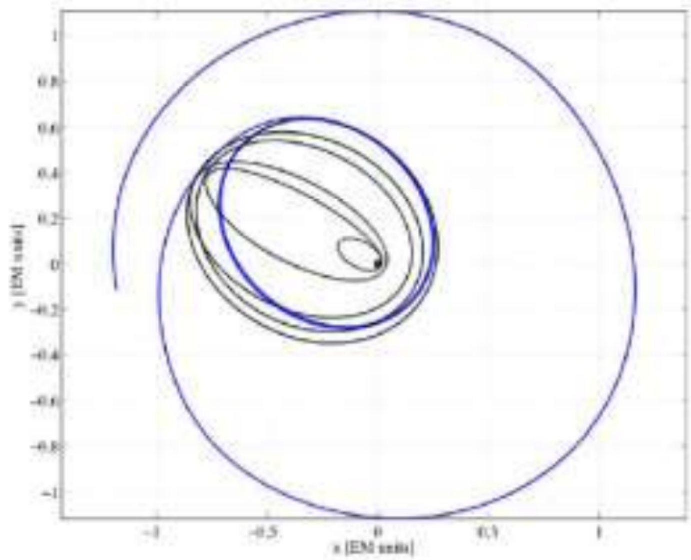


Figure 2: Interior low energy transfer trajectory. The blue line correspond to the Cruising phase and the black line to the Manifold insertion.

2.2.3 Final orbit

The final orbit is an Halo orbit around the Moon-Earth Lagragian point. 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 $\pm$ 7.75°, with a spacecraft orbiting at 64500 km in a circular orbit of 3500 km in radius, we can use a fixed lunar antenna with a minimal beamwidth of:

$$\theta = 2 * \arctan(\frac{3500}{64500}) * \frac{180}{\pi} + 7.75 \approx 14°$$

to guarantee that our spacecraft will always be in view of the lunar station without needing for antenna tracking.

Clear specification of mission type and class are missing. Redundancy description and

3 System overview

In this section we will go more in depth about the different system 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 have to implement to accomplish its mission.

Nice, but unreadable..

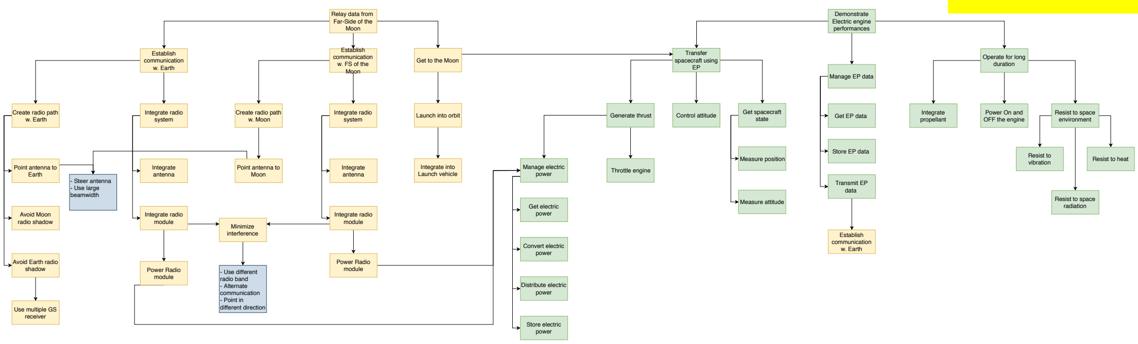


Figure 3: Functional diagram of the project. The communication part of the mission is in yellow while the propulsion is in green

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 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 system and subsystem needed to implement each functionality. This process is summarized in figure 4.

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

Before discussing the implementation of each system, it is important to describe the general timeline of the project. As it was mentioned above, the development phase of the project will last five years, as no additional equipment will have

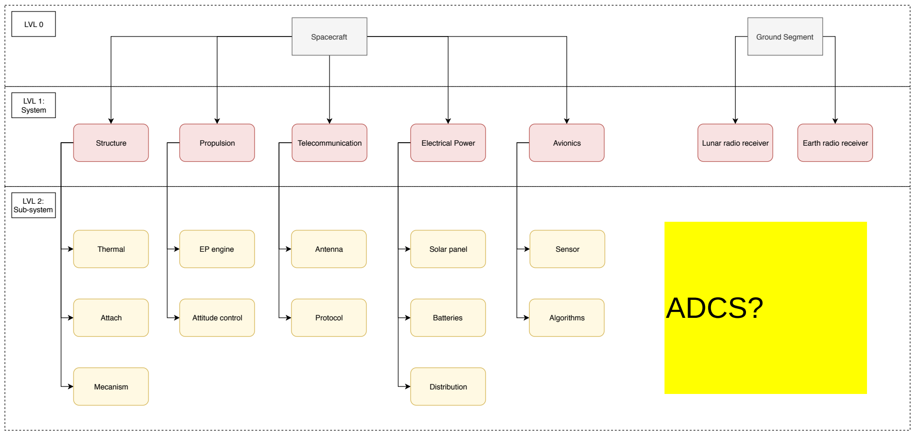


Figure 4: System decomposition

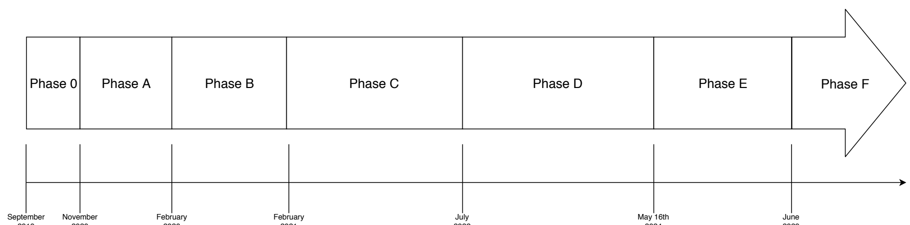


Figure 5: Complete planning of the mission. The longest phase is phase E, during which our spacecraft act as communication relay for lunar operations

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.

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 6. 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

System	Subsystem	Power during eclipse	Power during sunlight
Propulsion	EP engine	7092,5	7092,5
	Attitude control	193,32	193,32
	Total (20% cont.)	8743,0	8743,0
Structure	Thermal	20	140
	Mechanism	150	150
	Total (20% cont.)	204	348
Telecommunication	Antenna	27	27
	Total (20% cont.)	32,4	32,4
Avionics	Sensor	20	20
	Algorithms	5	5
	Total (20% cont.)	30	30
Electrical Power	Distribution	544,9	274,6
	Batteries	9428,0	-9428,0
	Solar panel	-18982,3	0

What are your assumptior

Mass budget missin

Figure 6: Power budget of the different systems and subsystems

3.2 Propulsion

3.2.1 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 engine

As the goal of this mission is to assess the performances of an EP engine,

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

I decided to choose 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

3.2.2 Attitude control

To control the orientation of the spacecraft, small motor will be placed at the edge of the spacecraft to generate torque. As we want to control the

Engine	Isp [s]	Power [W]	Min Impulse bit [mN.s]	Mass [kg]	Thrust [N]
MR-103G 1N	224	16.11	13.3	0.33	1.13-0.19

Table 6: Specifications of the MR-103G 1N engine

orientation of the three axes in both directions, without applying a net force on the spacecraft, we will need four motors per axes, so a total of twelve motors.

As these motors will only be used to correct perturbations of the orbit, we can use less efficient technologies like monopropellant, which has the advantages of being simple to use, small and lightweight. We can estimate at 20 m/s per year the cost of these maneuvers. As this is still very uncertain, we add 30% margin to be safe, which give us a total delta-V of 143 m/s. A possible candidate is the MR-103G 1N hydrazine engine, whose specifications are summarized in Table 6

You do not elaborate on

By using the Tsiolkovsky equation, we get that 6.7 % of the spacecraft mass should be hydrazine propellant.

3.3 Electrical power

3.3.1 Solar Panel

To accommodate for the large power of the NEXT engine, we will need large solar panels. These solar panels will have to get more energy that the spacecraft uses to compensate for the eclipse time during the first phase of the transfer. To mitigate the long period inside the Van Hallen belt, we will be using Indium Phosphide solar panel that have an efficiency of 18%. To find the power needed from the solar panel, we take the consumption of the spacecraft in daylight mode and multiply by a correctif factor k taking into account the degradation of the solar panel and a margin m. With all of this, we can calculate the size of the solar panel needed for the first phase of the transfer:

$$A = \frac{P_{SA} * m * k}{F * \eta} = \frac{18982 * 1.3 * 1.25}{1360 * 0.18} = 126m^2$$

which correspond approximately to the area spanned by four rectangular 2 by 16 meters solar panel. As the minimum diameter of the Vega fairing is about 2.2 meters, it is not possible to increase the width of the solar panel to more than 2 meters.

3.3.2 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. From the power budget, we can estimate the power consumption during eclipse to be 9428 Watt. To estimate the duration of the eclipse, a simple matlab script is used. The result is showed in figure 8. We see that the maximum eclipse duration is

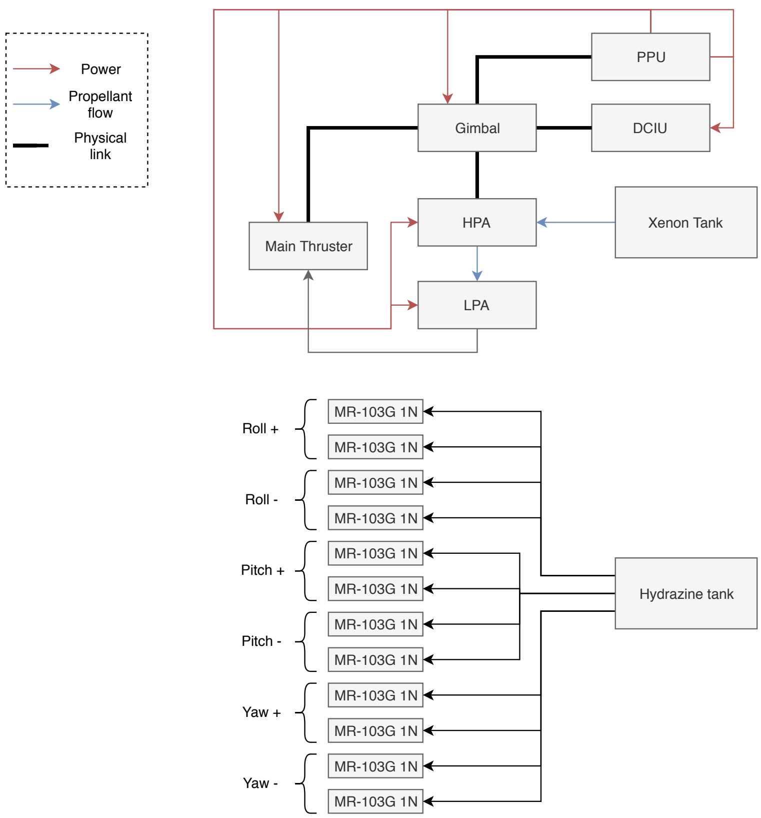


Figure 7: Harware diagram of the propulsion system, with the main engine and the attitude control engines

Battery type	Energy [Wh]	Mass [kg]	Volume [Wh/l]	Power output [kW]
LVP65	240.5	2.75	1.18	1.2
Battery pack	10101	115.5	49.56	50.5

Table 7: Specification of the LVP65 battery for one cell and for the complete battery pack

actually at the end of the first phase, because the period of the orbit is longer. This duration can be used to calculate the size of the battery:

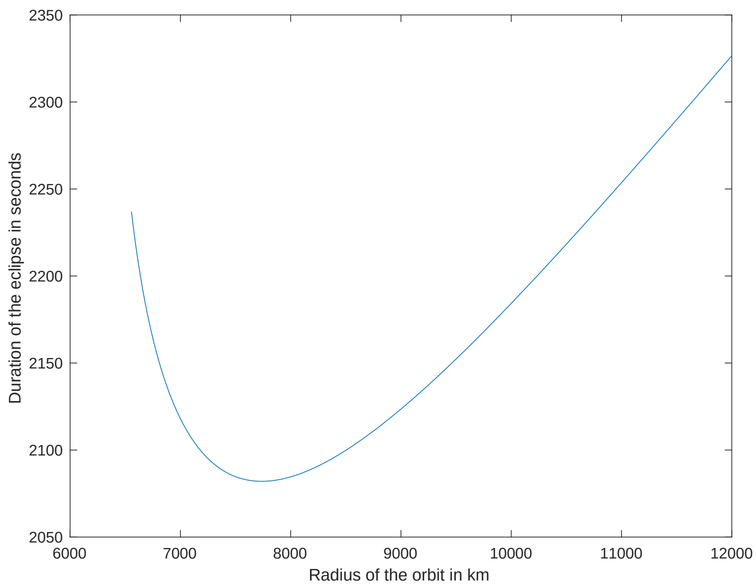
$$C = \frac{P_{Bat} * 1.3}{T_{max} * 0.8} = 9903Wh$$

with a recommended depth of discharge of 0.8 and a contingency of 30% This result allow us to size our battery. From the LVP serie datasheet (7) we find some important parameters listed in Table 7. From these parameters we can find the parameters of our final battery pack, composed of 42 cells.

3.3.3 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 18.9kW of power generated by the solar panel during daylight.

A possible candidate would be the PCU Next generation from Thales, 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.



We would like to see a computation of eclipse when you re

Figure 8: Time duration of the eclipse in seconds as a function of the altitude of the first phase

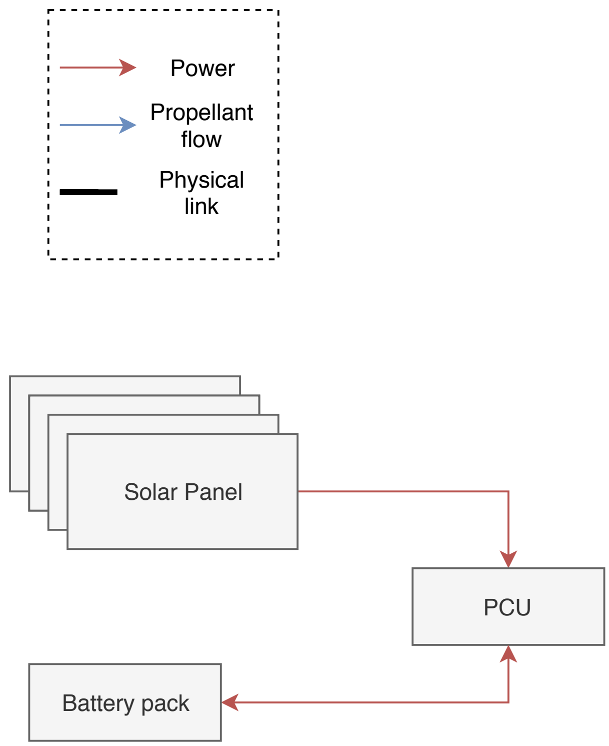


Figure 9: Hardware diagrma of the electrical power system

4 Conclusion

Now that the first part of the project is done, we have a more precise definition of its goal and requirements. In the following weeks, the other systems will be studied and a new iteration of the system engineer documents will be created.

ConOp missing.

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A Terms and abbreviations

EP	Electric propulsion
LEO	Low Earth Orbit
LV	Launch Vehicle
MS	Mission Requirements
NEO	Near Earth Object
PCU	Power Conditioning Unit

B Hardware diagram