Delft University of Technology

Faculty of Aerospace Engineering
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Aerospace Systems Design

Design of the JUICE Spacecraft

A Mission to the Giant of our Solar Systems

Mission: Jupiter's JUICE mission
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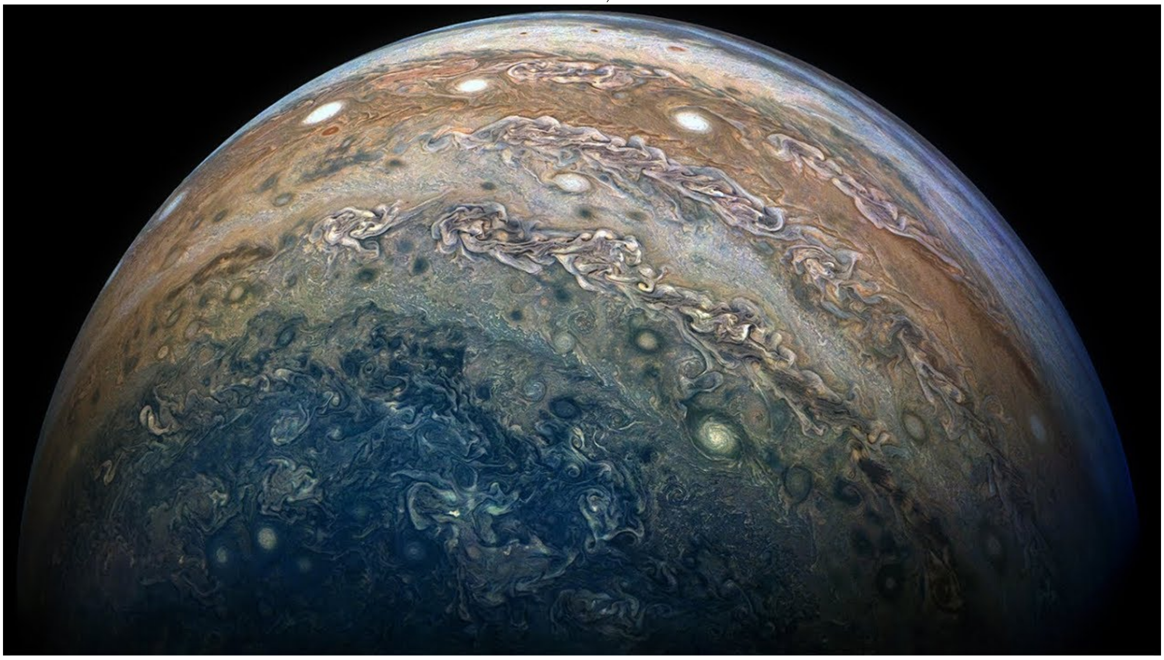

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Summary

The previous report outlined the general requirements, objectives and preliminary, first level estimates for the JUICE spacecraft. It is set to launch in 2025 for its twelve year long mission to explore the atmosphere and magnetosphere of the gas giant. The aim of this report is to describe the design of

the preliminary design of each subsystem the spacecraft consists of and to refine the mass, power and cost budgets. Requirements were generated for each subsystems and new concepts were discussed. One specific concept was then chosen and preliminary sizing of the components of the subsystem was performed.

The ADCS' requirements are driven by the payload and results in a three axis stabilized and controlled spacecraft. This is done through a selected set of sensors and actuators. The sensors consist of three star sensors, which provide high accuracy and redundant pointing knowledge. As a backup an additional sun sensor is used, while gyros are used for the time frame between calibrations using the star sensors. The actuation will be provided by four redundantly placed momentum wheels and momentum dumping on all three axis is performed using twelve thrusters.

The propulsion subsystem's requirements are mainly flowing from the mission profile and duration. As such the ΔV budget of the previous estimation was refined to be $1925.5m/s$, the LEROS 1b engine with nitrogen tetroxide and hydrazine as propellants was chosen based on this information and the mission profile. This yields a propellant mass, depending on the empty mass of 572.6kg to 985.15kg.

Design of the thermal subsystem first requires an analysis of the range of temperatures in which the components of the spacecraft can operate. Then the heat flux in and out of the probe depending on the stage of the mission was analyzed to find an equilibrium temperature of about 167.9 - 166.2K. This shows that active thermal control is required to heat up the spacecraft, to be able to keep all components running in their thermal window.

All subsystems need power provided by the EPS for proper functioning. Thus the requirements of the power subsystem are driven by the power need of the spacecraft as a whole. The source of power for this mission was selected to be solar power, using the TJ GaAs ultraflex type panels. This results in an area of $98.8m^2$ and $313.2kg$ of mass to generate the required maximum $263.43W$ at end of life. The Galileo CPDU of 16.3kg and a $778.8Wh$ battery of $5.86kg$ were chosen for the design.

The structure of the spacecraft has to withstand the launch, handle the loads and provide an interface for all components. The chosen material is carbon nanotube with a hollow cylinder structure of $3m$ diameter and thickness of $0.65mm$. The structure also incorporates two tanks used to hold the fuel and another for the oxidizer. They are assumed to be spherical, titanium tanks of $0.79m$ and $0.91m$ diameter, $1.25mm$ and $1.44mm$ thickness for the fuel and oxidizer respectively. The total mass of the tanks was estimated to be $38.8kg$. A preliminary design of the mechanism to deploy the solar panels has been laid out.

The central part of the mission is the observation of the natural phenomena of Jupiter. Thus multiple instruments will be carried on board of the JUICE mission. These are based on previous spacecrafts' instruments, Table 1 presents their details.

Table 1: Table with the mass, power and data rate of the scientific instruments of the JUICE orbiter

Instrument	**Mass** [kg]	**Operating Power** [W]	**Data Rate** [kb/s]
CAPS	12.50	14.50	0.50
MAG	15.30	2.0	4.0
JunoCam	3.70	5.90	0.33
JIRAM	8.0	16.70	~3.37
UVS	21.50	9.0	~1.77
Total	60.95	48.1	9.96

Once the subsystems have been preliminary designed the budgets can be revised. The result of this can be seen in **??** and Table 3.

Table 2: M_e Budget

Subsystems	**Mass [kg]**
ADCS	61.3
CDH	40.6
Communications	59.0
EPS	335.4
Harness	75.1
Payload	61.0
Structures and mechanisms	146.3
Thermal	30.0
Total without propulsion (81%)	808.61
Propulsion (19%)	165.7
Total without margin	974.3
22 % Margin[1]	215.83
Total with margin	**1190.1**

Table 3: Total power budget M_e

Subsystems	**Power [W]**
ADCS	14.1-17.0
CDH	19.2-23.2
Communications	38.6-46.4
EPS	2.6-3.1
Harness	-
Payload	48.1
Structures and mechanisms	6.4-7.7
Thermal	42.5-51.0
Propulsion	5.1-6.2
Total without margin	176.6
Total with margin	**202.7**

Finally the spacecraft layout and architecture have been preliminarily discussed. The spacecraft consists of a central cylinder containing all the systems inside. On one end there will be the main engine while the other end holds the antenna for communicating with the ground station. Solar panels will be deployed out radially from the cylinder. A rendering can be seen in Figure 1.

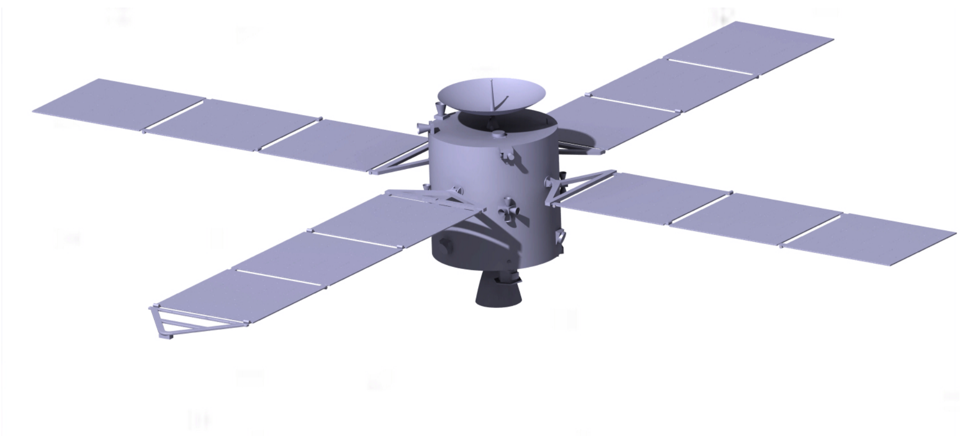

Figure 1: Render of the JUICE spacecraft

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List of Symbols

Latin Letters	Quantity	Symbol Unit
a	Semi-major axis	km
A_e	Emitting area	m^2
A_i	Area that catches albedo and planetary radiation	m^2
A_s	Area that catches solar radiation	m^2
A_{SP}	Solar panel area	m^2
b	Semi-minor axis	km
B	Magnetic field vector	T
c	Speed of light	m/s
$C_{S/C}$	Spacecraft cost	$
D	Distance Jupiter - Spacecraft	km
E_{BAT}	Energy in battery	Wh
E_{sp}	Specific energy	Wh/kg
E_ρ	Energy density	Wh/l
F	Force	N
F_a	Visibility	[-]
I_{sp}	Specific impulse	s
I_{solar}	Impulse from solar disturbing forces	Ns
$I_{magetic}$	Impulse from magnteic disturbing forces	Ns
J_a	Albedo	
J_s	Solar irradiance	W/m^2
J_{IR}	Planetary flux	W/m^2
M	Spacecraft residual dipole	$A-m^2$
M_avg	Average mass	kg
M_{BAT}	Battery mass	kg
M_{Dry}	Dry mass	kg
M_e	Empty mass	kg
M_{Loaded}	Loaded mass	kg
M_p	Propellant mass	kg
M_{PL}	Payload mass	kg
M_{SP}	Solar panel mass	kg
P_{BOL}	Power at beginning of life	W
P_{EOL}	Power at end of life	W
P_j	Planetary Power	W/m^2
P_{PL}	Payload power	W
P_s	Solar pressure	N/m^2

P_{spec}	Specific power	W/kg
P_t	Total power	W
Q_{IR}	Surface flux	W/m^2
$\dot{Q}_{in}$	Power in	W
$\dot{Q}_{out}$	Power out	W
$\dot{Q}_{absorbed}$	Power absorbed	W
$\dot{Q}_{emitted}$	Power emitted	
r	radius	m
R	Reliability	[-]
R_a	Apocenter	km
R_j	Radius of Jupiter	km
R_p	Pericenter	km
S	Frontal surface area	m^2
T	Orbital period	s
T_m	Magnetic torque	Nm
T_e	Eclipse time	s
V_{BAT}	Battery volume	l
$V_{S/C}$	Spacecraft volume	m^3
V_o	Volume of Oxidizer	m^3
V_f	Volume of Fuel	m^3
g_y	Load in longitudinal Direction	m/s^2
g_x	Load in Axial Direction	m/s^2
t	Thickness	m
E	E-Modulus	GPa
L_p	Length of primary structure of the spacecraft	m
L_c	Maximum length of spacecraft primary structure from center of mass	m
r	Radius of spacecraft primary structure	m
P_t	Pressure present in the Tank	MPa
$\rho_{propellant}$	Average density of propellant	kg/m^3

Greek Letters	**Quantity**	**Symbol Unit**
α	Absorptivity η	Efficiency
[-]		
ϵ_{IR}	Infrared emissivity	[-]
μ	Standard gravitational parameter	km^3/s^2
μ_{BAT}	Battery efficiency	[-]
ρ_r	Reflectivity	[-]
σ	Boltzmann constant	[-]

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0.1 General

ADCS - *Attitude Determination and Control System*
BOL - *Beginning of Life*
CDH - *Command and Data Handling*
DASML - *Delft Aerospace Structures and Materials Laboratory*
DOD - *Depth of discharge*
EPS - *Electrical and Power System*
EOL - *End of Life*
FAA - *Federal Aviation Administration*
HES - *High-efficiency silicon*
MMOI - *Mass moment of inertia*
PCDU - *Power control and distribution unit*
PCU - *Power conditioning and control unit*
PDU - *Power distribution unit*
RTG - *Radioisotope Thermoelectric Generator*
S/C - *Spacecraft*
TBD - *To be determined*
TJ - *Triple Junction* TT&C - *Telemetry, Tracking and Control*
TW - *Technical Writing* WP1 - *Workpackage 1*

1 Introduction

Mankind has always been seeking for a deeper and more thorough understanding of what is surrounding it. A first stepping stone to a better knowledge of the creation of the Universe starts with the solar system and its gigantic bodies, Jupiter being one of them. There have been only two past missions to actually orbit Jupiter, meaning that the gas giant is a relatively unexplored celestial body of the solar system. The JUICE mission aims to find out more about the atmospheric phenomena present of the planet's complex atmosphere with a special focus on the various cyclones occurring there. Additionally the intense magnetic field and radiation patterns in the neighbourhood shall be further analyzed, continuing the work previous probes like Juno.

The goal of the work of this group is to design the JUICE spacecraft that is to carry the scientific instruments to its final destination and make sure that the payload can observe the Jupiter to generate the data the scientific community is interested in. This report aims to refine the initial first level estimates generated in the previous work, by analyzing each subsystem individually and explaining the process with which each was designed. This results in new, refined mass, power and cost budgets as well as a more detailed general architecture of the spacecraft.

The report will have the following structure. In chapter 2, each subsystem will have its own section. Each section will describe the requirements of that subsystem, which then will lead to a preliminary design and sizing of that system. In chapter 3 the results of the design of the subsystem will be summarized to refine mass, power and cost budgets that have been set up in the previous report. chapter 4 shows the preliminary spacecraft architecture that illustrates the size of all major components on the spacecraft as well as their placement. Finally chapter 5 summarizes all the results obtained during the work of this group.

Subsystems Design 2

This chapter aims to provide the preliminary designs of the first most important subsystems of the spacecraft. Section 2.1 presents the ADCS subsystem design while section 2.2 gives an overview of the propulsion subsystem. Sections 2.3 and 2.4 provide a discussion of the thermal and power subsystems respectively. The design of the structure is approached in 2.5 and finally, the payload and the other subsystems are discussed in sections 2.6 and 2.7 respectively.

2.1 ADCS

The ADCS is responsible for acquiring and controlling the attitude of the spacecraft. As such it makes sure other systems such as the imagers or antennas pointing in the required directions and thus is essential to fulfilling the mission. This section details the preliminary design choices of the system. First the requirements are defined, after which the various operation modes needed are explained. This is followed by the disturbances the ADCS has to counteract. The types of sensors and actuators as well as their number for various axis and redundancy are explained, followed by a schematic explaination of the ADCS software operation.

2.1.1 Key Requirements and Functions

The functions of the ADCS consist in stabilizing the spacecraft and point the various subsystems such as the payload, antenna, solar arrays and engine in a certain direction. The level of accuracy of the pointing and general stability of the spacecraft are dictated by the requirements of the various other subsystems such as EPS and TT&C among others. The driving requirement can be selected as the most decisive one from the the other subsystems. Additionally, as it is the case for the complete spacecraft, the reliability should be kept to a maximum. As shown in [2] the ADCS is a common source of failure because of the mechanisms necessary for the attitude control, thus a redundant attitude actuation system should be implemented. Thus the following requirements can be defined from the Cassini spacecraft which has very similar instruments on board[3].

- ADCS1.1: The pointing accuracy shall be 2 mrad or better.
- ADCS1.2: The pointing stability shall be 8 $\frac{\mu rad}{s}$ or better.
- ADCS1.3: The maneuvering rate shall be 0.1 $\frac{rad}{s}$.
- ADCS1.4: The attitude actuators shall be in a redundant configuration.
- ADCS1.5: The spacecraft shall be 3-Axis stabilized.

2.1.2 Modes of Operation

Depending on the phase of the mission and the different required attitudes, the spacecraft will implement different modes of determination and control as described in [2]. Here the modes will be explained and why they are needed or are not.

- **The launch mode** is used during the launch when the spacecraft is still attached to the launch vehicle. During this time the spacecraft is not responsible for its own attitude and positioning, thus it should simply remain idle until it is released from the launch vehicle.
- **The de-tumble mode** is used just after separation or in the case that the spacecraft loses control of its attitude during its mission. This mode reduces the rotations around each axis to near zero, to be able to start or resume its mission.
- **The attitude acquisition** permits to determine the attitude from multiple sensors, such as sun or star sensors. These need to have their reference points in their field of view to be able to determine the precise attitude of the spacecraft. The attitude acquisition mode rotates the spacecraft so the sensors have their reference points in their field of view. It is activated every once in a while to reset the drift the gyros accumulate over time.
- **The operational mode** is the main configuration of the spacecraft, during which the mission objective will take place. This mode makes sure that the payload instruments are pointed and stabilized as required by the mission phase. a
- **The thrust mode** is used when the propulsion system is accelerating the spacecraft. This orients the engine to make sure the planned trajectory can be achieved.
- **The communication mode** is needed when the antenna cannot be aligned with earth during its 'normal' operations. In that case, there needs to be a separate configuration during which the antenna can be pointed correctly to enable a proper link between the ground station and the spacecraft. As this depends on how the antenna is aligned, this mode might not be needed if the operational mode already permits communication.
- **The safe mode** is a contingency in case anything goes wrong. This mode is supposed to stabilize the spacecraft and wait until further commands are received to take action.

2.1.3 External and Internal Disturbances

The spacecraft will endure many disturbances while in operation. In this section a deeper look into these will be taken. First, some causes of these disturbances are given in Table 2.1:

Table 2.1: Disturbances that could occur during operations

Disturbance source	**Explanation**
Gravity gradient	Torques can be generated due to a variation of the gravitational field along the spacecraft
Aerodynamic pressure	Due to an imbalance of aerodynamic pressure on different sides of the spacecraft a torque can be generated
Solar radiation pressure	Different amount of solar radiation on different sides of the spacecraft can cause disturbance torques
External magnetic field	Due to the external magnetic fields the spacecraft can experience a torque
Internally generated torques	Reaction wheels, solar panel orientation mechanisms and thrusters can cause additional disturbance torques

In the first stage near Earth, the spacecraft may endure many different types of disturbances such as due to a variation in the gravitational field, aerodynamic pressure from Earth's atmosphere, solar radiation, magnetic effects from the magnetosphere and internally generated torques. The latter are caused by thrusters when the spacecraft's orbit is being changed or the deploying of the solar panels among others.

During interplanetary cruise the spacecraft will not be close to any celestial body, meaning that it will mainly endure internally generated and solar pressure disturbance torques.

While the spacecraft is getting closer to Jupiter a lot more disturbances may come up. First, the spacecraft will gradually enter Jupiter's gravitational and magnetic field, which will cause disturbance torques. Furthermore, when JUICE will get closer, the pressure of Jupiter's atmosphere may cause additional torques to the spacecraft. However, these effects will vary greatly throughout the orbit as the orbit is highly elliptical and the distance to Jupiter varies. Torques due to solar radiation pressure may still be present, but these will be considerably smaller compared to when the spacecraft is near Earth. In addition, the internally generated torques will be similar to the other phases of the mission.

2.1.4 Attitude Sensors and Control Actuators

An accurate positioning of the spacecraft is of great importance for the success of the mission, therefore, multiple sensors are used to determine the exact attitude of the spacecraft. First the imagers and spectrometers require a correct orientation with respect to the surface of Jupiter, this is checked using three non-parallel star sensors. Two of those working in pair can very accurately determine the attitude of the spacecraft in three axis, the third one is added for redundancy. The drawback however is that these require more power than other types of sensors and require very low rotation rates to work reliably. Thus in case of entering safe mode or operating under higher rotation rates a coarse sun sensor will be used. This sensor is very light weight, simple and uses little to no power. Furthermore, three gyros are used to track the rotations of the spacecraft once the exact attitude of the spacecraft has been determined by the star sensors. Meaning that it is not always necessary to point the star sensors to the reference points, instead the gyros can indicate how far the spacecraft has rotated from that position to determine the current position. As the accuracy of gyros wanders over time, a periodic resetting of the attitude using the star sensors is necessary. [4]

Actuators are needed to adjust the attitude of the spacecraft. But those are mechanisms that fail rather often and it is thus wanted to reduce the nedd for those by using spin-stabilisation. A major disadvantage of this type of stabilisation is that it becomes difficult to point the payload and the solar panels in the right direction at all times. Due to the relatively complex pointing requirements of the payload, antenna and solar panels the JUICE spacecraft will not be spin stabilized. Thus a there axis stabilization system will be used and, for general pointing and stabilization four momentum wheels will be used. These provide highly accurate stabilization and torque to change the attitude to the required direction. Only three wheels are required for full three axis control and a fourth one is added for redundancy. A way of momentum dumping is required as those will saturate over time, this will be done using thrusters. For each direction of rotation per axis two thrusters are needed positioned equal distances away from the center of mass. This ensures that no translation will occur when the thrusters are used. As there are two directions per axis and three axis in total twelve thrusters will be required for full 3 axis control.

2.1.5 ADCS Software Block Diagram

The software block diagram as shown in Figure 2.1 illustrates the operation of the software that runs the ADCS.

Images/adcs block.png

Figure 2.1: ADCS software block diagram

2.2 Propulsion

The propulsion subsystem on-board of the JUICE orbiter should bring the spacecraft into orbit around Jupiter. Therefore, it shall be analyzed with great detail, to ensure a successful mission. Firstly, the subsystem requirements and their function will be established, then it is important to estimate and calculate eventual disturbing forces for the spacecraft. With these calculations the ΔV and the propellant mass can be deduced, which also helps selecting the thrusters and propellants. Lastly, all components of the propulsion system are listed and characterized, which will help in making a detailed sketch.

2.2.1 Key Requirements and Functions

It is crucial to list the main subsystem's requirements and functions to get a broader insight of what should be designed. Important to note is that this only covers the propulsion system of the final spacecraft without kick stages or launch vehicles.

- PROP1.1: *The ΔV budget shall be 1939.9.* The budget lists the required ΔV per maneuver, so in order to get a clear view on the ΔV budget, it should be known what maneuvers will take place.
- PROP1.2: *The maximum and minimum thrust level shall be TBD and TBD respectively (for now out of the scope of the report).* When considering the maximum thrust, it should be enough to hold disturbing forces for example. Furthermore, the minimum thrust level should be able to do the least force-intensive operations. Both should be determined for a proper selection of the main thruster.
- PROP1.3: *The maximum flight duration shall be 12 years.* The propellant of the propulsion system need to be storable for this time span in order to be operational for the complete life time.
- PROP1.4: *The propulsion subsystem shall perform at least TBD thrust periods during its complete life time(for now out of the scope of the report).* Since the propulsion system will not be active for the entire mission, it is essential that it is known how many thrust periods need to be performed in order to get the spacecraft into the orbit around Jupiter. The propulsion system needs to be reignited at least this number of times.
- PROP1.5: *The spacecraft shall be inserted into an orbit around Jupiter through the on-board propulsion subsystem.* This requirement will all affect the ΔV budget and because the subsystem design depends greatly on the ΔV budget, it is considered as a key requirement.
- PROP1.6: *The spacecraft shall perform three orbital manoeuvres (heliocentric orbit, Hohmann transfer and elliptical orbit).* It is necessary for the propulsion subsystem to know how many and what type of orbital manoeuvres are needed, to get the spacecraft into orbit. As this will later form the ΔV budget.

- PROP1.7: *The spacecraft shall use the propulsion subsystem for attitude control*: Again this will influence the ΔV budget and therefore, what is asked from the spacecraft's propulsion system.

- PROP1.8: *The spacecraft shall de-orbit at the end of the mission.* This has an impact on the ΔV budget.

- PROP1.9: *The spacecraft shall counteract disturbing forces with its propulsion subsystem.* The ΔV budget should include a buffer for (un)expected disturbances.

2.2.2 Disturbing Forces

During its mission, the JUICE orbiter will encounter some (un)expected disturbing forces. These ought to be estimated conservatively in order to make the propulsion subsystem successful. Therefore, this subsection will make an accurate estimation of the disturbing forces to overcome by propulsion.

Solar radiation

As the Sun is crucial for the heliocentric orbit, it is also responsible for solar radiation, which is one of the disturbing forces the spacecraft will experience. In order to calculate the solar radiation, one uses the equations 2.1 and 2.2 [2]:

$$P_s = \frac{J_s}{c} \quad (2.1) \qquad F_s = (1 + \rho_r) \cdot P_s \cdot S \quad (2.2)$$

First the solar pressure (P_s) is calculated using the solar intensity (J_s) that depends of the position in the solar system, and velocity of light (c). For the heliocentric orbit phase it was decided to use the worst-case scenario, so the closest point to the Sun, which is at the start of the heliocentric orbit, on leaving Earth's orbit (1,361 W/m^2). Thus, $P_s = \frac{1361}{3 \times 10^8} = 4.536 \mu N/m^2$. Using Equation 2.2, together with the reflectivity (ρ, which is assumed to be 0.5 as for most satellites), the solar pressure and the frontal area (S) of the spacecraft. The frontal area in the worst case scenario is estimated be the 98.8 $m^2$2.4.3 for the four solar panels added to 7.069 m^2 for the front of the cylindrical bus. The solar pressure at this position is thus estimated to be $F_s = 720.33 \mu N$. A similar procedure can be used to compute the solar pressure in orbit around Jupiter. Using $P_s = 0.1675 \mu N/m^2$, the force is found to be $F_s = 26.6 \mu N$.

Hence, the solar radiation force at maximum on the spacecraft will be 720.33 μN in the heliocentric orbit, and 26.6 μN in the elliptical orbit around Jupiter.

Magnetic torque

The magnetic field around Jupiter is enormous and much stronger than around Earth, hence the team will focus on the elliptical orbit around Jupiter for the maximum magnetic forces. It is therefore logical that the magnetic forces can affect the JUICE orbiter. At the point of the strongest magnetic field in the elliptical orbit, the spacecraft is 7,000km above Jupiter, being the closest it will ever get to the planet. The maximum torque experienced by the spacecraft is thus computed using Equation 2.3.

$$T_m = M \times B \quad (2.3)$$

Where M is the typical value of $0.1Am^2$ and B is the magnetic field vector of the position of the orbiter. B can be found in Table 2.7 in the ADSEE I reader[2], which shows that Jupiter's magnetosphere is about 20,000 times greater in magnitude than Earth's. Therefore, the maximum magnetic field strength that JUICE will experience is $0.000417T$. This is a substantial torque experienced by the spacecraft that definitely is to be taken into account.

Aerodynamic drag

Aerodynamic drag is expected to be negligible because of the high altitude from the start of the transfer to EOL. Even at the closest point around Jupiter, the gas density is negligible, it was therefore chosen to not include aerodynamic drag, when it comes to disturbing forces.

Gravity

As the JUICE orbiter will have a high-thrust strategy, the gravitational forces during maneuvers can be assumed to cancel out[2]. This comes from the fact that, at a high thrust to weight ratio, the ΔV can be assumed to be burnt instantly and therefore not letting gravity influence the spacecraft. Furthermore, the relatively high T/W-ratio will eliminate the gravity pulling the spacecraft out of the mission path[2].

2.2.3 Selection Thrusters and Propellants

Before selecting the thrusters and propellants for the propulsion subsystem, the ΔV should be revised, and refined if required. Indeed, the spacecraft may need more ΔV than expected because of the disturbing forces that may arise during the mission as explained earlier. A safe margin will ensure to have enough ΔV to counteract the disturbing forces and torques. The total ΔV estimated in last report [1] was 2,185m/s, a summary of this analysis is given in Table 2.2. Please consult the previous report in case more details are required.

Table 2.2: Summary of the total ΔV estimation

Maneuver	$\boldsymbol{\Delta V [m/s]}$
Transfer from Heliocentric to Jupiter including Gravity assists based on previous missions	1300
Initial Jupiter insertion	450
Station keeping and attitude control over 12 year mission life	$12 \cdot 30 = 360$
De-orbiting	75
Total	**2185**

This budget will be refined using the newly available information about the disturbing forces from subsection 2.2.2 assuming that those are constant during the relevant mission phases. The following subsubsections will assess the impulse generated by each significant disturbance and a total ΔV accounting for those will be computed.

Solar radiation ΔV

The JUICE orbiter will experience five years of $720.33\mu N$ and twelve years of $26.6\mu N$ because of the solar radiation. Equation 2.4 gives the relation to obtain the impulse generated by the disturbing forces.

$$I_{tot} = F \cdot t \tag{2.4}$$

This yields a total impulse of $I_{tot} = 123,661.481 Ns$ for the solar radiation effect.

Magnetic torque ΔV

The magnetic torque experienced by the spacecraft is expected to be about $0.06Nm$ in the elliptical orbit around Jupiter (twelve years), for the worst case scenario. Moreover, to get the total impulse from the magnetic torque, it should first be converted to a constant force. This can be calculated from Equation 2.5 using the radius of the counter-thrusters from the center-point of the spacecraft (1.5 m):

$$T = r \times F \tag{2.5}$$

This yields a force of $2.78 \cdot 10^5$N. Using Equation 2.4, the total impulse is then estimated to be about 10,519.52 Ns.

Revision M_e

In order to make the propellant mass calculation as accurate as possible, newly found masses, from next sections, will be used for the M_e. The ΔV itself will also be influenced by the mass, therefore, the calculated M_e from the mass budget will be used.

Total disturbing forces ΔV

Now that the the total impulse from the disturbing forces was calculated, the team decides to replace the "Station keeping and attitude control over 17 year mission life" with the upcoming ΔV. Since, that was intended to account for the disturbances, and thus may be replaced. Normally a safety margin is added, however as the calculations assume constant forces, which is not the case, and are worst case scenarios. Therefore, a margin has already been introduced during the calculations. The average mass can be calculated using the M_e retrieved from the mass budget of Table 5.1 and the iteration in Section 4.5.4. Equation 2.6 permits to compute the additional ΔV required using $I_{tot} = I_{solar} + I_{magnetic} = 134,181Ns$ and $M_{avg} = \frac{M_p}{2} + M_e = \frac{1112}{2} + 1196.26 = 1752.26kg$.

$$\Delta V = \frac{I_{tot}}{M_{avg}} \tag{2.6}$$

This yields a ΔV of 76.576m/s. Hence, the new summary of ΔV is given in Table 2.3.

Table 2.3: Summary of the total ΔV estimation

Maneuver	$\boldsymbol{\Delta V [m/s]}$
Transfer from Heliocentric to Jupiter including Gravity assists based on previous missions	1300
Initial Jupiter insertion	450
Station keeping and attitude control against disturbing forces	76.6
Margin for disturbing forces (50%)	38.3
Deorbiting	75
Total	**1939.9**

Selection thrusters and propellants

It was decide to use one main large thruster beneath the cylindrical bus of the JUICE orbiter, however further smaller thrusters will be needed to counteract the disturbing forces/torques. This is treated in section 2.1, the spacecraft will have one bi-propellant main propulsion engine using nitrogen tetroxide as an oxider and hydrazine as the fuel. Even tough these propellants are highly toxic and "green" propellants are making their way into the aerospace industry, both propellants are more (cost-) efficient than their "greener" options[5]. For example, the specific impulse reached by the chosen propellant is very high compared to the "green" version, which is rather important as the orbiter is using a high-thrust strategy. Moreover, chemical propulsion was chosen over an ion engine, since the orbiter will be relatively far positioned from the Sun, so it will be less affected by solar radiation.

2.2.4 Total Propellant Mass

For the propellant mass to be calculated the empty weight is needed, which is computed in the mass budgets of this report. So, the propellant mass of the JUICE orbiter will not have to be revised again during the mass budgets of this work package. To obtain the propellant mass, the rocket equation (2.7) is required:

$$M_p = M_e e^{\frac{\Delta v}{I_{sp} \cdot g_0}} - M_e \tag{2.7}$$

The newly calculated ΔV and the estimated impulse of $300s$ are also needed, besides the already used M_e.

$$M_p = 1196.26 \cdot e^{\frac{1939.9}{300 \cdot 9.80665}} - 1196.26 = 1116.83kg$$

Therefore the mass of the propellants onboard of the spacecraft will be $1044.8kg$, this estimation already includes a margin. Since the ΔV- margin was rather large, which will thus account for the propellant one as well.

2.2.5 Components of the Propulsion System

To conclude the propulsion system, all components ought to be listed and characterized to give a clear overview of the whole system:

- **Main engine**: The main engine of the propulsion system is responsible for creating the propulsion from propellants. The LEROS 1b engine was chosen as the main engine, since it has been successful for the (comparable) Juno orbiter and because of the $318s$ specific impulse delivered by this engine.[6] This is fairly close to the estimated impulse of $300s$, thus the engine fits nicely in the JUICE orbiter and gives some further margin.
- **Debris shield**: As the engine is very valuable for the mission, to get the spacecraft into orbit around Jupiter, it should be protected when not in use. Therefore, a debris shield is incorporated into the design, to ensure that the engine is safe during the total mission.
- **Main thruster**: The main larger thruster will be positioned underneath the cylindrical bus of the spacecraft. It is of paramount importance that the thruster is pointed away from all components of the orbiter, so the hot exhaust will not damage the spacecraft. Furthermore, it ought to be placed in the centre of mass, in order to avoid creating any torque.
- **RCS thrusters**: Twelve smaller Reaction Control System thrusters will be placed around the bus of the spacecraft, these will counteract any of the disturbing forces described in subsection 2.2.2, as well as attitude control. The JUICE orbiter needs twelve thrusters, since attitude control needs two thrusters per axis, of which there are three, and then twice that amount for the opposite direction.
- **Heat shield**: A heat shield will be incorporated into the propulsion system to pretect the rest of the spacecraft from the thermal radiation created by the thrusters.
- **Helium tank**: The helium tank is responsible for the pressurization of the JUICE orbiter's propulsion.

2.2.6 Sketch Propulsion Components

As can be deduced from the sketch below, there is one main thruster directly connected to the Leros 1b engine. As well as the three spherical tanks with a helium tank on top, from which two are fuel and one oxidizer. Furthermore, there are twelve RCS thrusters powered by the engine, of which only six can be seen because of perspective. Lastly, there is also a debris shield to protect the engine during the mission.

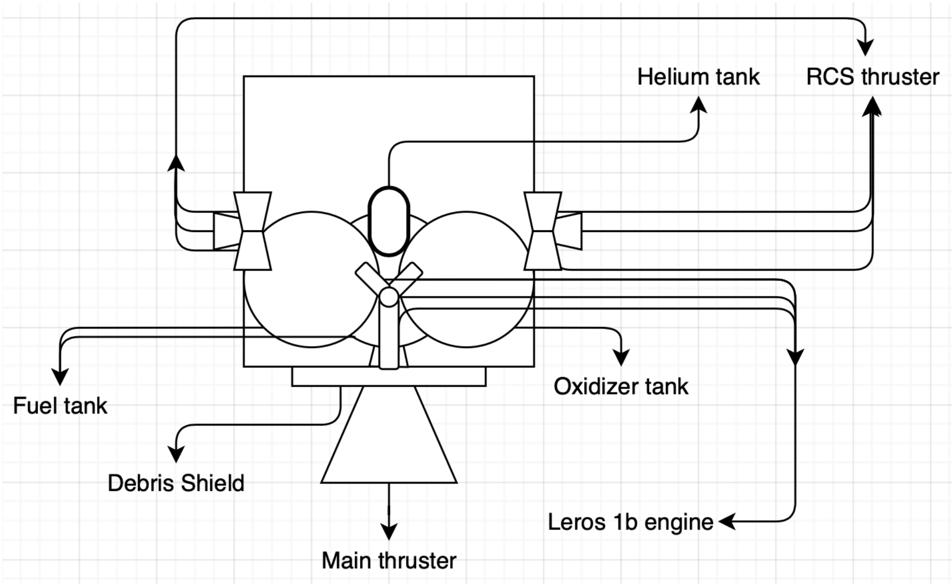

Figure 2.2: A general sketch of the propulsion system

2.3 Thermal Control

This section will cover the preliminary design of the thermal control of the spacecraft. First, the requirements of the subsystem are defined, next, the thermal environment will be defined and the heat generated in the spacecraft will be calculated. With this information the equilibrium temperatures can be calculated and a passive thermal control will be designed. From the analysis performed and the materials available it is concluded that the passive thermal control cannot satisfy the requirements. Active means need be taken into account to meet the requirements and an active thermal control is designed.

2.3.1 Requirements

The main purpose of the thermal control subsystem is to protect the instruments from the harsh environment by managing the temperature of the spacecraft and its instruments. To get a view of the important parameters the requirements and the functions of the thermal control are listed here:

- THERMAL1.1: *The temperature of the components shall be managed such that the temperatures of the components will be within the ranges given in Table 2.4 respectively.* This is mainly to prevent overheating or under cooling of equipment, since most of the instrumentation, such as batteries or propellants, are designed to operate at Earth's temperature and cannot function properly at very high or low temperatures.
- THERMAL1.2: *The temperature gradient over the components shall be managed such that these lay within the bounds given by Table 2.5 for their respective components.* This is to keep the instruments accurate. A difference in temperature over a component creates a gradient, which alters the geometry of the instrument. For instruments that need to be very accurate this can become a problem.

- THERMAL1.3: *The change in temperature shall be managed such that the difference of temperature of the respective component will fall within the ranges given by Table 2.5.* It is important to keep the temperature constant, because the scientific equipment of the payload must stay accurate and should not get damaged during the mission. Abrupt temperature changes can lead to distortion of instruments and calibration errors.

Table 2.4 lists the temperature ranges of the main spacecraft components.

Table 2.4: Temperature tolerance of main spacecraft components [2]

S/C component	T_{min} ($^{\circ}C$)	T_{max} ($^{\circ}C$)	**ΔT** ($^{\circ}C$)
Batteries	0	+20	20
Solar Arrays	-105	+110	215
Sensors	-30	+50	80
Thrusters	+7	+65	58
Mechanisms	0	+50	50
On board computers	-10	+50	60
Transponders/transmitters/receivers	-20	+60	80

Table 2.5 lists the allowed temperature gradients and variations of critical components that are sensitive to temperature gradients and variations.

Table 2.5: Typical allowed temperature gradients and variations [2]

S/C component	**-**	**Magnitude**
Across optical instrument (1.5 m)	ΔT	$< 5^{\circ}C$
Structural elements	$\Delta T/m$	$< 2^{\circ}C/m$
Between MMH and NTO tanks	ΔT	$< 5^{\circ}C$
For typical electronic unit	$\Delta T/\Delta t$	$< 5K/h$
For CCD camera	$\Delta T/\Delta t$	$< 0.1K/mn$
For cryogenic telescope	$\Delta T/\Delta t$	$< 100\mu K/mn$

2.3.2 Thermal Environment

To study and manage the temperature of the spacecraft it is required to know more about the environment of the mission, specifically how much heat is absorbed by the spacecraft. This comes with the knowledge that, in space, there are three sources of heat energy in total: direct radiation from the sun, albedo radiation and planetary radiation. Prior to the mission the orbiter has to travel to Jupiter and it will not be deployed before it reaches the planet. It will be carried by a kick stage and will be protected by a separate thermal control, therefore, it is not necessary to design thermal control for the mission phases prior to its arrival in Jupiter's orbit.

Orbit

Before the different types of power can be discussed in detail, it is important to know more about orbit about Jupiter. The spacecraft will orbit Jupiter for seven to twelve years and a sun-synchronous orbit was chosen. This means the spacecraft will be orbiting the planet with an inclination of 90° with respect to the incoming Sun radiation. Together with the fact that the distance between Jupiter and the Sun is nearly constant it translates into a constant solar flux throughout the whole orbit. At Jupiter the solar flux is 50.26 W/m^2 [7].

As depicted in figure 2.3 the orbit is highly elliptical, meaning that the distance to the planet will differ greatly. Therefore the spacecraft will experience great variations in albedo and planetary flux.

However the area where this flux radiates on can be assumed constant as the bottom of the spacecraft will nearly always be pointed towards Jupiter. From this information it can be determined that the most extreme environments are the pericenter, receiving maximum heat, and the apocenter, receiving the minimum amount of heat.

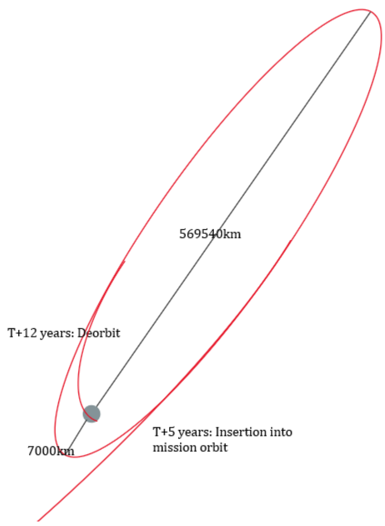

Figure 2.3: Orbit around Jupiter

From the distance between the orbit and the Sun, the solar flux could be determined. Next the albedo and the planetary flux will be covered. The exact value for the albedo flux can be calculated with Formula 2.8 and 2.9 [2].

$$J_a = a * J_s * F_a \qquad (2.8) \qquad\qquad F_a = \left(\frac{R_{planet}}{h_{orbit}}\right)^2 \qquad (2.9)$$

The albedo factor is 0.52 [2] and the visibility factor F lies between 0 and 1. It is 0 if the spacecraft is in the shadow of the the planet or can be calculated using Formula 2.9 when it is far away from the planet if $h_{orbit} > R_{planet}$, otherwise it's just 1 if $h_{orbit} < R_{planet}$. Another effect of the sun-synchronus orbit is that the orbiter will not be on the sunlit side or in the shadow, but in between. Half of the sunlit side is visible from the orbit and it assumed that half of the flux will reach the orbiter compared to when it would be in the sunlit side. To correct Formula 2.8 half of this flux should be taken as the albedo flux caught by the spacecraft. In the first case, using formula 2.8 and an orbit height of 7000km leading to a visibility factor of 1, albedo radiation has been calculated being $26.1 W/m^2$. At the apocenter, where the orbit height is 569540.6km, the visibility factor is 0.015 and the albedo radiation becomes $0.394 W/m^2$.

A different source of energy is planetary flux. Because the surface of the planet has a temperature it acts as a black body radiator. Part of this energy is absorbed by the orbiter and the amount depends on the temperature of the surface. The surface of Jupiter has an effective radiating temperature of 165.15 K [8]. Using the equations 2.10, 2.11, 2.12 from [2]:

$$Q_{IR} = \sigma \cdot T_{IR}^4 \qquad (2.10) \qquad\qquad P_j = Q_{IR} \cdot 4\pi \cdot R_j^2 \qquad (2.11) \qquad\qquad J_{IR} = \frac{P_j}{4\pi \cdot d^2} \qquad (2.12)$$

Gives formula 2.13:

$$J_{IR} = \frac{\sigma \cdot T_{IR}^4 \cdot R_j^2}{d^2} \qquad (2.13)$$

Where d = distance of the spacecraft to the center of Jupiter.
Filling in the values and using 7000km as it the closest point in the orbit around Jupiter, a planetary flux of $6.74 W/m^2$ was found. This is the closest distance to Jupiter. When the spacecraft reaches the apocenter, the distance from Jupiter increases greatly and the planetary flux reduces to $0.0974 W/m^2$.

2.3.3 Internal Heat

Next to the heat caught by the spacecraft from the environment heat will also be generated by the components. The components use electrical power and this power is eventually converted into heat. The total power used by the spacecraft is 263, 43 W. All of this power is assumed to be dissipated into the body. Next to this heat is also generated in the cables that transport this power, which is also assumed to dissipated into the body.

2.3.4 Equilibrium Temperature

The temperature will differ during the orbit about Jupiter. However to make sure that the payload is safe the temperature control should be able to manage the most extreme environments. This includes the apocenter and pericenter of the orbit. Here is when the least and the most heat is coming in respectively.

When the incoming flux changes, the temperature will start to change. In order to find the hottest and coldest temperatures of the spacecraft a minute by minute model is needed. This makes it really complicated to solve for the exact temperatures of the spacecraft. To simplify the problem the model is limited to the extremes, with this a range can be set up. These are the temperatures the body reaches in the coldest and hottest environment after a very long time, so when the equilibrium temperature is reached the power emitted is the same as the power received. With this information, and the ones about the spacecraft and the thermal environment these temperatures can be computed using the following Equation 2.14:

$$\dot{Q}_{in} = \dot{Q}_{out} = \dot{Q}_{absorbed} + \sum P_{dissipated} = \dot{Q}_{emitted} \tag{2.14}$$

As mentioned earlier the power dissipated is 263.43 W. The power that is absorbed consists of the solar-, albedo- and planetary flux. The mathematical formulation of this is given by Equation 2.15:

$$\dot{Q}_{absorbed} = \alpha_s \cdot J_s \cdot A_s + \alpha_a \cdot J_a \cdot A_i + \alpha_{IR} \cdot J_{IR} \cdot A_i \tag{2.15}$$

The area, which each source of power radiates on, depends on the orientation of the spacecraft and therefore the orientation of the instruments. The instruments that make measurements of Jupiter, such as the camera, are placed on the bottom. This means that the bottom of the spacecraft will be aimed towards Jupiter. Next to this the solar arrays are on the side of the body and are pointing towards the Sun. Therefore the Sun radiates on the side of the body, the inside of the antenna and the solar arrays. The antenna is assumed to be a semi-circle with a radius of 1.25m. The solar arrays are connected via a rod and the energy transfer between the arrays and the body is neglected as mentioned in 2.3.3. From the sketches of work package 1 ([1]) it is determined that $A_s = 11.5m^2$, $A_a = 7.1m^2$ and $A_{IR} = 7.1m^2$. Equation 2.16 will then be used.

$$\dot{Q}_{out} = \sigma \cdot A_e \cdot \epsilon_{IR} \cdot T^4 \tag{2.16}$$

The area that emits heat is separated in different parts. The body of the spacecraft, the antenna and the solar arrays. The body is a cylinder with a diameter of 3 m. Secondly, the antenna is assumed to be a semi-sphere with a radius of 1.25 m. The antenna is positioned at the top of the spacecraft, so the top of the body will radiate heat which is caught by the antenna and vice versa. However this amount of heat is very small and is therefore neglected. Next to this the antenna is assumed to have the same temperature as the body. Finally, the solar arrays are assumed to be thin plates. From this the equilibrium temperatures can be calculated. An equilibrium temperature of the solar arrays is found to be 137.7 K and the equilibrium temperature of the body is found to range between 148.4 and 146.9 K. Due to relatively small changes in incoming power the difference between the equilibrium temperatures is small. Namely 1.5 K. However the temperature of the arrays and body is not the optimum case. They have no protection, no insulation is applied and no other methods of managing the temperature of the spacecraft are applied. Before real calculations can be made it is important to look at ways to manage the temperature of the spacecraft.

2.3.5 Passive Thermal Control

Passive thermal control allows to control the temperature of the spacecraft without the use of electrically powered equipment. This is possible using materials and devices that manage the heat flowing in and out of the spacecraft. An overview of the different methods of passive thermal control in given in the diagram below, Figure 2.4.

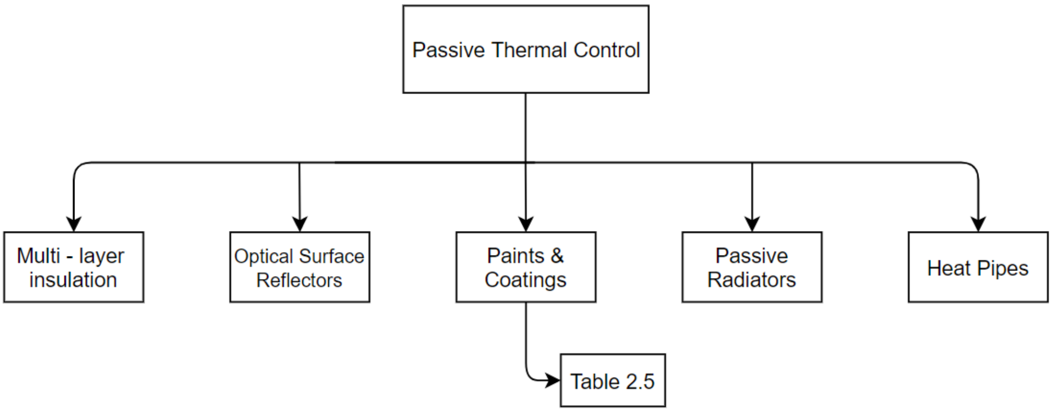

Figure 2.4: Main passive thermal control methods

To select the best way to control the temperature of the spacecraft, some calculations have to be performed. Using equations 2.14 and 2.16 combined, the temperature of the spacecraft body can be calculated with Equation 2.17:

$$T = \left[\frac{\alpha \cdot (J_s \cdot A_s + J_a \cdot A_i) + \epsilon \cdot J_{IR} \cdot A_i}{\sigma \cdot A_e \cdot \epsilon} \right]^{0.25} \tag{2.17}$$

Where α and ϵ are the absorptivity and emissivity respectively. These values are determined by the material on the outside of the spacecraft. In Table 2.6 the absorptivity and the emissivity of commonly used materials are given. In the last column the fraction is also given, the higher this fraction the higher the temperature the spacecraft will have. From this fraction it becomes clear that Aluminium klapton (metal side) will be the most efficient in keeping heat in the spacecraft. This is important, because we are far away from the sun and the equilibrium temperature of the body will be very low. Two different temperatures can be calculated, for the apocenter and the pericenter. The difference of the two, ΔT, is also important. The lower this difference is, the easier it is to keep the temperature consistent. This is important for on-board instrumentation which requires a low temperature gradient. The equilibrium temperature, also needs to be taken into account as the spacecraft will be far away from the Sun, immersed in a very cold environment.

Table 2.6: Materials for passive thermal control [2]

Material	**Absorptivity**	**Emissivity**	α/ϵ
Black paint	0.96	0.75 - 0.88	1.2
Aluminium teflon foil	0.14	0.78	0.18
Silver teflon foil	0.09	0.8	0.11
Aluminium kapton foil	0.42	0.72	0.58
Aluminium kapton metal side	0.17	0.82	2.4
White paint	0.17 - 0.38	0.82	0.36
OSR	0.09	0.76	0.12
Solar cell Si	0.75	0.82	0.91
Solar cell GaAs	0.91	0.81	0.12
CFRP	0.92	0.82	0.12

After performing multiple calculations with different materials, the following design was implemented. The spacecraft body is going to be covered in two different materials, multi-layer insulation (MLI) is going to be used for the side of the spacecraft that receives no radiation, both from the Sun and from Jupiter, and the rest of the body is going to be covered in aluminum kapton. This is because MLI has a very low emissivity, in the range 0.003 - 0.03 [2]. This will significantly reduce the heat loss for the back of the spacecraft. Aluminum kapton on the other hand is the best choice for obtaining a high equilibrium temperature, since it has the highest α/ϵ, absorbing 2.4 times more heat than it emits. This thermal design for the body gives it an equilibrium temperature that ranges from 167.8 to 166.1 K. This is a problem, because the components within the spacecraft require a higher temperature to operate (see Table 2.4). To solve this these components are thoroughly insulated and are heated up with their own dissipated power. With this a sensor would be needed and the temperature could be lowered using louvre's or by shutting the instruments down. Shutting down an instrument is not desired and has to be prevented by the louvre. This solution however is only viable for these specific components listed, not for the thrusters and the antenna. For this a different solution has to be found.

Next to this is the design for the solar arrays. In Section 2.14 it was determined that the equilibrium temperature is to low. To get a higher temperature MLI is applied on the back of the arrays. This raises the temperature to 163.7 K. This is also to low. A way to increase the temperature is to add heat pipes between the body and the arrays. However the temperature of the body is not high enough to raise the temperature of the arrays to its operating range. A different solution has to be found for the arrays aswell.

2.3.6 Thermal Control

The temperature of the arrays, antenna and thrusters are lower than the operating ranges given in Table 2.4. This is a big problem, because this means the spacecraft can not operate properly. To solve this active control thermal control is needed. As a last resort electrical systems are used to raise the temperatures of these components. First the thermal control for the solar arrays is improved, then the antenna and thrusters.

The solar arrays as mentioned earlier are thermally decoupled. So the only heat coming in is coming from the sun. However part of this energy is converted into electrical energy, this energy can also be used to heat up the solar arrays. From Section 2.4 it is known that the solar arrays generate 1331.1W of electrical power at the start of the mission. This value drops to 263.43W at the end of the mission. This amount of power is the maximum power that can be required at once by the spacecraft. So at the start of the mission 1067.7 extra watts are generated than actually is required from the solar arrays. This power can be used to heat up the arrays to a high enough equilibrium temperature of 175 K. This means at the start of the mission the solar arrays can operate within the temperature range given by Table 2.4. The only problem is that the solar panels degrade and not enough electrical power is generated to heat up the spacecraft. However because it degrades this energy is only not converted into electrical power. This means this power is still caught by the solar array and is directly converted into heat. So the solar arrays stay at the exact same temperature during the entire mission. The thermal control for the solar arrays is summarized in Figure 2.5.

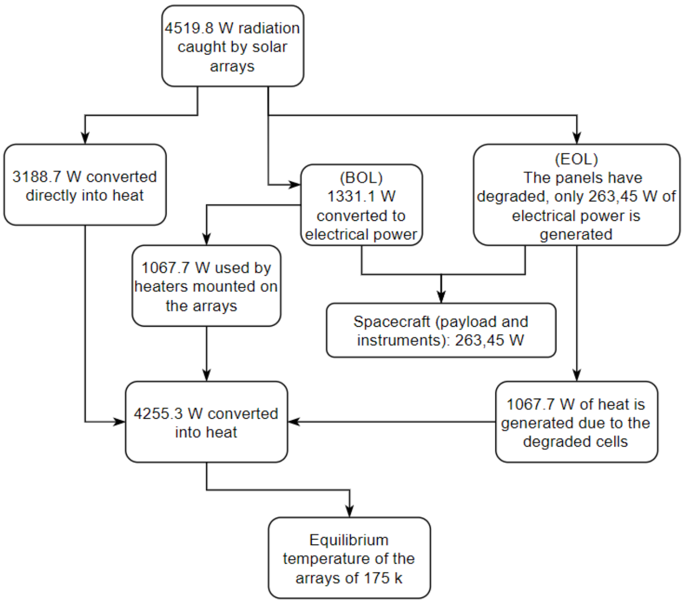

Figure 2.5: Thermal control solar arrays

To be more efficient the antenna is thermally decoupled. This is because the antenna needs to have a high temperature, but the body itself does not. By thermally decoupling it power, weight and money can be saved. The back of the antenna is covered with MLI and will have 5 RHS units, together they generate 1000 W. This is enough to heat the antenna up to 257 K. Batteries, sensors, mechanisms and on board computers will be thoroughly isolated using MLI and their own dissipated power reach their operating temperatures. With a sensor and a louvre the temperature of these components can be controlled to be within the ranges given in Table 2.4. This is done by a computer and costs almost no power. Finally the thrusters will have heating units that can heat the thrusters to 280 K. This is only done when they need to be fired. They will not need to fired often, just to correct the orbit. The material of the battery which can hold 778.8 Wh is considered enough. The body will have a maximum temperature of 167.9 K and a minimum temperature of 166.2 K. The relatively small difference makes it easy to keep the temperature of sensitive part stable.

2.4 Power Subsystem

This section covers the power subsystem of the JUICE spacecraft. This subsystem being vital for the mission, a careful analysis will be performed in order to determine its main characteristics. The structure of the section is as follows. First, the key requirements and the amount of power required during all mission phases are presented in subsection 2.4.1. The different available power generation sources are then explored in subsection 2.4.2 and one from those is selected in 2.4.3, a preliminary dimensioning follows from this choice. Subsection 2.4.4 will approach the different characteristics of the spacecraft batteries and a specific one from the market will then be chosen in subsection 2.4.5, in this same section the power control components will also be selected. Finally, subsection 2.4.6 will present an overview of the complete subsystem.

2.4.1 Requirements of the Power Generation Subsystem

First, the key requirements of this subsystem are listed and a rationale for each will be given. As an information for one of the main requirements, the amount of power required for each mission phase is given. The list of the requirements is given here, note that some values in the requirements come from the next subsections.

- POWER1.1: *The power generation unit (when active) shall produce at least 263.4W to operate the spacecraft safely and to fill the batteries (if any).* This becomes important in case the power generation source is not constantly available, such as it is the case with solar panels and an orbit having an eclipse time. In that case, the power generation unit should produce enough power to operate the spacecraft and fill the batteries for the next time interval where the power source is not available.

- POWER1.2: *The power generation unit shall generate at least 263.4W of power during the complete mission.* The mission is planned to be lasting from seven to twelve years but during that time the power generation unit will degrade and hence produce less power than at the beginning of the mission. It is thus important to take the possible degradation into account during the design to make sure that enough power will be produced at all times.

- POWER1.3: *The power generation unit should not harm the rest of the spacecraft components.* Extra care should be taken when using a unit such as a RTG as it produces non-negligible amounnts of radiation that could put the mission in danger.

- POWER1.4: *The batteries shall provide at least TBDW during the time the primary source of power is not available.* In case the primary source of power generation is not constantly available, the batteries (secondary source of power) should fulfill the needs of the spacecraft in terms of power.

- POWER1.5: *The power subsystem shall control and distribute the energy to all the other subsystems.* A component in charge of the distribution and the control of the power shall be present to permit a proper handling of the energy available.

- POWER1.6: *The power subsystem shall provide the required amount of power to the spacecraft during all mission phases.* The different phases of the mission will require different amount of power to be provided by the spacecraft because of the various functions that need to be fulfilled at different times. Table 2.7 gives the values of power required for each phase of the mission, it was computed using a 30% margin on all subsystems given in Table B.2. When the payload is operational, the estimated payload power from section 2.6 is used with a 30% margin.

Table 2.7: Power Required for each Mission Phase

Mission phase	Operational subsystem	Power required $[W]$
Launch	None	0
Heliocentric orbit	TT&C, Thermal, EPS, ADCS, CDH, Propulsion	167.0-200.9
Hohmann transfer orbit	TT&C, Thermal, EPS, ADCS, CDH, Propulsion	167.0-200.9
Distance from Jupiter $\leq 10,000km$	All subsystems	249-263.4
Distance from Jupiter $\geq 10,000km$	TT&C, Thermal, EPS, ADCS, CDH Propulsion, Magnetometer, Spectrometer	222.1-236.5

2.4.2 Energy Sources Available during the Mission

Different power generation sources are available for a deep space mission. In this subsection, the different energy sources will briefly be introduced and their respective (dis)advantages will be presented. Then, a comparative table based on the mass and size of the power system required for each power source is given to assist the choice of the energy source for the JUICE mission.

- **Solar Energy** is mostly used in space probes when the spacecraft is to stay at a reasonable distance from the Sun, this most importantly means not too far from the star as the solar irradiance drops by the square of the distance. The solar flux around Jupiter is about $50.3W/m^2$ which is low compared to the $1360W/m^2$ that the satellites orbiting the Earth receive. The Juno mission has shown that using solar panels for a mission in Jupiter's orbit is doable although those need to be quite large. It is also important to note that photovoltaic systems typically have a rather low efficiency of about 20-30% and they need to be oriented perpendicular to the sun rays for maximum efficiency. Another disadvantage of the use of solar energy is that it becomes unavailable when the spacecraft is in eclipse, but on another side, solar energy is a "free" energy that is there for use and can be a cheaper alternative to other sources. Lastly, the solar panels will degrade over time because of its environment. [2]

- **Radioisotope thermoelectric generator** or RTG are often used for deep space probes going to the outer-most places of our solar system and beyond, where the solar irradiance is so small that it is impossible to use solar energy. This source requires to take the energy on-board of the spacecraft and is designed for the peak power as well as the total energy, which means that it is also constantly operational since it does not depend on the environment the spacecraft is in. No secondary source of power is required. A downside of the RTG is that it does emit radiation that could harm the rest of the spacecraft and hence further structural considerations need to be taken to position the RTG in a way to protect the other subsystems. Furthermore, RTG often rely on the use of 238-Plutonium that is rather expensive: about $8.82M\$/kg$ FM2015 [9]. Lastly, the amount of power generated will decrease over time because of the natural decay of the particles and an RTG generates quite some heat in a general manner. [2]

- **Hydrogen fuel cells** are designed for the peak power and the total energy need (as the RTG), the reason is that it is wanted to have a constant power output throughout the mission. A downside of such design though is that the excess power generated should be dissipated using heaters but still avoiding any interference with the different sensors and other parts of the spacecraft. Another disadvantage, again, is that the energy should be taken on-board which costs mass and volume. [2]

- **Lithium ion batteries** can be taken on-board as a mean of power generation. This is often used in launchers since their operational life is quite small and most lithium batteries have a specific energy ranging from 300 to $550Wh/kg$ and an energy density between 600 and $1000Wh/L$. This means that the total mass and volume of the power generation unit can quickly get very large. [2]

The qualitative point of view of the different power sources was presented and a quantitative comparison based on mass, size and need for a secondary power source of the power generation unit (not the complete EPS subsystem) is now given in Table 2.8. For comparison, the 263.4W of peak power requirement from the mission requirements is used for input of the following table.

Table 2.8: Power source comparison

Power source	Minimum mass [kg]	Size	Secondary source
Solar Energy [1]	43.4	$27m^2$	Yes
RTG [2]	55	-	No
Hydrogen Fuel Cells [3]	13845.2	18.5 m^3	No
Lithium Ion Batteries [4]	55377.2	27.7 m^3	No

Please, note that this table is given only for comparison and more precise dimensioning will be performed in subsection 2.4.3. It is important to keep in mind that this comparison is performed neglecting any kind of degradation. Most likely, the mass and size of solar panels would greatly increase while the mass of the RTG would increase by a much less factor.

2.4.3 Chosen Primary Power Source and Dimensioning

In this section, the primary source of power will be chosen and an explanation therefore will be provided. Then the dimensioning of the power generation units will be presented.

Primary Power Source Dimensioning

Based on the discussion from subsection 2.4.2, it was decided to use solar panels as they offer a 'free' power source, do not risk to harm other parts of the spacecraft with radiation and are deployable devices. This means that the volume can be reduced to a minimum to fit into the launcher. The orbit previously chosen is a polar sun-synchronous one, meaning that it presumably has no eclipse time but a battery is still required for reasons of safety in case the orbit would need to be revised after launch and an eclipse would be present. This choice is also motivated by the large history of spacecrafts using solar energy as a lot of documentation is thus available and different technologies can be investigated. Figure 2.6 shows that photovoltaic panels can be used for the JUICE mission according to literature. Indeed, the mission duration shall be from seven to twelve years and the estimated peak power is 263.4W. Lastly, solar panels were preferred over a RTG in terms of cost and the batteries or fuel cells were no option because of their estimated mass and size.

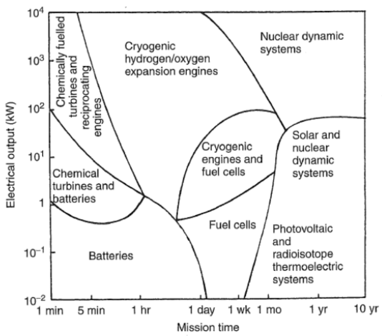

Figure 2.6: Primary source of power choice [2]

[1]Assumptions: perpendicular to panel, efficiency: 19.4%, specific power: $164.25W/kg$@1AU, solar irradiance @Jupiter: $50.26W/m^2$, no eclipse time(polar sun-synchronous orbit), neglecting degradation for now. All the data comes from [2]

[2]From one of the most recently designed system: the General Purpose Heat Source RTG. All the data comes from [2]

[3]Assumptions: density: $750kg/m^3$, specific energy: 0.5kg/kWh. All the data comes from [2]

[4]Assumptions (Best case scenario): efficiency: 95%, specific energy: $500Wh/kg$, energy density: $1000Wh/L$. The energy is given by the peak power time total mission time (neglecting transfer). All the data comes from [2]

Dimensioning of the Solar Panels

In order to dimension the solar panels used, a specific technology should be selected. Table 2.9 gives an overview of some different technologies available. Note that an assumed constant solar irradiance of $50.26W/m^2$ for the orbit around Jupiter is used for further computations.

Table 2.9: Available solar panel technologies [2]

Technology	Specific Power (BOL) [W/kg] [5] @ Efficiency [%]	Cost @ 1AU [$k\$/W$]
HES rigid panel	2.16 @ 19	0.5-1.5
HES flexible array	4.22 @ 19	1.0-2.0
TJ GaAs rigid	2.59 @ 26.8	0.5-1.5
TJ GaAs ultraflex	4.25 @ 26.8	1.0-2.0
CIGS thin film	10.17 @ 11	0.1-0.3
Amorphous-Si MJ/thin film	13.05 @ 14	0.05-0.3

The degradation of the solar panels is assumed to be like during the Juno's mission. This provides a conservative approximation that the degradation would be of about 13.5% per year [10]. Using all this information and the peak power, the dimensioning of the solar panel can be performed. A conservative approach under the assumption that the peak power should be generated at all times. This provides the possibility to refill the batteries during all parts of the orbit (always in sunlight) which do not actually need the peak power. The degradation factor will be taken into account over a mission duration of twelve years using Equation 2.18.

$$P_{BOL} = \frac{P_{EOL}}{e^{(-degradationfactor \cdot t)}} \tag{2.18}$$

Here, P_{EOL} is equal to the peak power, as mentioned earlier. In this way, the odds that the spacecraft will have enough power to operate safely after twelve years are high. Using the data in Table 2.9, it is then possible for each technology to compute the mass of the solar panels with Equation 2.19.

$$M_{SP} = \frac{P_{BOL}}{P_{spec}} \tag{2.19}$$

The total solar panel area can also be computed using the efficiency, the solar irradiance and the required power (here taken as the peak power). The relation between those parameters is given in Equation 2.20.

$$A_{SP} = \frac{P_{BOL}}{J_s \cdot \eta} \tag{2.20}$$

Using the method outlined above with the maximum peak power of 263.43W to ensure a conservative approach, the required power at the be begin of life is estimated to be 1331.1W. In Table 2.10 the information that will be used to determine the technology to be used is presented.

Table 2.10: Comparison of solar panel technologies

Technology	Mass [kg]	Area [m^2]	Maximum total cost [$M\$$]
HES rigid panel	616.3	139.4	53.99
HES flexible array	315.4	139.4	71.99
TJ GaAs rigid	514.0	98.8	53.99
TJ GaAs ultraflex	313.2	98.8	71.99
CIGS thin film	130.9	240.8	10.80
Amorphous-Si MJ/thin film	102.0	189.2	10.80

[5] The table[2] only provides information for 1AU, the data given here was computed by dividing the data found by $(5.2)^2$ since Jupiter is on average at 5.2AU from the sun

A trade off between area and mass should be performed using Table 2.10 since both should be minimized. On the one hand, the size of the solar panels should be small as MMOI directly impacts the amount of propellant needed for maneuvers. On the other hand, the mass should be kept minimum as sending anything to space is expensive. Having those ideas in mind, it is noted that the solar panels can (un)deploy, thus reducing the MMOI within a few minutes. That means the solar panels can be retracted prior to any maneuver to spare propellant mass, but the amount of hinges necessary for the solar panels to (un)deploy is a source of failure. Too many of those is putting the mission in jeopardy. Using the TJ GaAs ultraflex solar panels permits to have a relatively low mass and area for a reasonable cost. In Table 2.11, the final technology of the solar panels can be found.

Table 2.11: Characteristics of the solar panel technology used

Technology	**Mass [kg]**	**Area [m^2]**	**Cost $M\$$**	**Efficicency**
TJ GaAs ultraflex	313.2	98.8	71.99	26.8%

2.4.4 Main Characteristics of S/C Batteries

In this subsection, the primary sizing of the batteries of the spacecraft will be performed. Batteries are often used to provide energy to a spacecraft in case of an eclipse and to support the primary source of power during the peak power periods. In the case of the JUICE mission, a polar sun-synchronous orbit is used meaning that the S/C should always stay in sunlight and hence no eclipse time occurs. In theory, the solar panels are able to provide all the power required at all times but a battery should still be present in case an unexpected event occurs. A conservative approach would be to estimate the eclipse time in case the same orbit was used in the orbital plane of the planet, this was done in Appendix C. Here, only the result is used: the eclipse time for the same elliptical orbit in the orbital plane of the planet has an eclipse time of about 44.35 minutes. Equation 2.21 presents how to determine the total battery size.

$$E_{BAT} = \frac{P_{peak} \cdot T_e}{\mu_{BAT} \cdot DOD} \tag{2.21}$$

Lithium-Ion batteries are mostly used. Such batteries were also present on-board of the Juno mission[11], this shows that that they can resist the high amounts of radiation around Jupiter. The DOD can be found by approximating the number of cycles by assuming one cycle per orbit during twelve years. The number of cycles is thus given by $\frac{12 \cdot 365.25}{14} = 312.86$ cycles (with the orbital period of 14 days). Figure 2.7 shows how to estimate the Depth of Discharge to be 100%.

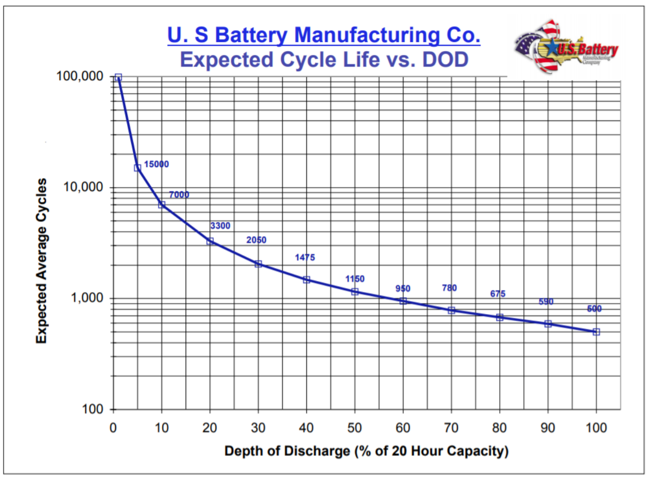

Figure 2.7: [12]

From Equation 2.21 and the information found in [2] and [12], the total energy that should be stored in the battery is found to be 778.8Wh at the most. Using equations 2.22 and 2.23, the volume and mass of the battery can be computed based on the stored power.

$$M_{BAT} = \frac{E_{BAT}}{E_{sp}} \qquad (2.22)$$

$$V_{BAT} = \frac{E_{BAT}}{E_{\rho}} \qquad (2.23)$$

The results of the sizing of the battery are presented in Table 2.12.

Table 2.12: Results of the battery sizing

Battery type	$\boldsymbol{E_\rho}$	$\boldsymbol{E_{sp}}$	**DOD**	**Efficiency**	**Capacity**	**Volume**	**Mass**
Lithium-ion	$321Wh/l$	$133Wh/kg$	100%	90%	$778.8Wh$	$2.43l$	$5.86kg$

2.4.5 Battery and Power Control Components Selection

In this subsection, the models of the battery and the power control components will be given. Those were found on the market in order to cope as most as possible with the needs described earlier. For both, the main parameters were the mass and volume that the component would have for the JUICE mission. The devices were chosen in order to minimise those parameters.

Battery Model

The *SLC-16050*[13] from *EaglePicher*^{tm} Technologies LLC was chosen as the battery for the JUICE mission. It is a Lithium-Ion battery that is was designed for space application in Earth orbit but also for exploratory satellite missions. Its characteristics are given in Figure 2.8.

Part Number	SLC-16050
Cell Design	Lithium-Ion Secondary Cell
Electrical	
Beginning of Life Capacity	62.5 Ah, 245 Wh at 20°C EOCV 4.1 CC/CV, C/2
Nameplate Capacity	52 Ah, 204 Wh at 20°C EOCV 4.1 CC/CV, C/2
Charge Current Limit	31.25A to 4.1 EOCV CC/CV
Discharge Current Limit	62.5A to 3.0 EODV
Pulse Current Limit	125A for 10 seconds
Mechanical	
Weight	2020 g/4.44 lb
Specific Energy	121.3 Whr/kg
Energy Density	305.4 Wh/l
Dimensions (less terminal)	173 x 81.5 x 56.9 mm (6.81 x 3.21 x 2.24 in)
Storage Temperature	-5°C to 5°C (23°F to 41°F)
Operating Temperature	10°C to 30°C (50°F to 86°F)
Vibration Limit	Random 24.7 g
Transportation	Class 9 ID number – UN3090*

Figure 2.8: Characteristics of the SLC-16050 battery

Power Control Component Model

For the JUICE mission, it was decided to use a PCDU instead of both a PCU and PDU. This permits to reduce the total weight of the power control components and to have an all-in device. To do so, the Galileo PCDU[14] was selected since it is compatible with Li-Ion batteries selected earlier and the estimated power of the spacecraft, another important point is the reliability provided which is of 0.983 for 12 years which is prefectly adapted for the mission. The specifications of the Galileo PCDU are given by Figure 2.9.

Dimensions (L x W x H)	508 x 235 x 156 [mm]
Mass	16.3 kg
Reliability 12 years	> 0.983
Regulated bus voltage	50V ± 0.5%
Output power capability, sunlight mode	up to 3000 watt
Output power capability, eclipse mode	up to 2400 watt
Output impedance	< 30 mΩ
Bus voltage transient regulation @ Δ10A	< ± 1%
Idle power consumption	< 28.3 watt
Solar array interface	20 sections of 3 ampere
Transfer efficiency	> 97.8%
Battery interface	Li-Ion, 22.5 to 39 volt
Discharge power	up to 4 x 600 watt
Discharge transfer efficiency	> 95.5% @ 30V
Equipment Power Distribution lines	49
Heater Power Distribution lines	96
Propulsion power lines	29 volt, 80 watt 4 lines
Thermal Knives Actuator lines	2 x 12
DMS Interface	2 x Dual MIL-STD-1553

Figure 2.9: Specifications of the PCDU used on-board of the JUICE mission

2.4.6 Block Diagram of the Subsystem

Figure 2.10 provides the block diagram of the subsystem. The various components that were designed and selected are present, as well as their efficiencies. The power flow is indicated with the direction of the arrow.

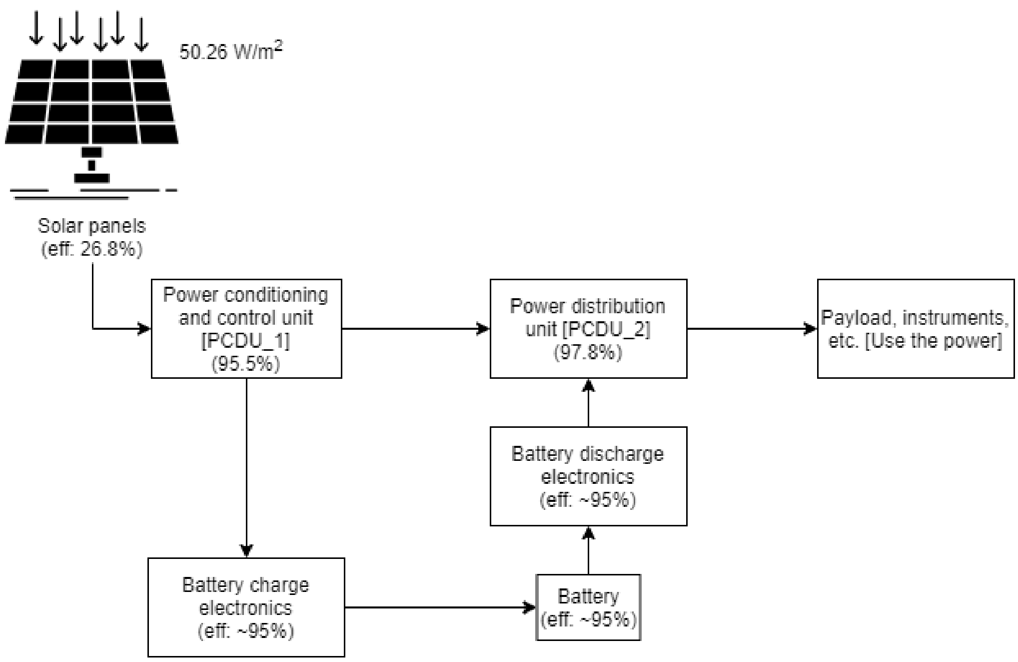

Figure 2.10: Block diagram of the EPS

The PCDU was broken down into its two main functions for clarity of the diagram. The efficiencies provided for the solar panels and the PCDU were found in the data sheets of the components chosen in subsection 2.4.5 and subsection 2.4.3. No such information was found for the batteries and thus typical data was taken from [2].

2.5 Structures and Mechanisms

This section is mainly about the Structures and Mechanisms of the Spacecraft. The section covers the requirements of the subsystems related to the structures and mechanisms, The definition of all loads acting on the spacecraft and the relevant safety factors, preliminary dimensions of the primary structure and the supporting structure for the solar panels will be computed depending on what the structure is designed for. In addition to that the design of the propellant tanks is taken into account. Finally the other mechanisms used in the spacecraft will be noted.

2.5.1 Key Subsystems Requirements

For the Structural requirements:

Handling Loads: The spacecraft should be able to withstand the Loads that are applied on the spacecraft during its mission.

Reduce Structure Mass: The Mass of the structure should be reduced to an optimum mass in order to increase payload mass and at the same time withstand the applied loads.

Material Choice: The material used in the spacecraft should be space compatible and durable.

Reliability and Cost: The materials used in the spacecraft should have a high reliability and a reasonable cost. paragraphLV interfacing:It is advantageous if the Spacecraft has the structural capability to interface with other types of launch vehicles in case a change of choice of launch vehicle occurs.

For the Mechanisms:

Friction: The mechanisms used for Deployment/Pointing/Sensing should have enough torque to counteract the friction forces.

Energy: Energy must be stored or added to keep the mechanism working

Mass, Cost and Reliability: The Mass, Cost and reliability should be reasonable for the Mechanism used

2.5.2 Loads

Launch Loads Table 2.13 indicates the Launch loads on the Spacecraft when using the Atlas V launcher. [15]

Table 2.13: Launch Vehicle Loads on spacecraft

	Steady-state Axial loads	**Steady-state Lateral loads**	**Dynamic Axial loads**	**Dynamics Lateral loads**
Launch	1.6g	0g	2g	2g
Flight Winds	2.4g	0.4g	0.5g	1.6g
Strap-on SRM separation	3g	0g	0.5g	0.5g
Booster engine cut-off end	5.5g	0g	0.5g	1g
Main engine cut-off (max axial)	4.8-0g	0g	0.5g	2g
Main engine cut-off (max lateral)	0g	0g	2g	0.6g

Handling and Transportation loads This table presents the loads on the structure when the structure is handled and transported in a certain way. The S means that the load occurs simultaneously however I means that the load occurs independently [16](citation for whole table).

Table 2.14: Loads During Transportation

Transportation Mode/ Vehicle	Load Occurrence	Fore/Aft g's	Lateral g's	Vertical g's
Water Craft	S	±0.75	±1.0	+2.5, -0.5
NASA Barge (MAF to KSC)	S	±0.75	±1.0	+2.25, -0.25
NASA Barge Inland Waterway	S	±0.5	±0.5	+1.4, +0.6
Airplane	S	±3.0	±1.5	+3.0, -1.0
Crash Landing	I	+3.0, -1.5	±1.5	+4.5, -2.0
Ground				
Truck or Air Ride Trailer	S	±2.0	±2.0	+3.0, -1.0
Rail (Humping)	S	±30.0	±5.0	±15.0
Rail (Normal Operation)	S	±3.0	±1.5	+3.0, -1.0
Dolly (Max Velocity, 5 mph)	I	±1.0	±0.75	+1.5, +0.5
Dolly (Hand Operated)	I	±0.2V	±0.15V	1
Forklift	S	±1.0	±0.5	+2.0, 0.0
Hoist	S	0	0	1.33
Hoist (Heritage Hardware)	S	0	0	1

Shock Loads The Maximum shock envirnoment for the space craft occurs when the seperation of the spacecraft from the launch vehicle happens. according to the atlas V launch manual, the maximum shock load in the Axial direction is 0.9g and in the lateral direction is 0.6g[15].

2.5.3 Preliminary dimension and design

Material trade off The most common materials used for the primary structure of the spacecraft is 6061-T6 or 7075 aluminum [17], these materials are used for the good E modulus and ultimate stress. however recently carbon nanotube based composites are being used for planetary missions. this material has many great qualities including high elastic modulus (0.8-3)Tpa, high thermal conductivity (3000-6000)W/mK,low density (1400kg/m^3) and its high electrical resistivity (3-20)$\mu\Omega cm$,[18] clearly this material has much better qualities than the other mentioned materials (the table below includes the properties of the mentioned metals), however the only downside to carbon nanotube based composites is the fact that they are expensive . Since the budget for the JUICE mission is 1.7 billion the cost of the material chosen for the primary structure is a good trade off.

Table 2.15: Material Choices for spacecraft Primary structure

	6061-T6 Aluminium [19]	**7075 Aluminium [20]**	**Carbon Nanotube based composite [18]**
E-Modulus	69GPa	71.7GPa	940GPa (average)
Yield Stress	276MPa	503MPa	20GPa
Density	2700kg/m^3	2810kg/m^3	1400kg/m^3
Thermal conductivity	167W/mK	130W/mK	(3000-6000)W/mK
Resistivity	4$\mu\Omega cm$	5.15$\mu\Omega cm$	(3-20)$\mu\Omega cm$
Cost	Low cost	Low cost	High cost

Initial Design parameters Since the primary structure consists of a Launch vehicle adaptor and the body structure of the spacecraft. therefore the chosen body structure is a hollow cylindrical shape and the Launch vehicle adaptor is a D1666 Atlas V standard payload adaptor [15]. this adaptor has been used since according to the ADSEE reader a 1.6m diameter payload adapter can support a 3.3m diameter spacecraft [2]. Based on the volume requirement ($21.6m^3$ B.1) and the shape that is used (cylinder) a suitable dimensioning of the spacecraft should be a cylinder with a diameter of 3 meters and a height of 3 meters.In this section of the report the Mass of the spacecraft used is approximately 2000kg, this mass is used in order to get appropriate dimensions for the spacecraft.

Designing for strength when designing for strength the maximum stress and the stress at which buckling occurs has to be taken into account (critical buckling stress) therefore for a material with an E-Modulus of 940Gpa and a Yield stress of 30Gpa the critical buckling stress could be computed using this formula:

$$\frac{\sigma_c}{E} = 9[\frac{t}{R}]^{1.6} + 0.16[\frac{t}{L_p}]^{1.3} \tag{2.24}$$

and the maximum stress applied on the spacecraft is computed using this formula, bearing in mind that the maximum axial loading is 5.5g and the maximum lateral loading is 2g:

$$\sigma_{max} = \frac{g_y M L_c}{I} + \frac{g_x M}{A} \tag{2.25}$$

furthermore both formulas are used in order to get the best possible thickness.

$$\frac{g_y M L_c}{I} + \frac{g_x M}{A} = E(9[\frac{t}{R}]^{1.6} + 0.16[\frac{t}{L_p}]^{1.3}) \tag{2.26}$$

using the fact that $I = \pi r^3 t$ and $A = \pi D t$ this formula becomes:

$$\frac{g_y M L_c}{\pi r^3} + \frac{g_x M}{\pi D} = E(9[\frac{9}{R^{1.6}}]t^{2.6} + 0.16[\frac{0.16}{L_p^{1.6}}]t^{2.3}) \tag{2.27}$$

$$21627 = 940 \cdot 10^9 (4.704t^{2.6} + 0.03836t^{2.3}) \tag{2.28}$$

with this formula we can get the root of the equation closest to 0 by the Newton-Raphson method [21]:

$$t_{n+1} = t_n + \frac{f(t)}{f'(t)} \tag{2.29}$$

f(x) represents the formula 2.28 when equated to zero, f'(x) represents the same formula after its differentiated. after iterating the Newton-Raphson method multiple times using $x_0 = 0.1$ the value of thickness that is achieved using this method is 0.62mm however after the addition of a safety factor of 1.25 the required thickness is 0.78mm.

Designing for Stiffness Taking into account that the launcher fundamental frequency requirement is 8 Hz lateral and 15Hz axial as a minimum, And an average Elastic Modulus of 940Gpa is used for the calculation.[22][15][2]

for the Lateral frequency

$$8hz \leq \frac{1}{2\pi}\sqrt{\frac{3E\pi r^3 t}{ML_p^3}} \leq \frac{1}{2\pi}\sqrt{\frac{0.94 \cdot \pi \cdot 1.5^3 t \cdot 10^{12}}{2000 \cdot 3^3}} \tag{2.30}$$

According to the lateral frequency the required thickness t is at least $1.369 \cdot 10^{-5}$ for the longitudinal frequency

$$15hz \leq \frac{1}{2\pi}\sqrt{\frac{E\pi Dt}{L_p M}} \leq \frac{1}{2\pi}\sqrt{\frac{\pi \cdot 0.94 \cdot 3 \cdot t \cdot 10^{12}}{3 \cdot 2000}} \tag{2.31}$$

According to the longitudinal frequency the required thickness t is at least $6.016 \cdot 10^{-6}$ and our chosen thickness is 0.78mm therefore based on the requirement of stiffness the requirement is full filled and the structure is designed for strength since it required a higher thickness than designing for stiffness. with the chosen thickness the mass of the primary structure calculated will be 46.3kg. the mass is calculated using this formula:

$$M = \rho_{CNT}((2\pi r L_p) + (2\pi r^2) \tag{2.32}$$

2.5.4 Propellant tank

According to the estimations preformed in B.1 the propellant mass is approximately 1108kg. First volume is calculated from density of propellant using the fact that the propellant has a density of $1400kg/m^3$ for the Oxidizer and $1010kg/m^3$ for the Fuel and the mass (Fuel and Oxidizer) has been separated by a mass ratio of 1.1 . [4]

$$V = \frac{M}{\rho} \tag{2.33}$$

However we have to consider a 10% increase in volume as safety.then we calculate the thickness of the tank based on the hoop and longitudinal stress and considering the fact that the maximum expected operating pressure in the tank is approximately 24 bar.[2] in addition to that, the material used for the propellant tank is titanium since it has a high young's Modulus (105-120Gpa) and a high elastic limit(830Mpa) however at the cost of the high density($4500kg/m^3$).

$$V_o = 1.1\frac{580.475}{1400} = 0.456 \tag{2.34}$$

$$V_f = 1.1\frac{527.7}{1010} = 0.575 \tag{2.35}$$

This volume computed is the volume of both the oxidizer and the fuel however since the design consists of 3 tanks and they (the oxidizer and fuel) have different volume proportions due to the oxidizer/fuel and difference in density then this volume will be distributed upon 1 oxidizer tank and 2 fuel tanks. after the distribution of volumes the volume of the oxidizer tank is $0.450m^3$ and the volume of each fuel tank is $0.2834m^3$, given the fact that a sphere has a volume of $\frac{4}{3}\pi r^3$ the diameter of the oxidizer tank is 0.95m whilst the diameter of both fuel tanks is 0.81m.

Design of tank the propellant tanks will be in the shape of a sphere because that is the most weight efficient shape as a propellant tank.

Material trade off the table below shows the common materials used for tanks.[4]

Table 2.16: Material choices for propellant tanks

	Aluminum Alloy	**Carbon Fibre composite**	**Fibre glass composite**	**Magnesium**	**Steel**	**Titanium**
E-Modulus	69GPa	530GPa	125-150GPa	45GPa	200Gpa	105-120GPa
Yield Stress	400MPa	-	-	100MPa	250-700MPa	830MPa
Density	2700	1800	2500	1700	7800	4500

The chosen material is Titanium since it has a very high elastic modulus which makes the thickness requirement much lower than the rest of the materials but this advantage comes with a disadvantage of a very high density.however since the elastic modulus is so high and therefore the thickness is low this gives rise to a thin structure that is capable of withstanding the required pressure loads and since its a thin structure the mass of the structure is lower compared to a structure built by another material with a smaller yield stress (that material would have a larger thickness and thus have a mass that is close to the mass of titanium). this is the case since titanium has the larges ratio of Yield stress to density according to the available data in the table.

Designing for Pressure the reason why the tank is designed for pressure is because pressure is the most influencing factor on the tanks since the pressure required inside the tank is around 24 bar.

Given the fact that the pressure inside the tank is approximately 24 bar and the Hoop stress is equal to the axial stress used for the calculation of thickness of the tank since it produces the higher requirement of thickness of the propellant tanks. additionally a margin of safety of 1.1 is used.

$$t = \frac{P_t.r}{\sigma_{yield}} \cdot 1.1 \tag{2.36}$$

first the calculations are done for the fuel tanks.

$$t = \frac{24 \cdot 10^5 \cdot 0.4092}{830Mpa} 1.1 = 1.30mm \tag{2.37}$$

then the thickness calculation is done for the oxidizer tank:

$$t = \frac{24 \cdot 10^5 \cdot 0.4770}{830Mpa} 1.1 = 1.52mm \tag{2.38}$$

thus the required thickness for the oxidizer tank is 1.51mm and for the fuel tank its 1.30mm. the design information of the tanks are summarized in this table:

	Oxidizer tank	**Fuel Tank**
Diameter	0.95m	0.82m
Thickness	1.52mm	1.30mm
Material	Titanium	Titanium
Mass	19.5kg	12.3kg

Table 2.17: Properties and dimensions of the propellant tanks

the mass was calculated using the formula $4\pi r^2 t$ and the total mass of all the tanks are 44kgs

Iteration something that equation (4.20) does not show is that when the propellant mass is changed the volume of the tanks are changed therefore the size and thickness of the tanks differ this causes a change in the mass of the tanks thus the whole structural mass changes which causes equation (4.20) to be calculated once again with different parameters. however this problem could be solved using iteration to find the propellant mass at which no changes occur to the system. this is done by means of an equation that is a function of its own variable which is represented below.

$$M_e = (1150.6 + ((\frac{M_p}{\rho_{propellant}} \frac{3}{4} \frac{1}{\pi}) 4\pi \frac{24 \cdot 10^5}{830 \cdot 10^6} 4500 \cdot 1.1^2) \tag{2.39}$$

which simplifies to

$$M_e = 1150.6 + 0.0395 M_p \tag{2.40}$$

however M_e is a part of a bigger equation which is:

$$M_{p+1} = M_e(e^{\frac{1}{300 \cdot 9.08665}(\frac{125607.5}{\frac{M_p}{2}+M_s}+1857.3)}) - 1) \tag{2.41}$$

after iteration The final value of M_p is 1112kg and for M_e its 1196.26kg. however since the propellant mass is only 4 kgs heavier than the propellant mass used no recalculation should be done.

2.5.5 Supporting structure of Solar Array

Since the solar arrays have to be mounted on the primary structure of the spacecraft is its required that the structure that supports the solar array is able to deploy the solar array to be able to generate the required power. For the supporting structure of the solar array a hinge will be used to mount the solar arrays to the primary structure. the design of the supporting structure of the solar arrays will include a hinge system, an array panel structure and a deployment system.

Hinge system: the hinge system is required to control, deploy and lock the solar array at each edge of the solar array. The design of the hinges includes a spring[23],dampener and a locking mechanism. The material used for the deployable and lockable mechanism is an aluminum alloy 2A12, and for the rotating hinge assembly stainless steel 1Cr18Ni9Ti is used.[24]

damping mechanism: since the locking mechanism of the deployment stage causes oscillatory/impulsive forces or moments it is necessary to include a dampening system to reduce the effect of these forces during the deployment stage, therefore an eddy current damper is used since its more advantageous when comparing it to other forms of dampeners (liquid-based viscous dampers, paraffin actuators and mechanical brakes) since they are non-contact, non-leakage and simple fabrication.[25]

Supporting structure: the purpose of the rigid planar structure is to have a stable and secure framework to provide strength to the solar array, this is done by means of mounting the solar cell array onto a structure that is stable and sufficiently strong. The materials that are used in this structure is a lightweight aluminum honeycomb core,Kapton and composite face sheets. The sizing of the aluminium honeycomb core and the two thin composite face sheets that are attached above and below the core is 18mm thick, while the Kapton that is responsible for insulating the solar cells has a thickness of 0.05mm[26][27], this structure would have approximately a total mass of 56kg, this is because the aluminum honeycomb structure has a thickness of 18mm and thus the density would be $28.8kg/m^3$ [28]. the figures below roughly presents the structure that the solar arrays will be mounted on, this includes a root hinge which is used to attach the yoke (triangular structure attaching the solar panels to the root hinge) to the primary structure. further more secondary hinges are used for the deployment mechanism (in order to fold the solar arrays during the launch of the spacecraft to decrease the space used).

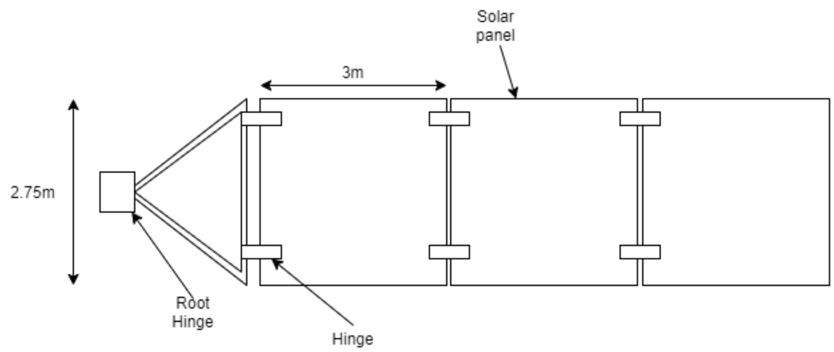

Figure 2.11: Hinge Distribution

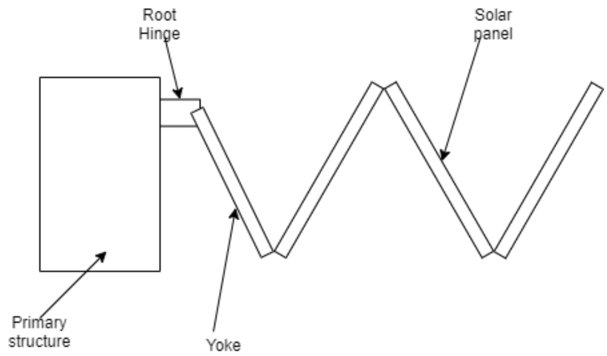

Figure 2.12: Deployment mechanism

2.5.6 Spacecraft Mechanisms

- Deployment [29]: For the solar arrays a hinge system is used which is explained in the previous subsection, for the antenna an antenna
- Pointing [29]: For the antenna an antenna gimbal is used to point the antenna towards to intended direction, for the solar arrays a solar array drive mechanism is used in order to point the solar arrays to the position required.
- Attitude mechanism [29]: to be able to orient the spacecraft in the desired direction or stablize the spacecraft an attitude mechanism is used, therefore for this spacecraft reaction wheels are used.
- Dampers & Load abosrbers [29]: for the solar arrays an eddy current damping mechanism has been used to make sure the solar array is deployed safely as explained in the previous subsection.

2.6 Payload

Based on the description of the JUICE-mission and payload data from previous missions, a detailed payload list can be generated.

Since a main mission objective of the JUICE orbiter is to analyze ion and plasma particles present around the planet, the orbiter should have a plasma spectrometer. The NASA's Cassini Spacecraft had a plasma spectrometer on board as well, the Cassini Plasma Spectrometer (CAPS). This scientific instrument consists of three sensors: an electron sensor, an ion mass spectrometer and an ion beam sensor. The instrument is able to measure the particle's kinetic energy, the direction the particle was travelling, and the mass of the studied particle. Due to the fact that this instrument can only detect particles that physically collide with its detectors, it should be placed on the outside of the orbiter in such a way that it can observe most particles as possible. The mass of the CAPS is 12.50 kg including the boom, the average operating power is 14.50 W and the data rate is 0.5 kb/s during survey to 16 kbps during encounters.[30]

Furthermore, the JUICE spacecraft needs to investigate Jupiter's magnetosphere, so a magnetometer is needed. Juno has measured the magnetic field of Jupiter using the MAG instrument of Juno [31] that consists of two fluxgate magnetometers, mounted to a boom on the end of the solar panel, creating the maximum distance from the spacecraft itself. This prevents the magnetometers from picking up the magnetic fields of the spacecraft. The magnetometer assembly attached to the end of the solar panel weighs a total of 15.25kg. FGMs use very little power, around 2W is typical [32]. Moreover, the average data rate is 4 kb/s.

Additionally, imagers will be on board to provide images of the atmosphere as well as of the cyclones that are to be studied. For this purpose both a visible light camera, as well as an infrared imager that can penetrate the surface 50 to 70km. Furthermore, another ultraviolet imager will be used to find out more about the atmosphere.

The visible light camera will be based on the JunoCam, which in its turn is based on the Mars Descent Imager from the Curiosity rover. It weighs 3.7kg and consumes 4.7W while idle, and 5.9W while imaging. It has an average data rate of 0.325 kb/s [33]. This image should be placed in such a way it point in the right direction and it has an unobstructed view.

The infrared imager will be based on Juno's infrared imager as well. The Jovian Infrared Auroral Mapper weighs around 8 kg and has a peak power consumption of 16.7 W [31]. Both of these imagers need to placed in such a way that they have an unobstructed view of the surface of the planet. It has an average data rate of 3.37 kb/s.

The ultraviolet imager will again be base on Juno's. It weighs 21.5 kg, consumes 9W and has an average data rate of 1.768 kb/s. [31]. As this is an imager, it too needs to have an unobstructed view of the subject, in order for it to be able to take pictures that generate valuable data. All the important details of the scientific instruments can be found in table 2.18

Table 2.18: Table with the mass, power and data rate of the scientific instruments of the JUICE orbiter

Instrument	**Mass[kg]**	**Operating Power [W]**	**Data Rate [kb/s]**
CAPS	12.50	14.50	0.5
MAG	15.25	2	4
JunoCam	3.7	5.9	0.325
JIRAM	8	16.7	~3.367
UVS	21.5	9	~1.768
Total	60.95	48.1	9.96

2.7 Other subsystems

There are multiple other subsystems on the spacecraft that need consideration.

2.7.1 Command and Data Handling

This system is responsible for sending commands to the different parts of the spacecraft. Additionally this system has to record all the data recorded from the instruments for transmission back to earth later, thus the requirements depend on the payload's data rate. This central computer is usually laid out redundantly as well.

- The CDH system shall record the payload data at 9.96 kb/s
- The CDH system shall be able to send commands to all the systems of the spacecraft.
- The CDH system shall have TBD memory to save the payload data before transmission.
- The CDH system shall be made up of two identical redundant computers.

2.7.2 Telemetry and Communication

This subsystem is crucial to the operation of the spacecraft. It is responsible for making sure the spacecraft receives the appropriate commands and can transmit the acquired scientific data back to earth. From the payload the following requirements can be defined.

- The downlink data rate shall be TBD or better.
- The uplink data rate shall be TBD or better.

- The spacecraft shall have one directional high gain antenna and one one non directional low gain antenna.
- The signal to noise ratio shall be 10 or more.
- The system shall have redundant transceivers.

For high data rates a parabolic, highly directional is used to be able to transmit the majority of the scientific data the spacecraft collects. This antenna requires the spacecraft to be very precisely aligned to the antenna on the ground. Thus in case the spacecraft is not able to be aligned due to having a malfunction or needing to point in a different direction for firing the engine, there needs to be another antenna. This will be a lower gain, less directional antenna. This will not allow high data rates, but can make sure the spacecraft will always be reached from the ground. In order to generate the signal going to the antenna and receiving the signals from the antenna transceivers are needed. These will also be redundant to make sure a single failure will not prevent the spacecraft from communicating with the ground station.

2.7.3 Navigation

The navigation subsystem shall allow the ground station to determine the position and velocity of the spacecraft. To do this there is usually a transponder integrated in the communications and telemetry system. This transponder re-transmits ranging tones coming from the ground station to allow the ground station to calculate position from the turnaround time. Velocity can be computed from the frequency shift between the transmitted and received signal.

Budget Revision 3

In this chapter, the refined mass and power budgets will presented and compared to the earlier estimations from WP1, also present in Appendix B. The results will then be compared to all the mission requirements and an iteration will be performed in case it is proved necessary.

3.1 Mass

The mass of the spacecraft consists of two parts: the spacecrafts dry and propellant masses, together forming the spacecrafts wet mass. As is calculated in subsection 2.2.4, the propellant mass is estimated to be 1112kg. While, the dry mass of the spacecraft consists of the mass of components of the ADCS, CDH, communications, the propulsion subsystem, thermal control, the power subsystem, and the structure and mechanisms of the spacecraft and the payload. The mass break down among the different subsystems is given in Table 5.1.

All subsystems ought to be included to the M_e, however some are exactly known yet at this stage of the design, so these will still have their maximum estimated values from Table B.2 These estimated subsystems are ADCS, Communications, CDH, harness, and propulsion. The rest of the subsystems will use their newly calculated values. For the thermal control subsystem, the MLI is estimated to weight 18kg, the 5 RHUs weight 1.4kg each and each louver has a weight of about 1kg. Up to five louvers may be needed so the total mass for this subsystem has been calculated being about 30kg.

Table 3.1: Calculation M_e

Subsystems	**Mass [kg]**
ADCS	61.3
CDH	40.6
Communications	59.0
EPS	335.4
Harness	75.1
Payload	61.0
Structures and mechanisms	146.3
Thermal	30.0
Total without propulsion (81%)	808.61
Propulsion (19%)	165.7
Total without margin	974.3
22 % Margin[1]	215.83
Total with margin	**1190.1**

Comparison to WP1

The budget created in this report is much more detailed, and will give a realistic view upon the mass budget. While, in WP1 the team based their budget mainly on estimations and less accurate calculations.

What can mainly be deduced, when comparing both budgets, is that the former mass budgets still had a lower and upper bound, which is not accurate, but common for the first stage of the design. The ranges were used to get an idea of what type of values could reasonably be expected while it is now more important to have a conservative upper bound estimate. Besides this major difference, it can be retrieved that the total dry mass is significantly higher than was estimated, even by the the upper bound estimation. Since, the upper bound estimate was $996.5kg$, while the newly calculated total dry mass is $1190.1kg$. Furthermore, EPS is mainly responsible for the increase, as it doubled its value, compared to last report. Hence, this makes the estimation in WP1 less realistic, this probably comes from the fact that interplanetary missions up to celestial bodies like Jupiter often used a RTG in the past and thus the estimation relations were not completely adapted to this mission. However, the use of an RTG was discussed in section 2.4 and it was decided to go with solar panels mainly because of the cost. Moreover, the propulsion subsystem slightly increased in weight compared to the last report, however it is not very significant. Lastly, most calculated values remained very close to their estimated counterpart, which shows that these estimations can be assumed as very realistic.

Comparison to mission requirements

The mass being more of an objective to be minimised on its own, the only mission requirement that considers an aspect of the mass budget is: *The payload mass shall be within 100 and 300kg.* In this case, the payload mass was found to be much less than expected: about 61kg. This is not considered as a problem since it is lower than expected and not higher, no iteration of any subsystem is thus necessary.

3.2 Power

Table B.2 shows the power budget for the spacecraft's bus from the Initial sizing report. However, in this power budget, the power used by the payload is absent. From table 2.18 we see that the payload consumes 48.1 W. This gives a new power budget as can be seen in table 3.2

Table 3.2: Total power budget M_e

Subsystems	**Power [W]**
ADCS	14.1-17.0
CDH	19.2-23.2
Communications	38.6-46.4
EPS	2.6-3.1
Harness	-
Payload	48.1
Structures and mechanisms	6.4-7.7
Thermal	42.5-51.0
Propulsion	5.1-6.2
Total without margin	176.6
Total with margin	**202.7**

In comparison to table B.2 only the power has been added to the budget. Since the power for the payload can be determined with great precision due to requirements of the scientific instruments, no margin is needed. The other subsystems will be more specified in a later design phase.

In the design requirements it is stated that the spacecraft shall be able to deliver between 50 W and 250 W to the payload. Since the power we determined is only 1.9 W of the upper limit, this won't form a problem. The total power with margin (202.7 W) is neatly within the design requirement of total power, i.e. the spacecraft shall be able to deliver 200 W and 800 W of total power.

4 Spacecraft Architecture

In this chapter the architecture of the spacecraft and the placing of different components is explained. First, in section 4.1, the architecture of the general architecture of the spacecraft will be motivated. In section 4.2, the placing of the components of the different subsystems is explained and finally, in section 4.3, a comparison will be made between the current spacecraft architecture and the architecture from work-package 1.

4.1 General structure

This time a cylindrical bus was chosen with a diameter of 3m and a height of 3m, such that the volume will be $21m^2$ which is in the high range of the estimated bus volume from work-package 1 [1]. The diameter of the antenna is still taken to be 2.5m based on the Juno mission [31], which had one of the same size. It is placed on top of the cylindrical bus. Refer to Fig. A.1 and table A.1 for a detailed image of the spacecraft architecture and the part names and their quantities.

The four solar arrays are attached to the sides of the bus (Fig. A.2). As written in section 2.5.5 the solar panel and its yoke fold in. Because the yoke is smaller than the three 3m long solar array parts it was best to attach the solar panel arm to somewhere in the middle of the spacecraft bus to fold the spacecraft as efficient as possible (Fig. A.3). However, when the solar array is then turned up vertically, which may need to happen in order to get the most sunlight, a lot of solar radiation will be blocked by the bus of the spacecraft. To avoid that, the solar panel arm is attached to the bus, one yoke-length down from the top of the bus (Fig. A.2). Now, if the solar array needs to be pointed vertically, only a small amount of solar radiation will be blocked by the antenna (Fig. A.4.

Furthermore, a small magnetometer boom is placed on top of one solar arrays. This strategy was also used on the Juno spacecraft [31] because now a long magnetometer boom from the spacecraft bus is not needed. However this will cause one solar panel to be longer and heavier than the others which might cause stability issues. So a possible solution may be to make the solar panel area of this solar array smaller and making the others a bit longer. Unfortunately, this also has its downsides, because longer solar panels mean more mechanisms which can lead to failure. So this needs to be considered better in the following work-packages.

4.2 Subsystems

In this section the placing of the components of the different subsystems will be explained.

4.2.1 ADCS

Sensors The three star sensors are placed perpendicular to each other on top of the bus of the spacecraft to provide 3 axis knowledge (section 2.1.4). Furthermore, three gyro's are placed on the spacecraft, one on top and two on the sides. So every gyro tracks the rotation of the spacecraft around one axis. Finally, one sun sensor is placed on top a solar panel.

Actuators There are a total of twelve thrusters and four reaction wheels placed on the spacecraft. From the four reaction wheels one is placed on top and three are placed on the side such that these are in a pyramid configuration. The twelve thrusters are all attached to the sides, such that every axis has four thrusters (two for each direction around an axis).

4.2.2 Propulsion

The placing of the following components is based on the information provided in section 2.2.5.

The main engine is placed in the center on the bottom of the bus. The Debris shield is located around the main engine. While the engine is in use, it is folded inside a pocket around the engine. The two fuel tanks and the oxidizer tank are placed inside the bus touching each other such that their center of mass coincides with the vertical axis of the cylinder. Finally, the helium tank is placed in the center of the cylindrical bus on top of the propellant tanks.

As can be noted, there are a total of 13 thrusters on the spacecraft (table A.1), which is a lot. The downside of this is that it is harder to place the payload instruments such that these are placed at a safe distance from all the engines and don't have a high chance of failure due to heat. This is something which needs to be further improved on in the next steps of the design process.

4.2.3 Payload and Power

The payload consists of five scientific devices listed in table 1. For the placing of the payload it is important to keep in mind that these usually need to be placed on the outsides of the spacecraft in order for these to take their measurements.

First, the plasma spectrometer (CAPS) is placed on the edge of the top of the bus.

The magnetometer (MAG) is attached to the tip of the magnetometer boom at around 13.5m from the center of the spacecraft in order to minimize the effects of electromagnetic fields generated by the spacecraft on the measurements of the magnetometer.

The imager (JunoCam) is placed on the bottom of the spacecraft, which is pointed to Jupiter at all times. It is placed such that the engine is out its field of view.

Furthermore, both the infrared imager (JIRAM) and the ultraviolet imager (UVS) are placed on the bottom of the spacecraft as these need an unobstructed view of Jupiter (Section 2.6).

Finally, the battery is placed on top of the spacecraft, far from the thrusters to minimise the chance of failure.

4.3 Comparison with Previous Design

As can be noted there are a number of differences to be found compared to the spacecraft design in work-package 1 [1].

First, the bus was changed from a box-shape to a cylindrical shaped one, as it was recommended in the project reader. Furthermore, the solar array area was increased dramatically, it changed from $60m^2$ [1] into $98.8m^2$ 2.11. The number of solar panels stayed the same, but the dimensions of each of them changed from 2.5m x 6.0m to 2.75m x 9.0m. The hinge systems of the solar panel arms has also changed, because, with the current design of the hinges, the spacecraft would not fit inside the launcher if the yokes were extended. Unfortunately, more mechanisms are needed here and the chance of failure increases.

In the current design there are also propellant tanks and a helium tank added, which weren't present in the first design.

The only change for the power subsystem is that the battery size was decreased as there will be no significant eclipse during JUICE's mission. There were however a lot of changes for the payload instruments. In this work-package a more detailed list of scientific instruments is obtained, so these instruments are designed with some more detail. Furthermore, the magnetometer boom from work-package 1 is switched with a smaller one attached on top of a solar array.

Finally for the ADCS subsystem three of the four sun-sensors were removed as these are not accurate enough at Jupiter. Instead of these, three star sensors are used. Furthermore, three gyro's are used. These also weren't present in the design of work-package 1. The spacecraft now also has reaction wheels which it didn't have earlier. The spacecraft now also has more thrusters such that the spacecraft can still rotate even if some reaction wheels or thrusters have failed.

Conclusion 5

The purpose of this report was to design the subsystems of the JUICE spacecraft to be able to set up budgets for the mass, power and cost. This process was worked on in parallel, as the team split up into teams that each worked on a subsystem. Each group first defined the requirements that particular system needs to fulfill. Then the subsystem could be preliminarily designed and sized. Once the results of the subsystems were known the revised budgets were set up. A preliminary spacecraft architecture was developed. The results of this work will be summarized here.

The ADCS requirements state that the spacecraft be 3 axis controlled and achieve a pointing accuracy of 2 mrad or better. This leads to a design using three star sensors, a coarse sun sensor and three gyros for attitude sensing, while actuation is achieved using four momentum wheels and 12 thrusters for momentum dumping. The use of 3 star sensors and 4 momentum wheels allows for redundancy, as the ADCS is a critical system that is vulnerable to failures.

The propulsion system's requirements are computed from the mission profile. It is found that it needs to be able to provide a delta v of 1925.5 m/s. This is achieved using the LEROS 1b nitrogen tetroxide/hydrazine engine. Depending on the empty mass a propellant mass of 572.6 kg to 985.15 kg is required.

The thermal system makes sure the spacecraft operates within the thermal limits of the components. First an equilibrium temperature is calculated based upon a passive system and the heat flux into and out of the spacecraft during the mission. This provides a temperature of 167.9 - 166.2K, which means active thermal control is needed.

The power consumed by all the subsystems comes from the power system. This will use a solar panels with an area of $98.8m^2$ and a mass of $313.2kg$ accompanied by a battery of $778.8Wh$ weighing $5.86kg$ to provide the maximum power of the spacecraft of $263.43W$ during all times of the mission.

The structure of the spacecraft consists of a hollow cylinder of $3m$ diameter and $0.65mm$ thickness made out of carbon nanotubes designed to resist all launch and handling loads and provide the mechanisms for the deployment of the solar panels as well as the spherical tanks to store the propellants of the engine. There are two titanium tanks for the fuel, each with a diameter of $0.79m$ and $1.25mm$ thickness, and one titanium tank for the oxidizer with a diameter of $0.92m$ of thickness $1.44mm$. Their mass in total is $38.8kg$.

The payload consists of multiple instruments derived from previous missions totaling a mass of $60.95kg$, a power consumption of $48.1W$. They produce a data rate of $9.96kb/s$. Using these subsystem specifications new refined budgets were set up, the results of which can be seen in **??** and Table 5.2.

Table 5.1: M_e Budget

Subsystems	**Mass [kg]**
ADCS	61.3
CDH	40.6
Communications	59.0
EPS	335.4
Harness	75.1
Payload	61.0
Structures and mechanisms	146.3
Thermal	30.0
Total without propulsion (81%)	808.61
Propulsion (19%)	165.7
Total without margin	974.3
22 % Margin[1]	215.83
Total with margin	**1190.1**

Table 5.2: Total power budget M_e

Subsystems	**Power [W]**
ADCS	14.1-17.0
CDH	19.2-23.2
Communications	38.6-46.4
EPS	2.6-3.1
Harness	-
Payload	48.1
Structures and mechanisms	6.4-7.7
Thermal	42.5-51.0
Propulsion	5.1-6.2
Total without margin	176.6
Total with margin	**202.7**

Finally a preliminary spacecraft architecture is designed. This illustrates the placement and size of the subsystems that have been designed. The architecture is based on a cylindrical primary structure. This cylinder is where all other subsystems attach to. The engine will be attached to one end of the cylinder, while the antenna is at the other end. Solar panels will fold out radially from the cylinder. A rendering of the spacecraft can be seen in Figure 5.1.

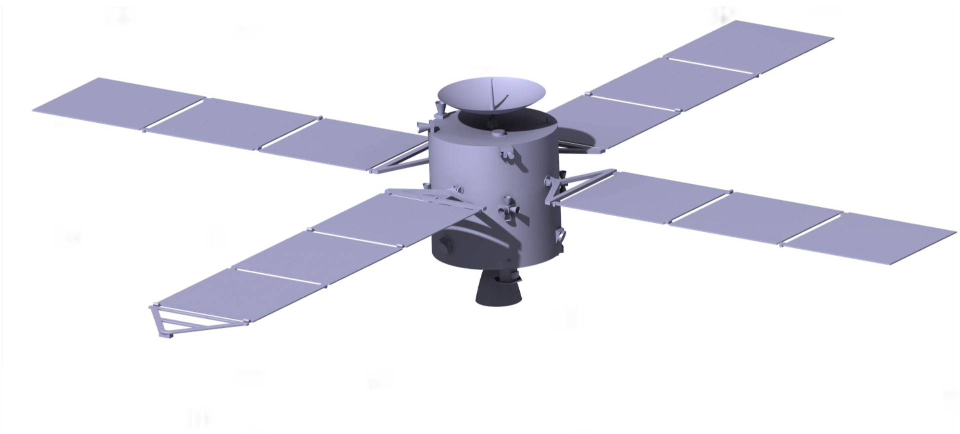

Figure 5.1: Render of the JUICE spacecraft

The next steps we recommend in realizing the spacecraft is to continuing the detailed design for the various subsystems. This involves making choices, such as whether to use commercially available solutions or designing new systems, followed up by selecting the right components or starting the process off designing the new systems. For some systems these choices have already begun, while others still need more attention and work.

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CATIA Drawings of the JUICE Spacecraft A

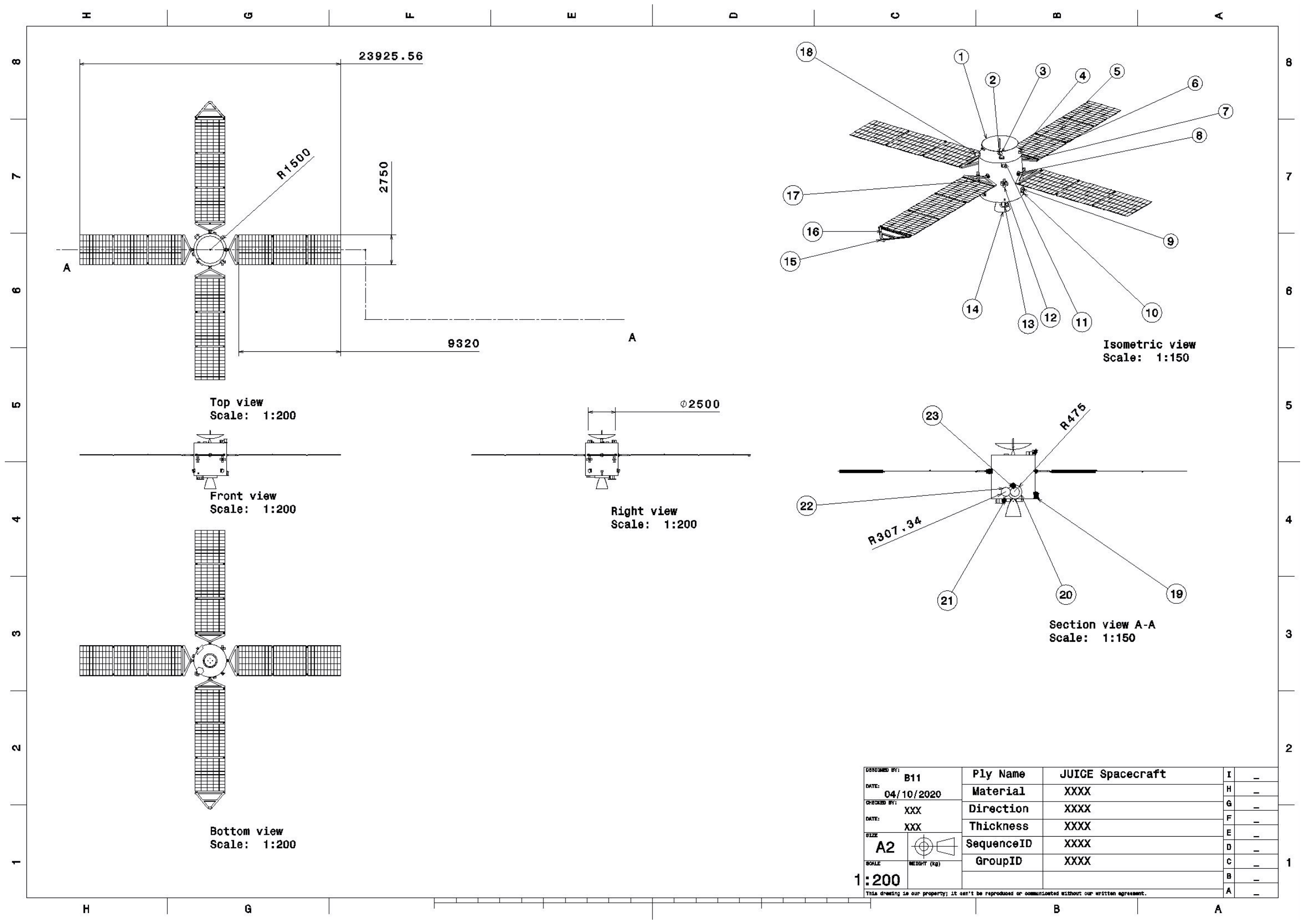

Figure A.1: CATIA Drawing of JUICE Spacecraft

Table A.1: The Parts of the JUICE Spacecraft

Part Number	Part Name	Quantity	Part Number	Part Name	Quantity
1	Antenna	1	13	JIRAM	1
2	Antenna Joint	1	14	Main Engine	1
3	Gyro	3	15	MAG	1
4	Battery	1	16	Magnetometer Boom	1
5	Solar Panel	12	17	Sun Sensor	1
6	Solar Panel Hinge	26	18	Star Sensor	3
7	Yoke	4	19	JunoCam	1
8	Solar Panel base Hinge	4	20	Oxidizer Tank	1
9	Reaction Wheel	4	21	Debris Shield	1
10	UVS	1	22	Fuel Tank	2
11	CAPS	1	23	Helium Tank	2
12	Small Thruster Group	4			

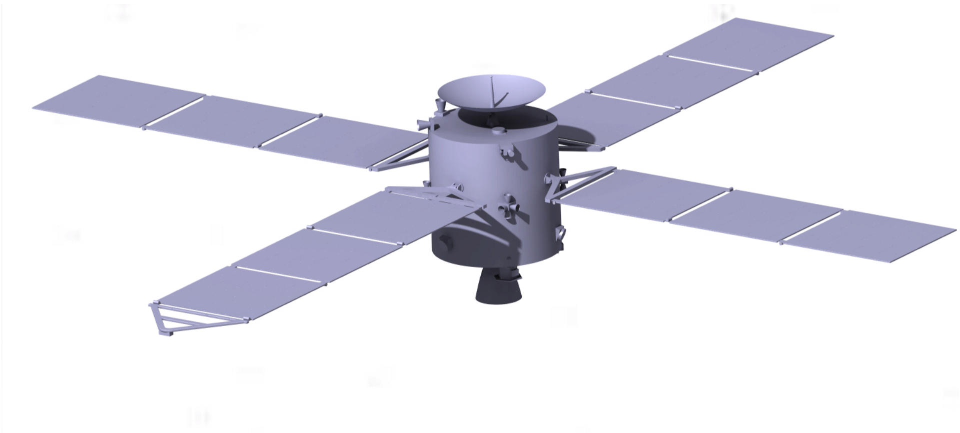

Figure A.2: CATIA Render of Deployed JUICE spacecraft

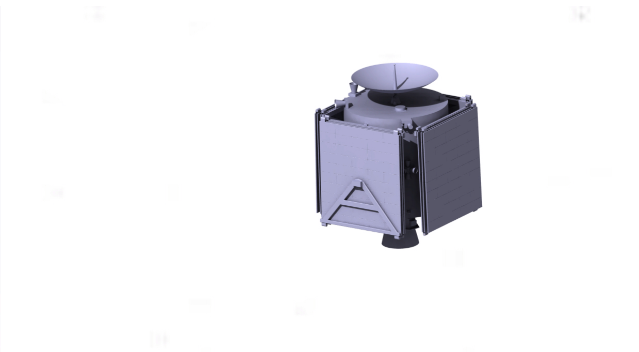

Figure A.3: CATIA Render of Undeployed JUICE spacecraft

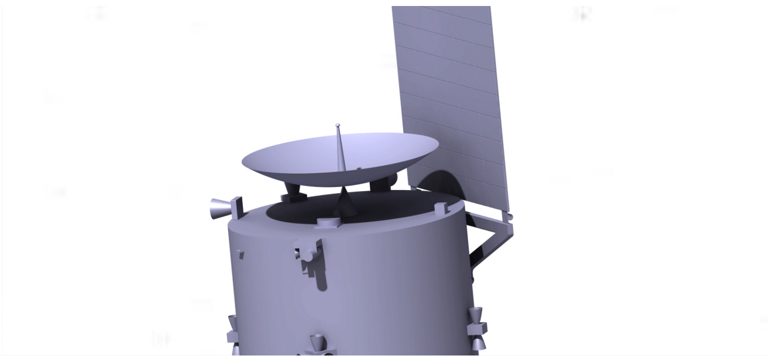

Figure A.4: CATIA Render of a Part of the Spacecraft with a vertical solar Panel

Estimations Performed in WP1 B

In this appendix, the mass and power budgets generated in the last work package as well as the general estimations are provided for reference. For further details about those results, please consult the previous report. Table B.1 provides the first order estimations performed in WP1 based on the requirements

Table B.1: Vehicle Level Estimations

Parameter	**Statistical Relation**	**Results Range**
Dry Mass	$M_{Dry} = 2.233 \cdot M_{PL} + 396.6$ [1]	619.9-1066.5kg
Loaded Mass	$M_{Loaded} = 15.909 \cdot M_{PL} + 24.7$[2]	1615.6-4797.4kg
Total Power (solar panels)	$P_t = 1.13 \cdot P_{PL} + 122$ [3]	178.5-404.5W
Total Power (RTG powered)	$P_t = 332.93 \cdot \ln P_{PL} - 1046.6$ [4]	255.8-791.7W
Size (solar powered)	$V_{S/C} = 0.0045 \cdot M_{Loaded}$ [5]	7.3-21.6m^3
Size (RTG powered)	$V_{S/C} = 0.0269 \cdot M_{Loaded}$ [6]	43.5-129.1m^3
Reliability	$R = \exp(-\lambda \cdot t)$ [7]	0.378 (7 years) - 0.189 (12 years)
Cost	$C_{S/C} = 0.3531 \cdot (M_{Dry})^{0.839}$ [8]	77.74-122.56 FY2000 M$

The preliminary mass and power budgets of the different subsystems are provided in **??**.

Table B.2: Mass and power budgets from WP1

Subsystem	**% total power**	**Power [W]**	**% total mass**	**Mass [kg]**
Propulsion	4	5.1-6.2	19	98.7-145.6
ADCS	11	14.1-17.0	8	41.6-61.3
Communications	30	38.6-46.4	7.7	40.0-59.0
CDH	15	19.3-23.2	5.3	27.5-40.6
Thermal	33	42.4-51.0	4.8	24.9-36.8
Power	2	2.6-3.1	22.4	116.4-171.7
Structures	5	6.4-7.7	23.0	119.5-176.3
Harness			9.8	50.9-75.1
Total S/C Bus		128.5-154.6		519.6-766.5
30% Margin		38.6-46.4		155.9-230.0
Total S/C Bus with margin		167.0-200.9		675.5-996.5

[1] Payload mass range: 8-365kg, R^2: 0.603, RSE:38.2%.
[2] Payload range: 8-180kg, R^2: 0.6686, RSE: 95.5%. The relation was extrapolated for the upper bound's estimate
[3] Payload power range: 5-1000W, the model is appropriate for all types of solar powered spacecrafts only.
[4] Payload power range: 75-250W.
[5] Slope range: 0.0024-0.0185, density range: 54-409kg/m^3, average density: 222kg/m^3, mass range: 286-3625kg.
[6] Slope range: 0.0149-0.0555, density range: 18-67kg/m^3, average density: 37.2kg/m^3, mass range: 258-5623kg.
[7] λ: 0.056-0.139, using worst case scenario. It is expected that the reliability estimates become larger once more data is available.
[8] Mass range:40-2350kg.

Orbital Parameters and Details C

In this chapter, the orbital parameters are given and corrected since workpackage WP1. Then, the code used to obtain the eclipse time of the same orbit but in the plane of the planet (0° inclination) is given. Note that the method used to determine the orbit was outlined in WP1 and won't be repeated here. Table C.1 presents the details of the orbit chosen.

Table C.1: Parameters of the Selected orbit

Parameter	**Value**
Pericenter	3270509.07km
Apocenter	78492km
Eccentricity	0.953
Semi-major axis	1674500.5km
Semi-minor axis	506664.4km
Inclination	90°

The eclipse time was estimated to 0.22% of the total orbital period using the following code. No further explanation is given about it, main strategy and important points are given as a comment in the code.

```
import numpy as np
from math import *
def orbdata(theta,e,a,mu):
    #orbit equation
    r = (a*(1-e**2))/(1+ e*cos(radians(theta)))
    #vis viva
    v = sqrt(-((mu)/a)+ 2*(mu/r))
    #omega = dtheta/dt
    # the velocity vector in (r, theta) has r-component dr/dt and theta-component omega*r. So
    v^2 = (dr/dt)^2 + (omega*r)^2
    #dr/dt= dr/dtheta * omega. We first compute dr/dtheta
    dr_dtheta = ((a*e*(1-e**2))/(1+e*cos(radians(theta)))**2)*sin(radians(theta))
    #So rearanging v^2 = (dr/dt)^2 + (omega*r)^2 we get:
    omega = sqrt(v**2/(dr_dtheta**2+r**2))
    H=(r**2)*omega
    return r,v,omega,H


e = 0.9531251269392532
a = 1674500.534610078 #km
b = 506664.3838491398 #km
mu= 126.687E6
t= 0
dt = 1
theta=0
T = 2*pi*sqrt((a**3)/mu)
rlst = []
vlst = []
omegalst=[]
tlst =[]
```

```
Rjup = 69911
ecltime = 0
eclthetalst =[]
boundlst = []
while t < T:
    datalst = orbdata(theta,e, a, mu)
    r = datalst[0]
    v = datalst[1]
    omega = datalst[2]
    rlst.append(r)
    tlst.append(t)
    vlst.append(v)
    omegalst.append(omega)
    dtheta = degrees(omega)*dt
    theta = theta+ dtheta
    H = datalst[3]
    #eclipse around pericenter. For eclipse around apocenter: instead of (theta > 270 or theta <90) -> 90 < theta < 270
    if abs(r*sin(radians(theta))) <= Rjup and (theta > 270 or theta <90):
        if (abs((a*(1-e**2))/(1+ e*cos(radians(theta-dtheta)))*sin(radians(theta-dtheta))) > Rjup) or (abs((a*(1-e**2))/(1+ e*cos(radians(theta+dtheta)))*sin(radians(theta+dtheta))) > Rjup):
            boundlst.append(theta)

        ecltime = ecltime + dt
        eclthetalst.append(theta)

    t= t+dt
eclminute= ecltime/60
eclpercent = (ecltime/T)*100
#time in minutes and percentage
print('The eclipse time in minutes is:',eclminute)
print('Percentage of total period:', round(eclpercent,2),'%')
print('The spacecraft is in eclipse between theta is', round(boundlst[0],1), 'degrees and theta is', round(boundlst[1],1), 'degrees')
```

Task Distribution D

In this appendix, Table D.1 presents the task distribution among the different team members.

Table D.1: Task distribution per member

Team member	**Deliverables**
Jonatan	ADCS, Payload, Chapter 6, Appendix A
Tarek	Complete structure section
Stefano	B.1, B.2, B.3, B.4, B.5, B.6, Budget Revision (Mass), Grammar and sentence structure, Proofreading: Payload, Other subsystems, Conclusion
Niklas	Summary, Introduction, Conclusion, ADCS, Payload, Other Subsystems
Antonio	C.1, C.2, C.3, C.4, C.5, C.6, PR Section 4.3
Lorenz	E.1, E.2, E.3, E.4, E.5, E.6, TW, Appendix B&C, Proofreading: Summary, Introduction, ADCS, Propulsion, Thermal, Power
Silvano	Payload, Budget Revision Power, General proofreading
Sam	C.1, C.2, C.3, C.4, C.5, C.6, PR Section 2.3