Delft University of Technology

Faculty of Aerospace Engineering
AE2111-I
Systems Design

Initial Sizing of the JUICE Spacecraft

A Mission to the Giant of our Solar Systems

Mission: Jupiter's JUICE mission
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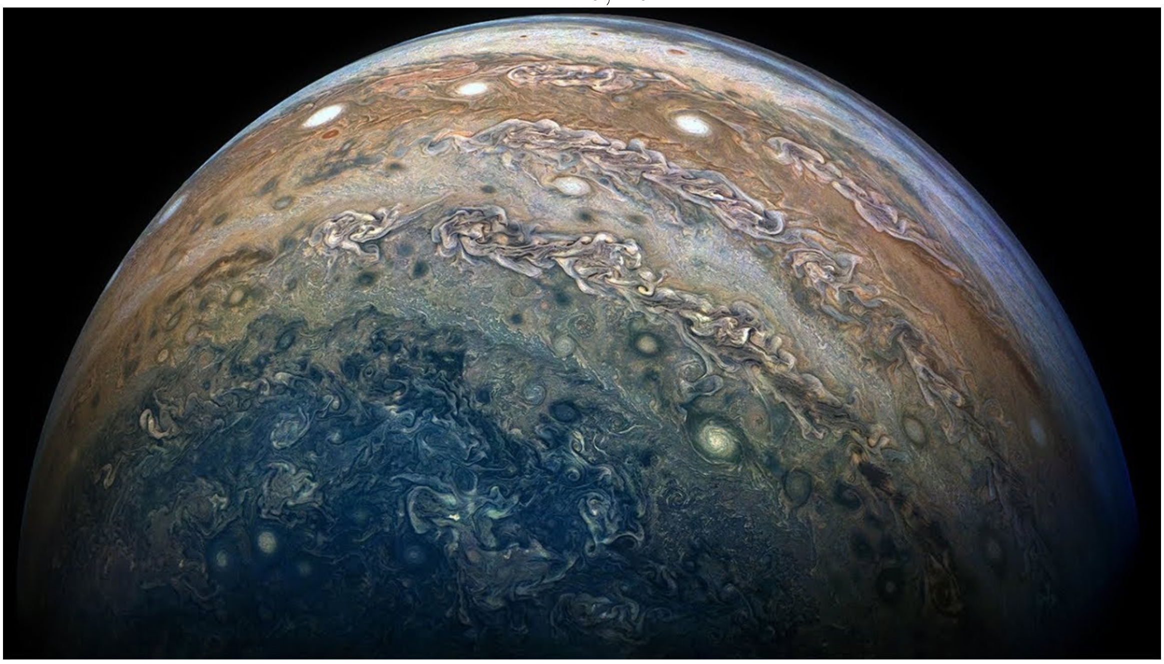

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Summary

Of all the planets of the solar system, Jupiter is by far the most massive. The elements that govern its violent atmosphere and its incredibly strong magnetosphere are still widely unknown to humankind. A research mission on this planet can expand our knowledge about the phenomenoa that shape this far distant world. One of these missions will be launched in 2025, the JUICE mission.

The aim of this report is to describe the initial sizing process for the JUICE mission. The main objective of this mission is to study the atmosphere of Jupiter and learn about the fields and charged particles that surround the planet. This will be done with the payload on board of the orbiter. The payload consists of multiple scientific instruments among which are magnetometers to map the magnetic field of Jupiter, spectrometers to measure charged particles and imagers to study the atmosphere. The spacecraft is scheduled to be launched in 2025 and the mission duration shall be between 7 and 12 years. The spacecraft should also have the following characteristics: the payload mass is between 100 and 300kg, the payload dimensions should be $1m \cdot 2m \cdot 1m$ and should receive from 50 to 250W of power and the total power generated should be between 200 and 800 Watts. Next to this it is required that the closest approach in the orbit is at least 10,000 km above the 'surface' of the planet and that the mission will not cost more than 1.7 billion euro's. Together these objectives and requirements set the mission and the mission profile and a preliminary design can be formulated.

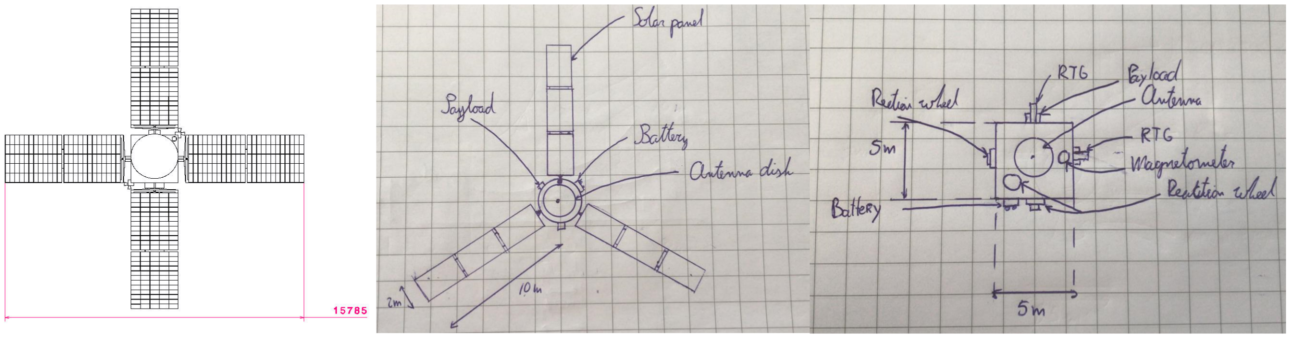
Using statistical relationships, the main spacecraft characteristics can be estimated. The spacecraft dry mass is in the range 620 - 1067kg, the total power and size is between 179 - 405W and 7.3 - 21.6m^3 for a solar power powered spacecraft and between 256 - 792W and 43.5 - 129.1m^3 for a RTG (Radioisotope Thermoelectric Generator) powered one. The estimated reliability lies between 0.378 and 0.189 at this stage but is expected to increase as more information becomes available at further stages of the design, while the cost was estimated being between 78 and 123 million dollars in Fiscal Year 2000. Using the escape velocity equation and ΔV budgets of other similar missions the ΔV budget of the JUICE mission can be approximated. Using this approximation and the rocket equation the propellant mass can be estimated. The total ΔV is 2185 m/s and the spacecraft chemical bi-propellant mass is 1620.2 kg

Concluding the report a power and mass budget can be calculated. This was done using statistical relationships once again and a mass margin of 30%. The results are summarized in Table 1.

Table 1: Mass and Power budgets

Subsystem	Mass percentage	Mass	Power percentage	Power
Propulsion	4.75%	50.7kg	4%	6.2W
ADCS	6%	64.0kg	11%	17.0W
Communications	4.75%	50.7kg	30%	46.4W
Thermal	8.5%	90.7kg	33%	23.2W
Power	30%	320.0kg	2%	51.0W
Structures	20%	213.3kg	5%	3.1W
CDS			15%	7.7W
Total S/C Bus		789.4kg		154.6W
30% Margin[5]		236.8kg		46.4W
Total S/C Bus with margin		1026.2kg		201.0W

Finally the design options for the orbiter are sketched. In total 3 design options are made. The first option uses four solar arrays to generate power and has a rectangular bus. The second design uses three solar arrays and has a cylindrical bus. Finally the third design uses RTG's to power the spacecraft and has a rectangular bus. A technical drawing has been made of the first design and a sketch has been made for the second and third design.



Design 1 Design 2 Design 3

Figure 1: Spacecraft Design Top View Sketches

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List of Symbols

Latin Letters	Quantity	Symbol Unit	Unit
a	Semi-major axis	km	kilometers
b	Semi-minor axis	km	kilometers
$C_{S/C}$	Spacecraft cost	$	dollars
I_{sp}	Specific impulse	s	seconds
M_{Dry}	Dry mass	kg	kilogram
M_{Loaded}	Loaded mass	kg	kilogram
M_{PL}	Payload mass	kg	kilogram
P_{PL}	Payload power	W	watts
P_t	Total power	W	watts
R	Reliability	[-]	[-]
R_a	Apocenter	km	kilometers
R_p	Pericenter	km	kilometers
T	Orbital period	s	seconds
$V_{S/C}$	Spacecraft volume	m^3	meters cube

Greek Letters	Quantity	Symbol Unit	Unit
μ	Standard gravitational parameter	km^3/s^2	kilometers cube per seconds square

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0.1 General

DASML - *Delft Aerospace Structures and Materials Laboratory*
FAA - *Federal Aviation Administration*
S/C - *Spacecraft*
RTG - *Radioisotope Thermoelectric Generator*
MMOI - *Mass moment of inertia*

0.2 Scientific Instruments Abreviations and Their Functions

CAPS - *Cassini Plasma Spectrometer, to measure the flux of charged particles*
CDA - *Cosmic Dust Analyzer, this instrument will investigate tiny dust grains near Saturn*
CIRS - *Composite Infrared Spectrometer, to measure infrared radiation*
CRS - *Cosmic ray system, this system determines the origin of cosmic rays and other information about the cosmic rays*
GS - *Gravity Science, it will measure Jupiter's gravity*
INMS - *Measures the composition of charged and neutral particles*
IRIS - *This system measures thermal energy and atmospheric composition of a planet*
ISS - *Two camera system that provides imagery with electromagnetic waves with wavelengths in the range of 280 to 640 nm*
JADE - *Jovian Auroral Distributions Experiment, an energetic particle detector that will investigate ions and electrons at low energies*
JCM - *JunoCam, a visible light camera*
JEDI - *Jovian Energetic Particle Detector Instrument, an energetic particle detector that will investigate ions and electrons at high energies*
JIRAM - *Jovian Infrared Auroral Mapper, this instrument will observe IR radiation in the upper layers of the atmosphere*
LECP - *Gathers information about the differential in energy fluxes and angular distributions of ions and electrons*
MAG - *Magnetometer, it will investigate the magnetic field of Jupiter*
MIMI - *Instrument that produces images and measures other data about particles trapped in the magnetosphere of Saturn*
MWR - *Microwave Radiometer, it will measure electromagnetic waves, to be specific radio waves*
PLS - *Measures electrons in the energy range of 5 to 1000 eV and investigates properties of plasma*
PPS - *Studies the radio-emission signals from Jupiter and Saturn using a sweep-frequency radio receive*
PWS - *Makes measurements of the electron-density profiles of a planet*
RPWS - *Radio and Plasma Wave Science Instrument, measures radio waves coming from the inter-*

action of the solar wind with Saturn and titan
RSS - *Radio Science Sub-instrument to observe radio waves as they went to various mediums*
UVIS - *Measures uv-radiation reflected by the surface of a planet*
UVS - *Ultraviolet Spectograph, it will measure ultraviolet photons and atmospheric properties*
VIMS - *Using visible and infrared light images of the surface was made to learn more about the composition of this surface*
Waves - *Radio and Plasma Wave Sensor; this instrument will measure radio and plasma spectral in the auroral region*
MS - *Magnetometer sensor used to detect and analyse how the magnetic field of the planet affects the satellite*
PI - *Plasma instrument used to detect low energy particles*
PWD - *to study electromagnetic waves generated by the particles*
HEPD - *high-energy particle detector*
CJDD - *detector of cosmic and Jovian dust*
UD - *an extreme ultraviolet detector associated with the ultraviolet spectrometer*
HIC - *heavy ion counter to assess potentially hazardous charged-particle environments the spacecraft flew through*

1 Introduction

Jupiter is the largest planet in our solar system and is a gas giant twice as massive as all the other planets combined. Its atmosphere is home to huge storms such as the Big Red Spot and its magnetosphere is the strongest in our neighbourhood at nearly 20000 the strength of Earth's one. It traps electrically charged particles, which constantly blast its nearby moons with high levels of radiation. The aim of the JUICE mission is to gather more information on the atmospheric phenomena of Jupiter, while also study the different types of fields and charged particles that surround the planet.

The goal of this report is to identify the mission's design requirements, estimate the spacecraft main parameters such as dry mass, power and size, provide a description for the mission profile and explain the spacecraft initial sizing's process. The mission top-level requirements have to be met as well: the payload mass should not exceed 300kg, the total power should not be above 800W, the temperature of the payload should lie between 150K - 250K and the duration of the mission is planned to be between 7 and 12 years.

The report will have the following structure. The first step of initial sizing of the spacecraft will be treated in Chapter 2. Here the functional analysis is made. After the objectives are set and it is clear what elements will interact with the mission, the requirements of the spacecraft will be defined. These can be found in Chapter 3. Next a mission profile will be made in Chapter 4 using data from similar spacecraft and statistical relations. The initial sizing of the spacecraft is concluded in Chapter 5. The report ends with a conclusion in Chapter 6, here the most important characteristics of the mission are summarized.

Functional Analysis 2

This chapter provides a preliminary functional analysis of the spacecraft. In section 2.1 spacecraft with similar missions were researched and some information about them was collected. Section 2.2 covers the different important elements of the mission interacting with the spacecraft. Based on these chapters and the description of JUICE the objectives were collected in section 2.3 and explained further in 2.4. The already known parameters of the orbiter were summarized in 2.5.

2.1 Similar Missions

In Table 2.1 a few spacecrafts with similar missions are listed. Their mission objectives, main elements, design characteristics and the performance of each element is given.

Table 2.1: Similar Missions and their Spacecraft

Name of Mission	Mission objectives	Elements that make up the mission	Design characteristics and preformance of elements
Galileo	Analyse the dynamics and circulation of Jupiter's atmosphere mostly in the upper atmosphere and the ionosphere. The spacecraft should also be able to analyse the interaction of the magnetic field and gravitational field of the planet with the spacecraft. Finally, the spacecraft should investigate the composition and distribution of minerals on the surface of Jupiter.	Payload: MS, PI, PWD, HEPD, CJDD, UD, HIC Electrical power: power comes from the radioisotope thermoelectric generators.	The magnetometer is extended 11 meters to one side The rest of the instruments weight 118kg. The radioisotope generators produce 570W at launch and 485W at the end of the mission
Voyager 2	The mission objective of voyager 2 is to study multiple planets during a flyby with its 11 scientific instruments. The planets the spacecraft studied are Jupiter, Saturn, Uranus and Neptune. Uranus and Neptune were in the original mission. Next to the study of the outer planets the Voyager 2 spacecraft was also designed to travel to the edge of our solar system to study the environment.	Payload:ISS, UVS, IRIS, PRA, PPS, MAG, PLS, LECP, PWS, CRS, RSS Electrical system: electrical power is provided by three MHW-RTGs.	Each MHW-RTG has a mass of 37.7kg and produces around 157 watts of power. Heat is provided by nine RHU's. These are small devices which make use of the decay of plutonium-238 to provide the thermal energy to make sure the equipment of the spacecraft can keep operating
Pioneer 10	The mission objective is to investigate and study Jupiter's atmosphere and surface. Following up the spacecraft enters an escape trajectory from the solar system.	Payload: Imaging photopolarimeter, helium vector magnetometer, infrared adiometer, quadrispherical plasma analyzer, ultraviolet photometer, charged particle instrument, cosmic ray telescope, Geiger tube telescope, Sisyphus asteroid, meteoroid detector, TRD The payload is used to successfully analyse and study the surface of Jupiter in addition to its atmosphere electrical power: 4 radioisotope thermoelectric generators	Each radioisotope thermoelectric generator produced 40W at launch payload consists of 11 science instruments. each present to preform a certain task explained in the table below.
Juno	The mission objectives for Juno is to study Jupiter's atmosphere composition, gravity field, magnetic field and polar magnetosphere. The main elements of the Juno mission are the payload, the propulsion and navigation subsystem of the spacecraft.	Payload: MWR, JIRAM, MAG, GS, JADE, JEDI, PWS, UVS, JCM Electrical power:Juno uses three solar panels symmetrically arranged around the spacecraft	Design characteristics Juno: The solar arrays are 9 meters long and have a width of 2.65 meters. Together the three solar panels generate 14 kilowatts at 1 AU and 400 watt at the distance Jupiter is from the sun.
Cassini	Cassini was sent to study Saturn and its system together with the lander Huygens. Huygens landed on Titan. Cassini also made a flyby of Jupiter, during which it made scientific measurements. A goal of the Cassini-Huygens mission was to increase the human understanding of Saturn along with its moons and rings and to investigate whether life might be found in this part of the solar system.	Payload: CAPS, CIRS, ISS, UVIS, VIMS, CDA, INMS, MAG, MIMI, RPWS, Radar, RSS Electrical system: The Cassini-probe has three GPHS RTGs	The 3 GPHS RTGs provide the electrical power needed for the payload and the spacecraft itself. The probe also has 82 RHUs, these provide heat to keep the instruments at a temperature where they can keep operating.

2.2 Mission Elements

There are multiple elements that interact with the spacecraft to ensure the success of the mission, and thus the design process of the spacecraft. These are shown in Figure 2.1 and will be defined here as given by [1].

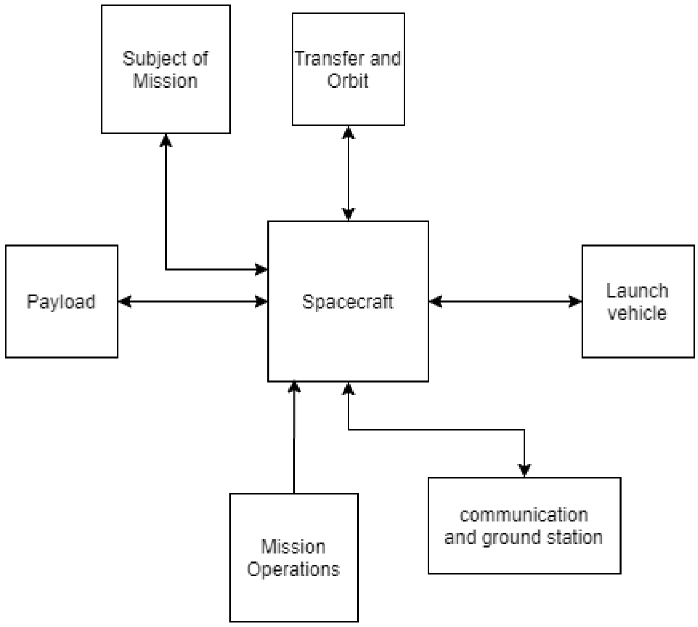


Figure 2.1: Block Diagram of mission elements

- **The payload** is central to the mission, it is the reason the entire mission exists. It consists of the sensors and experiments that deliver the data the scientists are interested in. Here, the payload will consist of magnetometers, spectrometers and imagers.
- **The subject of the mission** is the what the payload interacts with to generate the required data. In the case of the JUICE mission, Jupiter's atmosphere and magnetosphere are the main subject of the mission.
- **The ground system** is responsible of acquiring the scientific data collected by the payload through a RF communication link and handling the telemetry data.
- **The mission operations** are responsible for sending commands to the spacecraft and monitoring its systems, to ensure a successful mission. This system works in pair with the ground system and are sometimes located at the same facilities. Indeed, all commands sent by the mission operation are based on data received primarily by the ground system.
- **The launch vehicle**, and possibly a kick stage, will bring the orbiter from the surface of the Earth into space and then on-wards.
- **The transfer trajectory** determines both the final orbit possible, as well as the necessary ΔV required by the spacecraft. It should be considered in a very meticulous way to use necessary flybies in order to minimize the necessary propellant to be taken on board.

2.3 Main Objectives of the JUICE Mission

The overall objective of the JUICE mission is to obtain more knowledge the atmosphere and magnetosphere of Jupiter. Guided by the mission description in the Spacecraft Design reader [2], these objectives are listed in this section.

1. The spacecraft shall study the various atmospheric phenomena on Jupiter. This includes the cyclones, Great Red Spot and huge storms that cover the planet among others.
2. The spacecraft shall analyse and map the magnetic field of Jupiter which expands up to several millions of kilometers from the planet.
3. The spacecraft shall analyse the concentration in ion and plasma particles present around Jupiter.
4. The spacecraft shall study the interaction between the magnetic and gravitational field of Jupiter.
5. The spacecraft shall be released into orbit in 2025.
6. The total mission cost may not exceed 1.7 billion Euros (Fiscal Year 2020).
7. The mission duration shall be between 7 and 12 years.
8. The payload mass should be in the range 100kg - 300kg.
9. The total power used by the spacecraft should be between 200W and 800W.
10. The temperature of the spacecraft may lay between 150K and 250K.
11. The ΔV shall be enough to bring the spacecraft into orbit around Jupiter.
12. The spacecraft shall be put in an elliptical orbit, to be more efficient.
13. The spacecraft shall not fail due to loads being taken into account by the necessary design margins.
14. The total mass of the spacecraft shall be minimized for efficiency reasons.

2.4 Main functionalities of each objective

In order to realise the design objectives of the spacecraft, it is of paramount importance to specify the main functionalities that accompany the objectives. Hence, a list of functionalities per objective.

1. The spacecraft must carry imagers in order to realise its ability to study the atmospheric phenomena of Jupiter. A proper accuracy of the attitude determination and control system is required for a proper pointing of the imager.
2. The spacecraft must carry magnetometers to measure Jupiter's magnetic field.
3. The spacecraft must carry a spectrometer to analyse charged ions and plasma particles. Again, the ADCS subsystem should have a high enough accuracy to ensure a proper pointing of the spectrometer.
4. The planning and the personal must be aware of the fact that the deadline is 2025. Proper milestones and deadlines should be put to ensure that the final deadline is met.
5. The spacecraft must carry the necessary sensors to make sure that the minimum approach distance objective is met.
6. A budget-cap should be placed on the manufacturing of the spacecraft.
7. The durability of the materials and electronics used for the spacecraft should be of at least 7 years. This should take the high amount of radiation, a protective box can be used to protect the most important components.
8. The spacecraft structure should be able to carry a payload of mass up to 300kg. The instrumentation on the payload and its structure may not exceed a mass of 300kg.

9. The instrumentation on board of the spacecraft should use a minimum of 200W of power and not more than 800W.

10. Thermal control should ensure that the temperature of all components stay between the range 150K - 250K. That involves the use of heaters and coolers to avoid all overheating and over-cooling of the different subsystems.

11. Enough propulsion systems and propellant should be carried by the spacecraft, in order to meet the desired ΔV.

12. To ensure an elliptical orbit, the propulsion system should receive the information about when to create the certain ΔV, to reach the elliptical orbit.

13. Factors of safety should be taken into account when preforming structural dimensioning; Yield load ≥ 1.1 and ultimate load ≥ 1.25. The type of structure should be as efficient as possible in terms of load bearing, the different types still need to be investigated.

14. The carried magnetometers and gravity meters will be analysing the the interaction between both fields.

2.5 Design Parameters of the Orbiter

In this section, a list of all design parameters is given. Some of those are already known and others left to be assessed. The following items were given in the mission description [2]:

1. **Payload mass**: 100kg - 300kg. It is a driving parameter for the design process as it has a direct impact many aspects of the design such as the structural parts.

2. **The payload size**: $1m{\cdot}2m{\cdot}1m$is important as it should be taken into account in the arrangement of the different subsystems in the spacecraft. In the same domain, it is important to know if some sensors need a specific positioning inside the spacecraft (e.g. at the 'surface').

3. **Payload power**: 50W - 250W. The power required by the payload will have an impact on the design of the electrical system and on the energy storage/production devices used.

4. **Total power**: 200W - 800W. The total power needed for a proper functioning of the spacecraft will determine (among others) what type of energy production system and energy storage is to be used.

5. **Orbiter temperature**: 150K - 250K, this will partly determine the type of heating/cooling system used.

6. **Mission reliability**: 0.9. The mission reliability should be taken into account throughout the design process as it flows down from the reliability of each sub-system.

7. **Mission duration**: seven to twelve years. This design parameter is crucial for determining the type of power source to be used.

In addition to those, it will also be important to assess the following:

1. **Dimensions of the spacecraft**: The dimensions of the spacecraft need to be known for the launcher selection as it should fit in the payload bay.

2. **The type of orbit** needs to be chosen to determine the time the spacecraft will be in sunlight. This will have an impact on the power source selection and heating/cooling systems. Note that it is already known that the closest point of approach is 10,000 km but the type of orbit itself is still to be chosen.

Spacecraft Requirements 3

From the objectives in the previous chapter a set of requirements for the spacecraft will be formed. For that, general knowledge about Jupiter was collected in 3.1 as well as more data about similar spacecrafts in 3.2. With the help of that information requirements were formed in section 3.3 with the driving requirements being highlighted in 3.4 These serve as a guideline during the design to make sure the spacecraft is capable of performing its mission to the required level.

3.1 Scientific Objective and Characteristics of Jupiter

Table 3.1 gives an overview of important characteristics of the celestial body studied. The information needed to generate this table was found on the Jupiter fact sheet from NASA [3].

Table 3.1: Characteristics of Jupiter

Characteristic	**Value**
Equatorial radius	71 492km
Mass	$1,898.19 \cdot 10^{24} kg$
Bond albedo	0.343
Geometric albedo	0.538
Solar intensity received	50.26 W/m^2
Orbital period at 10,000km altitude	216.4387 minutes
Eclipse periods of S/C [1]	73.729 minutes
Eclipse time of celestial body	2h every 1.8 days [4]
Atmospheric density at 100kPa	0.16 kg/m^3
Gravitational parameter (μ)	$126.687 \cdot 10^6 km^3/s^2$
Average distance from the Sun	5.2 AU
Inclination axis	3.13°
Black body temperature	109.9K

The JUICE mission is aimed to study Jupiter's atmospheric phenomenon as well as the fields and charged particles that are present in the neighbourhood of the planet. In order to complete those research goals, the payload consists of imagers, spectrometers and magnetometers making the collection of the various relevant data possible. Vortices lasting from a few days to hundreds of years, the Great Red Spot and the circumpolar cyclones are atmospheric phenomenons [5] that, among others, can be investigated by the JUICE mission. The spectrometers will be used to analyse ion particles, plasma particles and magnetometers will be used in order to map the magnetic field of the planet. The collection of all this data is important for scientists, as it can be used for a better understanding of the formation of our solar system. °

[1] Assuming a circular orbit at an altitude of 10,000km above the 'surface'. Using trigonometry, it is found that 34% of the orbital period is in eclipse. The total orbital period is found to be 12986.326s and the proper ratio is applied.

3.2 Comparable Spacecrafts

In this section, five comparable spacecrafts are listed and their basic information is given. Those will be used as a mean to verify the accuracy of the estimations in 4.1.

Satelite	[6]Ulysses	Galileo	Huygens	New Horizons	Juno
Payload Info	Mass: 55kg Power: unknown	[7]Mass: 118kg Power: unknown	[8]Mass:43.8kg Power:unknown	Mass: 30kg Power:28W	Mass:unknown Power:unknown
Power Budget	Average:285W	[7][8]Average:570W	[8]Average:351W	Average:240W	Average;500W
Size	2m x 2m x 1m	[9]5.3m high 11 meters to one side when magnetometer extended	[10]2.7m Wide	0.7m x 2.1m x 2.7m	3.5m(height)x3.5m(diameter)
Communications	1.65m diameter parabolic dish	[9]4.8 meter high gain antenna		Antenna Diameter: 2.1m	Antenna diameter: 2.5m
Structure	honey comb structure	[9]5.3 meters from the antenna to the bottom of the probe, Dual spin stableized			Solar panel Area 60m^2
Mass	Launch:370kg	[7]Launch:2223kg Dry:1298kg	[10]Launch:318kg	Launch: 478kg	Launch: 3625kg Dry:159kg
Orbital parameters	Orbital inclination:80.2 heliocentric ranges between 1.34-5.4AU Period:6.2 years communication:	[11]Orbit period:5.6 years Altitude: 214000km			Inclination: Polar Pericenter: 4100km Period: 14 days

3.3 Design Requirements

In this section, the relevant requirements of the mission for preliminary design are listed. The list is divided in subsections to ensure its practicality in further usage.

1. The spacecraft shall be able to carry a payload with a mass in the range of 100 kg to 300 kg.
2. The spacecraft shall be able to support a payload having 1m x 2m x 1m as dimensions.
3. The spacecraft shall be able to deliver between 50 W and 250 W to the payload.
4. The spacecraft shall be able to deliver between 200 W and 800 W of total power.
5. The spacecraft shall be able to keep its temperature in the range of 150K to 250K.
6. The duration of the mission shall be between seven and twelve years.
7. The mission reliability shall be at least 0.9 in the seven to twelve year long mission.
8. The spacecraft shall have a high-gain antenna with a diameter of TBD m to communicate with earth.
9. The spacecraft shall be able to withstand without failure a TBD amount of radiations for a TBD amount of time per orbit.
10. The spacecraft shall be able to withstand without failure a TBD amount of charged particle for a TBD amount of time per orbit.
11. The spacecraft shall be able to achieve at least a TBD total ΔV to perform the necessary maneuvers and housekeeping.
12. The spacecraft shall be able to de-orbit to enter Jupiter's atmosphere and communicate further data during its last minutes fall.
13. The orbit of the spacecraft shall have an inclination of TBD °.
14. The spacecraft shall be collect data under 10,000km from Jupiter.

15. The orbit shall be sun-synchronous.
16. The spacecraft shall use solar panels or an RTG for power generation.
17. The spacecraft shall be able to deploy the solar panels if those are used.
18. The batteries shall be able to provide enough power during the complete eclipse period.
19. The batteries shall be able to refill in a TBD amount of time
20. The spacecraft shall be able to point the payload with a TBD arc-seconds accuracy.
21. The ADCS subsystem shall be able to perform pitch rotations at at least TBD deg/
22. The ADCS subsystem shall be able to perform yaw rotations at at least TBD deg/s
23. The ADCS subsystem shall be able to perform roll rotations at at least TBD deg/s
24. The spacecraft shall be able to endure launch loads of TBD N.
25. The spacecraft shall be able to endure launch vibrations.
26. The spacecraft shall be able to endure TBD transportation loads to the launch site.
27. The spacecraft shall be able to communicate relevant data from the payload and telemetry with the ground station at during a TBD portion of the orbital period.
28. The spacecraft shall be able to receive and execute commands (TT&C).
29. The spacecraft shall be able to collect reliable data from the magnetometer, spectrometer and imagers.
30. The spacecraft should have a total mass of TBDkg.
31. The spacecraft should have a maximum size of TBD m^3.
32. The spacecraft shall use a TBD stabilisation system (spin or 3-axis).
33. The TT&C subsystem shall transmit data at a TBD bit rate.
34. The TT&C subsystem shall transmit data at a TBD frequency.
35. The ADCS subsystem shall be able to determine the attitude of the spacecraft at a TBD accuracy.

3.4 Driving Requirements

When it comes to design requirements, some are more important than others, as they will play a major role in the design process. Therefore, it is of great importance to state these so-called "driving requirements" and discuss their part in the design.

1. **Payload mass**: The payload may be the most important design requirement, as it the consists of the load that needs to be brought to a certain location, in this case in an orbit around Jupiter. The payload is thus a driving requirement, since the spacecraft should hold the payload and the structure may not fail, under the payload. Therefore the payload influences the materials used and the structure inside the spacecraft. No payload means no mission.
2. **Dimensions**: The dimensions are a driving requirement, as they work as a certain constraint for the design, where it can not go over or under these values. The dimensions of the payload are one as this should fit in the spacecraft but the general dimensions of the probe should also be taken into account for the launch vehicle choice.

3. **Power**: The power that needs to be delivered to both the payload and the rest of the spacecraft is a driving requirement, as it determines the size and mass of the batteries and solar panels/RTG for the spacecraft.

4. **Temperature range**: When choosing materials for a spacecraft, one should know the temperature range, as this will influence the behaviour of the materials. Hence, the materials chosen for the spacecraft should function in the given temperature range, thus it will affect the design. Furthermore, this will determine the type of cooling/heating systems used.

5. **Duration**: The duration of 7 to 12 years is also a driving requirement as it will resolve, which materials to use, as these materials may not corrode during the mission. Some protective devices can also be considered as a mean to overcome radiations.

6. **Reliability**: This is a driving requirement, since the reliability of 0.9 will impact design choices of the spacecraft. This will mainly affect the materials and how much "back-up" payload is brought by the spacecraft.

7. **Antenna**: One of the requirements discusses the diameter of the antenna, which immediately influences the design, on the pointing accuracy and how to place the antenna (The diameter itself is still to be determined though).

8. **Resistance charged particles**: Some materials will degrade faster when in contact with charged particles, than other materials. So in the design-phase, one should select materials that will sustain these charged particles over the whole mission duration.

9. **ΔV**: In order to get the spacecraft into orbit, multiple manoeuvres and heliocentric transfers are needed. The ΔV resembles the propulsion needed for these transfers, thus determines how much propulsion should be taken by the spacecraft. Therefore, the ΔV is a driving requirement, as it will influence the size and weight of the propulsion system, hence the design of the spacecraft.

10. **Power generation mean**: As stated by the requirements, "The spacecraft shall use solar panels or an RTG for power generation", which means the design should anticipate on both at a first place (and later choose one design over the other), so this will heavily influence the design of the spacecraft and make it a driving requirement. Furthermore, the solar panels would need to be deploy-able, which again influences the design greatly. And the RTG (if used) should not harm the rest of the spacecraft.

11. **Launch loads and vibrations**: To withstand the enormous loads and vibrations during launch, the structure and the materials are chosen very carefully, so they will not fail during launch. As these materials and the structure are accurately determined by the team, it is a driving requirement.

Mission Profile 4

The mission profile defines the timeline as well as high level estimates of the spacecraft. To do this the general characteristics of the spacecraft as found in 4.1 need to be known as well as the desired orbital parameters from 4.2. With that data the total ΔV and propellant mass can be calculated in section 4.3. The launch vehicle is selected in 4.4. All pieces are put together in 4.5 to form a complete mission timeline.

4.1 First Vehicle Estimations and Comparison with Similar Spacecrafts

This section consists of the first vehicle estimations that will be used in further analysis. First, those estimations are performed and then a comparison with the spacecrafts listed in section 3.2 is provided as a matter of sanity check.

First Vehicle Estimates

The first vehicle estimations rely on statistical models studied in the context of AE1222-II [12]. The models were constructed based on statistical data to give a rough approximation of the main spacecraft characteristics, such as the dry mass, size, total power, reliability and cost. In Table 4.1 the chosen relations and the results of the estimation method are given. The input of the statistical relations are the mass or volume of the payload being 100-300kg and 50-250W respectively (the largest values were taken to guarantee a conservative approach).

Table 4.1: 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$

[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.

Note that as the type of power generation was not selected yet, the estimations of the total power and size were performed for both a solar powered and RTG powered system. The loaded mass was also computed as it is needed in order to estimate the vehicle size.

Comparison with Other Spacecrafts

To verify that the results from Table 4.1 they can be compared to similar spacecrafts. These can be found in section 3.2. However, from the lack of data this becomes difficult. Doing the calculations for the spacecraft from which enough data is known we can not draw any conclusions. For example, the launch mass is accurate when doing the calculation for the loaded mass of Galileo. It yields a launch mass of 1901 kg, which is only 14 % off. However, the calculation for Ulysses is off by 142%. So the verification of the data can hardly be done in this stage.

4.2 Orbital Parameters

In this section, the orbit of the spacecraft around Jupiter will be determined. First, a preliminary estimation of the orbit based on the past Juno mission by NASA will be made. Then, several points of importance will be given for further work on the orbit.

Determination of Orbital Parameters

Using Kepler's third law, knowing the period of the elliptical orbit and the relevant gravitational parameters, it is possible to find the semi-major axis of the ellipse. Using Equation 4.1 rearranged as Equation 4.2. Juno's spacecraft being a spacecraft that is similar to JUICE, it was decided to use its first planned polar 14-days orbit around Jupiter for preliminary estimations. Those estimations will be refined at a later stage of design.

$$\frac{a^3}{T^2} = \frac{\mu}{4\pi^2} \tag{4.1}$$

$$a = \left(\frac{\mu \cdot T^2}{4\pi^2}\right)^{1/3} \tag{4.2}$$

Using $T = 14 * 24 * 3600 = 86400s$ in Equation 4.2, the semi-major axis is found to be 288,270.32km. The pericenter is now chosen to be 7,000km as this gives some margin below the 10,000km required for the sensors to function properly. Using those information, the apocenter, semi-minor axis and eccentricity can be found using equations 4.3, 4.4 and 4.5.

$$r_a = 2 \cdot a - r_p \tag{4.3}$$

$$b = \sqrt{r_a \cdot r_p} \tag{4.4}$$

$$e = \frac{r_a - r_p}{2 \cdot a} \tag{4.5}$$

All relevant results can be found in Table 4.2.

Table 4.2: Orbital Parameters Estimation

Parameter	Value
Pericenter	7,000km
Apocenter	569,540.6km
Eccentricity	0.9757
Semi-major axis	288,270.3km
Semi-minor axis	63141km
Inclination	90°

Important Parameters

In the choice of the orbit, it is important to take different parameters into account, to ensure that the required life-time of the spacecraft is respected, but also that the results of the mission are good enough for further research. A list of the important items to be taken into account for any orbit selection is provided here:

- The ΔV of the mission should be kept to a minimum, as it means that less propellant is required. Less ΔV required means a less complex propulsion system and hence a spare in the mission costs. All orbital parameters have an influence to some extent on the ΔV required for the mission.
- The radiation belt in the neighbourhood of the giant planet is known as being a very harsh environment. This was confirmed by the measurements made by the Cassini spacecraft [13], the region within 300,000km of Jupiter is very dangerous radiation-wise and should be avoided as much as possible. To ensure that the spacecraft can operate for a duration of at least seven years 3.3, an orbit limiting the time intervals in the radiation belt should be chosen.
- The mission description requires to perform measures using magnetometers, spectrometers and imagers. The magnetometer is used to map the magnetic field, it would thus be relevant to use an orbit that makes the spacecraft change its 'altitude', in order to have a complete picture of the magnetosphere in the neighbourhood of the planet. In contrast, the spectrometers and imagers would require to be quite near the planet to make useful measurements.
- It is also given in the mission description that the nearest position to Jupiter should be at most at 10,000 km of the planet. This comes from the fact that some of the sensors need to operate very near the planet to produce useful measurements.
- The minimum operational time of the payload should also be taken into account to make sure that the time spent near the planet is adequate. A reliable set of data should be generated by the sensors at each orbit.
- The possible time of the eclipse by the planet should also be taken into account to ensure a large enough battery life and that this battery can be filled up to a certain level at each orbit.

All these parameters will be used in the detailed design phase to determine the exact orbit to be selected for the mission.

4.3 Total ΔV and Propellant Mass Estimation

Depending on the launch vehicle's capabilities the spacecraft will be placed either into an Earth orbit or already in a heliocentric orbit. To get into low earth orbit launch vehicles need to impart achieve around 9-10 km/s. Due to a relatively low expected launch mass the launch vehicle will most likely place the spacecraft immediately in the heliocentric orbit. In case the spacecraft needs to do this maneuver by itself the required ΔV from a low earth orbit (around 400km) to escape velocity can be calculated with Equation 4.6.

$$\Delta v = v_{escape} - v_{400km} = \sqrt{\frac{2\mu_{earth}}{a}} - \sqrt{\frac{\mu_{earth}}{a}} = 3.18\frac{km}{s} \tag{4.6}$$

Once in heliocentric orbit, a Hohmann transfer orbit from 1AU to Jupiter (5.2AU) is initiated. The ΔV related to that maneuver is given by Equation 4.7.

$$\Delta v = \sqrt{\mu_{Sun}\left(\frac{2}{r} - \frac{1}{a}\right)} - \sqrt{\frac{\mu_{Sun}}{r}} = 8.8\frac{km}{s} \tag{4.7}$$

Where r is the radius around the sun the spacecraft departs from and a is the semi-major axis of the transfer orbit. Once the spacecraft has arrived at its apohelion it will be inserted into Jupiter's orbit. Similar to Juno and Galileo, this will approximately take 600 m/s of ΔV [14]. Once this orbit is achieved one last maneuver will be required to get into the final orbit that will provide the conditions for the scientific data to be collected. This simple Hohmann transfer is relatively inefficient and requires high amounts of ΔV. The budget will thus be oriented towards the past missions analysed previously and similar trajectories using gravity assists to minimize the required ΔV.

As such, Juno required around 1300m/s distributed over 3 maneuvers to arrive in a highly elliptical parking orbit from heliocentric orbit [14]. Therefore, a similar ΔV of 1300m/s for the transfer is estimated for the JUICE mission. This means that a similar gravity assist trajectory is assumed to be available, taking around 5 years to arrive at Jupiter. Once in the highly elliptical parking orbit, the final scientific orbit needs to be reached, which similarly to Juno will require another 450m/s. Finally, station keeping as well as attitude control require around 30m/s per year, adding up to 360 m/s for the entire mission [12]. At the end of the operational life, it is required to deorbit the spacecraft into Jupiter to prevent contamination of the various moons and to get some last bits of data from the inner atmosphere. This demands another 75m/s.With the common margins [15] the a total of 2636m/s. The total ΔV breakdown is presented in Table 4.3.

Table 4.3: Summary of the total ΔV breakdown

Maneuver	ΔV
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 * 30 = 360
Deorbiting	75
Total	2185

As the mission will be a large distance from the Sun and thus not having an abundance of solar irradiance, chemical propulsion for the spacecraft will be used instead of ion engines. Using the rocket equation, the necessary propellant mass can be estimated. Assuming I_{sp} of 300s for a chemical bi-propellant engine [12] it can be calculated using Equation 4.8 and the dry mass estimated in 4.1:

$$M_p = M_e e^{\frac{\Delta v}{I_{sp} g_0}} - M_e = 1620.2kg \tag{4.8}$$

Thus the final first order estimated mass of the spacecraft will be 2686.7 kg, when released into heliocentric orbit by the launch vehicle.

4.4 Launch Vehicle

The launch vehicle is one of the most important facets of the mission, besides during launch being the most spectacular part. The launch vehicle is of paramount importance, since it will carry the spacecraft from Earth into space. Therefore, one should analyse with great detail, what the most efficient and effective launcher for the mission is.

Launcher and Launch Site

When comparing launch vehicles to find the best suited for this mission, there multiple criteria to keep in mind. Firstly, the reliability of 0.9 that ought to be met, so it should be an experienced launch vehicle with none to a few failures. Furthermore, the launch vehicle needs to be able to launch deep space spacecrafts into space, so the launcher should have experience with these kind of spacecrafts. Moreover, the most recent and successful orbiter around Jupiter is the Juno orbiter, hence the team could use the information available to its advantage. Accordingly, the Atlas V 551 launch vehicle is the best fit keeping this criteria in mind. When looking at the reliability, it has only had successful launches and a lot of deep space missions thus far, so it certainly meets that criteria. Also, the Atlas V 551 is the launcher that brought Juno into space, so this is a recent successful launch vehicle. The first '5', in 551, stands for a 5-meter fairing, so there will be enough space for the payload. The second 5 signifies '5' SRBs (solid rocket-boosters), which ensures that the launcher will deliver enough thrust, to bring the JUICE orbiter into space. Lastly, the '1' represents the one Centaur engine that should be enough. Cape Canaveral will be the most logical choice of launch site, as this is where the Atlas V mainly was launched from.

Maximum launcher loads

	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 seperation	3g	0g	0.5g	0.5g
Booster engine cut-off	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

Payload Adapter

Type of standard Atlas V Adaptor	A937	B1194	D1666	F1663
Mass	69.7-114.4kg	61.8-106.5kg	61.4-106.1kg	116.1-160.8kg
Height	0.7366-1.143m	0.5842-0.9906m	0.6604-1.0668m	0.9431-1.2987m
Separation System	LSPSS937 Low-shock Marmon Type Clampband	LSPSS1194 Low-shock Marmon Type Clampband	LSPSS1666 Low-shock Marmon Type Clampband	Four Separation nuts
Diameter	0.9453m	1.2150m	1.6661m	1.663m

Launcher Minimum Frequencies

Since there was no information present regarding the atlas V minimum fundamental frequencies, the frequency requirements of Ariane 6 A62 was used since both launch vehicles have similar characteristics, therefore the minimum lateral frequency is 6Hz whilst the minimum longitudinal frequency is 20Hz. [16]

Ascent profile

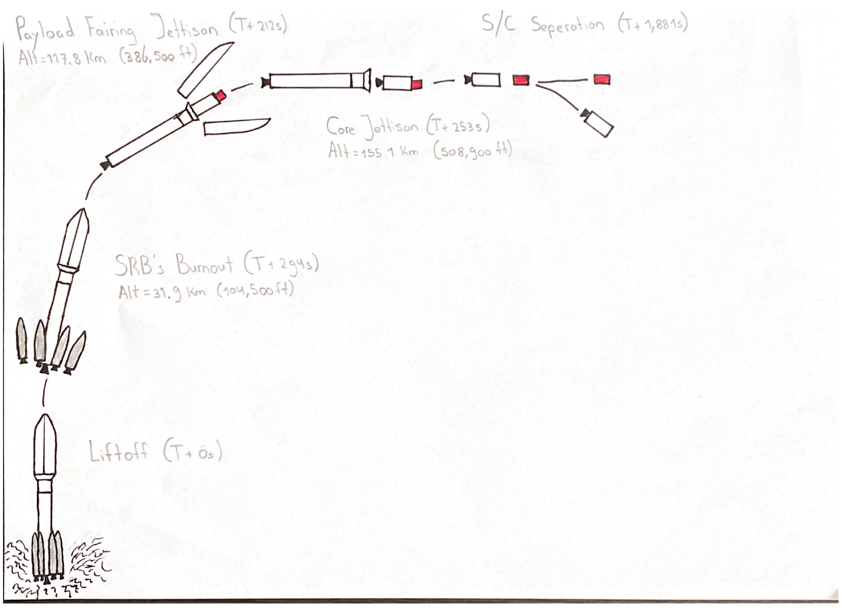
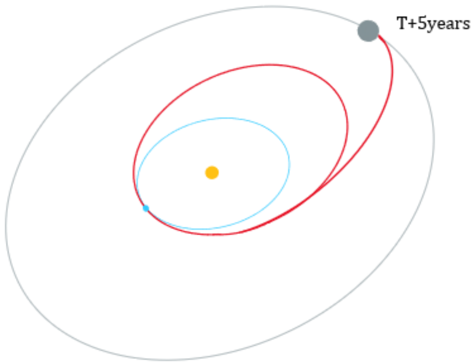
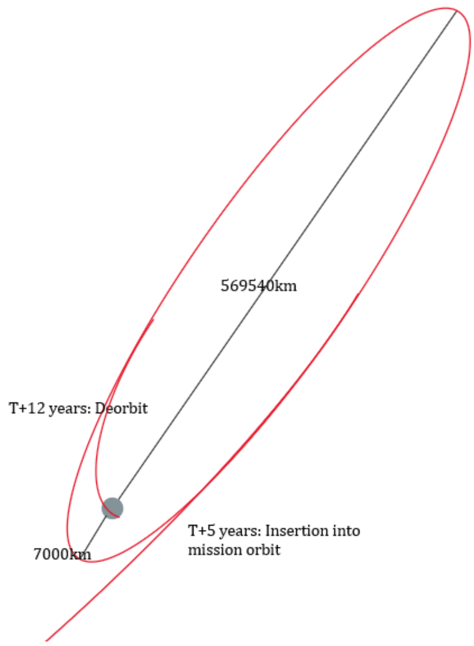


Figure 4.1: Ascent profile Atlas V-551

4.5 Mission timeline



(a) Earth to Jupiter transfer



(b) Jupiter insertion and end of mission

Figure 4.2: Mission trajectory

In 2025 Atlas V 551 will lift off from Cape Canaveral on its way to delivering JUICE to its destination Jupiter. After the 5 solid rocket boosters have seperated and the core stage shuts off at T+ 253s, the Centaur upper stage will start with placing JUICE into a preliminary parking orbit around earth. Once in the right position, Centaur will reignite its engine to push JUICE to escape velocity and away from the Earth. Once burn out has occured, the upper stage will separate and JUICE is on its way to Jupiter, it will then deploy all necessary items and switch to transfer operating mode. It will use a similar trajectory as Juno, gravity assisting on varying bodies as can be schematically seen in Fig. 4.2a. The exact trajectory is still to be planned, the planned time for this trajectory is around 5 years and will require 2-3 burns of the on-board propulsion system.

Once JUICE is approaching Jupiter there will be two maneuvers to enter its final mission orbit as can be seen in Fig. 4.2b. First a small burn to capture JUICE into a preliminary parking orbit. After that the orbit will be adjusted to be able to collect the scientific data to start its main mission phase at around T+ 5 years. This orbit will be maintained for the 7 remaining years of its mission until T+ 12 years, when the command for de-orbiting will be sent. This causes the spacecraft to adjust its orbit to attain a periapsis below Jupiter's surface, thus ending the mission.

Initial Sizing 5

This chapter concerns the initial sizing of the spacecraft. A general budget will be made for both mass and power in section 5.1. This will provide a general framework in which each subsystem can be designed. Additionally multiple spacecraft architectures will be sketched using CATIA in 5.2. For each of these, the mass moment of inertia will be computed in section 5.3, which is of utmost importance for the later design of the attitude determination and control system.

5.1 Mass and Power Budgets

Using statistical models provided in [12] p. 260, a mass budget was estimated and is presented Table 5.1. At this design stage the details of each component are still unknown, making preliminary estimations highly inaccurate. Thus a high mass margin of 30% was selected as suggested by [12], to ensure the spacecraft will be capable of completing its mission even though one or more of the systems may have become heavier than primarily estimated. The estimated dry mass of 619.9-1066.5kg and a payload mass of 100-300kg determined in section 4.1 was used to obtain the results given in Table 5.1. This means that the input masses are 519.9-766.5kg.

Table 5.1: Mass Budget

Subsystem	**Percentage**	**Mass**
Propulsion	19%	98.7-145.6kg
ADCS	8%	41.6-61.3kg
Communications	7.7%	40.0-59.0kg
Thermal	4.8%	24.9-36.8kg
Power	22.4%	116.4-171.7g
Structures	23.0%	119.5-176.3kg
Harness	9.8%	50.9-75.1kg
CD and H	5.3%	27.5-40.6kg
Total S/C Bus		519.6-766.5kg
30% Margin[12]		155.9-230.0kg
Total S/C Bus with margin		675.5-996.5kg

A similar procedure using table 8 from p. 262 in [12] was performed for the power budget in Table 5.2. The estimated total power of 178.5-404.5W from section 4.1 was used and similarly to the mass, a margin of 30% was chosen.

Table 5.2: Power Budget

Subsystem	**Percentage**	**Power**
Propulsion	4%	5.1-6.2W
ADCS	11%	14.1-17.0W
Communications	30%	38.6-46.4W
CDS	15%	19.3-23.2W
Thermal	33%	42.4-51.0W
Power	2%	2.6-3.1W
Structures	5%	6.4-7.7W
Total S/C Bus		128.5-154.6W
30% Margin		38.6-46.4W
Total S/C Bus with margin		167.0-200.9W

Note that in Table 5.2, the total S/C bus power is equivalent to the estimated total power without the payload.

5.2 Spacecraft Architecture Possibilities

With the gathered information about the spacecraft from the previous sections three different spacecraft architectures are designed. The drawings of which can be found in Appendix A. Two of them are solar powered and one of them has a RTG. For the first two design option the solar panel area has to be estimated first.
The maximum total power estimation came out to be 404.5W. The solar flux near Jupiter is about $50.26 W/m^2$ on average [3] and the orbital period is earlier estimated to be 86400s. So, assuming that the spacecraft is in direct contact with the sun (sun-synchronous polar orbit) throughout the orbit, the total power to be produced by the solar panels comes out to be 404.5W. The efficiency of the solar panels is estimated to be around 28% (taken from the Juno spacecraft). The solar panels of the spacecraft will not be able to point directly to the sun at all times and the solar panels may degrade a lot over time due to the very harsh environment of Jupiter. So for these reasons, a safety factor of 0.5 is taken in this preliminary phase. The result is a solar panel area of almost 60 m^2.

Design Option 1 For the primary design option, a rectangular box is chosen with four solar panel arms of around six meters long. Furthermore, the arms are easily foldable now. The volume of the bus is taken from the estimations and a conservative $20m^3$ is taken for this preliminary design. The antenna diameter is taken to be 2.5m. This is again a very rough estimate based on the Juno spacecraft and it is very likely that this dimension will change later on in the design process. For orbit operations, seven thrusters are used in the primary design option, a bigger one and six smaller for small attitude corrections. The small thrusters could be replaced by spinning wheels, or with a spin stabilisation mechanism. However, for now thrusters are chosen instead of spinning wheels, because the spinning wheels require more energy, which means that the solar panels need to be even bigger than they already are. It is in this case better to have a bit more propellant then. The different scientific sensors are put on the outsides of the spacecraft. On the top there is a ten meter tall fully retractile boom (Fig.A.2) with a magnetometer on top, this is to avoid interference between the rest of the S/C devices and the measurements of the magnetometer. Furthermore there are two imagers and a spectrometer on the bottom of the spacecraft pointing to Jupiter for atmospheric research and there is a spectrometer for analysing plasma particles and ions (Fig.A.1). The spacecraft also has four small sun sensors on top of each solar panel arm. These will register the angle of incidence of the sun. So the spacecraft is able to let the panels point to the sun in the best way possible.

Design Option 2 For the second design option a cylindrical bus is chosen on which three $10m \cdot 2m$ solar panels are mounted (Fig. A.3). The downside of this design compared to option 1 is that, it has a higher chance of solar panel failure as it is longer and has more mechanisms. The bus again has a volume of around 20 m^3 and the antenna also has the same dimensions as in design option 1. For orbit operations this spacecraft has an engine on the bottom and it is spin stabilised. The scientific instruments are put on the bottom of the spacecraft like spacecraft option 1 and the magnetometer is placed on top of a 10 meter long, fully retractable boom (Fig.A.4).

Design Option 3 The third spacecraft option is powered by a GPHS-RTG [17]. The main advantage of a RTG powered spacecraft is that there are no solar panels needed, which have to be very big deep in space, but only two 1.2m long reactors (Fig.A.5). One of these reactors produce at least 285W [17] at beginning-of-mission, so two of them will be sufficient to produce 404.5W for the entire mission duration . The downside of this RTG powered spacecraft is that the volume of the bus is very big. The preliminary estimations predicted a volume between $43.5m^3$ and $129.1m^3$, which is a really big spread. To be conservative a volume of $120m^3$ is taken for the third design option. So the bus of the spacecraft is designed to be 5m x 5m x 5m. Furthermore, the antenna has the same dimension as in the other designs. For orbit operations this spacecraft has a main engine mounted on the bottom side and it has three reaction wheels on the sides of the spacecraft for attitude control. The scientific instruments are attached in the same way as the ones in the other designs. So the imagers and spectrometers are mounted on the bottom pointing to Jupiter and the magnetometer is mounted on a 10m tall boom on top of the spacecraft (Fig.A.6).

5.3 Mass Moments of Inertia of the Possible Architectures

In this section the moments of inertia will be calculated for simplified versions of the three preliminary spacecrafts. The mass of the spacecraft used is 996.5kg and the mass of all solar panels is 283kg.

MMOI Estimates Design 1 First the Mass moment of inertia will be calculated through the paper from the front view of the first sketch (solar panels deployed). Assumptions:

- Mass is uniformly distributed
- Only MMOI of main components is taken into account (antenna, battery.... etc is not taken into account)

$m_{bus} = 996.5kg, m_{pan} = 70.75kg.$

$$I_z = \frac{1}{3}m_{pan}(2.5^2 + 6^2) + \frac{1}{6}m_{bus}(2.5^2) + 4m_{pan}(4.75^2) = 8419.6kgm^2 \tag{5.1}$$

Then The mass moment of inertia of the front view is taken through the page as well. however to preform the parallel axis theorem the COM has to be found. after calculation the COM was found to be 2.148 from the bottom of the spacecraft.

$$I_x = \frac{m_{bus}}{12}(2.5^2+3.5^2)+\frac{1}{6}m_{pan}(2.5^2)+\frac{1}{6}m_{pan}(6^2)+m_{bus}(0.3984^2)+m_{pan}(1.3765^2)+2m_{pan}(3.5^2) = 3745.1kgm^2 \tag{5.2}$$

The same process is repeated for the first sketch however in an undeployed solar panel state

$$I_z = \frac{1}{6}m_{bus}(2.5^2) + \frac{1}{3}m_{pan}(2.5^2) + 4m_{pan}(1.775^2) = 2077.0kgm^2 \tag{5.3}$$

however for the front view of the spacecraft the COM from the bottom of the spacecraft is 1.47m

$$I_x = \frac{1}{12}m_{bus}(2.5^2+3.5^2)+\frac{1}{6}m_{pan}(6^2)+\frac{1}{6}m_{pan}(6^2+2.5^2)+m_{bus}(0.28^2)+4m_{pan}(0.97^2) = 2803.4kgm^2 \tag{5.4}$$

MMOI Estimates Design 2 First, the moment of inertia around the axis through the center of the spacecraft out of the paper will be estimated (Fig.A.3), which we call the z-axis. This is assumed to be the center of mass due to symmetry. It will be assumed that the spacecraft consists of the cylindrical bus with uniform density and three rectangular solar panels. The moment of inertia through this z-axis will then be:

$$I_z = \frac{1}{2}m_{bus}r_{bus}^2 + 3(\frac{1}{12}m_{pan}(l^2 + b^2) + m_{pan}d^2)) \tag{5.5}$$

In which I_z is the moment of inertia around the z-axis, r_{bus} is the radius of the bus, m_{bus} is the estimated mass of the bus, m_{pan} is the mass of one solar panel, l is the length of a solar panel, b is the width of a solar panel and d is the distance from the center of mass of the solar panel to the center of mass of the bus. So the first term is the moment of inertia of the satellite bus and the other terms are for the three solar panels. The mass of the bus was estimated to be 997kg. For the solar panel size it is assumed that the density is the same as the density of Juno's solar panels. The total mass of Juno's $72m^2$ solar panels was around 340kg [18]. So based on that this spacecraft with a solar panel area of 60 m^2 will have a total solar panel mass of around 283kg. Filling in the values and ignoring the lengths of the solar panel arm joints (d = 1.5m + 5m = 6.5m), the following moment of inertia is obtained:

$$I_z = \frac{1}{2} \cdot 997 \cdot 1.5^2 + 3 \cdot (\frac{1}{2} \cdot \frac{283}{3}(10^2 + 2^2) + \frac{283}{3} \cdot 6.5^2) = 27794kgm^2 \tag{5.6}$$

When the spacecraft is undeployed the solar panels are folded along the length of the bus. For the moment of inertia around the z-axis, it will be assumed that the solar panels are thin walled plates with their center of masses at the rim of the bus oriented like the top view in Fig. A.2, but then with the cylindrical bus and the three solar panels of design 2. So the mass moment of inertia will be:

$$I_z = \frac{1}{2}m_{bus}r_{bus}^2 + 3(\frac{1}{12}m_{pan}b^2 + m_{pan}r_{bus}^2) \tag{5.7}$$

This gives an I_z of 2324 kgm^2, which is significantly smaller then in the deployed situation. Now consider the axis through the center of mass of the spacecraft into the paper in Fig.A.4 and along the top solar panel in Fig.A.3. This is taken to be the yc-axis. Now, besides the solar panels and bus, the magnetometer plays a role on the moment of inertia and center of mass. The z-coordinate of the center of mass has to be determined here. Let the bottom of the bus equal z = 0. The mass of the magnetometer with the boom was estimated to be 70kg based on the 2.9m and 10kg magnetometer MAGBOOM [19]. Furthermore it was assumed that its center of mass is in the center of the boom so at z = 3m + 5m = 8m. So using the dimensions in Fig.A.4 and noting that the center of mass of the bus is at z = 1.5m, the center of mass of design 2 deployed came out to be at z = 2.26m. For the moment of inertia calculation around this y-axis a simplification is made for the two solar panels at the sides in Fig.A.3. These are assumed to be positioned perpendicular to the yc-axis with their long axis. So the moment of inertia around this axis will be:

$$\begin{aligned} I_{yc} = \frac{1}{12}m_{bus}(3r_{bus}^2 + h^2) + m_{bus}(2.26 - h/2)^2 + \frac{1}{12}m_{mag}l_{mag}^2 + m_{mag}(l_{mag}/2 + h - 2.26)^2 \\ + \frac{1}{12}m_{sol}b^2 + m_{sol}(h - 2.26)^2 + 2(\frac{1}{12}m_{sol}l^2 + m_{sol}(h - 2.26)^2) \end{aligned} \tag{5.8}$$

Here, h is the height of the bus and the height of the solar panel attachment point with respect to z = 0, m_{mag} is the mass of the magnetometer, l_{mag} is the length of the magnetometer, m_{sol} is the mass of one solar panel, b is again the width of a solar panel and l is the length of a solar panel. So I_{yc} = 6533 kgm^2. For the undeployed case the magnetometer can be ignored as it is retracted inside the bus. The solar panels are retracted along the bus...

Finally the mass moment of inertia about the x-axis can be calculated. The mass moment of inertia of the body and antenna about the x-axis is the same as about the y-axis, but the mass moment of the solar arrays is different due to the geometry of the design.

$$I_x = \frac{1}{12}m_{bus}(3r_{bus}^2 + h^2) + m_{bus}(2.26 - h/2)^2 + \frac{1}{12}m_{mag}l_{mag}^2 + m_{mag}(l_{mag}/2 + h - 2.26)^2$$
$$+ \frac{1}{12}m_{sol}b^2 + m_{sol}(h - 2.26)^2 + 2(\frac{1}{12}m_{sol}l^2 + m_{sol}(h - 2.26)^2) \quad (5.9)$$

MMOI Estimates Design 3 First the Moment of inertia will be calculated at an axis through the paper in the top view of the third sketch. $m_{bus} = 996.5kg, m_{mag} = 7kg$.

Assumptions:

- Mass is uniformly distributed.
- Only MMOI of main components(solar panels,body,magnetometer) is taken into account (antenna, battery.... etc is not taken into account).

$$I_z = \frac{1}{6}m_{bus}(5^2) = 4152.1kgm^2 \quad (5.10)$$

Secondly the Moment of inertia will be calculated from the side of view with an axis through the paper once again.

$$I_y = \frac{1}{6}m_{bus}(5^2) + \frac{1}{12}m_{mag}(10^2) + m_{mag}(7.5^2) = 4210.4kgm^2 \quad (5.11)$$

6 Conclusion

This report aims at describing the process for the initial sizing of the JUICE mission. This process consists of different steps: collecting data from similar spacecraft, identifying the driving requirements for the mission, estimate the spacecraft's main parameters such as dry mass, power and size, describe the mission profile, size and sketch some of the spacecraft architecture. The mission also needs to meet the top-level requirements set for the mission: the payload mass should be at maximum 300kg, the total power consumption must not exceed 800W, the temperature of the payload should lie between 150K - 250K and the duration of the mission is set to be between 7 and 12 years.

To start off, the main objectives for the mission need to be determined, therefore similar missions were researched. After collecting information about the requirements and the elements of those missions, the objectives of the JUICE mission could be formulated along with the corresponding functionalities. In addition, a list was made with all the design parameters of the orbiter that were already available from the mission description. After the functional analysis the requirements can be established. In total 35 design requirements were formulated for the mission. To lay more focus on the important parameters the driving requirements have been defined as well. After this step the mission profile could be set up. The mission profile enables the team to plan ahead for the entire mission. The elements, which have been determined, are the first vehicle estimates, the orbital parameters, the delta V budget and the launch vehicle. From the orbital parameters and first vehicle estimations the propellant mass can be approximated, this was found to be 1620.2 kg. For the launch vehicle the Atlas V was chosen. Finally the mission timeline and ΔV budget can be found in the following Table.

Table 6.1: ΔV budget and the mission timeline

Maneuver	ΔV	**Timeline**
Transfer from Heliocentric to Jupiter including Gravity assists based on previous missions	1300	T + 1 day
Initial Jupiter insertion	450	T + 5 years
Station keeping and attitude control over 12 year mission life	12 * 30 = 360	T + 5 years
De-orbiting	75	T + 12 years
Total	2185	T + 12 years

With the information from the mission profile the initial sizing could be finalized. Using statistics a mass and power budged could be formed. These can be found in the following table.

Different design options for the spacecraft have been taken into account. The architecture of the spacecraft highly depends on the type of power generation used. Power generated by solar panels and power generated by RTG's are regarded as the best options. For this three different design were made. Two using solar arrays and one using RTG's for power generation. The first design consists of a rectangular box with four solar panels of length 6 meters attached to it, resulting in an estimated volume of $20m^3$. Propulsion is achieved thanks to six small thrusters and a bigger one. The second design of a volume of $20m^3$ consists of a cylindrical bus with three, 10 meters long panels and a single

Table 6.2: Mass and Power budgets

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

engine positioned at the bottom. The RTG - powered design is the biggest one having a volume of $120m^3$ and a single engine mounted at the bottom. The choice between the three design options is of main importance and cannot be made yet. The choice will be made in a later stage of the design.

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CATIA Drawings A

In this appendix, the drawings of the three possible design architectures are given. For design option 1 a deployed and undeployed CATIA drawings are given (A.1, A.2), the top and side sketches of design 2 (A.3, A.4) and the ones of design 3 (A.5, A.6)are given.

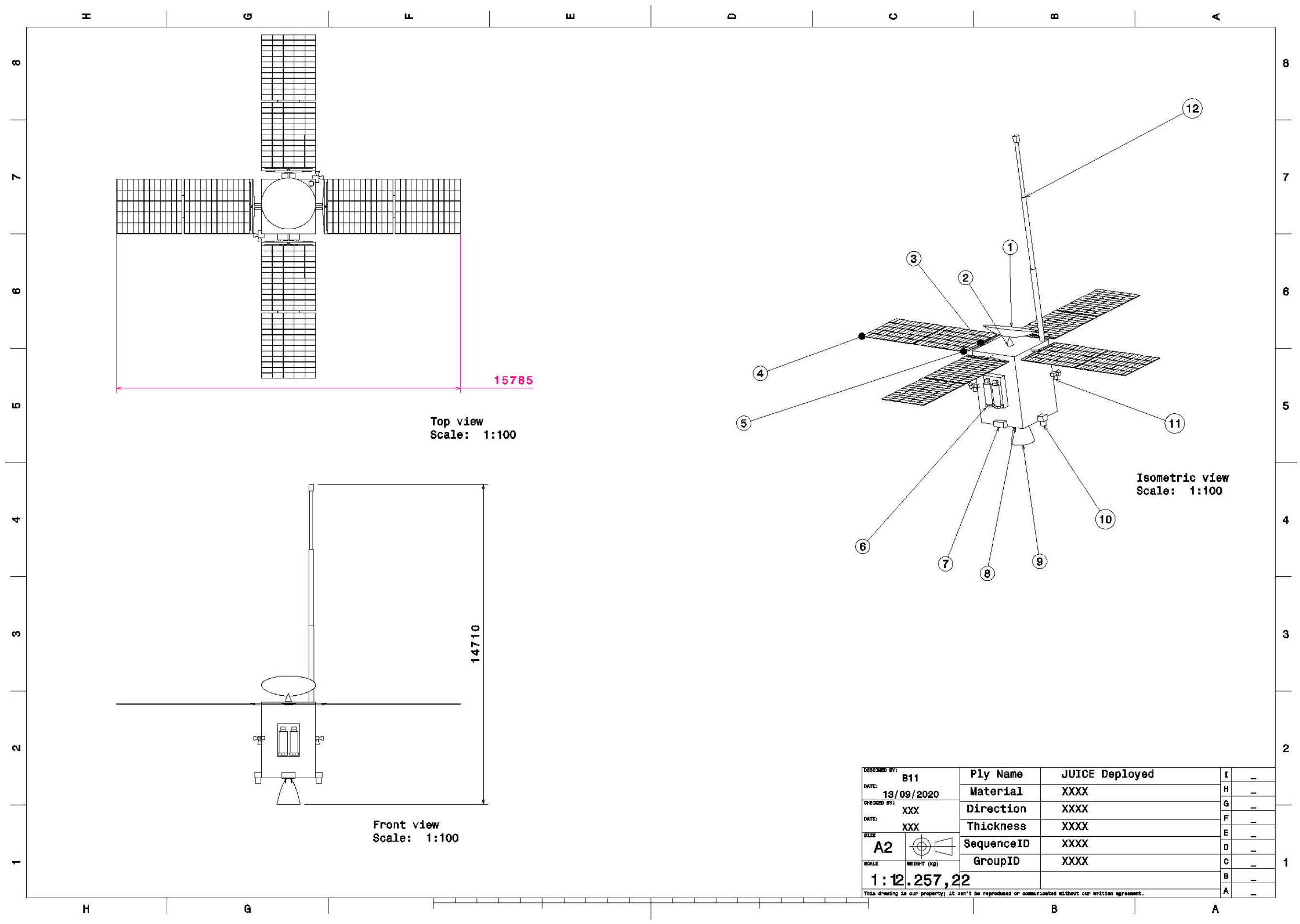

Figure A.1: Spacecraft 1 Deployed

Table A.1: Parts Present in the Assembly Drawing

Part number	Part name	Quantity
1	Antenna	1
2	Antenna Joint	1
3	Sun Sensor	4
4	Solar Panel	4
5	Solar Panel Joint	4
6	Battery	1
7	Spectrometer	2
8	Satellite Bus	1
9	Main Thruster	1
10	Imager	2
11	ADCS Thrusters	2
12	Magnetometer Boom	1

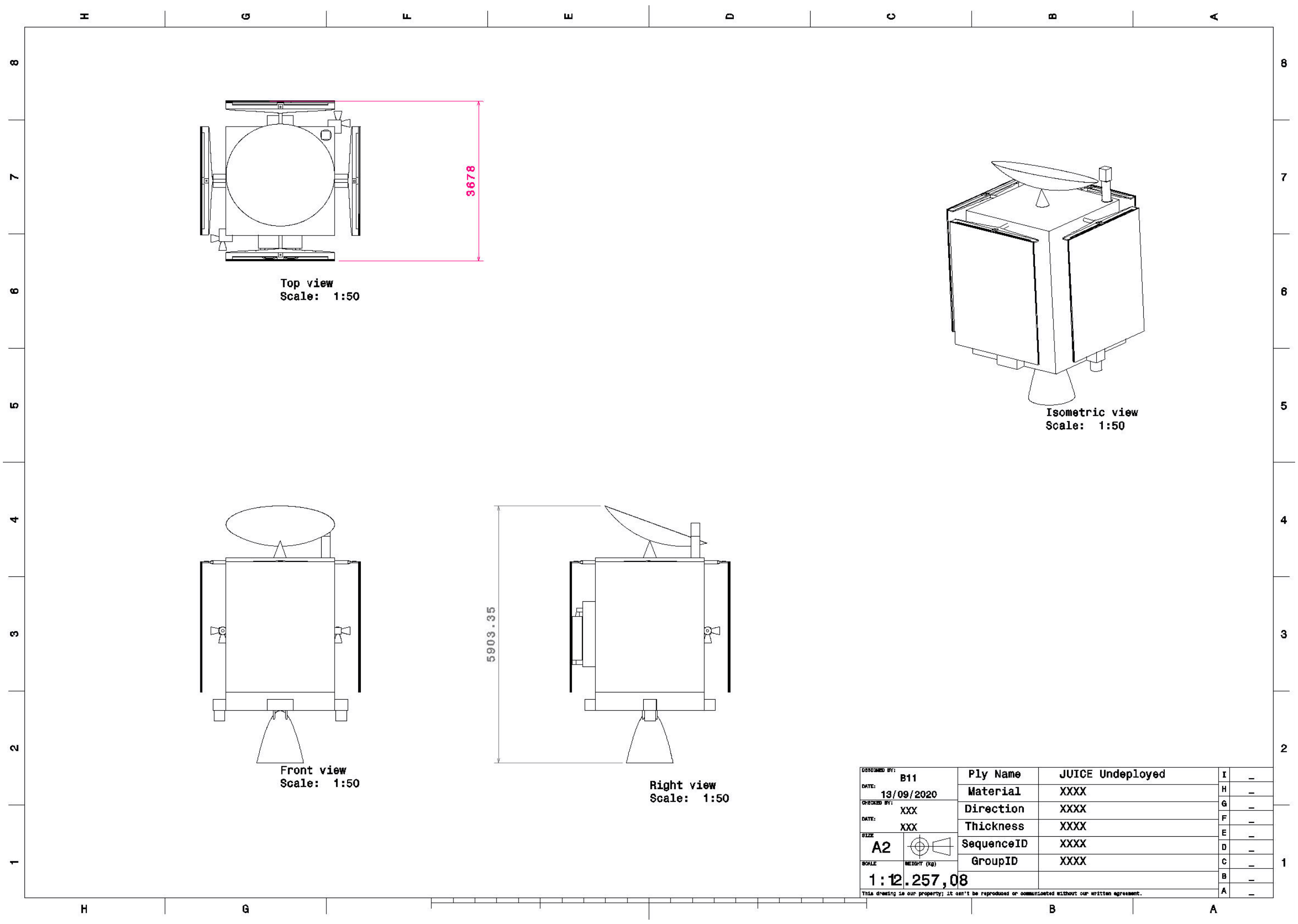

Figure A.2: Spacecraft 1 Undeployed

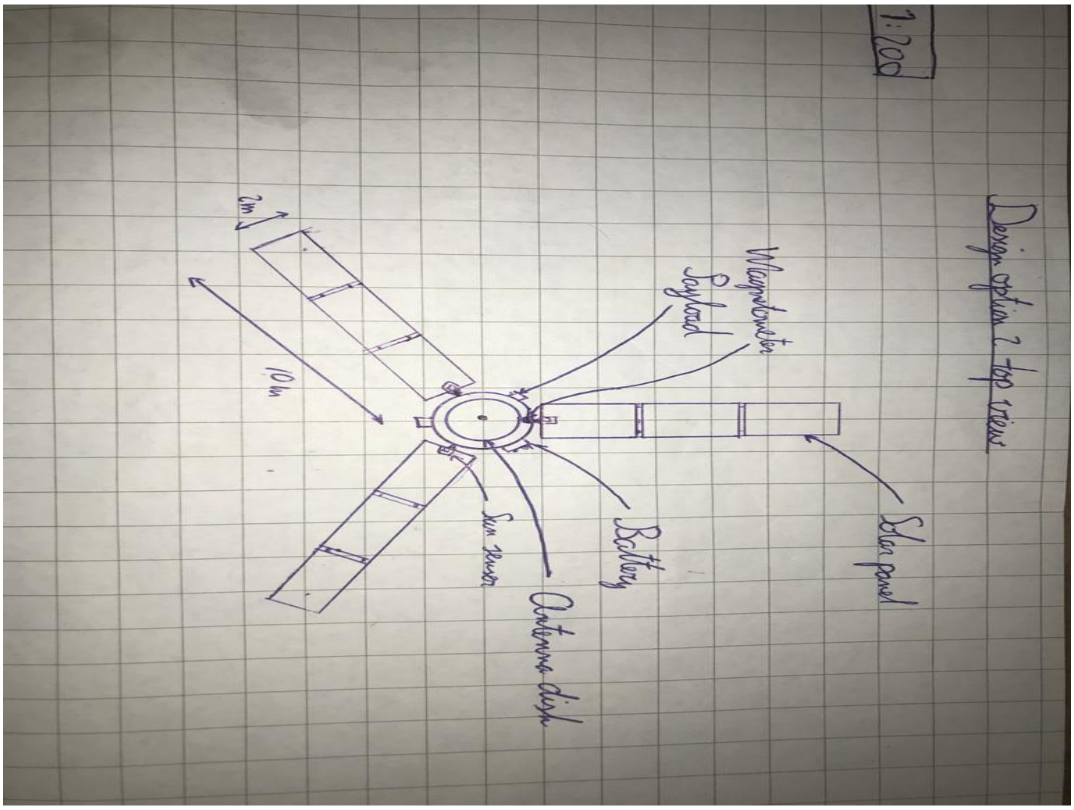

Figure A.3: Spacecraft 2 Top View

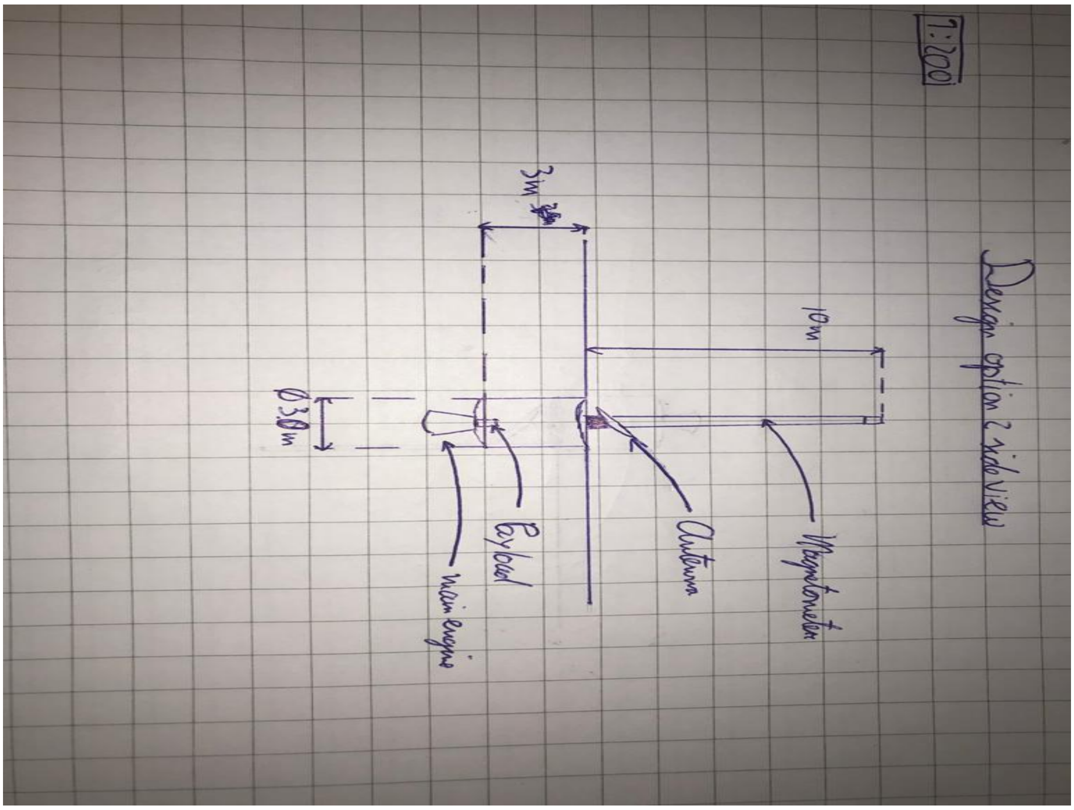

Figure A.4: Spacecraft 2 Side View

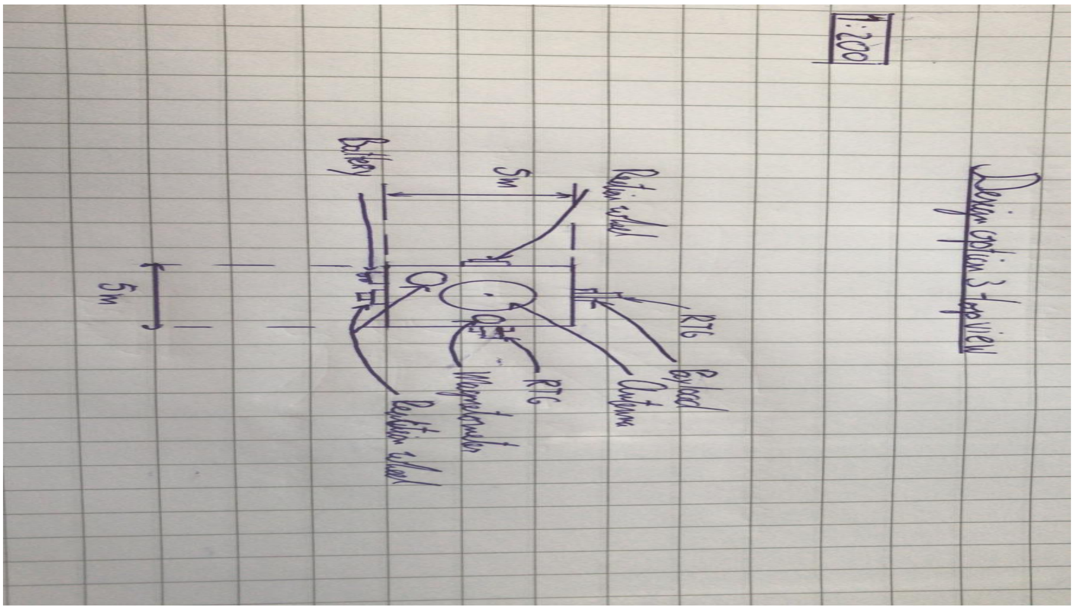

Figure A.5: Spacecraft 3 Top View

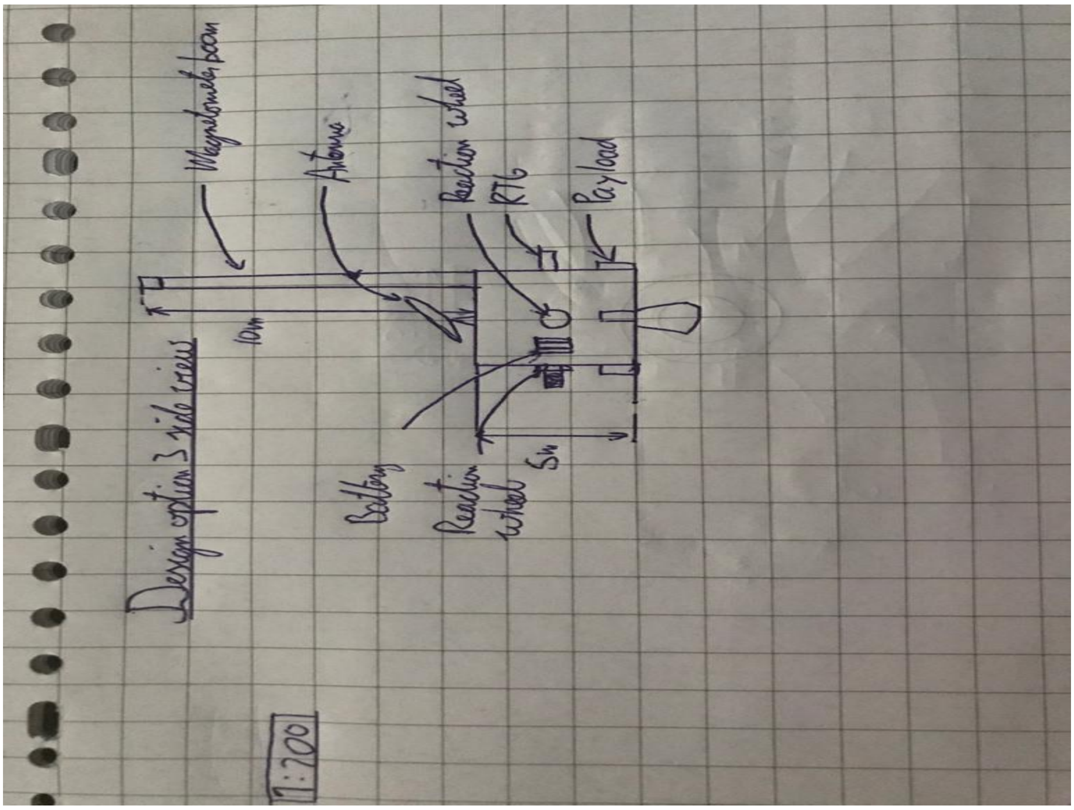

Figure A.6: Spacecraft 3 Side View

B Task Distribution

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

Table B.1: Task distribution per member

Team member	**Deliverables**
Jonatan	1.1.3, 1.1.4, 1.2.2, 1.2.3, 1.4.2, 1.4.3
Tarek	1.1.1, 1.2.2, 1.3.4, 1.4.3
Stefano	1.1.3, 1.1.4, 1.2.3, 1.2.4, 1.3.4, review grammar/punctuation
Niklas	1.1.2, 1.3.3, 1.3.4, 1.3.5, 1.4.1, chapter introductions, project leader
Antonio	Introduction, conclusion, summary, review technical writing
Lorenz	1.1.4(added explanations), 1.1.5, 1.2.1, 1.2.3, 1.3.1, 1.3.2, proof reading, technical writing
Silvano	1.1.1, 1.1.2, 1.2.1, 1.4.2, project secretary
Sam	1.1.1, 1.3.1, conclusion, summary, proof reading