# Space Mission Analysis and Design(2nd year, MEAer)

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# Preliminary Mission Design - Mission SMADGROUP2

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**Abstract**

SMADGROUP2 (Saturn Magnetosphere, Atmosphere and Dust from Gaps and Rings with Onboard Utilities and Probing 2) mission to Saturn aims to advance our understanding of the Planet's atmosphere, rings, moons and magnetosphere by leveraging state-of-the-art instrumentation and modern space technology. Building on the legacy of the Cassini-Huygens mission, this new mission will focus on high resolution imaging in both visible and infrared spectrum to map Saturn's surface, analyze its atmospheric composition and investigate its magnetosphere, focusing on its interaction with the solar wind by using magnetometers and a solar wind monitor. Additionally, the mission seeks to explore the composition and dynamics of Saturn's rings and collect dust samples for on-board analysis. Titan observations are planned due to the high number of flybys expected around the moon. The mission's scientific goals are mapping Saturn in both visible and infrared spectrum, investigating Saturn's magnetosphere and its interaction with the solar wind, exploring Saturn's rings and gaps composition and dynamics with onboard samples analysis and observing and imaging the Titan moon during multiple flybys. The spacecraft will carry an array of high-performance, radiation-hardened instruments, including high resolution visible and infrared cameras, infrared and plasma spectrometers, magnetometers and dust collector and analyzer. With advanced on-board memory and data compression systems, the spacecraft will be capable of storing and transmitting large amounts of scientific data over extended periods, ensuring efficient use of available bandwidth when in communication with Earth. The mission's operational orbit will be optimized for both equatorial and polar coverage, enabling a global survey of Saturn's magnetosphere and rings while capturing high-resolution images of the Planet. This mission will provide unprecedented insight into the dynamics and evolution of Saturn's environment, building on past missions while utilizing modern technological advancements to achieve new levels of scientific discovery.

*Keywords:* Saturn, atmosphere, rings, gaps, imaging, mapping, magnetosphere, solar wind, dust collection, on-board analysis, Titan, high-resolution, space mission.

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## 1. Introduction

### *1.1. Mission objectives and goals*

The proposed mission has one main objective and five independent secondary objectives. The main objective consists of mapping the surface of Saturn in the visible spectrum to get detailed information about atmosphere, cloud structures and surface features. Furthermore, the mission has multiple secondary objectives. First, the spacecraft shall take close pictures of the Titan's surface (during multiple flybys). Second, it shall collect and analyze particles from gaps between rings and from the thin rings (E ring, F-G rings gap, Cassini division) during the deorbiting phase by using onboard instruments. Third, it should investigate Saturn's magnetosphere focusing on its interaction with the solar wind. The fourth secondary objective is the investigation of the whole surface of the planet in the infrared spectrum to get more details about thermal properties of the Planet's surface. Fifth, it shall analyze rings' characteristics by observing them in the visible and infrared spectrum.

Establishing success criteria and mission metrics is essential for evaluating whether the mission meets its objectives. Success criteria define the key achievements necessary for the mission to be considered as a success, and are described in the following. The spacecraft shall achieve at least 95% global mapping of Saturn's surface in the visible spectrum. It shall achieve a thermal mapping of at least 85% of Saturn's atmosphere in the infrared spectrum. It shall collect and analyze detailed information of magnetosphere and solar wind interaction from the whole region of magnetosphere in the operational orbit. It shall provide high-resolution measurements of Saturn's magnetic field variations over the course of a whole Saturnian year (7 terrestrial years). It shall obtain high-resolution visible and infrared images of Saturn's rings and gaps. It shall collect and onboard analyze dust samples from E ring, F-G gap and Cassini division ensuring no sampling mix. It shall capture high-resolution images of at least 3 different relevant spots on Titan's surface or atmosphere such as hydrocarbon lakes and seas, or cryovolcanoes. It shall achieve 90% of the planned data transmission back to Earth. It shall maintain a stable orbit for at least 7 years while conducting scientific observations. It shall execute all planned orbital maneuvers to optimize flybys as predicted.

Mission metrics quantify the performance of the spacecraft and instruments, helping to assess the efficiency, effectiveness and health of the spacecraft during the mission, and are collected in the following. The spacecraft shall achieve at least 10 km/pixel resolution for global mapping of Saturn's surface in visible spectrum. It capture thermal data with a resolution of at least 10 km/pixel to detect temperature gradients in Saturn's atmosphere. It shall record measurements of magnetic field strength with a sensitivity of 0.1 nT across the magnetosphere. It shall analyze dust particles with minimum size form 0.1 to 1 micron It shall ensure all critical instruments are operational for $> 99\%$ of the planned operational time. It shall maintain average power consumption within the 5% of the predicted nominal power levels ensuring RTGs provide adequate power generation. It shall ensure that the spacecraft's internal temperature remains within the operational range of -20°C to +40°C exception for instrumentation that require specific cooling.

### *1.2. Background and motivations of the mission*

Saturn continues to be a focal point of scientific interest due to its complex system of rings, moons, magnetosphere and atmosphere. Previous missions, most notably the Cassini-Huygens mission, have provided significant insights into the planet's structure, atmospheric dynamics and its moons, but many aspects of Saturn's environment remain unexplored or only partially understood. This new mission aims to improve the discoveries made by Cassini, using modern instrumentation and techniques to probe deeper into Saturn's atmosphere, rings, magnetosphere, and dust environment. Saturn's atmosphere is known for its dynamic weather system, with storms and cyclonic features like the famous hexagonal storm at its north pole. Despite the Cassini mission's detailed study of these phenomena, many aspects of the planet's meteorology and long-term weather patterns remain poorly understood, especially in the deeper layers of the atmosphere. One of the goals of this mission is to map Saturn's surface allowing to study weather patterns, storm systems and their evolution, additionally, the use of modern infrared imaging will enable a better understanding of temperature variations in the atmosphere, including deeper layers. Saturn's rings system is among the most iconic features of the Solar system, but much about the origin and evolution of these rings remains unknown. This mission aims to advance our understanding of the rings and gaps between them, in particular utilizing a dust collector to gather samples from different regions of the rings environment allowing for an in-depth on-board analysis of particle size distribution and their composition, essential to understanding the origin of the rings. Saturn's magnetosphere is one of the largest and most complex magnetic environments in the Solar System and it is strongly influenced by the solar wind. Cassini provided important data on the magnetic field structure and plasma environment, but many aspects of how the magnetosphere interacts with the solar wind remain open to exploration; this mission will further study Saturn's magnetosphere by deploying magnetometers, plasma spectrometer and solar wind monitors, allowing the mapping of the magnetosphere's structure and observing how it responds to the solar wind, giving particular attention to the magnetotail formation area. Titan is Saturn's largest moon, it has a dense atmosphere and a surface covered by hydrocarbon lakes and seas; the Cassini mission landed a probe on Titan's surface, however much about Titan's atmosphere, surface activity and internal structure remains unknown; this mission will conduct multiple flybys around Titan, during which it is going to observe its surface, atmosphere and weather patterns over time, in particular searching for evidence of geological activity, such as cryovolcanism. In conclusion, the SMADGROUP2 mission will fill in crucial gaps left by previous explorations, by using advanced instrumentation to push the boundaries of our understanding of Saturn's environment.

## 2. Mission Concept Alternatives and Selection

### *2.1. Alternative mission concepts*

In this section, different mission concepts are taken into consideration and are analyzed in order to come up with the best solution in terms of reliability, scientific importance and feasibility. These alternative mission concepts are related to the way on how the spacecraft will be led to end its operational life also referred to as the deorbiting. The deorbiting is an essential phase of a space mission in order to not cause more space debris in the orbit of a planet.

### *2.2. Drivers identification*

As the trajectory and, therefore, the required $\Delta v$ will differ significantly based on the deorbit maneuver, the mass of the propellant depends on the selection of the deorbiting approach. This influences the selection of the launcher and thus, the costs of the mission. Since the mass of propellant necessary to perform the deorbit of the spacecraft is going to be carried through the whole mission, it would increase the total weight at launch and the required propellant for the first stage as well, this in turn could increase the total mass and the costs. Furthermore, when choosing the approach the ethical implications must be taken into account, particularly with regard to the potential environmental impact of introducing spacecraft components, including a nuclear power generator, into celestial bodies.

### *2.3. Considered options*

The two considered options are to crash the spacecraft either into Saturn's moon Rhea or into Saturn itself, both performed in a controlled manner.

### *2.4. Assessment and concept selection*

Both of the presented approaches have been simulated and the exact amount of $\Delta v$ was calculated. Based on the required $\Delta v$, the total mass of the propellant and of the tanks required for storing the propellant are determined Appendix A. This results in a total mass of the mission that needs to be brought into LEO of 34775.37 kg and 61918.32 kg for deorbiting into Rhea or Saturn respectively. This would lead, as further explained later, for both approaches to the Falcon Heavy as the launcher. So no significant advantage could be won by crashing into Rhea.

Crashing into Saturn is considered the more ethically favorable approach for several reasons. Saturn is a gas giant with no solid surface, meaning that the spacecraft, along with any radioactive material from the nuclear power generator, would be engulfed in its dense atmosphere. Over time, the spacecraft would disintegrate and descend deeper into Saturn's interior, where the high pressures and temperatures would further break down its components. This minimizes the risk of contamination to solid surfaces where life might potentially exist. In contrast, impacting Rhea presents greater ethical concerns. Rhea is a solid moon, and while there is no direct evidence of life, planetary protection protocols prioritize minimizing the risk of contaminating celestial bodies that might hold potential for future exploration or astrobiological studies. Crashing a nuclear-powered spacecraft into Rhea, considering also that it is much smaller than Saturn, could leave long-term contamination, which might interfere with future missions seeking to explore this moon or other icy bodies in the Saturn system.

Given the considerations of planetary protection, long-term space sustainability, and ethical responsibility, deorbiting into Saturn is the preferable option. This approach ensures compliance with both space debris mitigation guidelines and planetary protection policies while minimizing any long-term environmental risks. Therefore, the approach of deorbiting the spacecraft into Saturn is selected.

## 3. Conceptual Mission Architecture

### *3.1. Overview of the mission concept*

The SMADGROUP2 mission is an innovative mission designed to investigate Saturn's environment starting from the knowledge obtained by the previous missions, in particular the Cassini-Huygens mission, improving the quality of data by leveraging state-of-the-art instrumentation and modern space technology. The mission will focus on the observation of Saturn's atmosphere, rings, gaps, and magnetosphere, with an emphasis on dust collection and on-board samples analysis. Additionally, the mission will perform multiple flybys around the largest Saturn's moon, Titan, to study its features. Key mission objectives are listed in section 1.1. The spacecraft will be an orbiter equipped with a range of scientific instruments, well described in section 5.2, and will be powered with an RTG and a battery. After a journey of around 5.5 years, considering both interplanetary trajectory and flybys around Titan needed to be captured into the Saturn orbit, the spacecraft will enter a polar orbit with radius around 551000 km around the planet, enabling global coverage of Saturn's atmosphere, rings and magnetosphere. The polar orbit will allow continuous observations of both polar regions and key atmospheric features, while flybys around Titan are exploited to observe its surface and atmosphere.

### *3.2. Preliminary design options*

The preliminary design of this interplanetary mission involves careful evaluation of various spacecraft components to meet the specific requirements and constraints of the mission. This includes the orbiter as it must be designed to operate efficiently in deep space and in harsh environments around Jupiter and Saturn. Key considerations include radiation shielding, thermal control, power generation, and communication capabilities for long-distance signal transmission. The choice of propulsion system is crucial for providing the necessary $\Delta v$ for interplanetary maneuvers and orbital insertion. Out of many viable options, a bipropellant chemical propulsion system is the most feasible solution, considering mission constraints and requirements. Depending on the resulting mass of the spacecraft, a appropriate launcher needs to be selected influencing e.g the costs and risks of the mission. Furthermore, the design of the payload includes scientific instruments developed for studying Saturn's magnetic field, atmosphere, rings and Titan moon. A detailed analysis and design selection with respect to all requirements is done throughout this work and for details one refer to the respective section.

### *3.3. Mission phases*

The space mission to Saturn is divided into eight phases, as shown in Table 1. It begins with the launch to a 200 km LEO in phase 1. Phase 2 covers the cruise in LEO, where system checks, trajectory adjustments, and preparations for interplanetary transfer are conducted. In phase 3, a $\Delta v$ maneuver enables escape from Earth's orbit, placing the spacecraft en route

| | | | |
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| Phase 1 | Launch to LEO | Phase 5 | Transfer to Saturn |
| Phase 2 | Cruise in LEO | Phase 6 | Entry at Saturn |
| Phase 3 | Transfer to Jupiter | Phase 7 | Operation in working orbit |
| Phase 4 | Flyby at Jupiter | Phase 8 | Deorbiting |

Table 1: Overview of the mission phases

to Jupiter. Phase 4 involves a gravity-assist flyby at Jupiter, providing the velocity boost required for transfer to Saturn. Phase 5 follows, with the spacecraft continuing toward Saturn, performing any necessary course corrections. As the spacecraft nears Saturn, it enters a transfer orbit during phase 6, focusing also on secondary objectives related to Titan. Phase 7 involves scientific operations in Saturn's working orbit, collecting and transmitting data. Finally, in phase 8, the spacecraft is deorbited to avoid space debris, while collecting samples from the ring gaps.

### *3.4. High-Level Mission Timeline*

Following NASA's guideline about mission planning for original concept missions, we can break down the high-level mission timeline in 7 different phases, all of them composing the life-cycle of a mission. The Pre-Phase A provides the idea of plan for the mission proposed near the academic community, to be evaluated for method and feasibility. Phase A gives a preliminary design and project plan of the mission to establish some key concepts like what to build, when to launch, the course the spacecraft is to take, what is to be done during cruise, when the spacecraft will reach the target, and what operations will be carried out. Cost, spacecraft instruments, where system tests will be performed, who performs mission operations, what ground data system capabilities are required. Phase B is the phase where the definitions are converted into a baseline technical solution. Facilities are selected based on existing resources and past performance, and teams are composed to build and operate the instruments and to evaluate the data returned. Phase C/D is the design and development phase, where schedules are negotiated, and the space flight system is designed and developed. This phase begins with the building and integration of subsystems and experiments in a single spacecraft. In a process called ATLO (Assembly, Test, and Launch Operations), the spacecraft is assembled, integrated, tested, launched, deployed, and verified. Before the launch the spacecraft is also tested in a simulated interplanetary space environment. The ground systems that support the mission are also developed and tested alongside the spacecraft in this phase. Phase E is the operations phase, where the mission "starts". It includes flying the spacecraft and collecting the data for which the mission was designed. For the final phase of the mission, Phase F includes retrieving the data collected and analyzing it. The systems used in the mission are deorbited or disposed of, and all the systems are shut down. This is also called the "closeout phase".

## 4. Preliminary Mission Requirements

In this section are reported all the mission requirements and constraints.

*4.1. High level science requirements*

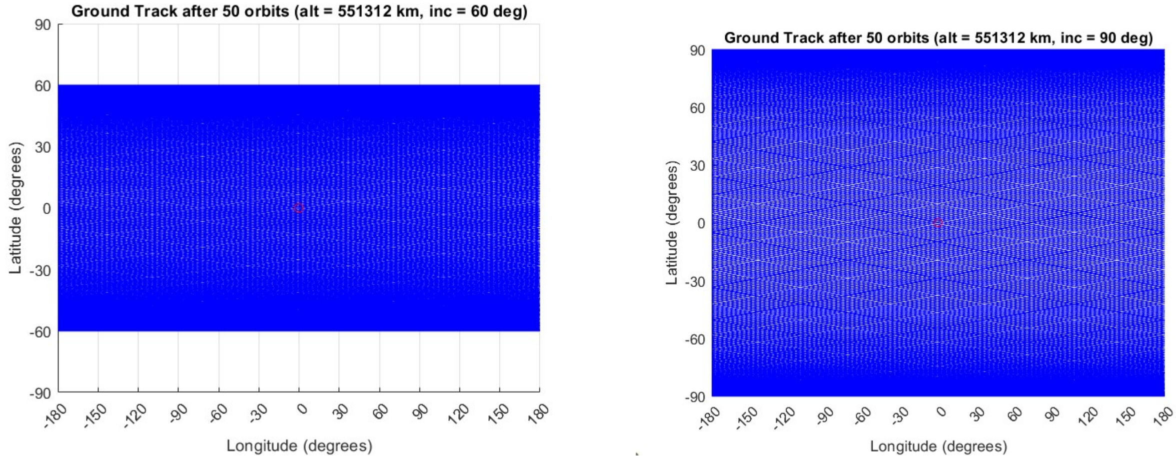
- The spacecraft must collect visible spectrum images of at least 95% of Saturn's surface.
- The mission shall conduct at least 7 flybys around Titan, capturing visible and infrared images of the surface with a resolution of less than 1 km per pixel.
- The spacecraft shall capture infrared maps of Saturn's atmosphere covering $\geq 85\%$.
- The spacecraft shall collect and correctly analyze dust samples from at least the E ring and the Cassini division.
- The magnetometer shall map the structure and dynamics of Saturn's magnetosphere over at least half of a solar activity cycle to better understand its influence.
- The spacecraft shall gather data on Saturn's plasma environment, at least in the area where the magnetotail starts its shaping.

*4.2. High level technical requirements*

- Spacecraft's dry mass shall be $\leq$ 900 kg to enable launch with Falcon Heavy.
- The spacecraft attitude control system shall guarantee a pointing accuracy of $\leq 0.05°$ for high resolution imaging.
- The spacecraft shall be equipped with RTGs to provide continuous power.
- The visible and infrared narrow cameras shall provide imaging resolution of 2 km per pixel at the operational distance from Saturn.
- The dust collector shall be capable of capturing particles from rings and gaps, avoiding the mixing of samples from different regions.
- Sensitivity of magnetometer measurements shall be $\leq$ 0.1 nT.
- The infrared spectrometer shall cover a wavelength range of 3000 - 14000 nm.
- The spacecraft shall have on-board data storage capacity of at least 64 GB, with sufficient compression capacity to store all scientific data between communication windows.
- The spacecraft shall have a delta-V capability sufficient to perform orbital insertion around Saturn and to perform the deorbiting maneuvers.

*4.3. High level operational requirements*

- The operational mission duration shall be at least 7 years.
- The working orbit shall be a polar one to guarantee poles coverage (see Fig. 1).
- Altitude of working orbit shall be $\geq$ 490,000 km to avoid impacts with ring's particles.
- The operational orbit shall have an altitude below 600,000 km to guarantee high-quality images of Saturn.
- The operational orbit shall be circular to take pictures always at the same distance.
- The operational orbit shall not pass through Rhea's sphere of influence to avoid orbital deviations (Rhea's orbit average radius is 527,000 km).
- Data transmission to Earth shall occur during designated communication windows.




(a) Inclination of 60 deg

(b) Inclination of 90 deg.

Figure 1: Plot of the ground tracks in the desired working orbit around Saturn having different inclinations based on a self-developed Matlab simulation.

### *4.4. High level environmental requirements*

- The flyby around Jupiter must have a perigee above 600,000 km to not pass through the high radiation area around the planet.
- The flyby around Titan must pass above Titan's atmosphere.
- The approach to Saturn and the orbits before the achievement of the operational orbit must pass through ring gaps or thin rings.
- All the cameras shall be protected with a mobile cap to reduce their degradation due to the space environment.
- The magnetometers shall be mounted on a long boom to isolate their measurements from the spacecraft noise.
- The infrared cameras' sensors shall be cooled to guarantee noise reduction.
- The spacecraft shall be able to withstand external temperatures from -200°C to +50°C, maintaining the internal temperature within the range 0°C to +40°C.
- The spacecraft shall be shielded from high-energy particle radiation, with a tolerance of $\geq$ 100 krad total ionizing dose.

### *4.5. Regulatory and policy*

- The spacecraft shall not contaminate any of Saturn's moons that may harbor life, such as Enceladus and Titan.
- The mission shall secure necessary frequency allocations for communications through the International Telecommunication Union (ITU), ensuring minimal interference with other deep space missions.

## 5. Preliminary Spacecraft Design Concepts

### *5.1. Spacecraft bus concepts*

The system uses a custom-built configuration to optimize space and instrumentation. The satellite bus must support payloads by providing structural rigidity, power, and data handling. It will supply power to subsystems (via RTGs and batteries), manage attitude determination and control for precise pointing, and ensure accurate antenna alignment for communication with ground control. Additionally, it will transmit telemetry and payload data to the ground station.

The Attitude and Orbit Control System (AOCS) ensures the spacecraft's orientation and position relative to inertial axes, crucial for both major maneuvers and precise instrument pointing. A three-axis stabilization approach is chosen, where main engines handle large $\Delta v$ changes, while attitude control is managed by momentum exchange devices (reaction wheels) for long-duration control without propellant use. The spacecraft will use 4 reaction wheels — 3 for control along each axis and 1 for redundancy [1]. Thrusters, assisting in higher-torque adjustments, are excluded at this phase but may be evaluated later. Star trackers will provide precise attitude data by tracking stars. Integrating 3 trackers ensures redundancy and high accuracy. PID control will be used for the reaction wheels, with Model Predictive Control (MPC) considered for more advanced control if computational capacity allows.

Structure is a key aspect of a spacecraft, serving as its backbone and ensuring good endurance of the harsh conditions faced in a space environment, as well as during launch. In terms of geometry, it is usually a tall structure shaped in a quasi-cylindrical form, made out of lightweight alloys (such as aluminium-lithium) and/or advanced composites (carbon fiber) in order to resist intense mechanical loads and vibrations while maintaining a low mass. The surface needs to be covered with radiators for heat control, micro meteoroid shielding, adequate thermal protection (via insulation and coating techniques), and radiation shields, to prevent damage to equipment. Secondary (internal) structures are designed to mount components such as instruments, electronics, and other systems. It is essential to place each component in the most pertinent way considering how they will interact with each other and behave as a whole in the spacecraft.

Power generation is critical for operating the spacecraft's systems, including its computational unit, antennas, and scientific instruments. Solar cells, typically used in space missions, are unsuitable for deep space due to Saturn's distance (9.59 A.U.), where solar irradiance is only 1.09% of Earth's, requiring impractically large solar arrays. Fuel cells are also impractical due to their weight, fuel storage challenges, and the risk of instability. The best option is Radioisotope Thermoelectric Generators (RTGs), which are reliable for low-power missions and space-proven. Batteries will supplement RTGs for peak power demands, staying near full charge from the RTG's continuous output.

The communication system is vital to the mission, with a High Gain Antenna (HGA) as the primary tool for transmitting high-data-rate science data to Earth and receiving commands. The selected HGA, a 4-meter parabolic dish weighing 35-50 kg, operates in the

X-band (8.4 GHz downlink, 7.2 GHz uplink) with a power output of around 20 watts allowing focused, narrow-beam transmissions of data to Earth. A Low Gain Antenna (LGA), an omnidirectional monopole weighing 1-2 kg, will handle low-data-rate communication, particularly during HGA unavailability and early mission phases. The system will additionally use a (typically custom made) Traveling Wave Tube Amplifiers (TWTA) for high-frequency amplification. The data transmission to Earth relies on NASA's Deep Space Network, with communication windows scheduled based on spacecraft position and DSN availability. Transmission prioritizes scientific data and employs both lossless (for sensitive measurements) and lossy (for larger datasets like images) compression to manage large data volumes effectively.

### *5.2. Payload concepts*

The mission aims to use instrumentation with proven flight heritage to guarantee optimal results while keeping the risk low, some upgrades to the instruments are planned to improve their functionality without changing their core, then avoiding the necessity of expensive tests (both in terms of money and time). To achieve the objectives, the following instruments have been selected.

#### *5.2.1. Wide Angle Camera*

This camera is necessary to achieve the main objective and one of the secondary objectives of the mission, allowing the mapping of Saturn's surface in the visible and IR spectrum. This camera is going to work in both the visible and IR spectrum using a dual channel system with a combination of different sensors, filters, and lenses to separate and capture light across these wavelength ranges and allow for simultaneous imaging. The collected images are used to find interesting spots, such as storms and thermal anomalies, to observe them with the other two narrow angle cameras. The selected sensors are a CMOS sensor for the visible spectrum and a HgCdTe (Mercury Cadmium Telluride) sensor for the IR observations, knowing that it requires cooling. A beam splitter allows to divide the incoming light into two separate channels, one for visible and one for IR, in which different filters allow only the desired wavelengths to pass, directing them to their specific sensor, leading to simultaneously imaging. The camera is equipped with different lenses optimized in the IR and visible spectrum, placed in the two different paths to guarantee better images with respect to the usage of dual-purpose lenses. This camera is using a FOV = 3.5 deg, is working in a wavelength range between 400 - 700 nm (visible path) and 3000 - 14000 nm (from mid to long wave infrared, for the IR path) and has a focal length of 200 mm. HgCdTe sensor is used because it works better with mid and long wave infrared observations that we are aiming for, and about the visible spectrum, a CMOS sensor (2048x2048 resolution) has been selected as better solution with respect to the previous missions' CCD sensor (1024x1024 resolution) allowing, in addition to the better resolution, a lower power consumption, higher frame rates (really useful during flybys), but knowing that a cooling system is necessary to reduce the noise. This kind of technology has already been used in missions as the Mars Reconnaissance Orbiter.

*5.2.2. Narrow Angle Cameras*

Two narrow angle (FOV) cameras, one operating in the visible spectrum and the other in the IR spectrum are planned, their objective is to take detailed visible and thermal images of interesting points of the surface,atmosphere and rings. These cameras are an improvement over Cassini's cameras since they have higher resolution, sensitivity and precision. The visible camera uses a multi-element lens system to guarantee high precision and minimize chromatic aberration, upgraded coatings with respect to the Cassini mission are planned to reduce glare and scattering. The selected sensor is a CMOS sensor, to guarantee less power consumption and higher frame rates with respect to the CCD sensor used in the Cassini mission. The camera is equipped with filter wheels to switch between different narrow-band filters allowing it to isolate specific visible wavelengths and allowing better imaging. The IR camera is intended to be used in the mid-wave / long-wave infrared, so a HgCdTe sensor is the best option, a more efficient cryogenic cooling system with respect to the one used in the Cassini mission is planned to reduce thermal noise. The optical system is going to be optimized in the IR spectrum to guarantee a higher resolution with respect to the Cassini mission and the filtering system is expected to have more filters to isolate different thermal wavelengths. These cameras are using a FOV = 0.35 deg, are working in a wavelength range between 400 - 700 nm (visible camera) and 3000 - 14000 nm (mid-long infrared camera), and have a focal length of 2000 mm.

*5.2.3. Medium IR spectrometer*

This instrument allows to measure thermal emissions and to analyze the composition of Saturn's atmosphere and rings by detecting IR radiation, is useful in understanding temperature gradients and determining the presence of specific compounds. The infrared radiation from the target is captured by the optical system and dispersed into its component wavelengths to measure the intensity of light of each wavelength in order to create a spectrum containing informations about which molecules are present in the target (e.g atmosphere or rings) based on the specific absorption features of those molecules in the infrared range. Recently launched missions, such as the James Webb Space Telescope, Herschel Space Observatory and Gaia demonstrated that higher resolution, sensitivity and on-board noise reduction can be implemented to improve the technology used in previous missions around Saturn.

*5.2.4. Magnetometers*

They allow the spacecraft to measure strength and direction of the Saturn's magnetic field, both a fluxgate and an helium vector magnetometers are going to be implemented on the satellite in order to obtain accurate measurements across a wide range of field strengths, covering both the stronger fields near Saturn and the weaker, more subtle variations in the outer magnetosphere. This is possible since fluxgate magnetometers are excellent for measuring medium/strong magnetic fields (from nanotesla to several gauss) and vector helium magnetometers are more sensitive to weaker magnetic fields being able to detect very small variations in the range of sub-nanotesla. Another advantage of using both of them is that fluxgate magnetometers measure the vector components of the magnetic field, and vector helium magnetometers can measure the magnitude of the magnetic field with high precision, so combining their measurements a very accurate value of the magnetic field is obtainable. All the achieved data are compared with the solar wind ones to assess their interaction.

*5.2.5. Plasma spectrometer*

It is based on the cooperation of 3 subsystems, an ion beam spectrometer (that measures the velocity and energy of ions), an electron spectrometer (that measures the velocity and energy of electrons), and an ion mass spectrometer (that identifies the composition of ions by measuring their mass-to-charge ratio), making the instrument is capable of measuring density, temperature, and composition of plasma in Saturn's magnetosphere. It is possible by collecting charged particles form the surrounding plasma environment through electrostatic fields, and then, by using another electric field (electrostatic deflector) it is possible to deflect them allowing the separation between particles based on their energy levels, the determination of their velocity and energy distribution and perform the analysis through the ion mass spectrometer to understand the composition. Since the operational orbit is at around 550000 km altitude, this instrument is useful to understand better the influence of Rhea moon on the magnetosphere and to map the whole outer layer of the magnetosphere, including the beginning of the magnetotail.

*5.2.6. Solar Wind Monitor*

Its task is to monitor the properties of the solar wind and combining its results with the previous instruments data it is possible to evaluate the interaction between magnetosphere and solar wind, achieving more reliable data with respect to previous missions around Saturn that were not equipped with this tool. its operation is similar to the one adopted by the plasma spectrometer allowing the assessment of density, velocity and energy of the particles in the solar wind, it is capable to filter the collected particles to analyze only the ones coming from the solar wind since they generally come from the same direction and have higher velocity and energy with respect to the magnetosphere particles.

*5.2.7. Dust Analyzer*

Its goal is to detect, measure, and analyze dust particles, in particular, when a dust particle collides with the sensor, it can detect mass, velocity, charge, and in some cases the composition of the particle by measuring the ions and electrons that are released during the impact. For this purpose a detector similar to the one used in the Cassini mission is enough, the upgrade of the measurements is due the comparison between the results from the dust analyzer and the results from the on board spectrometer.

*5.2.8. Dust Collector and On-board mass spectrometer*

The collector is designed to capture rings' dust particles for on-board analysis, it is important to not damage the particles to achieve detailed results, and to this purpose, aerogel material has been selected in order to gently slow down particles during their penetration into the aerogel. A similar approach has been implemented in the NASA Stardust mission. To avoid contamination of particles from different rings, different aerogel surfaces are going to be used. The on-board spectrometer purpose is to analyze the composition of the captured particles, identifying their chemical composition by measuring the mass-to-charge ratio of the ionized particles released by the samples.

## 6. Preliminary Mission Trajectory and Navigation

### *6.1. Initial orbital mechanics considerations*

The mission trajectory involves several maneuvers requiring a solid understanding of orbital mechanics. A basic understanding is assumed, with advanced maneuvers briefly reviewed. To achieve a heliocentric trajectory, the spacecraft must first escape Earth's gravitational influence from LEO by applying a high $\Delta v$. If the $\Delta v$ reaches escape velocity (about 12 km/s), the spacecraft transitions to a hyperbolic orbit and enters a heliocentric trajectory. Multiple gravity-assist flybys alter the spacecraft's velocity and trajectory by using a planet's gravity, enabling speed adjustments without additional fuel. The maneuver's effectiveness depends on two key parameters: the closest approach distance and the flyby entry angle $\theta$, both of which can be adjusted with small maneuvers. Upon reaching Saturn, the spacecraft must transition from a hyperbolic to an eccentric orbit by applying a negative $\Delta v$ or performing flybys around Saturn's moons. These maneuvers are more efficient at perigee, near Saturn's surface. To enter a polar orbit, the inclination must be changed, which requires a significant $\Delta v$. A more efficient approach is to simultaneously alter the orbit's shape, to save propellant. These maneuvers are best performed at the apogee for efficiency and are most feasible for small inclination changes.

Furthermore, a few assumptions are made in order to design the optimal trajectories. First, the patched-conic method is used. Second, when solving the Lambert's problem between Earth and Jupiter in order to calculate the required velocity at the LEO it is assumed that the spacecraft is in the center of Earth at departure and at the center of Jupiter at arrival. Third, the position of the spacecraft after every flyby is assumed to be in the center of the body around which the flyby was performed. Last, all flyby maneuvers are assumed to be performed instantaneous.

### *6.2. Propulsion system options*

An interplanetary mission from LEO to Saturn's periodic orbit, with a flyby around Jupiter, requires a highly efficient propulsion system capable of generating the necessary $\Delta v$ while minimizing propellant usage and delivering the payload within the mission timeline. The most prominent propulsion systems include chemical, electric, and nuclear propulsion. Although electric propulsion is considered efficient, implementing a successful electric propulsion system for this mission presents challenges. This makes chemical propulsion the most suitable choice. Within chemical propulsion, the type of fuel used for combustion plays a crucial role. For short bursts of high-energy thrust, solid rocket boosters are preferred. These boosters, such as Space Shuttle SRB and SLS SRB, are hypergolic and easy to store, which makes them a good option for providing the initial boost to escape Earth's atmosphere. However, their relatively low specific impulse ($I_{sp}$) and uncontrollable combustion make them unsuitable for sustained or precise maneuvers. For a more controlled flight with better maneuverability and higher $I_{sp}$, bipropellant liquid rocket engines are favored. These engines fall into three categories based on the type of propellant: hypergolic, non-cryogenic, and cryogenic. Hypergolic engines, such as the Aerojet Rocketdyne AJ10 and Viking engines, use monomethylhydrazine ($MMH$) as fuel and nitrogen tetroxide ($N_2O_4$) as the oxidizer. They are known for their ease of storage and reliable ignition but have lower efficiency due to a lower $I_{sp}$. Non-cryogenic engines, such as the Merlin 1D and RS-27A, use RP-1 (a refined kerosene) as fuel and liquid oxygen ($LOX$) as oxidizer. These engines provide a good balance between efficiency and storability but require separate ignition systems and careful management of $LOX$ storage. Cryogenic engines, such as the RS-25 and Vinci engines, use $LH_2$ and $LOX$, providing the highest $I_{sp}$ around 400-450s. However, the extremely low storage temperatures required for cryogenic propellants, as low as -253°C which make them difficult to use in long-duration missions, and the large fuel tanks needed due to $LH_2$'s low density present additional challenges.

Considering all these factors, a trade-off analysis was performed to select the best engines for each stage of the mission. The first stage of the mission, which involves traveling from LEO to Jupiter's atmosphere, requires a propulsion system with a balance of high $I_{sp}$ and manageable mass. After evaluating several options, the best choice for this phase is the Merlin 1D engine, which has an $I_{sp}$ of 311 seconds and a mass of 470 kg. This engine, used in the second stage of Falcon 9, is powered by RP-1 and $LOX$, making it a non-cryogenic system. The RD-140 engine, which has an identical $I_{sp}$ of 311 seconds but weighs 5,480 kg, was also considered. However, both would require an impractical amount of propellant ($\approx$ 62,000 kg) for the first stage, exceeding the Falcon Heavy's payload capacity. A more efficient alternative is the RL10CC1 engine, with an $I_{sp}$ of 453.8 seconds and a mass of 188 kg. The RL10CC1 uses cryogenic propellants, making it capable of achieving the mission's requirements with a significantly lower propellant mass. The main challenge with this engine, however, is the need for cryogenic storage over a long-duration mission. For the second stage, which covers the journey from Jupiter to Saturn's polar orbit, the RD-140 and RL10CC1 engines were again considered but were found to present storage or mass challenges. After further research, the R-4D engine was selected as the best option for this phase of the mission. The lightweight R-4D engine, uses hypergolic propellants offering easy storage and making it an ideal candidate for delivering the payload easily to its final destination. Details on the two selected engines can be seen in Table 2.

### *6.3. Preliminary navigation strategies*

For the preliminary navigation strategy, we would use a combination of both Star Trackers and Delta Differential One-Way Ranging (Delta-DOR). The Star Trackers are already being used on our spacecraft for attitude determination, but the data that they collect can also indirectly aid in position tracking, especially when we combine it with some other navigation method. The Delta-DOR method consists in having two widely separated ground stations, that are thousands of kilometers apart on Earth, that simultaneously receive signals from the spacecraft. By having them far apart from one another, the slight difference of the time that the signal is received will be precisely measured (to within nanoseconds). If we also measure a quasar (an extra galactic, stable radio source fixed compared to Earth), and use it

| Feature | RL10C-1-1 | R-4D |
|---|---|---|
| Status | Active (2021) | Active (used since Apollo missions) |
| Engine Type | Cryogenic ($LH_2/LOX$) | Hypergolic ($MMH/N_2O_4$) |
| Thrust | 105.98 kN (23,825 lbf) | 490 N (110 lbf) |
| Specific Impulse ($I_{sp}$) | 453.8 s (4.450 km/s) | 312 s |
| Mass | 188 kg (415 lb) | 3.63 kg (8 lb) |
| Propellant Type | Liquid Hydrogen ($LH_2$) / Liquid Oxygen ($LOX$) | Monomethylhydrazine (MMH) / Nitrogen Tetroxide ($N_2O_4$) |
| Engine Length | 2.46 m (8 ft 0.7 in) | 0.84 m (2 ft 9 in) |
| Engine Diameter | 1.57 m (5 ft 2 in) | 0.41 m (1 ft 4 in) |
| Mixture Ratio (O/F) | 5.5:1 | 1.65:1 |
| Chamber Pressure | 4.4 MPa (44 bar) | 14 atm (1.4 MPa) |
| Usage | Upper stage engine (Atlas V, Vulcan Centaur) | Orbital maneuvering, attitude control (Apollo, Space Shuttle, Cassini) |
| Applications | Long-duration burns, interplanetary, deep space missions | Orbital insertion, trajectory corrections, attitude control |

Table 2: RL10C-1-1 and R-4D Engine specifications [2–4].

as a fixed point of reference, we can also measure the difference in arrival times of this signal at the two ground stations. Combining both these measurements and time delay (of the spacecraft signal, and the quasar signal), we can accurately calculate the spacecraft angular position, relative to the quasar. For 3D position determination of the spacecraft in space, we just need to combine the angular position with Doppler tracking and two-way ranging. Doppler tracking can accurately give us the spacecraft radial velocity, while two-way ranging can determine the spacecraft's distance from Earth. This way, and by combining these methods, we get an extremely precise 3D position of the spacecraft in space. The only big downside of the Delta-DOR method is that it would require highly complex infrastructure and equipment, since it would require multiple ground stations to perform, and extremely accurate measurement equipment (like atomic clocks), and this could be challenging to acquire, while being very costly. It's also important to be aware of some error sources, like the radio waves traveling through the troposphere, ionosphere and solar plasma, and clock instabilities at the ground station [5].

### *6.4. Trajectories*

For finding the optimal trajectory of all parts of the mission, an extensive Python simulation environment is developed. Details on the structure of this project and on the different included simulations can be found in the Appendix A. In this section only the finally selected trajectories are presented.

#### 6.4.1. Preliminary interplanetary trajectory

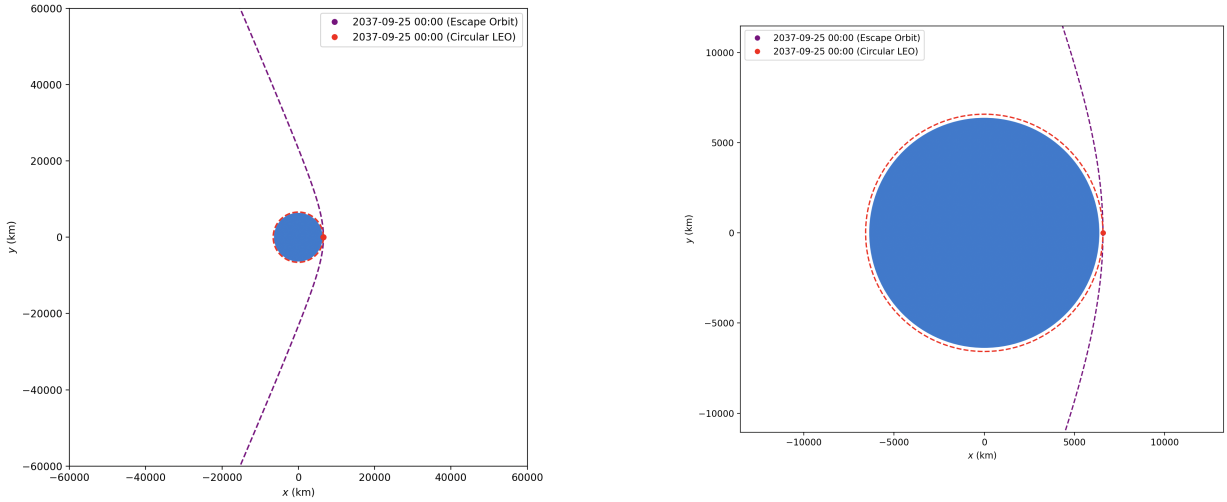
In general, the procedure of the interplanetary trajectory is as follows: First, the spacecraft launches to a Low Earth Orbit of 200 km using an external launcher. Then, a certain $\Delta v$ is applied in order to escape the gravitational field of the Earth and to get on a heliocentric orbit around the Sun. Afterwards, a gravity-assist flyby maneuver will be performed at Jupiter in order to gain velocity with respect to the Sun. Lastly, Saturn will be met on the new heliocentric orbit around the sun on which the spacecraft will travel after the flyby. This procedure was chosen based on the maneuvers performed by the four previous missions that traveled to Saturn. Three of them, namely Voyager-1, Voyager-2 and Pioneer 11, also used only one flyby maneuver around Jupiter to reach Saturn. The fourth one, the Cassini mission, performed in total four flybys: two around Venus, one around Earth and one around Jupiter. Due to the high complexity of designing an interplanetary trajectory of more than one flyby, this mission is oriented on the procedure of the other three missions and therefore, will perform only one flyby around Jupiter.

In order to find the optimal trajectory, a simulation was built that simulated the described steps and iteratively tried out hundreds of thousands different trajectories and further optimized promising candidates. The input parameters of the simulation that were varied to influence the trajectory are the date of departure, date of flyby, minimal distance of the flyby maneuver at Jupiter and the entry angle of the flyby. The trajectory was optimized with respect to the following aspects. First regarding the required $\Delta v$ for escaping Earth's gravity, in order to minimize the required mass of propellant and therefore the mass that has to be brought into Low Earth Orbit. Second, regarding the duration of the entire trajectory, in order to ensure that the scientific results achieved by the mission are available as soon as possible. And third, regarding the timing of all the mentioned maneuvers, in order to cross the orbit of Saturn with the spacecraft at the exact time when Saturn is located in this position of crossing.

The optimal trajectory was found for departing the Low Earth Orbit on the 25th of September 2037 and performing a flyby at Jupiter on the 29th of July 2039 with an entry angle of 5.8814 deg and a minimal flyby distance of 1159500 km. Without considering the

| Parameter | Value |
|---|---|
| Date of departure at LEO | 2037-09-25 |
| Date of flyby at Jupiter | 2039-07-29 |
| Minimal distance of flyby at Jupiter | 1159500 km |
| Entry angle of flyby at Jupiter | 5.881 deg |
| Required $\Delta v_1$ at LEO | 6.848 km/s |
| Date of arrival at Saturn | 2041-09-14 |
| Arrival velocity at Saturn w.r.t. Sun | $\left[-5.271 - 12.241 - 4.667\right]$ km/s |
| Arrival speed at Saturn w.r.t. Sun | 14.121 km/s |
| Arrival speed at Saturn w.r.t. Saturn | $v_{\text{inf}} = 9.931$ km/s |

Table 3: Summary of the parameters of the interplanetary trajectory to Saturn.




(a) Zoomed out.

(b) Zoomed in.

Figure 2: 2D plot of the LEO orbit and the trajectory on which the spacecraft escapes Earth's sphere of influence.

gravitational influence of Saturn on the spacecraft, the spacecraft would meet the center of Saturn with a closest distance of 7200.915 km. A summary of the parameters of the resulting interplanetary trajectory is given in Table 3. A 2D plot of the interplanetary trajectory is shown in Fig. 3 and of the LEO combined with the escape orbit can be seen in Fig. 2.

### *6.4.2. Working trajectory during data acquisition*

As presented earlier, a circular polar working orbit is chosen around Saturn. Based on the requirements, a radius of about $r_{desired} = 550000$ km is chosen as the desired working orbit radius. In order to find a feasible trajectory from the entry at Saturn to the final working trajectory, many different ideas and approaches has been tested. The most promising one is presented in the following. All the calculations, detailed results of each step and plots of the trajectories can be reviewed in detail in a Jupyter Notebook which presents the Python simulation with added explanations, formulas and interpretations step by step. For details, refer to Appendix A.

As described in previous sections, the spacecraft arrives with respect to Saturn on a hyperbolic orbit and needs to be slowed down in order to be captured by Saturn in an elliptical orbit. This can be done by either applying a $\Delta v$ in negative flight direction or by performing flyby maneuvers around the moons of Saturn. In general, the goal is to keep the $\Delta v$ which needs to be applied by the spacecraft's engines and so the mass of the propellant as low as possible. Furthermore, one of the secondary objectives of the mission is to acquire images of Saturn's moon Titan. Therefore, performing multiple flybys around Titan is chosen as an ideal method to analyze Titan and to slow down the spacecraft. As Titan is Saturn's moon with the largest gravity, each of the gravity-assist flyby maneuvers has an great impact on the spacecraft's velocity. Furthermore, the inclination of the spacecraft needs to be changed to 90 deg in order to achieve a polar orbit. As explained earlier, changing

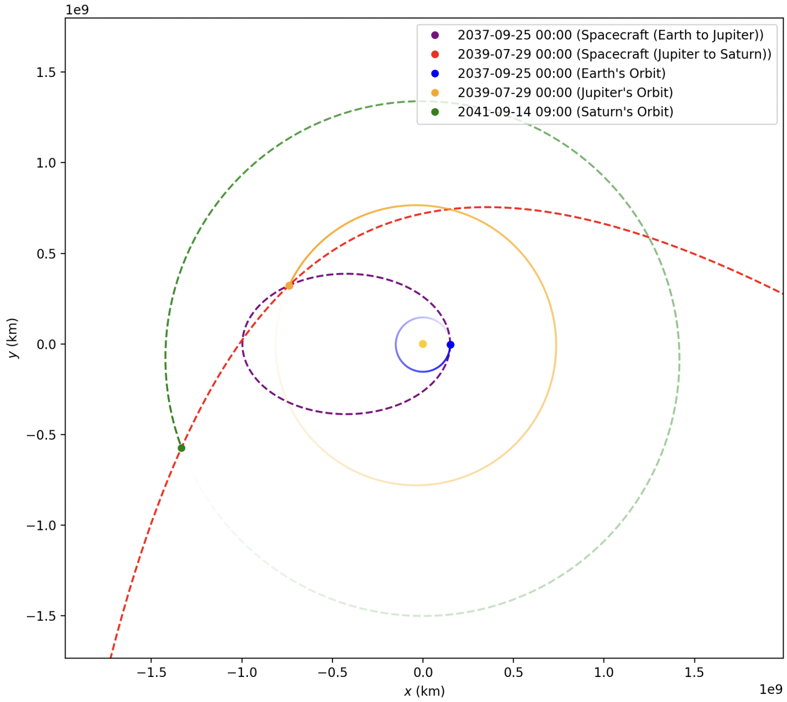



Figure 3: 2D plot of the interplanetary trajectory of the spacecraft and of the involved planets.

the orbit's inclination should be done at low speed, so at the apogee of an elliptical orbit and only about small angle changes $\Delta i$. However, an inclination change can also be achieved by doing a gravity-assist flyby maneuver if the maneuver is performed with the correct choice of the entry angle. With these considerations a sequence of maneuvers is designed in order to bring the spacecraft into the desired working orbit which are described in the following.

After arriving at Saturn as computed in the interplanetary trajectory optimization, the spacecraft will perform a small maneuver to increase the impact parameter $b$ (semi-minor axis) of the hyperbolic entry orbit around Saturn. The value of $b$ is set in a way, that the perigee lies within the F and G ring (140000 km to 166000 km) of Saturn. After running simulations $b = 370000$ km is chosen which leads to a perigee at 149076.620 km. We are assuming that the rings are in the ideal position at the time of entering the sphere of influence of Saturn so that the spacecraft passes the rings exactly at the perigee. This can be achieved by performing some small maneuvers in advance to wait for the right timing.

The first maneuver is then performed at the perigee. A $\Delta v_2$ is applied in the negative flight direction in order to decrease the velocity of the spacecraft. The magnitude of $\Delta v_2$ is chosen so that the the resulting orbit has a very high eccentricity of about $e = 0.98$:

$$e = \frac{r_a - r_p}{r_a + r_p} \rightarrow r_a = \frac{-r_p - r_p \cdot e}{e - 1} = 14.758 \cdot 10^6 \text{ km} \tag{1}$$

Based on the incoming speed and the required speed of the next desired orbit, the amount

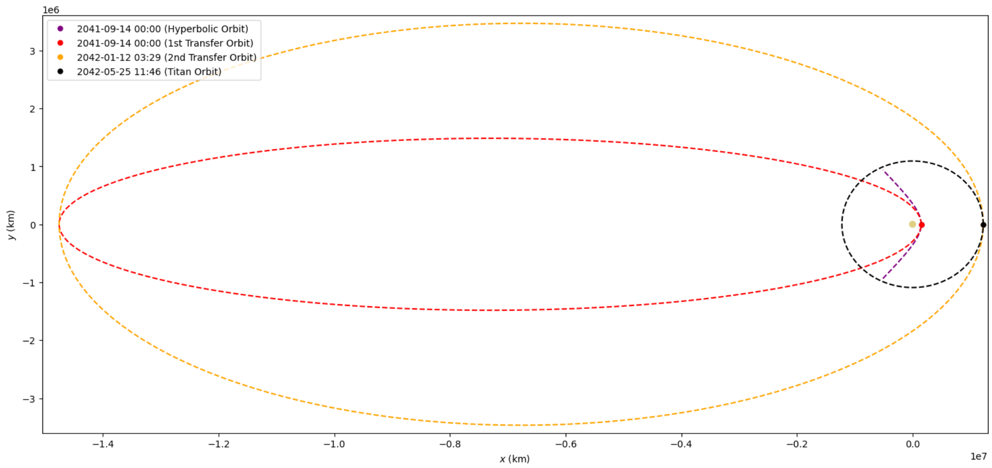



Figure 4: 2D plot of the first entry trajectories at Saturn.

| Flyby | 1 | 2 | 3 | 4 | 5 | 6 | 7 |
|---|---|---|---|---|---|---|---|
| Inclination [deg] | 65.12 | 67.48 | 70.17 | 72.71 | 77.43 | 82.53 | 88.17 |
| Apogee [$10^3$ km] | 5813.65 | 3798.86 | 2615.53 | 1953.45 | 1242.18 | 1221.87 | 1221.87 |
| Perigee [$10^3$ km] | 1221.87 | 1221.87 | 1221.87 | 1221.87 | 1221.87 | 822.39 | 551.31 |
| Orbit period [days] | 77.91 | 46.96 | 31.38 | 23.62 | 16.14 | 12.20 | 9.86 |
| Speed [km/s] | 7.16 | 6.85 | 6.51 | 6.18 | 5.59 | 5.00 | 4.39 |
| Flyby radius [km] | 4300 | 5900 | 5300 | 5800 | 3200 | 3200 | 3200 |

Table 4: Summary of the results and parameters of the different flyby maneuvers during entry. The parameters except of the flyby radius are related to the orbit after the performed flyby.

of $\Delta v_2$ can be computed:

$$\Delta v_2 = |24.648 - 22.445| \text{ km/s} \approx 2.203 \text{ km/s} \tag{2}$$

After that maneuver the spacecraft is captured on a high eccentric orbit around Saturn. The idea of the next maneuvers is the following. First, the inclination of the orbit is changed significantly by applying a certain $\Delta v_3$ at the apogee of the high eccentric orbit. This maneuver is combined with changing the radius of the perigee of the orbit to the distance that the moon Titan has to Saturn. As mentioned above this maneuver is optimal, as an inclination and orbit shape change is performed simultaneously at the same time at a very low speed. The goal of this maneuver is to get into an orbit where we can perform flyby maneuvers around Titan at the perigee of our orbit, to first decrease the velocity and thus lower the apogee, and second, to further change the inclination of the orbit towards 90 deg. After trying out different scenarios in the developed simulation, changing the inclination to a value of 62 deg leads to an optimal total $\Delta v$ that needs to be applied to get into the working orbit. As it is assumed that we arrive at Saturn approximately in the ecliptic plane, and Saturn is tilted to that plane with about 26.73 deg, the initial orbit with respect to Saturn already has an inclination of 26.73 deg. Thus, the inclination needs to be changed by $\Delta i = 62\,\text{deg} - 26.73\,\text{deg} = 35.27\,\text{deg}$. The $\Delta v_3$ that needs to be applied can be computed by the following formula

$$\Delta v_3 = v_1^2 \cdot v_2^2 - 2 \cdot v_1 \cdot v_2 \cdot \cos \Delta i = 0.461 \text{ km/s}. \tag{3}$$

The value of $v_1$ is the speed of the spacecraft at the apogee of the current orbit. The value of $v_2$ is the speed of the spacecraft at the apogee of the desired orbit that has its perigee at the same distance to Saturn as Titan ($r_{Saturn \to Titan} = 1221870$ km). A 2D plot of the resulting trajectories before performing the first flyby at Titan can be seen in Fig. 4. It is assumed that the entry at Saturn can be timed in a way that the Titan moon is exactly at the position of the perigee of the current orbit, when the spacecraft arrives there. If this is the case, then the spacecraft will perform a flyby maneuver around the moon. As mentioned earlier, by properly choosing the entry angle, this flyby can decrease the magnitude of the velocity with respect to Saturn and increase the inclination at the same time without applying additional $\Delta v$. After theoretical considerations of the B-plane, an entry angle of 90 deg leads to the desired behavior. Furthermore, the distance of the flyby needs to be chosen. Therefore, several aspects needs to be considered: First, as the moon has a radius of about 2500 km and has an atmosphere of about 600 km above the surface, it needs to be avoided to have a minimal distance of less than 3200 km. Second, as it is desired to perform multiple consecutive flyby maneuver around Titan, it needs to be ensured that the the new orbit period fulfills the following equation:

$$n \cdot T_{Orbit} = m \cdot T_{Titan} \quad \text{with} \quad n, m \in \mathbb{N}. \tag{4}$$

Otherwise Titan would not be met on the next perigee and the spacecraft would have to wait until luckily meeting Titan during future orbits. The appropriate flyby distance of each

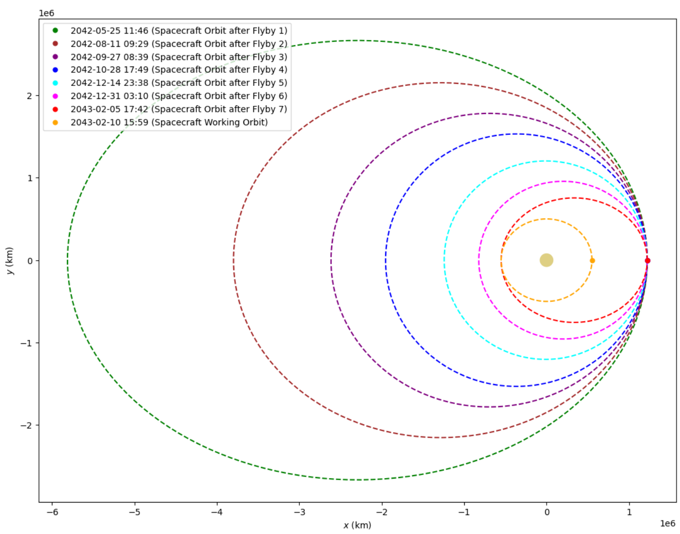



Figure 5: 2D plot of the different orbits after the flybys around Titan during entry and of the final working orbit.

flyby maneuver for achieving this is found by simply trying out different values. The orbital period of Titan is $T_{Titan} = 15.945$ days.

In total seven flybys are performed. An overview of the results is given in Table 4. As can be seen, the perigee distance of the last orbit is very close to the radius of the desired working orbit. The same holds for the inclination. Therefore, during the resulting orbit after the last flyby, a final $\Delta v_4$ is applied at its perigee in order to get into the desired working orbit. This is done by simultaneously changing the inclination to the desired 90 deg and the apogee to the same distance as the perigee. This leads to the desired circular polar orbit with a radius of 551312.59 km. The amount of $\Delta v$ can be computed again based on the following formula:

$$\Delta v_4 = v_1^2 \cdot v_2^2 - 2 \cdot v_1 \cdot v_2 \cdot \cos \Delta i = 1.471 \text{ km/s}, \tag{5}$$

where $v_1$ is the speed of the spacecraft at the perigee of the orbit after the last flyby and $v_2$ the speed of the spacecraft in the circular orbit. A 2D plot of the flyby orbits and the resulting working orbit is illustrated in Fig. 5. It has an orbit period of 4.83 days and the speed of the spacecraft is about 8.29 km/s.

### *6.4.3. Deorbiting trajectory*

After the spacecraft successfully observed the entire surface of Saturn in the working orbit, we have to ensure that it will be deorbited in a controlled way. Therefore, several flyby around Titan are used again to slow down the satellite so that it will crash into Saturn. This can be done by doing the flyby maneuver with an entry angle of 90 deg. The transfer

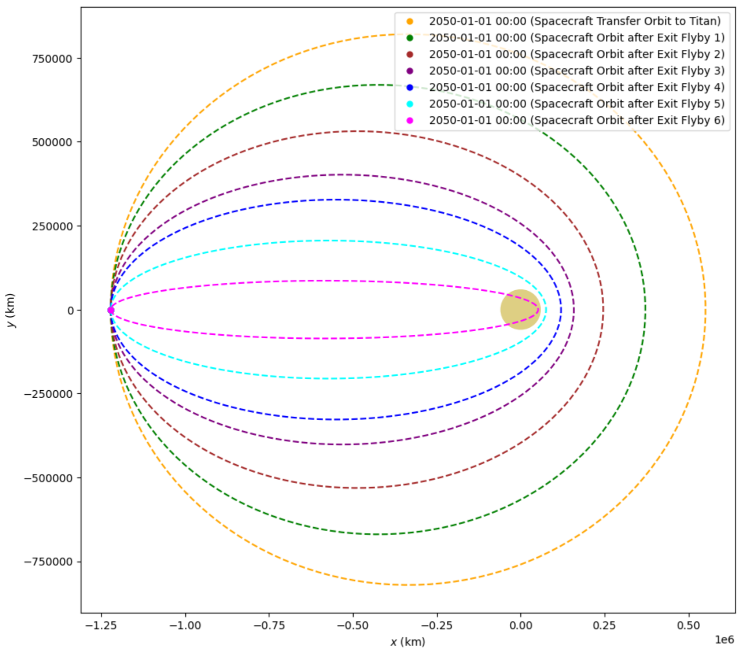



Figure 6: 2D plot of the different orbits after the flybys around Titan in order to deorbit the spacecraft.

| Flyby | 1 | 2 | 3 | 4 | 5 | 6 |
|---|---|---|---|---|---|---|
| Perigee [$10^3$ km] | 371.92 | 246.386 | 158.76 | 120.66 | 75.98 | 52.47 |
| Flyby radius [km] | 3200 | 3200 | 3200 | 5500 | 3200 | 3200 |

Table 5: Summary of the different flyby maneuvers to deorbit the spacecraft.

orbits are again influenced by the flyby distance. Thus, the flyby distance is chosen in a way that all perigees lie in gaps of the rings. This is done to ensure, that the spacecraft won't be hit by particles of the rings in order to lower the risk of losing the control of the spacecraft and increasing the amount of space debris. The only exception is made for the first two transfer orbits. They will pass the E-ring of Saturn, but as it only contains really small particles, it should not have a big impact on the spacecraft. First, the apogee of the working trajectory needs to be increased again to meet Titan. Thus, another $\Delta v_5$ is applied. This can be calculated based on the last orbit before circularizing and results in $\Delta v_5 = 1.443$ km/s. Afterwards, several flybys are performed and an overview is given in Table 5. No care is taken about the orbit period of the resulting orbits. As the mission objectives are successfully accomplished and it does not matter how long the deorbiting lasts, it is not necessary to meet Titan after each flyby as soon as possible, but the spacecraft can wait until it meets Titan at the perigee by chance again. The flyby distances are only chosen in a way that the spacecraft passes the gaps of Saturn. A 2D plot of the different trajectories can be seen in Fig. 6.

### *6.5. Launch window and mission duration*

In Table 6 the dates of the main milestones of the mission are listed. It can be seen that the spacecraft only requires 5 years and 5 months. Furthermore, as the launch date of the optimal trajectory is in about 13 years, this will give enough time for developing the final design, building the spacecraft as well as testing all modules.

### *6.6. Demonstration of fulfillment of requirements*

Based on the results of the simulations and calculations, all requirements concerning the trajectory are met successfully. An overview of these requirements is given in Table 7.

| Event | Date |
|---|---|
| Departure from Earth | 2037-09-24 |
| Departure from LEO | 2037-09-25 |
| Flyby at Jupiter | 2039-07-29 |
| Arrival at Saturn | 2041-09-14 |
| Arrival in working orbit | 2043-02-10 |

Table 6: Summary of the dates for the main milestones of the mission.

| Requirement | Fulfilled |
|---|---|
| The mission shall conduct at least 7 flybys around Titan | ✓ |
| The spacecraft must be on a polar orbit around Saturn to guarantee poles coverage. | ✓ |
| The operational orbit shall has an altitude above 490000 km to avoid impacts with ring's particles. | ✓ |
| The operational orbit shall has an altitude below 600000 km to guarantee high quality images of Saturn. | ✓ |
| The operational orbit shall be circular to take pictures always at the same distance. | ✓ |
| The operational orbit shall not pass through the Rhea's sphere of influence to avoid orbital deviations (Rhea's orbit average radius is 527000 km). | ✓ |
| The flyby around Jupiter must have a perigee above 600000 km to not pass through the high radiation area around the Planet. | ✓ |
| The Flyby around Titan must pass above the Titan's atmosphere. | ✓ |
| The approach to Saturn and the orbits before the achievement of the operational orbit must pass through ring gaps or thin rings. | ✓ |

Table 7: Overview of the requirements on the trajectory.

### *6.7. Delta-v budget*

A $\Delta v$ budget is a critical part of the mission planning. The $\Delta v$ is the measure of the change of the velocity the spacecraft needs to complete the mission objectives. Regarding the $\Delta v$ budget, three phases can be defined. For all phases some additional margins are taken into account as described in detail in Section 7.1. Phase 1 includes traveling from LEO to Jupiter's sphere of influence leading to $\Delta v_1$. Phase 2 includes small maneuvers and trajectory corrections during the flyby at Jupiter until Saturn. Phase 3 includes entering Saturn's sphere of influence, getting into the working orbit and deorbiting the satellite at the end of the mission leading to $\Delta v_3$. An overview of the the $\Delta v$ budget is shown in Table 8.

| | Phase 1 | Phase 2 | Phase 3 |
|---|---|---|---|
| Components | $\Delta v_1 = 6.848$ km/s<br>$\Delta v_{Margin1} = 0.150$ km/s | $\Delta v_{Margin2} = 0.100$ km/s | $\Delta v_2 = 2.203$ km/s<br>$\Delta v_3 = 0.461$ km/s<br>$\Delta v_4 = 1.471$ km/s<br>$\Delta v_5 = 1.443$ km/s<br>$\Delta v_{Margin3} = 0.600$ km/s |
| Sum phase | $\Delta v_{Phase1} \approx 7.0$ km/s | $\Delta v_{Phase2} \approx 0.1$ km/s | $\Delta v_{Phase3} \approx 6.2$ km/s |
| Total | $\Delta v_{total} \approx 13.3$ km/s | | |

Table 8: $\Delta v$ budget.

## 7. Budgets

### *7.1. Safety margins*

As noted earlier, not all maneuvers will be ideal, requiring small corrections. Additionally, small maneuvers are needed to ensure the correct angles and distances for flybys. Based on the advice of the supervisor, considering additional margins of 5%-10% should be sufficient. Thus, 0.6 km/s are considered as the margin for the $\Delta v$ phase 3 as many flybys are performed. During the first phase, the spacecraft only has to perform one big boost to get into the interplanetary trajectory to Jupiter, and so a smaller margin of 0.15 km/s is taken into account. As an exact flyby around Jupiter is crucial for the success of the mission, we consider an extra margin of 0.1 km/s to ensure to achieve this.

### *7.2. Propulsion mass estimation*

To calculate the required mass of the propellant for an interplanetary mission from LEO to Saturn's polar orbit, we use the Tsiolkovsky rocket equation. The key parameters involved in this calculation are the required change in velocity for each phase of the mission ($\Delta v$), the specific impulse of the engine ($I_{sp}$), the gravitational constant ($g_0 = 9.81$ m/s$^2$), the initial mass $m_0$ (including the spacecraft and propellant) and the final mass $m_x$ (dry mass of the spacecraft, excluding the propellant). The difference between initial mass $m_0$ and the final mass $m_f$ gives the propellant mass. The interplanetary mission from LEO to Saturn's polar orbit consists of two stages and three phases. Each phase has different $\Delta v$ requirements, leading to different propellant mass calculations for each segment. The calculation begins with the final phase (Phase 3), and its resulting propellant mass is incorporated into the calculations for the preceding phases to determine the required propellant at each stage. The propellant mass required for Saturn's polar orbit insertion is calculated first (Phase 3 - Stage 2). This mass is used as part of the final mass for Phase 2. The propellant mass required for the trajectory correction from Jupiter to Saturn's outer atmosphere is calculated next (Phase 2 Stage 2). The final mass from Phase 3 is considered in this step. The propellant mass required to travel from Earth's LEO to Jupiter's atmosphere is computed last (Phase 1 - Stage 1), incorporating the propellant mass for the first two phases. A Python program is developed to compute the propellant mass for different engine configurations and to ensure the total mass remains within the payload limitations. Detail can be seen in the Appendix A. Table 9 summarizes the results for each phase. The total mass of the propellant required for the entire mission from LEO to Saturn's orbit is 57540.32 kg, and the total liftoff mass, including all the structure and payload, is 61918.32kg, which is below the maximum payload limit of 63,800 kg of Falcon Heavy launch vehicle.

| Stage | Phase | $\Delta v$ (m/s) | $I_{sp}$ (s) | Propellant Mass (kg) |
|---|---|---|---|---|
| 2 | 3 | 6200 | 312 | 49067.46 |
| 2 | 2 | 100 | 312 | 312.21 |
| 1 | 1 | 7000 | 453.8 | 8160.65 |

Table 9: Propellent mass for stages and phases

*7.3. Power budget and power system estimation*

In order to determine a power budget and select the appropriate power generation components, we must identify each element that consumes power, and it must be established how much power said elements consume on average, as well as on maximum settings (peak power). The estimated values have been assessed by using previous missions as reference such as Cassini, Juice and Stardust. The values are reported in Table 10.

It is also important to ponder the activity period of each component, to understand how much power will be used constantly, and how much power will be required only at certain instances. In light of the above, it is established that both magnetometers, the solar wind monitor, the star tracker and the computer will always be on. This is to be expected given the critical nature of some of those components and/or the necessarily long periods of usage for effectiveness (e.g. the extensive mapping of magnetic fields with the magnetometer). Other components that are often used are the narrow angle cameras and the wide angle camera. Those cameras will work on periods designated for observation, and only one type of cameras will be on at a time (so we will consider the bigger power value between the two narrow angle cameras and the wide angle camera). The IR camera will also be on frequently, as well as the plasma spectrometer, the IR spectrometer, and the reaction wheels. This equates to 249 W of power that will be required most of the time, and should be our target in terms of average power generation. A correction of 10% is implemented to account for malfunctions, miscalculations, and other unpredictable power fluctuations, which results in 274 W of power. While those components essential to the mission will be on most of the time, there will be other components working at the same time occasionally, namely the dust analyzer and the Antennas during telecommunication periods, whereas some instruments will only be used during deorbiting, that is, the dust collector (will only

| Instrument | Average Power [W] | Peak Power [W] |
|---|---|---|
| Narrow angle camera (x2) | 25 (x2) | 40 (x2) |
| Wide angle camera | 15 | 30 |
| IR camera | 15 | 30 |
| Magnetometer (x2 ) | 5 (x2) | 8 (x2) |
| IR spectrometer | 30 | 50 |
| Dust analyzer | 15 | 25 |
| Plasma spectrometer | 15 | 20 |
| Solar wind monitor | 10 | 15 |
| Dust collector (Stardust) | - | 25 |
| Onboard spectrometer | 35 | 50 |
| Star tracker (x3) | 8 (x3) | 15 (x3) |
| Reaction wheels (x3) | 5 (x3) | 30 (x3) |
| Computer | 80 | 130 |
| Antennas | 20 | 28 |

Table 10: Power budget

be used at peak power) and the on board spectrometer. Moreover, at certain instances, some components will operate at higher power than their average consumption. Despite that, occasional spikes in power requirements do not warrant more RTG's (which can be quite costly and harder to manage), since they can be solved by the usage of batteries. In terms of power efficiency degradation, it can be more or less ignored when it comes to the Plutonium-238 decay, given that its half-life is 87.7 years, meaning a low decrease in thermal power output in its first few years, for example, 7.6% in the first 10 years, based on the exponential half-life model (which is approximately insignificant since the thermal to electrical power conversion is already a low efficiency process). However, the thermoelectric materials used to convert heat to electricity degrade at a higher rate, approximately 0.3% to 0.5% every 1000 hours of usage [6, 7]. This translates to a few tens of Watts over the course of years, and given how long deep space missions can last, we chose a single GPHS-RTG (about 300 W beginning of life) [8], as it allows a margin of error to account for degradation. Other RTG options were explored, and despite being lighter and cheaper as a singular unit, their low power output would require the usage of at least two units, which would be more problematic and expensive than the GPHS-RTG. A Nickel-Hydrogen 35 kg battery was chosen to store excessive RTG power, and to be used during extra power requirements that exceed the RTG capabilities. Although this type of battery has lower energy density than modern Lithium-ion batteries, it wins in terms of longevity and amount of charge/discharge cycles, which is crucial for deep space missions. During extremely rare occurrences, while most systems are working at/close to peak power, the battery is capable of aiding the RTG in meeting the necessary momentary requirements (about 500-600 W). Given that RTG's are usually separated from the main components of a spacecraft using low thermal conductivity materials, they should not be an issue in the taking of infrared pictures. It would still be pertinent to further investigate the situation in a future planning iteration, and even assess possible solutions such as filtering techniques if this becomes a problem.

### *7.4. Link budget and comm estimation*

The link budget accounts for all the gains and losses from the transmitter to the receiver in a communication system:

Received power (dBm) = transmitted power (dBm) + gains (dB) - losses (dB)

We'll calculate it as

$$P_r = P_t + G_t + G_r - L_t - L_r - L_s - L_a \tag{6}$$

$P_t$ is the transmitted power (in dBm or W), 20 W (43 dBm). $G_t$ is the Gain of the transmitting antenna (in dBi), calculated as

$$G_t = 10\log_{10}(\frac{A_e}{A:0}) \tag{7}$$

with $A_e = \eta \cdot A$, $\eta \in [0.55\ ,\ 0.7]$, $A = \frac{\pi D^2}{4}$, $D = 4m$, $A_0 = \frac{\lambda^2}{4\pi}$, $\lambda = \frac{c}{f}$, $f = 8.4Ghz$. This leads to a $G_t$ of 42 dBi. $G_r$ is the gain of the receiving/DSN antennaa (in dBi), calculated as:

$$G_r = 10\log_{10}\frac{4\pi A_e}{\lambda^2} \tag{8}$$

For the bigger 70m antennas, $G_r = 74$ dBi; for the smaller 34m antennas, $G_r = 64$ dBi.

$L_t$ is the losses in the transmission line from the transmitter (in dB), which account for losses in the cables, connectors, and other RF components between the TWTA and the high-gain antenna on the spacecraft. They're around 1 to 2 dB. $L_r$ is the losses in the transmission line from the receiving antenna (in dB), which accounts for losses in the cables, waveguides, and RF electronics at the DSN ground station. It's also usually 1 to 2 dB. $L_s$ is the free space path loss (in dB), calculated as:

$$L_s = 20 \log_{10} \frac{4\pi d}{\lambda} \tag{9}$$

The worst case scenario is when Saturn and the Earth are the furthest apart from each other, at around 1.8 billion kilometers. This leads to an $L_s$ of 296 dB. Finally, $L_a$ is the additional losses (in dB) which can come from atmospheric attenuation (minimal for space communications), polarization mismatch (if the transmitter and receiver antennas are not perfectly aligned), and other small effects like mispointing or noise in Earth's atmosphere. It's around 1 to 2 dB. With all this, our final $P_r$ is around -150 dB, which is a valid and expected value for a deep space mission. Deep Space Network receivers are highly sensitive, operating at noise floors near -170 to -160 dBm in the X-band. As such, we have a link margin of $P_r - P_{r\,\mathrm{min}} = 15$ dB.

### *7.5. Mass budget*

In this section, an estimation of the mass of the mission has been assessed, by using as reference previous missions that used similar components, such as Cassini, Juice and Stardust for the dust collector. The overview is given in Table 11. It can be seen, that this estimation stays below the requirement of 900 kg.

| Component | Weight [kg] | Component | Weight [kg] |
|---|---|---|---|
| Narrow IR camera | 40 | Onboard spectrometer | 30 |
| Narrow visible camera | 40 | Infrared spectrometer | 40 |
| Wide angle camera | 50 | Reaction wheels (x4) | 10 (x4) |
| Helium vector magnetometer | 4 | Computer/Hardware | 40 |
| Fluxgate magnetometer | 3 | Antennas | 44 |
| Plasma spectrometer | 15 | RTG | 57 |
| Dust collector | 10 | Battery | 55 |
| Dust analyzer | 20 | Structure | 380 |
| Star tracker (x3) | 5 (x3) | | |
| Total | | 883 kg | |

Table 11: Mass budget with estimated weights.

## 8. Launcher Selection

### *8.1. Launcher Selection*

Choosing the right rocket launcher for the Saturn mission is critical due to the spacecraft's weight of about 62,000 kg. Several launchers were considered, focusing on payload capacity and cost. SpaceX's Falcon 9, though widely used and affordable at under $30 million, can only carry 22,800 kg to Low Earth Orbit (LEO), making it unsuitable for our 62,000 kg payload. Falcon Heavy, also by SpaceX, offers a payload capacity of up to 63,800 kg to LEO, which meets the mission's requirements. At an estimated cost of $97 million (approx. €92 million), it provides a cost-effective and reliable option. Falcon Heavy has proven itself with missions like NASA's Europa Clipper, showcasing its ability to launch heavy payloads on interplanetary missions. The Atlas V, despite its reliability, cannot carry more than 18,800 kg to LEO and has a much higher cost, ranging between $110 million and $153 million, making it both insufficient and more expensive than Falcon Heavy. Ariane 6, still in development, offers a maximum payload of 10,350 kg to LEO in its A64 configuration, which is far below what is needed. With prices ranging between €70 million and €115 million, it remains an unproven and underpowered option for this mission. Given the payload requirements and cost considerations, Falcon Heavy is the optimal choice for the Saturn mission, offering sufficient capacity and competitive pricing.

### *8.2. Shared flight evaluation*

Given the sheer size and weight of our 62,000 kg probe, sharing the Falcon Heavy's launch with other payloads is not a feasible option. Falcon Heavy's total capacity to Low Earth Orbit (LEO) is 63,800 kg, and our spacecraft utilizes nearly the entire available payload capacity of the rocket. This leaves no room for additional satellites or secondary payloads, which are typically included in shared flights to offset launch costs. Furthermore, even if there were some marginal capacity available, integrating other payloads into such a high-stakes mission would introduce unnecessary risks. Shared flights often involve smaller, experimental satellites or commercial payloads, which can increase the complexity of the mission and introduce potential hazards, such as deployment issues or interference with our spacecraft. It's also important to mention the fact that the deployment in LEO will be made at 200 km of altitude which is not a desired altitude for almost any type of satellite since it's too low and the atmospheric drag is a really significant problem.

### *8.3. Launcher assessment and selection*

After evaluating all alternatives, it is clear that the Falcon Heavy is the only launch vehicle capable of delivering our 62,000 kg spacecraft to LEO. It provides the necessary payload capacity, reliability, and cost-effectiveness for such a mission. With the successful launch of the Europa Clipper on 14th October 2024 we get further confirmation of its suitability for deep-space exploration.

### *8.4. Insurance Considerations*

We have opted not to purchase launch insurance for this mission, as we are confident in the high reliability of the Falcon Heavy, which benefits from Falcon 9's proven technology and boasts a strong track record in successful missions. Given its 99% success rate, the likelihood of a catastrophic failure is minimal, and we believe the financial risk is manageable without insurance. However, if we were to secure insurance, it would typically be around 20% of the total mission cost.

## 9. Technology Assessment, Cost and Risk

### *9.1. Technology Assessment*

Interplanetary missions to Saturn require thorough technology readiness assessments to ensure all systems are prepared. Key instruments like visible and infrared cameras, and spectrometers, though proven in space (TRL 8-9), need further development over the next few years. Magnetometers, ranging from TRLs 7-9, and sample analysis tools at TRLs 6-7, also require enhancements. Propulsion is vital, with chemical propulsion well-established at TRL 9, while ion thrusters, more efficient but at lower TRLs (6-8), need further development. The spacecraft will rely on RTGs at TRL 9, though adjustments are needed for Saturn's cold environment. Navigation systems are advanced (TRL 8-9), but radiation-hardened versions and improved autonomous capabilities (currently TRL 5) will be required. Thermal protection using high-TRL insulation will need refinement for long-term use. Communication systems are mature, but improvements in data transmission and error correction will boost reliability. Despite maturity in many areas, propulsion, thermal management, and autonomous navigation need further refinement for mission success.

### *9.2. Cost and Schedule Estimation*

In this section an estimation about costs and schedule needed for the development of the technology involved in the mission is evaluated. Since the mission instrumentation is starting from known technologies, many of which have already been tested in space missions like Cassini, Juice and others, the estimation will focus on adapting and improving these instruments, rather than starting from scratch. According to literature information, it is possible to assess, at a high level, the time required for the development of the necessary instruments, in particular: instruments like magnetometers and cameras , which are highly flight proven could require around 3.5 years; more complex instruments, like the plasma and infrared spectrometer and the dust collector and on-board spectrometer could require more time, around 4.5 years. Since the required time is not short and the mission includes multiple instruments, a parallel development is the only option. Always referring to previous missions, in particular the Cassini mission, a high level cost estimation for the instruments can be assessed which is given in Table 12. Additionally, the cost of the propellant is estimated to 1 - 2 M €. Launching with Falcon Heavy will add a cost of about 92 M €.

| Instrument | Estimated cost [€] | Instrument | Estimated cost [€] |
|---|---|---|---|
| Narrow angle camera (x2) | 30 - 40 M | Wide angle camera | 40 - 60 M |
| Magnetometers (x2) | 5 - 10 M | Plasma spectrometer | 20 - 30 M |
| Dust analyzer | 20 - 25 M | Infrared spectrometer | 45 - 60 M |
| Dust collector | 10 - 20 M | On-board spectrometer | 35 - 55 M |
| GPHS-RTG | 60 - 110 M | Battery | 0.5 - 1 M |
| Star trackers (x3) | 4 - 10 M | Reaction wheels (x4) | 2 - 4 M |
| Computational unit | 300 - 400 M | Antennas | 2 -3 M |
| Propulsion | 15 - 25 M | Total | 632.5 - 935 M |

Table 12: Cost estimation

### *9.3. Risk Analysis and Mitigation Strategies*

Risk analysis and mitigation are crucial for ensuring mission success. Identifying mission-threatening risks early helps in developing strategies to minimize them. Technical failures pose a significant risk, including potential issues with propulsion, communication, or power systems. To mitigate these, redundancy is built into key systems, and thorough pre-launch tests and simulations will be conducted under conditions mimicking Saturn's environment, such as extreme cold and high radiation. Trajectory errors during the cruise phase are also a risk, as even minor deviations can prevent the spacecraft from reaching Saturn. To counter this, the mission will involve precise trajectory planning, real-time navigational corrections via NASA's Deep Space Network (DSN) and ESA's ESTRACK, and gravitational assists to ensure accurate orbit insertion. Space environment risks include cosmic radiation, micrometeoroids, and high radiation levels near Saturn. Radiation shields and segregated cradles will protect critical systems, while real-time space weather monitoring will adjust spacecraft operations to avoid harmful conditions. Cost overruns and delays are common in interplanetary missions. To mitigate this, the mission will follow strict project management practices with defined milestones, a 20-30% contingency budget, and regular internal and external reviews to stay on track. Political and regulatory risks could impact international collaborations, launch windows, or regulations. Global partnerships with NASA, ESA, and others will ensure access to key resources and infrastructure, with ongoing monitoring to remain compliant with evolving regulations

### *9.4. Legal, Ethical and Environmental Considerations*

Compliance with international space law, ethical standards, and environmental considerations is crucial for the Saturn mission. As the mission is developed in Portugal, it must follow international treaties, European regulations, and national laws. The 1967 Outer Space Treaty ensures the mission is scientific and non-military, prohibits harmful contamination of celestial bodies like Saturn's moons, and requires Portugal to register the spacecraft with the UN for transparency. Portugal is also liable for any damage caused by the spacecraft under the Rescue Agreement, Liability Convention, and Registration Convention. The mission must adhere to European Space Agency (ESA) guidelines, including strict sterilization protocols to prevent contaminating Titan and Enceladus. ESA's space debris regulations also require a clear end-of-life plan, such as deorbiting the spacecraft into Saturn's atmosphere. Nationally, Portugal's space law (Law No. 54/2019) mandates licensing from the Portuguese Space Agency and an environmental risk assessment focused on space debris. The mission must also ensure transparency and open access to data under international treaties, especially if signs of life are detected. Public engagement is essential for support, with regular updates on risks, progress, and discoveries. Portugal is committed to keeping the public informed through outreach initiatives, ensuring the mission's findings are shared with the global community.

## 10. Conclusion and Recommendations

### *10.1. Summary of findings from Pre-Phase A studies*

This document highlights the scientific relevance and the feasibility of the SMADGROUP2 mission, in particular demonstrating the fact that the goal orbit is reachable in a reasonable time and that the scientific objectives of the mission are both achievable and highly valuable. The mission leverages knowledge from previous missions and the usage of modern instrumentation, but starting from flight proven technologies, ensuring a cost-effective and efficient mission.

### *10.2. Recommendations for proceeding to Phase A*

Looking ahead to the next phase of SMADGROUP2 development from Pre Phase A to A, we as a community need to work on recommendations that inform how best we can proceed. These are the main guidelines for the transition to Phase A. While the initial mission concept has been laid out, Phase A will dive deeper into alternative mission designs and determination of the final architecture. The trade studies that were performed in Pre-Phase A need to be revisited and the areas of likely maximum impact (including propulsion systems, power sources and payload configurations) need choices refined. Is important that engineering teams closely work with their science stakeholders in order for both parties to develop a set of shared desired outcomes. As highlighted in the technology assessment, most of the mission's core systems, such as power, communication, and attitude control, are at high TRL levels,but some others, in particular instrumentation, since they must be upgraded, are generally recommended to have a Technology Development Plan (TDP) established prior so that they can be raised in readiness enough to serve the mission. Additionally, ideal opportunities available through technology partnerships need to be identified with specific entities or agencies like NASA/ESA who have gained experience in a similar domain. Phase A will proceed with a very detailed risk analysis, we suggest that a separate risk management team be set up to review new risks as the mission details develop. The team must also create a well-rounded contingency plan of how they will tackle delays and budget/technical issues that arise. Particular attention should be given to cost drivers that may impact the mission volume as well as ensuring adequate funding for high-risk components. One of the primary objectives during Phase A is to further develop and refine mission budgets/estimates. We have developed a rough estimate of magnitude cost but Phase A will go into the details for each element, conducting independent cost assessments and using pricing guidance reflecting potential commercial best practice to deliver savings in costs. The project should also characterize its funding sources at this stage and implement a process of cost control, which allows monitoring the development of mission budget expenditures in order to prevent deviations from initially planned costs.

*10.3. Identification of key areas for further study*

In the next phases, it is planned to build and test all the spacecraft components, after the choice of an adequate custom-built structure in order to develop the definitive accommodation strategy for all the subsystems and instrumentation to respect all the mission and launcher requirements. Further analysis of the spacecraft, such as FEM analysis, thermal and orbital analysis, shall be performed to consent a proper sizing of all the subsystems, such as Attitude and thermal control. Shielding from radiation and debris must be developed, such as protection of cameras' lenses. Additionally, cooling methods for infrared cameras and other payloads shall be properly selected, and compression data algorithms shall be developed. A deeper risk analysis must be conducted, to find out critical components and how to prevent their failure.

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## Appendix A. Structure of the Simulations

A detailed summary of the developed Python project which includes all the simulations can be found in the *README.md* file of the github repository. It also includes all details on the used software packages, algorithms and on how to setup and execute the simulation. However, a short summary will be given here as well. The main software package that is used is *poliastro*, which is a Python library providing functions for calculating tasks in orbital mechanics and the JPL ephemeris. All computations are performed in 3D.

*Interplanetary Trajectory*

In order to optimize the interplanetary trajectory, a simulation is developed which simulates the trajectory from the Earth to Saturn. This includes, first, solving the Lambert problem between Earth and Jupiter for a given date of departure and date of arrival. This provide the velocity of the spacecraft w.r.t. the Sun arriving at Jupiter as well as the velocity of the spacecraft w.r.t. the Sun required at Earth. Based on the needed velocity vector at Earth, on the velocity vector of the Earth itself at that time and on the velocity the spacecraft has at LEO, the $\Delta v$ is determined that needs to be applied to the spacecraft at LEO. Afterwards, based on the velocity of the spacecraft w.r.t. the Sun when arriving at Jupiter, on the velocity of Jupiter at that time w.r.t. the Sun, on a specified flyby distance and based on a specified entry angle of the flyby, the resulting velocity of the spacecraft w.r.t. the Sun is determined. This velocity is then used combined with the position of Jupiter at the date of flyby to propagate the orbit of the spacecraft. This is done for 3 years. Every hour the current distance between the spacecraft and Saturn is determined. The minimal distance during this propagation is saved. After a simulation run, it can be analyzed whether the spacecraft meets Saturn during its orbit after the flyby or not, and if it does, it can be analyzed how long it takes and with which velocity the spacecraft arrives.

This basic structure of the simulation is extended in a way that a range of the input parameters (date of departure, date of arrival, flyby distance and flyby entry angle) can be selected and all possible combinations are simulated. The results of each run are saved and can be compared afterwards. In that way an optimal trajectory can be found by iteratively simulating a high amount of possible trajectories.

*Entry, Working and Exit Trajectory*

In order to find a feasible trajectory for entering Saturn, getting into the working orbit and exiting the orbit after the mission many different approaches have been tried out. As there is not one best way that can be optimized, it was not possible to build a simulation structured as the one for the interplanetary trajectory. However, two possible approaches are simulated and presented using Jupyter notebooks. They consist of basic orbital mechanics considerations combined with many flyby simulations.

*Mass Estimation*

In order to estimate the required mass of the propellant and of the tanks for storing the propellant based on the total amount of $\Delta v$, a short Python script is implemented.

## Appendix B. Contributions

In the following, the contributions of each group member within this project are listed consisting of the covered sections as well as overall percentage of his contributions.

Joshua Redelbach (ist1112470) was responsible for the sections 2, 3.3, 4.3, 6.1, 6.4 - 6.7, 7.1 as well as Appendix A. Furthermore, he implemented all simulations regarding the trajectory and helped for calculating the mass of the propellant. Additionally, he reviewed this report at a final stage. His contributions equals 100%.

Matteo Bertolaso (ist1112776) was responsible for the sections 1, 3.1, 4, 5.2, 7.5, 9.1, 9.2, 10.1, 10.3. Furthermore, he has contributed particularly in the field of orbital mechanics and trajectory planning and design. Additionally, he reviewed this report at a final stage. His contributions equals 100%.

Bernardo Silva (ist199467) was responsible for sections 5.1 and 7.3. He also contributed in spacecraft system selection and considerations, as well as in reviewing the report at its final stages. His contribution equals 100%.

Virupaksha Thayavanagi Manjunatha (ist112621) is directly responsible for the sections 3.2, 6.2 and 7.2. Furthermore, he contributed to section 6.7 and discussions for the trajectory and orbit selection. His contribution equals 100%.

Miguel Ramos Fonseca (ist199536) was responsible mainly for communications, in sections 5.1 and 7.4. He also wrote sections 9.1, 9.3, 9.4, and 10.2. He also had a major contribution in reviewing the report at its final stages and transfering information into the final LaTeX document. His contribution equals 100%.

João Félix (ist197238) was responsible for sections 3.4 and 6.3, aswell as report revision of several chapters at the end stage. Was also responsible, alongside Matteo, for the development of the orbital mechanics calculations to determine the trajetory, which backed Joshua software development. His contributions equals 100%.

Pedro Sousa (ist199542) was mainly responsible for sections 8.1, 8.2, 8.3 and 8.4 related with the launcher selection and insurance assessment as well as a contribution on the overall report revision. Furthermore, was a contributor in discussions about critical parts of the projects including some parts of the orbit selection. His contribution equals 100%.