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MORPHEUS observatory located at the Sun-Mars L1 Lagrange point for data acquisition and protection of future Mars missions

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Abstract

The Mars-Sun Lagrange point L1 offers a strategic vantage for studying solar wind and solar activities with direct implications for planetary science and interplanetary exploration. This paper introduces MORPHEUS (Martian Observatory of Radiation Phenomena for the Heliospheric Environment and Unprecedented Shielding experiment), a proposed mission with multifaceted objectives: in-situ solar wind measurements, heliospheric observations, and an innovative experiment designed to advance protective technologies against space weather. MORPHEUS will deploy a novel mechanism to generate a localized magnetic field as a protective shield, simulating potential solutions for safeguarding future Mars missions. The impact and efficacy of the shield will be quantified using a Faraday cup mounted on a telescopic arm, enabling direct measurements of deflected solar wind particles. Additionally, MORPHEUS will carry instruments to observe the Sun and the heliosphere, providing valuable data to refine models of the Sun's interactions within the inner solar system. It will also be equipped with two antennas: one for Earth communication and another directed toward Mars' surface. This Mars-facing antenna could offer early warnings to mitigate imminent threats posed by solar activities to astronauts, rovers, or other Mars-based human artifacts. With a launch expected no earlier than August 2037, MORPHEUS aims to deliver unprecedented insights into solar wind dynamics, magnetic shielding technologies, and heliospheric science.

1. Introduction

Space weather monitoring studies the Sun and the Heliosphere. By means of observation, monitoring, analysis and modeling, it has several objectives: on the one hand, to understand and predict the state of the Sun and interplanetary or planetary environments, as well as the disturbances that affect them, whether of solar origin or not; on the other hand, to analyze in real time or predict possible effects on biological and technological systems. Many space weather missions exist to study the impact of Solar events on the Earth such as the SOHO mission launched in 1995.[1] However, with the increasing interest in Mars' exploration in the space community, and the aim to send humans[2] by 2050, it becomes important to study space weather around Mars.

Early missions around Mars sent back the first pictures of the planet (in 1965 by MARINER 4[3]), performed the first soft landing (in 1971 by Mars 3), conducted global mapping of the entire planet (in 1997 by Mars Global Surveyor[4]) and mapped the chemical and minerals that make up the Martian surface (in 2001, Mars Odyssey, still active[5]). Current active missions look at the history of water across the globe (in 2004, Mars Express[6] and in 2005, Mars Reconnaissance Orbiter[7]) and the possibility that the planet once hosted life (in 2012, Curiosity[8] and 2020, Perseverance[9]).

In 2014 the still active MAVEN probe[10] started orbiting Mars to examine how and at what rate the solar wind erodes away the atmosphere. While the spacecraft provides large amounts of information for scientists to design complex theoretical models, we miss empirical monitoring of the Sun as seen from Mars to complement this probe.

The **M**artian **O**bservatory of **R**adiation and **P**henomena, for the **H**eliospheric **E**nvironment and **U**precedented **S**hielding experiment (**MORPHEUS**) has for main objective to continuously monitor the Sun from a new perspective and provide new data. This will improve current space weather models, provide a better understanding of the impact

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of solar winds on the atmosphere of Mars,[11] and provide crucial data to protect spacecrafts and enable safe human exploration for the entire trip to and from Mars.[12] The data provided by the spacecraft could also help forecast weather on Earth, as in fact, the best viewing locations for Earth space weather forecasting is away from the Sun-Earth line.[13] This mission was developed as part of the course *Spacecraft design and systems engineering* given at EPFL for the minor in Space Technologies.

2. Science objectives

The MORPHEUS mission's objective is to answer three main scientific questions, which are listed below.

2.1 What are the properties of the in-situ solar wind reaching Mars?

Solar flares produce intense electromagnetic radiation and can accelerate particles to several hundred MeV/nucleon, or even a few GeV/nucleon. Solar Energetic Particles (SEP) events mainly consist of protons, with about 10% helium and <1% heavier elements. High-energy protons (>10 MeV) pose biological risks to astronauts on Mars, especially during extravehicular activities or in lightly shielded habitats. SEP events can occur during solar minimum, and could deliver a lethal dose on the transit to/from Mars, while their timing remains unpredictable. Surface and orbital measurements at Mars of the incident flux are required to validate the model predictions before crewed missions can be launched.[14]

2.2 How can we predict space weather on Mars?

Coronal mass ejections (CMEs) have been found to rotate, deflect, distort and interact with each other in the inner heliosphere. The understanding of their propagation and evolution in the heliosphere, further than 1 AU from the Sun, was improved by the observations of the heliospheric imagers onboard STEREO. The SECCHI instruments allowed the 3D reconstructions of the heliosphere provided by their multi-point-of-view imaging. However, those instruments have failed to establish an agreement between in-situ and imaging-based reconstructions of CMEs, the speed profile, particularly of medium speed (< 900 km s^{-1}) interplanetary CMEs (ICMEs), is difficult to establish.[15]

2.3 How could we use a magnetic shield to protect Mars' atmosphere?

Mars Express and MAVEN missions have found that Mars has been losing a large part of its atmosphere due to direct solar wind interaction, as it lacks a magnetic field to buffer its atmosphere. With a stronger atmosphere, Mars could support heavier equipment landings, protect against cosmic and solar radiation, extend the ability for oxygen extraction, and enable open-air greenhouses for plants. Models at the Coordinated Community Modeling Center (CCMC) simulate a magnetic shield by creating a magnetic dipole at the Mars L1 Lagrange point. Emerging research shows that a miniature magnetosphere might protect humans and spacecrafts.[16]

2.4 Identified environment constraints

Following the definition of the mission's objective, identified environment constraints on the spacecraft are listed below.

REF	Description
ENV-01	Interactions of the spacecraft with high energetic radiations of the Sun.
ENV-02	Large temperature gradient between sunward and anti-sunward side of the spacecraft.
ENV-03	Interferences between experiments, especially if large magnetic fields are generated.
ENV-04	Solar conjunction for data transmission, happening every 26 months, with maximum duration of 40 days.[17]

2.5 Principal stakeholders

The principal stakeholders who should be interested in the scientific return of the MORPHEUS mission are:

- Governmental agencies, to protect their rovers and astronauts during prospective missions on Mars.
- Space weather scientists for the validation of predictive models (HAFv.2 and SWMF) for Earth.[15]
- Private companies such as SpaceX and their Mars exploration program.

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2.6 Sustainable end-of-life

We made the deliberate decision to design the mission to be more sustainable, for example, by prioritizing a lower propellant toxicity. Additionally, we ensured a sustainable end-of-life disposal plan to minimize long-term impacts on the space environment and adhere to guidelines for debris mitigation (around the Moon, as no such guidelines exist for Mars).[18] Although this choice has considerably increased the amount of propellant required on board, we consider end-of-life to be an integral part of mission success.

3. Mission design

3.1 Basic mission requirements

The statements outline essential mission requirements based on stakeholder and scientific expectations. The choice to position the spacecraft at L1 between the Sun and Mars is particularly significant, as no current mission occupies this location. This location offers a unique new viewing point and is ideal for placing a magnetic shield or continuously monitoring solar activities. The requirements are described in Table 6 and Table 7 and are separated into 8 categories corresponding to the spacecraft' subsystems : Command and Data Handling (CDH), Electrical Power System (EPS), Propulsion & AOCS, Science Payload, Scientific Experiment, Structure, Telecom and Thermal.

3.2 Concept of operations (CONOPS)

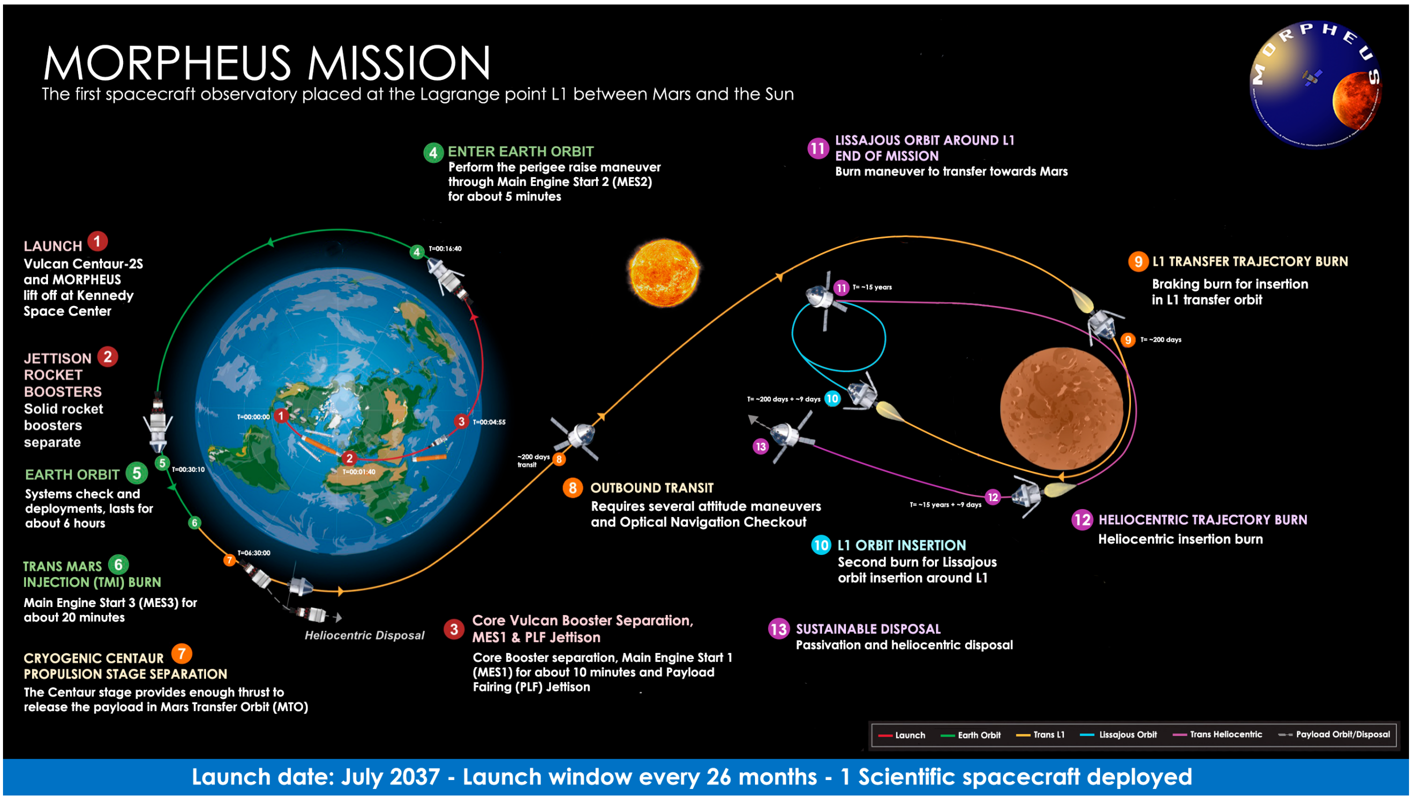

Figure 1: Diagram of the CONOPS. Inspired from[19]

3.3 Launcher and Launch Window

To maximize adaptability, the chosen launcher for MORPHEUS is ULA's *Vulcan Centaur-2S (VC2S)*. This rocket can deliver a payload of up to 3600 kg into a Mars Transfer Orbit (MTO), launching from Cape Canaveral Air Force Station in Florida. Because its payload capacity and fairing size are smaller compared to other Mars-capable launchers, such as Arianespace's *Ariane 6*, SpaceX's *Falcon 9*, and Blue Origin's *New Glenn*. Designing MORPHEUS for the Vulcan Centaur-2S ensures enhanced versatility for the spacecraft in terms of launch integration and availability. Launch opportunities to Mars occur approximately every 26 months, with optimal Δv conditions aligning roughly every seven

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synodic periods (about 15 years).[20] Based on these cycles, the optimal launch window for MORPHEUS is projected to be in July–August 2037, which is therefore considered the mission's target launch date.

3.4 Onboard Δv requirement to reach L1

The MORPHEUS spacecraft will be injected into its Mars Transfer Orbit (MTO) by the upper stage of the launcher. After separation, the remaining fuel will target the upper stage in a way that off-points it from Mars to ensure a low probability of it hitting the planet and sending it into a heliocentric orbit, which is considered sustainable.[18] The Vulcan Centaur-2S launcher delivers the payload into MTO without relying on the spacecraft's propulsion.

To reach the L1 orbit, a transfer braking manoeuvre at Mars periapsis will be performed, yielding a lower Δv compared to direct injection.[21] The precise Δv required depends on the launch timing, influenced by Mars' orbital inclination and eccentricity. Based on estimates,[21] the braking and orbit insertion at L1 will require (with margins) approximately Δv = 2 km/s.

3.5 Onboard Δv requirement for stationkeeping

The value of the Δv needed to maintain the spacecraft depends on the type of controlled technique used. While a precise control technique can maintain a periodic halo orbit, a loose one will make a Lissajous orbit. SOHO was loosely controlled and we assume the same here, because as long as it points towards the Sun the exact position has no consequence on the mission.[21] The stationkeeping for the Mars Lissajous orbit will be in the same order as for L1 Sun-Earth missions. SOHO used 2.3 m/s per year on average so we chose Δv = 3 m/s for the design of MORPHEUS.

Thus, the spacecraft will be placed in a Lissajous orbit, a three-dimensional quasi-periodic trajectory around a libration point (such as the Sun-Earth L1 or L2), where gravitational and centripetal forces are balanced, enabling stable observation or communication with minimal fuel usage.

3.6 Onboard Δv requirement for disposal

Following ESA mitigation requirements,[18] the recommended disposal of the spacecraft is a heliocentric orbit. By symmetry with the required velocity change to approach L1, we estimate Δv = 2 km/s to reach a heliocentric orbit at the end of the mission.

3.7 Mission timeline

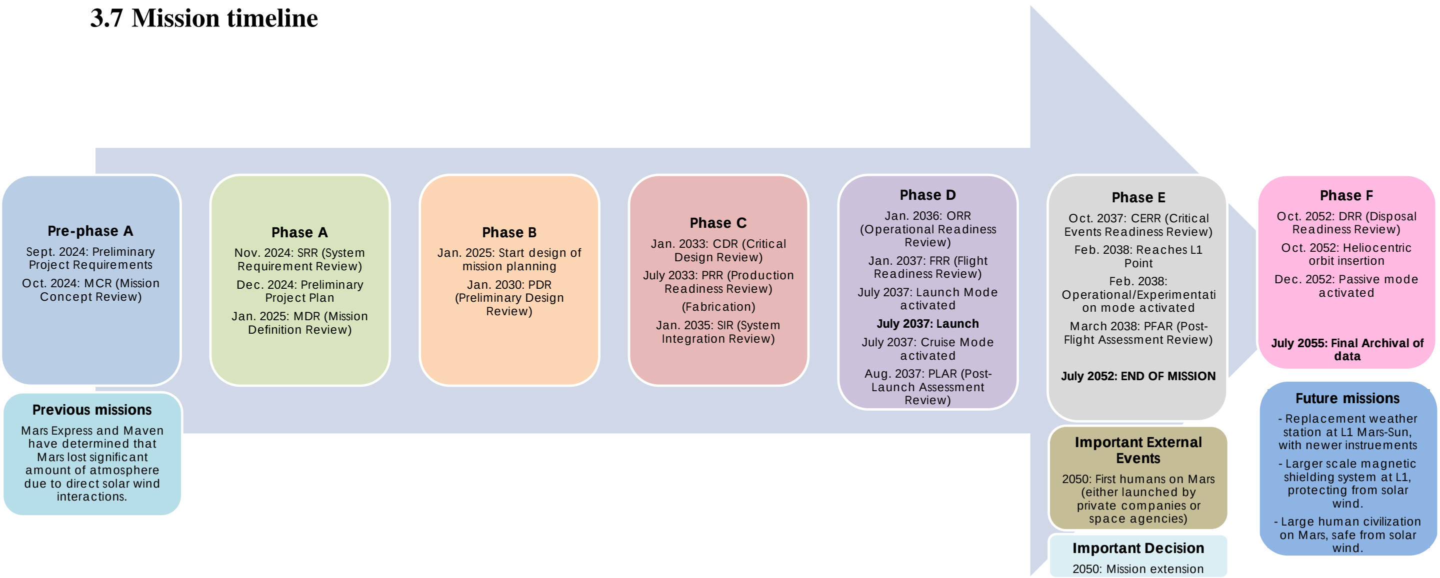

Figure 2: Mission timeline, including potential future missions. Following NASA's handbook[22]

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4. Mission architecture

4.1 Scientific payload

The scientific payload for MORPHEUS consists of instruments with proven spaceflight heritage, all at Technology Readiness Level[23] (TRL) 6, except for the experimental Magnetic Solenoid System (TRL 2). While some instrument specifications (indicated in blue in Table 1) were estimated based on photos due to incomplete available data, these overestimations ensured that all components would fit within the spacecraft's design envelope.

The main scientific questions were used to choose the different scientific instruments onboard MORPHEUS:

1. *What are the properties of the in-situ solar wind reaching Mars?*
   Instruments like the Light Ion Analyzer, Energetic Particle Detector, Magnetometer, and Standard Radiation Environment Monitor are tasked with measuring the velocity, energy, composition, and magnetic properties of solar wind particles, as well as the surrounding radiation environment.

2. *How can we predict space weather on Mars?*
   Solar observation instruments, including the Coronagraph, Heliospheric Imager, and Solar X-Ray Flux Monitor, will provide real-time data on solar activity, helping to forecast solar storms and their potential impacts on Mars.

3. *How could a magnetic shield protect Mars' atmosphere?*
   The Magnetic Solenoid System will generate an artificial magnetic field, while the Plasma Spectrometer mounted on a telescopic arm opposite to the Sun, will measure the density of charged particles after they pass through the generated magnetic field. This configuration will allow to assess the shielding effectiveness and study the interaction between solar wind particles and the artificial magnetic field.

4.1.1 Scientific Instrument Suite – TRL 6

Each instrument has previously flown on space missions and will undergo adaptations to meet MORPHEUS-specific requirements. Below are the key instruments and their primary functionalities:

- **Solar Wind Ion Composition Spectrometer (SWICS):** Determines elemental and ionic-charge composition, temperatures, and mean speeds of major solar wind ions (H through Fe) across solar wind speeds ranging from 145 km/s to 1532 km/s. The instrument measures total ion energies between 35 and 600 keV.[24]

- **Light Ion Analyser (LIA):** Measures the three-dimensional velocity distribution of solar wind ions, detecting low-energy particles in the range of 0.05 to 20 keV.[25] Some specifications have been estimated based on similar instruments.[26]

- **Energetic Particle Detector (EPD):** Analyzes spectral distribution, composition, temporal variability, and directional orientation of particles spanning 2 keV to 500 MeV per nucleon. It includes dedicated sensors for electrons (2 keV–30 MeV), helium ions (1.6–105 MeV), and protons (25 keV–107 MeV).[27,28]

- **Magnetometer (MAG):** Measures the solar magnetic field, particularly during coronal mass ejections (CMEs), supporting the mission's scientific objectives.

- **Standard Radiation Environment Monitor (SREM):** A compact detector that measures high-energy electrons (0.3–6 MeV) and protons (8–300 MeV), extending MORPHEUS's proton detection capabilities.[29]

- **Coronagraph (UV spectrum):** Employs an occulting disk to block the bright solar photosphere, enabling the observation of the faint solar corona. It is critical for studying the evolution of CMEs and understanding solar wind acceleration processes.[30]

- **Heliospheric Imager (HI):** Captures wide-field images of the solar wind in interplanetary space, operating in a 500–700 nm bandpass with a $40^\circ \times 40^\circ$ field of view.[31]

- **Solar X-ray Flux Monitor (SXFM):** Measures intensity fluctuations and spectral characteristics of solar X-rays across an energy range of 1–15 keV, with an energy resolution better than 180 eV at 5.9 keV and a field of view of $\pm 40^\circ$.[32]

- **Plasma Spectrometer (PLS):** Utilizes two Faraday-cup detectors positioned behind the spacecraft to measure changes in solar wind flux after shielding. The current measurement range is 5 eV to 1 keV, with plans to increase the upper energy limit for MORPHEUS.[33] Note that, the PLS is directly related to the MASS experiment (cf. next section).

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Table 1: Specifications of Scientific Instruments

Instrument	Mass [kg]	Data rate [bits/s]	Dimensions [m x m x m]	Power [W]
SWICS	5.58[34]	500[24]	0.500 × 0.500 × 0.500	5.00[35]
LIA (Estimated)	15.00	5000	0.400 x 0.400 x 0.700	20.00
EPD	15.72[27]	3600[28]	0.700 x 0.600 x 0.400	19.70[27]
MAG	3.53[36]	256[36]	0.200 x 0.100 x 0.100	7.20[36]
SREM	2.64[29]	100000[29]	0.095 x 0.122 x 0.217[29]	2.20[29]
Coronagraph (UV)	15.00[30]	245000[30]	0.220 x 0.197 x 0.354[30]	20.00[30]
HI	16.50[31]	20500[31]	0.627 x 0.247 x 0.400[31]	10.60[31]
SXFM	0.65[32]	2000[32]	0.234 x 0.112 x 0.133[32]	5.00[32]
	0.70[32]		0.150 x 0.127 x 0.055[32]	
MASS	(2×) 9.00	(2×) 0	(2×) Ø 0.100 × 1.000	(2×) 39.60
PLS	9.90[33]	32[33]	0.400 x 0.200 x 0.200	8.10[33]
TOTAL	**103.22**	**376888**	**0.509** m^3	**177**

4.1.2 Magnetic Solenoid System - MASS - TRL 2

The Magnetic Solenoid System (MASS) is a novel instrument developed specifically for the mission. MASS consists of two solenoids placed on each side of the spacecraft. The current in the solenoid is equal to 20 A as it is the current needed to generate a magnetic field of 15 μT on average (comparable to the one on Earth) at a distance of 2 meters around the spacecraft. Each solenoid has a diameter of 0.1 meters. The wire used for the solenoid is chosen based on the temperature balance, and depends on the length as well as the maximum number of turns available.

The Mars Express and MAVEN missions have determined that Mars has been losing a significant amount of atmosphere due to the direct solar wind interaction with the exosphere, ionosphere, and upper atmosphere, because it no longer has a magnetic field providing sufficient protection.[16]

For a long-term human presence on Mars to be established, serious thought would need to be given to terraform the planet.[37] The idea to create an artificial magnetosphere for Mars using a magnetic field generated at Mars L1, would allow to protect the planet from solar winds. While the total power required is still large, positioning the shielding magnet at L1 requires much less energy than if placed closer to the planet.[37]

As complex models are currently used to simulate a magnetic shield positioned at L1, experimental data would be welcome. Thus we place a Faraday cup on a long bow, to measure the remaining particles at a 2m distance from the body of the spacecraft.

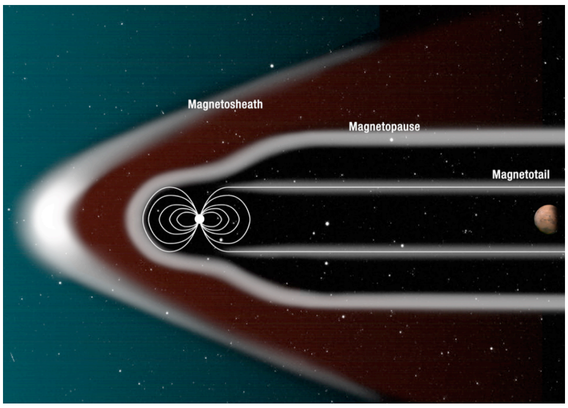

Figure 3: A Large artificial magnetosphere generated at L1, protecting Mars with its magnetotail.[16]

As mentioned previously, the wire of the solenoid has been chosen by doing a trade-off between the length of wire used and the amount of turns possible. By increasing the number of turns we increase the radial magnetic field but the diameter of the wire has to decrease for a given dimension of solenoid. On the other hand, the smaller the diameter of the wire, the higher the temperature gets and the power balance is harder to maintain. In addition, a larger wire has a higher mass, which is important to consider given the strict restrictions imposed by the launcher. Table 2 summarizes the different choices available and their corresponding values for the temperature, mass and magnetic field produced.

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Table 2: Trade-off for the solenoid wire choice.

Wire name	Number of turns	Cable length [m]	Cable mass [kg]	Magnetic field [T]	Temperature [K]
12 AWG	488	153	4.3	3.36E-05	365
6 AWG	*243*	*76*	*9*	*1.74E-05*	*223*
2 AWG	153	48	14.2	1.16E-05	157.7

The wire selected for the preliminary design of the MASS experiment is the 6 AWG as it combines both a sufficient magnetic field and an optimal power balance despite a higher mass than the 12 AWG wire.

4.1.3 Ethical considerations

Implementing a magnetic shield around Mars raises ethical considerations related to planetary protection and unintended consequences. It is crucial to evaluate potential risks to Mars' native environment, prioritize its scientific value for future study, and ensure such interventions align with mankind's vision of responsible space exploration.

4.2 Spacecraft main subsystems

The details of each subsystem are presented in their respective sections. A dash (−) is used when we do not know the information, and a 0 is used when we are sure it is zero. All important values have been estimated, and none of the dashed values are important for the final mass budget. Also, data rate only indicates the data generated by the subsystem, not the data passing through it.

- **CDH - Data storage :** During solar conjunctions, no communication can take place and the data need to be stored. With the calculated 420 kbps of data generated by the spacecraft, it requires a data storage of 1.5 Tb. This storage is made redundant. Since this system is already flying aboard existing space missions, the TRL of this subsystem is 9.

- **CDH - Architecture :** The Command and Data Handling needs to be tailored to the spacecraft. The functional block diagram of its system architecture is shown below, it is made to be redundant and only the antennas are not.

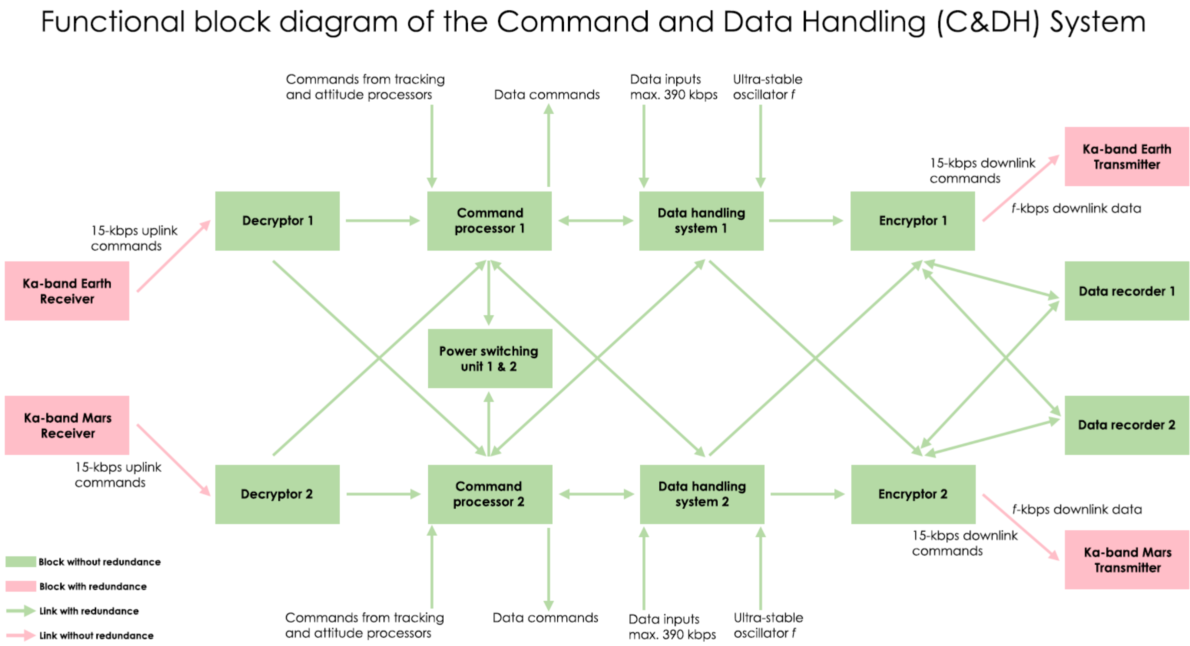

Figure 4: Command and Data Handling Architecture

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- **EPS - Solar Panels :** Solar arrays used in space have a specific power of around ∼ 100 W/kg in LEO.[38] At Mars distance, this value scales with the solar irradiance to give ∼ 40 W/kg. We want to use foldable solar panels for their compactness. Using the existing Roll Out Solar Array (ROSA)[39] that was also used on the ISS, we estimate at Mars and based on the value of their XTJ Prime Solar cells[40] that, the mass and area required. Assuming the cells work at $0.99^{15} = 86\%$ of their 30.7% BOL efficiency at the EOL (15 years), we estimate a 26% efficiency. With 10% margin from the calculated power consumption, knowing that the solar irradiance at Mars is[41] 586.2W/m^2, we need $\lceil \frac{1200\text{m}^2}{\frac{586.2\text{W}}{\text{m}^2} \cdot 0.26} \rceil = 8\text{m}^2$ of solar panels for a total mass of 31kg.

- **EPS - Batteries :** The two batteries are the same which equipped MAVEN, and each provides 2000 kWh.[42] In MORPHEUS only one battery is theoretically needed but another one is used for redundancy. Depending on the active mode, and if the solar panels can be used, the spacecraft can be battery-powered.

- **EPS - Cables:** The mass of the cables is estimated to be around 10% of the dry mass, so 60kg.

- **Propulsion & AOCS - Main propulsion :** The main propulsion system which requires a Δv of 4 km/s (separated in two burns of 2 km/s for going there and coming back), generally uses a hydrazine monopropellant, as seen on spacecraft such as MAVEN and SOHO. Hydrazine is highly reliable and offers good performance, making it ideal for precise in-orbit manoeuvres and attitude control. However, hydrazine is highly toxic, posing safety problems during handling,[43] which we consider unacceptable.

  Therefore, we plan on using a hypergolic monopropellant engine based on ammonium dinitramide (ADN), in particular the LMP-103S mixture.[44] In an ADN-based propulsion system, the propellant is typically thermally and catalytically decomposed, which provides higher performances than traditional monopropellant,[43] but also requires special reactor bed, thrust chamber and nozzle to withstand such temperature.[44] It is good to keep in mind that such parts made of non-traditional materials are potentially less sustainable to produce than regular ones.

  The propellant has a density of 1.24 g/cm^3, and will be contained in a standard tank of ∼ 2.4 m^3. Similarly to other missions,[45] the engines and thrusters will be pressure-fed, in this case with a helium tank (419 mm diameter and 1026 mm long, 119.7 liters)[46] which is enough to maintain the propellant tank at a reliable thruster operation above 100-psi.[47] Helium is chosen because of it is a non-flammable, inert gas with low molar mass, which we only require 4 kg onboard. Typically, an inert gas such as xenon would have required a 35× greater mass.

  We need 7× 440N-thrusters[48] to enable main burns to be shorter than 30 minutes:

  For a $\Delta v = 2\,\text{km/s}$ it means:

  $$F = m(t) \cdot \frac{dv}{dt} \rightarrow \frac{1}{m(t)} dt = \frac{1}{F} dv$$

  Knowing that the mass of propellant required for the first burn is ≈ 1800kg, we define $q = \frac{1800}{30 \cdot 60} = 1\text{kg/s}$ the constant exhaust mass flux during the burn. With $m(t) = m_0 - t \cdot q \approx 3600 - t \cdot \frac{1800}{30 \cdot 60}$, integrating gives:

  $$\frac{1}{q} \ln \left| \frac{m_0}{m_0 - t_f \cdot q} \right| = \frac{1}{F} \Delta v$$

  $\rightarrow F \approx 2886\,\text{N} \rightarrow$ We need $7 \times 440\,\text{N}$ main engines.

  The thrusters chosen as reference work with hydrazine but give the order of magnitude for the size and mass. We then have 6 × 27N-thrusters for translation and 8 × 1.12N-thrusters to rotate. The number of thrusters of each type is comparable to similar missions.[45] The thrusters used for station keeping are already used on existing space missions, however, the configuration required for the mission is new and therefore the overall TRL of the subsystem is 4.

- **AOCS - stationkeeping propulsion :** The stationkeeping propulsion requires 3 m/s per year over the duration of the mission. With the goal to not increase the complexity of the spacecraft, we decide to use the same ADN-based propulsion for the stationkeeping, but using 6× 27N-thrusters[49] around the spacecraft, for translation (in

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the three dimensions, acceleration and brake). Eight 1.12N-thrusters[50] are used for pointing, and rotating around 2 axis (need for 4 thrusters to provide torque around a single axis) which with rotational combinations, is enough to cover all the spacecraft orientations. Due to the instruments located on a single face of the spacecraft, the roll is not needed when the spacecraft is looking at the sun. The thrusters used for station keeping are already used on existing space missions, however, the configuration required for the mission is new and therefore the overall TRL of the subsystem is 4.

- **AOCS - Reaction wheels :** The chosen reaction wheel can store a momentum of 40Nms. It was chosen so that the spacecraft could perform a complete rotation along all of its three axes in less than 15 minutes. To estimate the time it would take to rotate the spacecraft along each of its axis, we first assumed that the spacecraft was a cylinder because theoretical formulas do not exist for a rectangle.

  By conservation of angular momentum, we have that: (s refers to the spacecraft and w to the wheel)

  $$\omega_s = -\frac{I_w}{I_s} \cdot \omega_w$$

  At constant velocity, the change of angular position over a time period t_m is:

  $$\Delta\theta_s = -\frac{I_w}{I_s} \cdot \omega_w \cdot t_m = -\frac{L_w}{I_s} \cdot t_m$$

  The angular momentum of the wheel: $L_w = I_w \cdot \omega_w$ in $Newton \cdot meter \cdot second$ can be found in the table above.

  Assuming the reaction wheels are positioned at the center of the spacecraft and act on the three rotational axes, we can extract the rotating time to make a full 2π rotation along each axis for the different reaction wheels configurations.

  $$t_m = \frac{2\pi \cdot I_s}{L_w}$$

  Thus $t_m = \dfrac{2\pi \cdot (\frac{1}{2} \cdot M_s \cdot R_s^2)}{L_w}$ along the longitudinal axis and $t_m = \dfrac{2\pi \cdot (\frac{1}{4} \cdot M_s \cdot R_s^2 + \frac{1}{12} M_s \cdot L_s^2)}{L_w}$ along the two other axis.

  With the spacecraft values: radius 0.9m and length 3.5m, we find that the W45 version of the reaction wheel allows for a complete rotation around each axis in about 12 minutes. Whenever the reaction wheel is near saturation, we offload it using the appropriate thrusters.

- **AOCS - Attitude determination :** The attitude of the spacecraft is determined using 2 rad-hard star trackers[51] positioned at 90°, providing sufficient information to obtain the absolute attitude of the spacecraft by comparing the observed stars with a known database. Those star trackers shouldn't be placed on the part that faces the Sun at L1. When star trackers face the Sun, these are blind and the use of the 2× deep-space passive sun sensors[52] are needed. Finally, a deep-space rad-hard fiber-optic gyroscope with additional accelerometers[53] is placed on the spacecraft to accurately determine attitude changes.

4.2.1 Structure - TRL 2

The structure of the spacecraft is made up of an aluminium frame on which honeycomb panels are fixed. This configuration guarantees the lightest structure possible while maintaining all the mechanical properties needed. Despite that, the materials are already being used for space applications, the structure of the frame requires to be developed especially for this mission and is therefore of TRL 2. Most of the payload instruments require a direct view of the Sun, and are thus all fitted on the Sun-facing side of the spacecraft. The instruments with this requirement are: the Coronagraph, the Heliospheric Imager, the Solar X-Ray Flux Monitor, the Magnetometer, the Energetic Particle Detector, the Light Ion Analyzer and the Solar Wind Ion Composition Spectrometer.

The propellant and the helium required for propulsion take up most of the weight on board the spacecraft and are placed as close as possible to the center of the structure to avoid mass imbalances when maneuvering. The thrusters used for station-keeping are placed on each side of the spacecraft to guarantee yaw and pitch axis motion. The antenna

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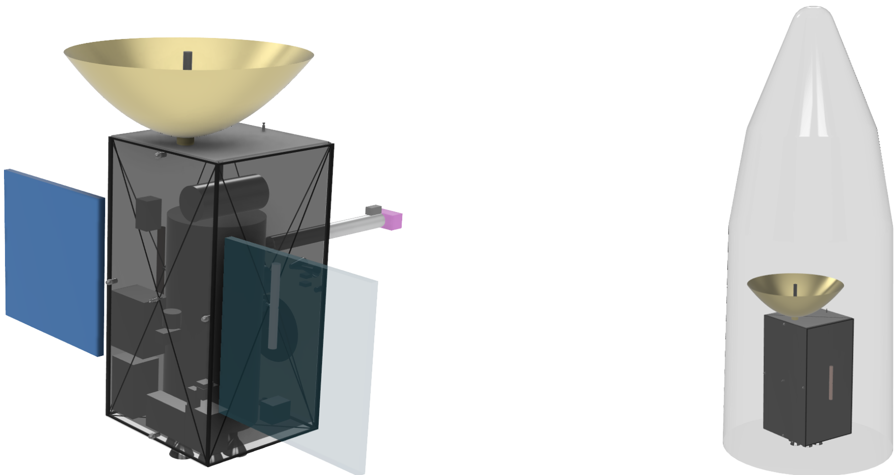
toward Earth, the biggest one, is placed on a gimbal for orientation on top of the spacecraft.

Since the spacecraft's orientation does not change during the Lissajous orbit phase of the mission (see CONOPS), the antenna for Martian communication is fixed and is placed on the opposite side to the Sun. To accommodate for the MASS Experiment, a boom is required to place the plasma spectrometer away from the spacecraft. This arm is also positioned on the side opposite to the Sun. The solenoids are placed on each side of the spacecraft, behind the solar panels. The total dimensions of the spacecraft, with the solar panel folded, are $1.8 \times 1.8 \times 3.5$ m^3. Figure 5a and Figure 5b show the preliminary configuration of the spacecraft with all the instruments, thrusters, solar panels and antennas.



(a) View of the preliminary spacecraft

(b) Spacecraft inside the fairing

Figure 5: Results from CAD on the Fusion 360 software.

4.2.2 Telecom - Entire subsystem - TRL 9

With the oversubscription of the NASA Deep Space Network (DSN),[54] it can be estimated from similar missions such as MAVEN, that contact with Earth will be allowed for about 12 hours per day (with margins and assuming Solar conjunction will be followed by days with more hours of data transmission).[55] From the data budget and the modes, we calculate that our spacecraft generates 380 (rounded up) kbps of data, plus an estimated 15 kbps of commands and telemetry (based on values from MRO mission)[17] from each antenna. We thus define our maximum data rate to be 410 kbps, which means that the antenna must be designed for 820 kbps, due to the communication window.

The frequency bands of the DSN for deep space, are the X and Ka-bands (S-band being unavailable at Madrid tracking station).[56] All communications with Earth for the up-link and down-link can fit into a single bandwidth of a parabolic antenna, as such antennas can have a bandwidth up to $30-40\%$ of their operating frequency[57] (here designed for 32 GHz). The intended connection channels with the DSN are the high-band ones, labelled H1-H34.[56] Assuming a TWTA gain of 200W (with 50% efficiency, underestimated as old ones already reached 62% efficiency),[58] and a maximum transmission distance[41] of 4.01×10^8 km between the Earth and Mars. Using a 1/2 turbo code encoded QPSK data stream, and 10^{-9} BER due to the high latency, the antenna has a diameter of 2.75m for a transmission link with 3.7 dB margin.

The size of the antenna has 820 kbps data rate at the furthest distance from Mars, but could reach up to 36 Mbps at its closest distance (54.6×10^6 km),[41] unloading the DSN. The antenna size is comparable to similar missions.[59]

The distance between L1 and Mars is 1.08×10^6 km,[60] we send the same data towards Mars as toward Earth, to use the same circuit for both antennas. Therefore, we work at the same frequency with the same error correction code, except that because of the lower latency (< 10 seconds), we accept to have 10^{-6} BER. Moreover, the atmospheric loss at mars (0.4 dB) and dust storm (1 dB) are lower than for Earth communications (2dB atmosphere, and 2 dB rain).[61] With a 20W TWTA (assumed 50% efficiency) at the transmitter, and assuming we can transmit data to a target on Mars 1/3 of the time (i.e. 8 hours per Mars day), a parabolic antenna of 0.85 m diameter in both the spacecraft and on the

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Mars asset is sufficient to allow 1230 kbps data rate with 4.1 dB margin.

All the data requires encryption (AES-256) to protect against unauthorized access to critical space vehicle functions,[62] and to protect the data recorded by MORPHEUS. Moreover, all antennas are right-hand circularly polarized, to be independent of angular orientation. The necessity for a low gain path antenna to use after launch shall be investigated but does not represent a trade-off at this point (its mass is a few tens of grams, and its size is limited).

4.2.3 Thermal - Heating system - TRL 2

Throughout the journey to Mars and the experimentation phase, the spacecraft faces different thermal constraints. We model the spacecraft as a rectangular shape of $1.8 \times 1.8 \times 3.5$, and assume that its temperature is uniform.

Using the value for the radius of the planets, the bond albedo and the temperatures of Mars and the Earth,[41] as well as the visibility factor for both planets,[63] we estimate the temperature of the spacecraft at five locations.

The five locations are: *LEO Sun shielded by Earth, LEO in front of the Sun, Transfer trajectory to Mars, Sun shielded by Mars, and L1 Mars-Sun*. The temperature at these five locations was used to determine which coating to use on the spacecraft. Knowing that the most important location is the L1 Mars-Sun as most of the time is spent at this location.

Using a tungsten vapor-deposited coating, the steady-state temperature of the spacecraft at L1 is 12.22°C. This temperature is within the functional ranges of the instruments, as listed in the requirements. The mass of the thermal subsystem estimated at 30kg, is an overstimation based on the Mars Odyssey mission[5] and it is assumed to be TRL 2 as the current design is only a preliminary analysis.

It should be noted that some instruments require independent heaters to work properly. When referenced in their description, the power consumption of these is detailed in Table 8 found in the appendix.

4.2.4 Thermal - Coating - TRL 9

Based on the thermal analysis, we use a tungsten coating (vapour deposited coating) of 10μm thickness. The deposited layer is placed on top of a sheet of aluminium that is placed on a honeycomb panel. The mass of the deposited layer is independent of the mass of the heaters.

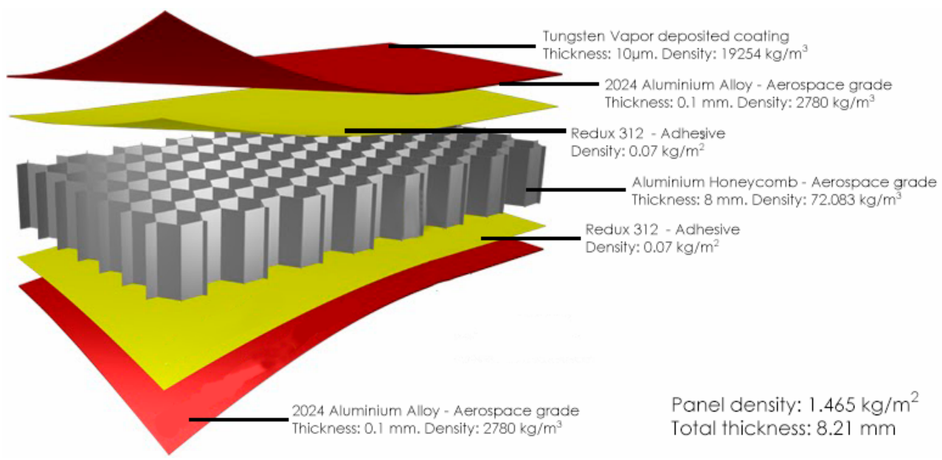

Figure 6: MORPHEUS design of the honeycomb panels.

4.3 Mission Modes

During the mission, each instrument and subsystem is either Fully active (I), Partially active (P), Standby (SB), or Off (O). These modes ensure the optimal performance and safety of the spacecraft throughout different mission phases. The table below outlines the status of each instrument and subsystem during various mission modes (cf. Figure 1).

The *safe mode* ensures the spacecraft's integrity, allowing the ground team to diagnose and resolve anomalies while preventing issues from propagating to other subsystems or instruments.

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Table 3: Mission Modes and Instrument/System Status (Color Coded)

Instruments/Systems	SWICS	LIA	EPD	MAG	SREM	Coronograph (UV)	HI	SXFM	MASS	PLS	CDH	EPS - Batteries	EPS - Solar Panels	Propulsion & AOCS	Telecom - Antennas	Thermal - Heaters
MISSION MODES																
Launch mode (1-4)	O	O	O	O	O	O	O	O	O	O	SB	SB	SB	SB	SB	SB
Partial deployment in LEO																
Cruise mode (5-10 + 12)	SB	SB	SB	SB	SB	SB	SB	SB	SB	SB	P	I	I	I	P	I
Complete deployment																
Operational mode (11)	I	I	I	I	I	I	I	I	SB	SB	I	I	I	I	I	I
Experimentation mode (11)	SB	SB	SB	P	SB	SB	SB	SB	I	I	I	I	I	I	I	I
EOL (heliocentric)																
Passive mode (13)	SB	SB	SB	SB	SB	SB	SB	SB	SB	SB	P	O	I	O	P	O
If critical issue																
Safe mode	O	O	O	O	O	O	O	O	O	O	P	O	I	O	P	I

5. System Engineering

5.1 Top level functionality

The functional tree of the mission is shown in Figure 7. The mission is divided between the scientific payload, which aims at answering the scientific questions and the main subsystems which make the mission possible.

While the scientific payload aims at improving the space weather knowledge at Mars as well as investigating the role of a shielding magnetic field, the main subsystems ensure propulsion and attitude control, provide energy to the spacecraft, ensure structural integrity and transmit data. The solution to each task is highlighted in blue.

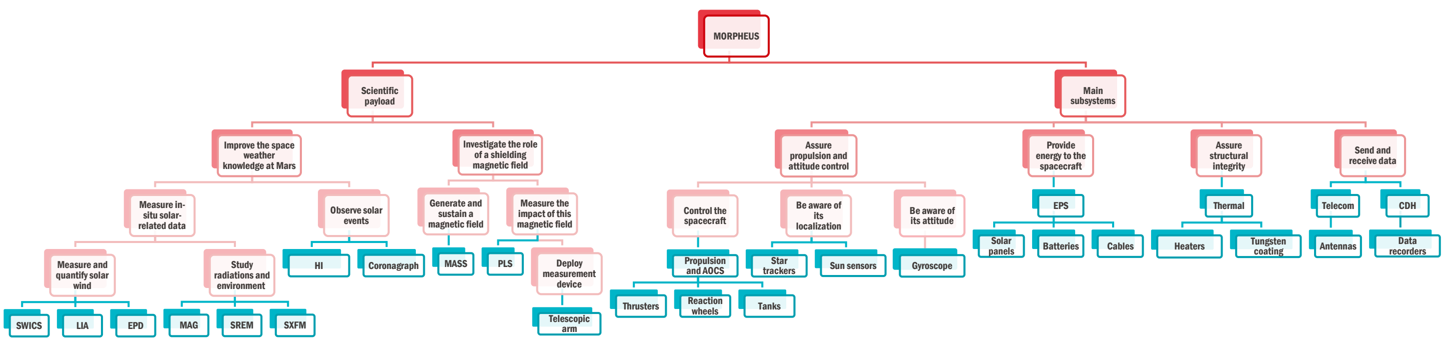

Figure 7: Functional tree for the MORPHEUS mission. A larger version of the functional tree is avalaible in the appendix.

5.2 Mission constraints and limitations

The mission is constrained by multiple elements. Since the spacecraft is going to be stationary and facing the Sun during the entirety of its Lissajous orbit, the solar panels are not required to be mobile, however, this constrains the mission during the cruise phase. In addition, always facing the sun can create damaging temperature gradients. One of the mission's success criteria is the spacecraft's communication with Earth and Mars, however, the antennas are single points of failure. In addition, solar conjunctions limit the availability of mission communications every 26 months.

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The limitations made during the study of the MORPHEUS mission are numerous and have to be kept in mind when looking at the results. For example, the thermal analysis uses simple geometry, sometimes existing systems are used to get a rough estimate of power/mass/data rate values for similar but non-identical subsystems. Moreover power estimations assume that the spacecraft delivers the full power required by each instrument at all times, which likely results in an overestimation of the power required during the mission.

5.3 Mission success criteria and measures of success

For the mission to be fully successful, the scientific questions driving the mission must be provided with data. In addition, the primary mission success criteria presented in Table 4 have to be fulfilled. Some additional measures of success can help quantify the success of the mission. For example, the yearly station-keeping required to keep the spacecraft in position should be less than 3 m/s during phases 10 to 11. A nominal data rate during these phases contributes to the mission's success. All single points of failure should be functioning and the magnetic field should be nominal during the MASS experiment. If those criteria and measures are not met, the mission could still be considered a partial success if the criteria defined under the "Secondary success criteria" in Table 4 are fulfilled.

Table 4: Primary and secondary mission success criteria

Primary success criteria	**Phase(s) of the mission**
Spacecraft is inserted in a Lissajous orbit around L1 point.	1 to 10
Each instrument gathers relevant data*	10 to 11
Spacecraft communicates with Earth	5 to 13
One run of the MASS experiment is performed and data is gathered	10 to 11
The sustainable disposal, according to the guidelines, is achived at the end of the mission.	12 to 13
Secondary success criteria	**Phase(s) of the mission**
Relevant data is gathered on the way to Mars**	5 to 10
Spacecraft is communicating with Mars	10 to 11
Communication possible after disposal	12 to 13

* The relevant data mentioned corresponds to the data type each instrument gathers. The data collected is described in subsubsection 4.1.1.
** The relevant data mentioned correspond to solar flux measurements and heavy particle detection.

The disposal referred in Table 4 is defined according to NASA Handbook[22] as after the Mission. The uniqueness of our mission is that the success of the Mission is not limited to the mission itself but extended to the end-of-life.

5.4 Preliminary risk assessment

The risks are described in Table 5. The category of the risk is determined by the likelihood of the risk happening and its consequences on the overall mission. Figure 8 shows the different risks and their likelihood of happening. The green color represents a low risk, the yellow color a moderate risk and the red color a high risk. Low risks have little or no potential in increase in cost, disruption of schedule, or degradation of performance. Actions within the scope of the planned program and normal management attention should result in the control of acceptable risk. An example of low mission risk would be a change of objective during the course of the mission. This risk can be moderated by examining the current objectives and planning their realization.

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Table 5: Risk assessment for MORPHEUS Mission

REF	Risk	Risk category	REF	Risk	Risk category
R-ATT-00	Guidance, navigation or control error	Moderate	R-O-00	Poorly defined requirements	Low
R-CDH-00	Radiation-induced glitches	Moderate	R-O-01	Unrealistic budget	Low
R-CDH-01	Communication system failure	High	R-O-02	Unrealistic planning	Moderate
R-CDH-02	Hackers take control of the spacecraft	Moderate	R-O-03	Loss of skilled staff due to mission duration	Moderate
R-POW-00	Power system failure	Moderate	R-O-04	Objectives regarding Mars changing during mission duration	Low
R-PR-00	Propulsion failure	Moderate	R-O-05	Mission ends prematurely	Moderate
R-SC-00	Failure to deploy instruments	High	R-O-06	Delay in developpment	Moderate
R-MASS-00	The magnetic field generated from the MASS instrument interfers with other instruments.	Moderate	R-O-07	Mode switch failure	Moderate
R-ST-00	Payload fairing separation number	Moderate	R-O-08	Conversion between american units and metric units	High
R-ST-01	Collision with micrometeorites	Moderate	R-O-09	Outgassing	High
R-TH-00	Thermal system failure	Moderate			

Moderate risk may cause some increase in cost, disruption of schedule, or degradation of performance. The MORPHEUS mission has a moderate risk of guidance, navigation or control error for example. These risks can be moderated by using redundant systems or having a safety mode on board the spacecraft. High risk is likely to cause significant cost increases, schedule disruption, or performance degradation. An example of a high risk to the mission is the failure to power instruments as it would lead to no scientific results to the mission. Significant additional action and high-priority management attention are required to handle those risks.[22]

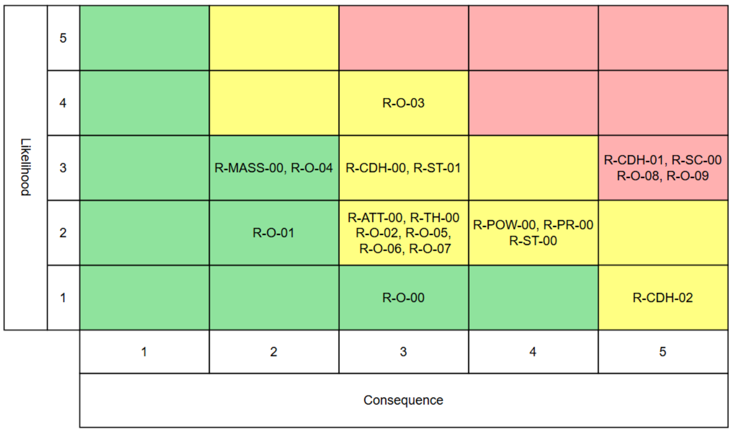

Figure 8: Risk matrix

6. Data retrieval

The main stakeholders for this mission are the space agencies and the private companies that are interested in going to Mars. If private companies provide funding to develop the mission, these would be allowed to receive live data from MORPHEUS. The data are otherwise made inaccessible to other private companies without a financial contribution.

7. Acknowledgment

We would like to express our gratitude to Emmanuelle David for his guidance and insightful feedback throughout the Spacecraft Design and Systems Engineering course at EPFL. We also thank the teaching assistants, Mathieu Udriot and Marnix Verkammen, for their continuous support and availability during the semester. Additionally, we are grateful to the guest lecturers whose contributions enriched our understanding of the many facets of spacecraft engineering.

8. Conclusion

In this article developped for the course *Spacecraft design and systems engineering*, we demonstrated the scientific importance of the MORPHEUS mission in the context of future human exploration of the planet Mars, scheduled to begin in the 2050 timeframe. We have also shown that the mission objectives are achievable using mostly existing instruments and launching on the most weight-constraining launcher. These aspects are important if we are to meet the ideal launch date of August 2037. Moreover, for the first time in an interplanetary mission, the importance of a

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sustainable end-of-life plan is entirely part of the mission's success criteria, a decision that implies a much larger wet mass at launch. Finally, MORPHEUS could be the first of a long series of spacecraft to come and could align itself with projects from space agencies or private companies on and around Mars.

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Appendix

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Table 6: Main requirement for MORPHEUS Mission Part 1

Payload System	REF	Description
CDH	CDH-00	The spacecraft shall communicate commands at a rate of 10 kbps
	CDH-01	The spacecraft shall communicate commands at a rate of 10 kbps
	CDH-02	The spacecraft shall communicate data at a rate of 400 kbps
	CDH-03	The spacecraft shall store all data during conjunction (Assumed to be max 40 days).
	CDH-04	Communications shall be encrypted with AES-256.
EPS	EPS-00	The main power bus shall have a nominal voltage of 28V
	EPS-01	The main solar panels shall be fixed, without rotation and always point towards the Sun.
	EPS-02	The main EPS subsystem shall ensure spacecraft power at all time.
	EPS-03	The solar panels shall be flexible.
	EPS-04	The batteries shall stay in temperature between -20 and +40°C.
Propulsion & AOCS	PRO-00	The tank shall contain all the necessary propellant for the entire mission
	PRO-01	The propellant shall be sustainable (less-toxic than normal)
	PRO-02	The reaction wheels shall enable the full rotation of the spacecraft in less than 20 minutes.
	PRO-03	The two star trackers shall be positioned on two different sides of the spacecraft.
	PRO-04	The sun sensors shall be positioned on two different sides of the spacecraft.
	PRO-05	The reaction wheels shall stay in temperature between -15 and +60°C
	PRO-06	The sun sensors shall stay in temperature between -40 and +90°C
	PRO-07	The star trackers shall stay in temperature between -40 and +65°C
	PRO-08	The gyroscope shall stay in temperature between -25 and +60°C
	PRO-09	The spacecraft shall have a stationkeeping of deltaV = 3 m/s.
	PRO-10	The spacecraft attitude shall be controllable without propellant
	PRO-11	The spacecraft attitude shall be measurable.
	PRO-12	The spacecraft's absolute position in space shall be measurable.
	PRO-13	The spacecraft shall be inserted in a Lissajous orbit around L1.
	PRO-14	The attitude of the spacecraft shall ensure sufficient solar power during cruise phase.
	PRO-15	The piping and valves of each engine shall be independently controllable.
Science Payload	SCI-00	The Energetic particle detector shall have sensors looking sun-ward and anti-sunward
	SCI-01	The Energetic particle detector shall be powered with 28V lines
	SCI-02	The Magnetometer shall be powered with 28V lines
	SCI-03	The Magnetometer shall be accommodated on a boom which deploys into the anti-Sun face
	SCI-04	The Standard Radiation Environment Monitor (SREM) shall be maintained between -20°C and +55°C
	SCI-05	The Standard Radiation Environment Monitor (SREM) shall be powered by 20V-50V DC
	SCI-06	The Coronograph (UV) shall face the Sun
	SCI-07	The Heliospheric Imager shall face the Sun
	SCI-08	The Heliospheric Imager shall be powered with 28V lines.
	SCI-09	The Heliospheric Imager shall detect CME in a field of view that includes the Earth.
	SCI-10	The Heliospheric Imager shall detect CME signal down to intensities of $3*10^{-15}$ B_0
	SCI-11	The Heliospheric Imager shall get data over the visible light spectrum.
	SCI-12	The X-ray flux monitor shall have an energy resolution of 180 eV
	SCI-13	The X-ray flux monitor shall measure energy between 1 and 15 keV.
	SCI-14	The X-ray flux monitor dynamic range for one second spectrum shall range from B2 to X5 class flares.
	SCI-15	The X-ray monitor shall be capable of measuring X-ray flux for flare above M5 flares temporarily.
	SCI-16	The X-ray monitor data shall be available in the form of full energy spectrum.

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Table 7: Main requirement for MORPHEUS Mission Part 2

Payload System	**REF**	**Description**
Scientific experiment	EXP-01	The magnetic field shall have an intensity of 40 000 nT at a distance of 20 meters from the spacecraft
	EXP-02	The Electromagnet shall take a current of maximum 20A.
	EXP-03	The Electromagnet shall be placed externally on the spacecraft
	EXP-04	The experiment shall run for period of 30 minutes at a time.
	EXP-05	The Faraday-cup shall not be shielded from solar particles by the spacecraft's structure.
Structure	STR-00	The (folded) spacecraft shall fit in the 15.5m rocket fairing of the Vulcan Centaur-2S
	STR-01	The spacecraft subsystems shall fit within the spacecraft structure.
Telecom	TEL-00	The spacecraft shall have two antennas
	TEL-01	One antenna shall be aimed at Earth
	TEL-02	One antenna shall be aimed at Mars
	TEL-03	The commands shall be transmittable for any attitude of the spacecraft
	TEL-04	The main Earth antenna shall have 2.75m of diameter
	TEL-05	The secondary Mars antenna shall have 1m of diameter
	TEL-06	Communications with Earth shall have a BER $< 10^{-9}$
	TEL-07	Communications with Mars shall have a BER $< 10^{-6}$
Thermal	THE-00	The Energetic particle detector shall have heaters of 16.36 W
	THE-01	The Magnetometer shall have heaters of 5.3 W.
	THE-02	The Heliospheric Imager shall have heaters of 2.5 W.
	THE-03	The Solar Wind Ion Spectrometer (SWICS) shall have heaters of 2W.
	THE-04	The spacecraft coating shall maintain the temperature within instruments requirements
	THE-05	Individual heaters shall maintain the temperature of each instrument within specifications.

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Table 8: Specifications of Subsystems Components

Subsystems	Mass [kg]	Data rate [bits/s]	Dimensions [m x m x m]	Power [W]
Command and Data Handling (CDH)	15.000[64]	0	$0.500 \times 0.500 \times 0.500$	250.0[65]
CDH - 2× 1.5Tb Data recorders	-	0	-	-
CDH - TOTAL	**15.000**	**0**	**0.125** m^3	**250**
EPS - 2× Solar panel arrays	31.000[66]	0	(2×) $2.000 \times 2.000 \times 0.050$[67]	0
EPS - 2× Batteries	(2×) 18.000[42]	0	(2×) $0.239 \times 0.301 \times 0.206$[42]	100.0[42]
EPS - Cables	60.000	0	-	0
EPS - TOTAL	**127.000**	**0**	**0.430** m^3	**100**
Propulsion - 7 × 440N engines	(7×) 1.860[48]	0	(7×) Ø 0.150×0.460[48]	0
Propulsion & AOCS - He tank (empty)	19.100[68]	0	Ø 0.419×1.026[68]	0
Propulsion & AOCS - He	4.027	0	0	0
Propulsion & AOCS - LMP-103S tank	70.000	0	Ø 1.158×2.515[69]	0
Propulsion & AOCS - piping & valves	10.000	0	-	0
AOCS - 6 × 27N engines	(6×) 0.480[49]	0	(6×) Ø 0.064×0.180[49]	0
AOCS - 8 × 1.12N engines	(8×) 0.330[50]	0	(8×) Ø 0.0343×0.150[50]	0
AOCS - 2× star trackers & electronics	(2×) 0.313[70]	10[70]	(2×) Ø 0.092×0.068[70]	(2×) 0.800[70]
	(2×) 0.450[70]		(2×) $0.100 \times 0.100 \times 0.040$[70]	(2×) 2.500[70]
AOCS - 2× sun sensors	(2×) 0.065[71]	-	(2×) $0.070 \times 0.082 \times 0.023$[71]	(2×) 0[71]
AOCS - reaction wheels & electronics	6.700[72]	10[72]	Ø 0.365×0.125[72]	29.00[72]
	4.670[72]		$0.258 \times 0.181 \times 0.143$[72]	
AOCS - gyroscope with accelerometers	4.500[73]	-	Ø 0.263×0.192[73]	13.50[73]
Propulsion & AOCS - TOTAL	**139.193**	**20**	**2.884** m^3	**49.1**
Scientific payload - TOTAL	**103.220**	**376888**	**0.509** m^3	**177**
Structure	134.760	0	$1.8 \times 1.8 \times 3.5$	0
Structure - TOTAL	**134.760**	**0**	**11.340** m^3	**0**
Telecom - Earth antenna	11.300[17]	15000	Ø 2.750	0
Telecom - Mars antenna	11.300[17]	15000	Ø 0.850	0
Telecom - TWTA	12.100[17]	0	-	440.0
Telecom - Electronics, gimbals, motors	53.300[17]	0	-	40.00
Telecom - TOTAL	**88.000**	**30000**	**-**	**480**
Thermal - Tungsten coating	7.000	0	0	0
Thermal - EPD heaters	-	-	-	16.36[27]
Thermal - MAG heaters	-	-	-	5.300[36]
Thermal - HI heaters	-	-	-	2.500[31]
Thermal - SWICS heaters	-	-	-	2.000[35]
Thermal - All heaters (estimate)	30.000[64]	-	-	-
Thermal - TOTAL	**37.000**	**-**	**-**	**26.16**
TOTAL	**644.173**	**406908**	**TBD**	**1080.26**

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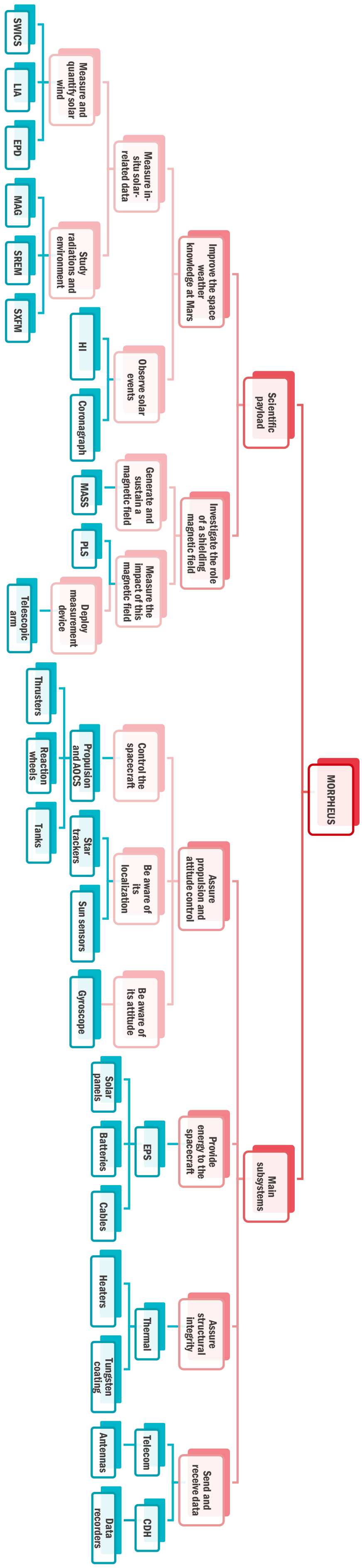

Figure 9: Functional tree

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