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# PRELIMINARY DESIGN OF A STAND-ALONE MARS CUBESAT MISSION INTEGRATING DLR IN-HOUSE TECHNOLOGIES


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## ABSTRACT

This work presents a design for a stand-alone Mars exploration mission utilizing a 12U CubeSat, showcasing the integration of DLR's in-house technologies. Focused on demonstrating the capabilities of miniaturized satellites in deep space, this innovative 4-year mission will travel independently to Mars propelled by a low-thrust propulsion system. Upon reaching Mars, the spacecraft will insert itself into a highly elliptical orbit, transitioning to a Primary Science Orbit (PSO) at 250 km altitude through aerobraking. This orbit is strategically chosen for its sun-synchronous and near-circular properties, optimizing scientific operations aimed at studying Mars' lower atmosphere and gravity field. The CubeSat features a 20.8 kg wet mass, a 6.3 km/s maneuvering capability, and can generate up to 90 W of power. It is equipped with a 2U infrared spectrometer, a 1U gravimeter, and a 12 Mpx CMOS camera for scientific data collection. Utilizing DLR's technology, including an integrated avionics stack combining communications, power, and onboard computer subsystems, the mission seeks to advance the CubeSat platform for interplanetary use, significantly reducing costs and fostering future exploration opportunities.


## 1 INTRODUCTION

The first wave of interplanetary CubeSats is already underway. In 2018, NASA's MarCO mission launched two 6U CubeSats into interplanetary space alongside the InSight Mars lander. These CubeSats enabled the success of the InSight mission by providing a crucial real-time communication link between the spacecraft and Earth during its entry, descent, and landing (EDL). In 2022, LICIACube carried out an observational analysis of the Didymos asteroid binary system after DART's impact on Dimorphos. More recently, ten 6U CubeSats built by universities and research centres were launched as secondary payloads of the Artemis 1 mission. These missions will independently explore cislunar and interplanetary space with the lowest cost up to date.

The primary motivation for this research is to propose the next significant advancement in the interplanetary CubeSat mission ecosystem. This next step shall encompass a stand-alone CubeSat mission to explore a planet in Earth's proximity. Since Mars was previously visited using CubeSats to support a larger mission, the most logical progression would involve a mission to the Red Planet with the purpose of conducting independent scientific research. To contribute to this goal, the Institute of Space Systems of the German Aerospace Center (DLR) has developed radiation-hardened small-satellite technologies, including communications, power, and onboard computer subsystems. Therefore, the mission will also serve as a technology demonstration for these technologies.


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## 2 MISSION ENGINEERING

An end-to-end systems engineering approach is used to design the mission. First of all, the following mission statement is proposed:

"A stand-alone CubeSat mission to Mars will be designed, leveraging DLR's in-house technologies and conducting an orbit analysis to demonstrate the feasibility of interplanetary small spacecraft."

From this mission statement, the main goals of the mission can be identified:

1. Demonstrate DLR's in-house technology for interplanetary small spacecraft.
2. Execute an Earth-Mars transfer trajectory.
3. Perform science in an orbit around Mars.

As can be seen, this is primarily a DLR in-house technology demonstration mission. The main technologies to be demonstrated are the Integrated Core Avionics (ICA) stack and the communications, power and onboard computer subsystems. On top of that, the spacecraft will also conduct scientific research in a Mars orbit. To do so, the spacecraft first needs to travel to Mars independently. This substantially increases the launch and transfer possibilities when comparing it to a mothership piggyback alternative.

In order to fully demonstrate the feasibility of interplanetary CubeSats, the mission shall also tackle relevant scientific objectives in Mars orbit. These knowledge gaps have been identified by the Mars Exploration Program Analysis Group at NASA. In their last report, published in 2020 [1], one of the higher-priority goals is to characterize the dynamics, thermal structure and distribution of dust, water, and carbon dioxide in Mars' lower atmosphere. On top of that, as a secondary scientific objective, the gravity field of Mars will be studied. As required by DLR, the mission shall also be able to carry out observational analysis of Mars.

The main customers for the mission are the German Aerospace Center and the Delft University of Technology. By identifying the other mission stakeholders and analysing their expectations, a first set of mission requirements can be proposed. Reference [2] can be consulted for the full list of mission requirements. Given these mission requirements, a mission timeline is proposed, which can be seen in Figure 1. This diagram represents the sequence of events that will enable the mission's success.

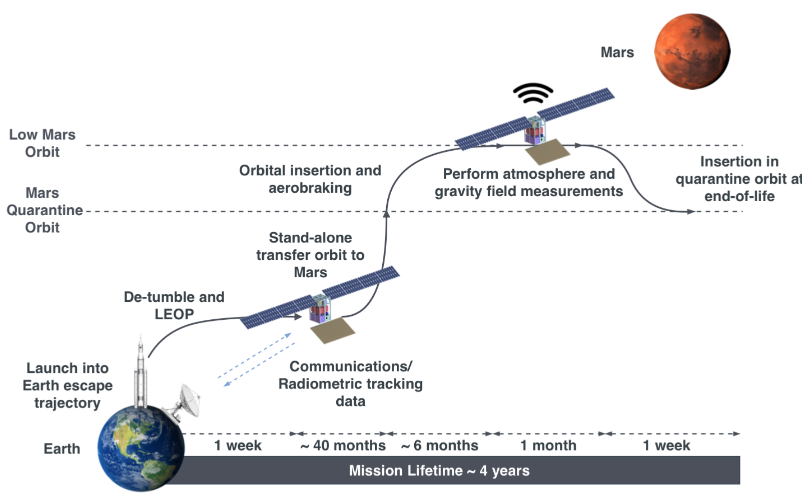



Figure 1. Mission timeline. Time periods are not to scale.

Once the system has been integrated and tested, the first step is to launch the SC into space. It is assumed that the launch vehicle will inject the SC into an Earth escape trajectory. This could be

achieved for example through piggyback on another Mars-bound mission. Once the SC is placed on a transfer orbit to Mars, it will start the Launch and Early Orbit Phase (LEOP). During this phase, the SC will deploy its solar arrays, de-tumble and perform a first contact with the ground segment. After this, the SC will start a stand-alone interplanetary cruise to Mars, which will take approximately 40 months. During this and all subsequent phases of the mission, a communication link shall be established between the SC and the ground segment to command the system and gather radiometric data for navigation purposes. These contacts will be more frequent during the mission's most critical phases, such as the orbital insertion around Mars.
Upon arrival at Mars, the SC will perform a low-thrust orbital insertion into a highly elliptical 2-sol orbit [3]. Once in orbit around the Red Planet, the SC will perform a 6-month aerobraking phase [4] to position itself around a Primary Science Orbit (PSO) at an altitude of ~250 km, where the science operations will take place. This PSO will be frozen and Sun-synchronous in order to take advantage of Mars' gravitational parameters to enable a near-circular orbit [5]. During this phase, the SC will perform measurements on the lower atmosphere and gravity field of Mars and carry out observational analysis for a nominal duration of 1 month. After this phase, another orbital change phase will be performed to raise the SC into a quarantine orbit [6]. This will allow the possibility of an extended science operations phase.
This ConOps enables the mission requirements to flow down into more technical system requirements. Reference [2] can be consulted for the full list of system requirements. By analysing the ConOps and the system requirements, the main functions and sub-functions for the system have been identified. These sub-functions can be associated with a certain physical component, which can be grouped into subsystems. This physical decomposition enables the creation of a physical diagram for the system, shown in Figure 2. In this diagram, the space, ground and launch segments are represented. The interactions among them and with the external environment are represented in yellow arrows (e.g., the loads on the space segment produced by the launch segment). The different connections between subsystems are also represented: mainly the power, data, Radio Frequency (RF) and propellant links. For the sake of clarity in presentation, the DLR ICA stack is omitted. Nonetheless, the components shown in light blue are integrated into the ICA stack.

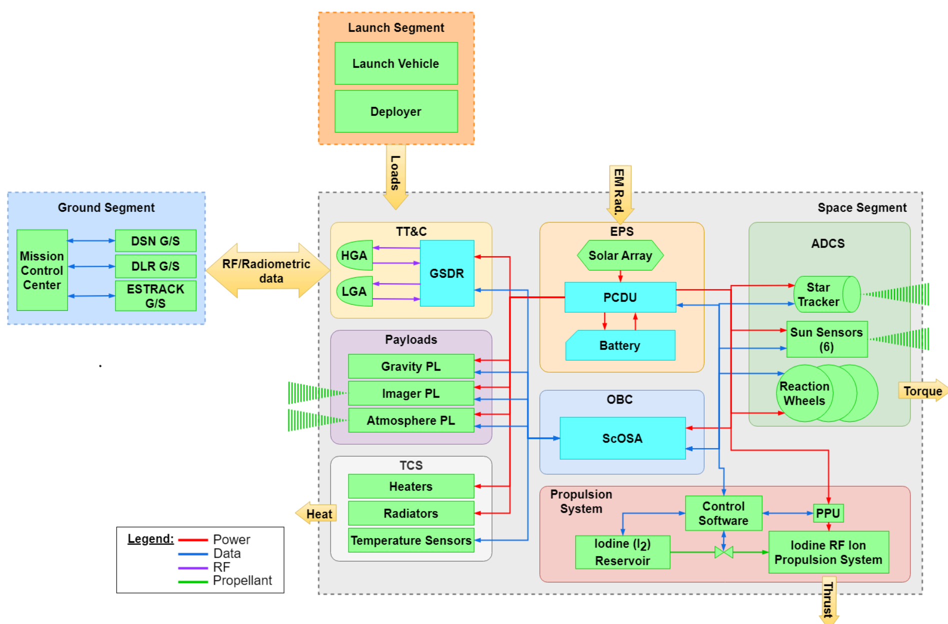



Figure 2. Mars CubeSat physical diagram. The light blue indicates components included in the ICA stack.

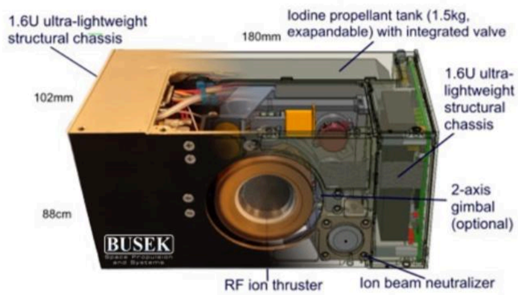
# 3 SYSTEM DESIGN

Having defined the architecture of the system, the different segments of the system can be designed. Within each segment, solutions that comply with the established architecture, ConOps, and system requirements are presented for their respective subsystems.

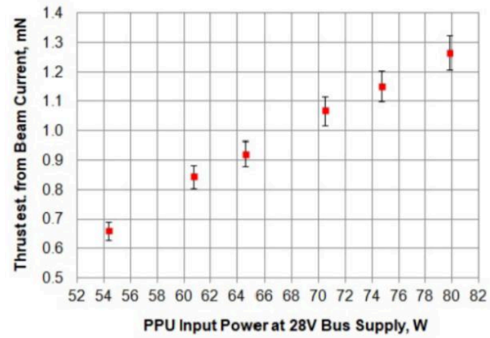
## 3.1 Space Segment

**Payloads**
The atmospheric payload of reference is the BIRCHES infrared spectrometer developed by NASA Goddard Space Flight Center [7]. This is a miniaturized version of the OVIRS infrared spectrometer used in OSIRIS-Rex. BIRCHES is a 2U, 2.5 kg, 12-25 W point spectrometer that operates in the 1 to 4-micron wavelengths. This allows the instrument to characterise and distinguish important volatiles (water, H2S, NH3, CO2, CH4, OH, organics) and mineral bands. BIRCHES is the primary payload of the Lunar IceCube launched in the Artemis 1 mission. Its main scientific goal is to study the composition and distribution of volatiles in the lunar environment. As this payload was designed for a mission outside Earth's orbit, it features its own cryocooler and radiation-tolerant hardware. Once it is operationally validated, its TRL will increase, making it an appealing option for an interplanetary CubeSat to Mars.
Another payload is needed to fulfil the objective of mapping Mars' gravity field. This has been done in the past by using tracking data from Mars orbiters or landers, such as the Mars Reconnaissance Orbiter [8]. This technique uses range and Doppler measurements to measure the changes in the spacecraft's state along its orbit caused by the variations in the planet's gravity field. Another approach could involve the use of a gravimeter. Miniaturized gravimeters have been demonstrated on CubeSat missions, such as the 1U gravimeter onboard the Star of Aoxiang, developed by Northwestern Polytechnical University [9]. Thus, a potential miniaturized 1U gravimeter developed by DLR will be used as the reference gravity payload for the mission. The readings of the test mass state will be used in combination with the spacecraft's radiometric data to study Mars' gravity field.
A camera will be used to take images of Mars. These images can have a variety of scientific applications, such as surface feature recognition, dust storm identification, etc. On top of that, images of celestial bodies can be used as input into the navigation algorithms. The DLR Institute of Space Systems has recommended using the 3D Plus 12 Megapixel CMOS Space Camera. This camera is to be used in the Mars Moons eXploration (MMX) rover, designed and integrated by DLR in collaboration with the Centre National d'Études Spatiales (CNES). This will greatly simplify the integration process and reduce uncertainties in the design process. This camera provides sufficient performance in a reduced volume of 40 x 40 x 39 $mm^3$, a mass of 120 g and a power consumption of 5 W. It is also highly radiation tolerant, with a TID of up to 40 krads.

**Propulsion**
For this to be a completely stand-alone mission, the spacecraft needs to perform an independent Earth-Mars transfer, achieve orbital insertion, and position itself into the PSO. The selected option is the Busek BIT-3, due to the utilization of the system in the Lunar IceCube and LunaH-Map CubeSats, both of which were launched as part of the Artemis 1 mission [10]. In principle, this propulsion system will allow CubeSats to carry out a low-thrust trajectory to lunar orbit. The BIT-3 (Figure 3a) is an RF ion thruster that features a two-axis thruster gimbal capable of ±10º slew. This characteristic allows for reaction wheel desaturation. The system can be throttled depending on the input power as can be seen in Figure 3b. At 80 W of input power, the system can operate at 1.25 mN of thrust and 2150 s of specific impulse.




(a) BIT-3 propulsion system and its main components.




(b) Real performance of iodine BIT-3 flight system.

Figure 3. BIT-3 RF Ion Thruster. [11]

**Attitude Determination and Control**

The most straightforward solution for ADCS is an integrated COTS system that includes the selected combination of sensors and actuators. With a focus on heritage, the best commercial option is the XACT family of integrated ADCS systems [12] developed by Blue Canyon Technologies. Their XACT-15 was used in relevant interplanetary CubeSat missions such as MarCO or Lunar IceCube. However, this model would not perform sufficiently when used in a 12U configuration. Therefore, the higher-tier XACT-50 system has been chosen, which provides a 1-sigma pointing accuracy of 0.007 deg. This system provides up to 50 mNms of momentum storage and a maximum torque of 0.006 Nm, which complies with the calculated disturbance environment.

**Guidance, Navigation and Control**

The standard approach to estimating the position of interplanetary spacecraft is using radiometric tracking data. This includes range, Doppler, and Delta-DOR measurements. While the range and Doppler data are suitable for line-of-sight coordinates, they have a weakness in determining the spacecraft declination component. In the past decades, Earth's deep space ground stations have been developing capabilities for long baseline interferometry measurements. This Delta-DOR technique provides data close to one-nanoradian accuracy with a reliability of 98% [13]. This would translate to an accuracy of around 300 m at Mars distance. To enable these measurements, the spacecraft needs to integrate a phase-coherent radio architecture that can input certain special navigational tones in the spacecraft carrier.

**Avionics Stack**

The DLR ICA developed in-house is considered for the avionics stack. A model of the ICA can be seen in Figure 4. It provides an interface to the avionic domains of onboard data handling, power, communication and software. The ICA was designed using a backplane-based configuration to be mechanically compatible with CPCI Serial Space but tailored to fit into a 3U CubeSat. Thus, the main advantage of ICA is its scalability for different applications and spacecraft classes, including CubeSats. It also implements a wireless intra-spacecraft communication system. This allows a simplification of harness design and routing, which can be used to reduce cables and the mass of the spacecraft.

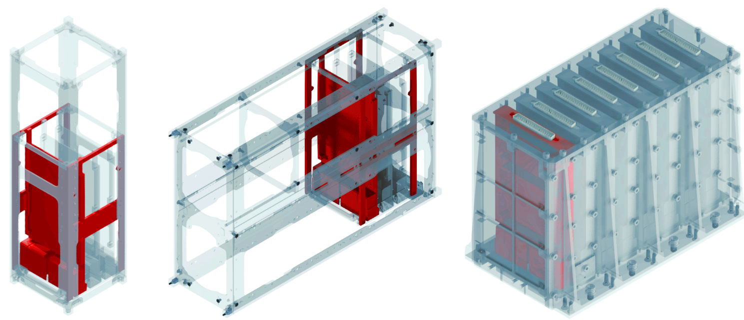


Figure 4. Application spectrum for ICA modules (in red) showing compatibility with 3U/6U CubeSat structures and integration with a stand-alone box for larger systems [14].

**On-Board Computer**

The DLR in-house developed Scalable On-board Computing for Space Avionics (ScOSA) is used for the spacecraft's OBC. The ScOSA consists of several computing nodes that are interconnected via a switched SpaceWire network. These computing nodes are reliable space-qualified or high-performance COTS components. The nodes are operated in a worker-monitor redundancy concept, where the functionality of the "worker" is supervised by the "monitor". On the one hand, reliable computing nodes (RCN) are space-qualified and radiation-hardened, which makes them suitable for interplanetary space missions. On the other hand, high-performance nodes (HPN) are COTS-based processing modules for application acceleration.

For this mission, a simple application of a single RCN and HPN is implemented. The RCN is the LEON3FT fault and radiation tolerant processing module used in the MASCOT mission [15]. Having been previously used by DLR, the implementation process of this board into the system will be more straightforward. The HPN is a COTS-based processing module with a higher performance. Given the higher performance of the HPN, the spacecraft capabilities can be extended to include higher autonomy or more advanced data processing.

The ScOSA implements DLR's Open modUlar sofTware PlatfOrm for SpacecrafT (OUTPOST) [16]. This flight software is independent of the operating system and the hardware it runs on so that it can be used in different stages of the mission design. OUTPOST includes several capabilities, including modules with Packet Utilization Standard (PUS) services, internal and external communication protocols, sparse logging, ground parsing and other mission control software.

**Telemetry, Tracking and Control**

The proposed transceiver is the Generic Software Defined Radio (GSDR) developed by DLR. The GSDR allows the operation of multi-band RF applications in the harsh environment of space, being designed with a notable focus on its radiation hardening [17]. This system is built around the concept of Software Defined Radio (SDR). The basic principle of SDR systems is to perform most parts of the signal processing through software, which has led to systems with much simpler and less heavy and costly hardware. This also means its functionalities are easily updated and upgraded through simple re-programming while using the same hardware. All these features make it an ideal option for a CubeSat spacecraft.

Given the requirements of the mission, at least two sets of antennas are needed for the mission. Regarding the LGA, the most viable option is a patch antenna. These antennas consist of a radiating patch that can be placed on the surface of a CubeSat. Therefore, they offer reduced dimensions and mass compared to other types of antennas. EnduroSat's X-Band 4x4 Patch Antenna is chosen as the reference LGA, as it provides better capabilities than its counterparts at a lower mass. Since the spacecraft requires full coverage using LGAs, two patch antennas are needed: one on the front and one on the back.

A reflectarray antenna, such as the one used for MarCO, presents itself as the most viable option for the HGA. Its reflecting surfaces rely on a simple mechanical deployment with spring-loaded hinges so that it can be folded against the spacecraft body at launch. This leads to an antenna that only consumes ~2% of the usable spacecraft volume (for a 12U CubeSat) with a mass of less than 1 kg [18]. The European Space Agency has been working on developing a reflectarray antenna to be used in its CubeSat missions through the GSTP Develop programme [19]. This design, led by TICRA, shall achieve a transmit gain of 28.5 dBi and a receive gain of 24 dBi. A preliminary link budget analysis shows that using the X-band and a bit rate of 2 kbps, the link can be closed with a 3 dB margin.

**Electrical Power System**

The solar arrays have been sized using the most demanding power mode, the Maneuver Mode, which has a power consumption in the range of 70 to 89 W. These solar arrays will be designed using Azur Space 32% Triple Junction 3G30C Solar Cells, as used on DLR's PLUTO mission [20]. Every individual solar cell, measuring 7 x 4 cm, has the capability to generate a maximum power output of 0.41 W at the End Of Life (EOL) when situated at Mars' distance from the Sun. Therefore, to meet the power demand of 89 W necessary for the full-range Maneuver Mode, a total of 218 solar cells are needed.With the dimensions of the 12U CubeSat configuration in mind, it is possible to distribute 24 solar cells across panels that match the size of the spacecraft's rectangular side. A linear distribution of these panels can be used to design the solar arrays. For this study, two 4-panel solar arrays will be assumed, which can allocate 192 solar cells. These solar arrays shall be equipped with Sun-orientation mechanisms to increase their efficiency. This configuration would ensure a net power generation of 79 W. The remaining 10 W can be gathered using body-mounted solar cells.

The PCDU used for DLR's PLUTO CubeSat will be used for the mission. This PCDU is designed to handle the 100 W generated by its foldable solar panel [20]. It implements two array power regulators (APRs) that handle the tracking of the maximum power point and convert the input voltage down to battery voltage. These APRs are based on GaN transistors in order to minimise the conversion losses in the PCDU. This system will need to be redesigned in order to account for the power generated by the solar arrays at BOL and Earth's distance from the Sun (i.e., > 200 W). The energy storage needed for the batteries can be calculated using the worst-case scenario for the Mars eclipse, giving the required battery capacity of approximately 60 Wh. Notably, the PLUTO PCDU integrates a 100 Wh battery comprised of commercial 18650 Li-Ion Cells. These cells are arranged in a 4s2p configuration, consisting of four cells connected in series to increase voltage and two sets of these series-connected cells connected in parallel to increase capacity. Although the battery's storage capacity exceeds the requirements, maintaining its current configuration is preferred to provide a safety margin and simplify the design process.

**Thermal Control System**

A preliminary thermal analysis for the mission's hot case shows that 0.16 $m^2$ of radiator area is needed. This area could be achieved with radiators on three of the rectangular surfaces of the CubeSat, with a margin of 0.02 $m^2$. These radiators need to have a low solar absorptivity and a high IR emissivity. The most efficient way to implement these radiators in terms of mass is using radiator films. For example, the Sheldahl Aluminum Coated FEP film can be used [21]. These FEP films are coated in aluminium on one side so that they can be used as second surface mirrors. For the 5-micron thickness option, this film offers an absorptivity of less than 0.15 and an emissivity higher than 0.85.

To comply with the mission's cold case, heaters need to be placed on the most thermally sensitive components. A suitable option could be the Minco Satellite All-Polyimide Thermofoil Heater [22]. These thin and lightweight heaters are constructed from entirely low-outgassing materials, incorporate pressure- sensitive mounting adhesive, and are fabricated in compliance with NASA GSFC S-311-P-841. By adding heaters that output a total of 35 W to the heat generated by the spacecraft, its temperature during Mars eclipse can be brought to -22 ºC. This temperature would ensure the survivability of all the spacecraft components.

**Structure**

The PLUTO mission being developed at the Institute of Space Systems of the DLR is planned to use a 6U structure from EnduroSat. The fact that a communication channel has already been opened between the DLR and EnduroSat provides yet another benefit from this option. Therefore, the EnduroSat 12U XL CubeSat Structure is chosen as the primary structure for the mission [23]. This structure is manufactured using Aluminium 6082, has a mass of 2 kg and its dimensions are 226.3 x 226.3 x 366 mm.

**Configuration**

The distribution of the various subsystems in the chosen configuration is mainly driven by the ConOps. Figure 5 shows the space segment interior as seen from the -X, -Y and +Z sides. The subsystem configuration will be explained from ‘top’ (i.e., side +Y) to ‘bottom’ (i.e., side -Y). First of all, both ICA stacks are located at the top, which include the GSDR, the battery, the PCDU, and the ScOSA OBC. The BIT-3 Propulsion System is located at the centre of the spacecraft, with its thruster pointing towards the -Z direction. The payloads are located in the lower region of the spacecraft. The atmospheric payload is positioned adjacent to the +X side, orientated in such a way that its sensor faces the -Y side. Consequently, while conducting science operations, the nadir will point towards the -Y direction. Both the imager and the gravimeter payloads are adjacent to the spectrometer. The sensor of the imager payload is aligned with the forward-moving direction so that it can easily be used for navigation purposes. Finally, the XACT-50 ADCS is situated between the imager payload and side -Y, with its star tracker pointing towards the +Z face. This enables the ADCS to have a direct line-of-sight to outer space during science operations. This also allows for the reflectarray antenna to block incoming radiation from Mars, which could interfere with the star tracker.

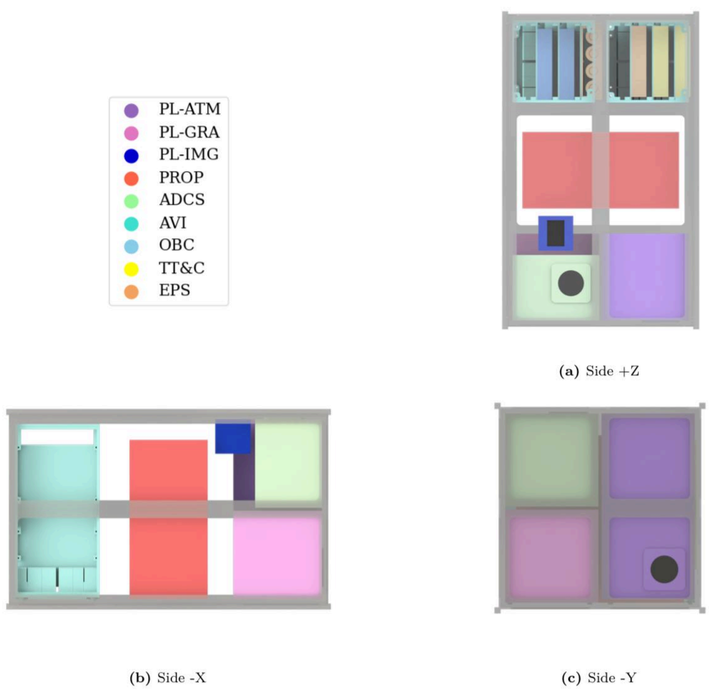



Figure 5. Subsystem configuration inside the space segment as seen from sides -X, -Y and +Z.

Finally, the complete space segment can be seen in Figure 6. This model features the spacecraft's linear solar arrays, as well as its reflectarray antenna. The total mass of the spacecraft is 20.8 kg, which complies with the mass requirement outlined by the CubeSat Design Specification (CDS) of 24 kg.

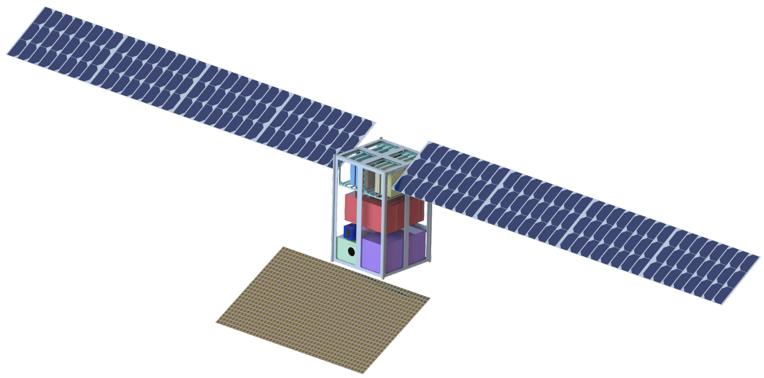


Figure 6. Mars CubeSat Model

### 3.2 Ground Segment

Several ground station networks on Earth offer the necessary capabilities to operate interplanetary spacecraft on the X-band. NASA's Deep Space Network, ESA's European Space Tracking (ESTRACK) network, and DLR's ground stations stand out as the most appealing choices. All three options offer different advantages and disadvantages. On the one hand, the DSN and ESTRACK provide high availability and communication capabilities; however, they come with a substantial cost. On the other hand, the DLR ground station can only be used to receive telemetry signals and gather radiometric data. Nevertheless, the operational costs would be considerably lower, as it would be managed by one of the main customers. Therefore, a combination of the three ground station networks shall be used in order to balance the communication performance and its underlying operational costs. During LEOP, interplanetary transfer and Mars Orbital Insertion (MOI) phases, the DSN and ESTRACK networks will be used due to their availability and transmitting capabilities. The DLR ground station will also be used during the interplanetary transfer to perform health checks and assess the need for transmitting contacts. Separately, during science operations, the DLR network will be used to downlink the scientific data. During this phase, the DSN and ESTRACK ground stations will be used when command uplink is required. Therefore, the ground station shall acquire the pertinent ITU radio frequency licences to operate on the three different networks.

### 3.3 Launch Segment

Three main options are considered in order to launch this mission. First of all, the last set of interplanetary CubeSats were launched as secondary payloads of the Artemis 1 mission. NASA's CubeSat Launch Initiative also sought proposals in 2019 to fly 6- and 12U CubeSat missions as secondary payloads of Artemis 2. However, all mission proposals were dropped in October 2021. Four more Artemis missions are planned in the next decade, which could open similar calls for proposals. While these calls have been limited to U.S. institutions and companies, the planned delivery of the I- HAB developed by ESA and JAXA on the Artemis 4 mission could be leveraged to include European institutions in the call. The main downside of this alternative is that an Earth-Mars transfer may not be feasible on these launches.

Next, the spacecraft could piggyback on another Mars-bound mission. The launchers used on missions headed for Mars usually provide launch energies up to C3 = 16 $km^2/s^2$. For example, NASA's Perseverance mission was launched using an Atlas V-541 with an excess energy of 15 $km^2/s^2$ [24]. This excess energy could contribute to the low-thrust Earth-Mars transfer by reducing the necessary Delta-V and increasing the feasibility of the mission. However, including a piggyback secondary mission could increase risks on the main mission. This could be an issue when negotiating with the project management team for these missions.

Finally, internal studies at DLR show a peculiar method for upper-stage disposal that could be potentially used to launch the mission. It has been observed that certain high-orbit launchers reach GEO with substantial excess Delta-V. In order to dispose of the upper stage, a final burn is performed in order to thrust it into deep space. The aforementioned studies show that this is a common practice with launchers such as the Atlas V, the Delta II and the Falcon 9. These launches happen regularly, which would be an advantage in terms of departure date flexibility. Thus, the CubeSat could be piggybacked on a launcher to GEO and launched into interplanetary space with a certain excess velocity during the disposal manoeuvre.

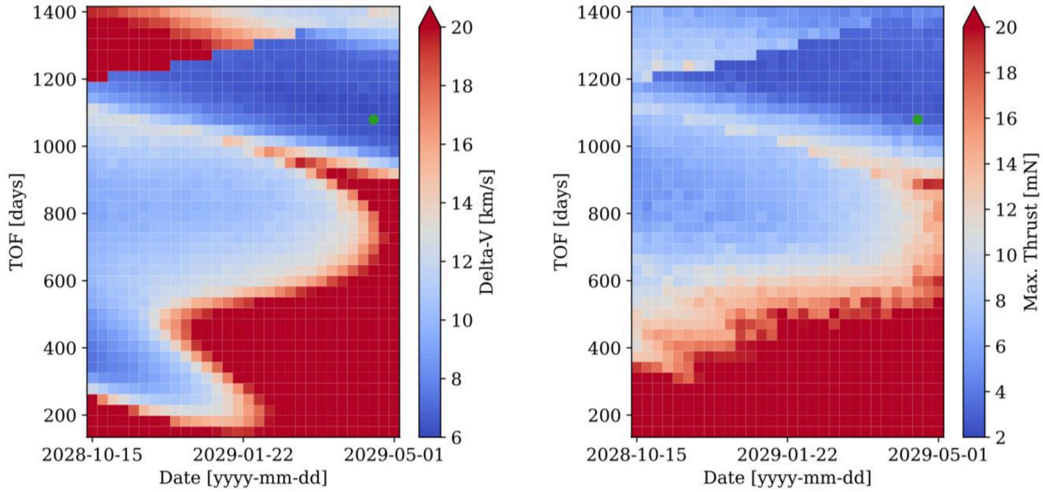
# 4 MISSION ANALYSIS

## 4.1 Earth-Mars Transfer

Here, the Mission Analysis is performed in order to study the feasibility of the designed system in the context of its trajectory design from the Earth to Mars. First, the launch windows were identified by assuming a piggyback on another Mars-bound launch. These missions usually perform ballistic transfer trajectories to Mars using launchers that provide excess energies of C3~15 $km^2/s^2$. Lambert's problem can be used to locate the launch windows that correspond to such excess energy. Therefore, the two departure date search spaces are: for the 2028/2029 launch window, from 2028-10-15 to 2029-05-01; and for the 2030/2031 launch window, from 2030-11-01 to 2031-06-15.

The hodographic-shaping method [25] is the main tool used to calculate the low-thrust Earth-Mars transfer trajectory for the mission. This method designs orbital trajectories by shaping the velocity components along the transfer. To do so, the velocity behaviour of the low-thrust trajectory is modelled using the shape of its velocity hodograph. This velocity profile is then integrated to find the change in position and the resulting trajectory. The proposed formulation for the hodographic technique offers nine different variables to characterise each trajectory. These include the departure date, the time of flight, the number of revolutions around the Sun (N), and six free coefficients that arise from the use of additional velocity base functions.

The Mission Analysis is performed using the Tudat Python library. This library supports astrodynamics and space research by implementing a wide range of functionalities, including the aforementioned hodographic-shaping method. First, the optimization procedure is performed on the complete design space allocated for the 2028/2029 launch window. A grid search has been implemented for these departure dates and a wide range of TOFs (and their inherent number of revolutions around the Sun). For each combination of departure date and TOF, an optimization procedure has been performed to identify the optimal set of free parameters that yield the most favourable trajectory in terms of Delta-V. Figure 7 shows the combined results of the optimization procedure for number of revolutions around the Sun = 0-2. On the one hand, Figure 7a shows the optimum values of Delta-V. As explained previously, three local minimum areas can be recognized, each originating from distinct values of N. The area that shows the most promising results is that corresponding to N = 2, which is consistent with the results presented by Gondelach and Noomen in a similar study [25]. This area shows great potential in Delta-V, with values around 6-7 km/s throughout the departure date range. Additionally, peak thrust values in this region decrease to 2 mN, a notable improvement over the results mentioned above for N = 0.




**(a)** Optimum Delta-V. **(b)** Maximum thrust.

Figure 7. Optimum Delta-V and corresponding maximum thrust on a 30 x 40 grid for the first launch window with N = 0-2. The green dot shows the most optimal trajectory in terms of Delta-V.

To assess the feasibility of these trajectories across the launch window, the best trajectory in the local refinement region for each of the analyzed departure dates is numerically propagated. The Delta-V and maximum thrust for each of these optimum propagations are shown in Figure 8. This figure clearly shows a downward trend in Delta-V as the departure is pushed to later dates. As this trend seems to continue into future departure dates, a lower Delta-V could be achieved if the launch window was widened using the other launch methods discussed previously.

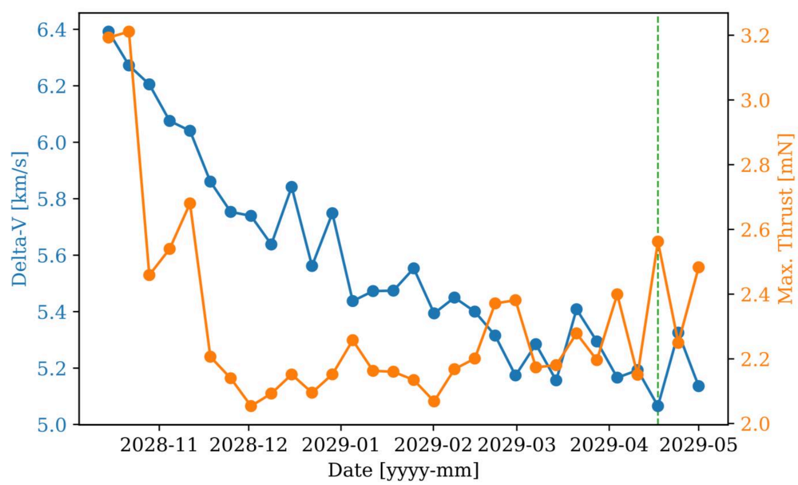



Figure 8. Delta-V and maximum thrust for the propagated optimum trajectory for each of the analyzed departure dates in the first launch window. The green dashed line shows the epoch of the best solution in Figure 7.

Regarding the behaviour of the maximum thrust, Figure 8 shows a significant drop to 2 mN after the 15th of November. For the following departure dates, a slight upward trend is observed, reaching ~2.5 mN at the end of the launch window. Therefore, there is a trade-off to be made: an earlier departure date requires a lower maximum thrust, whereas a later departure date is preferable in terms of Delta-V. In the context of this mission, the most critical requirement is for the maximum

thrust, making an earlier departure date preferable. Still, the lowest maximum thrust in this analysis is 2 mN, which is an increase of 60% over the requirement of 1.25 mN. Therefore, a strategy is necessary to further reduce this number. Further analysis of these trajectories shows that the average thrust of all the optimum propagations is below 1.25 mN. This means that the thrust of these trajectories could in principle be distributed over the length of the trajectory to mitigate these peaks.
The results of this analysis on the 2030/2031 launch window show similar results. Therefore, the Delta-V requirement for the Earth-Mars transfer shall be 6 km/s. This will allow feasible trajectories for 80% of the departure dates studied with TOFs in the range of 1000 to 1400 days. As explained, this number of feasible departure dates could be increased by extending the launch windows using other launch strategies. By adding the Delta-V required to place the spacecraft in a graveyard orbit at end-of-life to the value stated above, the total Delta-V for the system is 6.3 km/s. This Delta-V will result in a total propellant mass of 5 kg. Therefore, the final spacecraft launch mass is 20.77 kg, which complies with the upper limit for a 12U CubeSat of 24 kg established by the CDS [18]. Expanding the propellant tank will result in new dimensions for the propulsion system, measuring 88 x 180 x 180 mm. On top of that, the Earth-Mars transfer will have a duration of ~40 months. This will result in a mission lifetime of ~4 years.
All in all, the trajectory shown in Figure 9 stands out as the optimal choice in terms of Delta-V. This trajectory departs on 2029-04-17, has a TOF of 1079.49 days, and executes two revolutions around the Sun before reaching Mars. The Delta-V and maximum thrust of this transfer are 5.07 km/s and 2.56 mN. The results presented here have been verified against other low-thrust trajectory calculation methods.

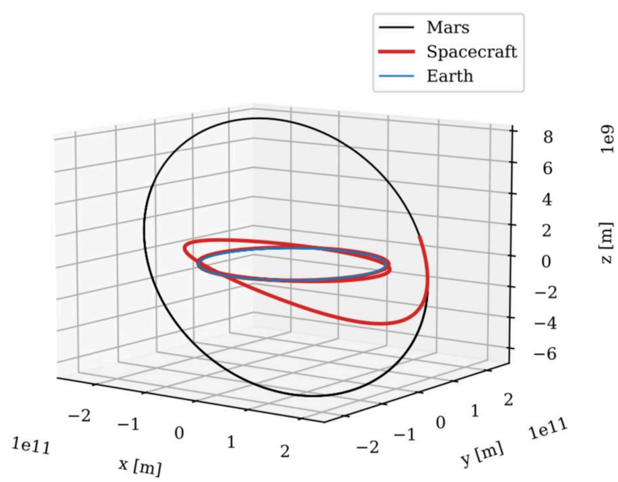



Figure 9. Trajectory of the best solution for a low-thrust Earth-Mars transfer for the mission.

### 4.2 Mars Orbital Insertion

Through the shape-based low-thrust transfer presented above, the spacecraft reaches Mars with zero hyperbolic excess velocity. Hence, executing an orbital insertion around Mars requires significantly lower Delta-V compared to a ballistic transfer trajectory. The insertion orbit is assumed to be a highly elliptical 2-sol orbit with a peri- and apoapsis of 300 and 57826 km, respectively. For the trajectory shown in Figure 9, an insertion Delta-V of 43 m/s is required. Techniques like ballistic capture could be used, potentially reducing the insertion Delta- V to zero. This is a phenomenon through which the spacecraft is captured into a temporary stable orbit about Mars, only by virtue of the natural attractions of the target body and the Sun [26]. Figure 10 shows a schematic of a ballistic capture around Mars. The spacecraft enters a highly irregular orbit about Mars after ballistic capture, and some energy needs to be dissipated to stabilise it. This could be done using a low-thrust propulsion system or through some other method for mechanical energy dissipation, such as aerobraking.

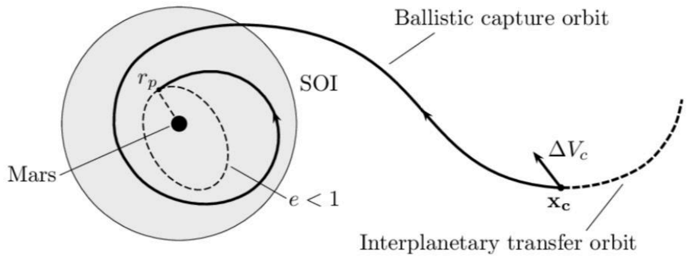



Figure 10. Schematic of a ballistic capture around Mars [26].

### 4.3 Mars Primary Science Orbit

As explained previously, the main scientific objectives of the mission around Mars' orbit are performing remote sensing on its atmosphere, conducting measurements of its gravity field and carrying out observational analysis of the Red Planet. This would be especially interesting in the lower regions of the atmosphere, where the characterization of the dynamics, thermal structure and distributions of dust, water and carbon dioxide pose a substantial knowledge gap [1]. An appropriate PSO around Mars is proposed that could enable these science operations.

One relevant case study is that of the Mars Reconnaissance Orbiter (MRO). The science orbit for this mission was designed to provide global access to Mars and satisfy its science and mission objectives while considering effects such as atmospheric drag [27]. The orbital parameters of this orbit, produced a Sun-synchronous ascending node at 3:00 pm local mean solar time; a periapsis altitude around 250 km; an apoapsis altitude near 320 km; a near-polar inclination and a combination of eccentricity and argument of periapsis that resulted in a frozen orbit. Since the MRO science orbit could enable the scientific objectives for our mission, it will be taken as the reference PSO in this preliminary study. In successive iterations of the mission design, a thorough analysis shall be carried out concerning the specific Mars science orbit required to achieve the proposed objectives.

## 5 CONCLUSIONS

A systems engineering framework has been used to design a stand-alone Mars exploration CubeSat mission that integrates DLR's in-house technologies. First, the stakeholders and their expectations have been analyzed. This has led to the identification of the main goals for the mission: demonstrating DLR's in-house technologies, executing an Earth-Mars transfer and performing science in a Mars orbit. The mission requirements have been formulated, which are the foundation blocks for the ConOps. The ConOps for the mission presents several phases. First, the spacecraft will be launched into a 40-month low-thrust transfer trajectory to Mars. Upon arrival at Mars, the spacecraft will execute a ballistic orbital insertion into a highly elliptical orbit. Once in orbit around the Red Planet, the spacecraft will perform aerobraking to position itself around a PSO at an altitude of 250 km, where the science operations will take place. During this phase, the spacecraft will tackle two of the most relevant scientific objectives around Mars: gathering data on its lower atmosphere and gravity field. Finally, the spacecraft will be placed into a quarantine orbit for disposal. This ConOps has been used to translate mission requirements into a set of technical system requirements. In turn, this has enabled the definition of the functional and physical architecture of the system. The system's main functions have been identified and linked to physical subsystems and components through a physical system diagram.

This has resulted in the design for the space, ground and launch segments. The space segment consists of a 12U CubeSat. The spacecraft has a wet mass of 20.8 kg and can generate up to 90 W of power at Mars using deployable solar panels. Its main payloads are a 2U infrared spectrometer, a 1U gravimeter and a 12 Mpx CMOS camera. The DLR ICA stack will be used, which combines multiple avionics systems (i.e., onboard computer, communications, and power) into a single unit. The DLR ScOSA onboard computer will be used, which integrates two computing nodes to ensure robustness. The reliable computing node comprises the OBC used on the MASCOT lander developed by DLR. The high-performance computing node consists of a COTS-based processing module for application acceleration. The spacecraft uses the radiofrequency X-band in order to communicate with its ground segment. The in-house developed GSDR is used as a transceiver. The system also features a main reflectarray antenna as used by MarCO and secondary patch antennas. Electrical power is generated using two linear solar arrays with 32% efficiency triple junction solar cells developed by Azur Space. The power control and distribution unit, as well as the batteries, will be adapted from those used by the DLR PLUTO mission. For other subsystems, COTS components have been chosen that satisfy the needs of the mission. The propulsion system is a Busek BIT-3 gridded-ion engine as used in the Lunar IceCube and LunaH-Map missions. This system features an expanded tank to accommodate the needed 5 kg of propellant. The XACT-50 developed by Blue Canyon Technologies is used as the integrated attitude control and determination system. Thermal coating and heaters are used for thermal management. All of these systems are housed in an EnduroSat 12U XL main structure. A ground segment architecture has been presented using a combination of DLR's antennas, ESA's ESTRACK infrastructure and NASA's Deep Space Network. These are also used for navigation using ranging, Doppler and Delta-DOR measurements. Different launch opportunities have been studied, favouring a piggyback on another Mars-bound launch.

An orbit analysis has been performed to calculate low-thrust Earth-Mars transfer trajectories using the Tudat library developed at TU Delft. These trajectories have mainly been computed using the hodographic-shaping method. In order to assess the feasibility of the system in terms of its propulsion capabilities, an extensive array of trajectories has been examined. Assuming that the mission is launched through a piggyback on a Mars-bound launch, two main launch windows are considered: one at the end of 2028 and one at the end of 2030. An optima region can be found in both launch windows for N = 2 and TOF = 1000-1400 days. This region includes trajectories with promising results in both the consumed Delta-V and the maximum thrust. Using this data, a total Delta-V of 6.3 km/s (including the Delta-V required to place the spacecraft in a graveyard orbit at end-of-life) leads to 80% of the departure dates resulting in feasible trajectories. This would translate to a total propellant mass of 5 kg. Nevertheless, these trajectories exceed the maximum thrust requirement of 1.25 mN in peaks at the end of the trajectories. Since the average thrust along these trajectories is lower than the requirement, a strategy can potentially be implemented to distribute this peak in thrust over the entire trajectory. Using these trajectories leads to a mission lifetime of ~4 years. The mission analysis is completed by defining a strategy for Mars orbital insertion using ballistic capture and defining the PSO around Mars. This PSO will be a frozen and Sun-synchronous orbit to take advantage of Mars' gravitational parameters to enable a near-circular orbit.

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