Final Report

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Final Report

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CHANGE RECORDS

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1 INTRODUCTION

1.1 SCOPE AND PURPOSE

The scope of this document is present a complete description of all the work done during the study. It covers the whole scope of the activity, from a comprehensive introduction of the context, the description of the work performed and the main results achieved.

1.2 APPLICABLE DOCUMENTS

Code / DRL	Title	Reference	Issue
[AD1]	Small Satellite Missions to mars - An Architectural Study-Mission Requirements Document	ESA-E3P-MARSX-RS-001	1.0
[AD2]	Early Phase Mars Studies-Environmental Spec	ESA-E3P-MARSX-RS-002	0.2
[AD3]	Early Phase Mars Studies-Margin requirements	ESA-E3P-MARSX-RS-003	1.0
[AD4]	SMARTieS CDF Study – Session 10 – Radiation Plots	-	-
[AD5]	Small Satellite Missions to Mars - An Architectural Study - EXPRO PLUS SoW	ESA-E3P-MARSX-SOW-001	1.0
[AD6]	Margin Philosophy for Mars Exploration Studies	ESA-E3P-MSR-RS-001	1.0

1.3 REFERENCE DOCUMENTS

Code / DRL	Title	Reference	Issue
[RD1]	SMARTieS CDF Study Final Report	CDF-205(A)	N/A
[RD2]	ECSS standards	https://ecss.nl	N/A
[RD3]	Ariane 6 User's Manual		1.0
[RD4]	Annex toTN1 – Mission Architecture Review and Selection Report	TN-SMARTIES-POMI	2.0
[RD5]	TN1 – Mission Architecture Review and Selection Report	TASI-SD-S2M2-ADD-004	4.0
[RD6]	TN2 – Mission Requirements Document	TASI-SD-S2M2-RQM-0018	1.0
[RD7]	TN5 – Programmatic Approach	TASI-SD-S2M2-TNO-0472	1.0
[RD8]	TN3 – Spacecraft Design and Operations Report	TASI-SD-S2M2-TNO-0471	2.0
[RD9]	TN4 – Spacecraft performance Analyses	TASI-SD-S2M2-TNO-0477	1.0
[RD10]	Annex to TN4 – Spacecraft Performance Analyses	TN-SMARTIES-POMI-MSO-MA	1.0
[RD11]	Castellini, F., Bellei, G., & Godard, B. (2019). Mars Aerobraking Operations for ExoMars TGO: A Flight Dynamics Perspective. In H. Pasquier, C. A. Cruzen, M. Schmidhuber, & Y. H. Lee (Eds.), Space Operations: Inspiring Humankind's Future (pp. 661–694). Springer International Publishing.	https://doi.org/10.1007/978-3-030-11536-4_26	

1.4 DEFINITIONS AND ACRONYMS

Acronym	Description
AD	Applicable Document
CDF	Concurrent Design Facility
CONOPS	Concept of Operations
COTS	Components Off The Shelf
DTE	Direct To Earth
DSP	Dispenser
ECSS	European Cooperation for Space Standardization
ERO	Earth Return Orbiter
ESA	European Space Agency
ESOC	European Space Operations Centre
FP	Final Presentation
GS	Ground Segment
GTO	Geostationary Transfer Orbit
HEO	High Elliptic Orbit
HGA	High Gain Antenna
IMU	Inertial Measurement Unit
ITT	Invitation To Tender
LEO	Low Earth Orbit
LGA	Low Gain Antenna
MAR	Mission Architecture Review
MCC	Marc Communications Constellation
ME	Main Engine
MMH	Monomethylhydrazine
MON-3	Mixed oxides of nitrogen
MSO	Mars Science Orbiter
MSR	Mars Sample Return
OIM	Orbit Insertion Module
OBC	On-Board Computer
PDR	Preliminary Design Review
POLIMI	Politecnico di Milano (Polytechnic of Milan)
PT	Pressure Transducer
QSO	Quasi-Satellite Orbit
RD	Reference Document
RCS	Reaction Control System
RCT	Reaction Control Thruster
ROM	Rough Order of Magnitude
SM	Separation Mechanism
SoW	Statement of Work
TAS	Thales Alenia Space
TASO	Trans-Aerocentric Orbit
TBC	To Be Confirmed
TBD	To Be Determined
TN	Technical Note
TRL	Technology Readiness Level
TUM	Tag-Up Meeting
UHF	Ultra-High Frequency

1.5 DOCUMENT OUTLINE

Section 1 describes the purpose, scope and application field of this document. In addition, it gives useful information for a better understanding of the document such as applicable documents, reference documents, specific definitions used or refer to in the document and at least the list of acronyms used. A brief presentation of its content completes this section.

Section 2 recall the main objective and mission requirement that shall be fulfilled by the proposed architecture. The two required mission options are presented with a general assessment of launch, transfer and propulsion strategies of the different options. It includes an overview of the trade-offs of the options from the system design point of view. All the candidates have been compared in terms of mass, cost, schedule, risk and performances.

Section 3 describes the architecture of the selected best candidate architecture, with a first evaluation of all mission segments and mission phases.

2 STUDY OVERVIEW

The focus of the study is the system level assessment of potential small mission architectures that could operate at Mars in the 2030's after or potentially in parallel with MSR mission. In particular two cases has been evaluated:

- Mission 1: Mars Communications Constellation (MCC), for supporting assets on the surface and providing science
- Mission 2: Mars Science Orbiter (MSO), a Science mission driven, remote sensing

On the following paragraph of this chapter, an overview of the 2 options is presented. At the MAR, the most promising mission has been selected and further analyzed during the study and it is detailed on chapter 0 of this document.

2.1 MISSION REQUIREMENTS

2.1.1 Mars Communication Constellation

The main objective of the constellation option is reported in the req. MIS-MCC-020 and influences the number of satellites needed in the architecture design (MIS-MCC-010), the orbital arrangement of the satellites (MIS-MCC-070) and the communication subsystem characteristics (such as low gain and UHF carrier frequency).
The communications architecture shall be based on the well know Mars Relay Network protocols (MIS-MCC-050). The CCSDS Protocols are considered as standards so that the compatibility with the external assets is ensured. The use of UHF for Proximity-1 communications between the landed assets and the relays guarantees a relaxation of the AOCS requirements, as well as robustness to multipath and to atmospheric degradation of the signal, at the cost of limiting the achievable data rates, so this protocol is considered in this study to ensure a reliable and consolidated link design.
Regarding the DTE communications, req. MIS-MCC-100 and MIS-MCC-110 specify the worst case uplink and downlink capabilities that affects the science data amount collected on-board, the on board data storage volume sizing, the transmitting power, carrier frequency and antenna size. This is the core of the Constellation option and it is analyzed in depth in the par. 2.3.1.
As reported on [RD1] with a ride-share launch the spacecraft becomes more complex to ensure a feasible design, off-setting the majority of launch cost savings and a dedicated launch option, for a minimal extra cost compared to the dual launch to GTO scenario provide launch opportunity flexibility, thereby reducing schedule risks. In par. 2.3.1.1 both the departure from a GTO (MIS-MCC-070) then a dedicated launch with Ariane 6.2 are analyzed and compared. For the sake of completeness the same solutions are evaluated with Ariane 6.4.
The minimum lifetime of 2 years (MIS-MCC-030) is deemed feasible for the solutions proposed and the goal of 6 years can be easily reached.
For the req. MIS-MCC-040, accordingly to the note included, the mass limit is considered as a reference value not binding, because derived from the cost estimation. Consequently the payload mass limit (MIS-MCC-090) seems not a driver.

2.1.2 Mars Science Orbiter

The main objective of the Scientific mission is to perform valuable scientific observations of Mars. As secondary objective, a data relay capability is required (MIS-MSO-020) that can support the Martian landed asset (MIS-MSO-050).
As reported on [RD1], to launch into GTO and ensure a technically feasible spacecraft design, a chemical propulsion kick stage is needed to get onto a Mars trajectory. Using our heritage solutions the total mission costs including a kick stage could be kept low and a dedicated direct launch to Mars could be in the cost cap. Consequently both solution, using Ariane 6.2 are analyzed. As for the MCC, the same solutions are evaluated with Ariane 6.4.

The possibility to have a shared launch (MIS-MSO-060) with a direct launch on a Mars Orbit Trajectory (MIS-MSO-070) implies the opportunity to have two medium Mars mission launched at the same time, and it seems a rare situation considering that MSR is scheduled on a dedicated Ariane 6.4 launch and no other Mars mission are foreseen in the 2026-2032.
The req. MIS-MSO-080 is intended as the capability to reach a closed orbit around Mars. The propulsion system is designed accordingly, considering eventually an aerobraking maneuver that reduces the ΔV needed at the insertion. However, the aerobraking maneuver adds design and operational complexity to the mission and could increase the cost of Operations.
Differently form the Constellation option, no downlink/uplink performances are given, underlining that the priority is on the scientific value of the mission. Nevertheless we assume to allocate similar target performance to the communications system that, will benefit from lower operative orbits than MCC ones.
The minimum lifetime of 4 years (MIS-MSO-030) is deemed feasible for the solutions proposed.
As for the Constellation Option, for the req. MIS-MSO-040, accordingly to the note included, the mass limit is considered as a reference value not binding, because derived from the cost estimation. Consequently the payload mass limit (MIS-MSO-090) seems not a driver.

2.2 PLATFORM SELECTION

2.2.1 Thales Platform Heritage

Thales Alenia Space represents a worldwide standard for space development: from navigation to telecommunications, from meteorology to environmental monitoring, from defense to science and observation. With 125 satellites built as prime contractor for three different constellations (Globalstar 2, O3b and Iridium® NEXT), Thales Alenia Space is clearly the world's preferred partner in low and medium Earth orbit (LEO and MEO) communications satellite systems. The Iridium® NEXT constellation is recognized as the highest performance telecommunications system in the world.

Thales Alenia Space has supplied military telecommunications systems as prime contractor for more than 30 years now. We are the exclusive domestic supplier in France, with four generations of Syracuse satellites, as well as providing two Sicral first-generation satellites plus the second-generation Sicral 2 for the Italian defense ministry, Athena-Fidus for military and dual (civil-military) telecommunications for France and Italy, and two Satcom BW satellites for Germany. Through Athena-Fidus and Sicral 2,Thales Alenia Space is at the heart of European defense collaboration and it has also exported products reflecting its dual telecom system expertise to Turkey, Brazil and South Korea.

Thales Alenia Space, offering both optical and radar very-high-resolution instruments, draws on over 30 years of experience to propose a complete range of observation systems designed to meet market expectations. For instance, Thales Alenia Space is prime contractor for Italy's COSMO-SkyMed radar-based Earth Observation system.

We also built Turkey's Earth Observation satellite, including its high-resolution instrument. Thales Alenia Space has been the exclusive supplier of all very-high-resolution optical instruments for French intelligence satellites, including Pleiades, Helios and CSO. In 2018, we were chosen by South Korea to supply four Earth observation satellites with synthetic aperture radars (SAR). Today, drawing on our unrivaled expertise in optical and radar technologies, we are developing brand-new Earth Observation products, including high revisit solutions. Moreover, Thales Alenia Space and the American startup Spaceflight Industries have created LeoStella LCC, an equally-owned joint venture fully reflecting the needs of the New Space environment. Their aim is to deploy the first constellation featuring short revisit times, comprising 60 high-resolution optical satellites.

Thales Alenia Space is major partner in this ambitious environmental monitoring and management program, as prime contractor for the Sentinel -1 and -3 families (4 satellites per family), in charge of the Sentinel-2 image ground segment, and manufacturer of the Poseidon-4 radar altimeter for the Jason-CS/Sentinel-6 mission.

Moreover, we have already supplied a wide range of Earth Observation satellites and instruments used for oceanography, altimetry, meteorology, mapping, crisis management, climatology and much more. All Meteosat geostationary satellites were built by Thales Alenia Space as prime contractor.

We have already produced seven firstgeneration Meteosat satellites and four Meteosat Second Generation (MSG) satellites, and we are now working on the third generation (MTG). The latest generation will comprise four imaging and two sounding satellites.

Thales Alenia Space is the overall prime contractor for the ExoMars mission, and a major contributor to the BepiColombo mission to explore Mercury, the most mysterious planet in the Solar System. The company led the Herschel & Planck science mission, deploying the largest space observatories ever developed in Europe. We also developed Corot, France's own low-orbit "exoplanet" hunter (planets outside the Solar System), and we will be heavily involved in a new program called PLATO, also tasked with tracking exoplanets, but from the Lagrange 2 point.

Thales Alenia Space built 25 of the 64 huge parabolic antennas (Europe's contribution) for the giant ALMA radiotelescope array located on the Atacama plateau in Chile. In addition, we played a lead role on the famous Rosetta-Philae comet mission [especially via assembly, integration and testing of the spacecraft], as well as on Cassini-Huygens. The Huygens space probe was built by Thales Alenia Space as prime contractor.

Also on the agenda at Thales Alenia Space is the European program Euclid, which will help us better understand dark matter. In the meantime, Europe is holding its breath in the run-up to the ExoMars 2022 mission. ESA's rover on this mission should touch down on Mars in 2023.

Fitted with a special drill built by Leonardo, the rover will take soil samples at a depth of two meters, in an attempt to discover evidence of past life (bacteria), while the Trace Gas Orbiter (TGO), launched in 2016, continues its mission in orbit around Mars, “sniffing” the Martian atmosphere to discover traces of methane gas in particular Table 2-1 and Table 2-2 provide an overview of main TAS platforms.

Table 2-1 – TAS Main platforms – 1/2

	NIMBUS	HE R-1000	PRIMA-S	PRIMA	IGSC	EliteBus
Spacecraft Launch Mass range	150 ÷ 200 kg	900 ÷ 1100 kg	1100 ÷ 1300 kg	1800 ÷ 2300 kg	2100 ÷ 2700 kg	800÷1000 kg
Launcher	Ariane 6, Vega	Vega, Soyuz, Ariane 6	Ariane 6, Delta, Soyuz, Vega	Vega, Soyuz, Delta, Falcon 9	Vega, Ariane 6, Falcon 9, Proton, New Glenn	Soyuz Fregat, Falcon 9, Vega Dnepr, Terran
Platform Dry Mass	< 105 kg	< 550 kg	< 650 kg	< 1100 kg	< 1400 kg	< 390 kg
P/L Mass	< 80 kg	< 400 kg	< 450 kg	< 1200 kg	< 600 kg	< 300 kg
Power Generated	900 W	< 2000 W	< 2000 W	4200 W	> 14.6 kW	< 2000 W
Average Power to P/L	300 W	< 900 W	< 600 W	1100 W	< 6000 W	< 1000 W
Delta - V	< 70 m/s	< 300 m/s	120 m/s	120 m/s	< 7000 m/s	< 380 m/s
Propellant Mass	< 8 kg (Hydrazine)	< 135 kg (Hydrazine)	< 130 kg (Hydrazine)	< 155 kg (Hydrazine)	< 700 kg (Xe)	< 170 kg (Hydrazine)
Orbit type	LEO	LEO	LEO/MEO	LEO/MEO	GEO	LEO, MEO
Orbit inclination	0 ° to 100 °	0 ° to 100 °	0 ° to 100 °	0 ° to 100 °	0 °	84°
Operative lifetime	> 5 years	7 years	7 years	7 years	15 years	12 years
Propulsion System	Chemical (Monoprop.)	Chemical	Chemical	Chemical	Electrical	Chemical (Monoprop.)

Table 2-2 – TAS Main platforms – 2/2

Elite Next Gen	Spacebus 4000	Spacebus Neo	Space Inspire	MILA (France)	MILA (Italy)
2000 kg	3000 ÷ 6000 kg	3000 ÷ 6000 kg	< 2500 kg	1000 ÷ 2000	1000 ÷ 2000
Falcon 9, Dnepr, Terran	Proton, Ariane 5 ECA, Falcon 9, Atlas	Ariane 5, Ariane 6, Falcon 9	Ariane 5, Ariane 6, Falcon 9	VEGA, Ariane 6, Soyuz	VEGA, Ariane 6, Soyuz
< 1000 kg	900 ÷ 2000 kg	< 650 kg	2000 kg	2060 kg	2060 kg
350 ÷ 550 kg	600 ÷ 750 kg	up to 2 Tonnes		500 ÷ 1200	500 ÷ 1200
	8700 ÷ 15800 W	< 20000 W	< 18000 W	0.5KW- 6.5KW	0.5KW- 6.5KW
2000 ÷ 4500	9 ÷ 14000	12 ÷ 20000	< 14000	1000 ÷ 4000	1000 ÷ 4000
< 4270 m/s	<3500 m/s	TBC	TBC	TBC	TBC
< 460 kg (Xe)	< 2000 kg	TBC	TBC	>300kg	>300kg
LEO, MEO	GEO	GEO	GEO	LEO (700 km)	LEO (700 km)
TBC	0 °	0 °	0 °	TBC	
TBC	15 years	15 years	15 years	7.5+5 years	7.5+5 years
Electrical	Chemical (Bipropellant)based on S-400 (16 thrusters)	Electrical	Electrical	Chemical	Chemical

2.2.2 Electric Platforms

Before going in the comparison of the different platforms, it is important to mention an electrical small platform that is done in partnership with Thales Alenia Space called PLATiNO. It is a SITAEL All-electric Multi-purpose 200kg small satellite platform, deployable in constellation and suitable for a wide range of multi-mission applications (Optical, SAR, Telecom, etc.).

The SAR payload on board PLATiNO-1, designed and manufactured by Thales Alenia Space in Italy, is an innovative Micro-SAR, able to image both in passive and active mode, that literally redefines the market performance of this technology.

Combined with the high power that the PLATiNO platform is able to provide, PLATiNO-1 can guarantee up to 5 minutes scan time per orbit, performance nowadays unmatched in the Micro-SAR sector, with ground resolution up to 1 meter.

The propulsion system is based on SITAEL low power HET electric propulsion (HT100). It is the smallest and lowest power HET ever developed in Europe, whose performance and characteristics represent the state-of-the-art of this technology. Based on permanent magnets, the HT 100 HET is designed to perform orbit control tasks on micro-satellites and AOCS tasks on mini-satellites. The HT 100 thruster unit and all the key sub-system components are fully based on European, ITAR-free technologies. The HT 100 thruster unit can be offered with its HC 1A S-type hollow cathode, 300 W max PPU and dedicated fluidics based on flight-proven valves.

This engine is not suitable for an interplanetary transfer, so the platform should be upgraded to HT5K that has been designed to meet the requirements of modern communication and navigation satellites, performing propellant-saving LEO-GEO/LEO-MEO transfers as well as station keeping tasks on large geostationary platforms.

Due to the Large Delta-V required for its mission in GEO, the preliminary configuration for IGSC adopts this kind of Hall Effect plasma thruster (HET) system with Xenon propellant. Two Thrusters, each one capable of providing a continuous thrust up to 290 mN, are mounted on 2 dedicated thrusters orienting robotic arm operating in redundant branch.

Despite the higher efficiency of this type of propulsion wrt chemical one, its utilization is not convenient for this particular mission scenario because the on board power generation will not be sufficient.

In order to lower the cost, we assume a S/C required power at Mars of 0.5 kW (with margins and without the electrical propulsion) to maintain a small size of the solar array (around 5 m^2). Referring to the IGSC specific case, the power request from the Electric Propulsion S/S is around 5 kW for moving a ~2500 kg spacecraft. Even assuming an half platform mass (with lower energy demand from its electric PRP S/S), the total S/C power request would not be satisfied in Mars Orbit.

2.2.3 Selected Platforms

Comparing the Thales platforms to the outcome of the analysis, we can consider:

- The Electric Platforms IGSC, Spacebus Neo, Space Inspire, EliteBus Next Gen are too massive (> 1000 kg) and incompatible with the cost cap.
- The Chemical Platforms Spacebus 4000, PRIMA-S and PRIMA are too massive (> 1000 kg) and hardly compatible with the cost cap.
- The EliteBus chemical platform, even if optimized for multiple launch, is too massive to be used in the MCC option, and due to the peculiar shape of the structure is less flexible to accommodate modifications than other TAS solutions (as HE-R1000)

We investigate also the possibility to build a recurrent TGO, removed EDM, as a possible Science Orbiter, as it is a spacecraft fully equipped to perform a scientific mission to Mars.

This architecture has demonstrated its robust design, optimum performance, and it is operative nowadays. The unusual external structure has been customized for specific scientific payload requirements that could be not applicable anymore(unless the same kind of payload would be needed, with similar requirements).

However, no matter the payload, this solution is quite massive (<3000 kg) and the implementation of a pure recurrent TGO, could have an high cost that exceed the target of this study. Last but not least, considering the operative time frame (2026-2032), there is a risk due to the obsolescence of the project itself that could require some critical changes that will cause a further increase of the costs.

To stay in the ExoMars perimeter we considered also the Carrier of the ExoMars RSP mission that is more recent than the TGO. In this case it cannot be considered as an alternative platform because it has a peculiar design, based on a expendable spinning spacecraft that with a ballistic trajectory deliver the DM on a collision path with Mars without performing any insertion maneuver around the planet. Due to its specific design, any modification will add non-recurrent cost and require to adapt the development plan, exceeding the programmatic and cost constraints.

Thales Alenia Space is also currently developing a European platform product line using a building-blocks approach to move towards transnational standard Earth Observation platforms. Thales Alenia Space, in cooperation with ESA, will use the experience and lessons learnt from the first three Sentinels and from constellation and multi-mission platforms (Proteus (TAS-F) and Prima (TAS-I)) to merge and harmonise these existing platforms into an European EO platform family called M.I.L.A that can be produced in either France, Italy or the UK. The schedule of this new platform may not be in line with the S2M2 needs, therefore it is not baselined, however if the platform or any equipment from this initiative becomes available and is suitable for S2M2 then this could be an option to be considered in future phases.

In conclusion, the most suitable platforms are NIMBUS for the MCC and HE-R1000 for the MSO.

2.2.3.1 NIMBUS

NIMBUS (New Italian Micro BUS), is a micro-satellite platform developed from the flight heritage of the PRIMA one. It is based on a modular multi-purpose platform, is therefore a highly flexible platform which allows to shorten the development times. NIMBUS is meant to serve different mission applications and operational scenarios with different payload (class 150-200 Kg, average power 100 W , peak power up to 1KW).

The satellite body is based on the 2-MF (Multi-Function Modular Frame, also named “Tray” Patented concept) modular approach that allows, through the stacking of the needed number and type of standard “trays”, a prompt and easy fulfillment of any mission exigencies in terms of:

- internal and external units accommodation volumes and I/Fs
- external interfaces for appendages and P/L linking
- interfaces for the launch vehicle (base ring or side attachment)

The Platform structure is based on the following preliminary Tray configuration for housing all PF units:

- 3 VFT typology (Volumetric Functional Tray);
- 3 GFT typology (General Functional Tray);
- 1 LVA + Bottom panel for the Propulsion S/S

The Structure S/S provides adequate mechanical performances for coping with the majority of the commercial launchers, effective radiation shielding, electrical Faraday cage and an isopotential external surface.

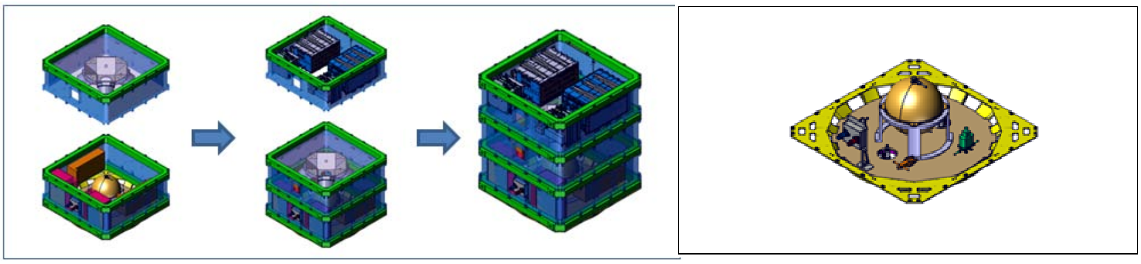


Figure 2-1 – On the left VFT (in blue) and GFT (in green). On the right is shown the LVA and the Propulsion Panel.

NIMBUS is three – axis stabilized, implements the IPAC Integrated Power, Attitude Control and the Data Handling Control Unit contains redundant modules:

- Multi Core Processor Module (MCPM)
- Telemetry, Telecommand, Mass Memory
- Reconfiguration Module (T2MR)
- Input & Output Module (IOM)
- Power Module (PWM)

As actuator, NIMBUS implements a Mini Control Moment Gyro assembly, designed for strongly enhancing attitude agility of microsatellites.

NIMBUS design is optimized for multi-launch: up to 8 satellites can be accommodated in a cylindrical dispenser, and keeps compatibility with Ariane 6 (as well as VEGA) see Figure 1 5 (in this picture VEGA is taken as reference). The dispenser would accommodate the satellite in several positions (e.g. top position and upper module).

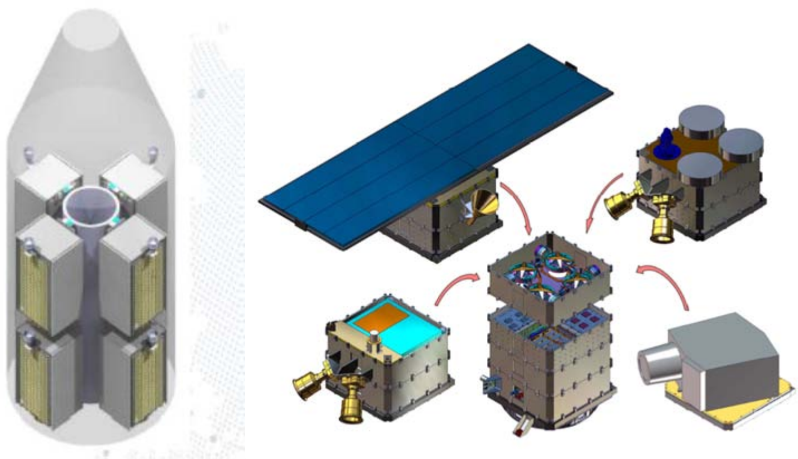


Figure 2-2 - NIMBUS modularity concept and VEGA accommodation (just for an indicative geometric representation)

The reference configuration for NIMBUS is sized for a Payload with following characteristics:

- Average Power per orbit: 100W

- Peak Power until ≈900 W for 3 min
- Reference scenario for EPS sizing a SAR PL
- Life Time 3 yrs
- Sun synch orbit

For this configuration the mass budget is reported on the following table.

Table 2-3 NIMBUS reference mass budget

NIMBUS	Mass [kg]	Margin	Mass with Margin [kg]
AVIONIC S/S	***25.02***		***26.27***
IPAC	*17.45*	5%	*18.32*
mini RW	*5.20*	5%	*5.46*
MGT	*1.20*	5%	*1.26*
MGM	*0.66*	5%	*0.69*
STT	*0.41*	5%	*0.43*
Solar MEMS SS	*0.10*	5%	*0.11*
COMM S/S	***9.11***		***9.56***
ICU	*6.36*	5%	*6.68*
S BAND ANTENNA	*0.20*	5%	*0.21*
RF COMPONENTS	*0.30*	5%	*0.32*
COAXIAL CABLES	*0.10*	5%	*0.11*
X BAND SSPA	*1.90*	5%	*2.00*
X BAND ANTENNA	*0.25*	5%	*0.26*
EPS S/S	***40.60***		***42.63***
PCDU	*7.00*	5%	*7.35*
BATTERY ASSEMBLY	*6.00*	5%	*6.30*
Solar Array	*17.40*	5%	*18.27*
DC & RF HARNESS	*10.20*	5%	*10.71*
Propulsion S/S	***3.68***		***3.87***
FDV	*0.11*	5%	*0.12*
FVV	*0.11*	5%	*0.12*
PROPELLANT FILTER	*0.08*	5%	*0.08*
PROPELLANT TANK	*1.29*	5%	*1.35*
LV	*1.10*	5%	*1.16*
PT	*0.28*	5%	*0.29*
RCT	*0.61*	5%	*0.64*
PIPING SET	*0.10*	5%	*0.11*
STRUCTURE (just PF)	***29.85***		***31.34***
LVA+Bottom	*8.00*	5%	*8.40*
VFT (volumetric)	*12.60*	5%	*13.23*
GFT (green)	*5.25*	5%	*5.51*
panels (Propulsion + mini RWs)	*1.00*	5%	*1.05*
Tertiary and Misc quota parte PF	*3.00*	5%	*3.15*
Platform Dry Mass	***108.30***		***113.67***

System Margin		*20%*	*22.73*
Total Platform Dry Mass with margin			*136.40*

The max propellant that can be loaded is 3 kg, so the total wet mass with margin in 139.4 kg

The development of NIMBUS is ongoing and the QSR is planned for October 2022, in line with our needs.

2.2.3.2 HE-R1000

In parallel to NIMBUS, we selected HE-R1000 that has been designed in order to be geometrically and mechanically compatible with a variety of commercial low-medium class launchers for single and for double launches. HE-R1000 is a 3 axis stabilization platform that integrates in a single main module all the BUS units, the propulsion subsystem and the payload equipment, including the pertinent appendages.

The HE-R1000 S/C has been designed taking into account the LEO reference orbit environment over the 7 Years operative mission lifetime (plus a 6 months for LEOP and commissioning).

The platform is organized into the classical subsystems sized for LEO EO missions but capable to extend, through the needed adaptations, its operating range to:

- Mission types: Science, TLC, Navigation.
- Orbit type: MEO and others;
- Payload accommodation: SAR, Optical, TLC, Scientific;
- Payloads resources requirements: variability in mass, power, signal/command lines, etc…;
- Commercial Launch vehicle: wide selection commercial launchers

Finally, the HE-R1000 physical architecture well pre-arrange the “stack ability” option for providing a three S/C launch set. This platform has an outstanding performance and customization capabilities to respond to the customer requirements, minimizing non-recurring cost, with a dry mass lower than 550 kg and a payload capability up to 400 kg. It offers reliability of 0.9@7 years, with a Power capability of 2000 W Solar array (BOL) at Earth and a 60-80 Ahr battery.

The propulsion capability is hydrazine based with a modular tanks supporting structure allowing polar/equatorial mounting and for a tanks capacity range from 35 to 160 kg. The system is a blow-down full redundant dual branch with 4, 6, or 14 RCTs.

The platform has high pointing accuracy (0.01°) and knowledge (0.003°), and allow high precision position real time knowledge better than 10 m, on 3 axes (3 sigma).

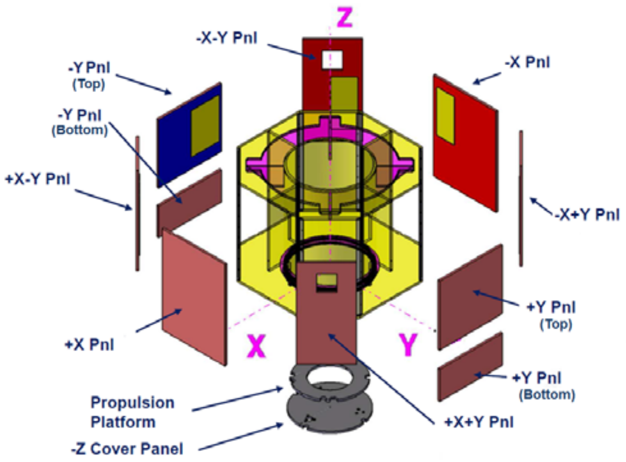


Figure 2-3 - S/C reference frame and panels names

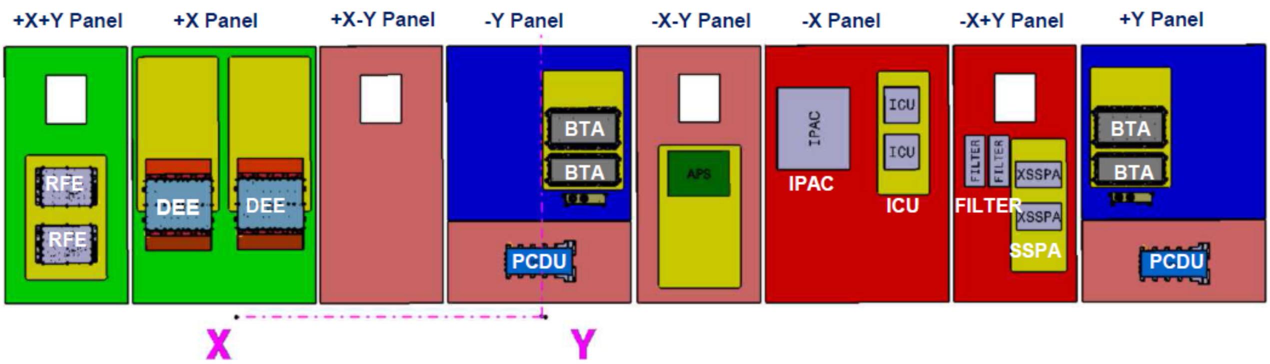


Figure 2-4 - Panels configuration for the S/C BUS (the figured available spaces are reserved to P/L equipment)

For the basic configuration, the HE-R1000 mass budget is reported hereafter.

Table 2-4 – HE-R1000 Mass Budget (without Payload)

S\S	Acronym	Description	Quantity	Unit Mass (Kg)	Margin	Unit Mass with Margin (Kg)	Total Mass with Margin (Kg)	
STRUCTURE			1	168.4	5%	176.8		176.8
TCS			1	19.1	5%	20.0		20.0
PROPULSION								23.4
	LT	Propellant Tank	1	16.1	5%	16.9	16.9	
	FDV	Fill & Drain Valve	3	0.2	5%	0.2	0.5	
	FVV	Fill & Vent Valve	1	0.2	5%	0.2	0.2	
	LF	Propellant Filter	1	0.1	5%	0.1	0.1	

	LV	Latching Valves	2	0.4	5%	0.4	0.8	
	PT	Pressure Transducer	1	0.3	5%	0.3	0.3	
	RCT	Reaction Control Thruster	8	0.4	5%	0.5	3.7	
	PIP	Propulsion Piping	1	1.0	5%	1.1	1.1	
HARNESS								**34.0**
			1	31.8		34.0	34.0	
EPS								**152.6**
	BTA	Battery Assembly	3	23.3	2%	23.8	71.4	
	PCDU	Power Conditioning and Distribution Unit	2	6.2	5%	6.5	13.0	
	APS		1	0.0		0.0	0.0	
	SAW 1482x1610	Area per wing 4.77 m2	4	16.3	5%	17.1	68.3	
AVIONICS								**26.1**
	IPAC	Integrated Power Avionic Communication	1	17.5	6%	18.6	18.6	
	STT	Star Tracker	4	1.8	2%	1.9	7.5	
	FSS	Fine Sun Sensor	4	0.0	2%	0.0	0.1	
	RW	Reaction wheels	4	6.0	8%	6.5	26.0	
TT&C								**16.3**
	ICU	Instrument Control Unit	2	3.2	10%	3.5	7.0	
	SSPA	Solid State Power Amplifier	2	1.8	10%	2.0	4.0	
	CHF	Channel Filters	2	1.5	10%	1.7	3.3	
	DLSM	Down Link Switch Matrix	1	0.2	26%	0.3	0.3	
	TT&C Hrn&Hyb	Telemetry Tracking & Command Harness	1	1.5	5%	1.6	1.6	
	SBA	S-Band Antenna	1	0.2	5%	0.2	0.2	
PAYLOAD								**0.0**
	TBC		1	0.0		0.0	0.0	
INTEGRATION								**19.8**
	MISCELLANEA			16.5	20%		19.8	
						Total Dry Mass		**469.1**
						System Margin	20%	93.8
						Dry Mass with Margin		**562.9**
						Propellant		**135.0**
						Total Wet Mass		**697.9**

The development of HE-R1000 is ongoing and by the end of 2021 the SM should conclude the mechanical qualification for a commercial customer delivery. The avionics is on development and no delays are foreseen.

All the mission architecture considered are based on the usage of chemical propulsion, and for some of the options the MSR OIM is considered as kick-stage to reach the operative orbit at Mars. Depending on the launcher transfer, the OIM perform all the maneuvers that are needed after the launcher separation to reach the final orbit.

HE-R1000 and NIMBUS cannot be used without modification to perform a mission to Mars because the TT&C is not sized for long distances communications with Earth. The propulsion system is not sized to host such a big amount of propellant as well as the Power subsystem capability is limited and not ready to an interplanetary mission.

2.2.3.3 KICK-STAGE SEGMENT

The OIM is a propulsive module that will be used in the MSR-ERO mission. The choice of selecting the OIM as a kick-stage is given by the fact that TAS is taking part to the MSR campaign with the responsibility for the OIM manufacturing, and its architecture could be taken without any modification leading to a pure recurrent cost. This aspect shows both technical and programmatic advantages, as summarized below.
The OIM is a bi-propellant MMH/MON-3 system with pressure regulation allowing it to operate both in regulated and blowdown modes.
The OIM system comprises of the following main components:

- 4 MEs and 16 RCTs arranged in 2 branches (prime and redundant)
- 4 MMH fuel tanks
- 1 MON-3 oxidiser tank
- 2 pressurant tanks containing high pressure Helium
- Pipes, service valves, PTs, filters, harness with all necessary support
- Thermal H/W (MLI blankets, radiator coatings)

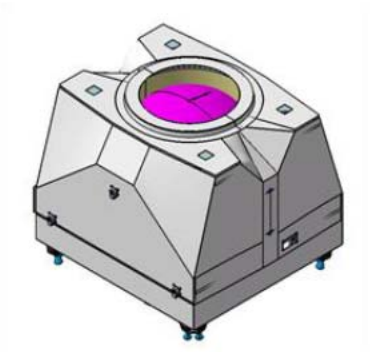


Figure 2-5 - OIM

The OIM main parameters are shown in table below.

Table 2-5 – OIM main parameters

	Data	Unit
Dry Mass (w/o margin)	520	kg
Propellant Mass	1921	kg
ME		
Thrust 2+2 engine	800	N
I_{sp}	320	sec
RCT		

Thrust (single engine)	22	N
I_{sp}	305	sec

The implementation of the OIM for the S2M2 scenario takes advantages by the simultaneous development in the frame of the MSR mission.

2.3 MISSION ARCHITECTURE

2.3.1 Mars Communications Constellation (MCC)

2.3.1.1 MISSION ANALYSIS

In the following paragraphs the Mission analysis for the MCC is presented. For the complete analysis and results discussion, see the MAR documents ([RD4], [RD5]).

2.3.1.1.1 Analysed configurations

The section introduces to the set of possible configurations studied for the constellation.

The following types of orbits are evaluated for the purpose:

- **Trans Areo-Stationary Orbit (TASO):**
  The TASO is a Keplerian, circular orbit, having a semimajor axis of 21000 km. Such trajectory, although not stationary with respect to Mars's surface (15° drift per Martian sidereal day), ensures a higher stability degree than the synchronous orbit.

- **Quasi-Satellite Orbits (QSO):**
  QSO are non-Keplerian trajectories hovering around the moon of the binary system (Mars-Phobos or Mars-Deimos) and are subjected to the attraction of the two bodies almost equally. The result is a formation-flying like trajectory around the moon, which however is stabilized by the latter, removing the relative drift that typically arises from formations of non-attracting objects (such as two artificial satellites).

- **Horseshoe orbits (HS):**
  HS orbits are non-Keplerian orbits with altitude and period similar to those of the moon of the system. A satellite placed on such orbit would have a very slow drift with respect to the moon (depending on the altitude differences between the two). When approaching the moon, a satellite on such orbit naturally executes a flyby of the moon, changing its semimajor axis and gaining distance from the moon itself, until the next encounter. Such low-paced drift may be exploited to minimize station keeping costs for relative phase maintenance between satellites. Despite the nominal long-term drift, perturbations may accelerate such process, therefore a proper optimization would be needed to have the minimum drift possible, as for TASO scenario.

The three types of orbits are explored to host a formation of 2 or 3 spacecraft:

- 2 satellites constellations:
  - QSO + HS (180° from moon) in Mars-Phobos system
  - QSO + HS (180° from moon) in Mars-Deimos system
- 3 satellites constellations:
  - TASO (0°) + TASO (120°) + TASO (240°)
  - QSO + HS (120°) + HS (240°) in Mars-Phobos system
  - QSO + HS (120°) + HS (240°) in Mars-Deimos system

2.3.1.1.2 Coverage and Communication Analysis

To assess the benefits of the previously described architectures in terms of data transmission, a coverage and communication analysis has been carried out.

The link parameters are the same reported in the CDF report, with the only difference in the transmitted power, which has been reduced to 4W (instead of 9W), and the introduction of the granular data-rate performed by Electra, that selects the discrete data-rate value (among 1/2/4/8/16 Kbps) based on the available signal-to-noise ratio. The power reduction is motivated by the selection of Electra EUT (e.g., used on ExoMars), instead of Electra Lite. It can indeed reach 5W, but 4W are used for the sizing to consider temperature and aging effects.

Visibility windows of the satellites have been simulated for a user at the equator. As reported in Figure 2-6, a satellite in Phobos' orbit performs 2.5 windows per day, with 4h duration per window. Deimos' satellites instead have a higher duration of 33h, due to their low relative velocity. On the other hand, windows' frequency is much lower, increasing the time gap between two successive windows.

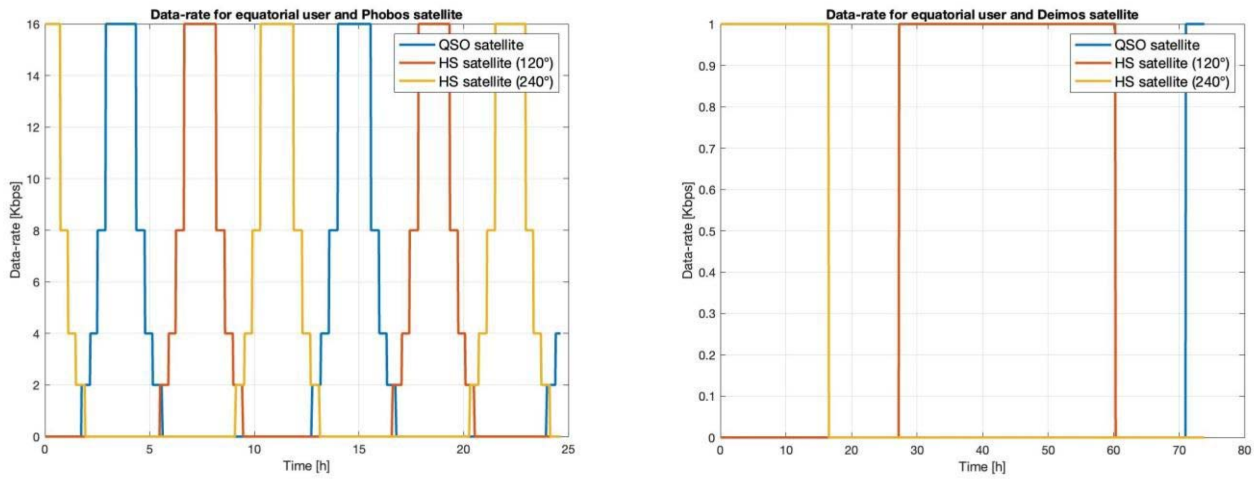


Figure 2-6: Coverage analysis for three satellites in Phobos (left) and Deimos (right) environments.

The duration of the gaps between two successive windows and their time occurrence is dependent on the specific architecture. In Table 2-6 a summary of the performances for each architecture is reported.

Table 2-6: Architectures telecommunication comparison.

Architecture	**Trajectories**	**Initial Phase Angle**	**ISL Nominal Distance**	**ISL Nominal Nadir Angle**	**Visibility Window Duration**	**# Visibility Windows / sat / day**	**Total visibility time**	**Consecutive windows gap**	**Data Return / day / sat**	**Data Return / day**
TASO	TASO	0°	-	-	24h	1	24h	0h	86.40 Mb / sat	259.2 Mb
	TASO	120°	42000km	30°						
	TASO	240°	42000km	-30°						
Phobos										
1	QSO	0°	-	-	4h	2.5	20h	1h - 6h	314 Mb / sat	628 Mb
	HS	180°	20000km	0°						
2	QSO	0°	-	-	4h	2.5	24h	0h	314 Mb / sat	942 Mb
	HS	120°	16000km	30°						
	HS	240°	16000km	-30°						
Deimos										
3	QSO	0°	-	-	33h	1	24h	33h	43.2 Mb / sat	86.4 Mb
	HS	180°	46000km	0°						
4	QSO	0°	-	-	33h	1	24h	9h	68.04 Mb / sat	204.12 Mb
	HS	120°	42000km	30°						

	HS	240°	42000km	-30°						

2.3.1.1.3 Transfer and Launch Strategy

The interplanetary trajectory is designed following the patched conics approach, and considering impulsive manoeuvres only. In order to take the Gravity Losses into account, dedicated Finite Thrust analysis have been performed for the escape and capture phases. For a more complete description of the developed strategies and the assumptions adopted see [RD4] and [RD5].

The adopted assumptions lead to the following division of the trajectory:

1) The **Earth escape:** in this phase the spacecraft shall escape from earth gravity field, matching the boundary conditions (mainly C3 and asymptote declination) required by the interplanetary trajectory. Different strategies have been considered:

   a **Direct Hyperbolic Escape**
   The easiest way to reach Mars: the launcher injects the spacecraft directly in the correct escape trajectory. The spacecraft shall perform only small correction manoeuvres, therefore the nominal ΔV for this strategy is close to zero. The drawback is that the cost of the launch is higher and the launchable mass is significantly lower with respect to the other scenarios.

   b **HEO departure**
   In this strategy the spacecraft is injected by the launcher in a High Elliptical Orbit (900000km x 250km) around Earth with a 6° inclination. The spacecraft shall perform a change plane (that, due to the altitude of the apocenter, is cheap) and a pericentre manoeuvre to reach the desired energy.

   c **GTO departure – unconstrained perigee**
   This is the most flexible strategy: launches in the GTO orbits are extremely frequent and are cheaper with respect to HEO and Direct escape launches. Moreover, the possibility to find a shared launch is high. Unfortunately, the departure cost from this orbit is quite elevate, even if a three manoeuvres strategy is applied (basically an intermediate HEO orbit is reached to reduce the cost of the plane change). In this scenario, the Local Time of the perigee is free.

   d **GTO departure – Local Time of perigee constrained**
   This is the most flexible strategy: launches in the GTO orbits are extremely frequent and are cheaper with respect to HEO and Direct escape launches. Moreover, the possibility to find a shared launch is high. Unfortunately, the departure cost from this orbit is quite elevate, even if a three manoeuvres strategy is applied (basically an intermediate HEO orbit is reached to

reduce the cost of the plane change). In this scenario, the Local Time of the perigee is forced to be midnight.

e **GTO departure with Moon-Earth Gravity Assists**
This strategy mitigates some drawbacks of the GTO departure. It includes a Gravity Assist of the moon to perform naturally the change of plane (and partially to increase the energy). The drawback is an increased complexity for the operations and the necessity to wait on the GTO until a correct phasing with the moon is reached.

2) The **interplanetary trajectory:** in this phase the spacecraft travels from the Earth sphere of influence to the Mars one.

3) The **Mars Orbit Insertion:** in this case the spacecraft shall close the trajectory around Mars and reach the correct operational orbit, matching the boundary conditions (mainly C3 and asymptote declination) required by the interplanetary trajectory.

Table 2-7 shows the summary of the whole ΔV for the MCC scenarios, including Margins and Gravity losses as in [AD6]. A complete treatment of the MCC scenarios ΔV breakdown can be found in [RD4] and [RD5].

Table 2-7 DV summary for the MCC scenarios

OIM Module						
Scenario	GTO	GTO constrained	GTO + GA	HEO	Direct	
TASO	3,372	3,854	3,494	2,561	1,810	GTO + GA escape could be lowered by 200 m/s if no constraints on pericenter are applied
Phobos QSO+HS	3,655	4.137	3.777	2.844	2,093	
Deimos QSO+HS	3,351	3,833	3,473	2,54	1,789	

2.3.1.1.4 Operational Orbits Costs

Constellation maintenance and phasing costs are evaluated per orbit type.

Phasing costs depend on the orbit type and on the constellation setup, therefore separate phasing scenarios for the same orbit are presented when applicable.

Station keeping cost is orbit-dependent only, therefore its value is analysed once for each trajectory.

Trans Areo-Stationary Orbits (TASO)

TASO are employed in a 3-satellite constellation only, therefore a 120° phasing is required by two satellites. Furthermore, relative altitude between spacecraft is relevant to the stability of the orbit, hence it is assumed to control two satellites only, to adjust their altitude with respect to the third one. As a worst-case scenario, it is assumed that each spacecraft cannot exceed a drift of 2 degrees from its nominal phase. It is worth noting that, although larger drifts can be achieved without losing coverage at the equator (a maximum value of 25°-30° is estimated), more stringent values are expected for higher altitudes, and the 2 degrees constraint can be considered a safe, margined value.

Quasi Satellite Orbits (QSO)

QSO have been verified to be very stable in high fidelity models (for both Phobos and Deimos), therefore no station keeping would be required. In scenarios involving non-Keplerian orbits, the QSO is always considered the target for the OIM injection, therefore no phasing is required from the satellite stationing in such orbit.

Horseshoe (HS)

Horseshoe orbits are employed in all non-Keplerian scenarios, and display different station keeping values depending on the reference binary system (Mars-Phobos/Mars-Deimos). Maximum phase error is dependent on the specific binary system: in Mars-Phobos scenarios, the shorter period of the orbit (with respect to Mars's surface) ensures good coverage up to a shift of 20°; in Mars-Deimos case, orbital frequency is comparable to TASO, therefore the same threshold of 2° is set.
Initial phasing cost is also dependent on the configuration, as in 3-satellite constellation a 120° shift is required, while a 180° phase is necessary for the 2-satellite constellation.

Table 3-12 reports the expected (margined) costs for phasing and station keeping (of a single satellite) in the described orbits, for a time span of 6 years. The reported costs are referred to the spacecraft already inserted in the operative orbits, and the corresponding manoeuvres are executed by the spacecraft only (OIM injection costs are not included).

Table 3-12: Delta V breakdown for each operative orbit (and for a single satellite) of the explored scenarios for the MCC (spacecraft maneuvers only, OIM injection not included). Costs are evaluated in a time span of 6 years.

<table>
<tr><th>Orbit</th><th>TASO</th><th>QSO (Phobos/Deimos)</th><th colspan="2">HS (Phobos)</th><th colspan="2">HS (Deimos)</th></tr>
<tr><td>Target Phase [deg]</td><td>120</td><td>0</td><td>120</td><td>180</td><td>120</td><td>180</td></tr>
<tr><td>Phasing cost [m/s]</td><td>21.2</td><td>0</td><td>15</td><td>17.5</td><td>22.6</td><td>28.8</td></tr>
<tr><td>Station keeping cost [m/s]</td><td>0</td><td>0</td><td colspan="2">4</td><td colspan="2">25.8</td></tr>
<tr><td>Total Cost [m/s]</td><td>21.2</td><td>0</td><td>19</td><td>21.5</td><td>48.4</td><td>54.6</td></tr>
</table>

Considering the costs per spacecraft and per orbit (as reported in Table 3-12), and the station keeping and phasing strategies adopted in the various constellation scenarios, the following operational orbit costs are obtained:

- TASO (3 satellites): 42.4 m/s
- Phobos QSO + HS (2 satellites): 21.5 m/s
- Phobos QSO + HS (3 satellites): 38 m/s
- Deimos QSO + HS (2 satellites): 54.6 m/s
- Deimos QSO + HS (3 satellites): 96.8 m/s

2.3.2 Mars Scientific Orbiter (MSO)

2.3.2.1 MISSION ANALYSIS

The present section describes the preliminary mission analysis for the Mars Scientific Orbiter. Further details can be found in MAR document and Annex ([RD4] and [RD5]).

2.3.2.1.1 Analysed configurations

The purpose of the MSO is to characterize the surface of Mars, and, in particular, human landing sites. Such objective requires the MSO to be at an altitude as low as possible to better characterize the surface, without reaching too high costs for station keeping. A good compromise is represented by altitudes around 300 km, as lower values would increase orbit maintenance costs above acceptable levels, while higher values would gradually reduce the quality of the observations.

The proposed orbit is a sun synchronous trajectory within the aforementioned altitude range. The out of-plane configuration allows to cover several regions of the whole Martian surface; however, altitude plays an important role in the actual coverage. Altitude values close to 300 km generate synchronicity with the surface, thus causing repeated passages above the same regions. By increasing the altitude to 320 km, it is possible to remove the synchronicity, and achieve a complete the full surface coverage in 7 days.

To maintain the properties of sun synchronous orbits, and ensure a node drift of 360° per Martian year, the following combination of orbital parameters is set:

- Mean semimajor axis: 3716 km
- Mean eccentricity: 0.009
- Mean inclination: 92.76 deg
- Mean argument of pericenter: 270 deg

It has to be noted that such low altitude, although optimal for surface observation, suffers from a relatively fast decay due to the planet's upper atmosphere.

An analysis of impacts of altitude increment on surface coverage, observation quality, orbital decay and station keeping, are performed in Section 3.1.1. The results will allow to assess the convenience of maintaining the current mean altitude (at the cost of a fast decay/higher station keeping costs) or increasing the altitude to a more stable orbit (at the cost of reduced observation performances).

2.3.2.1.2 Transfer and Launch Strategy

The interplanetary trajectory is designed following the patched conics approach, and considering impulsive manoeuvres only. In order to take the Gravity Losses into account, dedicated Finite Thrust analysis have been performed for the escape and capture phases. For a more complete description of the developed strategies and the assumptions adopted see [RD6].

The adopted assumptions lead to the following division of the trajectory:

1) The **Earth escape:** in this phase the spacecraft shall escape from earth gravity field, matching the boundary conditions (mainly C3 and asymptote declination) required by the interplanetary trajectory. For this analysis the HEO departure was considered
2) The **interplanetary trajectory:** in this phase the spacecraft travels from the Earth sphere of influence to the Mars one.
3) The **Mars Orbit Insertion:** in this case the spacecraft shall close the trajectory around Mars and reach the correct operational orbit, matching the boundary conditions (mainly C3 and asymptote declination) required by the interplanetary trajectory. To decrease the energy of the orbit, two different strategies have been analysed:

    a **Apocenter lowering using chemical thrusters:** in this scenario the final orbit is reached using a pericentric impulsive manoeuvre.

    b **Aerobraking:** in this scenario (further detailed in Sec. 3.1.1.3 and in [RD6]) the final orbit is reached by means of an Aerobraking manoeuvre. In this case the altitude of the apocenter of the initial Aerobraking orbit is a degree of freedom to balance the duration of the manoeuvre and the thermal/mechanical loads up to the spacecraft. In Particular, as presented in Sec. 3.1.1.2 and Sec. 3.1.1.3, a pericentric manoeuvre to reduce the apocenter of the initial Aerobraking orbit.

Table 2-8 and Table 2-9 show the transfer ΔV for the Aerobraking and powered descent scenarios.

Table 2-8 MSO ΔV breakdown – Aerobraking from 4 Sol MOI

Phase\Launch windows	11 Nov - 6 Dic 2028	2 Dic – 28 Dic 2030	8 Apr – 3 May 2033
Earth Departure	0.761 km/s	0.879 km/s	0.816 km/s
Moi Insertion	1.170 km/s	1.539 km/s	1.369 km/s
Aerobraking	0.071 km/s	0.071 km/s	0.071 km/s
TOTAL OIM	**1.957 km/s**	**2.489 km/s**	**2.256 km/s**

Table 2-9 MSO ΔV breakdown - Powered descent

Phase\Launch windows	11 Nov - 6 Dic 2028	2 Dic – 28 Dic 2030	8 Apr – 3 May 2033
Earth Departure	0.761 km/s	0.879 km/s	0.816 km/s
Moi Insertion	1.170 km/s	1.539 km/s	1.369 km/s
Pow. Descent	1.524 km/s	1.524 km/s	1.524 km/s
TOTAL OIM	**3.455 km/s**	**3.942 km/s**	**3.709 km/s**

2.3.2.1.3 Coverage Analysis

For this configuration, a visibility analysis has been performed for two test cases:

1. An equatorial user, at 0° latitude.
2. A near-polar user, at 88° latitude.

Visibility windows have been computed for 7 days, to catch the periodic behaviour of the orbit's ground track and are reported in Figure 2-7.

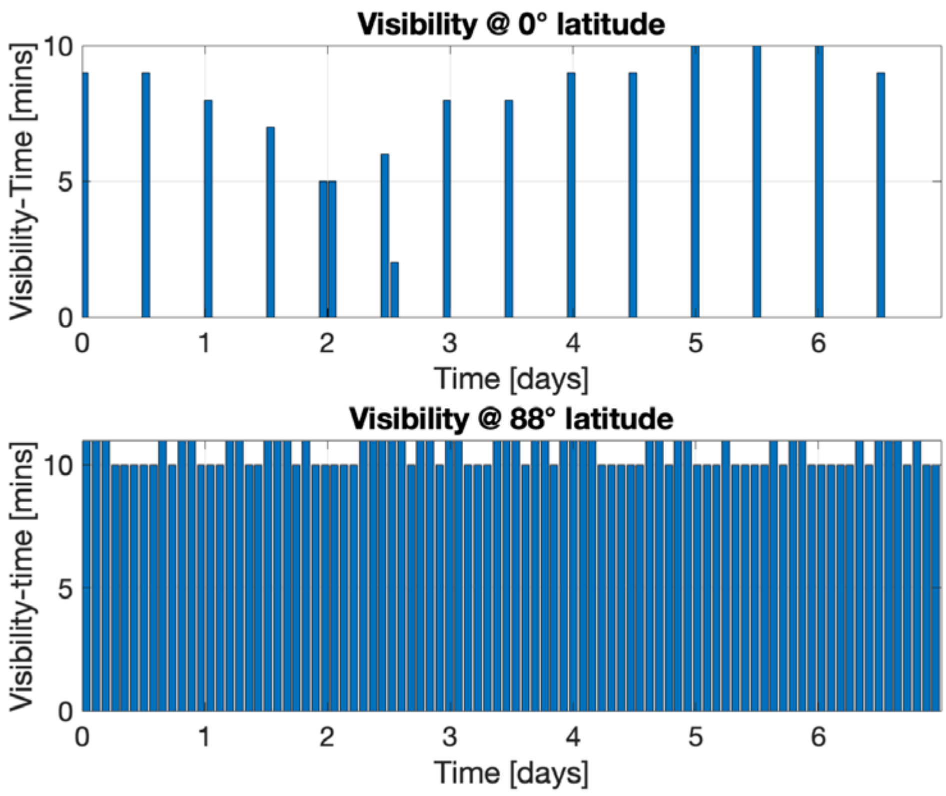


Figure 2-7: Visibility windows duration over time for equatorial and near polar test users.

The windows have been computed considering a minimum elevation angle of 10° and granular data-rate of the communication hardware. This leads to more frequent windows for a near-polar user as expected, with a maximum duration of 12 minutes at most.

2.3.3 Architecture comparison

From the preliminary review of the CDF Report, it is clear that some technically feasible solutions could not meet the cost constraint. At the same time, some cheaper solutions that should fit the cost cap, could not fit all the requirements, in particular those related to the launcher (i.e. MIS-MCC-060 and MIS-MSO-060).

In order to get rid of the problems highlighted in [RD1] and provide a comprehensive overview of possible solutions, we followed a top down approach starting from the mass available at Mars and then looking among the platforms built by TAS in the last years to evaluate their suitability. The outcome of this screening is evaluated for the recurrent cost, completeness and modularity.

2.3.3.1 LAUNCHER EVALUATION

The launcher performances are strongly reduced by requirements MIS-MCC-060 and MIS-MSO-060 that impose a shared launch with Ariane 6.2. On [RD3] there is no indication about the mass ranges of the DLS structure therefore as first evaluation we used a mass allocation of 890 kg; furthermore to be conservative, we considered as target the 50% of the launcher performance including the DLS mass.

It is understood that the scope of requirements MIS-MCC-060 and MIS-MSO-060 is to limit the launcher cost, even if it could pose some risk for programmatics aspects, because following requirement MIS-MSO-070 it is also required a direct launch and for the time being, no other small mission to Mars are foreseen in the international scenario in the next decade.

In order to increase the tight launcher performances and increase the chances of finding a shared launch, we investigate the usage of a shared launch with Ariane 64. It is clear that in this case, due to the high launcher cost, it is not possible to target the same sharing percentage, but a proper scaling could be done.

Figure 2-8 – Launcher cost comparison

As we don't know the exact cost of launchers, we normalize the cost assuming that Ariane 6.4 cost is 1.53 times the cost of Ariane 6.2 and we fix as target cost the 50% of the latter. For this cost ration, the breakeven point sharing percentage of Ariane 6.4 is around the 32.5%, as reported on Figure 2-8.

In order to increase the chances of finding a shared launch opportunity, keeping the same cost allocation for the launcher, we consider in the candidate evaluation both Ariane 6.2 (shared at 50%) and Ariane 6.4 (shared at 32.5%). On Table 2-10 we summarize the target launcher performance for each of transfers described on previous paragraph that are considered in the analysis, including the DLS mass.

Table 2-10 – Target launcher performances (including DLS shared 50%)

[kg]	GTO	GTO+GA	HEO	Direct
Ariane 6.2 [50%]	2250	2250	1650	750
Ariane 6.4 [32.5%]	3737.5	3737.5	2600	2242.5

Table 2-11 – Target launcher performances (without DLS)

[kg]	GTO	GTO+GA	HEO	Direct
Ariane 6.2 [50%]	1805	1805	1205	305
Ariane 6.4 [32.5%]	3292.5	3292.5	2155	1797.5

Table 2-12 – DLS ratio

DLS ratio	GTO	GTO+GA	HEO	Direct
Ariane 6.2 [50%]	24.7%	24.7%	**36.9%**	**145.9%**
Ariane 6.4 [32.5%]	13.5%	13.5%	20.6%	24.8%

It is clear that the DLS mass represent an important percentage of the available mass, in particular for Ariane 6.2, as reported on Table 2-12 where it is reported the ration between the launcher performance available (Table 2-10) and the assumed DLS mass.

For Ariane 6.2 this ration is quite high and represent the main part for the HEO and Direct launch, and consequently for that scenario, the main cost of the launch itself is used for launching on orbit an inert mass.

2.3.3.2 CONFIGURATIONS CONSIDERED

The C3 needed for the Mars transfer could be exploited by the usage of the electric propulsion or by a chemical kick-stage. Limitations in European technology for solar electric propulsion and solar arrays contribute to a required spacecraft mass in excess of what is possible for cheaper, lower mass launch options.

Regarding the chemical architecture, to launch in GTO and ensure a technically feasible spacecraft design, a qualified chemical propulsion kick stage is needed to get onto a Mars trajectory. On this aspect, from Mars Express we know that there are two key aspects:

- To take advantage of the procurement done in parallel with other development phases of concurrent, concomitant missions
- To limit qualification as much as possible, adopting as much recurrent equipment and subsystems as possible.

The two points above also allow, in turn, to fairly limit the project engineering process, as well as allow an implementation fast enough to stay within 4 years from phase B2 KO to delivery.

Differently from the CDF study, we have the opportunity to consider as kick-stage the OIM of MSR mission that presents technical and programmatic advantages. First of all it is a powerful propulsion module, sized in the MSR project to deliver more than 3000 kg of a Main Module in a 4 sol orbit at Mars and can be taken without any modification, leading to a pure recurrent cost. It is the enabler that allows the re-use of Earth platform in the Martian environment with minor changes. The production of this module for this mission could take advantage of the simultaneous development on the MSR mission, empowering the Mars Express approach, because not only the equipment, test benches, AIT, AIV and global development cost could be shared, but also the ESA project office itself could be optimized.

Taking advantage of the OIM as a sort of “off-the-shelf” cargo module that carries a user into Mars orbit, we have the freedom to select a small platform that has the best fit for each the required mission options. A detailed discussion of the OIM characteristics is provided on par. 2.2.2.

The considered MCC configurations are based on chemical propulsion (I_{sp}=320s):

- MCC-A: n°3 identical spacecraft (MCC1, MCC2, MCC3), released separately by the launcher upper stage. Each spacecraft performs each maneuver with its own propulsion system (no stacked cruise to Mars). The operative final orbit is the TASO.
- MCC-B (OIM): n°3 identical spacecraft (MCC1, MCC2, MCC3), mounted on top of the OIM. The stack is released by the launcher and the OIM performs all maneuvers with its propulsion system, including the MOI and subsequent constellation positioning. The operative final orbit is the TASO.
- MCC(Q)-A: same configuration of MCC-A but the operative final orbit is the QSO+HSE
- MCC(Q)-B (OIM): same configuration of MCC-B but the operative final orbit is the QSO+HS

The considered MSO configurations are based on chemical propulsion (I_{sp}=320s):

- MSO-A: n°1 spacecraft (MSO) mounted on the launcher upper stage. The spacecraft performs each maneuver with its own propulsion system. The operative final orbit is the SSO.
- MSO-B (OIM): n°1 spacecraft (MSO) mounted on top of the OIM. The stack is released by the launcher and the OIM performs all maneuvers with its propulsion system, including the MOI and subsequent positioning. The operative final orbit is the SSO.
- MSO-C: n°1 spacecraft (MSO) mounted on the launcher upper stage. The spacecraft performs each maneuver with its own propulsion system. To reach the final orbit, an Aerobraking phase is considered. The operative final orbit is the SSO.
- MSO-D (OIM): n°1 spacecraft (MSO) mounted on top of the OIM. The stack is released by the launcher and the OIM performs all maneuvers with its propulsion system, including the MOI and subsequent positioning. To reach the final orbit, an Aerobraking phase is considered. The operative final orbit is the SSO.

It worth notice that regarding the configurations proposed above, the OIM is quite massive and cannot be used in the Ariane 6.2 mission solutions. It could be utilized in the Ariane 6.4 mission solutions but it could have the best application if the dual launch requirement is removed.

For the propellant evaluation, the following assumptions has been considered:

- When present, the OIM performs all the maneuvers up to the final operative orbit (bipropellant, Isp=320s)
- In the other cases, two solutions are considered:
  - a simple Monopropellant system (Isp=220s), as it is the simplest and cheapest solution
  - a bipropellant system (Isp=320s), in order to reduce the propellant mass size

In some solutions with the OIM, the kick-stage imposes a limitation in the maximum propellant available (1921 kg) and consequently the launcher performance cannot be used completely.
Detailed table of the different configuration is provided on [RD5].

2.3.4 Dedicated launch

The dual launch is a driving requirement that constraint the available spacecraft mass in particular considering the mass of the massive DLS in the 50% of the launcher capability. If a different sharing percentage of the DLS could add more flexibility to find a suitable solution but it shall be evaluated with the launcher authority.

As suggested in the [RD1], a dedicated launch could offer more possibilities to find a proper low cost solution. For this reason, an evaluation of the performance reachable in this case is provided in the next tables.

Due to the risk of exceeding the total cost figure of 250 M€, only Ariane 6.2 dedicated launch is considered.

3 SELECTED ARCHITECTURE

Following the conclusion reported on [RD5], for the MCC case:

- No Ariane 6.2 shared launch feasible solution
- A feasible solution with an Ariane 6.2 dedicated launch should be technically feasible if a Direct launch, with a Kick Stage (OIM recurrent) is selected. Even if the OIM is considered pure recurrent and no modifications of NIMBUS propulsion system is performed, the overall cost largely exceed the cost cap.

Comparing the two solutions with and without the OIM, the former costs the 15% more than the latter, so the MCC case cannot pursued in compliance of all the constraints.

Concerning the MSO Case there are feasible solutions. In particular:

- MSO-C configuration (without OIM)
    - The GTO/GTO+GA, are not feasible due to the very high propellant load needed (comparable to the OIM one)
    - Direct solution has a lower launcher margin but is the cheapest solution
    - The HEO solution offers a good launcher margin (around 58% without DLS) but the higher propellant mass implies important structural modifications.
- MSO-D configuration (with OIM)
    - The Direct solution is not feasible due to the reduced launcher performance
    - The GTO/GTO+GA solutions require a sharing of the main maneuvers between the OIM and the spacecraft increasing the modification on the platform and there is an increased complexity from an operation point of view. Furthermore the GTO solution has been considered to find sharing opportunities but the launcher margins available are well below the required 50%.
    - The HEO solution allows a reduced impact on the propulsion system due to the presence of the OIM that performs all the main maneuvers up to the beginning of the Aerobraking phase.

As backup option, several feasible solutions with Ariane 6.4 shared launch are available, with interesting mass margins, pending the availability of frequent launch opportunities.
In conclusion the MSO Case is feasible with a dedicated launch on a HEO transfer (MSO-D) or on a Direct (MSO-C).
The MSO-D HEO (with OIM) is the proposed solution because even if is not the cheapest one, it is inside the cost cap and could allow a limited modification of the HE-R1000 platform, increasing the robustness of the schedule.

3.1 REFERENCE SCENARIO

3.1.1 Mission Analysis

3.1.1.1 OPERATIONAL ORBIT

The operative trajectory is a Sun-synchronous orbit which ensures a full coverage of Martian surface within 7 days. The orbital parameters are reported in .
Table 3-1.

Table 3-1: Operative trajectory orbital parameters.

Semi-major Axis [km]	Eccentricity [-]	Inclination [deg]	LTAN [h]	Argument of Pericenter [deg]
3716	0.009	92.76	15:00	270

For station keeping, a preliminary value of 2 m/s per year is considered, due to the very low orbital perturbations within the 6 years of the extended mission.

3.1.1.2 TRANSFER AND LAUNCH STRATEGY

The transfer phase starts just after the end of the LEOP and includes 5 main phases:

- **Earth Departure**
- **Interplanetary trajectory**
- **Mars Insertion**
- **Apocenter Lowering**

In the following paragraphs each phase is detailed for the three launch windows between 2028 and 2033. All the major maneuvers involved in the transfer between the Earth and the Operational orbit are performed by the OIM, therefore chemical propulsion only is considered. The most important data adopted during the mission analysis phase are reported in Table 3-2.

Table 3-2: OIM specifications

	Value
Thrust [N]	800
Specific Impulse [s]	320
Max. propellant mass [kg]	1921
Dry mass [kg]	624

Interplanetary Trajectory

The interplanetary trajectory between Earth and Mars consists of one Lambert's arc linking the two bodies, therefore natural motion only is considered in this phase. Some correction maneuvers will be necessary in order to correctly target the arrival asymptote, but they can be typically neglected during this phase. The position of the solar system bodies is computed using analytically ephemerides, valid between 1800 AD and 2050 AD. For Each launch windows considered (namely 2028, 2030/31, 2033), the declination and the C3 required at the escape from Earth and the C3 at the Mars encounter are mapped as function of the departure date and the interplanetary Time of Flight. The results are summarized in Table 3-3. For a complete treatment see [RD6].

Table 3-3 Boundary conditions for escape and departure

Launch Window	Departure C3 [km^2/s^2]	Departure decl. [°]	Arrival C3 [km^2/s^2]
11 Nov - 6 Dic 2028	10.5	40°	9.5
2 Dic – 28 Dic 2030	15	20°	12.3
8 Apr – 3 May 2033	12	40°	11.5

Earth Escape

The aim of the earth escape strategy is to inject the spacecraft in the correct hyperbolic trajectory. The main parameters that characterize the escape hyperbola are the Right Ascension and the declination of the asymptote, as well as the infinite velocity. In particular, the declination of the asymptote and the infinite velocity are fundamental to investigate the ΔV required. The selected strategy for the Earth escape consists of a launch in HEO orbit (900000x250 km @ 6° inclination), followed by a two-manoeuvres escape strategy. Table 3-4 shows the ΔV breakdown for the escape trajectory.

Table 3-4 Earth Escape DV breackdown

LW	Maneuver	ΔV	Gravity Losses Policy	ΔV (with Gravity Losses)	Margin Policy	ΔV(with GL &margined)
2028	1st man.	0.146 km/s	0%	0.146 km/s	10 m/s	0.156 km/s
	2nd man.	0.524 km/s	10%	0.576 km/s	5 %	0.605 km/s
	TOTAL	**0.670 km/s**		**0.722 km/s**		**0.761 km/s**
2030/31	1st man.	0.056 km/s	0%	0.056 km/s	10 m/s	0.066 km/s
	2nd man.	0.704 km/s	10%	0.774 km/s	5 %	0.813 km/s
	TOTAL	**0.760 km/s**		**0.830 km/s**		**0.879 km/s**
2033	1st man.	0.131 km/s	0%	0.131 km/s	10 m/s	0.141 km/s
	2nd man.	0.584 km/s	10%	0.642 km/s	5%	0.675 km/s

	TOTAL	**0.715 km/s**		**0.773 km/s**		**0.816 km/s**

Mars Orbit Insertion

The spacecraft arrives at Mars on a hyperbolic trajectory with a C3 that depends on the exact launch date and on the Time of Flight. The target Mars Insertion Orbit is a 4 Sol Elliptical Orbit, and the capture strategy includes two maneuvers:

- A first strong pericentric maneuver to close the orbit
- A second small apocentric maneuver to adjust the pericenter altitude.

Table 3-5 shows the ΔV breakdown for the Mars capture for all the launch windows.

Table 3-5 Mars Orbit Insertion DV breakdown

LW	Maneuver	ΔV	Gravity Losses Policy	ΔV (with Gravity Losses)	Margin Policy	ΔV(with GL &margined)
2028	1st man.	0.992 km/s	10%	1.091 km/s	5%	1.146 km/s
	2nd man.	0.014 km/s	0%	0.014 km/s	10 m/s	0.024 km/s
	TOTAL	**1.006 km/s**		**1.105 km/s**		**1.170 km/s**
2030/31	1st man.	1.232 km/s	10%	1.356 km/s	5%	1.423 km/s
	2nd man.	0.015 km/s	0%	0.015 km/s	10 m/s	0.025 km/s
	TOTAL	**1.247 km/s**		**1.371 km/s**		**1.539 km/s**
2033	1st man.	1.164 km/s	10%	1.280 km/s	5%	1.344 km/s
	2nd man.	0.015 km/s	0%	0.015 km/s	10 m/s	0.025 km/s
	TOTAL	**1.179 km/s**		**1.295 km/s**		**1.369 km/s**

Apocentric Lowering

In order to contain the time required by the Aerobraking and to reduce the thermal and aerodynamical loads, a first powered descent is introduced. Figure 3-1 shows the ΔV required to lower the apocenter of the MOI, together with the total ΔV required to close directly the orbit on the reduced MOI.

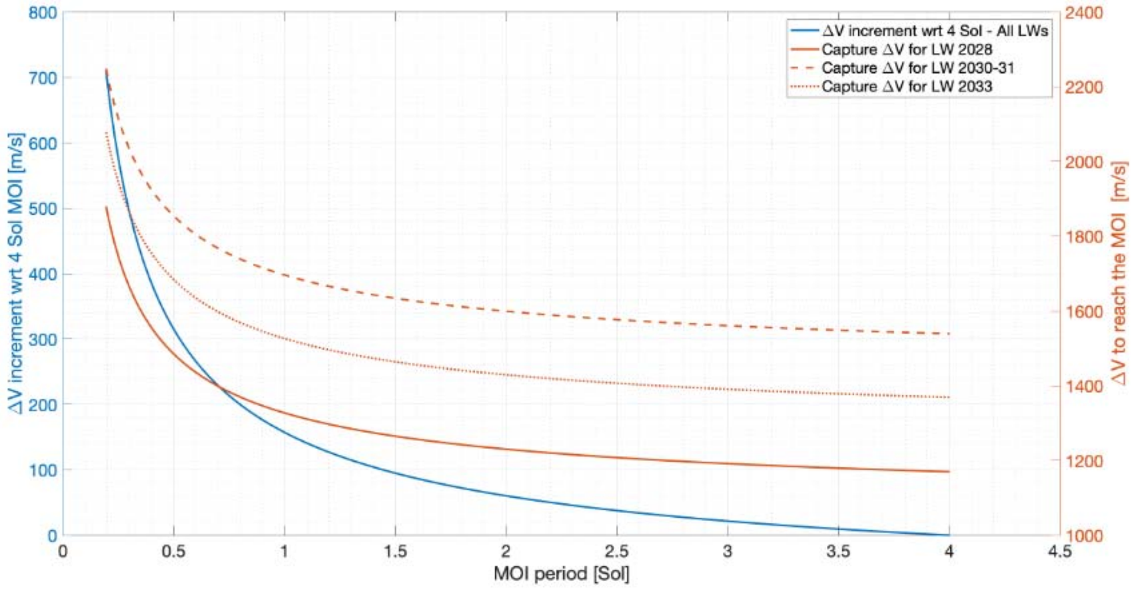


Figure 3-1 Cost to lower the Apocenter of the MOI

The cost to reduce the Period of thew MOI to 1 Sol is reported in Table 3-6. If the maneuver will be performed using the MSO propulsive unit, it can be split in two or more smaller consecutive pericentric maneuvers to contain the Gravity Losses, with no changes in the overall ΔV.

Table 3-6 MOI reduction cost

ΔV	Gravity Losses Policy	ΔV (with Gravity Losses)	Margin Policy	ΔV (with GL &margined)
0.136 km/s	10%	0.149 km/s	5%	0.157 km/s

3.1.1.3 AEROBRAKING ANALYSIS

An aerobraking maneuver is considered to reduce the ΔV costs to reach the operative orbit. To select the initial trajectory to perform the maneuver, three main drivers are considered:

1. Maneuvering time: considered starting from the initial orbit apocenter, up to the final circularization maneuver.
2. Dynamic pressure: the maximum dynamic pressure experienced by the satellite in the atmosphere, that expresses the maximum loads applied to the structure.
3. Convective heat flux: the flux is limited to a maximum value so that the heating of the satellite is controlled, and the temperature of the components is kept inside the appropriate boundaries.

The free variables considered to select the initial trajectory are the target radius of pericenter and the period of the initial elliptical trajectory. In Figure 3-2 maneuvering time, dynamic pressure and heat flux are reported for a set of the previously mentioned free variables. The spacecraft's parameters used for the analysis are reported in Table 3-7.

Table 3-7: Aerobraking analysis parameters.

Mass [kg]	Body Area [m^2]	Solar Panels Area [m^2]	C_D [ ]
768	3.7	9.45	2.2

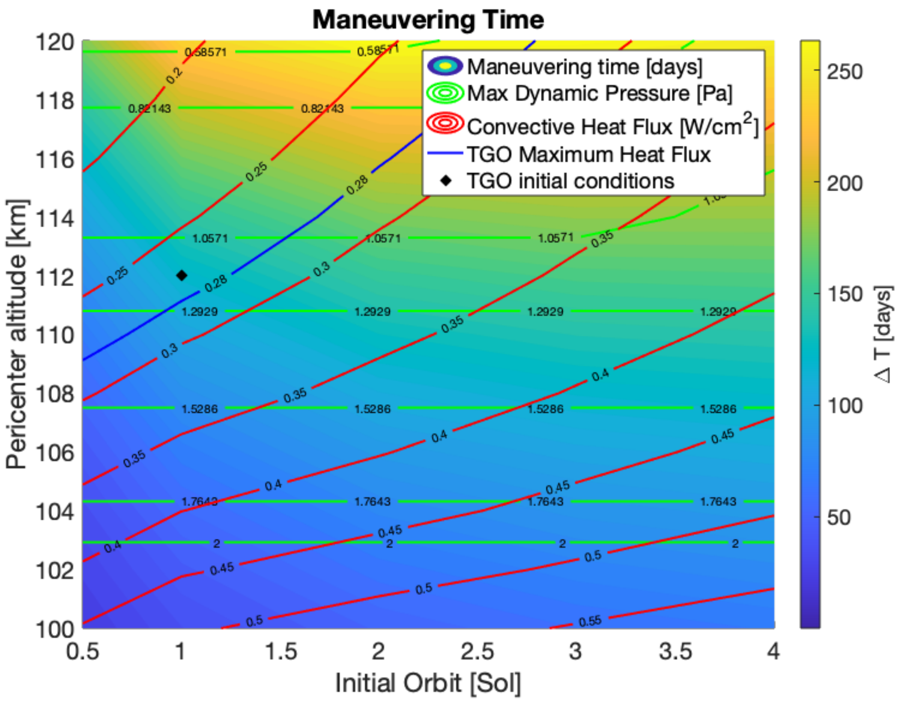


Figure 3-2: Maneuvering time, dynamic pressure, and convective heat flux for the possible combinations of target pericenter altitude and initial orbit period.

Reasonably feasible solutions have a maneuvering time in the order of 6 months to lower the maximum heat flux and dynamic pressure experienced. Indeed, TGO was able to stand a maximum heat flux of 0.28 W/cm^2 and a dynamic pressure of 0.7Pa (to which a safety factor of 150% is applied, leading to 0.28 Pa) [RD11]. Hence, for this analysis, a maximum dynamic load of 0.3Pa is considered. From Figure 3-2, it is possible to see that the maximum heat flux allowed of 0.28 W/cm^2 can be achieved only for target radius of pericenter of ~118km and initial orbit period of 1Sol.

For this condition, the experienced dynamic pressure is higher than the maximum. This is because the pericenter altitude is lowered at the end of the maneuver due to atmospheric drag, increasing the pressure loads. To avoid this situation, the following strategy is exploited:

1. Free motion of the satellite is simulated inside the atmosphere.
2. When the maximum dynamic pressure is reached, a tangential maneuver is performed at the apocenter to rise the pericenter altitude. The final altitude is selected to reach a desired value of dynamic pressure, that is equal to the maximum value multiplied for a scaling factor (set to 0.6 in this case).

The results in terms of dynamic pressure, heat flux and pericenter altitude over time are reported in Figure 3-3 for the case with 118km pericenter altitude and 1Sol period initial orbit.

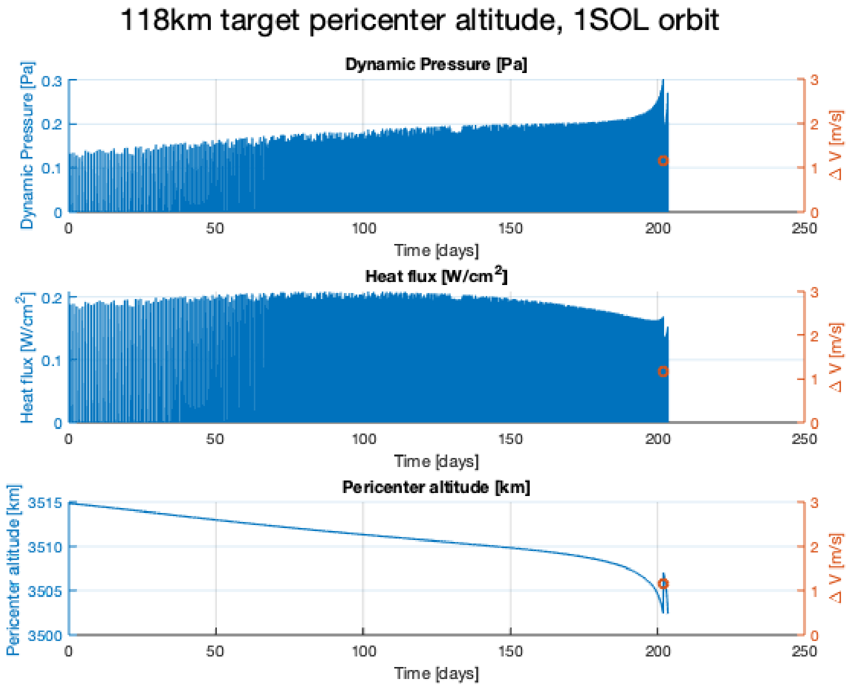


Figure 3-3: Dynamic pressure, heat flux and pericenter altitude over time for the aerobraking maneuver with 118km target pericenter altitude and 1Sol initial trajectory.

As can be seen, the maneuver allows to lower the dynamic pressure to the desired value while staying below the maximum heat flux. In this case, the total ΔV needed is 1.16m/s (without margins), and a single maneuver is required. Note that the temporal distribution of the ΔVs can be set to properly tune the duration of the overall maneuver, while keeping bounded the fuel consumption. Moreover, the target pericenter altitude can be lowered by considering correction maneuvers triggered by the maximum heat flow. In this case, new solutions can be considered as trade-off between the maneuvering time lowering and the fuel consumption increase.

In Figure 3-4 the maneuver with target pericenter of 118km and 4sol initial orbital period are reported instead. In this case, the maneuvers are three, with a total ΔV of 2.396 m/s. As can be seen, the total time is now almost 364 days, much higher than the value reported in Figure 3-2**Error! Reference source not found.**, due to the addition of the corrective maneuvers. Moreover, the constraint of maximum heat flux of 0.28 W/cm^2 is not respected as expected from Figure 3-2. This means that additional correction maneuvers shall be added to take into account also the maximum admissible heat flux.

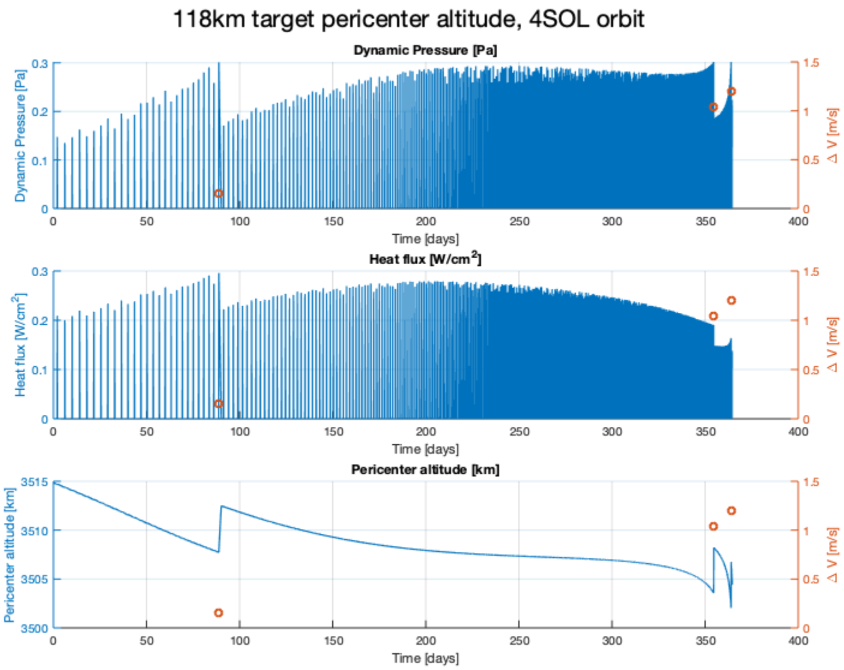


Figure 3-4: Dynamic pressure, heat flux and pericenter altitude over time for the aerobraking maneuver with 118km target pericenter altitude and 4 Sol initial trajectory.

Table 3-8: Aerobraking maneuver ΔV breakdown for the 118km pericenter altitude – 4Sol period maneuver.

Maneuver	ΔV	Gravity Losses Policy	ΔV (with Gravity Losses)	Margin Policy	ΔV (with GL & margined)	TGO
Walk-in	6.0 m/s	0%	6.0 m/s	5%	6.3 m/s	6.6 m/s
Pericenter raising	1.16 m/s	0%	1.16 m/s	100%	2.32m/s	4.55 m/s
Walk-out	60.30 m/s	0%	60.30 m/s	5%	63.48 m/s	31.1 m/s
TOTAL	**67.46**		**67.46 m/s**		**72.1 m/s**	**42.3 m/s**

	m/s					

3.1.1.4 DISPOSAL PHASE

MSO

The operative orbit, characterized by a low altitude, is subjected to a relevant effect from the Martian atmosphere. Consequently, it does not satisfy the minimum requirement of 50 years deorbiting from planetary protection guidelines.
Selecting a higher altitude operative orbit would solve the issue; however, it may introduce performance loss for surface imaging. For this reason, the proposed orbit is maintained as baseline trajectory, and final maneuvers are introduced to raise altitude and enter a disposal trajectory, to fulfill planetary protection requirements. Nevertheless, exploiting the disposal orbit also as operative orbit is still considered an option, as the impact of the low altitude and disposal maneuvers on operations and costs will need to be assessed in forthcoming mission phases.
From Figure 3-5 it is observed that 20 km above the operative orbit, the planetary protection requirement is satisfied, therefore such value is targeted as final disposal orbit.

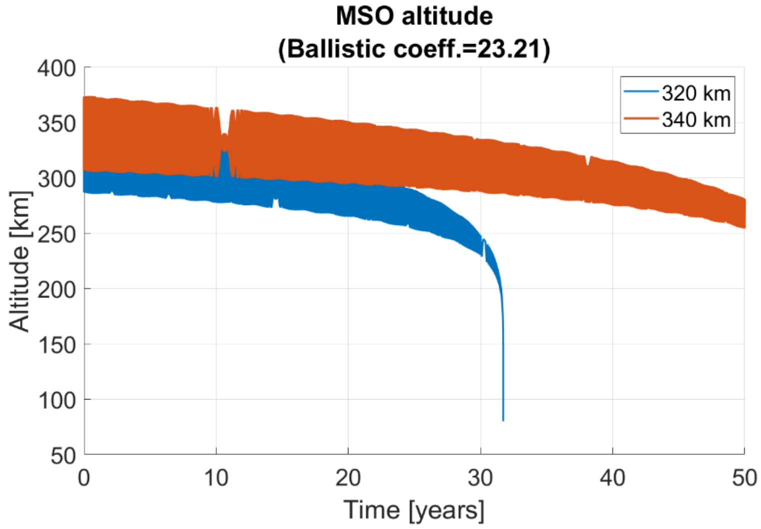


Figure 3-5: Orbit decay as function of mean altitude.

The transfer is modeled as a Hohman transfer, where first the pericenter of the operative orbit is raised by 20 km, then the apocenter is raised by the same value. The two maneuvers cost 4.6 m/s and 4.5 m/s respectively (without margins and gravity losses).

Table 3-9: Disposal maneuvers breakdown

Maneuver	ΔV	Gravity Losses Policy	ΔV (with Gravity Losses)	Margin Policy	ΔV (with Gravity Losses and margined)	Maneuver duration

Pericenter-raise	0.005 km/s	0%	0.005 km/s	10 m/s	0.015 km/s	>500 s
Apocenter-raise	0.005 km/s	0%	0.005 km/s	10 m/s	0.015 km/s	>500 s
TOTAL	**0.010 km/s**		**0.010 km/s**		**0.030 km/s**	**> 1000 s**

OIM

A separate analysis is carried out for the orbital decay of the OIM, to assess the compliance with planetary protection standards. A 4 Sol orbit is considered, with different pericenter altitudes, namely 300 km, 320 km, 340 km.

From Figure 3-6, it can be observed a negligible change in the semi-major axis of the trajectory, regardless of the pericenter altitude. The OIM is verified not to fall into the Martian atmosphere for the next 50 years after Mars's encounter, thus satisfying the requirements.

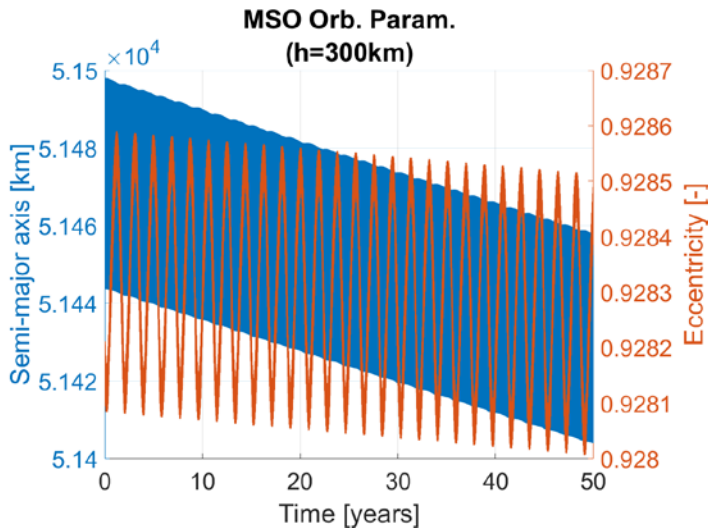


Figure 3-6: Orbital parameters variation of a 4 Sol orbit, with pericenter altitude at 300 km, due to SRP and atmospheric drag.

3.2 SPACECRAFT DESIGN

The MSO is based on HE-R1000 that is a 3 axis stabilization platform that integrates in a single main module all the BUS units, the propulsion subsystem and the payload equipment, including the pertinent appendages. The platform has high pointing accuracy (0.01°) and knowledge (0.003°), and allow high precision position real time knowledge better than 10 m, on 3 axes (3 sigma).

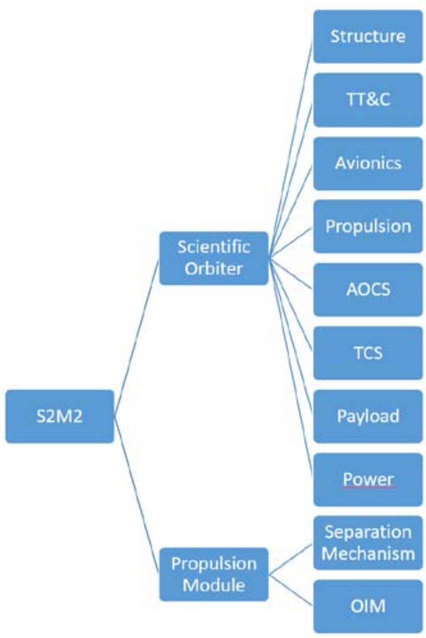
The Propulsion Module is a recurrent of the OIM that will be used in the MSR-ERO mission and will be procured without implementing any modification. The OIM is a bi-propellant MMH/MON-3 system with pressure regulation allowing it to operate both in regulated and blowdown modes.

In order to accomplish the mission the following subsystems are directly linked to specific objective or to critical phases:

- TT&C subsystem, as it is linked to the communications with Earth and to the secondary objective of performing Proximity Link with landed assets on Martian surface and Data Relay to ground.
- Power subsystem, as it has to support the high power consumption required by the TT&C and to provide resources to the payload
- The Propulsion System of the two elements:
  - OIM, as it is responsible to perform the Mars Orbit Injection
  - Orbiter, as it is responsible to support the Aerobraking phase and the Station keeping during the operational lifetime at Mars.

The overall MSO architecture is detailed in the TN3 [RD8] but a summary of the main subsystem is presented in the next paragraphs.

Figure 3-7 – S2M2 Product Tree



3.2.1 Subsystems Design

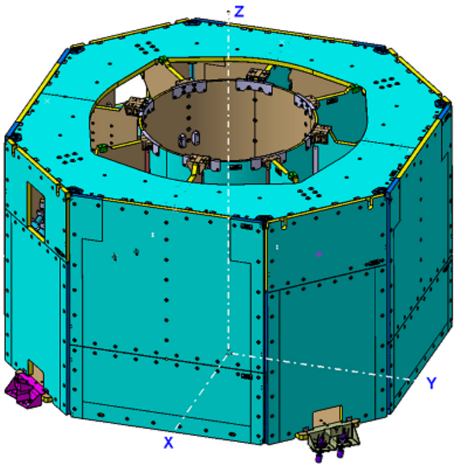
3.2.1.1 STRUCTURE

The structure of HE R-1000 platform is able to sustain the Martian space environment, so at today is not foreseen any modification in terms of materials and/or thickness for the panels that compose the main and the secondary structure of the S/C. For sure, a dedicated support has to be provided for the accommodation of the X-band HGA that would be probably mounted on the +Z panel.

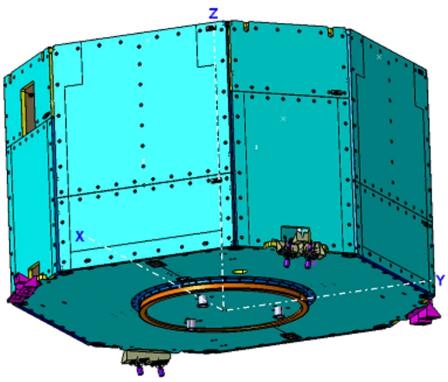
The spacecraft SDM (Structural Dynamic Model) will be qualified within June 2021.

HE-R1000 Satellite Structure is organized according to the classical split on:

- Primary Structure
- Secondary Structure
- Tertiary Structure



View from top



View from bottom

Figure 3-8 - HE-R1000 Overall Structure Configuration

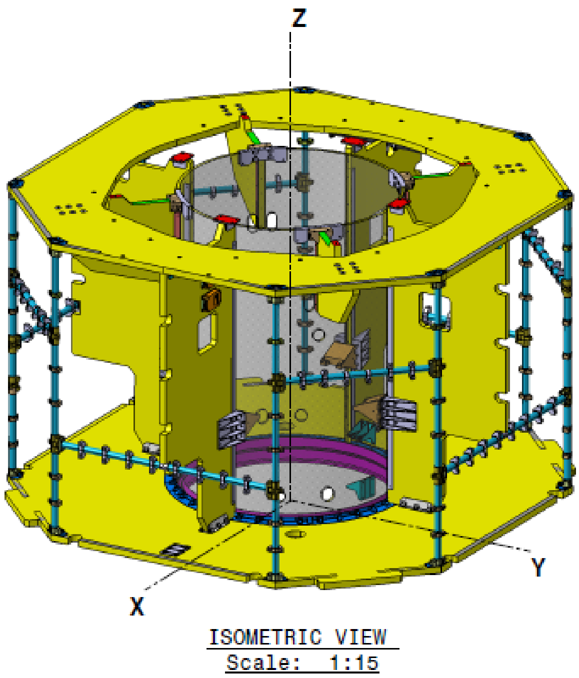


Figure 3-9 - HE-R1000 Overall Structure Configuration (internal)

No specific issues are seen in the accommodation of the mission dedicated P/L (units and antenna) even considering Custom tertiary structures to interface for instance the HGA.

About the software, existing "building blocks" will be used but a mission customization dedicated to the attitude control laws and sensor selection/usage should be necessary.

For the interface with OIM, the geometry of the current HE-R1000 CFRP cylinder is 937mm that is different from the baseline OIM interface ring that is 1194 mm. Consequently a transition cone shall be included in HE-R1000 and added to the mass budget.

The Tertiary Structure mass reported above is already considering 3 kg for the HGA support. In addition to those values, it's necessary to consider:

- 15 kg for the separation system (clamp band + separation spring);
- 20 kg for the interface adapter between OIM and HE R-1000.

3.2.1.2 AVIONICS

Regarding the Avionic hardware, the IPAC based architecture match the nedd of this kinf of mission.

3.2.1.3 AOCS

Regarding the AOCS, as preliminary assumption we consider the same used on HE-R1000 platform with the inclusion of a IMU that is mandatory for aerobarking, and the RW of 20Nms (max torque 0.215Nm). During the cruise, the platform commands the RCS on the OIM together with its own AOCS. After the jettison of the propulsion module, the platform AOCS is in charge of the attitude control up to the disposal.

3.2.1.4 TCS

Regarding the thermal control, as preliminary assumption we consider the same used on HE-R1000 platform, as well as for the OIM.

3.2.1.5 POWER

The heritage of the equipment part of the power subsystem are relevant to the HE-R1000 platform and the project relevant to the Main Propulsion module. Furthermore, the equipment that will be added to the previous main items (SADM, yoke and DC/DC converter) are those relevant to the application of the off-the-shelf (i.e. Exomars).

In order to provide the right orientation of the 2 solar array wings, in its preliminary configuration HE R-1000 needs to accommodate 1 SADM, 1 yoke and 1 hinge for each wing. As a reference, we can assume various models of SADM, i.e the one mounted on Exomars or the one mounted on Iridium Next.

3.2.1.6 TT&C

The Telecommunication subsystem has been designed for both the options: constellation and orbiter. For both of them, the design trade-offs and alternatives have been presented, with the aim to conjugate the maximization of the coverage of the needed performances for each mission, and the minimization of the costs.

The starting point for this design was the platform baseline telecommunication's capability. Indeed, both the considered platforms (NIMBUS and HE-R1000) have a TT&C S/S to cover the needs of a LEO satellite, i.e. requiring only one bidirectional link with Earth, at very high data rates (up to 10 Mbps) over short distances (thousands of km against 2/3 AU needed for Mars).

On the other hand, for the studied Mars missions, data relay is defined as the set of two full duplex communication links:

- Direct to Earth link (DTE), from S/C to Earth ground and vice versa
- Mars Proximity link, from S/C to Mars and vice versa

To the above links, an inter-satellite link (ISL) has to be added with a "mothership-children" configuration or can be added to improve Ground Control flexibility.

The conclusion of this comparison was that the platforms' TT&C S/S shall be heavily re-designed in order to respond to the necessities imposed by the Martian environment and by the mission architectures.

Generally speaking, the main choices to be performed on the TT&C subsystem, in order to assess the best architecture, are related to:

1. <u>Bandwidths to be used for the different links</u>

   The choice of the bandwidth will be driven by different factors:

   - Units' reusability and adaptation to existing assets

  - o Conformity to applicable standards
  - o Optimization w.r.t. the foreseen scenarios and environment
  - o Optimization of the on-board resources and management of multiple links in parallel, if needed

   The baselined choice was the one maximizing reuse and compatibility:
   - DTE in X band
   - Proximity in UHF band
   - ISL in Ka band (only for MCC)

2. <u>Redundancy policy</u>
   Usually, as the communications' function is essential for the mission, the related subsystem is designed at least to be single-point failure tolerant. This usually implies full redundancy of the transponder, amplifier, filters and all the critical units, ending up in duplicating the TX/RX chains up to the antennas.
   However, the constellation topology might use a "redundancy by spacecraft" policy, embarking a single TT&C chain per satellite, or on the children. In case of failure of a string, then the communications are however ensured by one of the other constellation satellites, even if a degradation of surface coverage has to be taken into account (1/3 in case of three satellites). The choice of the single-string configuration in this case is considered a good mass and cost saving solution (of course, only for the constellation option).

   The baselined choice was:
   - DTE in hot redundancy, two complete chains
   - Proximity in cold redundancy, only one antenns

3. <u>Bent pipe vs. Store&forward</u>
   The management of the data on board can be done in real-time (bent pipe) or managing delays (store and forward). The choice between the two options has implications on the On Board Data Handling and on the Satellite operations.
   The baselined choice was Store and Forward which minimizes operational constraints.

4. <u>Pointed or fixed antennas</u>
   Active Pointing Mechanisms have usually high mass and cost, especially the one for big and directive antennas. On the other hand, avoiding these units implies the need to point the S/C in order to align antennas for the link establishment and maintenance. This has impacts on the power S/Sand the AOCS.

   The baselined choice was to have an APM only for the HGA to minimize operational and AOCS constraints.

5. <u>Integrated transponders vs. stand-alone transponders</u>
   Single transponders, one for each link and working in the related bandwidth, allow higher reuse. On the other hand, duel band integrated transponders are under development, implying a mass saving. At the cost of a lower TRL, a potential cost saving can be also evaluated. Furthermore, having integrated transponders implies an advantage for the OBDH, especially in case of bent and pipe option.
   The baselined choice was to have separated transponders in order to maximize TRL and minimize costs.

In the following sections, a more detailed description of the subsystems covering the above three links is reported. Related budgets (only for the Orbiter option) are reported in sec. 3.4.5

3.2.1.6.1 DTE link – X band

The DTE link is designed in X band (around 8GHz) as per ECSS rules. It involves just the physical layer, while the coding/data layer is usually part of the OBC design.
DTE link is also used for radiometrics, providing support to Doppler and ranging services and DELTA-DOR for more precise measurements.

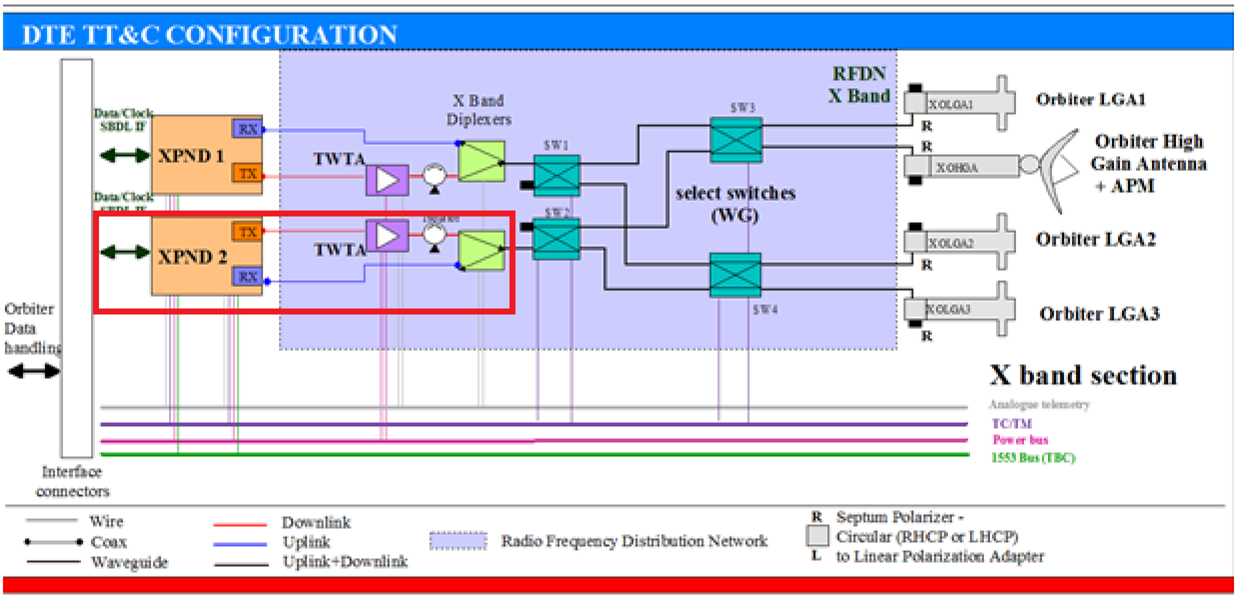


Figure 3-10 - DTE TT&C architecture for Mars applications

Figure 3-10 shows the designed full redundant DTE system, from TGO spacecraft. The architecture includes:

- 2 transponders for modulating, demodulating functions and radionavigation functions
- 2 external power amplifiers (TWTA)
- Radio Frequency Distribution Network to connect the transponders to the antennas, i.e. filter, switches and Wave Guides
- 3 Low gain antennas, for communications during LEOP, contingency, manoeuvres.
- 1 High Gain Antenna, for nominal (high data rate) communications and science operations, with active pointing mechanism

The TRL associated to the DTE equipment is the following:

- 2 X-DSTs: TASI has developed all transponders for ESA missions, for example observation satellites like Euclid, or deep space missions as the TGO. These XDSTs (TRL 9) already implement all features needed for Mars scenario.
- 2 TWTAs: the current heritage based on TAS-B design allows to choose the RF power up to 100-120W RF without requalification (TRL 9).
- RFDN: TAS-E is one of European leaders in RFDN design. The RF components are potentially available and qualified from Exomars (TRL 9), while NRE activity is foreseen for the routing of the waveguides (as per all spacecrafts)
- RHCP LGAs: low-mass, off-the-shelf units are already available from EXM TGO and RSP (TRL 9).

- HGAA (Antenna + Pointing mechanism): the RF part (2.2m reflector, feed) and pointing mechanism is available with TRL 9 from ExoMars, but some NRE activities can be foreseen to adapt the diameter size and qualify the unit to meet the thermal and mechanical environment at assembly level (in case of antenna + pointing mechanism).

3.2.1.6.2 Proximity link –UHF band

The link with Mars surface, named Proximity, is designed according to CCSDS 211.B Proximity-1 protocol, and is currently employed in UHF band (350-450MHz) in the existing Mars relay Network as physical/data layer between orbiters (MRO, MAVEN, TGO, ODYSSEY, MEX) and landers. The UHF link performances are very robust w.r.t. signal reflections and multipath due to Mars environment, and also w.r.t. the Mars atmospheric effects; on the other hand, they are highly affected by orbiter's orbit, in particular the orbiter altitude determines the visibility times and slant range, both affecting the overall data volume.

However, the current studies for the lunar environment start foreseeing also S-band for the proximity link. S-band is more sensitive to atmospheric and path losses, but on the other hand the involved antennas are smaller and lighter. Another current drawback is that actually Mars assets implement only UHF proximity links, so that moving to another band would imply reduced adaptability with the existing environment.

Figure 3-11 shows the designed architecture of a UHF Proximity link, which is made up by:

- 2 UHF transceivers, which implement all baseband and RF functions, including RF filtering. All the most recent Mars orbiters (MRO, MAVEN, TGO) have the P1 unit based on ELECTRA payload, which therefore represent the term of reference in terms of heritage (TRL9). No European alternative is currently available concerning UHF transceivers for orbiter applications with High TRL, but one is under development (TRL 3) and will be available for the mission.
- UHF antennas and cables (typically a fixed antenna) (TRL9). Concerning antennas, TAS has developed a full range of antennas for orbiter and lander applications in the frame of Exomars, optimized for large coverage (+/- 65° off-boresight for orbiter application). Optimization of the UHF LGA pattern is then still possible, as the altitude of the S/C implies that the needed field is restricted to about +/-10°. An improvement of about 2-3dB is expected with comparable mass and cost. TRL9. Cables and switches are fully available with TRL9, but, as usual, routing and detailed design will have to be tailored to the specific craft.

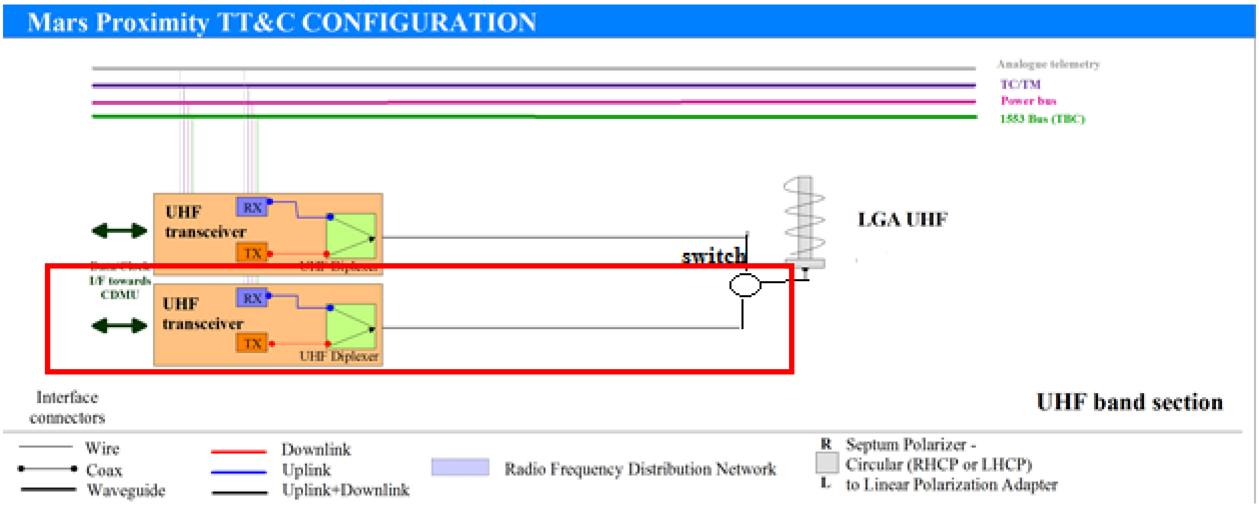


Figure 3-11 Typical TT&C architecture of Mars Proximity links

3.2.1.6.3 ISL – Ka band

For the constellation case, the most promising configuration of the single crafts is the “mothership-children” one. In this view, there is only one satellite equipped with X-band TT&C subsystem, able to communicate with Earth, while the other satellites only embark an inter-satellite link with which they send to the mother craft all the data gathered from the assets, and from which they receive the commands which shall be distributed to the Mars surface via UHF. This design allows the “children” to avoid mounting the X-band subsystem, especially the HGA, the accommodation of which are demanding for the small platforms. The “children” will mount only the ISL units (transponder, amplifier and small low gain antennas), and the redundancy of those subsystem can be the one at constellation level (i.e. a single string per each satellite). On the other hand, the “mothership” will have to be equipped with the suitable redundancy for the DTE subsystem: double chain as detailed in the previous section.

The ISL can be mounted, of course, also in the option of equal satellites. Indeed, the Mars full coverage requirement implies the will to allow a generic Mars asset, placed anywhere on the surface, to communicate with Earth anytime. In this case, if the orbiter in sight has no contemporary visibility with Earth, if another satellite of the constellation does, an ISL can allow to use the orbiter as relay and successfully deliver the data to Ground. The ISL is mandatory in case, for cost saving, the orbiters don’t provide the on board mass memory.

According to the Space Frequency Coordination Group, the recommended frequencies to be used for ISLs are S- and Ka-, the latter being more suitable for higher distances (which are compatible with a constellation with a low number of satellites) and data rates. Ka-band will be the standard frequency used for the incoming development of the lunar communication systems (i.e. Gateway), so at the time of development of Smarties, there will be available COTS units to use for the ISL (among them, the TASinI IDST as described here below will exploit the 25.5 – 27.0 GHz BW, with similar mass, envelope and power consumption of the X-band units, and also the TASinI KBT will be available).

The ISL S/S has to include also a signal amplifier (TWTA) to ensure coverage for high distances avoiding the need of big antennas with high gain. A majority of spacecrafts operating in Ka-band currently already rely on Thales TWTs.

Two Medium Gain Antennas (gain up to 25 dBi) will have to be also mounted in order to ensure omnidirectional coverage. Ka-band dishes antennas are relatively small and light: Weight: 2kg, diameter < 20 cm. Also horn antennas can be used for this link, they can be derived from the Cubesat heritages.

3.2.1.7 PROPULSION

During the first analysis carried out in the evaluation of the propulsion subsystem for the MSO, a baseline blowdown configuration was considered with a tank capacity ranging from 35L up to 160L. The HE-R1000 platform satisfies these preliminary requirements, and the final propulsion architecture is reported here below:

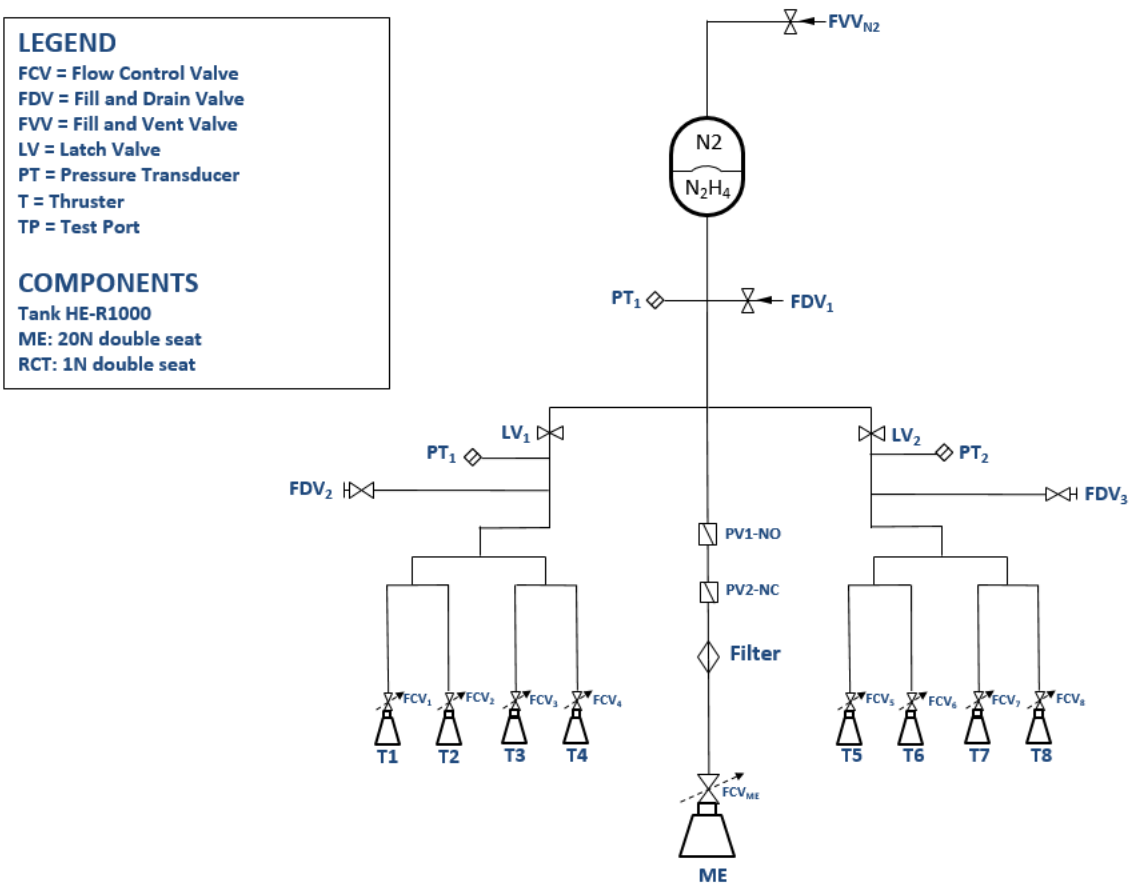


Figure X: MSO HE-R1000 propulsion subsystem configuration

The use of hydrazine and nitrogen as propellant and pressurant respectively deals with a full compatibility of the subsystem COTS units. The 135L tank is inherited by the platform, and the main difference of this configuration from the HE-R1000 original one is given by the 20N ME additional branch. The ME ensures a correct execution of the maneuvers during the A/B phase, and the 1N RCTs allow the management and control of the platform attitude during the station keeping operations. All the RCS thrusters are operative in order to maintain the platform stability along the three axis and guarantee a high pointing accuracy during the AOCS operations. A three isolation barriers level is ensured between the propellant tank and the thrusters.

3.2.1.8 Kick-Stage

The OIM is used as full recurrent and it is described on par. 2.2.3.3. It is equipped with its thermal control and it is interfaced with the upper module with a standard MIL-1553 bus. The power bus voltage is 100V and will be connected with a DC/DC converter on the SO.

The mechanical interface and the separation mechanism are described on par. 3.2.1.1.

3.2.2 TRL

No critical technologies are considered and all the equipment used has high TRL as reported in the following table.

Table 3-10 – TRL Table

ORBITER	S\S	Acronym	Description	Quantity	TRL	Supplier	Supplier Country	HE-R1000 equipment
	STRUCTURE			1				
			Primary Structure		7-9	TAS-I	Italy	✓
			Secondary Structure		7-9	TAS-I	Italy	✓
			Tertiary Structure		7-9	TAS-I	Italy	✓
			Miscellanea		7-9	TAS-I	Italy	✓
	TCS			1				
								TBC
	PROPULSION							
		TNK	Tank	1	9			✓
		FDV	Fill and Drain Valve	3	9			TBC
		FVV	Fill and Vent Valve	1	9			TBC
		LF	Line Filter	1	9			TBC
		LV	Latch Valve	2	9			TBC
		PV	Pyro Valve	2	9			TBC
		PT	Pressure Trasducer	3	9			TBC
		PIP	Pipes	N/A	9			TBC
		ENG	ME 20N (optional)	1	9	Ariane Group	France	No
		ENG	RCT 1N	8	9	Rafael	Israel	✓
	HARNESS							
		HRN	Harness	1				
	EPS							
		BTA	Battery Assembly	1	8	ABSL	England	✓
		PCDU	Power Control & Distribution Unit	2	7	TAS-B	Belgium	✓
		SADM+Yoke	SA Pointing Mechanism	2	7-9	RUAG	Switzerland	No
		SAW	Area per wing 4.77 m2	2	7-9	STI	Germany	✓
	AVIONICS							
		IPAC	Integrated Power Avionic Communication	1	7	TAS-I	Italy	✓
		STT	Star Tracker	4	7	Leonardo	Italy	✓
		IMU	Inertial Management Unit	1	7	TBC	TBC	No
		FSS	Fine Sun Sensor	4	7	Solarmems	Spain	✓
		RW	Reaction wheels	4	7	Rockwell Collins	Germany	✓
	TT&C							
		X-DST	[DTE] Transponder	2	9	TAS-A	Italy	No
		TWTA	[DTE] Amplifier	2	9	TAS-B	Belgium	No
		HGA	[DTE] High Gain Antenna	1	9	TAS-E	Spain	No
		HGA APM	[DTE] HGA Pointing Mechanism	3	9	Sener	Spain	No
		RFDN	[DTE]	1	9	TAS-I	Italy	No
		LGA	[DTE] Low Gain Antenna	1	8	Ruag	Sweden	No
		Transceiver	[Proximity]	2	9	NASA-JPL	USA	No
		LGA UHF	[Proximity]	1	9	Sener	Spain	No
	PAYLOAD							
		PAY	Payload allocation	1				-
	SEP. MECH.							
		Sep. Mech	Separation Mechanism	1	7-9	Ruag	Switzerland	No
OIM	S\S	Acronym	Description	Quantity	TRL	Supplier	Supplier Country	HE-R1000 equipment
	I/F			1				
		TRST	Transition Structure		7	TAS-I	Italy	No
	OIM			1				
		OIM	Orbit Insertion module (recurrent)		9	TAS-I	Italy	No

3.3 PRELIMINARY CONOPS

The preliminary S2M2 Concept of Operation, as described in the following paragraphs, is developed keeping in mind that the spacecraft operations are identified as a significant cost driver for small satellite missions.
In particular the reduction of the SC contact windows, the increase of the SC autonomy and the reduction of the critical mission phases (in particular the Aerobraking phase) are identified as the main areas in which the ground segment activities can be reduced for budget purposes.

It shall however be clear that any difference with respect to the classical approach shall be further analyzed, to evaluate the balance between the reduced operational costs and the possibly increased complexity with the related additional development costs. An additional point to be further evaluated is the development of a 2 level (fail-safe / fail-op) FDIR system for the management of the failures in the time critical mission phases. This point is further elaborated in paragraph 3.3.5. In any case a trade-off to evaluate the best balance between the two elements (increased autonomy vs. ground driven operations) can be performed only with the support of the Agency and/or the designated mission operations center.

3.3.1 Mission Phases

The S2M2 science orbiter will conduct a scientific mission on Mars lasting 4 years as minimum as reported on MIS-MSO-030, providing support for relay communications from surface assets on Mars (MIS-MSO-050).

3.3.1.1 EARTH DEPARTURE

The Earth departure phase will last around 40 days. The spacecraft is injected by the Ariane 6.2 launcher in a High Elliptical Orbit (900000km x 250 km) around Earth with a 6° inclination. The spacecraft shall perform a change plane and a pericenter maneuver to reach the desired energy. This preliminary phase require the Ground control to carefully evaluate the departure conditions and implement the corrections needed. The total ΔV required is smaller with respect to GTO departure.

It will include the Launch and Early Operations phase, the SCC commissioning and the 2 deterministic trajectory correction manoeuvers.

The launcher will release the SCC (SO + OIM) into an Earth escape orbit. All the subsystem required for the transfer will be checked through dedicated commanding and TM check sequences.

3.3.1.2 EARTH MARS CRUISE

After the end of the commissioning phase and the execution of the second trajectory correction manoeuver, the SC will start the 1 year long transfer to Mars. During this phase no deterministic manoeuvers are planned, and the SC is generally stable in its configuration.

The nominal attitude is controlled through the RWLs, the communication is granted through the LGA / HGA antennas.

Two checkout slots are planned: the first one 6 months after the start of the cruise phase, the second one before the arrival in the Mars.

3.3.1.3 MARTIAN PHASE

3.3.1.3.1 MOI

The insertion in an elliptic Martian orbit is obtained through a Mars Orbit Insertion manoeuver. Such manoeuver is split in two main events: a first pericentric manoeuver (1 hour of burn is currently planned) followed by a coasting lasting more than 2 sols, and a second apocentric manoeuver (also in this case a burn of 1 hour is expected).

3.3.1.3.2 Kick Stage Jettison

With the execution of the MOI, the OIM completed its main task and is jettisoned from the SC on a safe orbit compliant with the Planetary protection requirements.

3.3.1.3.3 Aerobraking

Once the OIM is jettisoned, the SC can start the Aerobraking phase which will allow the S2M2-SO to reduce the altitude of the apocenter and to reach its final scientific orbit.

The S/C orbital energy, period and apoapsis altitude are progressively lowered by means of the drag force acting mainly on the S/C Solar Array panels (the HGA will remain in the re-stowed position, so it will not be providing drag surface participating to the AEB), as the S/C passes through the upper layers of the Mars atmosphere.
The duration of the aerobraking phase depends on the ballistic coefficient of the spacecraft. Such coefficient depends on the SC attitude and in particular on its solar panels. For this reason the coefficient has not been frozen and the duration of the aerobraking can vary from a minimum of 10 days up to a maximum of 6 months.

A walk-in manoeuver is needed to lower the pericenter to the desired value.

A walk out manoeuver is needed at the end of the aerobraking phase to increment the pericenter to the operative orbit. Both the walk-in and walk-out manoeuvers are performed through the RCS system of the S2M2-SO.

During the aerobraking phase, the HGA is kept in stowed position, with the communication with Earth exploited via a guidance profile to be computed in such a way to position the S/C in order to guarantee the proper antenna orientation in the visibility phase.

3.3.1.4 OPERATIVE ORBIT

The operative orbit is a Sun-synchronous orbit with an inclination of 92.76° which ensures a full coverage of the Martian surface within 7 days. The altitude is 320km, with an apoares of 352.4 km and a periares of 285.6 km (e=0.009)

At the end of the mission the SC shall guarantee to not deorbit into Mars for at least 50 years. The operative orbit, being characterized by a low altitude, is subject to a relevant effect from the Martian atmosphere and it does not satisfy the planetary protection requirement.
For this reason a disposal maneuver shall be performed to increase the altitude of 20 km and ensure 50 years of orbital decay before its final atmospheric re-entry.

3.3.2 Ground Segment

It is assumed that the ground segment will use the following ground stations:

- 15m ground station for Launch, LEOP and commissioning
- 35m ground stations (ESTRACK from ESA and DSN from NASA) for all nominal cases and safe mode
- DSN 70m antenna (TBC, for not nominal SC emergency cases)

The ground stations coverage, on the basis of the ExoMars experience, will foresee a low level support during the transfers and will be increased for particular critical phases such as commissioning or manoeuvers.

It is assumed that the expected ground contact windows will change during the mission:

- One 8h-long Ground Station contact per week (TBC) during Interplanetary Cruise Phase
- One 8h-long daily Ground station contact during orbits around Mars
- Up to 24h a day coverage in preparation of the orbital manoeuvres and critical phases (LEOP, MOI, OIM jettison)
- One 45 minutes TBC contact for each aerobraking pass
- One TBD minutes contact per day for the scientific phase

3.3.3 System Operational Modes

The following picture reports the proposed approach for the S2M-SO operational modes and the allowed mode transitions.

The red lines represent automatic transitions triggered onboard. The automatic transitions can be triggered by OBC driven FDIR events (transitions to SAFE mode) or by specific onboard events (e.g. separation from the launcher). The green lines the transitions triggered by ground telecommands and the grey ones the transitions performed autonomously onboard.

The system mode logic is implemented in the onboard software executing within the OBC. The software contains all general system services and a range of applications for functions (such as for example thermal control, attitude control, etc.).

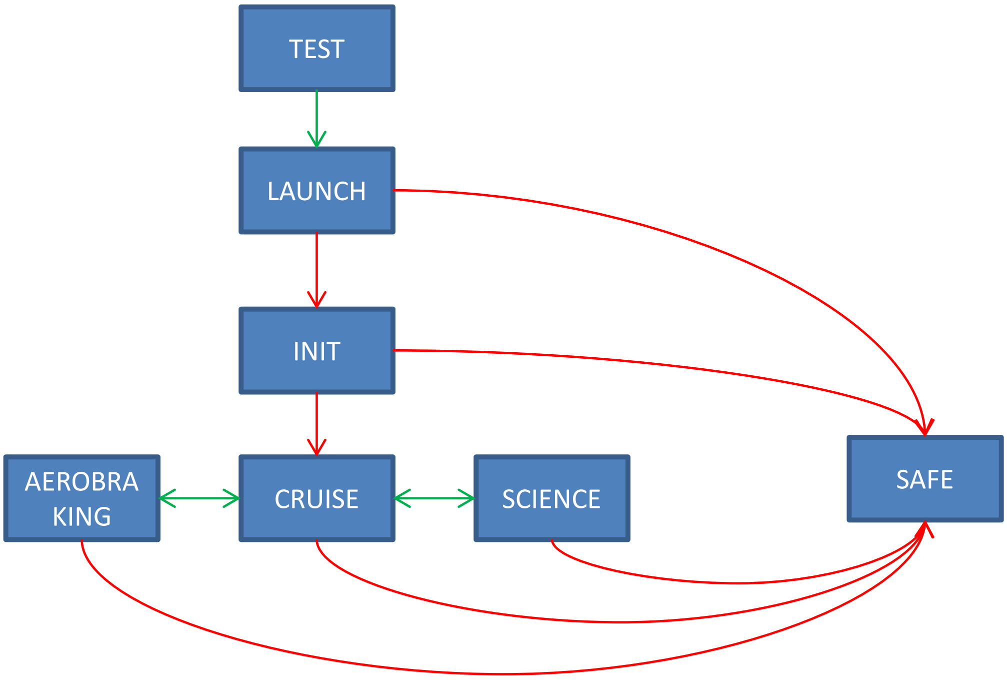


Figure 3-12 – System modes transitions

TEST MODE
It is a ground only mode, designed specifically for testing purposes. Only a subset of the avionic units is ON. The SC is ready to receive TCs and it generates TM packets as soon as the SW is up and running.

LAUNCH MODE
Launch mode is a pre-separation idle sub-mode maintaining the spacecraft passive on the launcher. When the separation is detected, the system mode manager switches automatically to the INIT Mode.

INIT MODE
The INIT mode is triggered at launcher separation. The spacecraft performs autonomously the reconfiguration sequence that prepares the SC to acquire a safe Sun pointed attitude and to receive further commands from Ground.
The exit from this mode is performed by Ground after the completion of the automatic activation sequence with a transition to Nominal mode.

A fail-Op strategy might be applied to ensure the execution of this sequence even in case of anomaly. Refer to paragraph 3.3.5 for further details.

CRUISE MODE
Cruise mode is the nominal mode for the transfer orbit. The SC attitude is controlled by the RWLs and the RCS, as needed. The trajectory correction manoeuvers are performed in this operational mode.

The AOCS actuation systems will be driven by the AOCS state machine which will be defined later in the project. In addition, for the delta V manoeuvers, two different orbital control modes for the normal manoeuvers and the critical manoeuvers (e.g. MOI) shall be defined, if a 2 level FDIR system is implemented, as described in paragraph 3.3.5.

AEROBRAKING MODE

Aerobraking mode is entered from cruise mode by Ground command. The purpose of the aerobraking phase is to reduce the energy of the highly elliptical orbit achieved by the propulsive planetary capture, transforming it into a low-eccentricity and low-altitude orbit by successive atmospheric passes.

In this mode a high level of autonomy is implemented onboard. Automatic means will be developed to guarantee safe SC operations during the atmospheric passes as already successfully implemented in the ExoMars TGO, e.g.:

- Automatic detection of the atmosphere
- Periapsis time estimator
- FDIR commanded pop-up manoeuvers (FDIR commanded or Ground commanded).

SCIENCE MODE

Science mode is entered by Ground command from cruise mode. The UHF communication system and the payload are active.

SAFE MODE

The safe modes is as much independent as possible from the previous spacecraft history to ensure the isolation against potential transmission of previous failure cases. SAFE mode will allow to transmit to Earth either with the HGA or with the LGA. The SAFE mode can be entered by an automatic FDIR driven transition or by Ground command.

When a SAFE mode transition is triggered the SC resets the command queue, switches off all the non-permanently powered units, points the Sun in order to generate power and resets the communication to Ground, ensuring uninterrupted power supply and safe thermal conditions.

In SAFE mode the attitude control is performed through the RCS attitude control thrusters, and if possible through the RWLs.

3.3.4 Operational Timeline

The following timeline describes the main mission events for each one of the mission phases. The dates included in the table are referred to the three analysed launch opportunities:
2028: 11 November - 6 December 2028
2030: 2 December – 28 December 2030
2033: 8 April – 3 May 2033

Table 3-11 – Mission timeline

Event	Duration	Description	2028	2030	2033
Earth departure	40 days		06/12/2028	28/12/2030	03/05/2033
Launch MSO+OIM					
LEOP	2 days				
SC commissioning start		1 month of SC commissioning (TBC)			
Coasting	20 days				
MAN#1		Apocentric manoeuvre			
Coasting	20 days				
SC commissioning end					
MAN#2		Pericentric manoeuver			
Interplanetary transfer	1 year		15/01/2029	06/02/2031	12/06/2033
Interplanetary trajectory		No deterministic manoeuvers are expected			
SC checkout					
Mars injection	9 months	(9 months figure includes the aerobraking)	15/01/2030	06/02/2032	12/06/2034
MOI pericentric manoeuver	1h				
Coasting	>2 sol				
MOI apocentric manoeuver	1h				
Coasting	>2 sol				
OIM separation					
Walk-in manoeuver	600s				
Aerobraking					
Aerobraking commissioning					
Aerobraking	<6months	Lower the apocenter			
Walk-out manoeuver	2000s	Final positioning in SSO (300km altitude)			
Science phase	6 years		15/10/2030	06/11/2032	12/03/2035
Science phase commissioning	TBD weeks				
Science phase execution	4-6 years				
Station keeping manoeuvers		Sun-synchronous orbit station keeping			
SC disposal	>50 years		15/10/2036	06/11/2038	12/03/2041
Pericenter raising	500s				
Coasting	1h				
Apocenter raising	500s				
Orbit decay	50 years				
Atmospheric re-entry					
End of mission					

3.3.5 Autonomy Concept

The spacecraft FDIR design will support the SC operations and autonomy in order to maintain quiet and stable conditions during the mission, put the spacecraft in safe conditions in case of major anomaly and maximize the probability of success of maneuvers (e.g. MOI) and mission critical phases (e.g. OIM jettison and science).

The commandability of the S2M2-SO mission is affected by the long time required by a TC to travel from Ground to the satellite and to get the response telemetry back (up to 15-20 minutes when close to Mars). This of course impacts the operational approach and the required autonomy of the SC, since no recoveries can rely on Ground intervention even when the SC is visible from Ground, in particular for the Aerobraking and Science phases.

The spacecraft is operated with a limited direct visibility from ground, with visibility windows durations depending on the mission phases.
The OBC implements standard SW services to store on-board and execute command queues when the S/C will not be in visibility and the telemetry data will be stored in a mass memory for later downlink.

Moreover, when ground contact is not available, the spacecraft is fully autonomous through an event-based set of commands reacting to triggering conditions. These commands are organized as high-level plans and can be invoked also in case of contingencies ensuring the safety of the spacecraft in case of off-nominal scenarios.

Outside critical phases described below, all anomalies are recovered at their proper level and can ultimately lead to SAFE mode. The SAFE mode does not need any ground contact to be maintained as long as the orbit stability allows it.

SC Autonomy shall be balanced with the development cost and risk. A higher level of autonomy could increase the overall costs for the development and testing of the SC. For this reason the suggested approach foresees a classical FDIR design.

The main mechanisms to ensure an autonomous operation of the are:
• The operation scheduling mechanism (MTL) and the OBCP mechanism in charge of the management of the programmed operations of the spacecraft
• The fault protection system (FDIR system) which guarantees the safe operation of the spacecraft in case of failure.

Further consolidation of the design of the SC might lead to implement a 2 level FDIR system as already implemented in the ExoMars 2016 TGO: the Autonomous Fail Operational (AFO) for category 1 and the Autonomous Fail Safe (AFS) for category 2.

During the Time critical operations, the failures are managed through a Fail-Op concept that forbids a transition to SAFE mode and forces the continuation of the nominal operations.
Transitions to SAFE mode are however still foreseen for the non-critical mission phases through a Fail-Safe approach.

3.4 SYSTEM BUDGETS

3.4.1 Mass Budget

The mass budget reported on Table 3-12 refers to the worst case scenario, corresponding to the 2030 launch opportunity. In this case, in order to reduce the Aerobraking duration, an Apoares lowering maneuver is considered. Consequently the maximum allocation for the payload is around 46.2 kg.

Table 3-12 Mass Budget (2030 case)

ORBITER	S\S	Acronym	Description	Quantity	Unit Mass (Kg)	Margin	Unit Mass with Margin (Kg)	Total Mass with Margin (Kg)	
	STRUCTURE								187.4
			Primary Structure	1	86.2	5%	90.5		
			Secondary Structure	1	70.7	5%	74.2		
			Tertiary Structure	1	19.4	5%	20.4		
			Miscellanea	1	2.2	5%	2.3		
	TCS			1	19.1	5%	20.0		20.0
	PROPULSION								28.3
		TNK	Tank	1	16.1	5%	16.9	16.9	
		FDV	Fill and Drain Valve	3	0.1	5%	0.1	0.2	
		FVV	Fill and Vent Valve	1	0.1	5%	0.1	0.1	
		LF	Line Filter	1	0.7	5%	0.7	0.7	
		LV	Latch Valve	2	0.5	5%	0.5	1.0	
		PV	Pyro Valve	2	0.4	5%	0.4	0.8	
		PT	Pressure Trasducer	3	0.8	5%	0.8	2.4	
		PIP	Pipes	1	2.9	5%	3.0	3.0	
		ENG	ME 20N	1	0.7	5%	0.7	0.7	
		ENG	RCT 1N	8	0.3	5%	0.3	2.5	
	HARNESS								28.1
		HRN	Harness	1	28.1		28.1	28.1	
	EPS								122.5
		BTA	Battery Assembly	1	23.3	2%	23.8	23.8	
		PCDU	PCDU	2	6.2	5%	6.5	13.0	
		SADM+Yoke		2	8.3	5%	8.7	17.4	
		SAW	Area per wing 4.77 m2	2	32.5	5%	34.1	68.3	
	AVIONICS								57.3
		IPAC	Integrated Power Avionic Communication	1	17.5	6%	18.6	18.6	
		STT	Star Tracker	4	1.8	2%	1.9	7.5	
		IMU	Inertial Management Unit	1	4.9	5%	5.1	5.1	
		FSS	Fine Sun Sensor	4	0.0	2%	0.0	0.1	
		RW	Reaction wheels	4	6.0	8%	6.5	26.0	
	TT&C								102.6
		X-DST	[DTE] Transponder	2	3.3	5%	3.5	6.9	
		TWTA	[DTE] Amplifier	2	2.1	5%	2.2	4.4	
		HGA	[DTE] High Gain Antenna	1	9.5	5%	10.0	10.0	
		HGA APM	[DTE] HGA Pointing Mechanism	1	47.6	5%	50.0	50.0	
		RFDN	[DTE]	1	14.3	5%	15.0	15.0	
		LGA	[DTE] Low Gain Antenna	3	0.2	5%	0.2	0.6	
		Transceiver	[Proximity]	2	6.3	5%	6.6	13.2	
		LGA UHF	[Proximity]	1	2.3	5%	2.4	2.4	
	PAYLOAD								46.2
		PAY	Payload allocation	1	46.2		46.2	46.2	
	SEP. MECH.								16.5
		Sep. Mech	Clamp band + sep. Springs	1	15.0	10%	16.5	16.5	
							Total Dry Mass		609.0
							System Margin	20%	121.8
							Dry Mass with Margin		730.8
							Propellant		37.2
							Total Wet Mass		768.0

OIM	S\S	Acronym	Description	Quantity	Unit Mass (Kg)	Margin	Unit Mass with Margin (Kg)	Total Mass with Margin (Kg)	
	I/F								19.3
		TRST	Transition Structure	1	17.5	10%	19.3		
	OIM			1					520.0
		OIM	Orbit Insertion module (recurrent)		520.0		520.0		
							Total Dry Mass		539.3
							System Margin	20%	107.9
							Dry Mass with Margin		647.1
							Propellant		1784.9
							Total Wet Mass		2432.0
							MSO Total Wet mass		3200.0

3.4.2 Power Budget

The initial power budget defined for both Constellation and Orbiter mission was based on the same configuration of the spacecraft (OIM Module and HE-R1000 platform) and the estimation of the power consumption was estimated at the same level for both mission. At the same time was been highlighted that the most energetic s/s was the TT&C.
These preliminary power budgets envisaged the same consumption for both missions and in both cases took into account the adoption of particular strategies to limit the power consumption of the TT&C sub-system.
The results of the assumptions performed about the Electrical Power System (EPS) are hereunder reported.

Table 3-13 Power Budget (for MCC mission)

System	Item	Power demand with OIM (W)	Power demand w/o OIM (W)
AVIONICS	BTA, PCDU, APS, IPAC, PCU, etc.	258	258
TCS	Heaters, Thermistors, etc.	91	91
MCC TT&C (not contemporaneous)	DTE + proximity	250	250
Propulsion	OIM + propulsion power	402	2
TOTAL		1001	601
Losses (Voltage drop, units dissipation,…) TBC		20%	20%
P/L power at Mars [W] considering the use of OIM only during cruise		319	319

Table 3-14 Power Budget (for MSO mission)

System	Item	Power demand with OIM (W)	Power demand w/o OIM (W)
AVIONICS	BTA, PCDU, APS, IPAC, PCU, etc.	258	258
TCS	Heaters, Thermistors, etc.	91	91
TT&C (not contemporaneous)	DTE + proximity	250	250
Propulsion	OIM + propulsion power	402	2
TOTAL		1001	601
Losses (Voltage drop, units dissipation,…) TBC		20%	20%
P/L power at Mars [W] considering the use of OIM only during cruise		319	319

In the next phase of the program, when the Orbiter mission has been chosen, the need to implement the system emerged in order to better consider the power consumption of each s/s the System. Following this criteria, has been adopted a different configuration of S/A, including

a SADM, a Yoke and a DC/DC converter in order to get the best performance from the power generation and power distribution. In addition has been made compatible the HE-R1000 platform architecture with the other s/s of the Spacecraft and, in particular, the OIM Module that need a power bus at 100V instead of 48V as foreseen as standard from the platform.
Furthermore, has been clarified that just one battery of the set provide by the HE-R1000 platform was enough for S2M2 System with a capacity of 2300Wh to provide the energy during the eclipse phase of the mission. Furthermore, has been clarified that the power allocation for the Payload shall be set as follows;

a. Maximum power consumption of the payload during the cruise to Mars; 5 W,
b. Maximum power consumption of the payload during the sun exposition at Mars; 100 W,
c. Maximum power consumption of the payload during eclipse at Mars; 5 W.

Anyway, all the hardware of the EPS s/s has been evaluated and defined to be in the range between TRL 7 and 9.
Finally, as described in detail in the TN4, the final power budget has been defined with more accuracy, with the values reported in the following Table 3-15. With respect to the previous phase, the DC/DC convert that was considered to be added to the system, the investigation performed has put in evidence that this component (as a board) is already include in the IPAC and should be modified or, at least, changed with a needed one.
In the final power budget has been also detailed the different phase of the mission and also for each orbit at Mars. In fact, with respect to the previous power budget, in addition to the “cruise” and “at Mars” phases, has been included 3 main different moment at Mars;

1. The phase of illumination when the TT&C is transmitting and also this phase has been splitted in two different moment, to optimize the power consumption;
    a. Transmission with DTE only (max 50 minutes for each orbit), with a power consumption of 751W,
    b. Transmission with full power and system (max 10 minutes for each orbit), with a power consumption of 848W.
    c. The illumination phase where TT&C is OFF, with a power consumption of 611W,
2. The eclipse phase, where all the electronic unit has been considered in a stand-by condition, to minimize the power consumption.

Table 3-15 Final Power Budget

S2M2 MSO-D HEO

ORBITER	S\S	Acronym	Description	Quantity	Power demand on cruise [W]	Power demand on Mars during SUN DTE ONLY [W]	Power demand on Mars during SUN DTE+UHF [W]	Power demand on Mars during ECLIPSE [W]
							~ 10 min per orbit	
	STRUCTURE	STR		1				
	TCS	TCS		1	50	50	50	97
	PROPULSION	PRS		1	2	2	2	2
		TNK	Tank	1				
		FDV	Fill and Drain Valve	3				
		FVV	Fill and Vent Valve	1				
		LF	Line Filter	1				
		LV	Latch Valve	2				
		PV	Pyro Valve	2				
		PT	Pressure Trasducer	3				
		PIP	Pipes	1				
		ENG	ME 20N	1				
		ENG	RCT 1N	8				
	HARNESS	HRN		1				
		HRN	Harness	1				
	EPS	EPS		1	65	60	60	40
		BTA	Battery Assembly	1	5	0	0	0
		PCDU	PCDU	2	40	40	40	40
		SADM+Yoke		2	20	20	20	0
		SAW 1482x1610	Area per wing 4.77 m^2	4	-	-	-	-
	AVIONICS	AVC		1	204	204	204	144
		IPAC	Integrated Power Avionic Communication	1	100	100	100	100
		STT	Star Tracker	4	9	9	9	9
		IMU	Inertial Management Unit	1	1	1	1	1
		FSS	Fine Sun Sensor	4	34	34	34	34
		RW	Reaction wheels	4	60	60	60	0
	TT&C	TT&C		1	210	210	291	70
		X-DST	[DTE] Transponder	2	48	48	48	32
		TWTA	[DTE] Amplifier	2	139	139	139	18
		HGA	[DTE] High Gain Antenna	1	0	0	0	0
		HGA APM	[DTE] HGA Pointing Mechanism	1	23	23	23	20
		RFDN	[DTE] Radio Frequency Distribution Network	1	0	0	0	0
		LGA	[DTE] Low Gain Antenna	3	0	0	0	0
		Transceiver	[Proximity]	2	0	0	81	0
		LGA UHF	[Proximity] Low Gain Antenna	1	0	0	0	0
	PAYLOAD	PLD		1	5	100	100	5
		PAY	Payload allocation	1	5	100	100	5
	SEP. MECH.	SEP		1				
		Sep. Mech		1				
			Total Power demand		536	626	707	358
			System Margin		15%	15%	15%	15%
			Losses		5%	5%	5%	5%
			Total Power demand with Margin		643	751	848	430

OIM	S\S	Acronym	Description	Quantity	Power demand on cruise [W]	Power demand on Mars during SUN [W]	Power demand on Mars during SUN [W]	Power demand on Mars during ECLIPSE [W]
	I/F	I/F		1				
		TRST	Transition Structure					
	TCS	TCS	Thermal Control System	1	350	0	0	0
			Main Propulsion TCS		350	0	0	0
	PROPULSION	PRS	Propulsion	1	50	0	0	0
			OIM Propulsion		50	0	0	0
	AVIONICS	AVC	Avionics	1	10	0	0	0
		PIU	Power Interface Unit		10	0	0	0
			Total Power demand		410	0	0	0
			System Margin		15%	15%	15%	15%
			Losses		5%	5%	5%	5%
			Total Power demand with Margin		492	0	0	0
			Total Power demand with Margin		1135	751	848	430

An evaluation of the performance of the EPS has been simulated to verify the compatibility among the settings of the s/s (S/A power generation and dimensioning of the battery) and the System needs. This evaluation has taken into account the worst case of the orbit that shall happen during the mission considering the information provided in the following Table 3-16.

Table 3-16 Eclipse/illumination phases

<table>
<tr><th>Earth Time (from MA):</th><th>Eclipse / illumination distribution per each orbit</th><th>If eclipse starts simultaneously with occultation</th><th>Eclipse / illumination distribution per each orbit</th><th>If eclipse starts simultaneously with visibility</th></tr>
<tr><td rowspan="4">09:15</td><td rowspan="2">35 minutes eclipse</td><td>20 min eclipse in occultation (TX OFF)</td><td rowspan="2">35 minutes eclipse</td><td rowspan="2">35 min eclipse in visibility (TX OFF)</td></tr>
<tr><td>15 min eclipse in visibility (TX OFF)</td></tr>
<tr><td rowspan="2">80 minutes illumination</td><td rowspan="2">80 min visibility in illumination (TX ON)</td><td rowspan="2">80 minutes illumination</td><td>20 min occultation in illumination (TX OFF)</td></tr>
<tr><td>60 min visibility in illumination (TX ON)</td></tr>
<tr><td rowspan="3">14:44</td><td>35 minutes eclipse</td><td>35 min eclipse in occultation (TX OFF)</td><td>35 minutes eclipse</td><td>35 min eclipse in visibility (TX OFF)</td></tr>
<tr><td rowspan="2">80 minutes illumination (worst case for power)</td><td>20 min occultation in illumination (TX OFF)</td><td rowspan="2">80 minutes illumination</td><td>55 min occultation in illumination (TX OFF)</td></tr>
<tr><td>60 min visibility in illumination (TX ON)</td><td>25 visibility in illumination (TX ON)</td></tr>
</table>

As highlighted in the previous Table 3-16, the worst condition for the power budget is those at Earth time (from MA) at 14:44 and when the eclipse starts simultaneously with occultation that has the minimum illumination phase (the most power consumption) for a shorter period with respect to the other situations and the same duration of the eclipse phase (the lowest power consumption). This case has been considered as worst because, due to limitation of the TT&C activation (max 60 minutes each orbit), is the used ones, instead of those with 80 minutes of illumination and visibility at Earth Time (from MA) at 09:15.
The simulation performed has given an DoD of 4% (refer to Figure 3-13) that confirm the possibility to recharge the battery having a little discharge, maintaining an high reliability of the battery itself, capable to supply the System all during the eclipse phase.

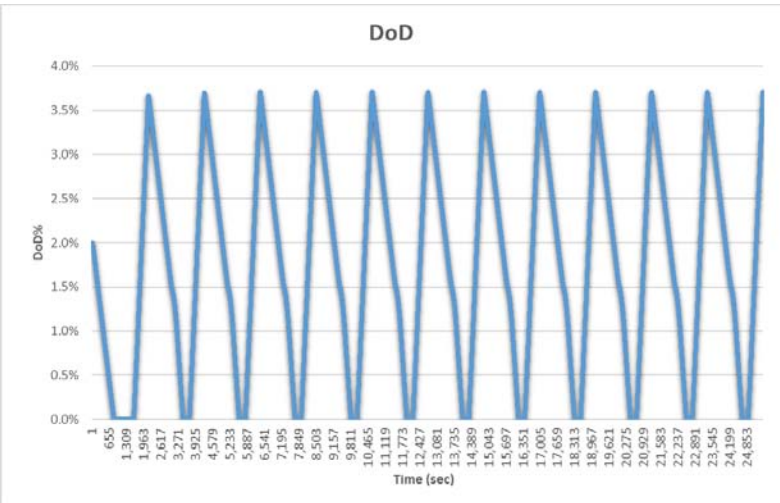
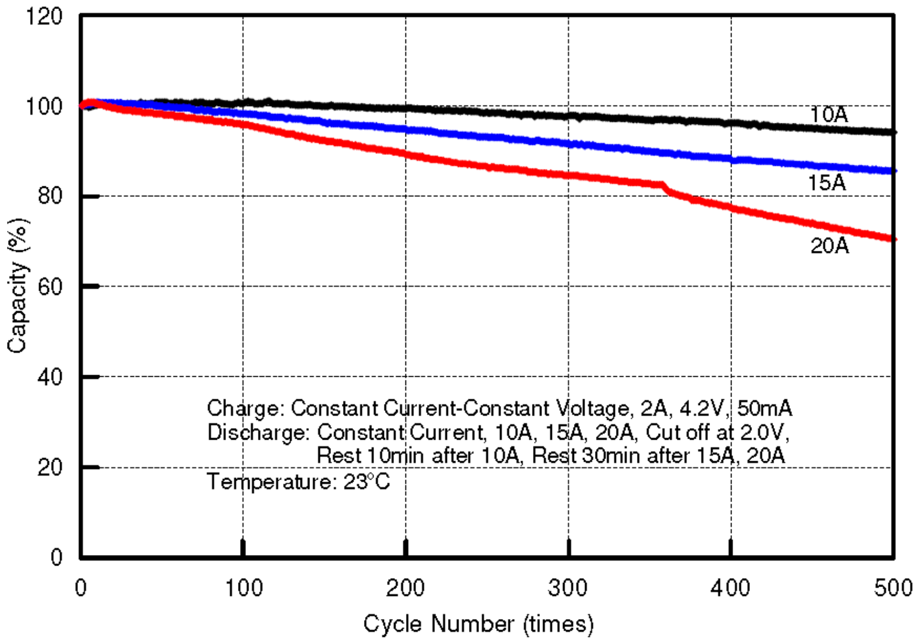


Figure 3-13: DoD of the battery

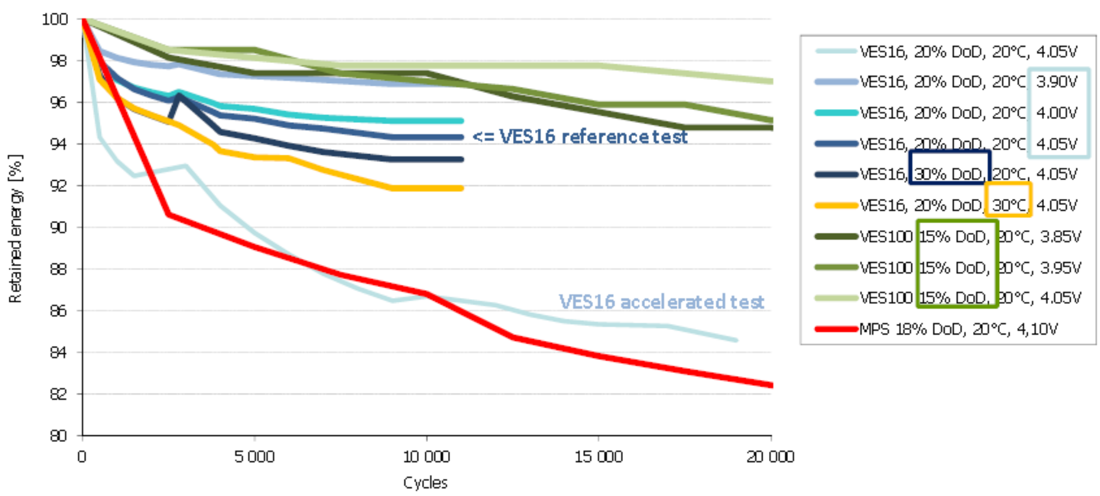
The battery modules adopted for S2M2 mission are those of the SAFT manufacturer, to get the best performance for Mars mission, even if has been maintained the same configuration of the basic module of HE-R1000 platform (16 strings in series and 20 in parallel). The choice to use the batteries of a different manufacturer (SFAT) wrt those in the baseline of the HE-R1000 platform (MOLICell manufacturer) has been done in order to get better performance in terms of durability (refer to following Table 3-17) and with heritage for Mars missions. As shall be noted comparing the information, even if not with the same values, is clear that the durability of the SAFT ones are higher wrt the MOLICell ones. In fact, the SAFT batteries (VES16 model) maintain about the 94% of their capacity after 12.000 cycles) while the MOLICell are at the same value after 500 cycles (but not with the same discharge current). However, to minimize modification of the platform, the same sizing (16x20 cells) has been maintained and has been considered to perform the power analysis.

Table 3-17 – Comparison of battery durability

MOLICell Datasheet



SAFT VES16 (NASA Battery Workshop - Hunstville: 2012, 6th to 8th November)



3.4.3 Delta-V budget

In this section the overall ΔV budget is presented for all the launch windows. In particular, Table 3-18 is referred to the launch window that happens in 2028, Table 3-19 is referred to the launch window that happens in 2030/31 while Table 3-20 is referred to the launch window that happens in 2033. The DV are referred to the missionphases defined on par. 3.3.1.

Table 3-18: ΔV breakdown for LW 2028

Phase	Maneuver	ΔV [km/s]	Gravity Losses Policy	ΔV (with Gravity Losses)	Margin Policy	ΔV (with Gravity Losses

				[km/s]		and margin) [km/s]
Earth Escape	Plane change	0.146	0%	0.146	10 m/s	0.156
	Pericentric maneuver	0.524	10%	0.576	5 %	0.605
Mars Capture	Pericentric maneuver	0.992	10%	1.091	5%	1.146
	Apocentric maneuver	0.014	0%	0.014	10 m/s	0.024
Aerobraking	Apocentric Lowering	0.136	10%	0.149	5%	0.157
	Walk-in	0.006	0%	0.006	5%	0.0063
	Pericenter maintenance	0.00116	0%	0.00116	100%	0.001218
	Walk-out	0.0603	0%	0.0603	5%	0.06348
Operational Orbit	Station keeping	0.002	0%	0.002	100%	0.004
Disposal	Apocentric maneuver	0.005	0%	0.005	10 m/s	0.015
	Pericentric maneuver	0.005	0%	0.005	10 m/s	0.015
TOTAL [km/s]	-	**1.891**	**-**	**2.055**	**-**	**2.193**

Table 3-19 ΔV breakdown for LW 2030

Phase	Maneuver	ΔV [km/s]	Gravity Losses Policy	ΔV (with Gravity Losses) [km/s]	Margin Policy	ΔV (with Gravity Losses and margin) [km/s]
Earth Escape	Plane change	0.056	0%	0.056	10 m/s	0.066
	Pericentric maneuver	0.704	10%	0.774	5 %	0.813
Mars Capture	Pericentric maneuver	1.232	10%	1.356	5%	1.423
	Apocentric maneuver	0.015	0%	0.015	10 m/s	0.025
Aerobraking	Apocentric Lowering	0.136	10%	0.149	5%	0.157
	Walk-in	0.006	0%	0.006	5%	0.0063
	Pericenter maintenance	0.00116	0%	0.00116	100%	0.001218
	Walk-out	0.0603	0%	0.0603	5%	0.06348
Operational Orbit	Station keeping	0.002	0%	0.002	100%	0.004
Disposal	Apocentric maneuver	0.005	0%	0.005	10 m/s	0.015
	Pericentric maneuver	0.005	0%	0.005	10 m/s	0.015
TOTAL	-	**2.222**	**-**	**2.429**	**-**	**2.589**

Table 3-20 ΔV breakdown for LW 2033

Phase	Maneuver	ΔV [km/s]	Gravity Losses Policy	ΔV (with Gravity Losses) [km/s]	Margin Policy	ΔV (with Gravity Losses and margin) [km/s]
Earth Escape	Plane change	0.131	0%	0.131	10 m/s	0.141
	Pericentric maneuver	0.584	10%	0.642	5%	0.675
Mars Capture	Pericentric maneuver	1.164	10%	1.280	5%	1.344
	Apocentric maneuver	0.015	0%	0.015	10 m/s	0.025
Aerobraking	Apocentric Lowering	0.136	10%	0.149	5%	0.157
	Walk-in	0.006	0%	0.006	5%	0.0063
	Pericenter maintenance	0.00116	0%	0.00116	100%	0.001218
	Walk-out	0.0603	0%	0.0603	5%	0.06348
Operational Orbit	Station keeping	0.002	0%	0.002	100%	0.004
Disposal	Apocentric maneuver	0.005	0%	0.005	10 m/s	0.015
	Pericentric maneuver	0.005	0%	0.005	10 m/s	0.015
TOTAL	-	**2.109**	**-**	**2.296**	**-**	**2.447**

3.4.4 Propellant Budget

In the following tables, the propellant budget for the 3 launch opportunities, with the Apoares Lowering maneuver down to a 1 SOL orbit.

Table 3-21 – 2028 (1 SOL) Propellant Budget

MSO (2028)		**HEO**	**LVA**
Launcher Performance	*A62*	3200	**100**
Launcher Margin		**0.0%**	
MSO		**920.1**	**Payload** **203.9**
MSO Propellant		**46.9**	
OIM Dry Mass		**647.1**	
OIM Prop. Mass		**1586.0**	
Launch Mass	[kg]	**3200.0**	

EVENT		**ΔV [m/s]**			**Mass [kg]**			
EVENT name		**ΔV [m/s]**	**ΔV Margin [%]**	**Total ΔV [m/s]**	**Main Propellant**	**Thruster Propellant**	**Dry Mass staged**	**Remaining Mass**
								3200
LEOP	Main	0	0%	0	0	0		3200
HEO departure	Main	761	0%	761	689	0		2511
MOI	Main	1170	0%	1170	781	0		1730
Apoares Lowering	Main	157	0%	157	84	0		1645
Staging	YES				*31*	-	647	967
Aerobraking	Thruster	71	0%	71	0	31		936
Station Keeping	Thruster	4	0%	4	0	2		934
Disposal	Thruster	30	0%	30	0	13		921
Residuals					***0***	***1***		**920**
TOTAL		**2193**		**2193**	**1586**	**47**	**647**	

Table 3-22 - 2030 (1 SOL) Propellant Budget

MSO (2030)		HEO	LVA
Launcher Performance	A62	3200	100
Launcher Margin		0.0%	
MSO		730.8	Payload 46.2
MSO Propellant		37.2	
OIM Dry Mass		647.1	
OIM Prop. Mass		1784.9	
Launch Mass	[kg]	3200.0	

EVENT		ΔV [m/s]			Mass [kg]			
EVENT name		ΔV [m/s]	ΔV Margin [%]	Total ΔV [m/s]	Main Propellant	Thruster Propellant	Dry Mass staged	Remaining Mass
								3200
LEOP	Main	0	0%	0	0	0		3200
HEO departure	Main	879	0%	879	782	0		2418
MOI	Main	1448	0%	1448	894	0		1524
Apoares Lowering	Main	157	0%	157	74	0		1450
Staging	YES				35	-	647	768
Aerobraking	Thruster	71	0%	71	0	25		743
Station Keeping	Thruster	4	0%	4	0	1		742
Disposal	Thruster	30	0%	30	0	10		732
Residuals					0	1		731
TOTAL		2589		2589	1785	37	647	

Table 3-23 - 2033 (1 SOL) Propellant Budget

MSO (2033)		HEO	LVA
Launcher Performance	A62	3200	100
Launcher Margin		0.0%	
MSO		795.9	Payload 100.5
MSO Propellant		40.5	
OIM Dry Mass		647.1	
OIM Prop. Mass		1716.4	
Launch Mass	[kg]	3200.0	

EVENT		ΔV [m/s]			Mass [kg]			
EVENT name		ΔV [m/s]	ΔV Margin [%]	Total ΔV [m/s]	Main Propellant	Thruster Propellant	Dry Mass staged	Remaining Mass
								3200
LEOP	Main	0	0%	0	0	0		3200
HEO departure	Main	816	0%	816	733	0		2467
MOI	Main	1369	0%	1369	872	0		1595
Apoares Lowering	Main	157	0%	157	78	0		1517
Staging	YES				34	-	647	836
Aerobraking	Thruster	71	0%	71	0	27		809
Station Keeping	Thruster	4	0%	4	0	1		808
Disposal	Thruster	30	0%	30	0	11		797
Residuals					0	1		796
TOTAL		2447		2447	1716	41	647	

In the following tables, the propellant budget for the 3 launch opportunities, without any Apoares Lowering maneuver; the aerobraking begins after the MOI from a 4 SOL orbit.

Table 3-24 - 2028 (4 SOL) Propellant Budget

MSO (2028)		HEO	LVA
Launcher Performance	A62	3200	100
Launcher Margin		0.0%	
MSO		1002.0	Payload 272.2
MSO Propellant		51.0	
OIM Dry Mass		647.1	
OIM Prop. Mass		1499.9	
Launch Mass	[kg]	3200.0	

EVENT		ΔV [m/s]			Mass [kg]			
EVENT name		ΔV [m/s]	ΔV Margin [%]	Total ΔV [m/s]	Main Propellant	Thruster Propellant	Dry Mass staged	Remaining Mass
								3200
LEOP	Main	0	0%	0	0	0		3200
HEO departure	Main	761	0%	761	689	0		2511
MOI	Main	1170	0%	1170	781	0		1730
Apoares Lowering	Main	0	0%	0	0	0		1730
Staging	YES				29	-	647	1053
Aerobraking	Thruster	71	0%	71	0	34		1019
Station Keeping	Thruster	4	0%	4	0	2		1017
Disposal	Thruster	30	0%	30	0	14		1003
Residuals					0	1		1002
TOTAL		2036		2036	1500	51	647	

Table 3-25 - 2030 (4 SOL) Propellant Budget

MSO (2030)		HEO	LVA
Launcher Performance	A62	3200	100
Launcher Margin		0.0%	
MSO		803.0	Payload 106.3
MSO Propellant		40.9	
OIM Dry Mass		647.1	
OIM Prop. Mass		1709.0	
Launch Mass	[kg]	3200.0	

EVENT		ΔV [m/s]			Mass [kg]			
EVENT name		ΔV [m/s]	ΔV Margin [%]	Total ΔV [m/s]	Main Propellant	Thruster Propellant	Dry Mass staged	Remaining Mass
								3200
LEOP	Main	0	0%	0	0	0		3200
HEO departure	Main	879	0%	879	782	0		2418
MOI	Main	1448	0%	1448	894	0		1524
Apoares Lowering	Main	0	0%	0	0	0		1524
Staging	YES				34	-	647	844
Aerobraking	Thruster	71	0%	71	0	27		817
Station Keeping	Thruster	4	0%	4	0	2		815
Disposal	Thruster	30	0%	30	0	11		804
Residuals					0	1		803
TOTAL		2432		2432	1709	41	647	

Table 3-26 - 2033 (4 SOL) Propellant Budget

MSO (2033)		HEO	LVA
Launcher Performance	A62	3200	100
Launcher Margin		0.0%	
MSO		871.5	Payload 163.4
MSO Propellant		44.4	
OIM Dry Mass		647.1	
OIM Prop. Mass		1637.1	
Launch Mass	[kg]	3200.0	

EVENT		ΔV [m/s]			Mass [kg]			
EVENT name		ΔV [m/s]	ΔV Margin [%]	Total ΔV [m/s]	Main Propellant	Thruster Propellant	Dry Mass staged	Remaining Mass
								3200
LEOP	Main	0	0%	0	0	0		3200
HEO departure	Main	816	0%	816	733	0		2467
MOI	Main	1369	0%	1369	872	0		1595
Apoares Lowering	Main	0	0%	0	0	0		1595
Staging	YES				32	-	647	916
Aerobraking	Thruster	71	0%	71	0	30		886
Station Keeping	Thruster	4	0%	4	0	2		885
Disposal	Thruster	30	0%	30	0	12		872
Residuals					*0*	*1*		871
TOTAL		2290		2290	1637	44	647	

3.4.5 Link Budget

3.4.5.1 X-BAND LINK BUDGET ANALYSIS

In the following Table, a sizing-case link budget for the X-band DTE (at the maximum distance) is reported.
An uplink datarate of 32kbps is always guaranteed.
The downlink performance, affected by solar losses much more than the uplink, is shown for three values of SES angle: 3°, 5° and 10° (at which the solar losses become almost negligible). It can be seen that the target of 50 kbps can be reached with a good margin only in this last case. When there are solar losses, a lower data rate for downlink will have to be considered and, in order to download all generated data, longer transmission windows will be needed.

Table 3-27: DTE link budget for X-band DTE. Assumptions: New Norcia Ground Station at 10° elevation, HGA with pointing accuracy 0.2°, 2.7 AU distance, GMSK 0.5, Turbo ¼.

	Uplink	**Downlink**		
Frequency [GHz]	7.19	8.44	8.44	8.44
TX power [W]	20000	65	65	65
EIRP [dBW]	107	59.2	59.2	59.2
TX antenna gain [dBi]	66 (G/S)	42.5 (HGA)	42.5 (HGA)	42.5 (HGA)
RX antenna Gain [dBi]	41.8 (HGA)	68.2 (G/S)	68.2 (G/S)	68.2 (G/S)
	Uplink	**Downlink**		
Path loss [dB]	281.70	283.10	283.10	283.10
Atmospheric loss	0.5	0.5	0.5	0.5

[dB]				
SES angle [°]	3	3	5	10
Solar losses [dB]	0.9	5.9	3	0.2
RFDN losses [dB]	2.58	1.48	1.48	1.48
Modulation index [rad-pk]	0.9	N/A (GMSK)	N/A (GMSK)	N/A (GMSK)
TC rate [kbps]	**32**			
TM rate [kbps]		**20**	**38**	**50**
Demodulation loss [dB]	2.2	0.9	0.9	0.9
PLL BW [Hz]	115	10	10	30
Carrier margin	**26.2**	**20.5**	**23.4**	**6.2**
TC margin	**5.3**			
TM margin		**3.0**	**3.0**	**4.6**

3.4.5.2 UHF LINK BUDGET ANALYSIS

The UHF link performance depends on the link distance between the satellite and the Mars surface asset. During the mission, those distances will vary periodically. They drive the maximum achievable data rates in UHF - rates are also driven by the Proximity-1 allowed data rates, which follow the protocol's granularity. Results are shown in the next Figure (referred to the datarate achievable when the spacecraft is at the maximum elevation).

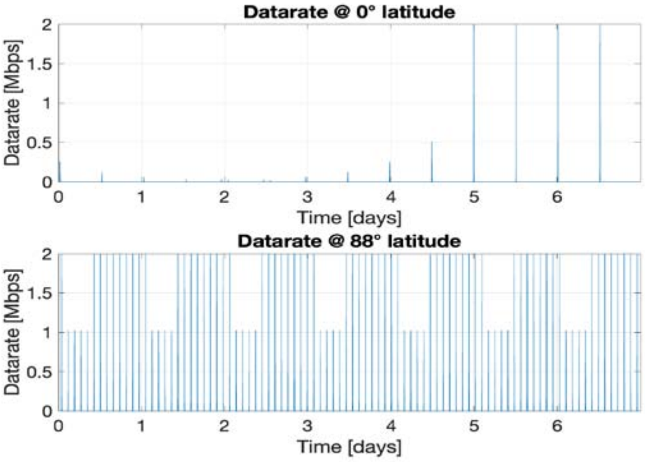


Figure 3-14: Return datarates from an Equatorial and a Polar user on the Mars surface, over 1 week.

3.4.5.3 DATA VOLUME ANALYSIS

From the above results, the amount of data exchange between the Mars surface and Earth can be estimated.

The X-band data volume analysis has been carried out starting from Mission Analysis input regarding the sequence of Earth visibility and Earth occultation times during each orbit.

Such times vary periodically during the mission (i.e. every about 2.5 years) between the following boundaries:

1. Earth time 09:15 → 20 minutes occultation and 95 minutes visibility.
2. Earth time 14:44 → 55 minutes occultation and 60 minutes visibility.

The obtained values will have to be intersected with the eclipse time during each orbit, because, for power budget reasons, transmission windows will have to be foreseen only when the spacecraft is in illumination (while instead it will not transmit during eclipses).
The eclipse duration, for each orbit, is of 35 minutes.

The worst case for what concerns the data volume, because it implies shorter communication windows, is when eclipse starts simultaneously with visibility. This is the case examined for the data volume computation.

Table 3-28 Communication wrt illumination

Earth Time (from MA):	Occultation / Visibility distribution	
09:15	20 minutes occultation	20 min occultation in illumination (TX OFF)
	95 minutes visibility	35 min eclipse in visibility (TX OFF)
		60 min visibility in illumination (TX ON)
14:44	55 minutes occultation	55 min occultation in illumination (TX OFF)
	60 minutes visibility	35 min eclipse in visibility (TX OFF)
		25 visibility in illumination (TX ON)

The resulting total communication time per day is:

Table-3-29: total X-band TX windows duration per day

Earth time	**09:15**	**14:44**
Occultation time per orbit [min]	20	55
Visibility time per orbit [min]	95	60
Orbit duration [min]	115	115
Orbits per day	12.52	12.52
Eclipse time per orbit [min]	35	35
Visibility time per orbit in solar illumination [min]	60	25
Tot TX duration per day [min]	751.30	313.04
Tot TX duration per day [hours]	**12.52**	**5.22**
Tot TX duration per day [s]	45078	18783

According to Earth position, communications can be established for 5.2 to 12.5 hours per day in total, divided in slots of 25 up to 60 minutes per orbit, i.e. when in visibility and illumination.

Of course, the above result has to be tailored with the Ground Station availability for S2M2, but this input is not available to TASI in this moment. The assumption retrieved from [RD1] shows a typical coverage of 6 hours, so the above result for 14:44 seems reasonable and can be used for data volume dimensioning. However, in the following analysis also the 09:15 input is used, in order to provide theoretical upper bound for the transmittable data volume with theoretical full GS coverage.

The computation of the data volume will have to take into account:

- the SCC housekeeping generated each day,
    - for S2M2, as an initial assumption, the TGO HK generation has been taken as assumed input: 200 Mb/day.
- the packets' overhead before transmission to Ground.
    - for S2M2, as an initial assumption, the EXM average frame efficiency of 89.68% has been considered.

Starting from the available data rate (dimensioned at the maximum Mars distance from Earth: 50 kbps), after removal of the overhead and of the housekeeping allocation, the remaining data volume results in 0.64 Gb/day in the worst case, up to 1.66 Gb/day with theoretical full GS coverage and can be allocated to the transfer to Earth of:

- data coming from the UHF link
- data coming from other payloads.

This volume has however to be reduced in specific periods, i.e. after a solar conjunction. Indeed, as explained above, the DTE link will be interrupted or corrupted during conjunctions, so that the data collected during this period will use part of the above data volume to be recovered as soon as communications will restart. The recovery time is about 5 days, but spreading this recovery on more days would allow an earlier restart of the UHF and payload relay functions (those functions are of course stopped during blackout periods).
A detailed analysis can be found in TN4.

The DTE data volume analysis has to be crossed with the UHF one.
According to the mission analysis, the visibility of the Mars Orbiter with an equatorial and a polar user vary periodically during one week. The resulting exchangeable data volume for each window (pass) and during the entire week is shown in Figure below (based on datarates reported in .Figure 3-14)

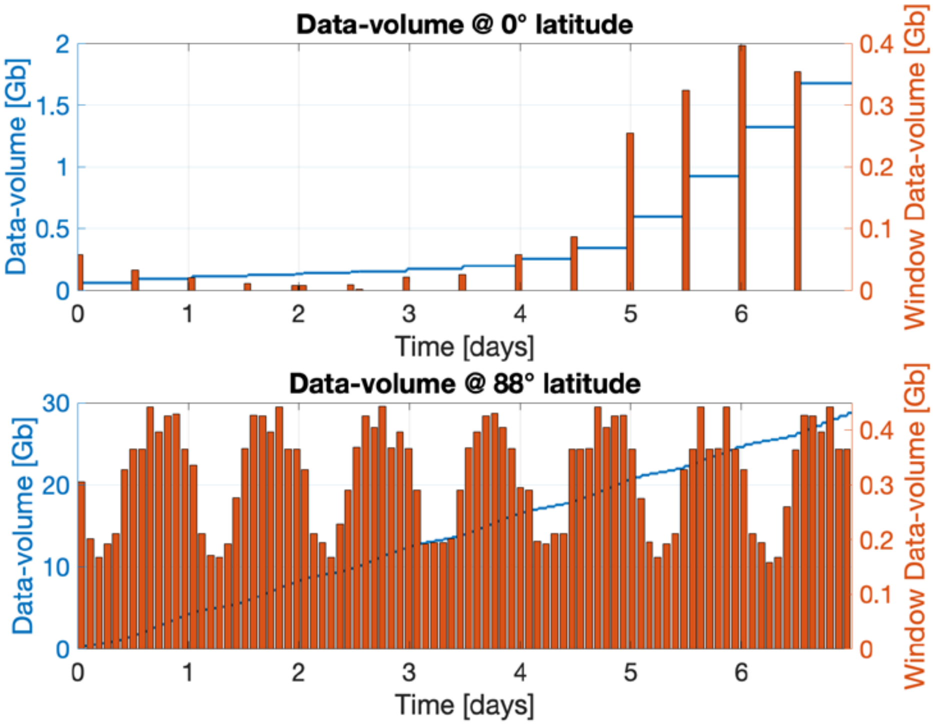


Figure 3-15 exchangeable data volume from an Equatorial and a Polar user on the Mars surface towards the orbiter, for each window (pass) and during the entire week

According to the DTE allocation to [UHF+payload] retrieved in the previous paragraph, not all the above data volume can be sent to Earth, even if every day the coverage to both users are guaranteed.
This is shown in the following Table.

Table 3-30 - Coverage

Earth time	Tot TX duration per day [hours]	S2M2 payload… allocation [Gb/day]	Mars User Latitude [°]	Average UHF transmittable data volume per day [Gb]	Average Covered passages	Best case Covered passages	Worst case Covered passages
09:15	11.52	1.82	0	0.23	4.6	all 3	all 3
			88	4.00	4	6 out of 13	3 out of 13
14:44	5.22	0.64	0	0.23	1.6	all 3	1 out of 3
			88	4.00	1.4	3 out of 13	1 out of 13

As a summary:
- With theoretical GS full coverage (Earth time 09:15), as an average, at least 3 passages are covered per day:
  - This covers all the available daily contacts with the orbiter from a user at 0° latitude

  - o This covers between 3 and 6 contacts with the orbiter from a user at 88° latitude, over 13 passages available per day.
- With 5.2 hours GS contact (Earth time 14:44), as an average, at least 1 passage is covered per day:
  - o This covers all the available low-rate daily contacts with the orbiter from a user at 0° latitude, and 1 out of 3 passages for the high-rate contacts.
  - o This covers between 1 and 3 contacts with the orbiter from a user at 88° latitude, over 13 passages available per day.

3.5 PAYLOAD ALLOCATIONS

The baseline configuration assign to the payload the following allocations.

3.5.1 Volume available

The payload could be mounted inside the spacecraft in the following volumes:

- +X Panels:
  - o Box 480 mm x 700 mm x 280 mm: 44 kg
  - o Box 480 mm x 250 mm x 280 mm : 16 kg
- -X Panels:
  - o N.2 Boxes 480 mm x 380 mm x 280 mm each: 48 kg each
- +Y Panels:
  - o Box 920 mm x 310 mm x 280 mm: 50 kg
- -Y Panels:
  - o Box 480 mm x 580 mm x 280 mm: 36 kg
  - o Box 450 mm x 310 mm x 280 mm: 18 kg

The mass reported in the list above is calculated considering an average density for the payload that corresponds to a total payload mass allowed (from a pure structural point of view) of 260 kg. The platform is capable of hosting on the top deck up to 320 kg of payload.

3.5.2 Temperature Ranges

In order to have a first estimation of the possible temperature ranges, the typical payload accommodated on the platform have the requirements reported in the following table.

Table 3-31 Temperature Ranges

	TEMPERATURE REQUIREMENTS									
	Qualification		Acceptance		Design		NOP			
	Operative		Operative		Operative		Not Operative		Cold Start	
SS/EQUIPMENT	MIN [°C]	MAX [°C]	MIN [°C]	MAX [°C]	MIN [°C]	MAX [°C]	MIN [°C]	MAX [°C]	MIN [°C]	Reference Element
SES										
RFE	-25	65	-20	60	-15	55	-40	75	-30	Baseplate
DEE	-25	65	-20	60	-15	55	-40	75	-30	Baseplate
FAA	-20	50	-15	45	-10	40				TRP

3.5.3 Power&Data Allocations

There are two Power BUS available:

- Regulated 28V
- Sun Regulated 67-100CV

The Power available is:

- 100 W on Sunlight
- 5 W on Eclipse

The data BUS is SpaceWire.

In order to increase the resources available without modifying the design some possibilities can be evaluated. Regarding the Power, as reported in the Power Budget (see **Error! Reference source not found.**), the high power consumption of the TT&C for the DTE and Proximity is a driver so limiting the communications windows (i.e. skip the communications on some pass) could increase the allocation of more power to the Payload.

3.5.4 Mass Allocations

The mass available depends on the Mission Profile selected because depending on the Launch Opportunity and the duration of the Aerobraking Phase, the total amount of propellant can vary of hundreds of kilograms.

In particular there are two main contributor to the increase of the propellant load (see par. 3.4.4):

- The 2030 launch opportunity DV, which is the highest in comparison to the others ones
- The presence of an additional Apoares Lowering maneuver performed by the OIM before being jettisoned.

Comparing the different combination of the points above, the payload allocation can vary as reported in the table below.

Table 3-32 – Payload mass allocation

Payload [kg]	2028	2030	2033
1 Sol	203.9	46.2	100.5
4 Sol	272.2	106.3	163.4

If can be excluded the 2030 opportunity as a backup solution, the minimum payload allocation increase up to 100 kg.

If a longer Aerobraking phase is sustainable, the payload allocation raise up to around 164 kg. As described on par. 3.5.1, all the payload could be accommodated inside the platform.

3.5.5 Data volume allocation

The final residual bandwidth allocated to a payload is based on the assumption to cover sent to Earth, every day, the UHF data collected during only 1 orbiter passage over a Mars user (estimated as 490 Mb as an average of the daily volume in Figure 3-15– please note that for Exomars 150 Mb/day are transmitted on average during every sol), and to allocate the rest of the available bandwidth to the payload.
This result is also reported for the periods after solar conjunctions, both when the recovery of the conjunction blackout is planned in 30 and in 60 days.

Table 3-33 – Data volume

TM rate [kbps]	Earth time	Tot TX duration per day [hours]	Transmittable data volume/day [Gb/day]	Transmittable data volume/day (excl. 89% frame efficiency) [Gb/day]	TGO HK/day [Mb/day]	S2M2 payload… allocation [Gb/day]	1-passage UHF return traffic to be sent to Earth (average) [Mb/day]	Residual data volume per day for a payload [Mb/day]	Residual data volume per day for a payload during 60 days' (worst) conj. recovery [Mb/day]	Residual data volume per day for a payload during 30 days' (worst) conj. Recovery [Mb/day]
50	09:15	12.52	2.25	2.02	200	1.82	425	1396.31	1342.83	1289.35
	14:44	5.22	0.94	0.84	200	0.64		217.21	163.73	110.26

As a result:

- With theoretical GS full coverage (Earth time 09:15):
  - 1.4 Gb /day can be allocated to a payload during nominal mission
  - 1.34 Gb /day can be allocated to a payload during the solar conjunction blackout retrieve if it's planned in 60 days.
- With 5.2 hours GS contact (Earth time 14:44) – worst case:
  - 217 Mb /day can be allocated to a payload during nominal mission
  - 163 Mb/day can be allocated to a payload during the solar conjunction blackout retrieve if it's planned in 60 days.

The result of sec. 3.2.1.6 is a dimensioning/worst case, indeed it has to be reminded that:

- It has been computed assuming that the eclipses always occur during visibility times, and this shortens the communications windows.
  - If the eclipses occur also outside visibility times, then the communication windows can be longer; however, this shall always be compatible with the SC power budget.

- 425 Mb/day are allocated to UHF data relay: this has been sized according to the UHF data traffic analysis of sec. 3.2.1.6 and approximates the best case passage, i.e. the one with the longest contact and higher data exchange. But: all the other contacts are shorter and foresee less data exchange, so that, planning the communication windows in other moments, with the same allocation more than one (shorter) contacts can be covered.
- 425 Mb/day allocated to UHF data relay is a huge amount w.r.t. the EXM reference which is 150 Mb/day, so another option is to decrease the UHF allocation per day: allocating less data to the UHF relay, for example arriving to the data exchange of ExoMars, of course would leave more bandwidth to the payload data:

TM rate [kbps]	Earth time	Tot TX duration per day [hours]	Transmittable data volume/day [Gb/day]	Transmittable data volume/day (excl. 89% frame efficiency) [Gb/day]	TGO HK/day [Mb/day]	S2M2 payload… allocation [Gb/day]	1-passage UHF return traffic to be sent to Earth (average) [Mb/day]	Residual data volume per day for a payload [Mb/day]	Residual data volume per day for a payload during 60 days' (worst) conj. recovery [Mb/day]	Residual data volume per day for a payload during 30 days' (worst) conj. Recovery [Mb/day]
0	09:15	12.52	2.25	2.02	200	1.82	150	1671.31	1629.45	1587.60
	14:44	5.22	0.94	0.84	200	0.64		492.21	450.36	408.50

As a result in this case:

- With theoretical GS full coverage (Earth time 09:15):
  - o 1.67 Gb /day can be allocated to a payload during nominal mission
  - o 1.63 Gb/day can be allocated to a payload during the solar conjunction blackout retrieve if it's planned in 60 days.
- With 5.2 hours GS contact (Earth time 14:44) – worst case:
  - o 492 Mb /day can be allocated to a payload during nominal mission
  - o 450 Mb/day can be allocated to a payload during the solar conjunction blackout retrieve if it's planned in 60 days.

3.6 PROGRAMMATICS AND RISK

The full approach for S2M2 is based on the possibility to procure two elements already developed for two other missions:

The propulsion module inherited from MSR ERO:, which is a full recurrent. It will be manufactured and tested starting close to the delivery of the OIM for MSR (in order to keep team, GSE and facilities, procurement of equipment and parts active in continuity with MSR ERO)

The Science Orbiter inherited from a HE-R1000 TAS LEO - platform used for Earth Observation (e.g. commercial products). In this case the S/C is maintained with some modifications which have to be introduced in order to comply with a mission to Mars, the major of which are:

- the TT&C which has to be completely refurbished,
- the introduction of an IMU
- a new larger battery
- SADM and Yoke to be applied to the solar arrays, made more robust for aerobraking
- a 20N additional engine
- a separation mechanism for detachment of the OIM after injection into Mars orbit

As for the **OIM**, the completion of its qualification process for the MSR ERO mission is currently expected by December 2025, exactly when it is expected to have the authorization to proceed to the implementation phases of the S2M2 programme. So, it will be possible to start the procurement of a full recurrent of OIM as needed for S2M2, upon programme selection.

The only change which will be requested is the interface adapter on top of the OIM in order to accommodate the Mars Science Orbiter : interface cone and ring with pyro (for separation at Mars) from the 1194 mm thrust cone of the OIM to the 937mm thrust cone of the MSO (HE-R 1000)

As for the **Mars Science Orbiter**, based on the LEO platform HE-R1000, this will partially qualified as well. This spacecraft is being procured for LEO missions, and is currently entering production for a specific case of earth observation. As not all its subsystems will be fully inherited for S2M2, we can assume that it will be provided with a partial qualification.

This approach is a pure design-to-cost one, based on:

- the adoption of two **existing and in-production platforms**, because ensures safe and reliable designs which depend on simple and proven concepts or technologies, supported by extensive early test & analysis verification task, simulations and trade-offs (all performed in the frame of the MSR ERO and HE-R1000 programs, so the costs will not rest on S2M2).

- the use of a **modular concept of the S2M2 Composite**, combining different elements to allow IVVQ interdependencies testing and flexibility in the AIV sequence, taking advantage to the maximum possible extent of the activities performed in the frame of the respective heritages;

- the use of separate modules implementation sequence: the OIM PFM and the HE-R1000 PFM complemented by ATB as needed and delta qualification as appropriate, resulting in more straight-forward implementation solution with a **unique S/C PFM test model**;

- minimum number of AIV&T models with maximum reasonable **re-utilization** on the basis of heritages also at **equipment** level (see par. 3.2.1)

- maximum commonality and **re-use of the GSE**, tools, facilities, including those for Phase E launch campaign (transportation) will be implemented.

As soon as the MSR ERO S/C has successfully concluded its QAR (by end of 2025), it will be possible to start with the procurement of the OIM recurrent for S2M2. Same approach shall be pursued for the Science Orbiter so as to reduce the delivery time.

In order to ensure a development duration in line with the constraints and cost reduction, the best approach would be to procure the needed items (equipment and parts) contextually with the MSR and the HE-R1000 ones, as soon as possible after the CMIN2025, based on direct negotiation or purchase order in order to shorten the duration of the procurement itself and to keep the heritage of the design, which allows to take advantage of the verification already performed and to get a more appealing price, by the same providers of the equipment and subsystems of the original programmes .

None products or parts or items under ITAR will be necessary. All the programme shall be ITAR free. Figure 3-16 schematically illustrates the assumed flows for integration, verification and testing after mating of the two modules.

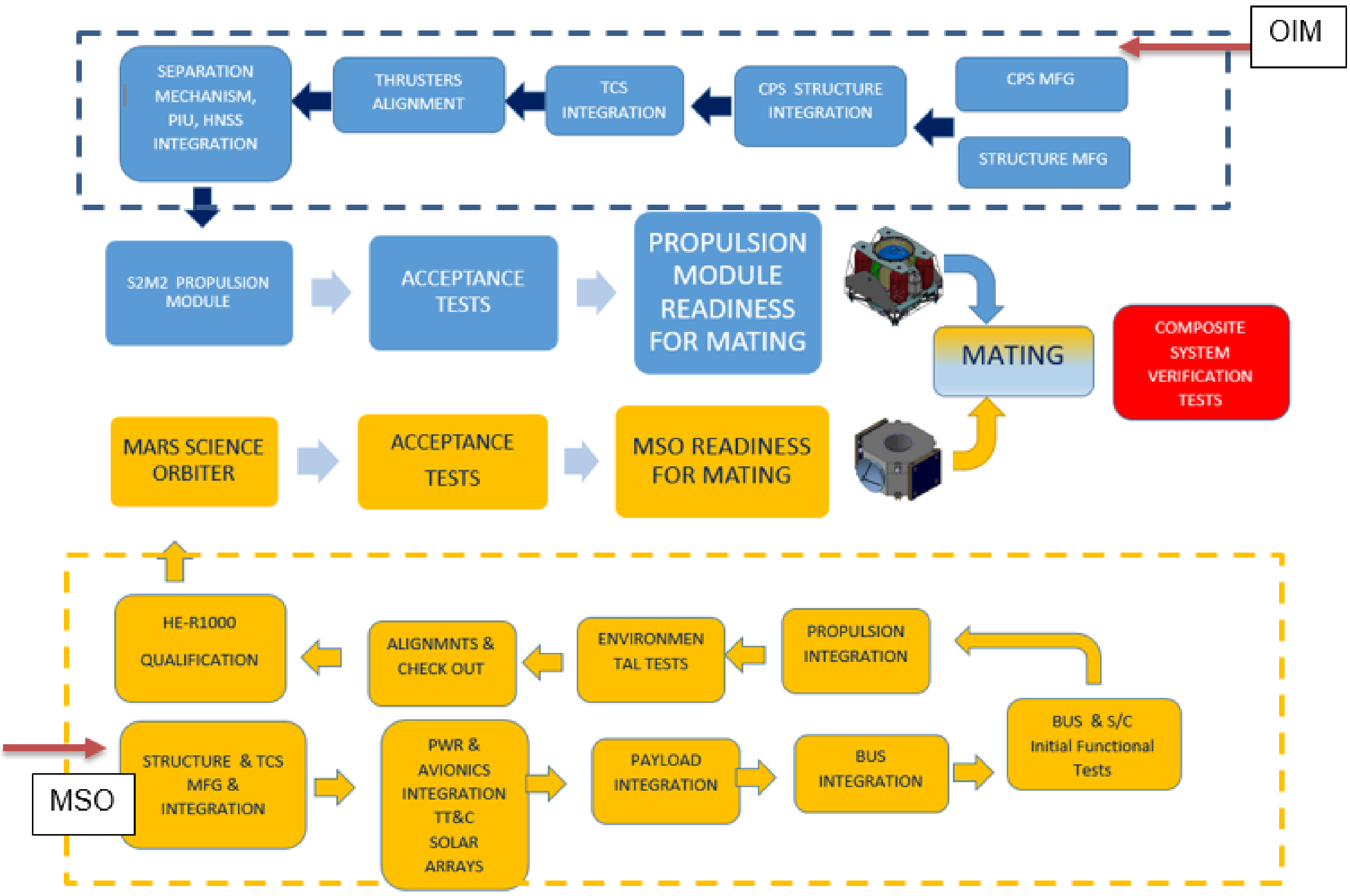


Figure 3-16 S2M2 PRELIMINARY DEVELOPMENT FLOWS SCHEMATIC

At Spacecraft Composite (OIM + HE R1000) level the following activities are envisaged:

- mating of OIM with HE-R1000
- execution of the system functional and environmental tests
  - Mechanical Test,
  - Qualification of the interfaces
  - TVTB,
  - EMC test

Environmental tests will consist in a selection of S/C level functional tests carried out in a S/C environment similar to that found during launch or in orbit

After the mating of the two modules into one single spacecraft, the acceptance testing will be done which essentially serves a dual purpose:

- to verify the integrity and function of the subsystem and its suitability for delivery
- to establish a performance baseline for the test to be done after environmental testing.

The Model Philosophy envisaged at this stage is based on a pure PFM –approach. However, given the needed refurbishment needed in the MSO (HE-R1000) , an EQM could be needed.

3.6.1 Schedule

The S2M2 schedule would be highly compressed in the implementation phase, so it will be necessary to review and prepare all the needed documentation during the preliminary phases. The major change (wrt a "usual" programme) is the duration of phase C.

Phase A : 9-months for detailed mission analysis and complete review of the two heritage projects documentation and customization to S2M2,

Phase B : 1 year for detailed review and preparation of the specification and the preparation of all the procurements documentations, agreements, contracts for all the needed items, and preparation of the CDR. All the papers shall be ready for signature. Proposal preparation.

Phase C: 4 to 5 months dedicated to: selection, review of the project and CDR

Phase D: 2,5 years (29 months) for manufacturing and testing

Margin: 3 months ESA margin before going into launch campaign

Launch campaign : 2 months

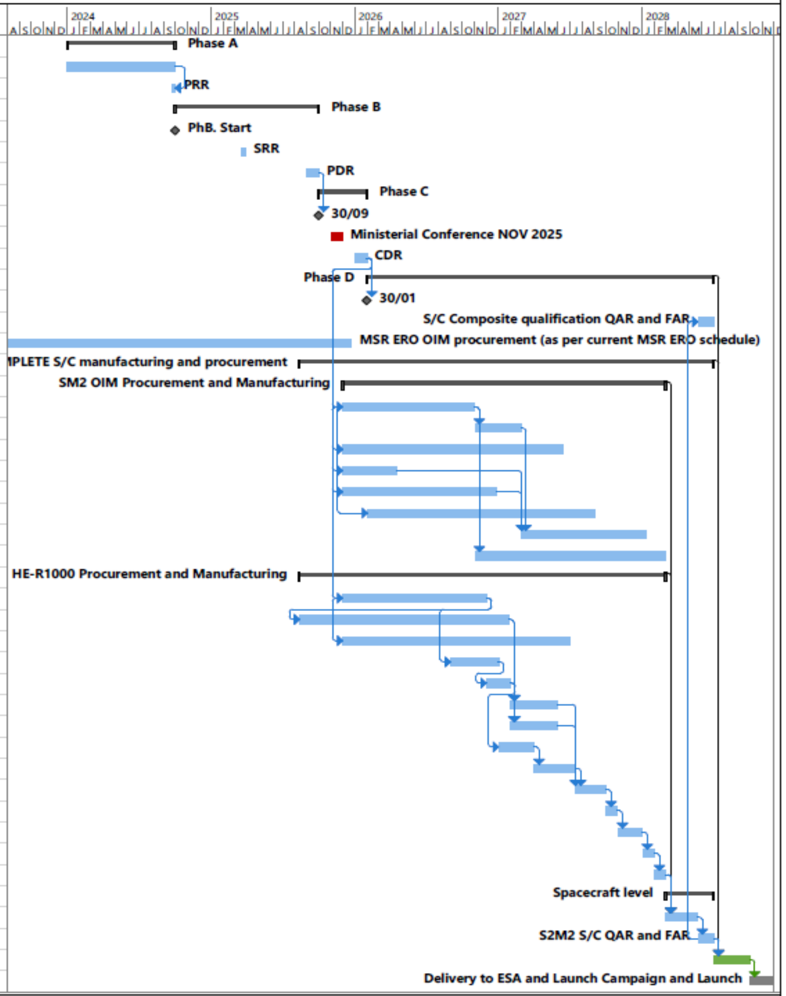
Schedule drivers are:

- CMIN on Nov 2025, before which no implementation can be carried out (otherwise said, Phase C could not be started)
- Launch date on Nov 2028,
- Availability of the CPS equipment (linked to an early ATP) soon after the CMIN
- Availability of the OIM Structure (linked to an early ATP) soon after the CMIN

Figure 3-17 – S2M2 Schedule

S2M2 Schedule 11062021

ID	Task Name	Duration	Start	Finish
1	**Phase A**	**196 days**	**01/01/24**	**30/09/24**
2	Establish feasibility through modules	196 days	01/01/24	30/09/24
3	PRR	4 days	25/09/24	30/09/24
4	**Phase B**	**261 days**	**01/10/24**	**30/09/25**
5	PhB. Start	0 days	01/10/24	01/10/24
6	SRR	10 days	17/03/25	28/03/25
7	PDR	22 days	01/09/25	30/09/25
8	**Phase C**	**88 days**	**30/09/25**	**30/01/26**
9	Ph.C Start	0 days	30/09/25	30/09/25
10	Ministerial Conference NOV 2025	20 days	03/11/25	28/11/25
11	CDR	22 days	01/01/26	30/01/26
12	**Phase D**	**630 days**	**30/01/26**	**30/06/28**
13	Ph. D start	0 days	30/01/26	30/01/26
14	S/C Composite qualification QAR and FAR	30 days	22/05/28	30/06/28
15	**MSR ERO OIM procurement (as per current MSR ERO schedule)**	**1407 days**	**31/07/20**	**22/12/25**
16	**S2M2 COMPLETE S/C manufacturing and procurement**	**753 days**	**13/08/25**	**30/06/28**
17	**SM2 OIM Procurement and Manufacturing**	**587 days**	**01/12/25**	**29/02/28**
18	CPS equipment procurement and equipping	240 days	01/12/25	30/10/26
19	CPS subassembly Integration and testing	85 days	02/11/26	26/02/27
20	TCS procurement	400 days	01/12/25	11/06/27
21	Long Lead Items proc.	100 days	01/12/25	17/04/26
22	Structure proc., mfg. & static test	280 days	01/12/25	25/12/26
23	Harness proc. Mfg. inst	412 days	02/02/26	31/08/27
24	OIM PFM AIT including CPS integration in OIM PFM	227 days	01/03/27	11/01/28
25	PIU FM procurement & MAIT and OIM functional test	347 days	02/11/26	29/02/28
26	**HE-R1000 Procurement and Manufacturing**	**665 days**	**13/08/25**	**29/02/28**
27	Structure manufacturing	261 days	01/12/25	30/11/26
28	Long Lead Items (P/L assumption: 18 months)	380 days	13/08/25	26/01/27
29	Harness proc. Mfg. inst	412 days	01/12/25	29/06/27
30	TCS integration	88 days	01/09/26	31/12/26
31	power Subsystem integration including SA and SADM	44 days	01/12/26	29/01/27
32	Avionics Test Bench integration	86 days	01/02/27	31/05/27
33	Telecomm subsystem integration	86 days	01/02/27	31/05/27
34	propulsion subsystem integration	64 days	01/01/27	31/03/27
35	Bus units integration (mech/elect) & alignment	76 days	01/04/27	15/07/27
36	BUS and S/C IFT	55 days	16/07/27	30/09/27
37	BUS & S/C prep to Environmnetal test & MOI-COG det	21 days	01/10/27	29/10/27
38	HE-R1000 Environmental testing	45 days	01/11/27	31/12/27
39	Alignments and check out	21 days	03/01/28	31/01/28
40	HE-R1000 QAR	21 days	01/02/28	29/02/28
41	**Spacecraft level**	**88 days**	**01/03/28**	**30/06/28**
42	Mating and testing	58 days	01/03/28	19/05/28
43	S2M2 S/C QAR and FAR	30 days	22/05/28	30/06/28
44	**Margin**	**65 days**	**03/07/28**	**29/09/28**
45	**Delivery to ESA and Launch Campaign and Launch**	**44 days**	**02/10/28**	**30/11/28**



3.6.2 Risk

According to the implemented schedule, the program has to be carried out in a timeframe of 31 months. The key point here is that this "new" approach is solid as long as the recurrence of the modules holds as described in the provide technical documentation [AD 7], [AD8].

Schedule risks: given the constraint of the November 2025 Ministerial Conference and the launch date by 2028, the phase C and D are compressed in two years and a half. This is quite challenging, but can be pursued provided a pre – agreement for the procurement of the two modules be already in place and ready for signature immediately after the Ministerial Conference 2025. Contracts have to be already in place and discussed, as well as the agreements for equipment procurement. All the procedure (RFI delivery, discussion with candidate providers, specifications, NDAs, answers and possible selections, and any other bureaucratic fact) for all the needed equipment will have to be carried out *before* November 2025. Thus, provided the authorization be given, we will be ready with the starting of the manufacturing as soon as the mission is selected . The LLI (mechanical structure of the OIM, CPS, some TT&C equipment) need some pre-agreement and need early ATP.

IVVQ activities: the only risks we can envisage at this stage are (again) on the schedule and are late Availability of S/C Hardware to be integrated and tested and Payload late delivery.
Facilities, GSE and tools: the assumption of being able to reuse all the means used for the "heritage" programmes is to be verified with the respective owners of the facilities/GSE/tools, and back-up solutions have to be found, in case.

Payload requirements (unknown for the moment) might lead to a more complicated development (e.g. due to very tight attitude constraints, to a very low temperature on the sensors, or due to tight magnetic cleanliness…).

Another procurement risks, which is outside the heritage programmes concerns the European UHF, for which no information are available yet. The usual 6-months taken as ESA margin between the delivery of the S/C and the FAR are reduced to 3 months. The launch campaign will have to be done in 2 months. The QAR and the FAR are coincident (FAR as conclusion of QAR)

Risks related to the heritage programmes: problems on MSR would impact also S2M2, so it will be important to be informed on the status of the MSR ERO OIM procurement in case of changes, issues, deviations, in order to timely implement the needed modifications.
In particular, the CPS (*Chemical Propulsion System)* is a LLI for which we'll have to assume an early Authorization to proceed. The CPS procurement would take about 660 working days. We can presume that there could be some margin for a more accelerated process, but anyway still it will be an issue to stay within the schedule. It will be necessary to have some authorization for starting an advance procurement.

3.6.3 Costs

A proper Cost estimate has not been done yet. However, a ROM evaluation of the real possibility to have the spacecraft developed within the cost target indicated by the SOW has been done.

We considered:

- Only phases C-D-E (assuming phases A and B differently funded)
- a pure design-to-cost approach
- in – house expertise and heritages, focusing on recent programmes, in order to take the advantage of both personnel experience, knowledge and hardware available in short times close to the “heritage” missions themselves.
- Model Philosophy: 1 PFM for each module
- Dedicated models are expected at equipment level for few specific items only (new entries)
- High TRL for all equipment (TRL 6 or higher at start of the phase B). No technology development costs are foreseen.
- Lessons learned related to OIM (in the frame of MSR ERO) and to HE-R1000 will be hold and checked for applicability. In this sense, a management reserve shall be considered in order to cover possible discrepancies which might show up in the ERO-OIM design at a later stage

A ROM evaluation of costs for the procurement of the two modules includes:

- Industrial cost of Modules (Platforms) MAIT during the implementation phase from KO after the CMIN2025 (Phases C and D)
- Industrial costs risks (mainly managerial and financial risks, technical are TBD but should be extremely limited due to the choice of recurrent, already qualified hardware
- Phases A and B excluded as usually covered with dedicated contracts and covered in specific ESA General Studies budget
- Payload excluded as unknown at this stage

Given all the above, our very high-level evaluation get to the result that the programme could be developed, as for the industrial part, within the cost limitation as indicated in the SoW. The distribution of the costs could be as illustrated in Figure 3-18. A detailed costs estimate has in any case to be performed, this report by no means can be considered as a commitment.
Following a ROM allocation estimated for each subsystem, the foreseen splitting of the costs (in %) can roughly be expected as in next figure.

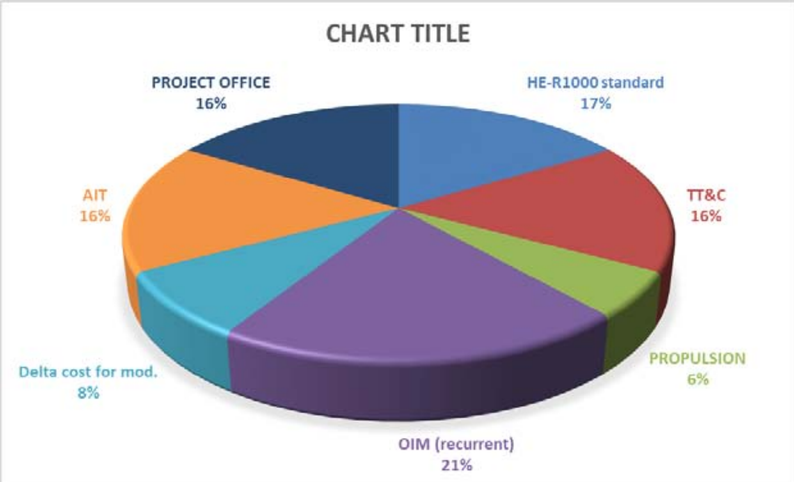


Figure 3-18 Expected, preliminary cost sharing for the S2M2 mission based on TAS heritages

We highlight that , at this stage, there are many possibilities which could be investigated, in terms of schedule, with shorter / longer duration of the preliminary phases. Though, this would not change much the approach, because as long as the constraints are the CMIN 2025 outcomes and the launch epoch, the implementation phases are very much constrained in a couple of years and a half in the best scenario (that is, accepting a reduction in ESA margin). In terms of design to cost, the key point for this mission is to bring about the procurement of HW in the wake of the same procurement done for the two heritage programmes. This is key to lower the cost.

The major, expected cost drivers are:

<u>On HE-R1000:</u>

1. TT&C is expected to be the most expensive subsystem due to:
   - The large number of items to be procured
   - Usually costly equipment which need qualification for deep space
   - Unknown cost of the European UHF
   - Verification and AIT activities to be performed end to end (more detailed investigations on the equipment may lead to an improvement of these aspects).

   The European UHF Transceiver cannot be evaluated as far as now. An ITT with parallel studies is ongoing, too premature to have information, so our cost evaluation and comment do not take this equipment into account.
2. Power subsystem: mainly due to the necessary new entries in the design (SADM, Yoke) this cost is going to increase wrt the baseline TT&C usually adopted by the HE-R1000 for the LEO missions.

<u>On the OIM:</u>
The Chemical Propulsion System : though it is likely that there will be some savings wrt the original programme, this is expected to be the most expensive subsystem of the Kick stage. The main reason being that for a propulsion system, all the due relevant tests have to be repeated in any case

END OF DOCUMENT