Mars Sample Return - Earth Return Orbiter: ESA's next Interplanetary Electric Propulsion Mission Concept

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Abstract: The return of physical samples from Mars to Earth for analysis continues to be a top priority of the international planetary science community. NASA and ESA are currently collaborating to explore options for conducting a joint Mars Sample Return (MSR) campaign, consisting of three flight missions, capable of delivering a variety of atmospheric and soil samples collected on Mars for analysis on Earth. In the current scenario under study, samples collected and cached by the NASA M2020 Rover would be collected by ESA's Sample Fetch Rover and launched into Mars orbit by NASA's Mars Ascent System. ESA is currently conducting parallel Phase A/B1 industrial studies on the Earth Return Orbiter (ERO) mission concept. The ERO is an ESA procured spacecraft carrying a NASA/JPL payload that will detect, rendezvous with and capture the orbiting sample and perform a safe transfer and return of the samples to Earth. The spacecraft is based around a Hybrid propulsion architecture, with both a high-power electric propulsion (EP) system and a 1 kN-class apogee boost system. The EP system is fully solar electric and will be capable of generating up to 1N from around 35 kW at Earth and half of this at Mars. The basic mission design foresees an Earth-Mars transfer using EP, an initial Mars Orbit Insertion using Chemical Propulsion (CP), spiral down/up phases using EP, and Mars-Earth transfer using EP. To achieve this between 2 and 4 EP thrusters will fire simultaneously, with the overall five-year mission

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requiring a lifetime of around 20,000 hrs per thruster. The basic spacecraft architecture builds on ESA's experience from the successful BepiColombo spacecraft, which is currently flying to Mercury using up to 2 EP thrusters firing in parallel.

Nomenclature

AU	*Astronomical Unit*
BPP	*Backwards Planetary Protection*
CCRS	*Capture, Containment and Return System*
CP	*Chemical Propulsion*
EDL	*Entry Descent and Landing*
EES	*Earth Entry System*
ERO	*Earth Return Orbiter*
EP	*Electric Propulsion*
FCU	*Flow Control Unit*
Ls	*Solar Longitude*
MAS	*Mars Ascent System*
MOI	*Mars Orbit Insertion*
MRSH	*Mars Returned Sample Handling*
MSR	*Mars Sample Return*
NSSK	*North-South Station Keeping*
OS	*Orbiting Sample*
PPU	*Power Processing Unit*
RIT	*Radio Frequency Ion Thruster*
SEP	*Solar Electric Propulsion*
SFR	*Sample Fetch Rover*
SRL	*Sample Return Lander*

I. Introduction

MARS Sample Return (MSR) continues to be a high priority in the planetary science community and is a decades-long goal of international planetary exploration programs[1]. The NASA Mars 2020 sample-caching rover mission[2] is the first component of a potential Mars Sample Return (MSR) campaign, so its existence constitutes a critical opportunity to return those samples to Earth for subsequent scientific analysis[3]. ESA and NASA are currently working together to explore concepts for the retrieval missions of this potential international MSR campaign, with launches of the mission planned as early as 2026. Key elements for this campaign are a NASA-led Sample Retrieval Lander (SRL), which would be responsible for retrieving the Mars 2020-collected samples with the aid of an ESA Sample Fetch Rover (SFR), loading them into an Orbiting Sample (OS) container, and launching the OS into a stable Mars orbit on a Mars Ascent System (MAS), and an ESA-led Earth Return Orbiter (ERO), which would be responsible for locating and capturing the OS in Mars orbit and ensuring its safe return to Earth. Figure 1 below illustrates the four-element campaign (including the ground-based element for handling the samples from landing through to curation; referred to as Mars Returned Sample Handling or MRSH), indicating the cross-mission contributions by each Agency.

By acquiring and returning to Earth a rigorously documented set of Mars samples for investigation in terrestrial laboratories, scientists will have access to the full breadth and depth of analytical science instruments available in these laboratories. The investigations will be free of the mass, volume, and power constraints that limit in-situ instruments. The resulting investigations of these returned samples will enable breakthrough advances in a) the search for ancient and/or extant life on Mars; b) understanding the origin and evolution of Mars as a geological system; c) understanding the processes and history of climate on Mars; and d) preparing for human exploration.

Backward Planetary Protection (BPP) is a critical design driver for a potential MSR campaign, with the ERO mission being the first to be categorised as a Category V restricted Earth return mission since Apollo 11. All aspects of the campaign and element architectures under study are intended to reflect guidance and principles derived from NASA and ESA planetary protection policies and requirements. The objective of the BPP design and implementation is to prevent the exposure of unsterilized and uncontained Mars material to the Earth's bio-sphere. This requires a strategy for the use of analysis, design, and test of the campaign elements and systems that would be implemented and validated/certified to deliver sample tubes to an Earth-based receiving facility, while containing and/or sterilizing any other Mars material that might reach the biosphere of Earth.

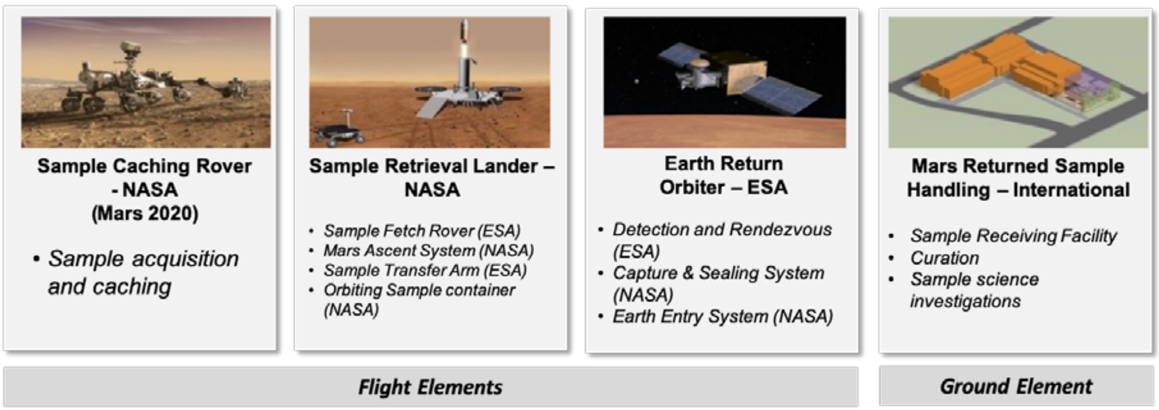

Figure 1:Mars Sample Return notional architecture.

Figure 3 illustrates the current reference mission scenario under joint study between ESA and NASA showing the interrelationship between the 3 flight elements and the ground element. The campaign timeline reflects the unique nature of this collaborative effort where each mission depends critically on the other in terms of functionality as well as phasing of operations. In particular, the timeline is driven by the need to ensure that the surface mission of SRL begins at the earliest opportunity during favourable seasonal conditions on Mars for the solar-powered spacecraft and favourable atmospheric and lighting conditions for Entry, Descent and Landing (i.e. early spring from Ls=0). Therefore, the landing of SRL is timed accordingly while the ERO mission has to be in its place in Mars orbit to provide the necessary Entry, Descent and Landing relay communications as well as to serve as the primary relay communication orbiter for the surface assets (SRL lander, SFR, Mars 2020 rover and the MAS).

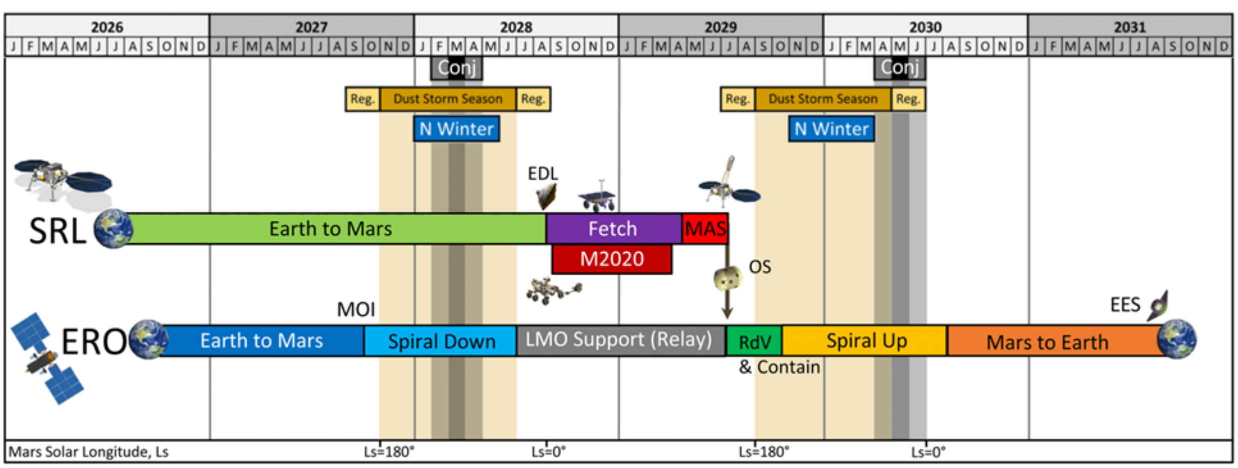

Figure 2: MSR Campaign Timeline for the 2026 launch opportunity.

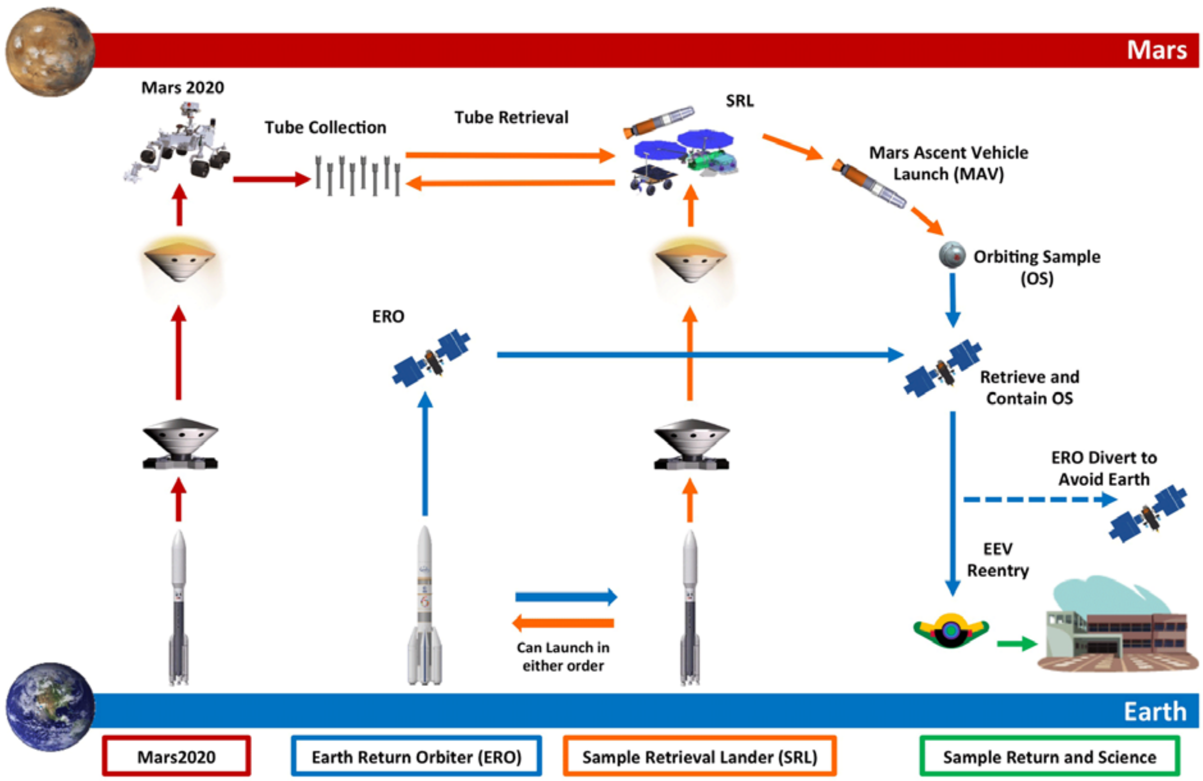

Figure 3:Reference MSR Mission Scenario for joint studies. Arrow colours indicate roles: Red is NASA Mars 2020, orange is NASA SRL, blue is ESA ERO, and green is international sample receiving, curation and analysis.

Figure 2 shows the reference campaign timeline for SRL and ERO coordinated operations for the 2026 launch opportunity. A similar timeline is available for both 2027 and 2028 launch opportunities. Coordination between the flight missions is synchronized according to the Martian dust storm season.

II.The Proposed ERO Mission

The information provided about possible Mars Sample Return architectures is for planning and discussion purposes only. NASA and ESA have made no official decision to implement Mars Sample Return.

A. Mission Concept

The ERO mission concept foresees a launch on an Ariane 6.4, after which it makes use of a continuous low-thrust transfer to Mars, using Solar Electric Propulsion (SEP). On arrival at Mars, it will use a bi-propellant chemical propulsion system to perform Mars Orbit Insertion (MOI), entering into a highly elliptical orbit around Mars. Again using its SEP in order to save propellant mass, ERO will spiral down into a circular target rendezvous orbit approximately 400 km above the surface of Mars. The transfer will be timed to provide relay telecommunication support to the Mars 2020 and the Sample Retrieval Lander (SRL) missions including Entry, Descent, and Landing (EDL) and the ESA provided Sample Fetch Rover (SFR). At the end of the surface mission, prior to the Northern hemisphere autumn and subsequent potential global dust storms, the ERO will be positioned to provide tracking and monitoring of the launch of the Mars Ascent System (MAS) and release of the Orbiting Sample (OS). ERO then detects and locates the OS in Mars orbit using a primarily optical on-board sensor suit. Following the detection and orbit determination of the OS, ERO will synchronise its orbit, rendezvous, and captures the OS. The Capture, Containment, and Return System (CCRS) on-board the ERO will isolate the OS from any Mars material and transfer the samples into the EES. It would provide a sealing between 'dust contaminated' and 'sterilised' sides within the system, ensuring that possible residual Mars material external to the OS does not enter the 'sterilised' side of the

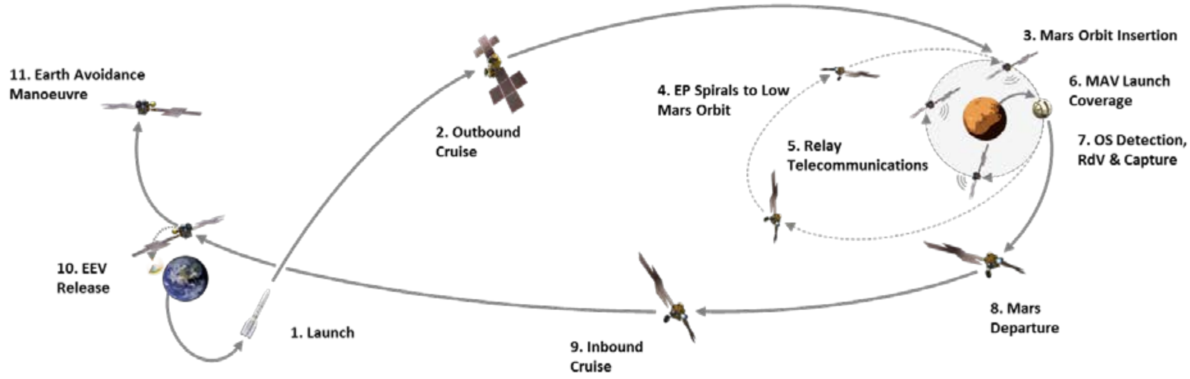

Figure 4:Illustration of the ERO mission concept

CCRS. Subsequently all potentially contaminated and unneeded portions of the CCRS would be jettisoned. The return module and the retained parts of the CCRS, not jettisoned into Mars orbit, will then begin the Mars departure by using the SEP to spiral out of its operational orbit and make the inbound journey to Earth. Upon arrival at Earth the ERO will release the EEV and perform an Earth avoidance manoeuvre to prevent any potential contact of residual Mars particles on the spacecraft with the Earth's atmosphere. This mission concept is illustrated in Figure 4. Launch dates as early as 2026 with Earth return by 2031 are under study.

B. Orbiter Concept

The ERO concept is based around a hybrid EP/CP spacecraft consisting of three modules: the chemical propulsion Orbit Insertion Module (OIM), which is separated at Mars; the Main Module, which contains the electric propulsion system with 4 x 8.5 kW thrusters, is powered by 41 kW solar arrays (1 AU equivalent) and returns to Earth; and the NASA provided Capture, Containment, and Return System (CCRS), which includes the Earth Entry System (EES). The total stack would be around 4.5 m tall and have a wingspan of more than 40 m. A notional artist's impression of the spacecraft is shown in Figure 5.

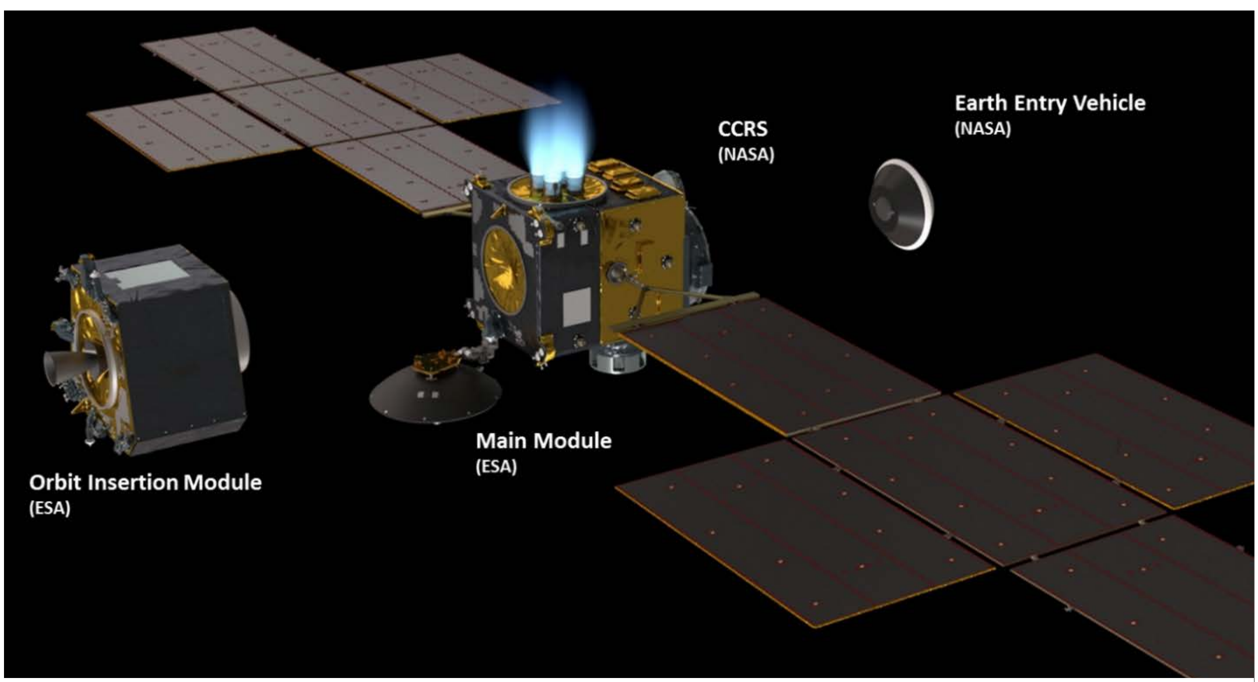

Figure 5:Artist's impression of possible ERO design

C. Orbiter Concept Trade-Offs

The proposed mission concept is the result of an extensive orbiter concept trade-off, which investigated and compared different mission architecture options, including all-chemical, all-electric, and hybrid chemical/electric propulsion architectures[4, 5]. The trade-space was then further increased by studying different Electric Propulsion (EP) systems (various gridded options and Hall-Effect thrusters) and by introducing the possibility to stage the spacecraft to reduce the mass for Mars operations and the Mars-Earth return leg, including different staging opportunities. The objective of these trade-offs was to identify a mission concept that minimizes cost and complexity, while still complying with the demanding Campaign timeline requirements and presenting a feasible (i.e. launchable) solution.

The trade-off resulted in the selection of a hybrid EP/CP concept for further detailed study, as it presented a good compromise between the all-chemical and all-electric propulsion options. The all-EP options were all-in-all very mass efficient, however were not able to achieve the tight mission timeline with the EP systems and Solar Arrays currently available in Europe. On the other hand, European technologies that could meet the technical requirements did not have the requisite TRL to meet the challenging spacecraft development schedule.

The all-CP architecture initially appeared attractive due to the benefits of a faster transfer and reduced complexity. However, due to the large delta-V required for a planetary return mission and penalizing multiplication factor for any mass (incl. propellant) carried to Mars and back to Earth, staging and long aero-braking phases around Mars are required to become mass effective and be even close to a feasible mission. Due to this “gear-ratio” of the CP architecture, CP concepts are inherently inflexible to mass growth or major changes in the mission strategy, where more propellant is required. All-CP concepts also carry significant programmatic risk related to mass margin management and hence ultimately scientific return for the Campaign. Nonetheless, the all-CP option still provides a number of important programmatic benefits such as low complexity, high heritage, cost efficiency, and fast transfers, making it attractive as a back-up. The cost of a CP back-up would be the need for heavy lift launcher, not available in Europe.

Taking all of this into account, alongside quantitative assessments of timeline, spacecraft composition and mass delivery capabilities, a hybrid EP/CP option, which relies on EP for the outbound transfer to meet launch mass, a chemical manoeuver for Mars Orbit Insertion to reduce time performed by a jettisonable stage, EP spiraling to the final orbit to reduce propellant mass, and an all-EP inbound transfer, provided the overall best mission architecture.

D. Mission Analysis

Fundamental to perform the mission analysis for ERO is to have accurate models to predict the output of the EP system at any time along the mission. The design driver is the size of the solar array for an output power of 41 kW at 1 AU. The total power produced by the solar arrays is a function of the distance to the Sun. A formula that corrects the $1/r^2$ dependency of the incoming power per unit area is used. It models the slight improvement of the decay in power for distances further away than 1 AU observed for solar panels. In addition, the effect of solar array degradation is considered at a rate of 1% power loss per year.

The total solar array power available gets deducted with margins and power needed for other subsystems to determine the maximum available power for the engines as the spacecraft moves further from the Sun as shown in the following Figure. Depending on the solar panel technology, some Beginning-Of-Life (BOL) losses are subtracted, in addition to the degradation towards the End-of-Life. Operational margin is used to account for the discrete steps in thrust level settings versus the use of a continuous curve in mission analysis, as well as some further operational flexibility. Losses are then included into the power conditioning and harness before being fed into the power processing unit for the thruster. Because the thruster is being developed and tested, additional maturity and system level margins are applied on the thruster itself.

The power to thrust dependency of the EP engine allows to obtain the total thrust as a function of the available power to the EP system. The resulting available thrust against the Sun distance is shown in Figure 6 for two levels of solar array degradation, 0% being applicable to the beginning of mission and 5% to the end of mission. Up until 1.15-1.18 AU 3 engines are planned to be used simultaneously at full power (though 4 could also be considered). Above this Sun distance 3 engines are used at lower power until 1.38-1.44 AU when the system has to transition to 2 engines. When the spacecraft is in orbit around Mars, it will be able to use only 2 engines firing simultaneously.

In addition, the mission analysis needs to consider a duty cycle penalty. The duty cycle accounts for planned outages of the thrust due to ground contacts dedicated to orbit determination and other spacecraft maintenance activities requiring the engines to be off, and for the effect of ramp-up of the thrust after an eclipse by Mars. Currently a 95% duty cycle is used across the mission.

In order to ensure that the heliocentric transfer trajectories are robust against unplanned thrust outages (e.g. safe mode) a 60-day coast arc prior to arrival at Mars or Earth is enforced as a constraint in the nominal trajectory design.

Dedicated missed thrust analysis using a Monte Carlo approach is performed to determine the potential additional propellant and time impacts expected from the thruster outages encountered along a given trajectory[6].

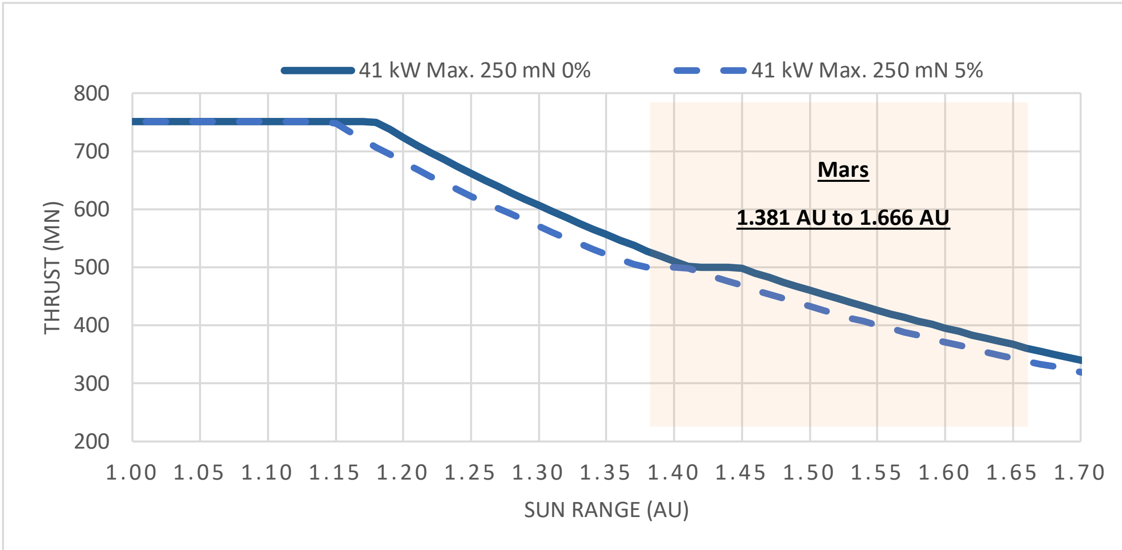

Figure 6:Thrust as a function of Sun distance for 3 x RIT-2X @ 41kW

The mission analysis work requires additional assumptions for the launch and at system level. Performance models for Ariane 64 are used to estimate the maximum spacecraft separated mass at launch as a function of the escape velocity and declination. A 21-day launch period with fixed launch conditions is assumed. In addition the spacecraft dry mass is necessary to perform the computation of the return leg, while the mass jettisoned at the staging points in orbit around Mars, i.e. the CP stage and the part of the CCRS that is dropped in low Mars orbit, is needed for the computation of the outbound leg.

With this information, a complex trajectory optimization process is performed by the ESOC mission analysis team that involves the use of the DITAN tool[7] for the heliocentric transfer trajectories and the in-house LATOP tool[8] for the EP spiral phases around Mars. LATOP considers the interruption of the EP operations due to the solar conjunctions encountered during the spiral-down and spiral-up phases (exclusion Sun-Earth-Spacecraft angle assumed 3 deg). It also models accurately the impact of switching off the thrust during eclipses.

The outbound heliocentric trajectory for the baseline launch is optimized to have the largest possible spacecraft mass whilst arriving at a Low Mars orbit for data relay at the time of the SRL landing. This requires optimization of the launcher escape velocity and declination, the chemical propellant for the Mars Orbit insertion burn, and the Xenon propellant mass and duration of the transfer and further spiral-down to the data relay orbit[9]. For some scenarios, it might be required that the EP system is switched off to repoint the spacecraft for providing data relay.

After the MAS is launched, the ERO will determine the orbit of the OS (using optical images[10] of the MAS and OS as well as RF signals[11] from the MAS). Following this, the ERO will use its EP system to maneuver to within a few kilometers of the OS[12]. From here, the ERO will use its RCS system to for the proximity operations, culminating with the capture of the OS.

For the return journey, the spacecraft has to spiral-up until the conditions are reached that allow leaving Mars with the desired heliocentric inbound transfer. It is of interest for the mission to be able to leave the low Mars orbit as late as possible in order to provide ample margins of time for the surface and MAS missions, as well as for the rendezvous operations, with the ultimate requirement to support MAS launch as late as Ls 180 deg. Later start of the heliocentric inbound transfer requires an increasing Xenon mass and involve significant penalty for the missed thrust. Mission analysis performs an overall optimization of the return journey including these effects such as to estimate the latest date to depart from the low Mars orbit.

A summary of the mission analysis results is given in Figure 7.

States	Hybrid Outbound				EP Inbound			
	Launch	Mars SOI	Post TOA	LMO Arrive	LMO Depart	Mars SOI	Earth SOI	Earth Return
Date	Sep-30-2026	Oct-03-2027	Oct-13-2027	May-17-2028	Oct-14-2029	Aug-11-2030	Sep-21-2031	Sep-21-2031
Mass [kg]	**6229**	5886	3603	3439	**3058**	2841	2645	2542
= Dry	**3294**	3294	2670	2670	**2418**	2418	2418	2418
+ Xe	**1048**	706	706	542	**517**	299	117	114
+ Biprop	**1639**	1639	0	0	**0**	0	0	0
+ RCS	**248**	248	227	227	124	124	110	10
Orbit	Vinf = 2.185	Vinf = 1.975	500 x 12500	400 x 400 Ls = 311	400 x 400 Ls = 204	Vinf ≈ 0	Vinf = 4.5	Ventry ≈ 12.1

Steps					(202 days)		
	E-M Cruise	MOI+TOA	Spiral Down	At Mars	Spiral Up	M-E Cruise	E Approach
Duration [d]	368	10	218	515	300	406	0
– Drop [kg]		624		252			
– Xe [kg]	342		163	26	217	182	3
– Biprop [kg]		1639					
– RCS [kg]		21		104		13	100
Info:							

Figure 7:Spacecraft state and timeline throughout the complete baseline mission

III. Overview of the ERO Electric Propulsion System

A. ERO Electric Propulsion System Requirements

The ERO EP requirements are driven by the Campaign timeline requirements and the launcher performance. In particular, the orbiter must be in an operational orbit around Mars by Ls = 0 (roughly 2 years after launch) in order to provide relay support to the EDL and later to the rover and lander missions on the Mars surface. Given the maximum escape velocity achievable by the Ariane 6.4 for the planned ERO mass and launch declination, this puts strong time constraints on both the EP Earth-Mars transfer and the spiral down legs. Equally, in order to allow the maximum surface time for the landed missions the Campaign requires the ERO to leave Mars operational orbit as late as possible (target Ls=180) again putting time pressure on the spiral up phase to meet the Mars departure window as well as to perform the Mars-Earth transfer in order to secure an Earth re-entry for the EES.

The duration of the Transfer and Spiral legs is a function of the launcher performance, spacecraft mass and total available thrust. Given the selected Ariane 6.4 launcher the launch mass is limited to a little over 6 tons (for the v_{inf} and declination required for the ERO transfer). Detailed mission analysis has shown that with a 41 kW Solar Array (equivalent performance at 1 AU) a continuous thrust range of 130 mN – 250 mN, with a thrust resolution of better than 1 mN, is required to guarantee maximum thrust and Isp is achievable throughout the EP phases in a way that minimizes the number of required thrusters and maximizes individual thruster lifetime as a function of available power. A single chain architecture is selected for the SEP system without cross-strapping to reduce complexity and verification overhead. The redundancy concept relies on carrying a full spare thruster chain (thruster, power processing unit and propellant flow control unit). In order to meet the mission requirements, up to 3 simultaneously firing thrusters are required at any given time. The ERO concept therefore baselines 4 thrusters, where 1 is spare. In practice, all 4 thrusters will be utilized cycling through in groups of up to 3 thrusters at a time in a manner to share life time across all thrusters and minimize wear.

At sun distances of less than 1.2 AU it is expected that there is sufficient power and the spacecraft architecture is such that 4 thrusters can be fired simultaneously increasing total thrust and therefore robustness against missed thrust or other potential anomalies during the Mars-Earth transfer (although the mission design does not rely on this).

Given the EP system is powered only by solar energy, and the power requirements of a firing EP system do not allow for extended operation on battery power in eclipse (assuming a reasonable battery size), during the spiral down/up phases ERO can only generate thrust in the illuminated phases of each orbital period. In the initial stages of the spiral down, or later stages of the spiral up, the orbit is highly elliptical and eclipses represent only a small fraction of the total orbital period. Non-thrust periods during eclipse, therefore, have a relatively minor impact. However, as the spiral is circularized and the apoares is reduced eclipses represent an increasingly significant portion of the orbital period. The problem here is two-fold. Firstly the SEP cannot be operated at full (or even reduced) power in eclipse without an unfeasibly large battery and secondly SEP switch-on time from cold could require a large fraction of the already limited period of illumination per orbit. These two factors combined become a major constraint for the mission design because orbits close to Mars offer only a relatively short amount of time in which to apply useful orbit lowering/raising thrust between eclipses. This puts pressure on the time required to complete the spiral phase. The ERO Study Team is investigating various options to overcome this limitation together with the EP providers.

To complete all EP phases of the mission, the analysis currently assumes each thruster must process around 350 kg's of Xenon (which is worst case assuming loss of 1 thruster at launch). This translates to about 20,000 hours of firing time assuming a mission average thrust of around 200 mN.

B. Overview of the ERO Electric Propulsion System

The need for quasi-continuous thrust modulation over a broad range of thrust at a (constant) high specific impulse requires fine flow control of the Xenon propellant and closed loop control of the plasma discharge power. In order to achieve this the ERO Electric Propulsion System architecture foresees the following elements:

- *Thruster/Discharge Unit:* Depending on the type of the thruster a cathode/anode or a RF generator (RFG)/coil combination is used to generate the discharge plasma. High-Voltage grids extract positives ions from the plasma discharge to create thrust. In Kaufman/Ring Cusp type thrusters the discharge is created by the collision of electrons emitted from a cathode with the propellant gas; the current flow is controlled by a positively charged anode. A magnetic field confines the plasma, reducing losses and controlling the plasma density profile. In RF thrusters, an RFG converts DC current into the required AC current for the coil mounted on the outside the thruster discharge chamber, inductively coupling to the discharge plasma electrons. The RFG includes a matching network to ensure maximum power transfer to the plasma at each thruster operating point.
- *Neutraliser:* Gridded ion thrusters extract and accelerate solely positive ions. Therefore the thruster's ion current has to be compensated with an equivalent amount of electron current. Hollow Cathode type Neutralisers are normally used for this purpose.
- *Power Processing Unit:* Operation of the thruster requires both positive and negative High Voltage power supplies to charge the grid system and a discharge power supply for the plasma discharge unit. The PPU also includes all necessary auxiliary power supplies for the unit's internal power buses as well as a power and signal lines to drive the Flow Control Unit (FCU) and Neutraliser. The PPU will be supplied by both 100 V and 28V regulated power buses. The PPU is planned to have a relatively high level of autonomy for quick reaction to thruster state changes but also interfaces with the spacecraft On Board Computer via 1553 MILBus to receive high level commands and to transmit telemetry.
- *Propellant Management System:* The system includes Xenon storage tanks and a propellant pressure/flow management system. The propellant flow management consists of two steps: Pressure regulation and flow control. The pressure regulator reduces the high pressure inside the Xenon tanks down to typically a few bar for input to the Low Pressure Flow Control System. Each FCU provides finely controlled Xenon flow to a Thruster and Neutraliser pair.

C. Candidate Electric Propulsion Systems

Taking into account the requirements of the Electric Propulsion system, especially high Isp, very high lifetime, relatively high thrust (over 200mN) and the programmatic need for high Technology Readiness Level (TRL) for all subsystem elements, the choices available in Europe and worldwide are limited. Both Gridded Ion Engines (GIE) and Hall Effect Thruster (HET) systems were investigated.

With-in Europe two GIE options meet the aforementioned requirements, namely the Ariane Group RIT-2X and the QinetiQ T6. The RIT-2X thruster is under development and qualification for commercial GEO applications and is being adapted for interplanetary applications at ESA in preparation for MSR. The T6 benefits from recent BepiColombo flight heritage and its development program but needs to be modified for operation at higher thrust levels and longer lifetime. A separate technology development program is running to prepare the T6 as a possible candidate for MSR.

D. Thruster Modelling and Life Time Assessment

For the Phase B1 of the Earth Return Orbiter of Mars Sample Return (MSR-ERO) a numerical model has been developed to predict the performance values of the Ariane Group RIT-2X thruster for its potential use in MSR-ERO.

The RIT-2X thruster is under development and qualification for GEO missions. The performance parameters, therefore, are adapted to the requirements of commercial applications, which translate to high thrust, low specific impulse for Orbit Raising and a (very) low thrust high/moderate specific impulse for North/South Station Keeping (NSSK).

These performance values are not suitable for the ERO mission, which requires high thrust together with high specific impulse. However, the thruster is well capable to cover the necessary thrust/Isp/power/lifetime performance matrix for MSR, without major changes in system design, including its subsystems (PPU, Neutraliser, Xenon FCU).

The mathematical model predicts the performance values of the thruster (thrust, specific impulse, power, taking estimated lifetime into account) based on the preliminary ERO requirements. Preliminary engineering testing has confirmed the accuracy of the model.

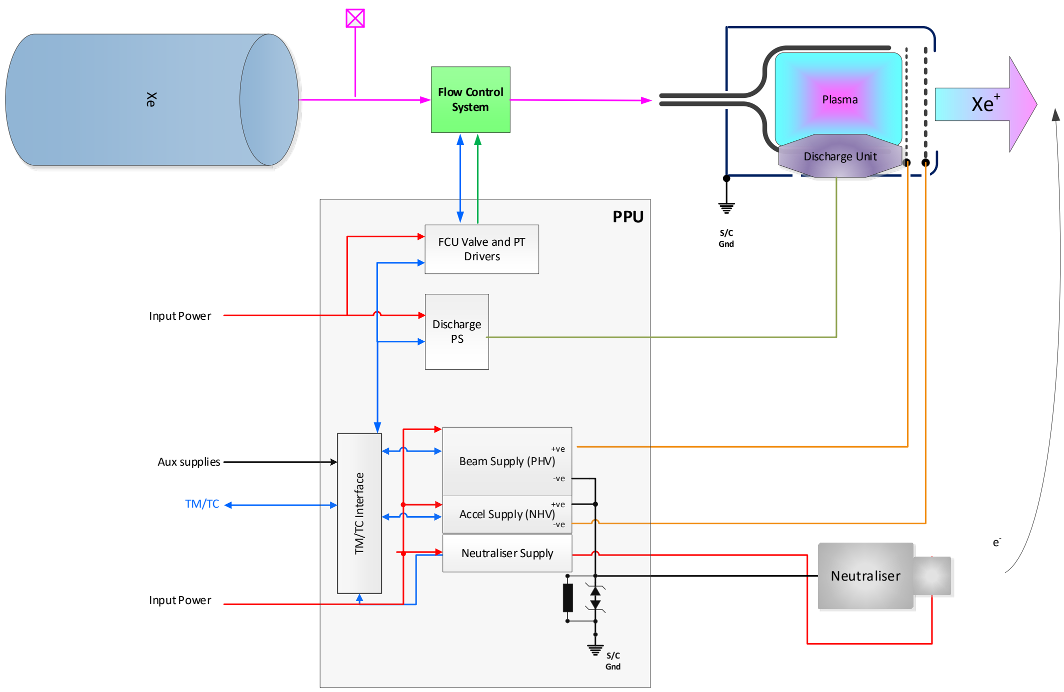

Figure 8: Diagram of possible ERO EP architecture assuming a GIE system.

IV.Conclusions

ESA is studying an Earth Return Orbiter concept to return Martian samples back to the Earth's surface as part of NASA's potential future Mars Sample Return Campaign. The concept implements a hybrid EP/CP propulsion approach, relying on both high power, long life time, gridded ion engines as well as a 1 kN bi-propellant chemical system. EP is used for the Earth-Mars and Mars-Earth transfers, spiraling down/up from Mars operational orbit as well as phase synchronization, orbit matching and rendezvous with the Orbiting Sample whilst CP is employed for the Mars Orbit Insertion.

Several orbiter concepts were studied including full chemical and full electric propulsion options. Whilst these options were found feasible under certain strict conditions (e.g. bigger launcher), it was found that the hybrid propulsion approach provided the best compatibility to the Campaign requirements and flexibility to future evolutions, whilst being compatible with Europe's Ariane 6.4 launcher.

Gridded Ion Engines are baselined for the ERO concept and both the Ariane Group RIT-2X and QinetiQ T6 systems are under consideration. Specific engineering tests and numerical modelling is in progress to assess the performance and life time of the thrusters.

Acknowledgments

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