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*Review Article*

# Small Mars Mission Architecture Study

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While the vast majority of ESA's funding for Mars exploration in the 2020s is planned to be invested in ExoMars and Mars Sample Return, there is an interest to assess the possibility of implementing a small mission to Mars in parallel with, or soon after, the completion of the MSR programme. A study was undertaken in the Concurrent Design Facility at ESA ESTEC to assess low-cost mission architectures for small satellite missions to Mars. Given strict programmatic constraints, the focus of the study was on a low-cost (<250MEuro Cost at Completion), short mission development schedule with a cost-driven spacecraft design and mission architecture. The study concluded that small, low-cost Mars missions are technically feasible for launch within the decade.

## 1. Introduction

ESA's current Mars exploration programme consists of the flying orbiters Mars Express and the ExoMars TGO, while the ExoMars rover *Rosalind Franklin* is planned for launch in 2022. The Nov 2019 ESA Council of Ministers meeting, Space19+, approved ESA contributions to a Mars Sample Return programme, led by NASA, with a launch of the sample retrieval missions planned to occur as early as 2026. ESA's primary MSR contributions include the Earth Return Orbiter as a dedicated mission, and the Sample Fetch Rover and Sample Transfer Arm of the NASA-led Sample Return Lander mission.

While the vast majority of ESA's funding for Mars exploration in the 2020s is planned to be invested in ExoMars and Mars Sample Return, there is an interest to assess, at Phase 0-level, the possibility of implementing a small mission to Mars in parallel with, or soon after, the completion of the MSR programme, to further the exploration of Mars in areas not addressed by MSR.

Missions to Mars at the small scale have not been greatly studied within ESA since Mars Express two decades ago and preliminary concepts for a Mars Micro Mission as an "Arrow" mission of the Aurora programme [1]. Since then, the landscape of technologies (in particular those relevant for small Low Earth Orbit platforms and instrumentation) and launch capabilities (e.g., rideshares) have matured significantly, offering promising new opportunities for low-cost implementations of interplanetary missions. The ESA programmatic framework with the advent of the Aurora programme, now European Exploration Envelope Programme (E3P), and approach to low-cost planetary missions has thus evolved over the years [2–5].

A study was undertaken in the Concurrent Design Facility at ESA ESTEC to assess low-cost mission architectures for small satellite missions to Mars. Given strict programmatic constraints, the focus of the study was on a low cost, short mission development schedule and with a cost-driven spacecraft design and mission architecture.

This paper presents an overview of the mission architectures considered and the results of mission and system-level design trades used to select a reference scenario for each mission case.

The final report of the CDF study [6] provides further details of all the subsystem design and performance analyses carried out by the team.

## 2. Mission Architecture

*2.1. Approach to Mission Architecture Assessment.* Due to the unique opportunity from which this study originates, the initial driving constraints were largely programmatic (rather than scientific or technical) and do not impose any particular kind of mission architecture. With a launch timed to complement the upcoming Mars Sample Return missions at the end of the 2020s, applying these constraints would mean having an opportunity for a European mission to Mars that provides *in situ* science data return to the Mars science community at a time when currently no new *in situ* science data are expected, as well as serving exploration goals towards the preparation of human Mars exploration.

Given the large trade space of potential options that could be considered for such a mission, a strategy with which to approach the mission architecture definition was devised. Initially, an assessment of the programmatic and cost constraints was developed into a set of high-level mission requirements and the design drivers formulated. A consultation with ESA Martian science experts revealed some mission themes that were turned into three distinct mission cases. These mission cases are representative of three different types of missions and are general enough to cover a wide variety of scenarios that scientists might like to see in a small Mars mission.

The architectural trade space was then analysed for key components of the mission architecture, such as launch scenario and the means of transfer to Mars. Various trade-offs were conducted at mission level to condense the options into a set of reference mission scenarios; one for each mission case.

*2.2. Programmatic Constraints and Design Drivers.* The following programmatic constraints were used to further limit the scope of study:

(i) The mission should be designed to cost

(ii) The project should envisage a fast development time, where a project phase of 4-5 years is considered

(iii) The time of transfer to final Mars orbit should be constrained to 3 (Earth) years

(iv) To limit the need for extensive developments, only equipment and units that can reach TRL 7/8 by PDR will be considered

The above constraints indicate that this mission is heavily cost and schedule driven. The study therefore aimed to address what can be achieved for a certain cost when there are no initial performance requirements placed on the resulting space segment. Whilst these design drivers remain the priority for ensuring a low-cost, short development time mission, it is still important that there is useful and attractive science return available from the mission. With this in mind, the science team helped to guide the evolution of the spacecraft design, ensuring that valuable science could be produced within the mass, power, and data envelopes under consideration and suggesting representative target orbits at Mars that would enable such missions, as well as reasonable targets for minimum payload allocations.

*2.3. Selection of Mission Cases.* Mission cases were selected to represent a broad range of missions that are of current interest to the ESA Mars Exploration Programme. Three representative mission cases were selected for study and are illustrated in Figure 1:

(1) *Mars Communications Constellation*. A three satellite constellation with an objective to provide data relay with continuous coverage to ground assets. The satellites also contain secondary science instrumentation.

(2) *Mars Science Orbiter*. A single science orbiter with a primary science objective and a secondary objective to provide a data relay.

(3) *Mars Hard Lander*. A demonstration mission of a carrier module and a number of hard landers.

*2.4. Architecture Options.* The study assessed a wide range of mission architectures. Given the wide scope of architectures that would be available for a cost-driven mission to Mars and the various mission cases under study, there are many potential options for the launch and transfer scenario. In order to condense these options, an initial qualitative assessment was made of launch vehicles, initial orbit injection options, and propulsions technologies that are likely to be available in the given timeframe.

The initial orbit into which the spacecraft is injected dictates the $\Delta V$ requirements needed for transfer to Mars. Options put forward for trade-off were LEO, GTO, the Earth-Sun L2 point, and direct trans-Mars Injection (TMI). The launch vehicles and corresponding injection orbits depicted in Figure 2 were considered. Whilst other similar options may become available opportunistically, the launch scenarios given here may be considered as representative for alternative launch vehicles of similar cost and performance.

In the case that direct trans-Mars Injection cannot be provided by the launch vehicle (e.g., due to insufficient performance or in case of a rideshare scenario), the spacecraft requires the capability to transfer to Mars from an initial geocentric orbit of its own accord. There are fundamentally three options available. The propulsion architecture options put forward for trade-off are shown in Figure 3.

Commercial rideshare options to GTO from which the satellite(s) would transfer to Mars using either chemical or electric propulsion were evaluated alongside dual and dedicated launch scenarios. The notable launch and transfer scenarios considered are depicted in Figure 4.

SEP-based transfers departing from LEO were not assessed as these were initially considered to take too long to achieve Earth escape conditions and impose significant radiation exposure on the spacecraft through extended dwelling in Earth's Van Allen Belts. Similarly CP-based transfers to LMO were assumed to necessitate aerobraking to limit overall mission $\Delta V$ requirements.

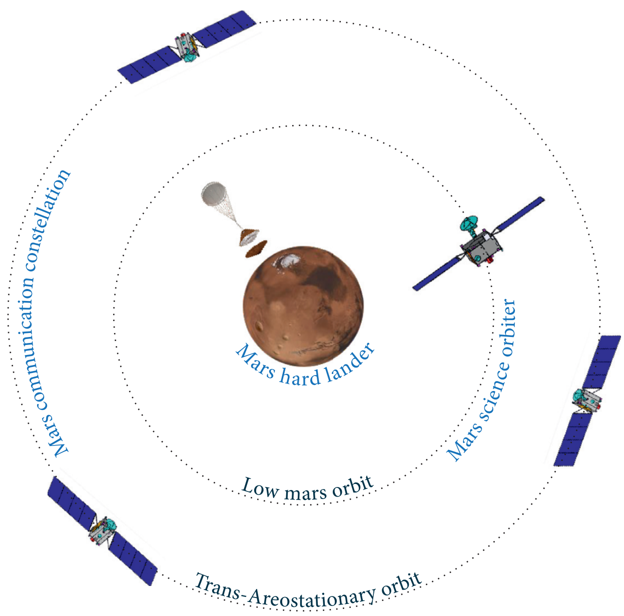


Figure 1: An overview of the three different mission cases studied in the CDF (not to scale).

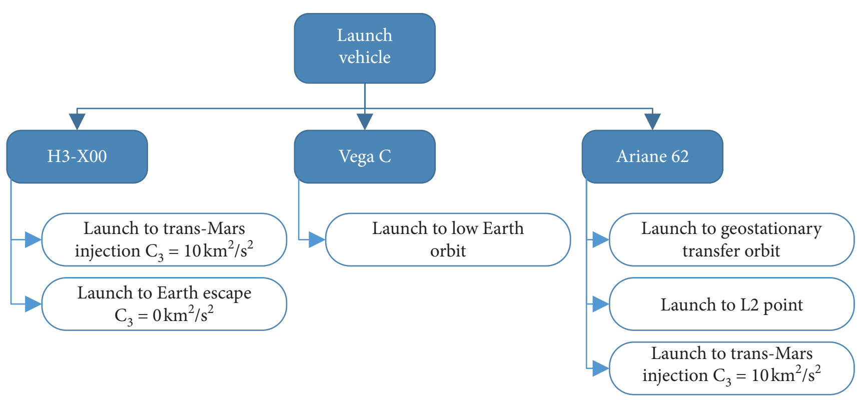


Figure 2: An overview of the launch vehicle options and corresponding injection orbits considered for study.

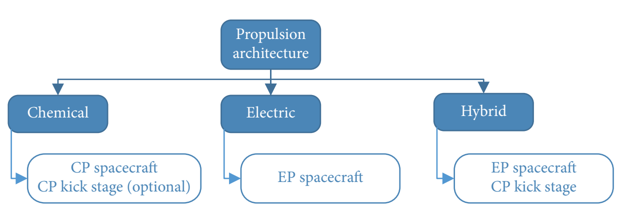


Figure 3: An overview of the propulsion architecture options considered for study.

Hybrid concepts were initially considered; however, these were later excluded, as the overall costs for the development of both a chemical kick-stage and a dedicated electric propulsion system were considered prohibitively high for the mission scenario.

An initial trade-off was made for each of the remaining combinations of launch and transfer options based on cost, performance, availability, operational complexity, and transfer time to Mars. From these results a reference architecture was selected for each mission case. The $\Delta V$

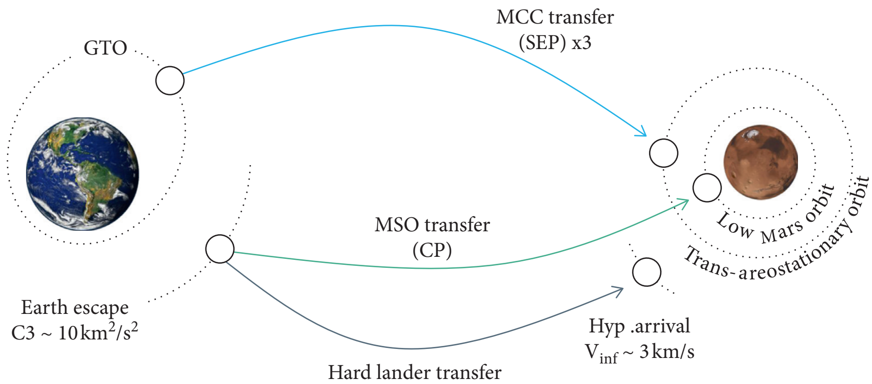


Figure 4: Overview of the propulsion architecture options considered for study. Solar electric propulsion (SEP) and chemical propulsion (CP) transfers are considered to reach low Mars orbit (LMO) and specific areosynchronous (AEO) orbits such as Trans-Areostationary (TASO).

combinations considered for both the Chemical Propulsion (CP) transfer cases and the Solar Electric Propulsion (SEP) transfer cases are provided in Tables 1 and 2.

## 3. Mars Communications Constellation Mission

Despite the long range, high altitude orbits are considered to be particularly useful for providing data relay to surface and orbiter missions due to their ability to provide long access times. This is complementary to the short range, short duration passes afforded by orbiters at low altitude. Areosynchronous orbits may be especially useful for this purpose since they have an orbit period similar to a Martian sol and are therefore a good candidate for the provision of continuous coverage of assets. Notably, such class of orbits can provide continuous coverage with far fewer spacecraft than would be required for a low Mars orbit (LMO) constellation.

Satellites in an areostationary orbit are subject to natural perturbations, which will incur added station keeping costs to maintain the spacecraft within prescribed mission required boundaries. There exist four regions of longitudinal stability for areostationary satellites, which require minimal station keeping costs [7]. However, these locations are evenly distributed and continuous coverage from these points can only be ensured by a constellation of minimum four spacecraft.

Following an initial trade-off on orbit design, a Trans-Areostationary Orbit (TASO) was selected for the Mars Communication Constellation (MCC) mission. This type of orbit has a slightly greater semimajor axis than areostationary orbits and allows the constellation to drift slowly around Mars, maintaining continuous coverage of the surface with minimal station keeping costs. Uninterrupted coverage is provided up to latitudes of $\pm 70^{\circ}$ for an elevation angle of $10^{\circ}$, as shown in Figure 5.

A Trans-Areostationary Orbit enables near-global simultaneous and continuous full-disk observation up to high latitudes (excluding the polar regions) of Mars. The view from the three satellites allows the monitoring of dynamical phenomena rapidly evolving in space and time. Various science and exploration knowledge gaps could be addressed from the vantage points such as the exchange between the surface and the atmosphere (e.g., energy and mass balance), atmospheric phenomena (e.g., dust storms, water, and $CO_2$ clouds), interaction of solar wind with Mars's upper atmosphere, and the Martian moons [8]. Figure 6 shows the science activities enabled at different orbit configurations, target distance, and a range of angular resolutions of any imaging instrument.

The primary payload of the Mars Communications Constellation mission is the telecommunications package; however, an allocation is also made for a secondary science payload suite.

To select the science objectives, European and US priorities as stated in the MEPAG Science Objectives [10] were considered, with the aim of closing as many knowledge gaps as possible. Additionally, there is an aim to fit the science objectives to instruments that would be useful from a 17,600 km altitude TASO orbit, i.e., without the need for fine spatial resolution, considering payloads having high TRL and low mass and benefitting from the near-global (excluding the polar regions) simultaneous and continuous view from the three satellites, and observations throughout the full diurnal cycle.

The rapidly evolving dynamics of meteorological phenomena such as dust storms (timescale spans from a few hours to months) and water/$CO_2$ ice clouds (timescale spans from half an hour or less) could extend from mesoscale up to the planetary scale. They affect the energy balance and the distribution of aerosols in the atmosphere and support an argument for continuous and simultaneous observations across the planet [8].

The solar radiation energy balance at the surface depends on local topography, albedo, and spatial and temporal variations of atmospheric aerosols, which results in rapid changes of the lower atmospheric column. The mechanism of dust lifting, vertical mixing, transportation, and sedimentation are dependent on the diurnal variability [11]. Transportation of dust can reach the mid atmosphere within hours [12] and can significantly grow by a factor of 10-20 in area in a week or two ([13–16]). Water and $CO_2$ ice clouds form in topographic lows such as canyons and large impact

Table 1: Overview of the approximate transfer ΔV (m/s) assumed for different mission options for chemical propulsion (CP) concepts.

| Transfer from | Transfer to: Hyperbolic arrival orbit | 4-Sol orbit | TASO | LMO | LMO (with aerobraking) |
|---|---|---|---|---|---|
| 4-Sol orbit | n/a | n/a | 640 m/s | 1,380 m/s | 270 m/s |
| MTO/TMI | n/a | 1,100 m/s | 1,740 m/s | 2,480 m/s | 1,370 m/s |
| GTO | 1,980 m/s | 3,080 m/s | 3,720 m/s | 4,460 m/s | 3,350 m/s |
| LEO | 3,970 m/s | 5,070 m/s | 5,710 m/s | 6,450 m/s | 5,340 m/s |

Table 2: Overview of the approximate transfer ΔV (m/s) assumed for different mission options for solar electric propulsion (SEP) concepts.

| Transfer from | Transfer to: Earth escape $C_3 \sim 0\,km^2/s^2$ | Mars arrival $V_\infty \sim 0\,km/s$ | TASO | LMO |
|---|---|---|---|---|
| Mars arrival $V_\infty \sim 0\,km/s$ | n/a | n/a | 1,000 m/s | 3,000 m/s |
| Earth escape $C_3 \sim 4\,km^2/s^2$ | n/a | 4,000 m/s | 5,000 m/s | 7,000 m/s |
| Earth escape $C_3 \sim 0\,km^2/s^2$ | n/a | 5,700 m/s | 6,700 m/s | 8,700 m/s |
| GTO | 3,700 m/s | 9,400 m/s | 10,400 m/s | 12,400 m/s |

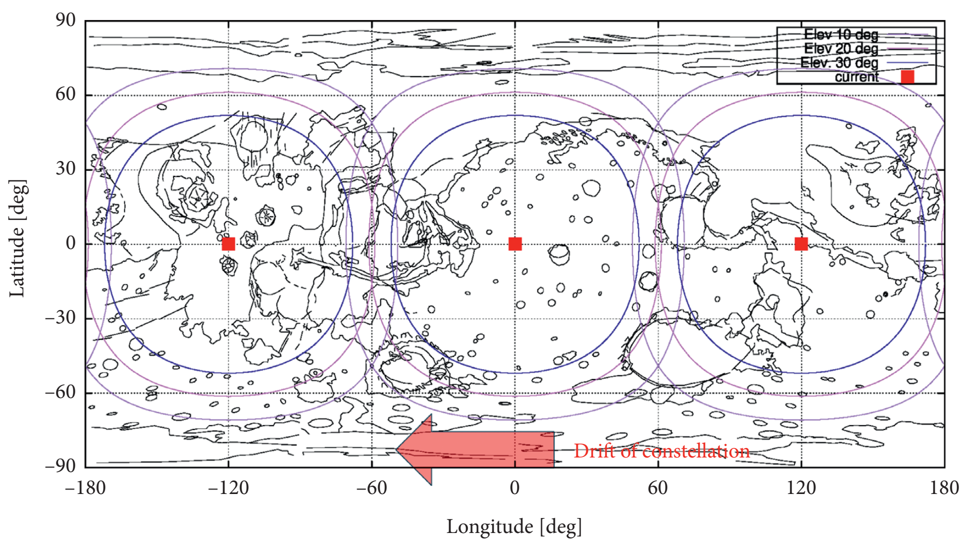


Figure 5: A Trans-Areostationary Orbit three-satellite constellation has Mars coverage up to latitudes of ±70°.

basins during the night and dissolve in the morning, implying an important exchange between the atmosphere and the regolith [17]. Additionally, cloud-tracking can provide information about the dynamic nature of dust storms and clouds.

The main science goal of the mission is to understand the present-day climate and dynamics of atmospheric processes. The following science objectives were considered:

(i) OBJ-01: characterize the volatiles (e.g., water and $CO_2$ clouds) and dust exchange (e.g., dust storms) between the surface and atmospheric reservoirs

(ii) OBJ-02: determine the spatial and temporal variation of key atmospheric gases

(iii) OBJ-03: measure the energy balance of the atmosphere

The science objectives (A2/1; A3/1, 2; A4/1, 3) from MEPAG Goal II (Atmospheric Science) [10] can be at least partially addressed by the proposed MCC mission.

Trade-offs were also made on the configuration of the spacecraft constellation during transfer, including consideration of three independent spacecrafts, a mother/daughter craft configuration, and the use of a disposable kick stage. Chemical, electric, and hybrid propulsion scenarios were all considered.

The resulting reference launch and transfer scenario for this mission is to utilise an Ariane 62 rideshare to a geostationary transfer orbit (GTO) and each of the three satellites will transfer to Mars independently, by means of onboard electric propulsion. This scenario, along with approximated ΔV needs, is illustrated in Figure 7.

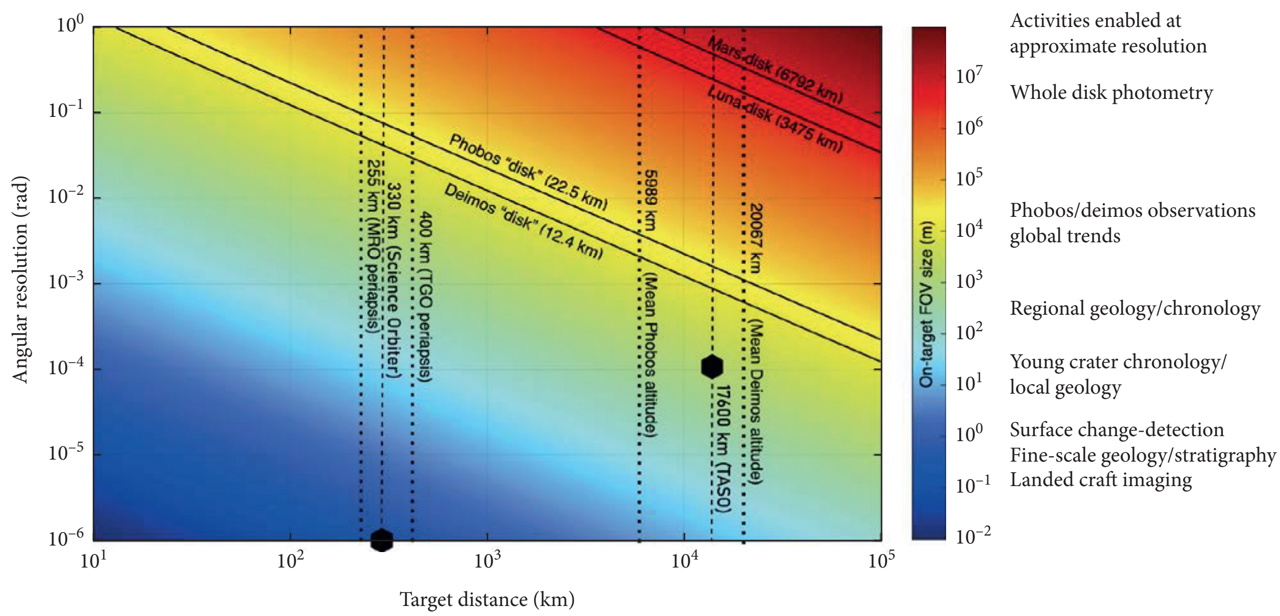


FIGURE 6: The science activities enabled at different orbit configurations; target distance is plotted against the angular resolution (=imaging capability of a camera). TASO at 17600 km altitude with current system requirements ($10^{-4}$ angular resolution of the optical instrument, marked with black hexagon) enables full-disk monitoring and observation of regional processes (e.g., dust storms and clouds). The Science Orbiter (see Section 4) at 320 km altitude with current system requirements ($10^{-6}$ angular resolution of the optical instrument, marked with black hexagon) supports high-resolution imaging of the surface (e.g., fine-scale geology, mineralogy, resources, and high-resolution topography), figure modified from [9].

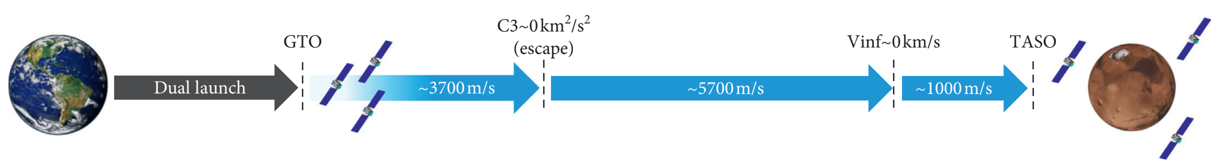


FIGURE 7: Launch and transfer scenario for the Mars Communication Constellation mission.

The end-state architecture of the constellation, including the communications concept, is illustrated in Figure 8. The satellite constellation must support communications with the current surface assets (rovers, probes, etc.) and therefore the Ultra High Frequency (UHF) band is selected as the communications link between the constellation and surface assets. Upgrades to S-band or X-band, to be in line with potential future mission needs, are to be investigated in the next design phases.

The instruments and their associated science objectives are given in Table 3.

The three satellites in the constellation are identical. A preliminary design exercise shows that a wet mass of ~610 kg per satellite is feasible within the programmatic constraints. This allows for a total launch mass that is consistent with an Ariane 62 dual launch opportunity to GTO and has a transfer time to Mars of 2.13 years. Mass reductions could be envisaged if the requirement to have a technology readiness level of 7/8 by PDR is relaxed or if a later launch opportunity is used.

The mass constraints placed by a dual launch opportunity in conjunction with the programmatic constraints mean that the performance of the telecommunications package is relatively low given the long range of the TASO orbit. This performance could be optimized, but it is important to note that even a low data rate link with continuous coverage fills an existing data gap. Mass budget information is provided in Table 4.

## 4. Mars Science Orbiter Mission

Following an initial trade-off on orbit design, a 320 km mean altitude low Mars orbit that is sun synchronous (SSO) was selected for the Mars Science Orbiter (MSO) mission. This orbit was selected because it does not have a strong synchronicity between orbital period and Mars rotation. This means that the ground track tightly covers the entire Mars surface after 7 days. Lower altitude orbits result in lifetime and planetary protection issues, whilst significantly higher orbits diminish the resolution of science data.

Trade-offs were also made on the Mars transfer scenario. Chemical, electric, and hybrid propulsion architectures were all considered as well as chemical kick stages and direct injection scenarios. Using a chemical kick stage would be an

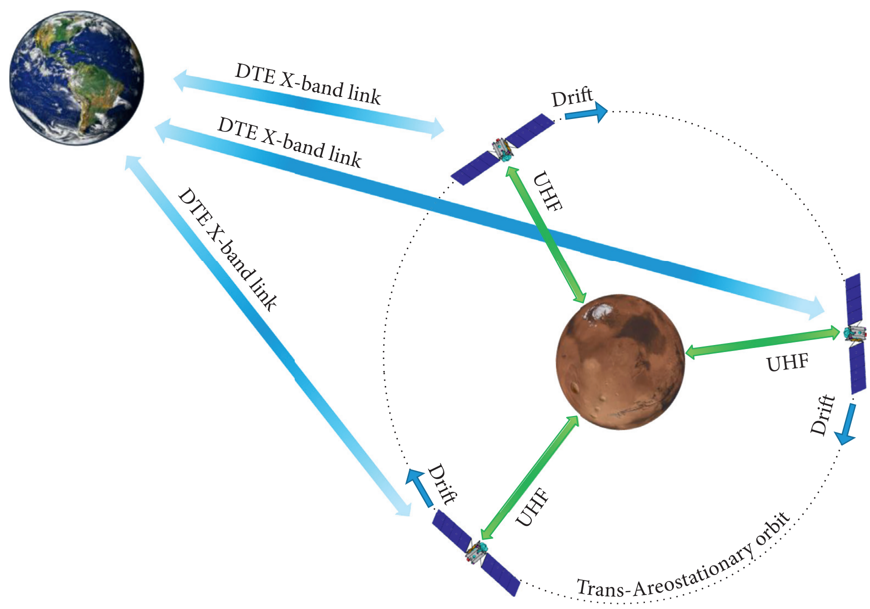


Figure 8: End-state architecture and communications concept for the Mars Communication Constellation mission.

Table 3: Science payload suite and associated objectives for the Mars Communications Constellation.

| Instrument | Heritage | FOV (°) | Spatial Res. at TASO (m) | Science objective | Ref. |
|---|---|---|---|---|---|
| Wide-angle camera imaging suite (VIS) | VMC (Venus Express) | $25 \times 25$ | 3840 | Daily weather monitoring, dust storm, clouds | [18, 19] |
| Wide-angle camera imaging suite (NIR) | VMC (Venus Express) | $25 \times 25$ | 3840 | Atmospheric composition $O_2$, $H_2O$, CO, $CO_2$, N-species | [18, 19] |
| Wide-angle camera imaging suite (UV) | VMC (Venus Express) | $25 \times 25$ | 3840 | Ozone (250–270 nm), aurora effects | [18, 19] |
| Thermal IR radiometer | MARA (MASCOT) | $5 \times 5^*$ | $1.5E6^*$ | Temperature of the atmosphere | [20] |

Each satellite in the constellation contains an identical science payload suite. *An orbital version of MARA would need a modified optical design to obtain a $5^\circ \times 5^\circ$ FOV instead of the original $18^\circ \times 18^\circ$. An upgrade to thermal IR imaging capability would be preferred if resources allow.

attractive option and one that would enable a rideshare launch to GTO. However, the costs and complexities involved in repurposing existing technologies to meet the mission requirements mean that it is no more costly and also programmatically simpler, to use a dedicated Ariane 62 launch to its full capability [21] with a large launch margin remaining for other opportunities.

The resulting reference launch and transfer scenario for this mission is thus to utilise a dedicated Ariane 62 launch to Earth escape, with the MSO satellite performing Mars Orbit Insertion (MOI) by way of an on-board chemical propulsion system. This scenario, along with estimated $\Delta V$ needs, is illustrated in Figure 9.

The end-state architecture of the constellation, including communications concept, is illustrated in Figure 10.

To select the science objectives, European and US priorities as stated in the MEPAG Science Objectives were considered [10]. The Mars Science Orbiter enables high-resolution imaging of the surface (e.g., fine-scale geology, mineralogy, resources, and topography) at two local times above the ground. High spatial resolution data (e.g., optical images, NIR infrared spectral data, thermal infrared, and terrain models) cover only a few percent of the surface. To better understand the planetary evolution of Mars and facilitate the selection of a scientifically rich and resource-rich landing site for human exploration missions, it is essential to fill the gaps in the spatial coverage and provide higher resolution data than is currently available.

Understanding the current distribution and form of water (e.g., liquid surface water, deep aquifers, water ice, and mineral-bound water) on the surface and in the subsurface of Mars is critical for interpreting the past aqueous history and the related paleoclimate. Orbital assets have identified various locations with hydrated mineral deposits such as phyllosilicates, sulfates, iron hydroxides/oxyhydroxides, carbonates, zeolites, and opal (e.g., [22, 23]). However, there is an uncertainty in the abundance, the composition variations at metre-scale resolution, and the mechanical properties of these deposits. This knowledge would be needed for in-situ resource utilisation (ISRU) purposes.

Additionally, the thermophysical properties of the regolith can provide information about the composition, grain

Table 4: Preliminary mass budget information for the Mars Communication Constellation mission.

| | | Mass (kg) |
|---|---|---|
| Attitude, orbit, guidance, navigation control | | 8.0 |
| Chemical propulsion | | 6.9 |
| Communication (UHF proximity link, DTE link: X-band, 50 W TWTA, 1 m HGA) | | 30.1 |
| Electric propulsion (T6 engine) | | 70.4 |
| Instruments | | 7.0 |
| Power (2.8 kW @Mars for electric propulsion) | | 158.9 |
| Structures | | 56.6 |
| Thermal control | | 13.3 |
| Data handling (0.5 Gb/day science data) | | 5.3 |
| Harness | | 17.8 |
| *Dry mass* | | **374.3** |
| System margin | 20% | 74.9 |
| *Dry mass incl. system margin* | | **449.2** |
| CPROP propellant mass | | 4.7 |
| CPROP propellant margin | 2% | 0.1 |
| EPROP fuel mass (10.4 km/s $\Delta V$) | | 153.8 |
| EPROP fuel margin | 2% | 3.1 |
| *Wet mass (per satellite)* | | **610.9** |
| *Constellation total dry mass incl. system margin* | | **1347.6** |
| *Constellation total wet mass* | | **1832.7** |
| Launch adapter | | 360 |
| *Launch mass (wet mass + adapter)* | | **2192.7** |

The bold values refer to summation values of the figures above them.


Figure 9: Launch and transfer scenario for the Mars Science Orbiter (MSO) mission.

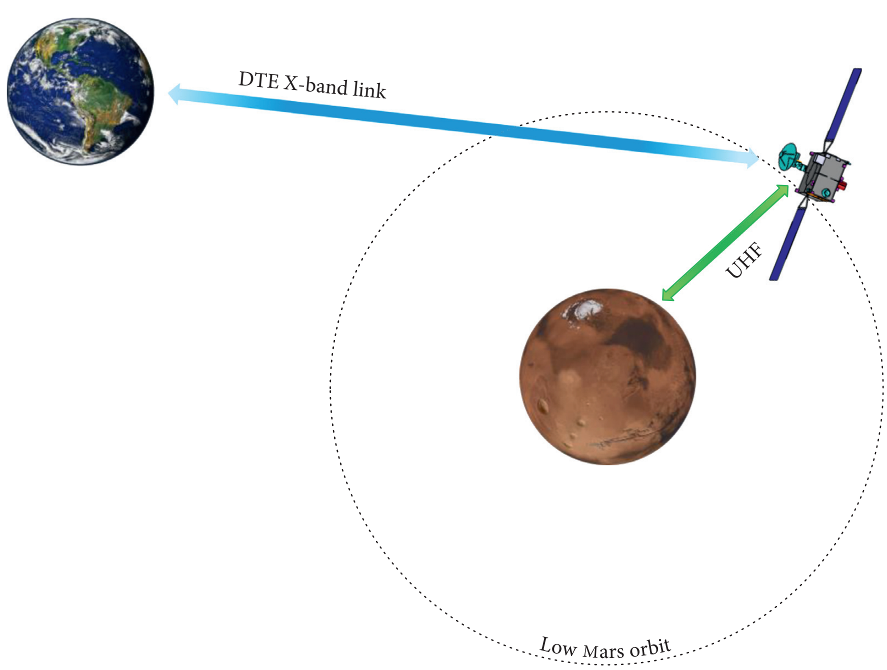


Figure 10: End-state architecture and communications concept for the Mars Science Orbiter mission.

sizes, rock distribution, surface roughness, porosity, and geological history of the surface ([24, 25]). The diurnal and seasonal changes in surface temperature are controlled by the thermal inertia. Thermal inertia depends primarily on the physical structure of the surface layer and is defined as a function of the thermal conductivity, heat capacity, and material density.

Moreover, change detection on the surface can allow monitoring of the dynamics of current surface processes including dune and ripple migration, landslides, dust deposition, and the recent impact flux.

Thus, the primary science goal was selected to map the thermophysical properties and the composition of the surface, focusing on hydrous minerals and to characterize surface hazards (e.g., rock abundance and high slopes) for future landed missions. The secondary science goal was to observe changes on the surface (e.g., new impacts, activities) inferring the current impact rate, dynamics of the surface processes, and the exchange between the surface and the atmosphere. As a summary, the following science objectives were considered:

(i) OBJ-01: characterize the thermophysical properties of the surface

(ii) OBJ-02: determine the spatial distribution of hydrated minerals on the surface

(iii) OBJ-03: constrain the timeline of geological history and habitability of Mars

(iv) OBJ-04: characterize surface hazards (e.g., rocks, slopes and incoherent material) to landing human scale systems

The science objectives from MEPAG Goal III (understand the origin and evolution of Mars as a geological system) (A1, A2, A3, A4) and Goal IV (prepare for human exploration) (A3/1, 2; C2/1) [10] can be addressed by the proposed MSO mission.

The primary payload of the Mars Science Orbiter mission is the science payload; however, the satellite also includes a data relay capability of similar sizing to the MCC satellites. To select the science payload, high TRL instruments from ESA's heritage planetary missions and European contributions to non-ESA missions were taken into account. Trade-offs were performed between (1) mass vs. performance and (2) maintaining heritage and optimising for the mission. The instruments and their associated science objectives are given in Table 5.

A preliminary design exercise shows that a wet mass of ~600 kg is feasible within the programmatic constraints. This allows for a total launch mass that is well within the capabilities of a dedicated Ariane 62 launch and also allows consideration of potential rideshare opportunities for small satellites or CubeSats. Mass reductions could be envisaged if the requirement to have a technology readiness level of 7/8 by PDR is relaxed or if a later launch opportunity is used. Conversely, mass increases could also be considered if it were to result in a significantly improved performance, so long as the programmatic constraints are still met. Key mass budget information is provided in Table 6. Note that a larger systems margin (30%) is used for chemical propulsion-based spacecraft than for an electric propulsion-based spacecraft (20%). This is the systems margin philosophy taken at this early phase of study and is due to mass growth seen in similar chemical systems over their development lifecycle.

## 5. Mars Hard Lander Mission

The study of the Mars Hard Lander (MHL) mission focussed mainly on mission analysis, entry, descent and landing (EDL), and cost. The reference mission architecture was chosen to comprise three hard landers, each with an entry mass of 70 kg and aiming to land 50 kg on the surface with an impact velocity of less than 20 m/s.

The entry mass of 70 kg draws large similarities with the *Beagle 2* lander on ESA's Mars Express mission (2003) which performed a semihard landing [31]. The study focussed on analysing if all the requirements could be met using the same aeroshell, parachutes, and EDL control as *Beagle 2*. The impact velocity requirement of less than 20 m/s was also derived from *Beagle 2* heritage. Using heritage equipment is in line with the main drivers for the mission: cost and schedule.

During the CDF study, the EDL trajectory and EDL equipment of the hard landers were analysed. Additionally, a high-level design of the carrier spacecraft was performed and a high-level cost estimation was made.

For the launch and transfer scenario, a dedicated Ariane 62 launch to $C_3 \approx 10\,\mathrm{km^2/s^2}$ was used as a reference case, putting the landers on a ballistic coast towards Mars hyperbolic entry. This scenario is illustrated in Figure 11.

The landers are carried by a chemical propulsion carrier module with a top-level design allocation provided by a reduced capability version of the Mars Science Orbiter satellite (with a lower $\Delta V$ capability, for example, since there is no need for a Mars orbit insertion manoeuvre). Three hard landers were studied for redundancy reasons and to enable science that benefits from simultaneous measurements in different locations (e.g., weather monitoring, seismology).

Modelling of the EDL trajectory shows that, for a reference Mars arrival date of 2$^{\text{nd}}$ October 2029 and an entry velocity of 5.6 km/s, a flight path angle (FPA) of between $-11^\circ$ and $-14^\circ$ allows the impact velocity requirement to be met (≤20 m/sec at 0 km MOLA).

Using the peak heat flux and total heat load calculated, it could be analysed whether the thermal protection system (TPS) used on *Beagle 2* would be sufficient for the MHL. Due to the use of a shallower FPA (*Beagle 2* entered at -15.8° FPA), the MHL exceeds the heat load that *Beagle 2* was designed for. Therefore, additional TPS material would need to be added to the aeroshell. The mass of the additional material ranges from 5.17 to 0.67 kg for a FPA of $-11^\circ$ to $-15^\circ$, respectively. The same pilot parachute and ringsail main parachute as *Beagle 2* are assumed and they have a diameter of 8 m and 10.4 m, respectively. Due to the high landing speed and high $g$-loads, a crushable attenuation structure is added to the lander. It was assumed to use the same material as ESA's Schiaparelli lander, which is an aluminium honeycomb sandwich structure. A trade between

Table 5: Science payload suite and associated objectives for the Mars Science Orbiter.

| Instrument | Heritage | FOV (°) | Spatial res. at 320 km altitude (m) | Science objective | Ref. |
|---|---|---|---|---|---|
| Thermal IR radiometer/ Imaging spectrometer, imaging spectrometer, radiometer | MERTIS (BepiColombo) | 4 × 4 | <200 (imaging spectrometer), ≤2000 m (radiometer) | Surface composition, temperature, thermal inertia, rock abundance, atmospheric science | [26, 27] |
| Visible imaging system | CaSSIS (TGO) | 1.35 × 0.85 | 3.2 | High resolution colour imaging of the surface, change detection, stereo imaging, geological context | [28] |
| NIR spectrometer | MacrOmega (Mars Moon explorer) MicrOmega (Phobos-Grunt, Hayabusa-2, ExoMars 2022 rover), | * | * | Mineralogy | [29, 30] |

The satellite also includes a data relay capability. * An orbital version of MicrOmega would need a different optical design as it would be focussed at infinity and use solar illumination; specification of the FOV and spatial resolution would depend on instrument-level design trade-offs.

Table 6: Preliminary mass budget information for the Mars Science Orbiter mission.

| | | Mass (kg) |
|---|---|---|
| Attitude, orbit, guidance, navigation control | | 8.0 |
| Chemical propulsion | | 41.4 |
| Communications (UHF proximity link, DTE link: X-band, 50 W TWTA, 1 m HGA) | | 34.3 |
| Instruments | | 34.2 |
| Power (310 W @Mars) | | 74.5 |
| Structures | | 63.4 |
| Thermal control | | 9.8 |
| Data handling (730 Gb science data volume) | | 5.3 |
| Harness | | 13.5 |
| *Dry mass SC* | | **284.4** |
| System margin | 30% | 85.3 |
| *Dry Mass SC incl. system margin* | | **369.7** |
| CPROP fuel mass (1370 m/s ΔV) | | 124.8 |
| CPROP fuel margin | 2% | 2.5 |
| CPROP oxidizer mass | | 97.7 |
| CPROP oxidizer margin | 2% | 2.0 |
| CPROP pressurant mass | | 1.0 |
| CPROP pressurant margin | 2% | 0.0 |
| *Total wet mass SC* | | **597.7** |
| Launcher interface | | 64 |
| *Launched mass* | | **661.7** |

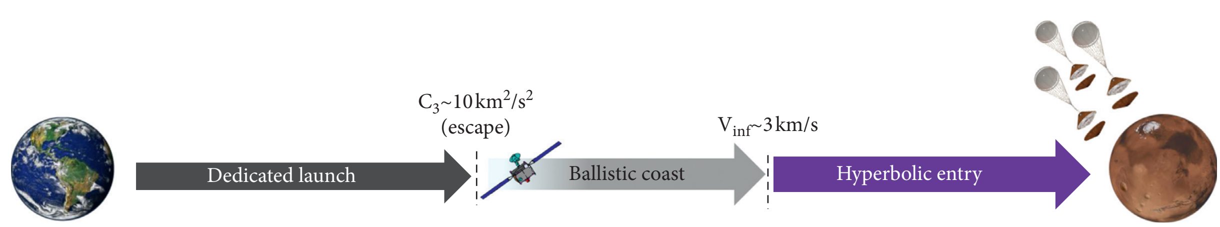


Figure 11: Launch and transfer scenario for the Mars hard lander mission.

the maximum allowable $g$-loads and the mass of the crushable structure led to a first estimation of allowable $g$-loads of 80$g$ and 120$g$ in the horizontal and vertical direction, respectively. Regarding EDL control, the same equipment as *Beagle 2* is considered for the Mars Hard Lander.

Using these assumptions, a preliminary mass budget is provided in Table 7. This budget is valid for an entry at flight path angles between −14° and −13°.

In order to achieve a spread in landing sites between the different hard landers, they need to be deployed from the carrier vehicle sequentially, whilst also leaving several

Table 7: Preliminary mass budget for the Mars Hard Lander mission.

| | |
|---|---|
| *Launch mass* | **513 kg** |
| Wet mass carrier vehicle | 303 kg |
| Entry probes (3) | 210 kg |
| *Entry probe* | **70 kg** |
| EDL system | 28.5 kg |
| Landed mass | 41.5 kg |
| Science payload allocation | ~9 kg |

days between each lander deployment for navigation corrections and to adjust and confirm accuracy of the carrier trajectory. Therefore, the first lander probe has to be released up to 10 days before entering Mars orbit, meaning that it must survive without power from the carrier until it can deploy its solar panels on the surface of Mars. Thermal control by way of RHUs and the implementation of a low power (<10 mA) timer are enablers for the survival of multiday coasting landers. Additional fuel also has to be accommodated on the carrier in order to enable the trajectory adjustments.

## 6. Conclusions

Overall, the study identified a wide range of potential small Mars mission architectures including orbital and lander missions. From the resulting analysis, it appears that small, low-cost Mars missions are technically feasible for launch within the decade. Three main themes emerged in the conclusions of the study concerning the launch scenario, required technology developments, and the mission operations. In particular, a robust development schedule and selection of high-maturity technologies are critical to meeting the programmatic constraints of the mission.

A commercial rideshare to Earth orbit, while reducing launch costs, significantly drives the spacecraft design and mission operations. Any launch cost savings that are made when using a rideshare opportunity can be easily offset by additional spacecraft development and operations costs. Consequently, there are key technology developments in Europe that would help realise the benefits of commercial rideshare launches and reduce the overall cost of small Mars missions. These include high power, low mass solar arrays and low cost, low power and long lifetime EP thrusters. Until these developments are achieved, dedicated or dual launches will be more likely. Finally, the classical approach to mission operations becomes a substantial cost driver for small missions, especially for long duration transfers and time spent aerobraking. For a cost-driven mission, the approach taken to mission operations becomes a critical mission architecture design driver.

The work completed in this study of small Mars mission architecture will continue in 2021 with further study conducted by European industrial contractors.

*6.1. The Concurrent Design Facility.* The Concurrent Design Facility (CDF) is a state-of-the-art facility equipped with a network of computers, multimedia devices, and software tools, which allows a team of experts from several disciplines to apply the concurrent engineering method to the design of future space missions. It facilitates a fast and effective interaction of all disciplines involved, ensuring consistent and high-quality results in a much shorter time. It is primarily used to assess the technical and financial feasibility of future space missions and new spacecraft concepts (e.g., internal prephase A or Level-0 assessment studies). During this study, the CDF successfully transitioned to distributed remote operation mode, made necessary due to COVID-19 restrictions.

The Concurrent Design Facility was established at ESTEC in November 1998 within the framework of the General Studies Programme.

## Data Availability

The Small Mars Mission Architecture Study (SMARTieS) CDF study report is available upon request from the authors.

## Disclosure

An oral presentation of this manuscript was made at the Europlanet Science Congress (EPSC), 2020 [32].

## Conflicts of Interest

The authors declare that there are no conflicts of interest regarding the publication of this paper.

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