Mars Sample Return with Electric Propulsion

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Abstract

The present paper takes a fresh look at future Mars Sample Return Mission including electric propulsion (EP) for the transfer. The standard mission scenario includes two spacecraft (S/C) launched separately from Earth: an orbiter and a lander. The lander sets down on the red planet together with an ascent vehicle to collect samples. The ascent vehicle would then take off from the Martian surface into Mars orbit with traditional chemical propulsion to transfer the samples to the orbiter waiting there for the return trip to Earth. The results of the system analysis identify EP for the orbiter as most beneficial in terms of launch mass, enabling a launch into Geostationary Transfer Orbit (GTO) by a relatively modest launch vehicle like the Soyuz-Fregat. Concerning the lander, a separate transfer with chemical propulsion appears more advantageous compared to an electrical one. Such a hybrid version of the sample return mission could be conducted within 1150-1300 days. In an advanced scenario, the lander could even ride the electric orbiter, piggy-back style, to the Red Planet.

Keywords

low thrust electric propulsion; mars sample return

1. Introduction

Within ESA's Aurora program the Mars Sample Return (MSR) mission is a flagship mission and envisioned to take place in the timeframe of 2020-2025. Previous studies [1, 2, 3, 4] developed a mission architecture consisting of two elements, an orbiter S/C and a lander S/C, each utilizing chemical propulsion and a heavy launcher like Ariane 5-ECA. The lander transports sampling equipment (e.g. a rover) and a Mars ascent vehicle with a sample container to the Martian surface. After completion of surface operations, the samples are transferred to the sample container, which is launched by the ascent vehicle into Mars orbit. The orbiter performs a separate impulsive transfer to Mars, conducts a rendezvous in Mars orbit with the sample container and returns the samples back to Earth in a small Earth entry capsule. Because the launch of the heavy orbiter by Ariane 5-ECA makes an Earth swing by mandatory for the trans-Mars injection, its total mission time amounts to about 1460 days.

This paper addresses the feasibility of a MSR mission using EP for the transfer. Detailed S/C models for orbiter, lander and ascent vehicles together with general mission and system analyses of the space transportation elements were developed within a diploma thesis at DLR and RWTH Aachen University [5, 6]. There, an extensive parametric study based on trajectory calculations and optimizations of interplanetary transfers, Mars entries, descents and landings as well as Mars ascents investigated the implications from specific impulse, thrust level, power system performance and transfer strategy. Overall goal was to reduce launch mass requirements – thus decreasing launch costs, to help identify areas worth of further investigations and to provide sufficient data for the preparation of a down-selection of different mission architectures as well as S/C design options. It was, however, not intended to define final mission architecture.

Here, we concentrate on the presentation of basic assumptions, discuss selected results and evaluate most interesting options. As reference, the conventional chemical scenario according to [3] is chosen. Major characteristics of this reference scenario are summarized in Figure 1 for a launch around 2020.

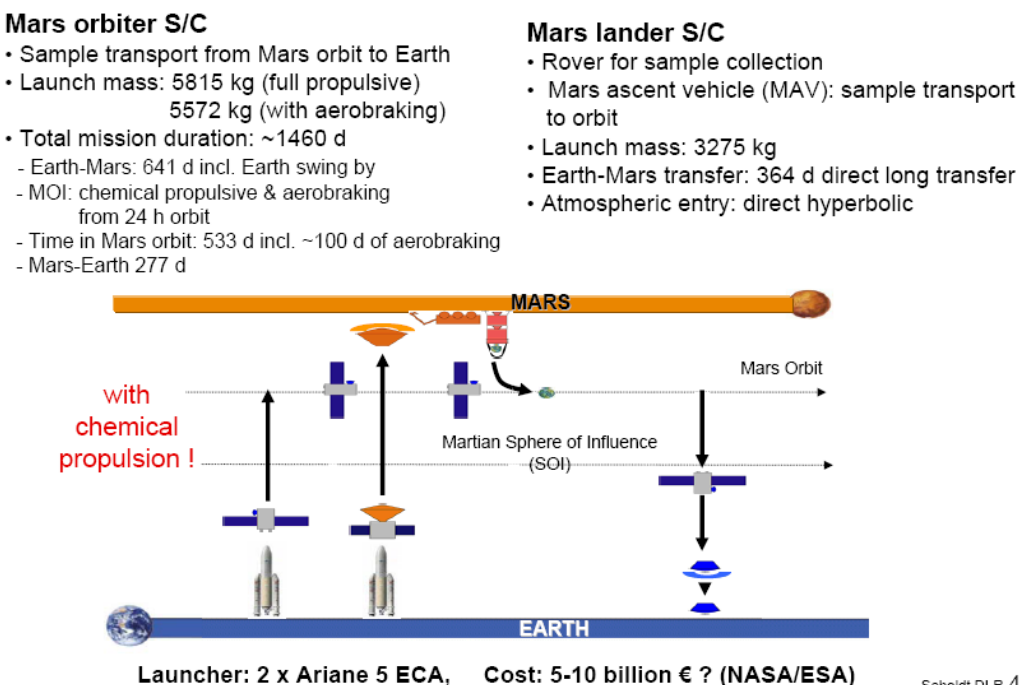

Figure 1 – Mission Characteristics of the Chemical Reference Scenario [3]

Figure 2 shows one of the investigated interesting mission scenarios for an orbiter with EP and a conventional chemical lander, each launched separately from Earth.

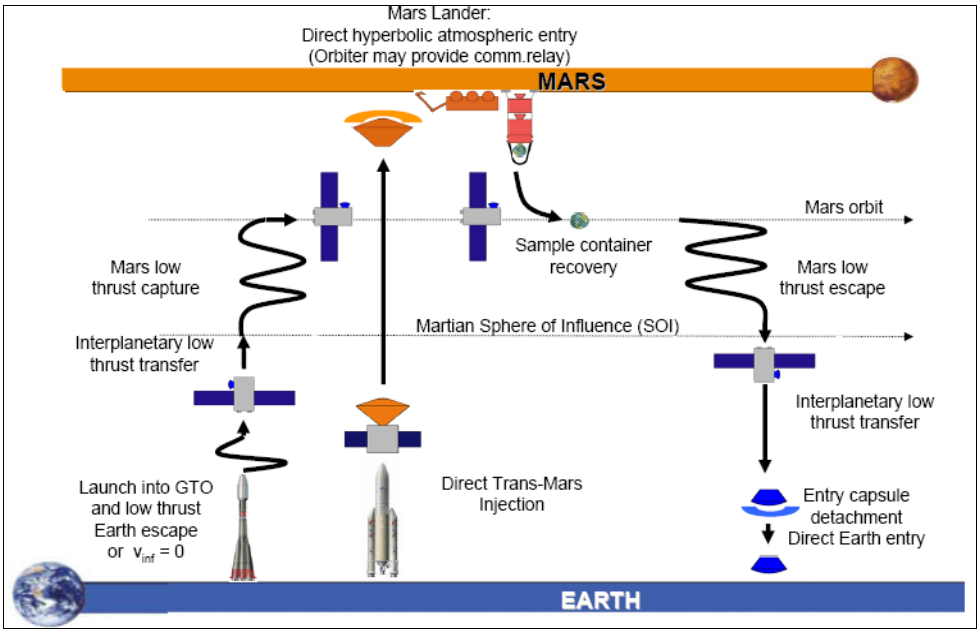

Figure 2 – Hybrid Electric Mission Architecture

In such a hybrid version of the MSR mission, the orbiter would be launched either into GTO, from where it would escape with its own low thrust EP system, or launched directly to Earth escape with velocity v_{inf}=0. The lander would get a direct trans-Mars injection from a separate launcher and perform a direct hyperbolic entry at Mars without using EP. In an advanced scenario, the lander could even ride the electric orbiter, piggy-back style, to the Red Planet (see below). In the following the main characteristics of the mission analysis and S/C models are outlined.

2. Orbiter Mission and Spacecraft Design

The electric orbiter mission has been divided into five segments (in time reversed order):

- Interplanetary Mars → Earth transfer ('inbound')
- Mars Escape
- Mars Capture
- Interplanetary Earth → Mars transfer ('outbound')
- Earth Escape (only considered when GTO is chosen as launch orbit)

Due to the fact, that only the mass of the samples, the sample container, the Earth entry capsule and the orbiter bus is known at the beginning of the mission analysis, the mission phases were examined in reverse order. The mass of the orbiter core bus part (without main propulsion/power/capsule components), containing the avionics, the attitude control system, the thermal control system, the part of the power system supporting avionics and thermal control and the corresponding carrying structure has been estimated based on MARS EXPRESS heritage with 450 kg (for the remaining parts see below and Table 3). To enable the return of the Mars samples, the Earth entry capsule mass - capable of transporting 500 g of samples – has been estimated by [3] to 112 kg. In contrast, we assume an entry capsule mass of 200 kg to double the returned sample mass.

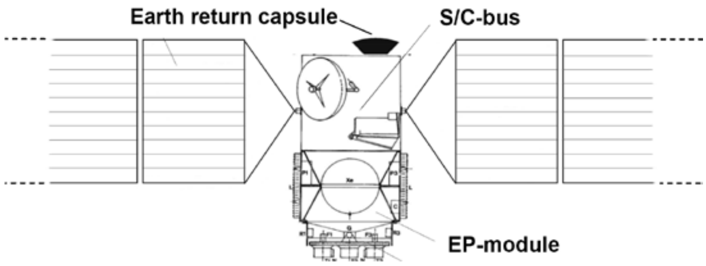

Figure 3 – Sketch of the orbiter with Earth return capsule and EP module

Figure 3 shows a sketch of the electric orbiter. In order to calculate the wet mass of the EP module, low thrust trajectories are optimized using the low thrust optimizer InTrance (Intelligent Trajectory optimization using neuroncontroller evolution) [7, 8]. InTrance combines Artificial Neural Networks and Evolutionary Algorithms. The Neural Networks are used to steer the low thrust S/C. A nearly global optimal solution is found by training the Neural Networks with the Evolutionary Algorithms. Trajectory integration is performed by utilizing the Runge-Kutta-Fehlberg method (RKF54) [9] and using JPL's DE405 ephemerides for Earth and Mars. A separate trajectory optimization is conducted for each segment, considering interface constraints to the neighbor segments, especially with respect to the S/C velocity and distance to the planets. All mission segments, except Mars capture, are optimized with respect to a minimal flight time. In contrast, the trajectory optimization of the Mars capture aims to maximize the allowable relative velocity at the border of the Martian sphere of influence which will be addressed below in more detail. For each mission segment, the required propellant and the corresponding tank/structure masses are determined iteratively and the masses are fitted together w.r.t. the mission phase order. For simplification, it is assumed that the propellant is distributed to several tanks and that after each mission phase an empty tank and its corresponding carrying structure are jettisoned. During the interplanetary outbound low thrust transfer the orbiter has to adjust its velocity in a way that it can be captured at Mars. Because of the low thrust of electric engines, the orbiter must arrive at the border of the Martian Sphere Of Influence (SOI) with a limited relative velocity which should not be exceeded. At the same time, the velocity should be as large as possible to give less restrictive 'rendezvous' conditions, simplifying the interplanetary trajectory optimization and shortening the transfer time. To find out the acceptable rendezvous conditions, corresponding capture trajectory calculations were performed backwards in time, starting from circular low Mars orbit (baseline 1000 km, also considered 250 km and 10000 km) and aborted when the orbiter reaches the border of the SOI. With the resulting velocities - typically around 0.55 km/s - a low thrust capture of the orbiter and a spiraling down to final Mars orbit can be achieved.

The return trip to Earth is similar (low thrust escape of the orbiter from Mars and interplanetary transfer) with an energy optimized launch date at Mars about a year later. For the 'rendezvous' with Earth, conditions concerning distance and relative velocity (w.r.t. Earth's SOI) must be defined accordingly. It is assumed that the orbiter jettisons the Earth return capsule during flight within Earth's SOI, so that the capsule with the samples performs a direct Earth atmospheric entry. Hence, a maximal distance of 10^6 km is assumed. In reality, this distance could be drastically reduced by implementing small correction maneuvers far from Earth. The relative velocity at Earth's SOI determines the atmospheric entry velocity which is limited by the used thermal protection system and the allowable deceleration loads. A relative velocity of 5 km/s (equivalent to an entry velocity of 12.3 km/s) is chosen as upper limit. In order to minimize the transfer time, the optimal trajectories approach the upper limit of the relative velocity. The corresponding mission phases are shown in Figure 4 (left: spiraling out from GTO; middle: interplanetary outbound transfer to Mars; right: spiraling into low Mars orbit) and Figure 5 (inbound transfer).

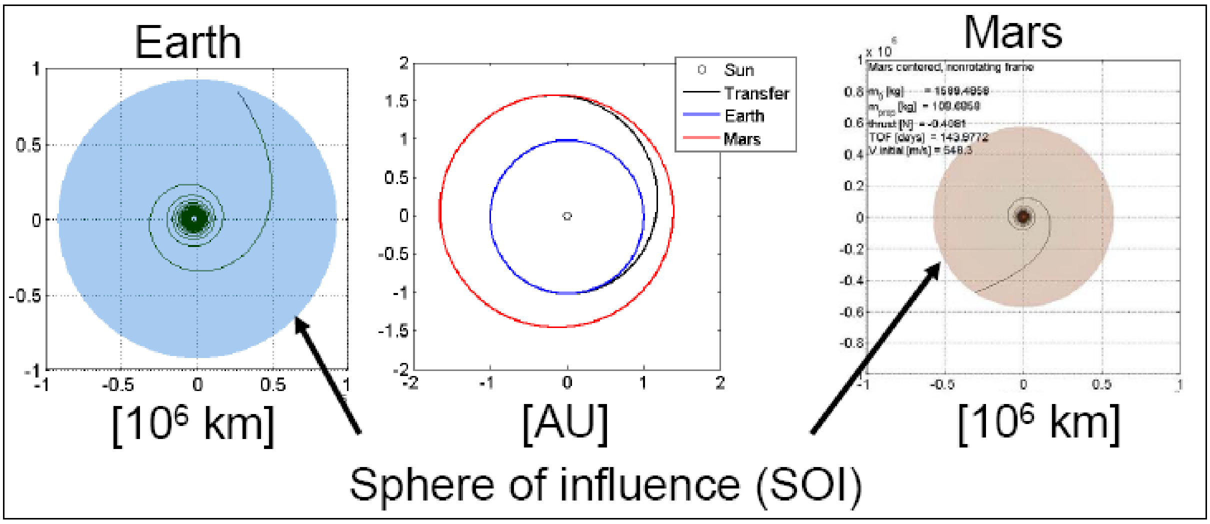

Figure 4 – Spiraling out from GTO, outbound (Earth→Mars) transfer and Mars capture

Clearly, a launch of the orbiter into Low Earth Orbit (LEO) would give the lowest launch vehicle requirements (but would increase the mission time), while a launch to $v_{inf}=0$ would result in a much faster solution. A good compromise, however, seems to be a launch into GTO.

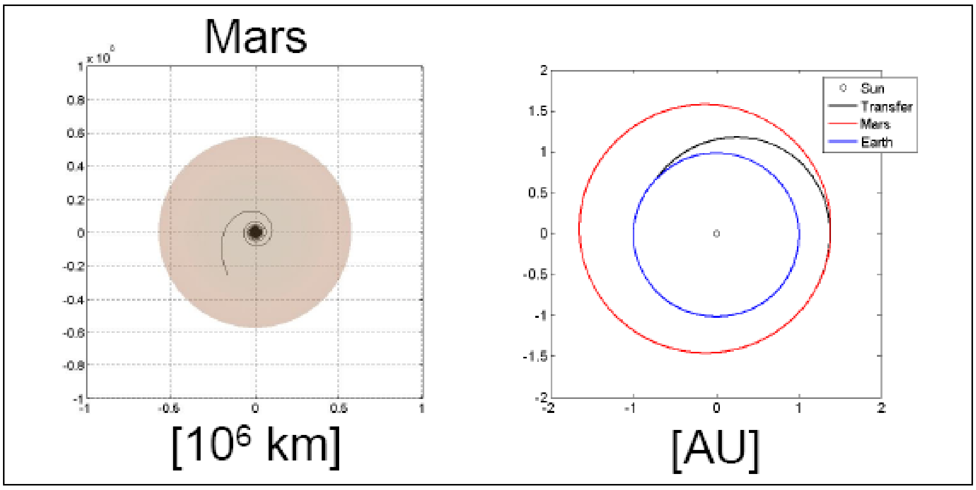

Figure 5 – Spiraling out from low Mars orbit and inbound (Mars→Earth) transfer

For the EP we choose the RIT-22 developed by the UNIVERSITY OF GIESSEN and EADS ASTRIUM. RIT stands for Radio frequency Ion Thruster, using a radio-frequency generator to ionize propellant atoms (Xenon). The charged ions are extracted due to their positive charge and accelerated by the electric field between plasma holder grid and accelerator grid. This has been demonstrated under space conditions by the RIT 22's precursor thruster RIT-10 during the EURECA and ARTEMIS missions [10]. In order to perform parametric investigations, the RIT-22 thruster is modeled for different specific impulses I_{sp} and assumed to be arranged in clusters of 2 to 8 engines. Characteristic data of the different considered engines are given in Table 1.

Table 1

RIT-22 thruster characteristics for two different types of RIT engines [11]

RIT 22	Low I_{sp}	Medium I_{sp}
Beam voltage, kV	1.25	2.1
Specific impulse, s	3704	4763
Power consumption, kW	4.02	6.2
Thrust force, mN	135.5	175
Engine mass, kg	21.3	28.5

Total low thrust velocity increments ΔV for the electrical orbiter mission show up to range between 20 and 30 km/s (direct launch into Earth escape respectively into LEO). Therefore, specific impulses below 3500 s are expected to be suboptimal and not considered. Correspondingly, RIT with high specific impulses beyond 5000 s are expected to be beneficial only for mission ΔV higher than for a MSR mission because they require large power and therefore large solar arrays. Hence, only 'low' and 'medium I_{sp}' engines are considered individually for parametric investigations in the present paper.

The orbiter should be provided with power by a pair of triple-junction solar cell arrays. According to [12], a specific mass of 5 kg/kW may be possible for future solar arrays. Considering further the mass of the high voltage power distribution system, a total specific mass of 10 kg/kW is assumed. This does not include, however, the low voltage power distribution and batteries for the S/C bus which are covered separately in the bus mass of 450 kg. To reduce the total mass and to simplify the orbiter design, it is assumed that the engines are not operating during eclipse phases on spiral orbits around Earth or Mars. Hence, only small batteries are required for the operation of the orbiter bus during the eclipse phases and are contained in its mass. To correct for flight time extensions due to eclipse phases, 30 days are simply added, e.g. to the Earth Escape flight times of S/C launched to GTO.

The electric power provided by the solar arrays is calculated using

$$P(R) = A\ \eta(R)\ W(R) = A\ \eta(R)\ 1.370 \times \left(\frac{1\text{AU}}{R}\right)^2 \approx P_{1\text{AU}} \left(\frac{1\text{AU}}{R}\right)^n \tag{1}$$

with:

P(R) - electric S/C power [in kW], provided by solar arrays at the solar distance R [in AU];

A - area [in m^2] of the solar arrays (assumed to be perpendicular to the Sun);

η(R) - conversion efficiency of the solar arrays, depending on temperature and therefore on distance to the Sun;

W(R) - solar flux at the distance R from the Sun [in kW/m^2];

P_{1AU} - electric S/C power [in kW], provided by solar arrays at the solar distance of 1AU;

N≈ 1.6 to 1.8 (for R ≥ 1AU).

Due to the fact that the conversion efficiency η depends on the solar cell temperature and thereby on the solar flux, the conversion efficiency of relatively cold cells at large distances from the Sun is higher than at Earth distance. To allow the determination of the electric S/C power using a constant efficiency, it is helpful to assume for the S/C power in equation (1) an approximate dependence on distance with an exponent *n* different from 2, e.g. n = 1.6 to 1.8, as correction for the cell temperature. We use the optimistic value of 1.6, and we note that up to Mars distance the difference to the more conservative value of 1.8 is less than 10 %.

Because of the decreasing solar power with distance from the sun, two different power systems are examined:

- standard power (stpo) system, sufficient to operate all electric thrusters with full thrust at Earth's distance (1AU)
- enhanced power (enpo) system, providing 1.7x the standard power and sufficient to operate all electric thrusters with full thrust even at Mars' distance (~ 1.38 AU)

In the case of the stpo-system, some engines must be throttled or shut down during the transfer between Earth and Mars, resp. at Mars, due to decreasing available power. In contrast to that, it can be shown, using equation (1), that the enpo-system is able to operate all engines at full thrust up to a distance of 1.38 AU. This is also the largest distance from the Sun of the transfer trajectory. Hence, the engines can operate at maximal thrust during the complete powered flight phase. If the available power is increased by more than a factor of 1.7 over the stpo-system, a further thrust increase is not possible, while the S/C dry mass increases, resulting in a flight time extension.

In combination with the abbreviations stpo and enpo we also use lo or me to characterize the chosen thruster type with low or medium I_{sp} (e.g. me-stpo). For illustration, Fig. 6 shows for the return transfer from Mars (inbound) the thrust profile of a 3 engine orbiter with a lo-enpo-system (a corresponding trajectory is displayed in Figure 7) compared to that of a lo-stpo-system. It can be seen that all thrusting maneuvers occur in the initial phase of the inbound trajectories. During this maneuver, the electric engines cannot provide their full thrust with stpo-systems (4 kW/engine at 1AU, line labeled with stpo). By increasing the power system output for a constant number of engines, each engine is used more efficiently (e.g. 6.7 kW/engine at 1AU, line labeled with enpo). Orbiters with enpo-systems reach higher accelerations for Mars capture and escape (for details see below and [13]). Therefore, such a power system seems desirable for operations around Mars. Hence, in addition to the I_{sp} the available power per engine at 1AU is also studied as a parameter.

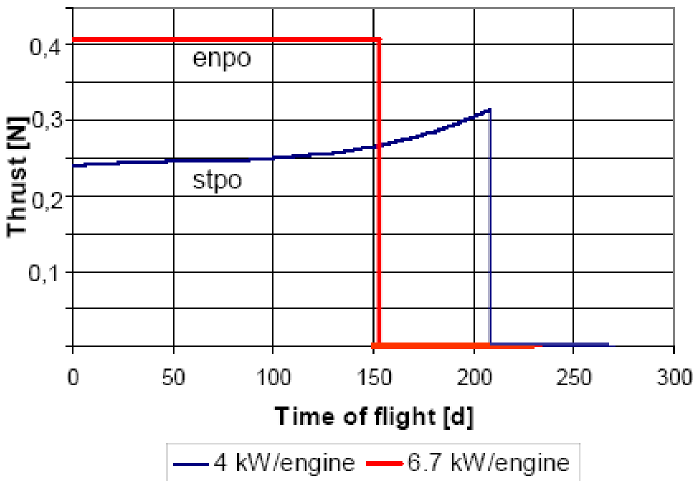

Figure 6 – Inbound trajectory thrust profiles (v_{rel}=5 km/s at Earth's SOI) for an orbiter with 3 engines and lo-stpo-system (4 kW/engine at 1AU), respectively lo-enpo-sytem (6.7 kW/engine at 1AU)

Our system analysis and evaluation of the most promising configurations are based on the following criteria:

- Launch mass / required launch vehicle / S/C complexity;
- Mission time;
- Stay time in Mars orbit (for the orbiter only).

The stay time in Mars orbit should be maximized for two reasons: An early arrival enables the orbiter to provide communication relay and navigation support to the lander which would be helpful especially during the lander's entry/descent/landing (EDL) sequence. Furthermore, a late departure date of the orbiter permits the extension of the surface operations, leaving more time for rendezvous with pre-deployed assets or for sampling.

Figure 7 (left) shows a typical low thrust outbound trajectory (lasting 354 days) for the lo-enpo orbiter with 3 engines which – as discussed below – appears very interesting. The corresponding inbound transfer starting about 14 months later and lasting 232 days is displayed in Figure 7 (right). The arrows indicate the direction and magnitude of the low thrust.

3. Lander Mission and Spacecraft Design

In principle, a similar approach can be used for the investigation of the lander S/C with an EP stage. But instead of the relatively small Earth entry capsule, a Mars lander has a significant higher entry mass. Therefore, two landers with different payload capabilities have been considered (for details see [5, 6]):

- a large lander, able to deliver a two stage ascent vehicle and a rover to the Martian surface;
- a small lander, able to transport only an ascent vehicle or a rover.

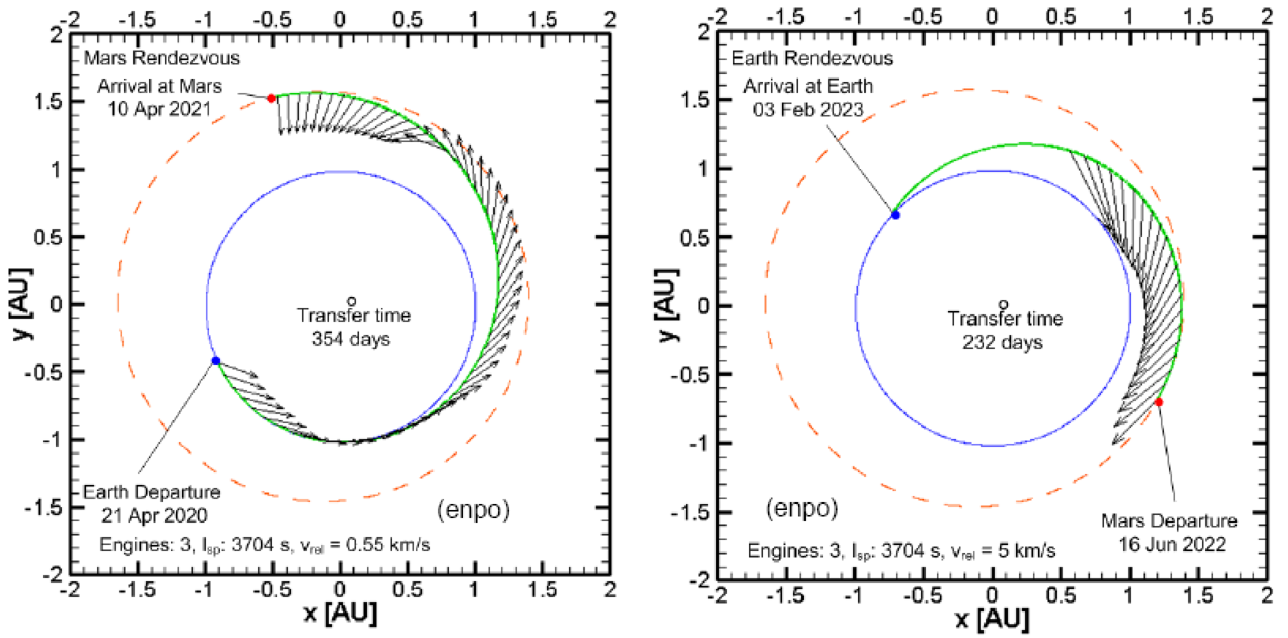

Figure 7 – Outbound (left) and inbound (right) low thrust transfer trajectories of a 3 thruster RIT-22 lo-enpo orbiter with a relative velocity of 0.55 km/s at Mars' SOI and 5 km/s at Earth's SOI

In case of the small lander two separate missions are assumed to deliver the sampling rover and the Mars Ascent Vehicle (MAV) to the surface of Mars (see also Figure 8).

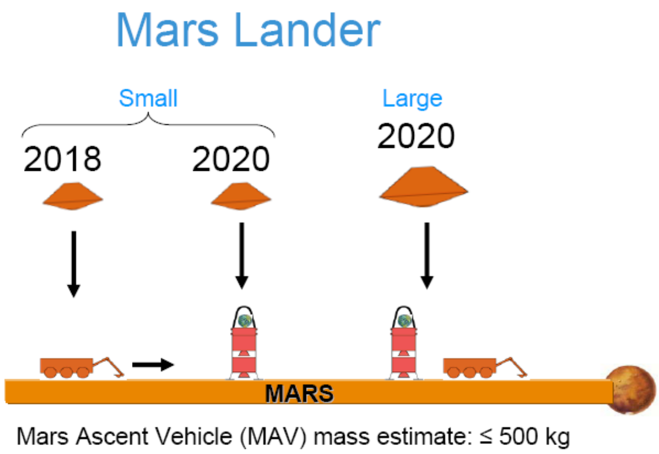

Figure 8 – Mars Lander Scenario

For the MAV an ascent trajectory optimization has been carried out and a brief mass estimation led to a launch mass of ~ 465 kg. Due to the high ascent ΔV between 4 and 5.5 km/s to low Mars orbit (baseline: 1000 km altitude), the wet ascent vehicle mass highly depends on the dry mass which is quite unsure at the actual project stage. Hence, the assumed mass has to be seen as an initial guess, enabling the comparison of different transportation concepts to the Martian surface which is the main focus of this study. Furthermore, it has to be pointed out that all landers include a S/C bus (370 kg), enabling an autonomous landing and surface operations. This bus can also provide some support during Earth-Mars transfer. Hence, the lander S/C propulsion stage requires only a simplified additional S/C bus, mainly consisting of a solar array, attitude control engines and navigation equipment. Its mass has been estimated based on data from [14] to 180 kg (without the separately included low thrust propulsion and power supply). For the determination of complete lander masses, a detailed entry, descent and landing system optimization has been conducted under following assumptions (for details see [5, 6]):

- an entry velocity of 5.6 km/s at an altitude of 135 km (derived as upper limit from a direct Hohmann like transfer with a hyperbolic excess velocity of 2.66 km/s at Mars SOI, see also [15]); in principle, for low thrust propulsion a reduction of the hyperbolic excess/entry velocity is possible at the expense of additional flight time and ΔV, but 5.6 km/s was considered also for the low thrust transfer as the upper limit;
- aerodynamic braking, using heat shield and parachutes; powered descent, using liquid rocket engines N_2O_4/MMH, I_{sp}= 318 and landing on legs with a velocity $\leq$ 2.5 m/s.

The main characteristics of the two lander designs are outlined in Table 2. Because a direct hyperbolic entry for the Mars lander is intended, the lander can approach the Martian SOI much faster than the orbiter. We assume a value of 2.66 km/s for the relative velocity, corresponding to a conventional chemical transfer. Furthermore, attention has to be paid to the specific impulse of the lander engines. Considering the relatively low mission ΔV as well as the effort for a higher specific impulse in terms of mass and cost, a low specific impulse of 3704 s and standard power system appear to be more suitable.

Table 2

Mars Entry Capsule Characteristics

Lander type	Small	Large
Heatshield ∅ VIKING shape [m] Entry flight path angle [°]	3.06 -15.6	4.5 -12
Supersonic parachute		
Diameter [m] Deployment dyn. pressure [Pa]	16.8 750	6 900
Subsonic parachute		
Diameter [m] Deployment dyn. pressure [Pa]	N/A N/A	23 270
Rocket engine thrust [kN]	18.7	28
Flight time [s]	353.6	775.9
Mass break down [kg]		
Heat shield Supersonic parachute Subsonic parachute Rocket engine Propellant Tank Landing gear EDL system subtotal	109 38 77 84 12 42 362	286 5 59 94 121 18 70 653
Structure	382	636
Spacecraft bus	370	370
Payload mass	≤ 500 (MAV or rover)	≤ 980 (MAV + rover)
Entry mass	1600	2640

As an example, such a transfer of the small lander with 4 engines and 16.1 kW total power at 1AU is displayed in Figure 9 with a transfer time of 339 days. Due to the relatively small required ΔV for the lander, however, a conventional chemical transfer as discussed, e.g., in [16] may also be a reasonable or even more promising alternative. Figure 10 displays such a transfer with an Ariane 5-ECA launch and a mass of up to 4000 kg at Mars arrival, covering even a large lander.

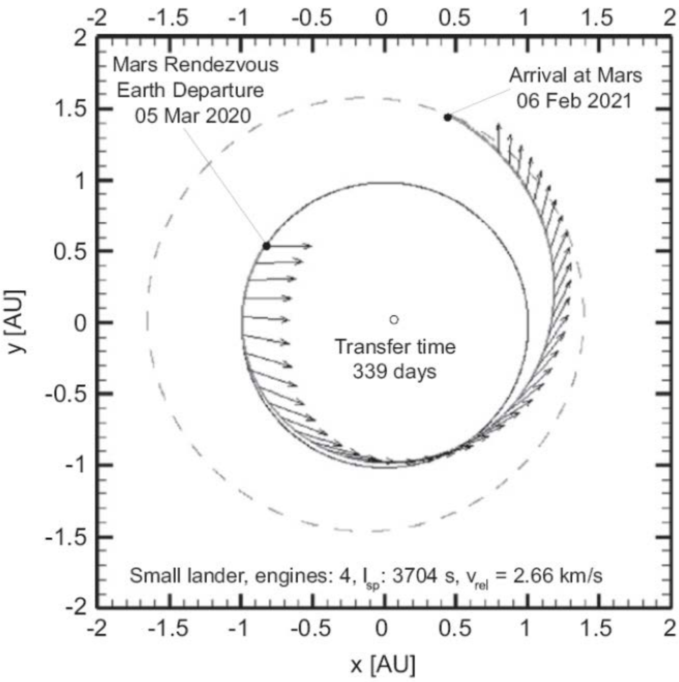

Figure 9 - Small electric Mars lander with 4 RIT 22 lo-stpo engines (16.1 kW total at 1AU)

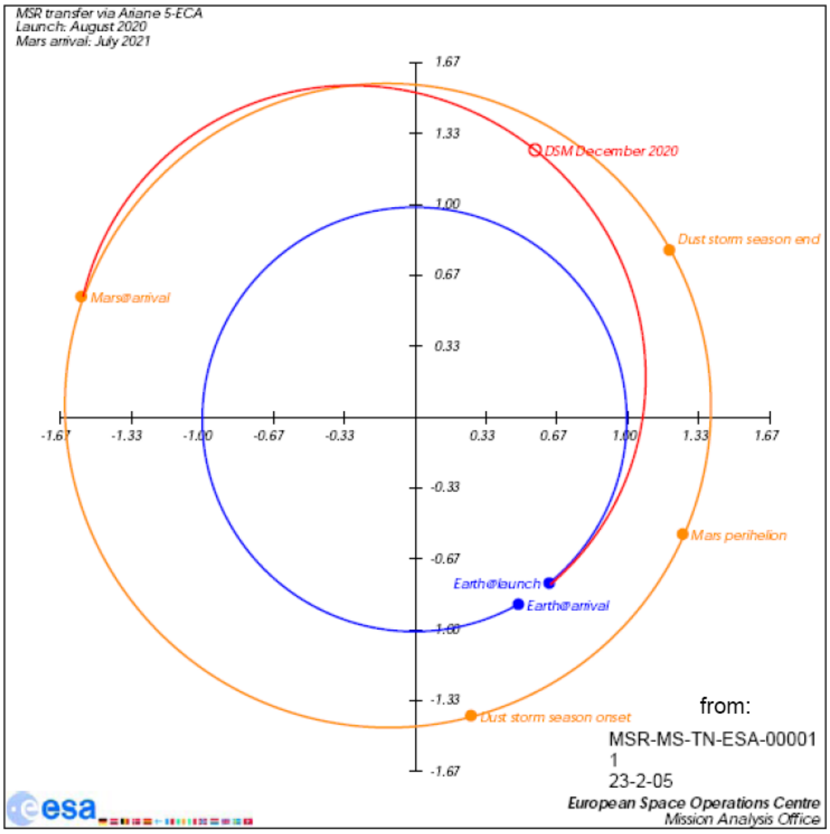

Figure 10 - Conventional impulsive long direct transfer of a Mars lander (launch Aug. 2020 with Ariane 5-ECA, mass at arrival in July 2021 ≤ 4000 kg) [16]

4. Results

Figure 11 displays the orbiter total mission time in case of a launch into GTO in dependence of the number and type of selected electric engines as well as the chosen power supply option.

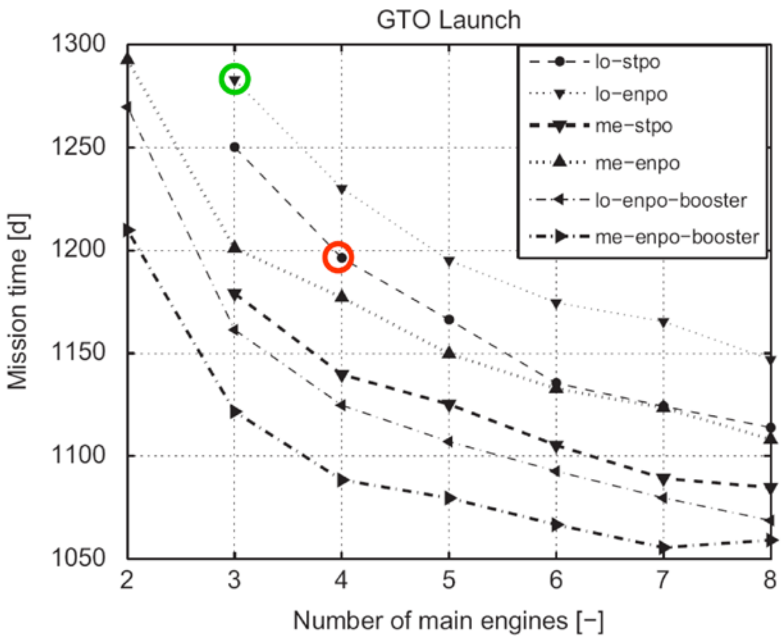

Figure 11 – Total mission time for various electric orbiter configurations with a launch into GTO and a Mars target orbit of 1000 km

The orbiter configuration has a significant impact on the total mission time. Configurations with stpo-systems have shorter total mission times compared to enpo-systems. This can be explained by the fact that the latter provide only lower accelerations during Earth escape and interplanetary transfer: they are oversized for operations at Earth distance and lead to disadvantages in mass to thrust ratios and time for Earth escape. On the other hand, they can allow faster Mars capture and escape due to their higher power and thrust performance at Mars distance, so that the stay time in Mars orbit is significantly increased, which is an important selection criterion. For compensation of the disadvantages at Earth distance, additional engines could be considered. These engines would be used as long as the power system can support them. After that, the engines would be detached at the border of the Martian SOI – equivalent to a staged propulsion system. According to the naming of launcher strap-on propulsion systems, the additional electric engines are called (electric) "booster" (lo-enpo-booster, me-enpo-booster; compare corresponding curves in Figure 11). More details for this option were investigated in [5, 6]. It can be concluded that the implementation of a booster stage reduces the total mission time further by 50 to 100 days, but the higher complexity has to be considered as an adverse factor. Therefore they are not recommended here. We keep in mind that stpo configurations require mission times between both extremes of enpo and booster configurations, but due to their shorter stay times in Mars orbit they seem less desirable (for details see [5, 6]).

The initial masses for a GTO launch range between 2000 and 4000 kg (see Figure 12), and most of the configurations could be launched by a Soyuz Fregat vehicle. The most interesting options as displayed in Table 3 below are emphasized by circles.

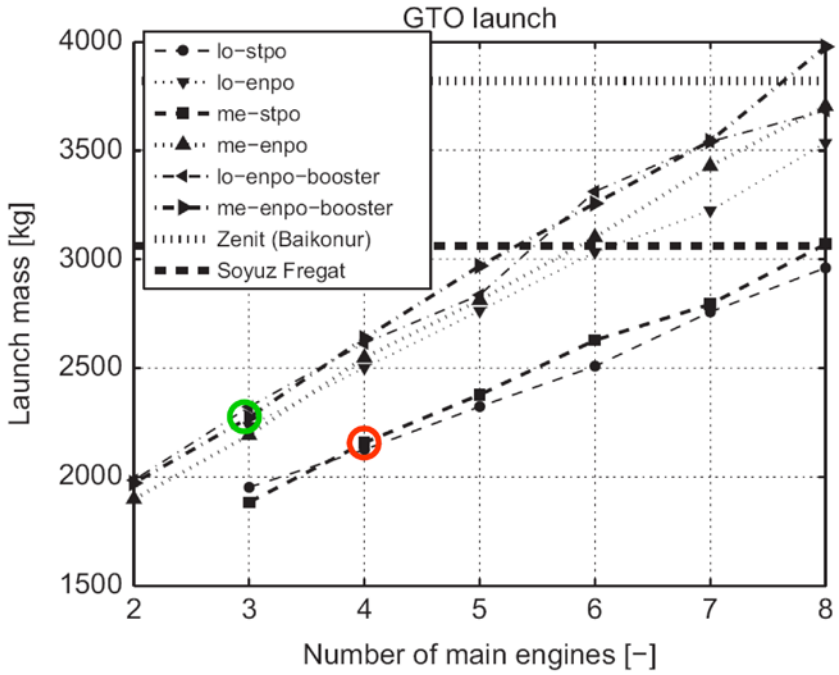

Figure 12 – Initial masses for various electric orbiter configurations with launch into GTO and a Mars target orbit of 1000 km

In case of a launch to v_{inf}=0 km/s, the corresponding analysis in [6] shows that the mission times with about 1,000 days are a bit shorter. However, the evaluation of the most attractive S/C configuration will be mainly based on the launch mass and in particular on the choice of the launch vehicle. The resulting launch masses in [6] of 1.7 to 4 metric tons into an Earth escape trajectory with v_{inf}=0 km/s range between the capabilities of Ariane 5 (7 tons) and Soyuz Fregat (1.6 tons). Because a dedicated Ariane 5 launch is oversized and a shared one seems not feasible for direct injection into an escape trajectory, a launch into GTO is preferred.

The major characteristics of two possible and interesting sample return orbiters (with GTO launch and 1000 km Mars orbit altitude) are displayed in Table 3. For comparison, it presents the configuration with the lowest power requirements of 16.1 KW at 1AU but still providing sufficient stay time in Mars orbit (137 days) for a rendezvous with a sample container. On the right hand side, the characteristic data of the smallest feasible enpo configuration are outlined. The 3 engine lo-enpo orbiter offers a significantly increased stay time in Mars orbit (192 days) with only slightly raised launch mass (2240 kg) and total mission time (1283 days). But it requires 20.25 kW at 1AU. Such a solar array would have similar power requirements as today's GEO communication satellites like the ALPHA-bus.

A further important aspect of low thrust missions is the life time of the engines. Based on the flight experience of the RIT 10 engine on the Artemis mission and 5000 hr ground testing of the RIT-22, the minimum expected life time has been estimated to 23,000 hrs [17]. However, real life times are expected to be even longer (especially if the thrusters are throttled). Such a throttling has to be implemented for the standard power configuration in any case. Alternatively, engines could be switched off. Because no back up thruster is included in the actual design and the thruster operation time may approach 24,500 hrs, this question remains to be examined further.

Table 3

MSR orbiter characteristic data with GTO-launch and 1000 km Mars orbit altitude, (total mission time includes 30 d margin, e.g. for power shortage during eclipse phases)

	4 lo-stpo	3 lo-enpo
No. of engines [–]	4 RIT 22 lo	3 RIT 22 lo
Power at Earth (kW)	16.1	20.25
Total mission time (d)	1196	1283
Time in Mars orbit (d)	137	192 !
Thruster operation time (h)	<24,700	24,500
Mass budget (kg)		
Earth return capsule	200	200
Spacecraft bus	450	450
Power	160	205
Propulsion	85	65
Sample container capture	105	105
Tanks, feeding system	85	95
Structure	180	195
Propellant	855	925
Dry mass	1265	1315
Wet mass	2120	2240

For the small lander with 4 engines and 16.1 kW total power at 1AU, a detailed analysis (see [6]) resulted in a total launch mass of 2500 kg including low thrust and power system for a launch to v_{inf}=0. The latter could be achieved, e.g., with a Zenit sea launch. This would not be possible, however, for a large lo-stpo lander, requiring at least 6 engines to keep transfer times below one year. Its launch mass would be around 4000 kg. Due to the relatively small required ΔV for the lander and its considerable complexity concerning the propulsion system, it remains doubtful whether a solar EP scenario presents advantages for the lander over a conventional chemical transfer as discussed, e.g., in [16]. In Figure 13 the mission duration data for the different phases of the investigated most interesting hybrid electric scenario are added for illustration.

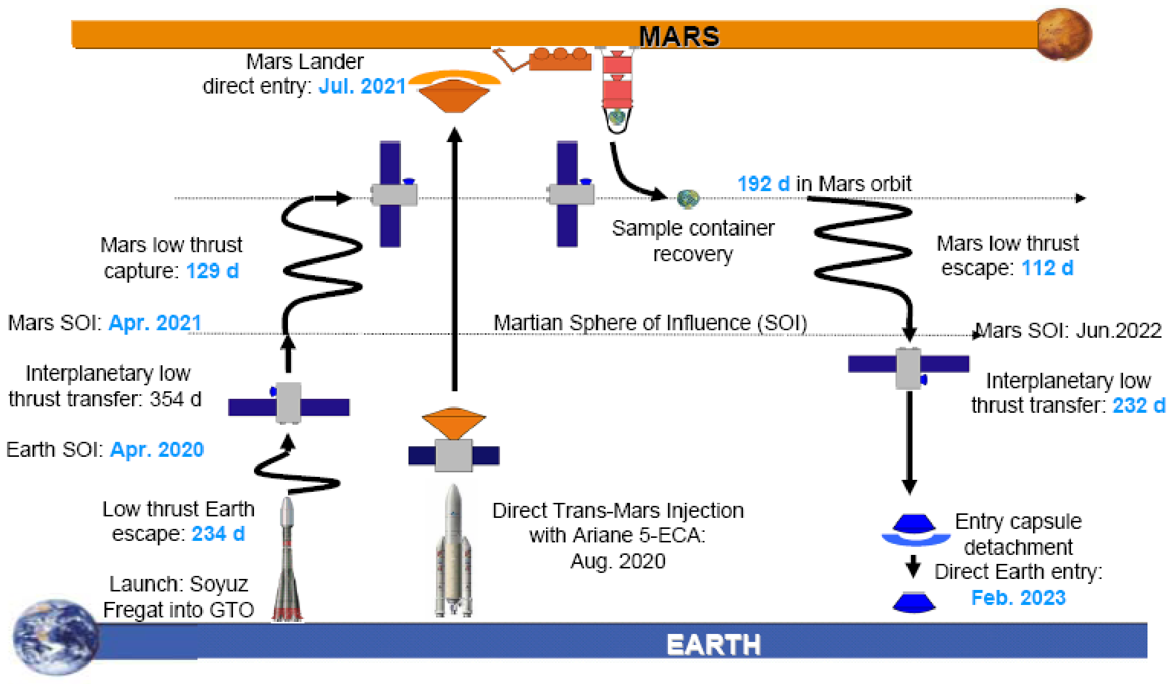

Figure 13 – Mission duration data for the hybrid electric case with a 3 engine lo-enpo orbiter launched by Soyuz Fregat into GTO and a separate conventional impulsive direct transfer for the lander

Finally, we mention shortly a very advanced scenario for an orbiter launched together with an attached lander, riding piggy-back style to Mars, which was investigated in [5, 6]. To keep engine numbers as low as possible, only the more powerful RIT-22 me is considered as main engine. The outbound trajectory design is based on the 4 and 5 RIT 22 me engine orbiters with enpo (plus additional booster engines). Those orbiters transport the attached small or large landers to the border of the Martian SOI, where the lander gets detached and injected by a cruise stage into a descent trajectory. The corresponding mission analysis varied the number of additional orbiter (booster) engines. Table 4 presents feasible configurations for a combined S/C with a small or large piggy-back lander. The launch masses to v_{inf} =0 km/s vary between 4330 and 5960 kg. The launch including orbiter plus piggy-back small lander can be accomplished with an Atlas 5 431 or even a Proton M Breeze. The more expensive Ariane 5 launcher enables the transport with the large lander. Hence, this mission could perform sampling and sample return on its own and would only require communication support for the lander EDL phase. Unfortunately, this configuration requires 8 electric engines and 52 kW at Earth. Anyway, both combined configurations with small and large landers offer sufficient surface and orbit stay times in combination with acceptable total mission times (see Table 4).

Table 4

Combined MSR orbiter characteristic data (including piggy-back small and large lander)

No. of RIT 22 me [–]	4+2 booster	5+3 booster
Power at Earth (kW)	41.6	52
Total mission time (days)	1010	1018
Outbound transfer time (days)	341	357
Time in Mars orbit (days)	max. 283	301
Surface stay time (days)	max. 335	345
Mass budget (kg)		
Mars lander	1600	2640
Cruise stage+Adapter	160	260
Earth return capsule	200	200
Spacecraft bus	450	450
Power	415	520
Propulsion	170	230
Sample container capture	105	105
Tanks	95	115
Structure	230	290
Propellant	905	1150
Dry mass	3425	4810
Wet mass	4330	5960
Launcher	Proton M	Ariane 5

5. Summary

The results of the system analysis identified EP for the orbiter as most beneficial in terms of launch mass, leading to a reduction of launch vehicle requirements and enabling a launch by a Soyuz-Fregat into GTO. As launch orbit the GTO seems an appropriate compromise between a launch to v_{inf}=0 km/s (most time saving option) and a low thrust escape from a LEO (option with least requirements on launcher). To maximize the stay time in Mars orbit, the power system should provide sufficient power to operate all electric engines with full thrust at Mars. Such a sample return mission could be conducted within 1150-1300 days. Concerning the small lander, a separate launch in combination with electric propulsion leads to a significant reduction of launch vehicle requirements, but would involve probably two landing missions. On the other hand, a large single lander would require a large number of engines and correspondingly a big power system. Therefore, a large lander launched by an Ariane 5-ECA and performing a separate chemical transfer could possibly be more advantageous. Alternatively, a second mission architecture has been developed, requiring only one heavy launch vehicle (e.g. Proton). In that case the lander is transported piggyback by the electrically propelled orbiter.

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