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Human mission to Mars: The 2-4-2 concept

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Basic idea:
We propose a simplified but efficient scenario for a human mission to Mars. The idea is to select a crew of only 2 astronauts, to bring in situ resource utilization systems on the surface of Mars and to duplicate the mission as it was proposed by Von Braun.

Preliminary introduction:

This report presents the details of the 2-4-2 scenario for a human mission to Mars. There are 2 parts:

- The first part is the draft version of the paper that was submitted to Acta Astronautica and finally published in 2011:
  J.M. Salotti, "Simplified scenario for manned Mars missions", Acta Astronautica, vol. 69, p. 266–279, 2011.
- The second part is the draft version of a new submission, which includes all the details of a revised version of the scenario.

The scenario is called two for two, or two four two because there are always 2 astronauts ready to help the 2 others and there are 2 astronauts in each vehicle, then 4 on Mars and 2 in each vehicle once again for the return.

Acknowledgements:
The author would like to thank Richard Heidmann and Alain Souchier for their valuable comments and suggested corrections of the paper. Special thanks also to the students of the Ecole Nationale Supérieure de Cognitique who worked on some aspects of the scenario.

Biography

Jean Marc Salotti is Professor in computer science at Ecole Nationale Supérieure de Cognitique, Institut Polytechnique de Bordeaux, France. He is a member of the Intégration du Matériau au Système CNRS laboratory. He has been involved in space related projects for ten years. He graduated from the ISU Master of Space Studies in 2002 and developed Marsbase, a multi-agents platform that helps understanding organizational and technological issues for human activities in extreme environments. He is also a member of Association Planète Mars, the French chapter of the Mars Society.

PART I

Simplified Scenario for Manned Mars Missions

List of acronyms:
DRM: Design Reference Mission, 1.0 or 3.0.
DRA: Design Reference Architecture 5.0
EDL: Entry Descent and Landing
ERV: Earth Return Vehicle
EVA: ExtraVehicular Activity
IMLEO: Initial Mass in Low Earth Orbit
ISRU: In Situ Resource Utilization unit
ISP: Specific Impulse
LEO: Low Earth Orbit
MAV: Mars Ascent Vehicle
MO: Mars Orbit
NTR: Nuclear Thermal Rocket
TEI: Trans-Earth Injection
TMI: Trans-Mars Injection
TPS: Thermal Protection System

Abstract
We propose a simplified but efficient scenario for a manned Mars mission. The idea is to select a crew of only 2 astronauts and to bring in situ resource utilization systems in the same vehicle. For security reasons, we suggest duplicating the mission as it was proposed by Von Braun. At every moment of the journey, the two vehicles would stay close so that each crew could provide help to the other. We show that this scenario is much simpler than the last design reference architecture proposed by NASA. The initial mass in low Earth orbit is minimized and the risks are also reduced. The total cost could be in the order of 40 billion dollars.

Keywords: Manned missions, Mars Direct, Mars semi-direct

1. Introduction
A critical point of manned missions to Mars is the total payload that has to be sent to the planet [16]. Its impact on complexity, risks and cost is very high. In the last Design Reference Architecture (DRA) for manned missions to Mars from NASA, it is suggested that at least seven launches of an Ares V class launcher are required to assemble different rockets in LEO (Low Earth Orbit) and send them to Mars [9]. While many aspects of the architecture have been clearly justified, the organization as a whole seems very complex. It probably explains why no manned mission is currently planned by NASA. Our aim is to propose a simpler mission that could be accomplished very soon at lower risks and costs. An important issue is to determine the parameters that help minimizing the payload mass. Many variables have an impact on the total mass. A large part of the payload is the propellant needed for the return. In all scenarios, a key idea is to use Martian local resources to produce all or part of that

propellant. For instance, it is possible to send liquid hydrogen on Mars and to use carbon dioxide extracted from the atmosphere to produce methane and oxygen, which is a good propellant for rockets [4]. Other variables play an important role, especially the size and mass of the payload that has to be sent to the surface of Mars and the size and mass of the payload that has to be sent back to Mars orbit. Both of them are correlated to the number of propulsion systems, the number of habitable modules, the number of astronauts and the amount of consumables. In Mars Direct, the payload sent to the surface of Mars seems optimized [19]. However, when Zubrin exposed his concept at NASA Johnson Space Center in 1992, some engineers claimed that the payload and volume of the habitats were underestimated [21]. They also suggested that 6 astronauts would be preferable instead of 4. Despite his efforts to maintain his original concept, Zubrin and Weaver finally proposed Mars semi-direct, which has been the driving concept of all successive NASA design reference missions for the exploration of Mars [20], [12], [8], [9]. In this category of scenario, the direct return is abandoned and a MAV (Mars Ascent Vehicle) is added to join an ERV (Earth Return Vehicle) waiting in MO (Mars Orbit). In the DRA scenario, which is the last proposal from NASA, it is thus proposed to send 4 habitable modules to Mars [9]. The first one is the surface habitat (SHAB) sent in a cargo two years prior to the launch of the crew. The second is the Orion capsule used to join the Mars transfer vehicle. The third is smaller. It is integrated in the ascent vehicle (MAV) for the return to MO. The last one is the module used for the transit between the Earth and Mars. Four habitable modules with their life support system, their consumables and an attached propulsion stage are therefore sent to Mars. In terms of number of habitable modules and propulsion stages, Mars semi-direct is less optimized. Obviously, another habitable module might be required for security reasons. In the DRA, the redundancy of habitable modules enables the Abort to Orbit (ATO) option if there is a problem on the surface. However, during transit, only one habitable module is available and there are long periods of time during which the habitable modules are left unoccupied. Moreover, compared to Mars Direct, the production of propellant using in situ resources has been highly reduced. All in all, if we compare the DRA scenario to the original Mars Direct concept, the payload on the surface of Mars has been multiplied by a factor of 2 (214 tons vs 98 tons) and an ERV waiting in MO has also been added. Is Mars semi-direct the best option for the first manned missions to Mars? Let us investigate another concept. In 1969, Von Braun proposed a different concept [16]. Two ships would fly in convoy from Earth orbit to Mars and back. In this scenario, two habitable modules for 6 astronauts each are used during the transit phases and two others on the surface of Mars, each one being a possible backup solution for the crew of the other. However, Von Braun’s proposal is not practical because it requires a huge IMLEO (Initial Mass in LEO) (1450 tons). The number and mass of habitable modules and propulsion systems are once again very high.

An efficient tradeoff between Mars Direct and Von Braun scenarios can nevertheless be made. In our scenario, we propose a single habitable module for the entire mission, the use of the same propulsion system for the landing and the return to Mars orbit and a duplication of the mission as it was suggested by Von Braun. In order to make that scenario simple and feasible, we propose a crew of 2 per vehicle. That scenario is very interesting for several reasons:

- A crew of 2 astronauts minimizes the needs in terms of mass and size of the spacecraft, which have a great impact on the mass of heat shield and propellant for landing on the surface of Mars. The entry, descent and landing stage is thus simpler and probably safer.
- If we choose a short number of astronauts per vehicle, the mass of accommodations and consumables is reduced. For the outbound stage, it might be possible to take in each vehicle a chemical unit and associated power systems for in situ propellant

production. There is no need to send other spacecrafts to Mars. In addition, compared to the DRA scenario, the risk of landing too far from the ERV is eliminated.
- The preliminary automatic mission that is required in Mars Direct or in the DRA to produce propellant before the arrival of the crew can be avoided. Each vehicle would indeed carry its own system to produce propellant, which would serve as a possible backup for the second.
- The deployment and maintenance of ISRU (In Situ Resource Utilization unit) and power systems are facilitated. The use of large and flexible solar arrays might even be enabled only if astronauts are present on the surface.
- Finally, if a major incident occurs during the trip, for instance an explosion of an oxygen tank like in the Apollo 13 mission, since two vehicles are sent to Mars and return to the Earth at the same time, it is possible to undertake a rescue mission and proceed to a transshipment of the crew onto the second vehicle.

The idea is further developed in the paper, which is organized as follows. In Section 2, the main stages of the scenario are given and the impact of the crew size is discussed. In Section 3, each stage of the mission is detailed. In order to determine IMLEO, it is proposed to estimate first the mass of the Mars ascent vehicle, then the mass of the vehicle before landing on Mars and finally the total mass of the vehicle before TEI (trans-Earth injection). In order to assess our proposal, a comparison with the DRA is proposed and discussed in Section 4. In conclusion, we make some recommendations to check the feasibility and relevance of our scenario.

2. Main idea

2.1 Synthetic description of the scenario

Our objective was to define a manned mission to Mars as simple, cheap and secure as possible. Several principles have been followed:

- A conjunction-class mission is more appropriate.
- IMLEO should be minimized to reduce the number of launches, which has a major impact on mission risks. In order to minimize IMLEO, it is important to minimize the mass of the payload on Mars and to maximize the production of propellant on the surface of Mars.
- The robotic mission proposed in DRA and Mars Direct to produce propellant for the return and consumables before the arrival of the crew is very complex and does not enable all technological options. An all-up scenario is preferable.
- Small crews might be a good solution to minimize the payload on Mars and reduce the risks during EDL (Entry, Descent and Landing) on Mars.
- The risk of loosing the crew during the transit between the Earth and Mars (or Mars and the Earth) or on the surface of Mars can be mitigated by sending another vehicle on the same trajectory at the same time.

From these principles, we came to a different concept for manned missions to Mars, which is a tradeoff between Mars Direct and Von Braun's scenario. The idea is to send everything in a single spacecraft, to produce on Mars most of the propellant for the return and to duplicate the mission. Zubrin and other designers probably had a similar idea before. However, as it will be soon explained, the complexity, the risk and the cost of the scenario are tightly linked to the crew number and neither Von Braun nor Zubrin suggested a scenario with only 2 astronauts per spacecraft. In this new scenario, the spacecraft landed on Mars is a set of four modules:

- The first is the main propulsion stage. In order to optimize the payload, the same stage is used for the landing on Mars and for the takeoff and return from the surface to MO. Since the requirements are of the same order, the use of the same engine for landing and takeoff is possible (the idea had already been proposed in the DRM scenario [8]). The propellant is LCH4 and LO2. H2, CH4 and O2 are produced on Mars.

- The second is a habitable module for a crew of two astronauts with consumables for four. Accommodations are taken for a period of approximately three years. A crew of two astronauts is preferred to minimize the payload and reduce EDL risks. It also enables taking the chemical unit in the same vehicle (see next sections). On top of the habitable module, some containers are indeed needed to store the components of a small chemical unit and its associated power systems for the production of methane and oxygen as well as land vehicles and other tools. ISRU (In Situ Resource Utilization unit) is deployed on the surface at the beginning of the stay. In order to minimize the payload for the return, the containers are left on Mars.
- The third is the propulsion stage for TEI (trans-Earth injection). It is already filled with CH4/O2 brought from Earth. After aerocapture, it is left in Mars orbit. After surface operations, the main propulsion stage is used to perform a rendezvous. The habitable module is then connected to the TEI propulsion stage and the ascent one is jettisoned. This return scenario is in fact similar to the return of the Apollo program with the rendezvous in lunar orbit between the lunar module and the command and service module.
- The last module is the Earth reentry capsule that is used at the end of the journey. Since there is no need to bring it to the surface, it is left in Mars orbit. Most consumables for the return are also left in MO. They are stored in containers (see Figure 1). After surface operations and launch from the surface, there is a rendezvous in MO. The capsule is then attached to the spacecraft and the content of the containers are integrated into the habitable module.

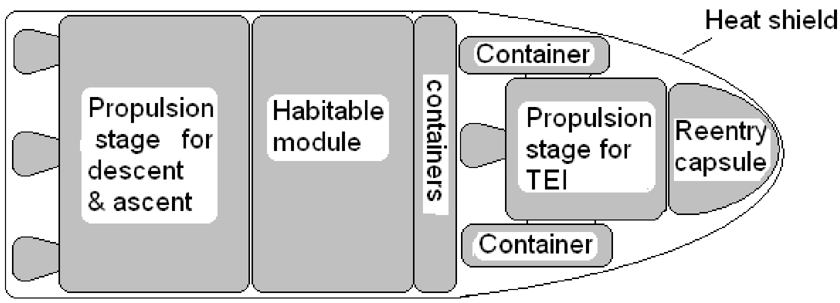

Figure 1: Drawing of the spacecraft during outbound. There are two propulsion stages but only one habitable module. The propulsion stage for trans-Earth injection, some containers (consumables for the return) and the Earth reentry capsule are left in Mars orbit after aerocapture maneuvers. At the end of surface operations, there is a rendezvous in Mars orbit to permute propulsion stages. NB: T he drawing does not necessarily respect the size of the modules and packaging constraints. It is given to illustrate the concept.

The minimization of the payload landed on the surface of Mars allows a reduction of the ballistic coefficient, which increases the braking in the early stages of the descent and improves parachutes efficiency. Less propellant is needed for landing, less propellant has to be produced for the ascent, which in turn has an impact on the mass of the tanks, on the mass of the ISRU and associated power plants and finally once again on the propulsion system for landing. More details are given in the next sections. The number of habitable modules is minimized and rather small propulsion stages are used. However, since there is no backup a single vehicle would be risky. In order to minimize the risks, as it is proposed in Von Braun's scenario, we suggest sending two vehicles at the same time [16]. During transit, the two vehicles should stay very close the one from the other so that in case of an emergency, it is possible to tranship the endangered crew in the safe vehicle. See Section 4 for a discussion on that topic. The same principle applies in MO, on the surface of Mars and during the inbound. An overview of the scenario is presented in Table 1 and Figure 2.

Table 1: Summary of our scenario.

Action	Start	End	Number of days	ΔV (km/s)
Launch with heavy launcher, crew 1, spacecraft 1	Day 1	Day 1	0	9.3[1]
Spacecraft 1 waiting in Earth orbit	Day 1	Day 14	13	0
Launch with heavy launcher, crew 2, spacecraft 2	Day 14	Day 14	0	9.3[1]
Check up of systems	Day 14	Day 14	0	0
TMI for both spacecrafts	Day 14	Day 14	0	3.6 x2
Both spacecrafts on the way to Mars	Day 14	Day 254	240	0
Aerocapture, both vehicles in MO	Day 254	Day 257	3	2.3x2
Check up of systems	Day 255	Day 255	0	0
Separation with TEI propulsion stage	Day 255	Day 255	0	0
Mars descent and landing vehicle 1	Day 256	Day 256	0	4.1
Mars descent and landing vehicle 2	Day 257	Day 257	0	4.1
Number of days on Mars surface	Day 257	Day 757	500	0
Mars surface, propellant production	Day 258	Day 588	330	0
Takeoff for both vehicles, Mars Orbit	Day 757	Day 758	0	5.6x2
Junctions with TEI propulsion stage	Day 757	Day 758	1	0.1x2
TEI, vehicle 1 and vehicle 2	Day 758	Day 998	240	1.57x2
Capsules separation.	Day 998	Day 998	0	0
Earth landing with capsules.	Day 998	Day 998	0	11

[1] including gravity losses.

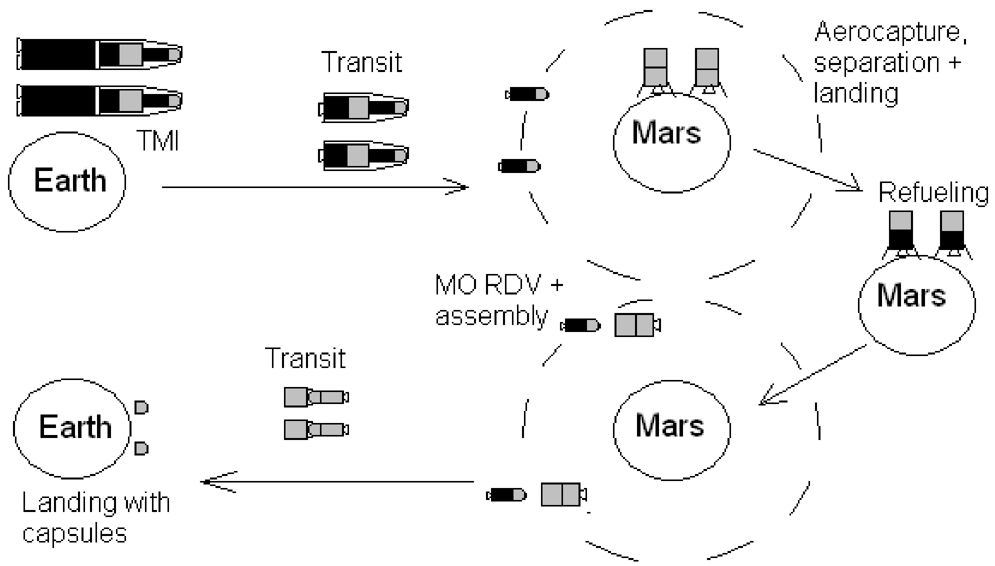

Figure 2: Main steps of the proposed scenario. Empty tanks are painted gray while filled tanks are painted black.

2.2 The right number of astronauts

In 1969, Von Braun proposed twelve astronauts [16]. In Mars direct, Zubrin and Baker proposed a crew of four with two astronauts specialized in operations and maintenance of the spacecraft and two scientists specialized in geology and biology [19]. In NASA design reference missions and in the HUMEX study from ESA, six astronauts were preferred [8], [9], [10]. What is the motivation for these numbers? In general, it is assumed that a small number of astronauts is imposed by the necessary minimization of the payload that has to be launched from Earth. The objective is then to find an optimal number for a good performance in terms of combination of skills, field exploration and science return [5], [7]. A detailed analysis has been undertaken by NASA in 1991 with further considerations in 2009 [9], [12]. The majority of crew activities fall into one of four categories: training, science and exploration, systems operations and maintenance, and programmatics (communication, reporting and real-time activity planning). In the same report, the authors explained that they focalized first on what the astronauts should do on Mars before considering the architecture. The crew size and composition was determined in a top-down manner: objectives => functions => skills => number of crew. The choice of 6 astronauts for the mission was therefore one of their first important assumption and the architecture of the scenario was designed according to that choice. However, what is the impact of the number of astronauts on the total mass that has to be sent to Mars, on the risks and eventually on the architecture of the mission? In the 1998 DRM (Design Reference Mission) report, a Three-Magnum scenario based on a simplification of the nominal NASA mission is mentioned [8]. Considering rather optimistic technological improvements and a simplified crew with 4 astronauts instead of 6, the proposed scenario enabled important modifications of the whole architecture with only 3 launches from Earth. Nevertheless, apart from that short study, the impact of the crew number on the architecture has not been addressed by NASA. That point is clearly stated in the addendum of the DRA report ([9] page 151): "In an attempt to reduce the overall IMLEO and costs that are associated with the mission, consideration must be given to reviewing the number of crew members that are required for this mission". Let us discuss some aspects of the problem. The first important issue is the minimum number of astronauts. Theoretically, all navigation maneuvers can be made automatic. On the surface of Mars, at least one astronaut is required to drive the rover, explore the surface and undertake scientific experiments in an efficient way. As Zubrin already stated, a single astronaut could be sufficient for the whole mission [20]. However, as it is the case in aeronautics, two persons are usually required for the maneuvers to come up with the eventuality of a health problem for the first pilot. In the Apollo program, they were two for the landing on the moon. The Russian space station MIR was also operated by two astronauts during several years. The same principle applies here. For every spacecraft maneuver, experiment on Mars or field exploration, the task is more secured if there is a second astronaut who can help in case of an emergency. Is there another significant reduction of the risks if we send more astronauts to Mars? There is no clear evidence for that. Two astronauts could therefore be the minimum number considering operational risks.

Another problem is to maximize the skills of the crew. The most important skill is the ability to use and repair the numerous systems onboard the spacecraft. Since there are many sensors and many devices for the production, processing or exploitation of different elements (oxygen, water, carbon dioxide, propellant, etc.), we suggest sending two persons with primary skills in chemistry, thermodynamics, mechanics and electricity. They would have to perform a long training phase for the use of onboard systems and other relevant domains, especially medicine, biology and astronautics. Eventually, a doctor with a long training in other domains might be preferred. As it is also stated in the DRA report, we have to find efficient solutions to mitigate health problems due to long stays in microgravity [9], [17]. Zubrin suggests spinning the spacecraft to simulate Martian gravity [21]. A more simple solution is to bring a centrifuge onboard [1], [2]. Some experiments have been conducted on Earth. For instance a centrifuge is currently used at MEDES (Institut de médecine et de physiologie spatiale in Toulouse, France), but the concept has not been tested in space. We assume in this paper that a small centrifuge would be onboard and would help in minimizing the physiological effects of microgravity. Other counter measures would also help in maintaining muscular strength and crew health. Even if the skills of the astronauts enable a good exploitation of the spacecraft, we also have to consider scientific skills to perform experiments in space and on the surface of Mars. The 4 astronauts of our scenario can work together on the surface of Mars but in comparison with a crew of 6 with different specialists, there would be a lack of expertise in some domains with a possible negative impact on scientific returns. However, if that drawback enables strong simplifications of the scenario, which in turn would make manned missions less risky and affordable, is that not the most suitable option?

The last issue concerns the risk of interpersonal conflicts [14]. In order to minimize them, the selection can be based on the compatibility of the two personalities. Such a selection is very difficult when considering large crews but for crews of 2 astronauts, the tests are easier and it is possible to select 2 persons who already know each other very well, for instance friends or brothers/sisters. Finally, is a crew of 2 riskier than a crew of 6? The problem is complex. Professors in experimental psychology who were solicited to give us an advice on the problem said that statistical data on long term cohabitation of people in confined environments are not abundant [5]. It is therefore uncertain that a crew of 2 would be riskier than a crew of 6 or the other way round.

2.3 Transhipment

The Apollo program was a great success. All astronauts returned safely to Earth. However, during the Apollo 13 mission, a terrible explosion occurred during the transit between LEO and the moon, which caused abort of the mission and endangered the crew. Considering the DRA or Mars Direct scenarios, if a similar problem occurs during the transit between Earth and Mars or Mars and the Earth, there would be no abort option to save the crew. In most scenarios, there are abort options while the crew is in LEO, in MO or on the surface of Mars but not during the transit phases. Is it reasonable to assume that the risks during the transit periods are very low? And if not, what can be done to reduce them? We propose to address both questions. Concerning the risks, the Apollo 13 problem suggests that there are not negligible. However, it is difficult to make an accurate estimation of the risks because many systems are involved and humans have strong capacities to mitigate the problems and repair the defected devices. In the DRA report, risk drivers have been examined [9]. The reliability of crewed equipment has been identified as a major issue for Mars manned missions. However, the uncertainty is very high and the risks highly depend on the modularity and reparability of the systems. In the case of the ISS, it has been clearly established that the probability of catastrophic loss is an order of magnitude smaller than the risk of evacuation. The astronauts could survive several days or weeks waiting for the rescue. Let us suppose a dramatic event occurs and there are only several days left to evacuate the vehicle. How to implement a rescue mission during that period? Obviously, during the transit stage, it is not possible to stop and come back to Earth. The only solution is to have another vehicle on the way to Mars as it was suggested in Von Braun's scenario [16]. Can a rendezvous on an interplanetary trajectory be easily accomplished? A similar question has been addressed in the DRA report in order to assess the “all-up” option [9]. The conclusion was that it is not an easy maneuver but "is probably manageable". Let us assume that only one vehicle is able to move and join the other and that the distance between the two vehicles is low to allow a rescue in a relatively short time with small ΔV requirements. Docking might not necessarily be an easy task. Providing that the damaged vehicle is safely reachable, not spinning for instance, a docking maneuver could be undertaken. The docking ports could be placed at the location of the airlock used for the exploration of Mars. Eventually, the astronauts of the damaged vehicle might have entered the capsule normally used for Earth landing. If this is the case, they should proceed to the ejection of the capsule first and try a docking maneuver afterwards. Once the rescue is accomplished, the mission would be abandoned. If the problem occurs early on the way to Mars, the safe vehicle could be placed on a free return trajectory with the help of the TEI propulsion stage [21]. If a free return trajectory is not possible, they would have to land on Mars in order to produce propellant for the return. If we allow the transhipment of a crew, what are the consequences on the specifications of the habitable module? A detailed analysis has to be undertaken to check the sustainability of the supplementary lives. Some elements depend on the volume of the structure but others have to be resized or added. In first approximation, we propose to keep the same habitable volume, to increase the mass of the life support system and to duplicate all consumables. Another issue is to allow more astronauts in the Earth re-entry vehicle at the end of the journey. Two persons should have access to the interface of the systems and it should be possible to add two passengers. Interestingly, a Soyouz Earth re-entry capsule can land with 2 astronauts and 150kg payload or 3 astronauts and 50kg payload. If no payload at all is taken onboard, a Soyouz based capsule with a reinforced heat shield and a little more volume and life support would practically be acceptable for a landing with four astronauts. Two important conclusions can finally be drawn. First, each of the two vehicles sent towards Mars can be a backup of the other. Second, in comparison with the DRA scenario, the volume and mass of the two habitable modules can be maintained to the strict minimum. Some mass savings are therefore expected for the proposed concept.

3. Details of the proposed scenario

3.1 Direct return or rendezvous in Mars orbit?

Since carbon dioxide is present in the Martian atmosphere, Zubrin and Baker proposed to send liquid hydrogen (LH2) to the surface of Mars and a chemical unit that would produce methane and oxygen, a possible propellant for rockets [19]. The amount was in the order of one hundred tons. In NASA reference missions, other options have been considered with a rendezvous in MO and less propellant produced on Mars. What is the best option? Let us examine three scenarios:

(1) In the first scenario, all propellant for the return is entirely produced on Mars and there is a direct return to Earth (Zubrin Mars Direct concept).

(2) In the second scenario, there is a propulsion stage waiting in MO with all propellant for TEI (DRA concept).

(3) The third scenario is a tradeoff between the first and the second. The idea is to produce all propellant for the return on the surface of Mars but to leave some elements in MO. A direct return is therefore not possible. The Mars ascent vehicle must perform a rendezvous in MO to join the other elements before TEI.

An important question is to determine the elements that can be left in MO. In order to optimize scenario 2 and 3, we propose to leave in orbit all systems that are not necessary on the surface of Mars, eventually at the expense of complex reorganizations of the spacecraft before TEI. The first evident element is the atmospheric Earth reentry capsule, which weighs in the order of 4 tons (sized for 4 astronauts). The second is the sum of consumables for the return. In astronautics, it is well known that two-stage launch vehicles are usually more efficient than single stage ones. Such a configuration is assumed for the 3 scenarios.

In order to determine the best scenario, an estimation of each IMLEO has been performed. Mass calculations follow a specific order:

- Mass of payload on the surface of Mars and eventually in Mars orbit that has to be sent back to Earth after surface operations.

- Mass of propellant, engines and tanks for the Mars ascent propulsion stage and for the TEI propulsion stage. It depends on the mass previously calculated.
- Mass of ISRU systems and associated power plants to produce the required amount of propellant.
- Mass of payload required on the surface of Mars and eventually mass of the payload waiting in Mars orbit. It depends on the payload and ISRU masses.
- Mass of the spacecraft to perform entry descent and landing on Mars. It depends on the size of the different modules and previous calculations.
- Mass of the spacecraft for TMI.
- IMLEO.

Parametric calculations have been made with Excel. The results are given in the next sections.

3.2 Estimation of the payload mass

Since the number of astronauts per vehicle is only two, we expect a reduction of the payload that has to be brought back to the Earth. In the DRM 3.0 report, the payload of the ERV is given with great details ([8], table A4-2 page 30). The numbers have been reproduced in Table 2. The total mass of the habitable module has been estimated to 21.7 tons without the capsule and 27 tons capsule included. In the DRA report, a similar table is proposed but the numbers correspond to the mass of the habitat at TMI and it is not clear how many tons of consumables remain on board for the return.

Table 2: Mass of the MAV at departure from Mars (day 757). The numbers of the first column have been copied from the table of the DRM report. They correspond to the mass of all systems present in the habitable module before TEI. A rough extrapolation has been made for a crew of 2 with estimation of the supplementary consumables to sustain a crew of four.

<table>
<tr><th>Subsystem</th><th colspan="2">Mass (kg) for 6 astronauts (DRM)</th><th>Mass (kg) for 2 astronauts</th><th>Suppl. mass (kg) for 2 more</th></tr>
<tr><td>Life Support System</td><td colspan="2">4661</td><td>3000</td><td>500</td></tr>
<tr><td>Crew accommodations and consumables</td><td>12058</td><td rowspan="2">4666</td><td rowspan="2">1555</td><td rowspan="2">1555</td></tr>
<tr><td>Consumables jettisoned before TEI
(it is stated in DRM that these consumables were needed in case the crew had to leave the surface before the expected date)</td><td>-7392</td></tr>
<tr><td>EVA equipment</td><td colspan="2">243</td><td>150</td><td>50</td></tr>
<tr><td>Comm/info management</td><td colspan="2">320</td><td>320</td><td>0</td></tr>
<tr><td>Power prod. 30 kWe P.V.A.</td><td colspan="2">3249</td><td>1000</td><td>200</td></tr>
<tr><td>Thermal control system</td><td colspan="2">550</td><td>400</td><td>0</td></tr>
<tr><td>Structure</td><td colspan="2">5500</td><td>3000</td><td>0</td></tr>
<tr><td>Science equipment (left on Mars for the new scenario)</td><td colspan="2">600</td><td>0</td><td>0</td></tr>
<tr><td>Spares</td><td colspan="2">1924</td><td>1000</td><td>0</td></tr>
<tr><td>Total without capsule and crew</td><td colspan="2">21713</td><td>10425</td><td>2305</td></tr>
<tr><td>Earth reentry capsule + crew and samples
(also Mars ascent vehicle in DRM)</td><td colspan="2">5329</td><td>3000</td><td>1000</td></tr>
<tr><td rowspan="2">Total (kg)</td><td colspan="2" rowspan="2">27042</td><td>13425</td><td>3305</td></tr>
<tr><td colspan="2">16730</td></tr>
</table>

In all scenarios, we assume that the volume of the habitat is defined for a crew of two and the consumables for a crew of four. Such hypotheses have an impact on almost all subsystems. A rough extrapolation of the payload is proposed in Table 3. Concerning accommodations and life support, as it is suggested in DRM3.0, regenerative life support systems should be used, especially to recycle air and water. It is assumed that the mass of consumables and accommodations is linear with the number of astronauts. It is important to note that the mass of photovoltaic arrays (PVA) can be reduced using state-of-the-art technology [18]. 175 We/kg have indeed already been achieved (AMO Standard 1357 W/m^2). Since Mars is further to the sun than the Earth, even if that specific power value is divided by 3, which is less than the worst case (Mars aphelion), 60 We/kg can be achieved. A 30 kWe PVA would weigh in the order of 500kg. This is the reason why the proposed extrapolation is much lower than the value from the DRM report. In first approximation the mass of the spacecraft is divided by two. Moreover, for scenarios 2 and 3 the Earth reentry capsule, 90% of the consumables for the return, some equipment (a set of solar arrays for instance) and some spares are left in MO. It is difficult to make an estimation of the remaining payload for the Mars ascent vehicle. In first approximation, it is assumed that the total mass of the Earth reentry capsule and the consumables and equipment waiting in MO could be in the order of 7 tons. With these hypotheses, it is possible to estimate the mass of the propulsion stage and compare the three scenarios.

3.3 Estimation of the propulsion stage mass

Let us make an estimation of the mass of propellant according to the payload mass for Mars ascent. We note M_p the mass of propellant (CH4/O2), M_s the mass of the propulsion stage (dry) and M_h the mass of the payload (habitable module). We therefore have $M_{total} = M_h+M_p+M_s$. We assume the following parameters:
- Two stages are used for Mars ascent and then TEI.
- Specific impulse of propellant CH4/O2 (ISP) is 370s. In the DRM report, it was assumed an ISP of 379s and other authors proposed an ISP of 368s or 369s [4], [8], [13].
- The structural mass to propellant mass ratio r = M_s/M_p is set to 12%.

M_p is linearly linked to M_h thanks to Tsiolkovsky equation. See Equation (1) and (2)

$$\Delta V = ISP \cdot g \cdot \ln\left(\frac{M_h + (1+r)M_p}{M_h + rM_p}\right) \qquad (1)$$

Let $K = e^{\frac{\Delta V}{ISP \cdot g}}$, we get: $$M_p = \frac{K-1}{(1+r) - rK} M_h \qquad (2)$$

For scenario 2, the rendezvous in MO is similar to the one proposed in DRA with the same ΔV requirements for MO and TEI 5.625 km/s and 1.57 km/s respectively [9]. For scenarios 1 and 3, a two stage propulsion system is also used but it is preferable to split the ΔV differently in order to minimize the total mass of the spacecraft. For the direct return, the total ΔV is smaller by 0.2 km/s because there is no need to secure a rendezvous in MO.

For a given payload mass on the surface of Mars and another payload in MO, it is possible to calculate the total mass of the vehicle according to the 3 scenarios. The assessment is presented in Tables 3a. At this point of the study, scenario 3 minimizes IMLEO. The propulsion stage waiting in MO with all the propellant for the return is indeed a mass penalty for scenario 2. However, the mass of the propulsion stage for ascent from the surface is much heavier in scenario 1 and 3. The difference between the 3 scenarios is finally less than 4 tons, which is very small. Let us examine the mass of other systems have not been taken into account. The heat shield, the thermal protection system and the propellant for the descent depend on the volume and mass that have to be sent to the surface. In first approximation, the mass of the thermal protection system is indeed proportional to the surface of the spacecraft and the mass of propellant is roughly proportional to the mass of the spacecraft. According to DRA, the mass of heat shield and propellant is typically in the order of 50% of the total mass of the descent vehicle: the payload mass plus the dry descent stage mass equals 57 tons and the total mass of the spacecraft on Mars arrival is 115 tons [9, p. 227]. All in all, as it is shown in Table 3b, in first approximation, it is expected that scenario 2 would be in fact the one that minimizes IMLEO. In addition, in scenario 2, the mass of propellant that must be produced on the surface is half the amounts of the two other scenarios. Other mass savings are therefore expected concerning ISRU systems. Different parameters have been chosen for the mass of the payload, the structural mass to propellant mass ratio and ΔVs but the results were similar. Last but not least, as it will be soon explained, EDL (Entry, Descent and Landing) risks are tightly linked to the mass of the landed spacecraft. Scenario 2 is also preferable for that important reason. The first important conclusion is that in order to minimize IMLEO and EDL risks it is preferable to send a propulsion stage already filled with propellant for TEI than to produce all the propellant on the surface and perform a direct return. What is also interesting is that the mass of propellant that has to be produced in scenario 2 is an intermediate value between the one suggested in DRA and the one calculated by Zubrin for Mars Direct.

Table 3a: Earth return comparison for the 3 scenarios.

All values are given in tons		**Scenario 1:** Direct return	**Scenario 2:** Propulsion stage + 7 tons payload waiting in MO	**Scenario 3:** 7 tons payload waiting in MO
Total payload mass for TEI from Table 2		16.7	16.7	16.7
Propellant for TEI	ΔV=1.57 km/s	N/A	9.7	N/A
	ΔV=3 km/s	25.4	N/A	25.4
TEI propulsion stage dry mass (12% of propellant)		3.1	1.2	3.1
Total mass for TEI		**45.2**	**27.6**	**45.2**
Payload waiting in MO		0	7.5	7.5
Propulsion stage (wet) waiting in MO		0	10.8	0
Payload for Mars ascent, TEI propulsion stage included for scenario 1 and 3		45.2	9.2	37.7
Propellant for Mars ascent	ΔV=5.625 km/s	N/A	61.7	N/A
	ΔV=4 km/s	119.8	N/A	100.0
Mars ascent propulsion stage dry mass (12% of prop.)		14.4	7.4	12
Total mass of propellant that must be produced on Mars		**165.2**	**61.7**	**125.4**
Total mass imported from Earth (1)		**34.2**	**35.0**	**31.8**

Table 3b: Results of the comparison. Scenario 2 minimizes IMLEO.

All values are given in tons	**Scenario 1:** Direct return	**Scenario 2:** Propulsion stage + 7 tons payload waiting in MO	**Scenario 3:** 7 tons payload waiting in MO
Landed payload and propulsion stages (tons)	34.2	16.6	24.3
Estimated impact on the mass of the thermal protection system, RCS and descent propellant. (+ 100%) (2)	34.2	16.6	24.3
Total mass from Earth when all elements are taken into account (1)+(2)	**68.4**	**51.6**	**56.1**

3.4 Additional payload mass for Mars exploration

In the previous section, an estimation of the mass of the habitable module at the end of the stay on Mars has been given. In order to estimate the total payload mass for landing, the mass of consumables needed for the previous period of the journey, the mass of some surface equipment and the mass of a chemical unit for the production of propellant have to be added.

- Mass of consumables

The mass of most consumables linearly depends on time. On the surface of Mars it is possible to use local resources. For instance, some water can be extracted from the soil. However, if it is not possible to land on Mars there should be enough consumables for the round trip. In Table 3, the estimation was only for the return. For 500 days on the surface of Mars, the increase in consumables is close to 3.1 tons. In order to secure transshipments in every situation, we propose to double that amount. The increase is therefore 6.2 tons per spacecraft.

- Pressurized or unpressurized rovers

In order to explore the surface of Mars and perform scientific experiments, astronauts need EVA equipment and rovers. A pressurized rover typically weighs several tons while unpressurized ones could be in the order of one hundred kilograms. Since its weight is not negligible, the choice of the pressurized version should be clearly justified. There is a need for a pressurized rover in order to be able to explore at reasonable distances from the habitat. In all Mars semi-direct scenarios, a pressurized rover is imposed to come up with the risk of landing far from the ERV [8], [9], [12], [20]. The maximum distance strongly depends on the size and reliability of the vehicle. In the DRA report, a modest pressurized rover is proposed. It would be able to support a crew of two during two weeks and travel for a total distance of 100 km [9]. What is the best choice for our scenario? Zubrin argues that unpressurized rovers may facilitate the exploration of rough terrains because they can be easily hand-moved [21]. In addition, it is very easy and quick to make a stop for a few minutes in order to take a sample of rocks or to access the top of a hill by walking. In a pressurized rover, a short stop is not simple because the astronauts have to put their spacesuit on before EVA. Furthermore, even when the pressurized rover is not used, its habitability has to be maintained. In order to extend field exploration with unpressurized rovers, a promising idea could be the use of inflatable structures that would provide temporary habitats. The question is therefore complex. In order to keep the mission as simple as possible, we prefer the option that minimizes the payload and suggest the use of small unpressurized rovers. The additional mass of that equipment is finally extrapolated from the DRM report [8]. See Table 6.

3.5 ISRU for propellant

A chemical unit and a power plant must be sent to the surface of Mars in order to produce propellant for the return. In NASA reference missions or in Mars Direct, there is a risk of landing the chemical unit too far from the crew or in an inaccessible zone. It is important to note that such a risk has been clearly identified as a major concern in NASA reference missions with the implication of landing a heavy long range rover [8]. That risk does not exist in our scenario since ISRU is stored in the same spacecraft.

It is suggested here to produce all the propellant for the return using in situ resources. The propellant is CH_4/O_2. Our estimation of the additional payload mass is based on the DRA report, in which the production of CH_4 and O_2 is detailed [9]. With a 1:3.6 CH_4 O_2 mixture ratio (best ratio for optimal ISP [4]), considering that 62 tons of propellant have to be produced, the mass of H_2 that has to be transformed into CH_4 is 3.3 tons. In DRA, CH_4 is brought from Earth but the option of producing all the propellant using ISRU has been investigated [9, section 6.2]. H_2 is obtained by water electrolysis and water can be extracted from the Martian surface using excavation tools and other devices. For 3.3 tons of H_2, 30.1 tons of H_2O have to be extracted. An estimation of the tools mass for water extraction is given in Table 4. Methane can be obtained from carbon dioxide and water by means of water electrolysis and Sabatier reaction. An estimation of the ISRU mass has also been made, see Table 5.

Table 4: Mass of systems for water extraction

	DRA estimated mass (kg) for 2146 kg H2O [DRA p 266]	DRA estimated mass (kg) for 16788 kg H2O [DRA p 266]	2 points linear extrapolation for 30100 kg
Excavation subsystem, soil with 3% H2O, 8 hours operations per day	852	1970	3000
H2O extraction subsystem, soil with 3% H2O, 8 hours operations per day	182	669	1100

Table 5: Mass of systems for Sabatier reaction and electrolysis.

	DRA estimated mass (kg) for 31458 kg of CH4/O2, 24 hours production per day [DRA p 263]	Extrapolation for 8 hours production per day (x3)	Linear extrapolation (kg) for 62 tons of CH4/O2, 8 hours production per day
Sabatier water electrolysis system for the production of CH4/O2	479	1437	2810

In the new scenario, the requirement is to be able to produce the propellant in less than the duration of the surface mission, which is 500 days. We need large margins for security reasons, so we propose 300 days which is the same period of time as in NASA scenario.

3.6 Powering ISRU

Regarding power requirements for ISRU, it is suggested in DRM3.0 that a nuclear reactor would supply 32 kWe only for the production of propellant. According to DRA estimations for the systems presented in Tables 4 and 5 and our extrapolation for the production of 62 tons of propellant, the requirements for the proposed scenario are in the order of 51 kWe for a continuous production. However, is a nuclear reactor the best option or are solar arrays preferable? In a recent comparison, it is claimed that solar arrays coupled with batteries or regenerative fuel cells may be the best option regardless of power requirements [6]. In the DRA report, there is an assessment of both technologies. The nuclear reactor option is preferred. In the proposed scenario, a small nuclear reactor may not be a good solution because its volume would not be negligible (impact on the size of the spacecraft, and therefore on TPS and heat shield) and it would be difficult to deploy it on the surface of Mars. Solar arrays have several drawbacks. The deployment of large scale arrays is complex and the dust that may cover the arrays time after time has to be periodically removed. However, with humans on the surface of Mars, such problems can easily be overcome. Another important issue is the availability of sunlight power. It is possible to use batteries that would be charged during the day for a continuous exploitation of ISRU systems. It is suggested in the DRA report that the best option would be to use ISRU systems daylight only with arrays supplying power 8 hours per day. The mass of ISRU systems for 8-hours operations has already been estimated in Tables 4 and 5. Power requirement for 8-hours operations is 153 kWe. What would be the mass of such solar arrays and what would be the volume for storage? According to recent advances in solar arrays technologies, 175 We/kg can be achieved in LEO with arrays supplying 7 kWe from an array surface of 23.7 square meters [18]. The efficiency of the array is 295 We/m^2. Mars is further from the sun and the arrays are not always facing the sun during the 8-hours exposition. For the sake of simplicity, let us assume that a quarter of that amount e.g. 75 We/m^2 can be supplied all day long. The specific power at the surface of Mars would therefore be close to 44 We/kg (≈175/4). According to specialists of the domain, with the help of humans for the deployment and maintenance of the arrays, much better performances might be achieved using large flexible arrays stored in rollable blankets [15]. Further investigations are needed to determine the limit of the technology in the context of the Martian surface. It is proposed in this paper to consider that 40 W/kg is achievable for the power plant based on solar arrays everything included (wires, electrical devices and eventually batteries to keep critical ISRU systems warm). For 153 kWe, the total mass is therefore around 3.8 tons. The size of the set of solar arrays would be huge, about 2000 m^2. It might take several days to deploy the arrays on the surface of Mars. The problem has to be seriously addressed so that the work can be done in a reasonable time. Concerning the volume of the arrays, there already exist solutions for a compact stowage volume close to 33 kWe/m^3 for space missions in LEO [18]. On Mars, the same arrays per cubic meter would supply only 8.2 kWe. For 153 kWe, 19 cubic meters are therefore needed, which seems manageable. More importantly, solar arrays may be stored in compact containers on top of the habitable module with a small impact on the size of the spacecraft. If flexible arrays can be used, the volume might even be greatly reduced. Solar arrays are therefore the preferred option for the proposed scenario.

3.7 Entry, descent and landing

Finally, we have to consider the mass of heat shield, TPS (Thermal Protection System), propellant for the descent and other landing mechanisms. EDL (Entry Descent and Landing) challenges have been examined by Braun and Manning [3]. They suggest the use of large supersonic parachutes to decelerate heavy payloads and minimize the propellant mass for the last propulsive phase. However, such parachutes can be deployed only if the velocity of the spacecraft is not too high, which means that the ballistic coefficient should be low enough to enable strong decelerations in the upper layers of the Martian atmosphere. According to the authors, provided that the parachutes can be deployed, the propellant mass fraction would be in the range 12-18%.

Table 6: Mass of the rocket for the landing on Mars

Habitable module (kg)	Mass at launch from Mars (Table 3.a)	9200
	Additional consumables mass	6200
	Unpressurized rovers (from DRM3.0)	375
	EVA suits and consumables (from DRM 3.0)	462
	Science equipment	500
	Subtotal	**16737**
ISRU, chemical unit, LH2 and solar arrays (kg)	Excavation systems	3000
	Water extraction systems	1100
	Sabatier reactor and electrolysis unit	2810
	Power systems	3850
	Subtotal	**10760**
Total payload Habitat + ISRU and power systems (tons)		**27.5**
Additional landed mass (tons)	Propulsion stage (Table 4)	7.4
Total landed mass		**34.9**
Additional entry mass: TPS, heat shield, propellant, RCS, parachutes=50% (tons)		**34.9**
Total mass for descent vehicle, rounded (tons)		**69.8**

In the DRA report, a detailed analysis has been performed [9]. The estimation is based on a specific configuration with 40 tons payload and a 30 meters long spacecraft with elliptic shape. In the proposed scenario, the payload mass is only 27.5 tons and the size of the spacecraft is smaller. Since the same propulsion stage is used for the descent and the ascent, its dry mass has already been estimated. It is equal to 7.4 tons (from Table 3). For the other systems, it is difficult to make an estimation of the supplementary mass. In first approximation, if we assume that the ratio between that supplementary mass and the total mass of the spacecraft for EDL is a constant, an extrapolation can be performed with the values proposed in the DRA report. That assumption is probably pessimistic because heating and decelerating constraints are lower for a lighter and smaller spacecraft [3]. As it was already mentioned in Section 3.3, in the DRA, the payload mass plus the dry descent stage mass equals 57 tons and the total mass of the spacecraft on Mars arrival is 115 tons [9, p. 227]. Assuming the same ratio (50%), the supplementary mass is 34.9 tons. The total mass of the spacecraft before EDL is therefore around 70 tons (see Table 6).

3.8 IMLEO

The total mass of the vehicle for MOI can be easily inferred. It is equal to 88.1 tons (see Table 7). There are different ways to send spacecrafts to Mars. Let us examine the simplest one with the use of chemical propulsion stages based on LO2/LH2. The IMLEO of our scenario has been estimated using the same procedure described in Section 3.2. The only differences are the payload mass, the total ΔV for TMI which has been set to 3.6 km/s and ISP that has been set to 450s. The resulting IMLEO is 246 tons for each rocket. It is a huge payload for a single launcher. A simple solution could be to split the TMI spacecraft in two parts, the propulsion stages and the payload and to launch them separately. An assembly of the TMI spacecraft would be needed in LEO but the requirement for the heavy launcher would be in the order of 155 tons in LEO, which seems realistic.

Table 7: Mass for MOI and IMLEO.

<table>
<tr><td colspan="2">Total mass for descent vehicle (tons)</td><td>69.8</td></tr>
<tr><td colspan="2">Total mass waiting in MO (tons)</td><td>18.3</td></tr>
<tr><td colspan="2">Total mass for MOI (tons)</td><td>88.1</td></tr>
<tr><td colspan="2">Additional consumables for outbound trip (tons)</td><td>3.1</td></tr>
<tr><td colspan="2">Total payload mass for TMI</td><td>91.2</td></tr>
<tr><td rowspan="6">TMI spacecraft
2 propulsion stages, ΔV=1.8km/s for each</td><td>Mass H2/O2 2nd stage (tons)</td><td>48.9</td></tr>
<tr><td>2nd stage dry mass (tons)</td><td>5.86</td></tr>
<tr><td>Payload 1st stage (tons)</td><td>146</td></tr>
<tr><td>Mass H2/O2 1st stage (tons)</td><td>89.8</td></tr>
<tr><td>1st stage dry mass (tons)</td><td>10.8</td></tr>
<tr><td>Total IMLEO</td><td>246.5</td></tr>
<tr><td colspan="2">Total IMLEO for proposed scenario with 2 spacecrafts sent to Mars (tons)</td><td>493</td></tr>
</table>

4. Assessment

4.1 ISRU risks

In our scenario, a critical point is that the Mars ascent vehicle is not ready for launch when the astronauts land on Mars. A pre-deploy strategy would be possible but it would require a new spacecraft with its own propulsion stage and complex robots to deploy, start and maintain solar arrays and ISRU systems. In addition, the manned spacecraft would have to land very close to the pre-deployed cargo and long range surface vehicles would be required. An all-up scenario is therefore preferable because it is simpler and minimizes IMLEO. How high is the risk that the astronauts fail in producing the required amount of propellant with ISRU? ISRU systems are based on well-known chemical reactions and thermodynamic properties. A Sabatier reactor is currently used in the ISS and electrolysis units are used in all submarines but in the harsh environment of Mars, the conditions are totally different. The specifications and performance of ISRU systems in the context of Mars have already been studied [9], [12]. Simple experiments have been conducted on Earth but not on Mars and not at the right scale. Nevertheless, it was sufficient to perform an extrapolation and predict the mass and

performance of the systems for the production of propellant on Mars. The elements and the performance of the systems are detailed in several tables of the DRA report with different requirements and constraints [9]. Before any manned mission to Mars, ISRU systems will have to be tested and validated at the right scale in simulated Martian environments, dust storms included. Since solar arrays are proposed, a major issue could be to achieve the production of the required amount of propellant if one or several dust storms significantly reduce sunlight during a long period of time. In the proposed study, we already took into account in the order of 200 days margin for the production period (300 days for the production time, around 500 days for the entire stay). The longest dust storms last several months but they are rare. One of the longest occurred during a Viking mission. Its duration was approximately 100 days. In the proposed scenario, there are 200 days margin and the sunlight would not be totally absent during the production period. The risk of an insufficient production would therefore be very low. However the deployment of power systems, dust storms effects, flight qualifications, extreme temperatures and every other possible problem will have to be examined with great details. If the proposed margins are not sufficient, new margins would have to be considered. And if there are still uncertainties on the properties of the environment and on the performance of the systems on Mars, a precursor robotic mission should be programmed to test and validate ISRU technologies in a Martian environment.
After validation of the systems, since the components are relatively simple, the risk of loosing ISRU capability should be several orders of magnitude lower than the risk of loosing a spacecraft during launch or landing. An interesting question is to understand why in all NASA scenarios should the MAV be ready for launch before the manned spacecraft is sent to Mars? In the DRA report, a pre-deploy strategy (cargo with ISRU systems sent to Mars 2 years before the manned mission) is compared to an all-up strategy (cargo and manned spacecraft in the same launch window) [9, p. 54]. As in previous NASA scenarios or in Mars direct, the pre-deploy strategy is preferred for redundancy reasons [8], [9], [21]. In the pre-deploy strategy of the DRA, the backup SHAB and MAV are indeed the SHAB and MAV of the second mission. Such backups are not possible in the all-up strategy examined by NASA. The risk of loosing ISRU capability is therefore real but it is not the key reason why the pre-deploy strategy is preferred by NASA. In the scenario proposed in this paper, since the redundancy already exists in a single mission, the pre-deploy strategy is not justified.
Even if the robotic mission is a success, the risk of loosing full ISRU capability will still exist. It is mitigated by the duplication of the systems and the human ability to repair defected parts. Once again, many reparations scenarios have to be simulated on Earth to minimize the risk. If the two ISRU systems are nevertheless out of order, the astronauts would have sufficient consumables to live at least another period of 500 days on the surface. In addition, they would have tools for soil excavation, water extraction and water electrolysis. It would therefore be possible for them to get water and oxygen from Martian resources and increase their stocks. Provided that it is programmed by space agencies, they would be able to wait for the next mission to Mars. During their stay on the surface, they would have time to determine the exact cause of the problem with the help of experts on Earth. Once the problem is identified, the next mission (manned or automatic) could bring to them what they need to repair or replace the defected parts.

4.2 Comparison with DRA

We propose a comparison between the DRA scenario and ours. The main differences are presented in Table 8 for the payload. As it was expected, even though our estimation is approximate IMLEO is much smaller in our scenario and the gain is significant. It can be explained by several factors. First, in the proposed scenario, there are 2 habitable modules, but they are small since they are sized for 2 persons each, while in the DRA there are 2 big

habitable modules for the transit phases and on the surface of Mars and another one for the ascent vehicle. Second, the mass of propellant produced on Mars is higher. The savings are very important because the total mass of ISRU systems is low and it also has an impact on the volume and mass of the spacecraft, which in turn have an impact on TPS and propellant for EDL. Third, there is an optimization of specific elements. For instance, in DRA, the capsule used for reentry in the Earth's atmosphere is Orion, which is a multi-purpose capsule sized for six astronauts and provides life support for several days. In the proposed scenario, the capsule is only used for the reentry in the Earth's atmosphere. If its size is optimized, its mass could be less than one third of the mass of the Orion capsule.
Regarding the risks, it is stated in the DRA report that “current design philosophies and technologies would not provide an acceptable level of reliability for a Mars mission” [9 p.145]. Let us consider the main risk drivers listed in the report.
- According to NASA, the most important risk for the DRA mission is the failure of the launch and integration stage because of the required level of mass to LEO. The number of launches is indeed in the order of ten and there are strict constraints for the launch windows. The average risk of failure has been estimated to 26% [DRA page 142]. In the scenario proposed in this paper, the number of launches is only four. The risk of failure is thus highly reduced for that specific stage. Moreover, the assembly of the two spacecrafts can be made in LEO and the technology of chemical propulsion stages is mature.

Table 8: Comparison between the DRA scenario and ours.

	Item	**DRA scenario**	**Proposed scenario**
On the surface of Mars	Number of vehicles landed on Mars	2 (hab + MAV)	2
	Number of propulsion stages on Mars	4	2
	Total mass sent to Mars, aeroshells excepted (tons)	127	69.8 (34.9x2)
	Total mass for descent vehicles (tons)	214 (DRA, table 4-5 p. 141)	140 (69.8x2)
	Mass of propellant produced on Mars (tons)	7.5 (only O2)	124 (62x2)
In space	Payload waiting in MO (tons)	51	15 (7.5x2)
	TMI core stages of dedicated transit vehicles, payload and propellant not included (tons)	NTR cargo1: 96.6 NTR cargo2: 96.6 NTR crew: 106.2	22 (10.8x2)
	Total Mass after TMI (tons)	700 (approx.)	182.4
	IMLEO (tons)	849 (table 4-1 p. 27)	493

- The second most important risk referenced in the DRA report (18%) is linked to Mars EDL. It is clearly stated in the report that for heavy vehicles the feasibility of the entry, descent and landing stage has to be proven. Braun and Manning have indeed shown that it could be very difficult to brake in the Martian atmosphere if the ballistic coefficient is too high [3]. For a given aeroshell with a diameter in the order of 15 meters, light payloads can aerodynamically decelerate to Mach 2 at reasonable altitudes so that the parachutes can be opened, the aeroshell jettisoned and the last propulsive phase started. Heavier payloads would reach Mach 2 at lower altitudes and the timeline for the last descent maneuvers would be very tight with very short margins if feasible at all. Larger aeroshells might be used, but there would be packaging problems and it would be more difficult to deploy them. Eventually, the propulsive phase might start earlier without use of parachutes but this would require much more propellant and more powerful engines. The mass of the payload has therefore an important impact on the complexity and risks of EDL maneuvers. In our scenario, the descent vehicles are smaller and lighter because the habitable module is sized for 2 astronauts and everything

that is not necessary on the surface is left in MO. EDL maneuvers are simpler and risks are therefore lower in our scenario.
- Another important risk referenced in the DRA report concerns the reliability of the equipment for the total duration of the mission. Since the surface habitat is sent two years prior to the crew, the lifetime of the systems must be in the order of 5 years. In our scenario, the duration of the mission is only 3 years. The associated risks are therefore lower. Moreover, the duplication of the mission allows the loss of some equipment without impact on scientific returns or crew health.
Another risk exists at takeoff from the surface of Mars. It is possible that the first spacecraft succeeds while the second has a problem. In order to minimize the risks, we suggest that the two spacecrafts try to takeoff simultaneously. All steps of the launch procedure have to be followed at the same time in the two spacecrafts and the final Go has to be allowed only if the other spacecraft is also ready for immediate launch.

4.3 Cost estimation

What would be the cost of the proposed mission? For the first semi-direct scenario elaborated by NASA, the total cost had been estimated at 55 billions dollars [12]. Taking into account US inflation for the last 16 years, that amount rises up to 80 billion dollars. The DRA scenario is probably more expensive, but we are not aware of any estimation of the cost. Compared to the semi-direct scenario proposed in 1994, our scenario is simpler:

- IMLEO is divided by 2: 438 tons vs 850 tons. (In DRA, IMLEO is also greater than 800 tons.)
- No NTR has to be built.
- There is no need for a robotic mission to produce propellant before the arrival of the astronauts.

We recommend a preliminary unmanned mission to test the landing of a heavy spacecraft, the production of propellant using in situ resources and the launch of a spacecraft from the surface of Mars. However, such tests should be performed for all the scenarios, eventually at smaller scales for ISRU systems and the launch from Mars.

We did not perform a detailed analysis of the savings. Based on a rough estimation of the above simplifications and considering that one third of the cost is tightly linked to IMLEO (32% of the total cost for DRM 1.0 [12]), we believe that the cost of our scenario could be 50% less than the cost of DRM. The total cost would therefore be in the order of 40 billion dollars. Moreover, most systems of the proposed scenario are based on mature technologies. The risk of costs overruns is thus rather limited.

4.4 Programmatic issue

The first manned mission to Mars should not be a one shot mission. Conversely, it should be integrated in a consistent long term programme of Mars exploration. Is the proposed scenario appropriate for that? If it is desired to explore several regions, the same scenario might be duplicated several times. However, it might be interesting to send a long range rover or a heavy payload to the surface. Would we have to reconsider the size of the spacecraft for that, to perform new tests and eventually to change the scenario? As it has already been explained, the size and mass of the vehicle that has to be sent to the surface play an important role in the feasibility and risks of the EDL stage. Moreover, if the spacecraft is heavier at launch from Mars, all ISRU systems have to be reconfigured and in turn the heat shield, TPS and the amount of propellant for the descent. It is therefore recommended to keep the same configuration of the spacecraft to avoid new calculations, new tests and new risks. If an exceptional payload is desired, a simple solution would be to create a small cargo module, to attach the cargo to the spacecraft for the outbound stage, but to perform a separated EDL

phase. The cargo would have to land very close to the landed manned spacecraft and the astronauts would have to join it by means of their unpressurized rovers. For the launch from Mars, since the cargo and its content can be left on the surface of Mars, the size and mass of the vehicle is still minimum and optimized.
If more astronauts are desired on the surface of Mars, more than two vehicles can be sent to Mars. For instance, if 6 astronauts is the prefered number, the best option would be to send 3 vehicles to Mars with 2 astronauts per spacecraft. That configuration would probably still be simpler, cheaper and less risky than the one proposed by NASA.
Another issue is to enable important evolutions of the spacecraft in order to speed up the transit between the Earth and Mars. For instance, efficient ion engines can be used to accelerate the spacecraft from LEO to Mars. Such evolutions are indeed possible without reconfiguration of the vehicle that has to land on the surface of Mars.
The proposed scenario can therefore be integrated in a robust strategy for Mars exploration.

5. Conclusion

The main idea of the proposed scenario is to reduce the number of astronauts in each spacecraft, to choose an all-up strategy and to duplicate the mission. In order to minimize IMLEO, it has been shown that the best option is to leave a propulsion stage and some elements in MO after MOI. The proposed architecture minimizes the size and the mass of what is sent to the surface, which in turn allows a reduction of the mass of engines, tanks, TPS, heat shield and ISRU systems (less propellant has to be produced). As in the Apollo Program, a rendezvous must be performed in MO in order to join the Earth return propulsion stage and other elements needed for the last stage of the mission. According to the first investigations presented in this paper, the proposed scenario is simpler, cheaper and less risky than the last reference mission from NASA. The total cost of the new scenario has been estimated to 40 billion dollars. Science returns might be less important than those expected with NASA scenario, but for the same price 2 missions might be programmed. Nevertheless, our proposal leaves numerous questions open and several assumptions have to be checked. The same propulsion stage is used for the descent to the surface and the ascent to MO. Is that realistic? For the mass of the habitable module landed on Mars, our estimation is in the order of 10 tons. Is it a correct estimation? The empty mass to propellant mass ratio is also an important parameter. Is 12% a reasonable assumption? Other important questions have to be addressed. We propose the use of large solar arrays on the surface of Mars to supply power to ISRU systems. What would be the exact mass? Would it be difficult to deploy the arrays? How much time would it take? How to mitigate dust storms effects? Finally, prior to any manned mission to Mars, it is important to test the landing of a heavy spacecraft, the production of propellant using in situ resources and the launch from Mars. Depending on the results of the tests, new scenarios may have to be proposed to overcome the difficulties. For instance, if it is imperative to reduce the mass of the spacecraft for the EDL stage, a solution could be to land ISRU systems separately. The proposed study is not intended to prove that the best scenario should follow all options presented in the paper but to provide a fair first estimation of the mass of each system, to bring new ideas and to arouse new works in that domain.
After publication in Acta Astronautica, the author carried on working on the scenario. There was a specific concern with the mass of the landing vehicles. The author investigated EDL options and imagined another strategy: What about splitting the landing vehicle in 2 smaller landers, a cargo vehicle that could bring ISRU systems on Mars 2 years before the manned mission and a habitat lander with the astronauts? These options are presented and discussed in the second part.

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PART 2:

Revised Scenario for Human Missions to Mars

List of acronyms:
EDL: Entry Descent and Landing
ERV: Earth Return Vehicle
IMLEO: Initial Mass in Low Earth Orbit
ISP: Specific Impulse
ISRU: In Situ Resource Utilization
LEO : Low Earth Orbit
MAV: Mars Ascent Vehicle
SLS: Space Launch System (NASA heavy launcher)
TEI: Trans-Earth Injection
TMI: Trans-Mars Injection

Abstract
We present a revised version of our scenario for human missions to Mars. The idea is to take into consideration the difficulties and constraints for entry, descent and landing by splitting the heavy vehicle into two smaller ones. The standard capsule shape is thus possible for aerocapture and landing on Mars. It is suggested to use the largest possible diameter such that the ballistic coefficient is minimized and the lift to drag ratio is kept small. The maneuvers for the descent and landing are then simplified and the risks are minimized. The scenario has been modified to cope with the new constraints. Different options have been taken into account. It is possible to land a small Mars ascent vehicle or to reuse the habitat lander for Mars ascent. All options perform as well as the others for the criterion of the initial mass in low Earth orbit. However, reusing the habitat lander allows a significant reduction of the size of the Earth return vehicle, which otherwise requires a huge launching capability.

1. Introduction
Human missions to Mars have been studied by many authors [12], [13], [16], [20], [27], [28]. In a recent work, a new scenario for human missions to Mars has been proposed [21], [22]. It is based on Von Braun's original idea with the number of crew limited to two astronauts in each vehicle, the exploitation of local resources to produce propellant for the return and a capsule left in Mars Orbit to minimize the payload on the surface of Mars. In NASA reference missions or in Zubrin's Mars Direct scenario, the pre-deploy option is preferred [11], [12], [14], [27]. The Mars ascent vehicle is pre-deployed and ready to take off from Mars before the first manned vehicle is sent to the planet. However with this option, it might not be possible to choose the most efficient technologies that minimize the initial mass in low Earth orbit (IMLEO). When humans are on the surface of Mars, ultra light solar panels can be deployed and eventually cleaned after a dust storm, complex in situ resource utilization (ISRU) systems can be assembled and if robotic excavators are trapped or damaged, they can be repaired and replaced in the appropriate position [6], [22]. In the new scenario, the all-up option is thus preferred. Thanks to the ISRU options, a reduced number of astronauts, aerocapture for all vehicles, the choice of unpressurized rovers and other adequate choices, the proposed scenario is simpler than the NASA reference architecture and minimizes IMLEO even though no nuclear thermal propulsion system is used. Nevertheless, the savings may not

be sufficient to reduce the overall risk of the mission because a huge payload still has to be sent to Mars. In the previous study, it was suggested to perform an assembly in low Earth orbit (LEO) and to land two heavy vehicles on Mars. We now believe that more interesting options exist. The idea is to land four small vehicles instead of two. As it will soon be explained, this choice avoids LEO assembly and facilitates the entry descent and landing (EDL) phase. The next section is specifically dedicated to EDL difficulties. In sections 3 to 9, the consequences for the other parts of the mission are examined and discussed.

2. Reducing EDL risks using small vehicles

In the last NASA reference architecture for human missions to Mars, the loss of mission risks have been examined [12]. One of the main risk drivers is the reliability of the EDL systems with a failure risk close to 20% and high uncertainties concerning the feasibility. Thanks to numerous robotic missions, there is a good level of experience regarding landing small vehicles on Mars but most technologies used for these missions are not adequate for human missions, which require much heavier landing vehicles [1], [9]. The problem of landing heavy vehicles has been addressed by several authors [3], [8], [9], [18]. The objective is to reduce the velocity by atmospheric braking and then to land by means of a propulsive phase. The key parameters are the ballistic coefficient, see equation (1), and the lift to drag ratio (L/D).

$$\beta = \frac{M}{CS} \tag{1}$$

The ballistic coefficient is proportional to the mass of the landing vehicle and inversely proportional to the reference surface that faces the flux. Since mass is roughly proportional to volume, the ballistic coefficient tends to grow rapidly according to the mass of the vehicle. As a consequence, if the lift is negligible, light vehicles brake efficiently in the upper layers of the Martian atmosphere while heavy ones brake very late and keep falling with a high velocity. In order to help slow down the vehicle, parachutes can be deployed. However, their use is constrained by their size and specific pressure limits. According to Braun and Manning, upgraded parachutes could perhaps be deployed at Mach 3 but not at higher velocities [3]. In addition, the deployment of very large parachutes requires several seconds, while the schedule of the following EDL maneuvers is very tight. The propulsive descent can also start early but at the expense of huge amounts of propellant, which is not desirable. As has already been done with the space shuttle, some lift can be added to the vehicle to help reduce the vertical speed. The problem is illustrated Figure 1, which is a synthetic view of different graphs found in Braun and Manning's paper [3].

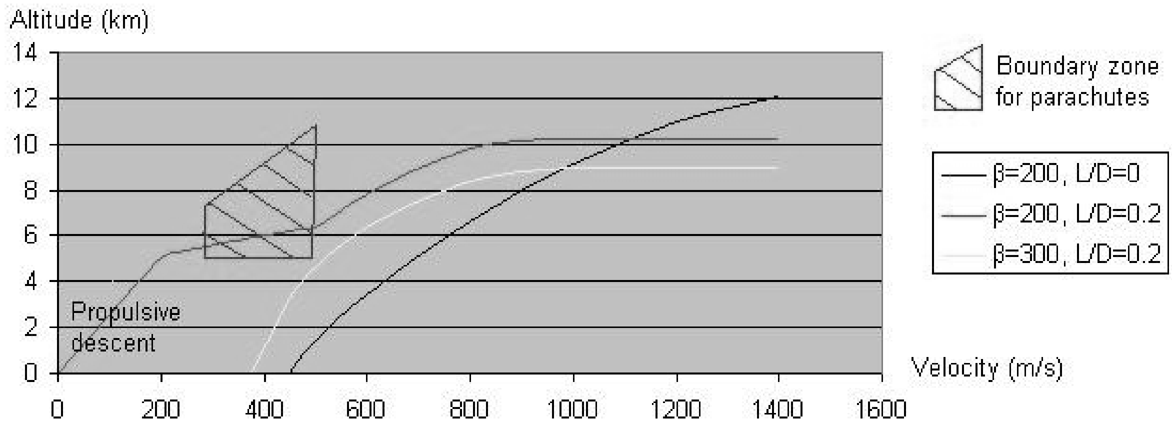

Figure 1: Descent configurations according to different ballistic coefficients and L/D values.

If the ballistic coefficient is above 200 kg/m^2 with no lift, the vehicle falls to the surface without time for parachute deployment. If the lift to drag ratio is 0.2 with a ballistic coefficient greater than 300 kg/m^2, there is still no time left for parachute deployment before the surface is reached. More lift can be obtained with complex shapes and orientations but at the expense of the complexity of the EDL maneuvers. A strong lift must be controlled with high accuracy to follow the right trajectory and the vehicle has to be reoriented for parachute deployment to avoid swinging movements. In addition, if the lift to drag ratio is high and no parachute is used, the drag direction may be very different from the thrust direction of the main engines, thus complicating the start of the propulsive descent. Last but not least, the heat shield has to be ejected and pushed away before the engines are fired. The problem is that unless very large parachutes are used, it is pressed against the bottom of the vehicle. Similar remarks can be found in several papers [3], [8]. If the risks of the EDL phase have to be minimized, a small lift is probably preferable, so that the descent is mostly ballistic and most of the previous problems are avoided.

Table 1: Qualitative assessment of different EDL options.

Option	**Mass**	**Ball. coef.**	**L/D**	**Attitude control**	**Trajectory control**	**EDL procedure**	**Constraints on mission architecture**	**Cost**	**Risks**
All propulsive descent	very heavy	high	low	simple	simple	ED: simple L: ?	high: unpractical	very high	high
NASA 5.0 Cigar shaped	heavy	high	high	complex	complex	very complex	low	high	high
Inflatable heat shield	heavy	low	low	simple	TBD	complex	landing accuracy?	med.	high
biconic shape 2-4-2 scenario	light: < 32t	low	low	simple	simple	simple	other vehicles needed	low	low

The best shapes for such low EDL risks are the capsules with 70° sphere-cone heat shields that have been used in all robotic Martian missions to date [3]. In fact, with the exception of the shuttles, all Earth re-entry capsules also share the same shape: Apollo, Soyouz, Dragon, Orion, etc. Let us assume that the maximum acceptable L/D value is around 0.2 (typical 70° cone) and that the use of parachutes is a good solution for the next maneuver of the descent, because it is a light and robust system. The maximum value for the ballistic coefficient is therefore close to 200 kg/m^2, otherwise there is no time left to open the parachutes and land safely on Mars. With these constraints in mind, it is interesting to look at the mass of the landing vehicles and to check the feasibility of using that shape for Mars landers. In most scenarios proposed for human missions to Mars, the mass of the vehicle entering the Martian atmosphere is in the range of 70 to 100 metric tons. Assuming a 70° cone shape, L/D = 0.2, β=200 kg/m^2 and M=70 tons, the diameter of the vehicle would be 17.8 meters, which is unrealistic. Inflatable heat shields might be an interesting solution [18]. However, as already pointed out by Braun and Manning, the control of the descent might be more difficult, another critical system is added and the risks would therefore be increased [3]. A synthetic view of our qualitative assessment is presented in Table 1.

If it is desired to minimize the risks of the descent and to avoid the use of inflatable systems, the last option is to choose a heat shield with the maximum acceptable diameter and to reduce the mass of the vehicle to comply with the ballistic coefficient constraint. The diameter of the vehicle depends on many parameters. Let us assume that the vehicle is directly placed on top of a launcher. The diameter of the Space Launch System, which is currently developed by NASA, is only 8.4 meters. The diameter of the Saturn V, the largest rocket ever built, was 10 meters. It is difficult to build larger launchers because of transportation and aerodynamic

constraints. If an effort is nevertheless required, it could perhaps be increased to 12 meters. Let us assume that this is the largest acceptable diameter. With the constraints previously cited, the maximum mass for the vehicle can be calculated. It is equal to 32 tons. This very simple calculation does not take into account the maximum deceleration constraints and other parameters such as the limit on the size of the parachutes. In a first approximation, it is nevertheless in agreement with the values found in other sizing studies [8]. It is also worth noting that according to Braun and Manning, a 30 m Mach 3 parachute allows for subsonic propulsive deceleration at an altitude of 2 km if the entry masses are below 33 tons [3].
32 or 33 tons is significantly lower than the 100 tons proposed in the NASA scenario. As already been pointed out by Christian *et al.*, a six-person crew would require a habitable volume and mass of consumables that are not compatible with a capsule shape [5], [8], [10]. In a recent study, we proposed a two-person crew and the landing of lighter vehicles but the mass was still around 70 tons, which is still much more than the 32 ton limit [21]. However, it is close to twice the maximum. An interesting idea is therefore to imagine a similar scenario, in which the landing vehicle is split into two smaller ones. This is not difficult. The payload can be divided in a habitable module with most consumables and a cargo module with all ISRU systems, power systems included, and contingency consumables. Three important points must be discussed:

- First, a landing with very light vehicles reduces the risks of creating an unstable crater below the lander, which has been identified as a major problem in the NASA scenario. In addition, if there are four engines below the lander as it is suggested by Christian *et al*, that effect is also mitigated by the dispersion of the plume around the landing zone due to the large diameter of the propulsion system [8].
- Second, this configuration also reduces the mass of each vehicle for LEO and avoids the need for LEO assembly.
- Finally, third, one may argue that since two vehicles are landed instead of one, there is a greater risk of losing one vehicle. This is true. However, the risk of losing the crew is decreased because the number of manned vehicles is the same and the EDL sequence is assumed to be safer. It is worth noting that the overall risk of mission loss is not strongly impacted since the 2-4-2 scenario is entirely duplicated [21].

Reducing the size and mass of the landing vehicles is therefore a key idea to reduce EDL risks and to eliminate the assembly phase. The inconvenience is to multiply the number of landings and to impose that both the cargo and the habitat land in the same zone. Pin-point landing might be an issue. However, according to the specifications of the Mars Surface Laboratory mission, uncertainties on pin-point landing are relatively small, in the order of a few kilometers [25], [26]. Nevertheless, if a cargo lander lands far from the habitat lander, several strategies exist to solve the problem (see next sections). The trade-off seems therefore favorable to small vehicles. In a recent study, it was stated in the conclusion "Future work should look into incorporating a detailed EDL design into existing Mars design reference missions to assess the system level impacts" [8]. This is the starting point for the revisions proposed in the next sections of the present paper.

3. Different options

As explained in Section 2, it is assumed that two types of vehicles are sent to Mars: a habitat lander and a cargo lander. Since the mission is entirely duplicated, there are four landing vehicles. In this new scenario, two main options must be discussed:

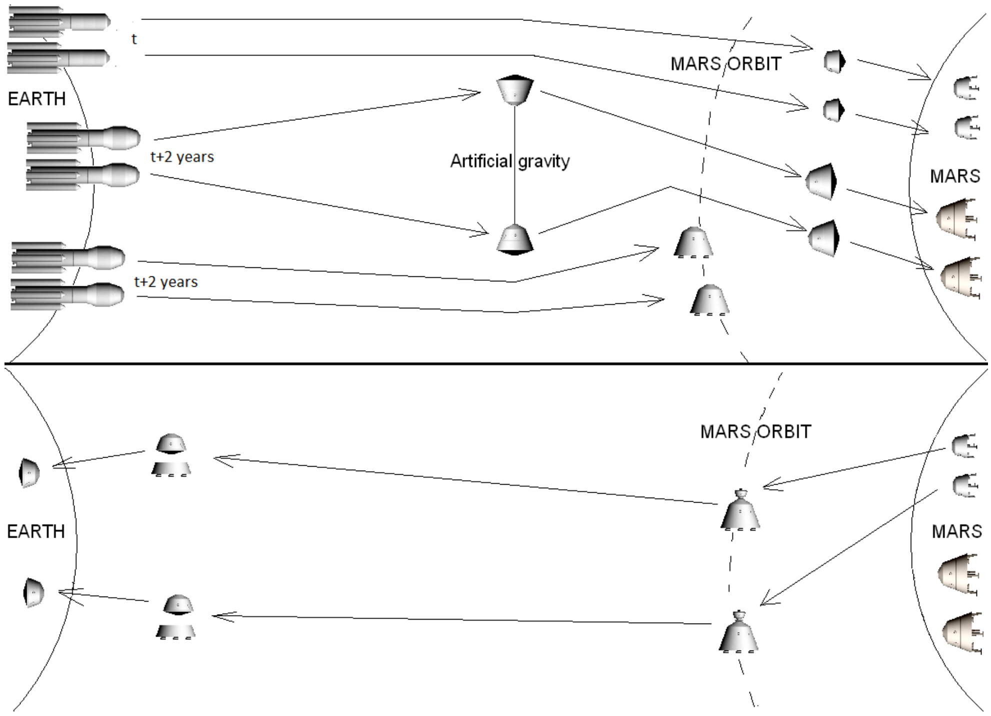

Figure 2: Standard option. There are six launches. Two cargo vehicles and two habitat vehicles are sent to the surface of Mars while two ERV are sent to Mars orbit. For the return, small MAV are launched from Mars to join the ERV, which finally bring the astronauts back to Earth.

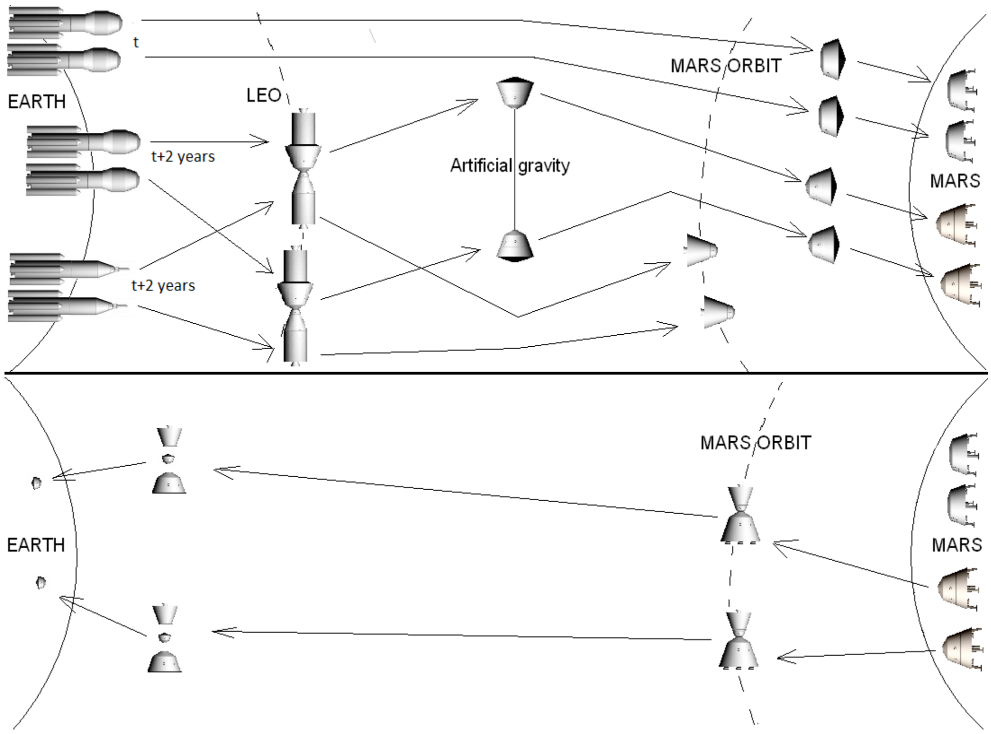

Figure 3: Single habitat option. The ERV sent to Mars orbit are smaller. Also, the habitat vehicles landed on Mars are the two Mars ascent vehicles. The Earth re-entry vehicles are small capsules.

- As in other Mars semi-direct scenarios, a light Mars ascent vehicle can be sent to Mars onboard the cargo lander.
- As suggested in our previous proposal, the habitat lander can also play the role of the Mars ascent vehicle [21].

The first option is called the "standard option" by reference to other semi-direct concepts and the second is called the "single habitat option". An overview of the scenarios is presented in Figures 2 and 3. Both options are further examined and discussed in this paper. For easy reading, the next sections of the paper are structured according to the chronology of the mission. In Section 4, the different types of vehicles are presented and the main features of the launcher are given. In Section 5, the transfer between the Earth and Mars is addressed. Aerocapture is then discussed in Section 6. The deployment of ISRU and power systems, the production of propellant and the use of rovers on the surface of Mars are discussed in Section 7. Back-up strategies and risks assessment are presented in Section 8.

4. Vehicles

4.1. Earth Return Vehicle

In both options, three types of vehicles are sent to Mars. The first one is called the Earth Return Vehicle (ERV). By analogy with the Apollo program, the ERV is similar to the service module and the command module that were waiting in lunar orbit while the lunar module was on the surface. The service module is a wet propulsion stage that is used at the end of the mission for trans-Earth injection (TEI) from Mars orbit. The command module is a habitable module with the capability of re-entering the Earth's atmosphere. The ERV could resemble an Orion capsule and its service module, which are currently developed by NASA. However, the NASA vehicle is not appropriate. In order to optimize the scenario, an aerocapture is assumed. The vehicle has to enter the Martian atmosphere before Mars orbit insertion and Orion specifications do not meet the requirements (the service module has no heat shield). As in the last NASA reference mission, we assume that the ERV is waiting in a high elliptical Martian orbit [12]. The ΔV budget for trans-Earth injection (TEI) is close to 1.5 km/s. The return habitat must sustain a crew of four during eight months (two in a nominal mode, but four if necessary). In addition, it should be able to wait in a sleeping mode during almost two years. Depending on the option, the habitable module waiting in Mars orbit is either small or relatively big.

- Standard option: Since the Mars ascent vehicle is very light, the habitable module of the ERV is the main habitat for the return leg of the mission. Moreover, it should be able to re-enter the Earth's atmosphere at the end of the journey. The command module is therefore a heavy capsule with accommodations, enough consumables and spare parts for eight months.
- Single habitat option: The habitat lander is also the Mars ascent vehicle. Since the main habitat comes from the surface of Mars, the habitable module waiting in Mars orbit can be a small capsule. In a recent proposal, it was suggested to store the consumables for the return in that capsule [22]. However, many other objects and systems can be stored in the same way. If it is imperative to reduce the mass of the vehicle for the launch from Mars, it is possible to leave some elements of the power systems, some personal affairs, spare parts and perhaps some elements of the life support system on the surface. Inevitably, if these objects or systems are needed for the return, they have to be duplicated and stored in the capsule before the capsule is sent to Mars. The single habitat option can therefore be split into two other options.
  A: Light capsule: Only consumables for the inbound trip are stored in the capsule.

B: Heavy capsule: Many elements are left on the surface but the same elements are duplicated in the capsule.

The advantage of the standard option is to minimize the mass landed on Mars, while the advantage of the single habitat option is to minimize the mass of the vehicle waiting in Mars orbit. It should be noted that if it is required that the astronauts must be sent to LEO in a capsule with an escape system on top of the launcher, the standard option requires another launch because the habitat is too heavy to play that role. In the case of the single habitat option, however, it is possible to use the capsule of the ERV and to proceed to transshipment onto the main habitat after rendezvous in LEO, as illustrated in Figure 4.

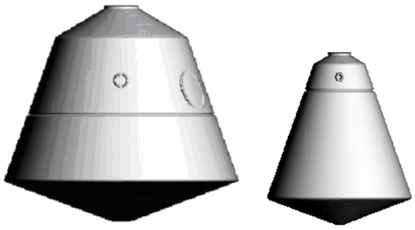

Figure 4: Illustration of the vehicle sent to Mars orbit. For the standard option, the vehicle is on the left and for the single habitat option, it is on the right. The separation between the command module and the service module is visible. The heat shield at the bottom protects the vehicle during Mars aerocapture. The heat shield of the command module for Earth atmosphere re-entry is hidden behind the service module.

Table 2: Mass of the vehicle that has to be sent to Mars orbit to prepare the return.

		Standard	Single hab A	Single hab B
Command module	Command module dry mass	8400	4200	4200
	Consumables for the return	3100	3100	3100
	Other consumables, accommodations and systems	0	0	3000
Subtotal command module		**11500**	**7300**	**9300**
Service module	Propulsion system, engines and tanks (kg)	3000	2900	2900
	Propellant (kg)	25000	24000	24500
	Heat shield for aerocapture (kg)	4000	2000	3000
Subtotal service module		**32000**	**29000**	**31500**
Subtotal (kg)		**44500**	**37300**	**40800**
Margin 20%		**8900**	**7460**	**8160**
Total for TMI, rounded (kg)		**53000**	**41000**	**49000**
Transfer in LEO		0	-3100	-6100
Launch escape system		0	1000	1000
Total at launch from Earth (kg)		**53000**	**39000**	**45000**

In order to make an estimation of the mass of the vehicle, we propose to look at the specifications of the Dragon capsule developed by Space X. The trunk module and the propulsion system are obviously not appropriate for our purpose. They should be replaced by a large service module with enough propellant for TEI. The dry mass of the capsule is only 4.2 tons. For the standard option, the volume of the capsule does not meet the standards and both the power systems and the life support should be enhanced to support the lives of the astronauts during eight months [13]. It is difficult to make an accurate estimation of the supplementary mass. In a previous paper, our estimation, which was based on values found in a NASA report, was 12.7 tons all consumables included [21, table 2]. Let us make a simplification and assume that the dry mass of the upgraded command module is simply doubled (without consumables). The mass of consumables is 3100 kg for two astronauts, taking into account a possible transshipment [21]. For the single habitat B option, it is assumed that three tons of life support or power systems are left on Mars. As a consequence, three supplementary tons are added in the capsule. For the service module, the estimation is based on the mass of propellant required for TEI (24 tons using CH_4/O_2 with an ISP equal to 370s [4]) and on a structural mass to propellant mass ratio equal to 0.12. A small amount of propellant is also taken into account for the post-aerocapture maneuver. For the structure, it is assumed here that the shape of the vehicle is optimized to enable aerocapture. The mass of the heat shield for aerocapture has been roughly estimated. It is assumed that the mass of the heat shield for the lightest ERV is half the mass of the heat shield for the heaviest one. An illustration of the vehicle is presented in Figure 4. See Table 2 for a detailed estimation of the mass of the vehicle that must be sent to Mars orbit. The mass of the Mars ascent vehicle is discussed in the next sections.

4.2. Cargo lander

Preliminary remark: The simplest solution would be to send all the systems in the same landing vehicle and to avoid the use of a cargo lander, as suggested in our previous study, but the mass of the lander might be very high and at the expense of EDL risks (see section 2). The cargo lander and the habitat lander are therefore distinct vehicles.

The second vehicle of our proposal is thus the cargo lander. In the standard option, it is used to land a small and dry Mars ascent vehicle (MAV), a chemical unit and power systems on the surface of Mars. The objective is to produce propellant for the MAV using in situ resources. In the single habitat option, the cargo lander is used to send only a chemical unit and power systems to the surface of Mars. In that configuration, there is no need for a MAV since the ascent vehicle is also the habitat lander, which is presented in the next section.

A dedicated Mars ascent vehicle can be very light because there is no need for a heat shield during the ascent and the life support systems can be much simplified and operational only for the launch. The configuration can be very similar to the lunar module of the Apollo program. Assuming a capability of four astronauts and two days of life support, the mass of the habitable module can probably be around two tons and the mass of the dry propulsion stage can also be around two tons. An illustration of both options is proposed in Figure 5 and a comparison is provided in Table 3. As has been done in a previous work (see [21] Tables 4 and 5), the mass of the ISRU systems has been estimated according to an extrapolation of the numbers proposed in the last NASA report describing the reference architecture and the mass of power systems has been extrapolated from another study on flexible solar panels for the Martian surface [6]. For the dedicated MAV, 15 tons of propellant have to be produced, while for the single habitat option A and B, 61 tons and 41 tons have to be produced, respectively (see explanations in Section 4.3). For the EDL systems, the following parameters have been taken into account:

- Cross range maneuver: ΔV = 265 m/s ([8] page 5).
- Propellant mass fraction: 15% ([3] page 15: 12-18%).
- Heat shield (dual use) mass: 12% of total mass ([8] page 8).
- Mass of four engines, total thrust 1MN: 1500 kg ([8], equation 3).
- Reaction control systems, parachutes, tanks and structure: 10% of total mass.
- Margin: 20%.

Table 3: Mass estimations for the cargo lander.

		Standard	Single hab A	Single hab B
Payload	Mars ascent vehicle, structure and LSS (kg)	2000	0	0
	Mars ascent vehicle propulsion system (kg)	2000	0	0
	Excavation systems (kg)	750	3000	2000
	Water extraction systems (kg)	270	1100	730
	Sabatier reactor and electrolysis unit (kg)	700	2810	1870
	Power systems (kg)	960	3850	2570
	Structure and packaging (kg)	1000	1000	1000
	Backup consumables (kg)	3100	3100	3100
	Science equipment (kg)	600	600	600
	Total payload, margin included (kg)	**11380**	**15460**	**11870**
EDL systems	Propulsion system, engines (kg)	1500	1500	1500
	Propellant (kg)	345	4600	3600
	Structure, tanks and other systems (kg)	2300	3100	2400
	Heat shield (kg)	2760	3700	2800
	Subtotal (kg)	**10010**	**12900**	**10300**
	Margin 20% (kg)	2000	2580	2060
	Total EDL (kg)	**12000**	**15480**	**12360**
Total, rounded (kg)		**23000**	**31000**	**24000**

The mass of the cargo lander for the single habitat option strongly depends on the choice made for the return. In a first order approximation, in terms of the landed mass, it seems equivalent to use a light dedicated MAV or to produce much more propellant but to reuse the habitat lander for the ascent back to Mars orbit (option Single hab B).

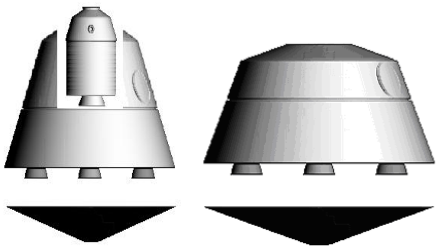

Figure 5: Illustration of the cargo landers. On the left, the payload of the standard option is mainly a light Mars ascent vehicle (dry). On the right for the single habitat option, there are only the ISRU and power systems but they are heavier in order to comply with the constraint of producing more propellant. In both cases, the shape is optimized for aerocapture and then entry descent and landing with a large heat shield located below the lander.

4.3. Habitat lander

The third vehicle of the scenario is called the habitat lander because it is used to send the astronauts and their habitat to the surface of Mars. In both options, the vehicle is first inserted into Mars orbit using an aerocapture maneuver, which eliminates the need for a large amount of propellant. Then the vehicle enters the Martian atmosphere and lands on Mars. In the standard option, the habitat is left on the surface of Mars at the end of the mission. In the single habitat option, after refueling using ISRU systems, the same vehicle is used to send the habitat back to Mars orbit. It is assumed here that it is possible to use the same engines for the landing and ascent. A similar hypothesis was made in the 1998 NASA study of the Design Reference Mission [11]. However, in order to bring the vehicle back to Mars orbit, the required amount of propellant is much higher for the ascent than for the landing. Therefore, there is an impact on the mass of the tanks and the size of the vehicle. The required amount of propellant (CH_4/O_2) has already been calculated [21]: for the single habitat A option, it is 61 tons and for the single habitat B option, using the same method, it is approximately 40 tons. In order to calculate the additional mass for the tanks and structure, the structural mass to propellant mass ration is set to 12%.

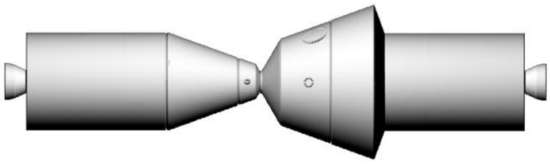

Figure 6: Junction in LEO between the ERV and the habitat lander. The astronauts are transferred to the main habitat while consumables for the return are transferred to the capsule. The TMI propulsion stage is still attached to the vehicles. At the end of the stay on Mars, the habitat lander will have to come back to Mars orbit and connect with the ERV in a similar way (without TMI propulsion stages).

The advantage of the single habitat option is that a small capsule can be waiting in Mars orbit instead of a full habitat. Moreover, that capsule can be used at the beginning of the mission to store the consumables for the return leg of the mission. As illustrated in Figure 3, there should be a rendezvous between the ERV and the habitat lander in LEO to transfer the astronauts from the capsule to the main habitat. At the same time, the consumables needed for the inbound trip can be transferred from the habitat to the capsule. Since the ERV is going to wait in Mars orbit, it is indeed the simplest solution to avoid landing these consumables on Mars and make them available for the return.

Table 4: Mass of the habitat lander at Mars entry.

	Subsystem	Mass estimation (kg)
Habitat	Life Support System (kg)	3500
	EVA equipment (kg)	200
	Comm/info management (kg)	320
	Power prod. 30 kWe P.V.A. (kg)	1200
	Thermal control system (kg)	400
	Structure (kg)	2000
	Consumables for Mars surface (kg) (3100 kg also in cargo)	3100
	Small rovers (x2) (kg)	380
	EVA consumables (kg)	400
	Subtotal for Mars surface (kg)	**11500**
	Margin 20%	2300
	Total for Mars surface (kg)	**13800**
EDL systems	Propulsion system, engines (kg)	1500
	Propellant, 15% of entry mass (kg)	4200
	Structure, tanks and other systems, 10% (kg)	2800
	Heat shield, 12% of entry mass (kg)	3400
	Subtotal (kg)	**11900**
	Margin 20% (kg)	2380
	Total EDL standard option (kg)	**14280**
Total, rounded (kg)		**28000**

It is also possible to store them in the capsule at the beginning of the mission but it might be preferable to avoid this in order to minimize the mass of the capsule for the launch and enable an emergency procedure with a light launch escape system located on top of the launcher. Once the transfer is done, the two vehicles are sent to Mars separately. This idea is illustrated in Figure 6. An estimation of the mass at Mars entry is presented in Table 4 and the total mass of the vehicle for the launch is given in Table 5.

Table 5: Mass of the habitat lander at departure from Earth.

Option		Standard option	Single hab A	Single hab B
Mars surface	Payload for Mars surface	13800	13800	13800
	EDL systems	14280	14280	14280
	Additional tanks and systems for ascent	0	3000	1700
	Total Mars entry	**28000**	**31000**	**29700**
LEO	Consumables for outbound	3100	3100	3100
	Total LEO	**28000**	**34100**	**32800**
Launch	Transfer in LEO	**0**	**3100**	**6100**
	Total, rounded (kg)	**28000**	**37000**	**39000**

4.4. TMI stage and launcher

In this scenario, the idea is to perform a rendezvous on the surface of Mars and to avoid LEO assembly. In order to send the six vehicles (two ERV, two cargo landers and two habitat landers) from LEO to Mars, a TMI stage has to be used. Depending on the option and the vehicle, the payload for LEO is different. Let us make some simplifications and assume the following parameters:

- ΔV = 3.6 km/s from LEO to Mars
- ISP = 450s for the propulsion system
- M_u is the payload mass (useful) and M_p is the mass of propellant
- r, the structural mass to propellant mass ratio of the propulsion system, is set to 12%
- Two stages for the TMI propulsion systems (1.8 km/s for each stage)

Tsiolkovski's equation allows us to estimate, in a first order approximation, the mass of the propellant for TMI according to the payload mass:

$$\Delta V = ISP \cdot g \cdot \ln\left(\frac{M_u + (1+r)M_p}{M_u + rM_p}\right) \qquad (1)$$

Let $K = e^{\frac{\Delta V}{ISP \cdot g}}$, we get: $$M_p = \frac{K-1}{(1+r) - rK} M_u \qquad (2)$$

Table 6: Initial Mass in LEO. Values are given for one ERV, one cargo lander and one habitat lander without taking into consideration the fairing of the launchers. For the total IMLEO of the scenario, the values have to be doubled.

		Standard option	Single hab A	Single hab B
ERV	Payload	53000	39000	45000
	1st stage	50900	37500	43200
	2nd stage	31800	23400	27000
	Total (kg)	**136000**	**100000**	**115000**
Cargo lander	Payload	23000	31000	24000
	1st stage	22100	29800	23000
	2nd stage	13800	18600	14400
	Total (kg)	**59000**	**79000**	**60000**
Habitat lander	Payload	28000	37000	39000
	1st stage	26900	35500	37400
	2nd stage	16800	22200	23400
	Total (kg)	**72000**	**95000**	**100000**
IMLEO total		**264 tons**	**274 tons**	**269 tons**

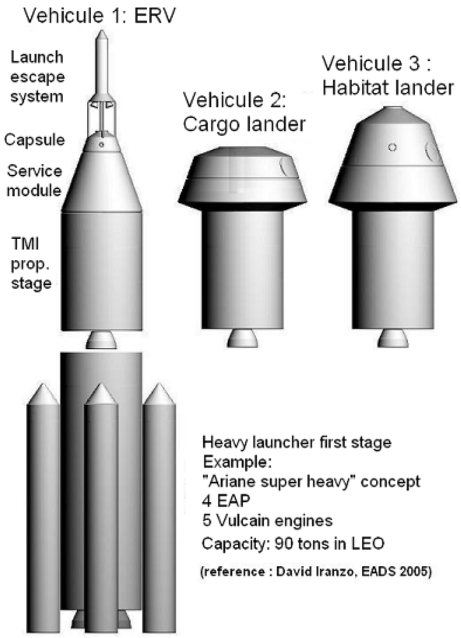

Figure 7: Heavy launcher, TMI propulsion stage and three types of vehicles.

An estimation of the payloads for the launch from Earth is provided in Table 6. In a first order approximation, all options perform as well as the others although some additional work is needed to confirm the results. An important issue is the possibility of using the same type of launcher without having to perform a LEO assembly. Considering that criterion, the standard option is penalized because there are important differences between the different vehicles and sending the ERV to LEO requires the use of a huge launcher with LEO capability above the one of the SLS launcher currently developed by NASA. The best option would be the second, although it is less competitive in terms of mass. The total mass of the TMI vehicles, propulsion stage plus payload is between 79 to 100 tons. The fairing elements (about 10% of the payload) are jettisoned early during the ascent to LEO but they have to be taken into account in the specifications of the launcher. The Space Launch System currently developed by NASA would be sufficient for all launches of the single habitat option. An illustration of a heavy launcher and the three types of vehicles is presented in Figure 7.

4.5. Launching sequence

In the NASA reference mission, the assembly phase is so complex that it has been identified as a risk driver for the loss of mission. An important advantage of the proposed scenario is that there is no assembly required in LEO. Nevertheless, six launches have to be programmed for a single mission. In order to speed up the launching sequence, two launch pads might be required. Another interesting idea is to undertake an international collaboration and to split the mission according to the necessary duplication of the vehicles. Many systems would be similar but the launcher and other important systems could be developed separately.

Let us describe the launching sequence of the single habitat scenario. It is reasonable to assume that the crewed vehicle has to be launched at the end. Cargo landers might be launched first a few weeks before the other vehicles but a more interesting idea is to send them to Mars two years before as in other pre-deploy scenarios. The term "pre-deploy" is however inappropriate in this case. The equipment of the cargos would remain stored in the vehicle until the arrival of the crew. This option is very interesting because there would be insurance that the cargo is safely landed before sending the crew. Another advantage for the single habitat option is that the Mars ascent vehicle (the habitat lander) is not included in the cargo. As a consequence, it would not spend a lot of time waiting on the surface as is the case in the NASA reference mission [12]. Providing that the cargos have safely landed on Mars, two years later, during the next launch window, the habitat landers are sent to LEO and placed in a parking orbit. Approximately three weeks later, the ERVs are launched with the crews. In the single habitat option, the rendezvous maneuvers are immediately performed and the astronauts are transshipped into the habitat landers while the consumables for the return are transferred into the capsule. Then all vehicles are sent to Mars. A strong constraint of this scenario is that the two habitat landers have to stay close to one another. Therefore, the TMI engines of the two habitat landers have to be fired almost simultaneously. The ERVs should also be very close because they allow important backup strategies. For instance, if a problem occurs during transit, a free return trajectory can be implemented or the astronauts could stay in Mars orbit and use the consumables stored in the ERVs [29].

5. Transit between the Earth and Mars

The proximity constraint between the two habitat landers is an advantage of this scenario because it enables the rescue of a crew through a possible transshipment onto the safe vehicle during the outbound trip. In the NASA reference mission or in the Mars Direct scenario, there is no rescue solution during that transit phase. The risk of losing life support during that eight month period may be low but the Apollo 13 experience suggests that such a risk has to be seriously considered. The impact of allowing transshipment onto the mass of consumables has already been discussed [21]. The numbers proposed in the previous tables already take a possible transshipment of two other astronauts into the same vehicle into account. The guiding principle is that there are consumables for a crew of four in each habitable module but there are no additional consumables for contingency situations. In addition, it is also assumed that less habitable volume and less comfort are acceptable if an emergency situation occurs and leads to a transshipment procedure.
An important problem of the transit between the Earth and Mars is the physiological impact of a long stay in microgravity. In the previous paper, a small centrifuge was proposed [2]. In the new version of this scenario, another solution inspired from Mars Direct is suggested [29]. Since the two habitat landers have to stay close to each other, it is possible to connect them by means of a long cable and to make one rotate around the other to simulate artificial gravity. However, the efficiency of the concept still needs to be demonstrated. An illustration is proposed in Figure 8.

Figure 8: The two habitat landers during the transit phase.

6. Aerocapture

Aerocapture is an interesting option to reduce the mass of propellant needed for Mars orbit insertion. In the last NASA reference mission, that option is chosen for the cargo but not for the vehicle with the habitable module because its shape is very complex and its shielding would be difficult. Let us make an estimation of the mass savings if aerocapture is used. Without aerocapture, a propulsion system to reduce the velocity of the spacecraft arriving from Earth should exist. The required ΔV depends on many parameters such as the trajectory between the Earth and Mars and the targeted Martian orbit. For a high elliptic orbit, the ΔV budget is in the order of 1.5 km/s (NASA reference mission). For the sake of simplicity, let us assume that the mass of the vehicle is 50 tons (payload, tanks and engines included) and that the propulsion system is based on CH_4/O_2 with a specific impulse equal to 370 seconds. According to Tsiolkovsky's equation, the propellant mass for a 1.5 km/s ΔV is 27 tons. By comparison, the mass of the heat shield and other supplementary systems needed to reach that orbit would be less than 7 tons (the mass of the heat shield is about 12% of the total entry mass, according to Christian *et al*. [8]). Furthermore, in order to send 20 supplementary tons from LEO to Mars, around 30 tons of propellant is required in LEO. In addition, this amount must be multiplied by the number of vehicles. The mass savings are therefore considerable if an aerocapture maneuver is performed. In the NASA reference mission, the savings are not as large because the specific impulse of a nuclear propulsion system is much higher. However, the key point here is that aerocapture is one of the better solutions to reduce IMLEO. It should not be regarded as an option that could be chosen if and only if the shape of the vehicle is appropriate for that. It should be defined as one of the main assumptions of the scenario and some guiding principles must be established to determine the other parameters of the mission, such as the shape or the size of the vehicles. This is clearly another reason to support the choice of small vehicles with standard conic shapes, which are adapted to aerocapture maneuvers.

7. Surface exploration

7.1. Preparing for departure

As suggested in the last NASA reference mission, a conjunction-class mission is preferred for the revised version of our scenario. There are two important reasons for this. First, there is plenty of time left for the exploration of Mars. Second, when the astronauts arrive on the surface of Mars, the Mars ascent vehicle is not ready for launch. The astronauts have to deploy the solar panels and the ISRU systems and start the refueling of their vehicle. This process may last 300 days. In the NASA reference missions, the ISRU systems are pre-deployed and the MAV is ready for launch before the crew is sent to Mars. In the proposed scenario, it is possible to use the same strategy if the standard option is chosen. In the single habitat option, since the MAV arrives with the crew, this strategy is not possible. Irrespective of the option, we suggest that the best idea would be to wait for the arrival of the crew. The main advantage of human presence is to optimize the exploitation of local resources for refueling the ascent vehicle. Thanks to their presence on the surface, huge fields of ultra light solar panels can be easily deployed and the work of excavating robots driving on irregular terrains can easily be controlled [6]. As already noted in previous studies, such technologies minimize the payload needed on the surface of Mars [6], [7], [24]. No H_2 or CH_4 stocks are needed and a complex nuclear power plant can also be avoided. In comparison with the NASA scenario, the mass savings are in the order of 10 to 20 tons. In addition, the ISRU systems can provide EVA and backup water and oxygen to the astronauts. They are only backup because we have to consider the case of a possible problem in Mars orbit and no landing on the surface. Also, if the payload is minimized, as has already been explained in Section 2, EDL is easier and safer. The inconvenience of this strategy is the risk that the ISRU systems do not produce enough propellant for the ascent. However, the implication is simply that ISRU systems have to be tested and qualified very strictly and that the duplication of the mission is clearly justified.

7.2. Surface mobility

In the previous paper, there was no need to reach another lander. In the revised version of the scenario, it is mandatory for the crew to reach the cargo and to bring the ISRU and power systems back to their landing site. In our proposal, the rovers are not pressurized. The justification of this choice has already been presented [21]. However, if the rovers are not pressurized, the distance range is reduced and there is a risk for the crew that the cargo is too far away. Such a risk is low because the control of the lift during hypersonic aerobraking and the propulsion systems of the landing vehicles allow accurate landings. Mars Science Laboratory, for instance, is equipped with the same guided entry mechanism that was used in the Apollo program and the error is reduced to 20 kilometers maximum with 95% certainty [26]. Second, it might be possible to transport and deploy a light inflatable structure a few kilometers away from the habitat in the direction of the cargo. In a first approximation, a portable habitat and two rovers are functionally equivalent to a pressurized rover. As already been suggested in a NASA report, some life support can directly be provided by specific devices placed onboard an unpressurized rover [15]. This option could significantly expand surface mobility. It would be less efficient in terms of ergonomics and efficiency, but much lighter, smaller and easier to store in the lander and easier to deploy on the surface. For the first manned missions to Mars, safety issues are more important than efficiency issues. Therefore, since EDL risks and other important risks are tightly linked to the total mass that is sent to the surface of Mars, the guiding principle for the choice between two options should primarily be the minimization of this mass.

8. Back-up options and scenario assessment

Our preliminary results show that it is possible to reduce the total mass of the landers at Mars entry. The greatest mass is obtained with the habitat lander of the single habitat A option with 31 tons, which is lower than the proposed 32 tons limit. Therefore, it seems possible to use a standard capsule shape and large parachutes for the descent and landing on Mars. It is important to remark that such a strategy is not possible in the NASA scenario, even with four astronauts instead of six, because of the constraints on mass and volume. Two astronauts is probably a minimum for a crewed vehicle but it is a maximum for a habitat lander that is supposed to provide life support during a long period of time. Also, it is worth noting that the mass of the cargo lander plus the mass of the habitat lander is less than the mass of the heavy vehicle that was proposed in the previous scenario. This is due to the simplification of the EDL systems (70° cone heat shields and parachutes), which are much lighter than the ones proposed in the NASA reference mission.

Different backup options exist. Let us consider the different phases of the mission:

- First of all, the two cargos are sent to Mars and land on the surface two years prior to the departure of the crewed vehicles. If a cargo mission fails, another cargo can be sent to Mars in the next launch window and the launch of the crewed vehicles can be delayed until two years later. In comparison with the NASA reference mission, the advantage of this scenario is that there is no deployment of the ISRU systems or storage of the propellant before the arrival of the crew. The payload delivered on the surface of Mars is not critical and the maintenance of the systems during an extended period of time is therefore easier.
- Once it is assumed that the cargos have safely landed, the two ERVs and the two habitat landers are sent to LEO. If a problem occurs during this phase, the mission is aborted and the astronauts return to Earth using the capsules attached to the ERVs. If a capsule is out of order, there is a connection between the vehicles and the four astronauts come back with the safe capsule.
- After trans-Mars injection, the two habitat landers perform a rendezvous and they are attached to one another with a cable to provide a kind of artificial gravity. If a vehicle becomes unsafe during the outbound trip, transshipment is performed onto the safe vehicle and the mission is aborted. Several options are then possible. The simplest one is to reach Mars, to perform an aerocapture maneuver, to wait in Mars orbit until the

Earth return window opens, to connect with the ERV and finally to come back to the Earth. Another option is to continue the mission and to land on Mars. This option may seem risky because there is no backup habitat on Mars. However, there are two cargos waiting on Mars. If there is a special need in terms of life support or energy, this option might be preferable.

- One of the two crewed vehicles may not land safely. For instance, the vertical velocity could be too high or the terrain could be unstable with the result that the vehicle could be damaged and the habitat could no longer be pressurized. Due to the full duplication of the mission and provided that the astronauts are still alive, there are backup options. If the two landers are not too far from each another (and they should not be), the astronauts of the damaged vehicle can simply switch to the safe one. Eventually, the astronauts of the safe vehicle may have to quickly deploy the rovers on the surface in order to undertake a rescue mission. These backup options do not exist in the NASA reference mission.
- On the surface of Mars, if the life support of a habitat is not working properly, it should be possible to use a backup system. If the problem regards the composition or the healthiness of the atmosphere and the limits are reached, it is suggested in the NASA reference missions to remove it and to replace it with a backup atmosphere. This procedure is exceptional and can be performed only once. If the problem regards the water quality, a similar procedure is possible. However, this extreme option is not urgent in our scenario because the astronauts can live in the second habitat while they try to solve the problem. The second habitat plays the role of a safe haven that provides time to the astronauts to understand the failure and find a solution. If no solution is found and if the backup systems have already been used, the last option is to transfer the crew in the safe habitat.

Table 7: Qualitative risks comparison

	NASA DRA 5.0	**2-4-2 scenario**
Launches	8 to 10	6
LEO assembly complexity	high	no assembly
Outbound trip, life support system failure	no backup	transshipment
New propulsion system	yes: nuclear propulsion	no: all chemical
EDL	heavy landers, complex descent maneuvers	small landers, conic shapes, simple descent maneuvers
Landing damages	no rescue	rescue possible
Propellant production failure	mission aborted	backup options: pooling or solved with next mission
Launch from Mars system failure	no backup	transshipment
ERV system failure for TEI	no backup	transshipment onto 2nd ERV
ERV life support failure	no backup	transshipment

- The ISRU systems have to be tested and qualified for the mission but we have to consider a possible failure. Theoretically, they have to be deployed as soon as the astronauts are on the surface so that the vehicles are refueled after a period of 300 days (200 days before take-off). If there is a problem with the performance of the production on both sites, it is possible to pool everything and to concentrate on the production of propellant for one vehicle. If the problem is more serious, for instance if the amount of propellant is far from the requirements, backup options still exist. The first idea is to determine a technical solution and to send tools and additional life support to the astronauts so that they can live two more years on the surface and solve the problem. Another extreme solution might exist. If a significant amount of propellant has been produced, it is possible to reduce the mass of the vehicle to enable the launch. This strategy may be realistic under strict conditions that have to be examined before the mission is undertaken.
- For the launch from Mars, if the two vehicles have been successfully refueled and seem to be still functional, we have already suggested that the countdown must be done simultaneously for both [20]. If a system fails during the launching sequence, the launches should be aborted so that all astronauts can be transshipped onto the safe vehicle.
- Back in Mars orbit, both ascent vehicles perform a rendezvous with their own ERV. If there is a problem with an ERV, both have to join the one that is safe and works properly. In order to achieve that goal, the astronauts of the first MAV have to be transferred into the ERV. Then, their habitat has to be detached to enable a connection with the second vehicle. One habitat would be lost.
- On the way back, if there are two ERVs and one becomes unsafe, transshipment has to be undertaken.

Because of the numerous backup options, the proposed scenario is very robust. We do not have the methodological tools to assess the risks in great detail. However, a qualitative comparison with NASA reference mission is presented in Table 7.

The IMLEO of the single habitat A option is 548 tons, while it is estimated at approximately 900 tons in the NASA reference mission for the chemical propulsion case study [12, table 5-16]. How to explain the mass savings?

- First of all, as already explained in Section 6, aerocapture is very important to reduce the mass of propellant needed for MOI. In the NASA scenario, the shape of the crewed vehicle is very complex and there is no aerocapture. The loss is between 50 and 100 tons.
- Second, there are six astronauts instead of four. It is difficult to determine the impact on the mass of all systems. There is a direct impact on accommodations and consumables but there also are impacts on the volume of the habitat, the mass of life support and power systems, the size and mass of the Mars ascent vehicle, the mass of ISRU systems, the mass of heat shields, etc. However, there is also a mass penalty in the proposed scenario because in each habitat lander, there should be enough consumables for four astronauts. On the surface of Mars, if we consider the habitable modules (without the rovers and ISRU systems), the total mass is 30 tons for the reference mission and it is 26 tons for our scenario. All in all, the difference might be in the order of 10 to 15% of the total IMLEO mass, which approximately represents 100 tons.
- Third, it is suggested by NASA to send 6 tons of methane to the surface of Mars and to use the ISRU systems only for the production of oxygen. In our scenario, we rather suggest to manufacture on Mars all the propellant of the Mars ascent vehicle. This is justified because there are two ascent vehicles and the total amount of propellant is

higher, especially for the single habitat option. It has been shown that the standard and the single habitat options are equivalent in first order approximation for the IMLEO criterion. As a consequence, the proposed ISRU systems do not provide any significant advantage over the NASA reference mission.

- Fourth, in the proposed scenario there are only unpressurized rovers. The mass savings are in the order of 20 to 30 tons for the IMLEO because there is a direct impact on life support, power and also stowage and tools for the deployment on the surface and indirect impacts on heat shields, structure and propellant for TMI.
- Fifth, the scenario itself avoids some drawbacks. In the NASA reference mission, the same transit habitat is used for the outbound trip and the return. Therefore, there are enough consumables for the totality of the mission in case it is not possible to land on Mars and there are consumables in the surface habitat. Because of this redundancy, 8 tons of consumables may have to be jettisoned before TEI. In the proposed scenario, the same habitat is used for the outbound trip and on the surface. If landing is not possible, the crew stays in the same vehicle, there is no need to duplicate the consumables in the ERV. The mass savings are probably more important than just 8 tons because in the NASA mission the redundancy is required for all systems of the life support. The IMLEO impact may be 20 to 30 tons.
- Finally, the most important part of the mass savings is due to the EDL systems. The capsule shape and the 70° cone heat shields are very efficient because the heat is concentrated on one side only. There is no need to add other heat shields and thermal protection systems everywhere. The use of parachutes also saves an important amount of propellant. In addition, the gains are cumulative. The lighter the landers, the lighter the heat shields, the less powerful the engines and the less propellant is needed to land on Mars. In the NASA reference mission, the mass of the heat shield for the landing vehicle is 22.5 tons, which represents 21% of the entry mass. It is only 12% of the entry mass in our scenario (Christian *et al* [8]) and the requirements for the thermal protection systems are much lower. All in all, the impact of the EDL systems (heat shields, thermal protection systems, structure, engines and propellant) on the IMLEO may be in the order of 100 tons.

9. Conclusion

A revised version of the 2-4-2 scenario has been proposed. Considering that a major risk driver is the choice of the systems and procedures for EDL, it is suggested to split the vehicles into two smaller ones and to choose a standard capsule shape with a large diameter. This choice enables a significant reduction of the EDL risks and avoids LEO assembly.
Our proposal is clearly less ambitious than the NASA reference mission with six astronauts and heavy rovers. However, there are backup options at every step of the mission and the risks are minimized. The details of the scenario still need to be examined. In this paper, several options have been proposed for the choice of the Mars ascent vehicle and the size of the ERV. If a dedicated small MAV is sent to Mars, there is a large impact on the size of the ERV and consequently on the capacity of the launcher. If the habitat lander is used for the ascent from Mars, a large amount of propellant has to be produced and the ISRU systems are heavy. A trade-off is possible. The habitat lander can be used for the ascent but many objects can be left on the ground. If that option is chosen, the objects have to be duplicated in the ERV. The best choice depends on complex parameters such as the capacity of the heavy launcher and the feasibility of using the same habitat for the entire mission.
Other interesting options might exist. Therefore, we recommend that the community working on human missions to Mars carry on investigating new ideas on the subject. The main

conclusion of this work is probably that such missions are not as complex and expensive as suggested by the last NASA study.

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