Launchers and Landers for a Manned Mission to Mars

O. Hag, J. Rabault, T. Olergård, S. de Vauplane, G. Balmer
Royal Institute of Technology, 100 44 Stockholm, Sweden

20th March 2014

A key issue in conducting an expedition to Mars is making sure that all the necessary payload can be carried safely back and forth. As group 2 of the Red team, we have been in charge of finding technical solutions to transfer payload through the atmostpheres of both planet Earth and Mars. Our tasks consisted in defining the best way for going from Earth ground to LEO, from Mars orbit to Mars surface, back from Mars surface to Mars orbit, and finally from earth orbit to earth surface. The presence of the atmosphere of the planets, and the very specific low pressure conditions on Mars, impose to push further some current technologies to fulfill this objective. As a consequence, we focused on analysing which technologies should be adapted to the requirements of the mission in order to provide the safest and most cost-efficient solution.

1 Introduction

1.1 Motivation

Mars has inspired humanity for centuries – it is probably the most intriguing planet in our solar system, and it a popular subject of both science and science fiction. This popularity is mainly due to the idea that Mars may have hosted life at some point, a question which is still unanswered.

This interest for Mars and its potential life forms has sparked a desire to explore it, and even before Yuri Gagarin's pioneering spaceflight there were first ideas for manned missions to Mars. Six decades have passed since, and still no human has set foot on Mars, but space technology has advanced, and many believe that the the first mission will launch within about 25 years.

1.2 Task and Scope

This paper is part of a group project with the goal of drafting a concept for a first human mission to mars. Many challenges have to be met to successfully complete such an endeavor, and the taks have been distributed across several student teams. This paper deals with the transport to and from the surface of the planets. For the Earth part, both launch and landing, there are proven existing systems and more are being developed. For the Mars part, however, things get more challenging, partly due to Mars' particularities, as the thin atmosphere, and party due to technical constraints, mainly the need to keep the mass low, but also for example the lack of a navigation system.

A different group is designing the transplanetary vehicle, and the boundaries of the work were established as follows: this paper concerns the parts from surface to a stable orbit and vice versa; the injection into the transplanetary trajectory as well as the orbit insertion belong to the transplanetary vehicle.

2 Access to Low Earth Orbit (LEO): technical and industrial situation

2.1 Constraints for technical solutions

Access to LEO is the first step to any space mission [5]. Today, the only way to achieve access to LEO is to use expandable chemical rockets [1]. Their main specifications are determined by the mass they can put on LEO. Other critical characteristics of those rockets are volume constraints (in particular, limited diameter of the payload) and most important price per kilo in LEO.

The use of chemical rockets for this part of the mission is the result of several requirements. First, a very high thrust is required to counter the gravity and accelerate the payload to the LEO orbital velocity of about $7.8\,\mathrm{km\,s^{-1}}$. This constraint discards some of the high Isp thrusters, such as plasma and electrothermal (Vasimir) engines, that have a low thrust to mass ratio. Moreover, since the trajectory starts from the ground and goes through the low atmosphere, pollution and environmental issues are also critical. This raises issues against launchers that use toxic fuels, such as the Proton. This means also that thermal nuclear propulsion, which is the only high I_{sp}, high thrust to mass ratio engine technologically mature today, cannot either be used for this part of the flight.

2.2 Criteria for launch system choice

As a consequence, the only solution available for this part of the mission is traditional chemical rockets. The I_{sp} of the chemical fuels used can slightly vary, but the corresponding order of magnitude is always around 300 s. This has dramatic consequences on the costs of transferring mass to LEO. For all the considerations related to access to LEO, the cost of putting 1 kg in LEO will be used as the main indicator of performance. We will not study in details the mechanisms of staging and the design of the rockets, since the mission we are planning corresponds rather to an end-user standpoint on this question. As a consequence, cost efficiency and safety for the crew will be the critical factors for this part of the mission.

A choice that drastically impacts the costs of launching is the choice between 'government' and 'private' rockets. Since the private industry has proven in recent years its ability to succeed in performance, budget and schedule constraints, both solutions will be considered without distinction as long as they fulfill the requirements for the mission [3]. This approach leads to considerable cost savings.

2.3 Launchers available for a Mission to Mars, 2025

In the perspective of a mission to Mars around the year 2025 to 2030, and due to the current space budget limitations, it is necessary to use available or in development rockets. Table 1 presents the different options available.

2.4 Launcher choice and scenario for access to LEO

The diameter of the payload has proven, after discussion with the concerned group, not to be a critical issue as long as modules of diameter superior to 5 meters can be lauched. As a consequence the main focuses are on price, payload and availability. Since several launches will anyway be necessary for the approach our team is considering, the payload issue is not critical. As a consequence, cost efficiency becomes the leading factor.

Cost efficiency considerations lead to the choice of the "Falcon" rockets as best rockets for our project. Since the development of the Falcon XX is still hypothetical, we choose to design the Mars Mission resorting to the Falcon Heavy [2]. The Falcon Heavy fits our criteria for performance, and will be tested in 2014 [4]. Moreover, it is very similar to the still hypothetical Falcon reusable. As a consequence we can design our mission using Falcon heavy char-

Table 1: Summary of rockets considered for a mission to Mars

Rocket	Mass to LEO	\$ per kg	Payload diameter	Status	Other
Falcon 9 v1.1	13 t	4300 \$/kg	5.2 m	operational	New versions in development
Proton-M	22 t	4500 \$/kg	4.15 m	operational	Toxic fuel (DMHD)
Soyuz	7.8 t	6150 \$/kg	3.35 m	operational	3 crew
Long March 9	130 t	?	? (> 5 m)	developement	governmental
SLS	130 t	?	? (> 5 m)	development	governmental
Falcon heavy	53 t	2200 \$/kg	5.2 m	development first flight 2014	
Falcon reusable	≈ 35 t	800 \$/kg	5.2 m	developement	reusable first stage: test 2014
Falcon XX	140 t	?	? (> 5 m)	project	development uncertain
Angara A7V	40.5 t	?	? (> 5 m)	development	Proton successor

acteristics, and decide if the development of Falcon reusable succeeds to finally use it. We can expect that the inferior payload of Falcon reusable will be increased through performance improvements before 2025, so that it could in 10 years fully reach the performances of the current Falcon Heavy. Off nominal cases should not be encoutered and no laucher should be lost, since the corresponding rockets should be used and tested during a lot of flights before the mission. However we should plan that if problems were encountered with some non manned launch, spare capacities would be available for launch. The manned lauch will resort to the usual escape tower technology to ensure safety of the crew.

The scenario for departure from earth for the mission will thus rely on the use of several (10) Falcon 9 heavy. Orbital Rendez-vous maneuvres will be necessary to build the final spacecraft. For further analysis, the price to LEO will be fixed 2200 \$ per kg. We however keep in mind the possibility that the price could go down to 800 \$ per kg if the development of Falcon reusable succeeds. Figure 1 presents the configuration of the first stage of Falcon heavy. It consists of three identical cross feeded first stages, each of them very similar to the Falcon 9 presented in picture. An artist view of the global Falcon heavy is also exposed with some technical features.

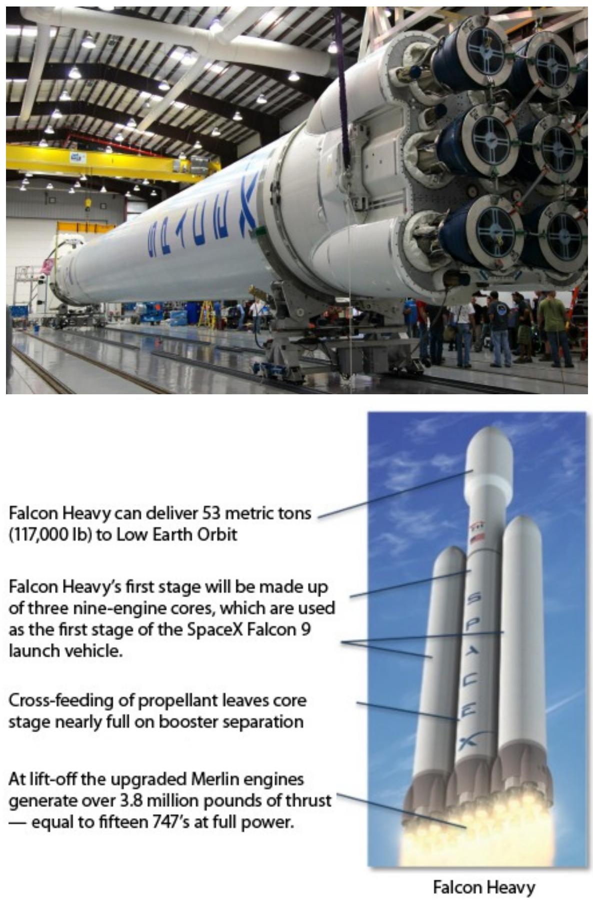

Figure 1: First stage of the Falcon 9 (top). The first stage of the Falcon heavy consists in three such stages. Artist view of the Falcon heavy in flight (bottom). Both images from [2]

3 Mars operations overview

Two of the most critical phases of the mission will be the Mars landing and ascent.

The scenario for our mission consists of several landings on Mars. In particular, the Mars habitat and the life support system module will land before the crew capsule (MES, Mars Excursion System). As a consequence, precise landing is absolutely necessary to ensure that the crew will be able to go to his habitat. In order to ensure precision landing, a network of radio balises analogous to the GPS network will be automatically deployed on the landing area before any other event.

Landing will take place the same way for all units except the guidance one. It will resort to aerobraking and parachutes to break most of the speed while saving fuel. Rockets will be used in the very final part of the landing, to accurately choose the landing location using balises guidance and land smoothly. The maximum payload that can be landed on Mars with the design we investigated is 45 tons, which imposes to land separately the habitat and the Life Support / ISRU (In Situ Ressource Utilization) module.

The ascent will take place in the same vehicle that was used for landing (MES). The MES will be refilled on Mars surface, using ISRU technologies available on the corresponding unit to produce oxygen and methane. In order to optimize its efficiency, the MES will resort to parallel staging for the ascent from Mars.

4 Mars Navigation

The crew lander and service modules has to touch down at a location close to the habitat, at least within 100 meters, in order for refueling, access to hardware and life support system to be possible and therefore, mission success. This introduces a navigation problem since landing with that kind of accuracy is currently not possible without first introducing a system for accurate navigation close to Mars.

A solution to this is to introduce navigation beacons, or pseudolites, to the surface of Mars. If the relative distance between the beacons is known, navigation relative to the beacons and other objects communicating with the system, such as the cargo landers, becomes possible.

The concept consists in sending a probe to Mars, before anything else is sent. This probe brings the radio beacons to Mars and spread them over the desired landing location for the primary Mars mission. These beacons should be separated by a few hundred meters to give good enough accuracy. The number of fully functioning beacons needs to be at least four. Considering their relative low weight, and how crucial the good function of the navigational system is to the primary mission, at least two extra beacons should be deployed to assure good reliability and resilience to off-nominal scenarios.

The positioning system is called pseudolites, or pseudo-satellites. They work in a way much like the satellite GPS system, replacing the satellites with radio beacons. To find the relative distance to the beacons, the time it takes for the signal to be transmitted is converted into distance $d = ct$ were t is the time and c is the speed of light. To calculate a position within meters, the time signal must be accurate within nanoseconds. This requires atomic clocks, on all beacons as well as the landers or units carrying this navigational system.

The purpose of this navigational system is to have a high accuracy for finding the relative positions between the different landers and beacons. To achieve this high accuracy, synchronization is required between the beacons in order to accurately determine the relative distance between them with an accuracy of about a centimetre. The atomic clock chosen for this mission is the NASA Deep Space Atomic Clock (DSAC). It was chosen for its reliability a low mass (3 kg) [20].

The landers can have a regular quartz clock instead of atomic clocks, which would save power as well as avoiding fragile components. A quartz clock is not as accurate, but can continuously reset itself from other beacons. This option would require four instead of three fully operational beacons, so each unit has three other beacons to synchronize with.

A complication for this strategy is power. The atomic clocks requires a power source, but the broadcasting needs a signal strong enough so that the beacon can be received by the lander which need accurate position information. Solar panels charging a set of batteries is a possible solution. The battery may very well be the component with the highest mass but a higher capacity means higher power for radio emission and therefore increased range. A signal strength of 3 W should be strong enough for the landers and beacons all to receive a strong enough signal.

Since the power is limited, the signal should only be broadcasted at mission crucial periods. Before the signals are needed the beacons can be in a hibernating state, not broadcasting. When the crucial period arrives, ie. incoming lander, it can be woken up by receiving a wake-up signal.

The battery should be able to retain its charge for a longer so that when the landers arrive and a strong signal is required, the battery is fully, or at least close to fully, charged. A battery of type lithium thionyl chloride (Li $SOCl_2$) is chosen which has a specific energy of 200 Wh/kg[19]. A battery of 3 kg would then suffice to have power for 20 hours without any charge from the solar panels. Using the same solar panels as the Mars Exploration rovers, with a power output on Mars of about $55\,W/s^2$, the sufficient area for the array should be about $1\,m^2$. That should be well enough to power the unit as well as charge the battery.

The complete beacon units would have an estimated mass of 8 kg each, including atomic clocks, battery, solar panel and antenna. To send this payload of total mass of approximately 50 kg to the primary landing site of the mission in advance is a mission that has been done several times before and will not be discussed further in this paper.

5 Entry and Descent

5.1 The ΔV needed to land to Mars

The ΔV needed to slow from a hyperbolic approach trajectory to a useful science orbit is:

$$\Delta V = V_{hyp} - V_{circ}$$

Moreover, due to the rocket equation, the mass increases exponentially with the ΔV for a propulsive capture. This is why we should take aerocapture to save fuel, mass and therefore costs.
This is all the more so critical since *after calculations, the entry velocity to break is between 5 to 8 km/s.*

5.2 The heat shield

5.2.1 Introduction

Thermal Protection Systems (TPS) are designed for space crafts and space capsules to protect them from the heat peak when they enter the atmosphere.[15] Indeed, during entry, temperatures can reach around two thousands degrees celsius. We have to decide between two main classes of TPS, reusable TPS, and ablative TPS.

Reusable TPS cannot resist the most extreme temperature but they can easily withstand moderate heat. On the contrary, ablative shields for TPS are based on Carbon-Carbon materials, that have a higher density and resist a higher heat thanks to ablation mechanisms. Thus we will decide the kind of TPS depending on the use that we want to make with it.

5.2.2 How to design the heat shield?

There are four major and critical parameters that drive us for designing the TPS for the phase of Martian atmosphere entry.

1. Peak heat flux
2. Stagnation pressure
3. Total heat load

4. Peak deceleration

Here are the different graphics that resume the last missions in space.

5.2.3 Solution needed for our mission

In our case, we have to understand that aerocapture and entry from Mars orbit produce different thermal consequences, and the TPS need to whithstand the two different situations. Indeed, the aerocapture peak heat rate[14] is more than 9 times greater than the EDL peak heat rate (38 W/m^2 vs 4 W/m^2. Moreover, the aerocapture dynamic pressure is twice as the EDL's, and the aerocapture total peak load is 6 times grater than EDL's.

After aerocapture, the crew landing capsule will begin his entry when it reaches the Martian atmosphere, around 130 km from the surface. Then its temperature will exceed 1600 °C, and this imposes to choose a heat shield that can be configurable for a dual-use: aerocapture and EDL.

Because of the high temperature and the high heat peak rate, we decide to use an ablative TPS that will used first for aerocapture, and later for entry. The SLA-561V[16], a specific ablative material begins significant ablation at a heat flux of about 110 W/cm^2. As a conclusion if fits our requirements and it could be a good solution for our mission. The heat shield is very needed for the mission and for the atmospheric entry, to save fuel and allow aerocapture.

5.3 The parachute

5.3.1 Why do we need some parachutes?

After the vehicle enters the upper atmoshpere at around 7 km/s, its velocity will decrease a lot, but, even 2 minutes later, the crew capsule will still be at nearly 400 m/s.[13] The parachute, even if its efficiency is limited, will reduce reduce capsule's velocity by a factor of 6 (slowing down to around 65 m/s), which represents huge fuel savings.

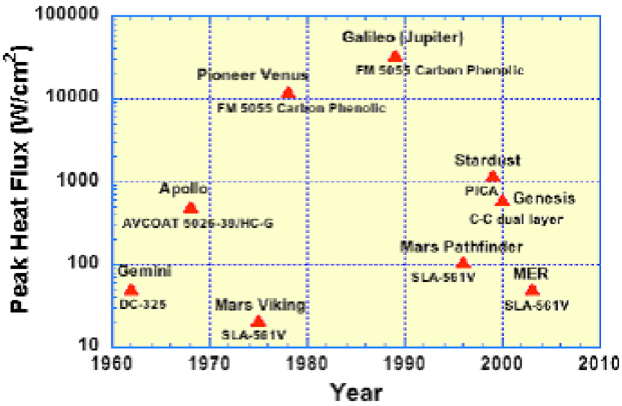

Figure 2: Chronology of ablative TPS for NASA entry missions

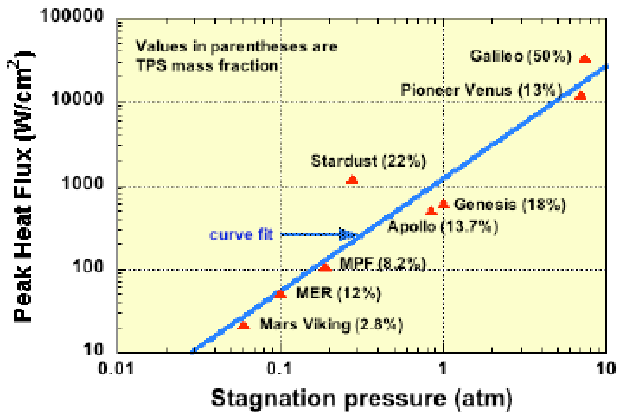

Figure 3: Mission environments for ablative TPS application

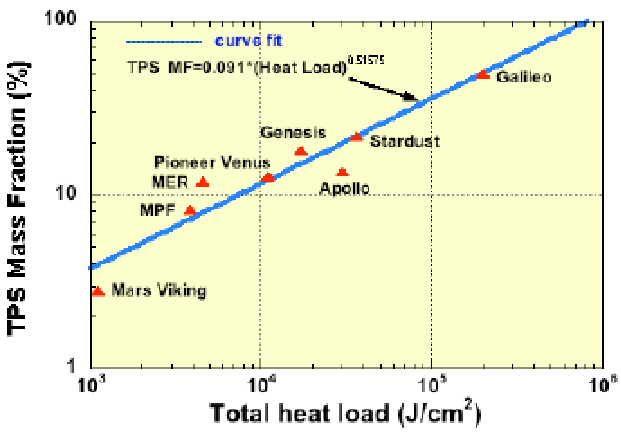

Figure 4: TPS mass fraction for prior ablative TPS missions

5.3.2 Solution needed for the mission

The materials used to make the parachute must be strong, but not heavy to be fitted into a small aera and not to increase the weigh of the capsule. In our case, we take the option of 3 supersonic parachutes of 30 m of diameter that will be used as aerodynamic decelerators from supersonic to subsonic speeds (that is to more than 340 m/s to less than 70 m/s). Rocket assisted descent motors will work in tandem with the parachute for the final phase of the landing.

5.3.3 Critical analysis of the solution

Today, such parachutes do not exist and this is maybe the biggest issue our groupe has to face. However, after atmospheric entry vehicle's velocity is around 400 m/s and we have only two solutions:

1. use supersonic descent propulsion directly to land to the martian surface. Such a propulsion doesn't exist and would use a lot of fuel

2. use supersonic parachutes of 30 m of diameter in order to reach subsonic velocity and then use subsonic propulsion for the last deceleration. Such supersonic parachutes exist but NASA never tested such large scale parachute. Nevertheless, some test have been done and this technology is more advanced than supersonic's propulsion. Moreover, with this proposition, less fuel is needed.

Those considerations explain why we finally chose the parachute system, which seems to be closer to achievement even if further developments are necessary. Such system would also be less fuel consuming.

5.4 Conclusion

Aerocapture phase:
The heat shield used will be the ablative SLA-561V.

Entry phase (130 km to around 30 km):
The heat shield used will be the same ablative SLA-561V. The velocity then goes from around 7 km/s to 400 m/s

Parachute phase (30 km to Mars surface):
3 supersonic parachutes of around 30 m of diameter are needed in this critical phase. The velocity goes from 400 m/s to 60 m/s.

6 Powered landing and Mars launch

The main issue at hand with propulsion systems in operation today is their efficiency relative to thrust and weight. When selecting the desired propulsive system for the spacecraft, several factors must be taken to into account. The objective of placing humans on the surface of Mars and then returning them back to Earth involves several different steps. The most efficient choice would be to use different propulsion systems with different characteristics depending on each situation. The major aspect covered here is propulsion systems needed for landing on Mars and ascent to Mars orbit. The best choice is most likely chemical engines, since electric propulsion has very low thrust to weight ratio and nuclear has a high system mass. This does not narrow much the options, and there are several different types of chemical engines available with current technology, based on a multitude of different types of propellants. [17]

The chemical engine does have its drawbacks. It has relatively low efficiency and it is also hard to start. All the different engines are different regarding those characteristics. For instance some engines have higher efficiency but are even harder to start compared to others. This means that engine selection has to be dependent on what type of situation they are going to be used for and which are the problems inherent with them. The mission is split up into three separate descents to Mars surface. The first one aims at landing the habitat, the second one is the landing of the services and equipment module and the third one is the Mars lander vehicle for the crew. All three have different requirements which in turn means all three will almost likely need different types of propulsion.

6.1 Habitat

The habitat is probably the first module to land on Mars which means it has the lowest requirement in terms of accuracy when it comes to its exact position on the landing site. This means less requirement on ΔV for the vehicle since it does not need to be able to adjust its position very precisely by drifting a lot during its descent through the atmosphere. On the other hand it's also the heaviest module, so it requires a system which is highly reliable but still has low enough mass to be able to be launched from earth. As a consequence, the probable solution would be a hypergolic propulsion system. The typical efficiency of this type of systems is around 315 s for I_{sp}. Hypergolic propellants are reliable and storable system. Those propellants put a lower requirement on the mass of the tanks but also on the mass of the rocket engines themselves. If one takes a closer look at the only operational hypergolic rocket in use today, the proton rocket [18], one can see that it has very low structural weight. But the real benefit of using these types of systems is the abillity to store the propellant in space without any further requirements on the tanks in terms of cooling equipment and/or extra shielding from heat. This means that a compact but reliable propulsive landing system can be designed for the habitat module.

6.2 Service module

This module weighs slightly less than the habitat module but will have to land relatively close to the habitat module in order for the operation to work. This means a greater requirement on delta V needed for the craft.

6.3 Lander

The lander module has the biggest requirements when it comes to designing the propulsion system because the fuel for the ascent will most likely be manufactured on Mars, taking advantage of In Situ Ressource Utilization. That means there's only one choice when it come to its propellant: liquid oxygen and liquid methane. It also has the highest requirement on ΔV because it must land close to the two other modules (within 100 meters, which means ability for important drift time)

The design of the lander will be based on parallel stage system with cross refueling. This is the same principle behind the Falcon Heavy rocket from SpaceX [2]. The main advatages of this design is that the center stage will use its engine but will remain fully fuelled untill the outer stages seperate. That means a lesser amount of engines and thus lower mass. Unlike the Falcon Heavy, this spacecraft will have 4 parallel boosters instead of 2 for a total of 5 tanks and engines.

The crew cabin will consist of a DragonRider (crewed version of SpaceX dragon capsule). It has the benefit of having an intergrated launch abort system. This gives us the possibility to save the crew in case of a failiure when returning to mars orbit, and provides safety when it comes to off nominal cases. The capsule with crew weighs about 8000 kg [9].

The needed ΔV to reach Mars orbit is estimated to 4500 m/s. The structual mass of the stage is assumed ot be 10% of the total weight. With all of this assumed the total empty weight of the vehicle becomes 11 t. The fully fuelled vehicle is 33 t. A scale drawing based on these calulations can be seen in figure 5.

6.4 Landing ΔV

The amount of ΔV needed for each landing is dependent on two factors. The first one is the speed left after all the atmospheric deceleration has happened. This quantity is quite hard to evaluate in the early concept phase but some preliminary estimated numbers are available. Some sources estimate this quantity to roughly 65 m/s [ΔV_{rest}] if parachutes are used to their full effect. The second factor for the ΔV is the flight time needed to compensate for the drift that occurred during the decent through the atmosphere and the inaccuracy of the orbital calculations. This "drift time" [Δt_d] is then

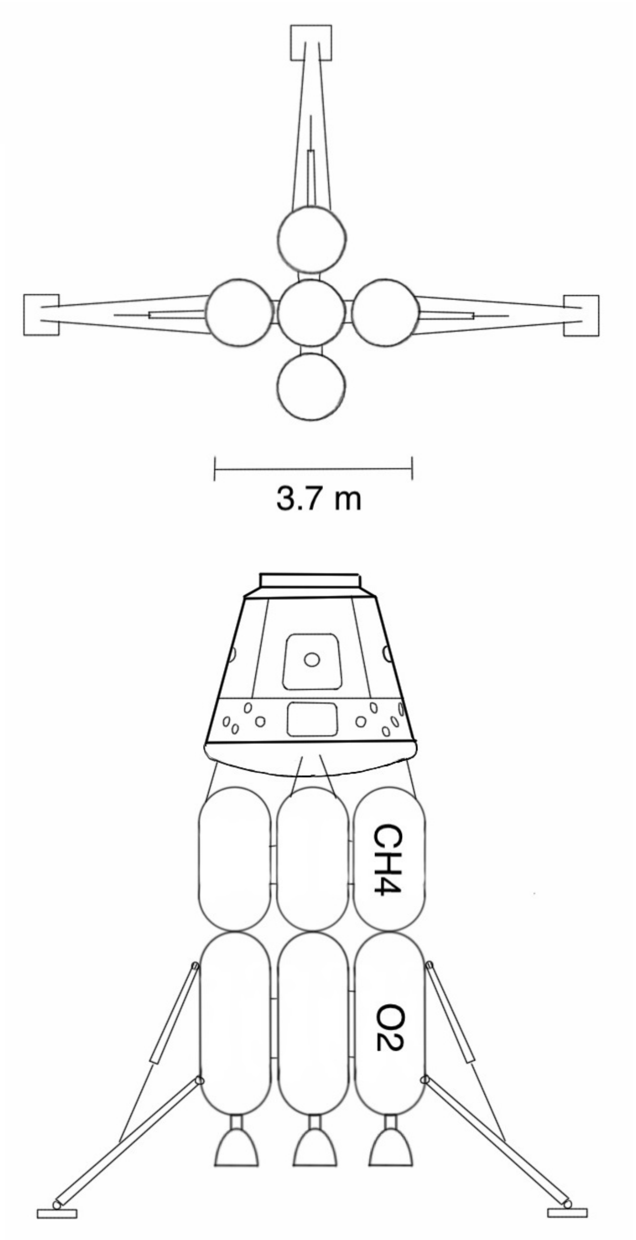

Figure 5: Dimensions of the lander based on the densities of liquid oxygen and liquid methane.

multiplied with the gravity factor of mars to get the needed ΔV for compensation of the drift. The original equation for those calculations is Eq 1.

$$\Delta V_{drift} = g_{mars} \cdot \Delta t_d \tag{1}$$

$$\Delta V_{tot} = \Delta V_{drift} + \Delta V_{rest} \tag{2}$$

The mass of the propellent is caculated from Eq. 3.

$$m_{ps} = \frac{m_c - A \cdot m_c}{ratio \cdot A - 1} \tag{3}$$

Where m_c is mass of the cargo (mass of the habitat for example), A being

$$A = e^{\Delta V_{tot}/v_e} \tag{4}$$

where the ratio is the propulsion system mass relative to the propellant mass:

$$ratio = m_{ps}/m_p \tag{5}$$

As stated earlier the drift time will be dependent on which type of module is landing.

Table 2: Estimated data

Cargo	drift t	$ratio$
Habitat	15 s	0.2
Service module	30 s	0.15
Mars lander	60 s	0.1

Numbers shown in table 2 are rough estimates. Flight times are opened to optimization. Since the habitat is very heavy it will need relativly high thrust, thus resulting in a hihger mass fraction in the propulsion system.

Using the estimated the numbers from table 2, the equations presented previously, and assuming an I_{sp} of 315 s for the propulsion system a mass table can be calculated. This is only done for the habitat and service module. The lander mass is based on the assumption stated in this section together with the calculated ΔV based on the required drift time.

Table 4: Mass table

Module mass	System mass	Propellent mass
50 t	2.3 t	1.9 t
30 t	2.0 t	1.7 t
11 t	-	0.9 t

6.5 Conclusion

The main conclusion to draw here is that it is possible to find a propulsive landing system that fulfills the requirements corresponding to the mission plan, while at the same time keeping the mass low enough to save costs. All the masses given here are estimates, and may thus have to be adjusted in more precise estimates.

7 Earth re-entry and landing

At the end of the return flight from Mars is the last challenge: entering the Earth's atmosphere and landing. A small, reliable vehicle will bring the crew, and potentially some rock samples, back to Earth. To land the whole transplanetary vehicle would be an unnecessarily big effort, and it can either be put into an Earth orbit (which would require propulsion), or discarded and burned up in the atmosphere.

The crew return vehicle has essentially the same requirements as an orbital manned spacecraft, i.e. space for the crew and some cargo, an adequate life support system, a heat shield for atmospheric entry and a parachute for landing. Therefore the easiest solution is to use a spacecraft that already exists or is being developed. Using an existing system will save the costs for development and testing.

The only manned spacecraft that are currently operating are Soyuz and the similar Shenzhou. Both can accommodate three astronauts and only very little cargo, if any. Other spacecraft are in various stages of development, notably in the US, where several spacecraft are being designed as part of NASA's commercial crew development programme. Table 3 lists the candidates that are likely to be available for a mission around 2025-2030.

Many factors must be considered for choosing the best solution. Mss is one of the main issues. This parameter is particularly important for the choice of the crew return vehicle since the vehicle will be transported from Earth to Mars and back. Therefore a lot of propellant can be saved by using a lightweight vehicle. The second fundamental property is reliability. There is no spare, so the return vehicle has to perform absolutely flawlessly, even after being in space for several years.

Further to consider is the payload and volume. Since this vehicle is only used for the relatively short flights to and from the surface, comfort is not an issue, therefore the volume can be quite limited. But the payload should be sufficient for 6 astronauts and some scientific cargo.

Last but not least, the price is a deciding factor. However, since the crew return vehicle is only a small part of the total budget for a Mars mission, the price is not too crucial. The key requirements are low mass and high reliability.

Comparing the vehicles in table 3, one can make the following observations: the Soyuz is significantly smaller than the other vehicles. It is very light, but because it seats only three, two capsules would be needed to return the whole crew. If there is some scientific cargo to return another more vehicle is needed, and the Soyuz quickly loses it's advantage of the very low mass. Among the other vehicles the Dragon is the lightest, and it is large enough for the

Table 3: Comparison of several crew return vehicles. Only the re-entry modules are considered in mass and volume, as a service module will not be required for this mission. The cost indicated is for a reference flight to the ISS. Data sources: [6] [7] [8] [9] [10] [11] [12]

Vehicle	Mass	Volume	Crew	G-load	Cost/Flight	Status
Soyuz TMA	2900 kg	3.5 m^3	3	4-5 g	~ 180 M$	operational
Dragon	~ 8000 kg	10 m^3	7	3.5 g	140 M$	cargo version operational
Dream Chaser	~ 9000 kg	16 m^3	7	1.5 g	?	in development
Orion	~ 9500 kg	9 m^3	6	?	?	in development
CST-100	~ 10000 kg	?	7	?	?	in development

whole crew. Moreover, the cargo version of the Dragon is already operational. This gives the Dragon a head start, the first manned flight is expected to take place in 2-3 years. This gives confidence that by the time of this Mars mission, Dragon will have been thoroughly tested and will have proven that it is a reliable vehicle.

Based on these considerations, we have chosen the Dragon as our crew return vehicle. It is safe to say that it can be used for our purpose with only small modifications, and although it is difficult to find information about the costs of different vehicles, we can assume that the Dragon's cost is similar, if not less than other systems.

Figure 6 shows a drawing of the manned Dragon with a crew of seven.

Figure 6: The Dragon capsule with a crew of seven (Image source: [8])

8 Conclusion

Going safely through the atmospheres of planets Earth and Mars is one of the challenges that must be overcome to achieve any manned Mars mission. Our group has investigated which technical solutions should be used for those critical phases. Launch from and re-enty to Earth are well mastered with current technologies and some private companies will be able to provide relatively cheap while highly reliable rockets and capsules for those phases. Similar operations on Mars are more challenging due on the one hand to the low pressure of its atmosphere for re-entry, and on the other hand the limited mass available for launch back to space. To complete efficiently those two phases, current technologies must be pushed further. Big supersonic parachutes must be developped to accomplish re-entry without the need for large quantities of fuel. Powerful rocket engines using methane must be developped to benefit from In Situ Ressource Utilization. However those elements are currently under development and no fundamentally new technology is necessary to build them. In parallel, to allow splitting of the Mars units in several landings, accurate landing must be performed. We come with a reliable and smart solution for this point, namely the installation of a ground based navigation system during a precursor mission.

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