Mount Olympus Mons Ascension Mission Space Vehicles - Team Red

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***Abstract*—This paper investigates the design of the space vehicles required to perform a mission to Mars which aims to climb Olympus Mons in 2038. The mission has to bring at least one human to the summit and allow a safe return to Earth. Most of the designs depend on the outcomes of the other groups and are the result of an iterative process. Several vehicles have been produced for the purpose of this project. A modified version of SpaceX's Super Heavy vehicle was used for launching from Earth. A total of three launches are scheduled before the arrival of the astronauts on Mars. The first one with supplies and rover is expected to land on the starting point of the ascension to Olympus Mons. The second and the third ones are planned to bring two fuel tanks to resupply the Crew Capsule before it performs a hop toward Gusev Crater. The Crew Vehicle has the duty to bring the astronauts from Earth to Mars's surface. Two different tanks, T1 and T2, are required to perform all the orbital maneuvers and the ballistic hop once on Mars. The habitable part of the crew vehicle is the Crew Capsule, which is shielded against radiation and is designed the same way as SpaceX's Dragon Crew. The Crew Capsule has the duty to transfer the explorers from Earth's surface to Low Earth Orbit and from Low Mars Orbit to Mars's surface. Finally, a Crew Station based on inflatable technology was designed to be launched into LEO before the mission starts, on board a commercial launcher. It is docked to the crew vehicle before performing orbital maneuvers. The station contains every life support system for long duration space missions and was studied to provide an artificial gravity environment in order to counteract the effects of a long-term weightlessness, thus allowing the astronauts to get used to Mars's gravity. The mass of each vehicles and the fuel requirements were calculate and can be found in the results.**

NOMENCLATURE

LEO	Low Earth Orbit
LMO	Low Mars Orbit
g_e	Acceleration due to gravity at sea level on Earth
g_m	Acceleration due to gravity at sea level on Mars
ISS	International Space Station
I_{sp}	Specific impulse
EDL	Entry, Descent and Landing

I. INTRODUCTION

A. Mission Description

The goal of the mission is to be the first team to reach the summit of Mount Olympus on Mars, the highest mountain of the solar system, with at least one human. The mission is planned for no earlier than 2038. There are also some restrictions, because it is not allowed to use a flying vehicle above 10km below the peak and that the last 1000m of altitude has to be done without any motorized vehicle. The goal of the space vehicle team is to design all the vehicles that are needed to perform the mission. Those vehicles have to enable the crew to perform interplanetary journey, to survive in space and on Mars, and to come back safe on Earth at the end of the mission.

B. Design Process

It is of primary importance to understand all the inter-dependencies between our design choices and the development of other groups work. A too harsh modification of the mass needed by Mars Operations could prevent a certain vehicle design from working, thus greatly affecting the masses of the vehicles, therefore jeopardising the logistics and forcing the mission design group to rethink the whole timetable. Our team made the choice of designing new and innovative vehicles, in order to extend our vision beyond the current existing programs. Therefore a lot of requirements had to be considered:

- From Mission Design [1] : the detailed ΔV budget for transfer maneuvers. Values of LEO and LMO, 400 km and 230 km, respectively.
- From Human Aspects [2] : the mass and volume required for the life support system and the shielding.
- From Mars Operations [3] : the location of the key places on Mars, the landing sites for the supplies and how to perform the hops

The final design of the Space vehicles and their logistics result from a long iterative process, done in conjunction with the other groups of red team. As it is not possible to specify every intermediate design, nor the complete design process, this paper will focus on the final mission design [1], its station used for the interplanetary journey, the launchers and their requirements in terms of technology and fuel, and the different operations that the flying vehicles have to perform. This paper doesn't include the design of the rover.

C. Assumptions for the space vehicle

Several assumptions are made for this study:

- Solar panels have an efficiency of 60 %.
- The drag is neglected on Mars for most operations.
- The thrust provided by the engines is constant during each phase.
- Every rocket stage is assumed to use methane/LOX and have an I_{sp} of 450 s.
- Structural ratios ϵ of all stages are set to 0.05, and account for the masses of every subsystems, unless specified otherwise.

In the design, subsystems are not studied extensively (the communication system, attitude control system and other system for example). They are included in the structural mass but a complete design of those system is not performed. For the simulations, our vehicles are considered as point mass, and only two body problems are considered as the distance scale is big with regards to the rockets scale and the influence of Earth or Mars are the main ones.

II. Methods

The design method that was employed in this project was an iterative progression from top to bottom. In a general way, it means that we designed all the upper stages and their requirements before going to the stages below. Furthermore as none of the requirements coming from other groups were completely settled during the design process, the vehicles had to be designed around variables, so that the resulting vehicles would "adapt" to the final mass requirements input. Generally speaking, the design of the vehicles depends mainly on the usual mass ratios used in rocket science [4], and the Tsiolkovski rocket equation, defined as:

$$\Delta V = I_{sp} g_e \ln \frac{m_0}{m_f} \tag{1}$$

Where ΔV is the velocity increment, m_0 being the initial mass, m_f the final mass and I_{sp} the specific impulse of the rocket.

A. Modules for descent on Mars

As one might expect, the crew, the rover and the supplies must be able to reach the Martian surface. It was decided in this study to tackle these aspects first, before going into any transfer vehicle considerations.

For the crew, it was decided to have a three seat crew capsule, built in the fashion of Space X's Dragon Crew [5] , which would be modified and shielded for the purpose of this martian mission. As the main designed constraints posed for the vehicles are about mass, the shielding, provided by the Human Aspects group [2] is characterised, in our study, by its areal mass of 20 gcm^{-2}. In order to protect the crew from Solar Particle Events, the capsule was designed to have such a shielding on half of its surface.

On the other hand, rover and supplies do not have the need of a human rated vehicle, and it was decided to combine them in a single descent vehicle. Therefore the entry, descent and landing (EDL) system used on Curiosity, and more recently Perseverance [6], was selected and scaled up for the study's purposes through a mass factor K_{EDL} shown below:

$$K_{EDL} = \frac{m_{EDL}}{m_{Rover}} \tag{2}$$

inducing our total EDL system with a mass defined as:

$$m_{EDL} = K_{EDL}(m_{Rov} + m_{Sup}) \tag{3}$$

Later on, in the results section, one can see that the value of $K_{EDL} = 2.5$ was implemented.

B. A Habitable module with artificial gravity for long travel times

Another important aspect of the mission is the transfer time between Earth and Mars. This poses several health aspects, including micro gravity. In addition to this, according to the general outline of the mission decided by Mission Design group [1], the stay on Mars is going to be very short. For this reason, once on Mars's surface, the explorers will not have time for training to compensate the effects of long term weightlessness. To tackle this problem, it has been decided to design a space station that is able to provide an artificial gravity environment. From this point on, this habitable module will be referenced as Crew Station.

The design of the Crew Station strictly depends on the need to create artificial gravity and at the same time to keep the total mass as low as possible. In this section, the general features of the Crew Station in terms of design, materials and masses will be analyzed.

The centrifugal artificial gravity design chosen for the station creates gravity acceleration due to the spin of the whole Crew Station about its centre of gravity. This design includes a central core and two arms at the end of which two Crew Modules, i.e. the facilities where the astronauts will spend most of their time during interplanetary travel, are allocated. The general overview of the Tantalus Crew Station can be later seen in the results, Figure 2. In order to keep the astronauts fit for the operations that they will perform on Mars, a gravity acceleration of $0.3g$ is sufficient. This value is expected to allow the astronauts to compensate for the muscles and bones loss due to a complete lack of gravity and also to get them used to Martian gravity.

It is now clear that the shape and the design of the whole crew station is based on the balance between a sustainable rotational speed and a length of the arms that does not impact too much on the total mass. In fact, a short arm allows to have a small structural mass but with a very high spin rate. A significantly high angular velocity generates strong 'Coriolis' forces which might affect the astronauts capabilities to perform linear and smooth movements, affecting their daily work and training, as clearly stated in [7]. Several studies have been performed to investigate which rotational speed and which arm length are optimal to provide the astronauts a 'comfort zone' in which they can perform all their tasks and duties without being affected by the Coriolis effect. In the literature it is suggested not to exceed a rotational speed of 4 rpm [8], preferably staying closer to 3 rpm.

In order to accomplish the strict mass requirements that the Crew Station had to satisfy, a rotational speed of 3.5 rpm was chosen. This value allows the crew to experience minimal negative side effects [9].

By considering that:

$$1 \text{ rad/s} = 9.55 \text{ rpm}$$

$$a = r \times \omega^2 \tag{4}$$

and accounting for $a = 0.3$ g $= 2.94$ m/s^2 and $\omega = 0.3665$ rad/s , a value of around 22 metres is obtained. This represents the distance of the Crew modules from the centre of gravity. As a consequence the total length of the Crew Station is expected to be 45 metres. As clearly stated in [7], with these values for the length of the Crew Station, the artificial gravity that should be achieved and the rotational speed, a comfortable environment should be produced.

The problem of the gravity gradient has also been addressed. According to [9], a gravity gradient of 6 % can be well sustained by the astronauts. The gravity gradient can be computed by using the following equation:

$$A_{gradient} = (\frac{\pi \times \omega}{30})^2 (R_f - R_h) \tag{5}$$

where R_f and R_h are the distance from the centre of rotation of the feet and of the head of the astronauts respectively. A gravity gradient $A_{gradient} = 4.6\%$ with respect to the gravity on the floor of the Crew Modules was found.

The design of the two Habitable Crew Modules and the two arms requires the use of new and innovative solutions in order to keep the mass as low as possible. The feature of both the arms and the crew modules is the extensive use of inflatable materials. These materials are currently under development and can be found on board the ISS in the Bigelow Expandable Activity Module (BEAM). This technology is a direct derivative of the TransHab Technology developed by NASA in 1990s [10] [11]. The inflatables consists of a metallic truss-like core both for the arms and the Crew Modules and a deployable pressure sheld.

The metallic core is needed to maintain structural rigidity and to carry any thrust loads or docking port required [12]. The Crew Station is assembled once in LEO. During launch the inflatables will be folded in the launcher and deployed once in LEO. Such a concept has several advantages:

- Low weight due to the strong lightweight structure;
- Larger living space for the crew with respect to a classic metallic structure for the same mass;
- Modular approach with no particularly large launchers required;

It is noteworthy that the internal floors as well as the internal layout of the modules will fold out after inflation. Furthermore, all the vital systems, such as the Life Support System and Avionics, must be allocated inside the metallic core before launching. The chosen design includes an interface of the inflatable materials directly on the cylindrical walls in the so called 'Radial Transit configuration' [13].

The inflatable structures are made of non metallic materials including graphite composite and high-strength fabrics. They are able to carry and sustain similar loads with respect to a classic metallic structure but with considerably important savings in launch mass and volume [14]. The multi-layer configuration of these materials also allows to have thermal and space debris/micro-meteroids protection included in the layers. According to [10], the structure of the inflatable materials consists of:

- an inner layer made of Kevlar and Nomex which will provide a first flame-resistant barrier that interfaces directly with the habitable environment;
- a triple redundant system called 'Bladder';
- a Restraint layer responsible for carrying the high radial loads caused by the pressurization of the Crew Modules;
- a Micrometeroid/Space debris protection layer;
- a Thermal protection system;
- a deployment system;
- an atomic oxygen protection layer.

The actual thickness required by these materials is around 40 cm even if due to technological improvement it is expected to be reduced by the starting date of the mission.

The central metallic module of the station is crucial not only because it structurally ties the two Crew Modules, but it also carries loads between the vehicles and permits crew transfer from one part of the station to the other. The main feature of this module is the presence of two de-spun platforms. In the first one the communication system (i.e. antennas) and the solar panels are allocated while in the second one, a docking facility is required to allow docking with the crew capsule. Due to the presence of these platforms there is no need of implementing antennas and solar panels that are able to track the Earth or the Sun while spinning at 3.5 rpm [8]. The sizing of the Crew Station has three guiding factors:

- The total length has to satisfy the requirements for the generation of artificial gravity with the given rotational speed as previously explained;
- The total habitable volume should be sufficiently high to allow the astronauts to have a comfortable journey;
- The mass requirements must be accomplished;

According to NASA's Human Research Program (HRP) [15], the minimum acceptable habitable volume for a a long journey in deep space is 25 m^3 per astronaut. However, by accounting for this value, the structural mass of the whole Crew Station was well above the limits given by the performance of the rocket. As a consequence of this, the habitable volume per astronaut, in accordance with Human Aspect team ([2]), and relying on the explorative nature of the mission, has been lowered.

In order to compute the pressurized volume both the Crew Module and the arms have been approximated as cylinder. A factor of 30 % has been used to compute the habitable volume from the pressurized volume as suggested in [16]. The final values for the dimension of the Crew Module (i.e. inflated diameter, height, habitable and pressurized volume etc.) can be found in the Results section.

One of the main challenges was to provide a detailed mass breakdown structure for the Crew Station. The mass estimation of the inflatable structures has been done by comparizon and scaling from the Bigelow Expandable Module currently in use on the ISS. According to [17], the BEAM has a total mass of 1 413 kg for a total pressurized volume of 33 m^3. The total mass of the BEAM includes an Aluminium structure and 2 metal bulk heads which may be compared to the central metallic truss needed in the Tantalus Crew Station.

From this value a mass-per-volume ratio of 42.81 $\frac{kg}{m^3}$ was computed. A scaling factor of 80 % has been taken into account in order to include future technological improvements that will allow the use of new and lighter material giving a final mass-per-volume ratio of 35 $\frac{kg}{m^3}$. This value has been used to compute the total structural mass of the Crew Station.

For the central metallic core a mass estimation was made based on the paper from the American Institute of Aeronautics and Astronautics (AIAA) titled 'Modular Space Vehicle Architecture for Human Exploration of Mars using Artificial Gravity and Mini-Magnetosphere Crew Radiation Shield' previously cited. The final mass breakdown structure, accounting also for the outputs from Human Aspect team, is presented in the Results section.

The spacecraft needs power supply during the whole mission to power life support system, control system and any other system needed. The power required is supplied by solar arrays attached to the habitable module. The estimate of the power required is scaled on the ISS pressurized volume. Moreover, solar power decreases non-linearly with regards to the distance from the sun. It means that the solar arrays need to be able to supply the power required at Mars' aphelion. The fact that solar power is conserved over a sphere, and the use of Stefan-Boltzman law lead to the surface of the solar arrays (S) being equal to :

$$S = F_s(\frac{R_s}{a_M})^2\frac{P_{req}}{\eta} \tag{6}$$

where F_s is the solar radiant exitance, R_s the solar radius, a_M is Mars' aphelion and η is the solar arrays efficiency. The mass of the of the arrays is calculated from the surface mass: $m_{SA} = \rho_s S$.

C. Needs for a "hop" on Mars surface

In the current vehicle design, it was decided to perform a 'hop' from the landing site on Mount Olympus Mons, to the refueling base located in Gusev Crater, at a distance of 3 321 km and a difference in altitude of 13 000 m. As it will be later discussed in results, under "The Surface Refuelers", this allowed a considerable reduction in fuel mass needed to be sent to Mars surface and it enables to resupply in water and oxygen at the base located on the Gusev Crater. It is assumed that there is no drag on Mars. It is also assumed that Mars is flat and the propellant required to turn the rocket at the top of the trajectory is neglected. The variation of the acceleration due to gravity with the altitude is also neglected. To perform the hop, the rocket is used and will perform the same trajectory as a ballistic missile. It means that a first burn will give an initial speed to the rocket. At burnout, the rocket is given an angle, here 45 because it enables to optimise the distance the rocket travels during the ballistic trajectory. The rocket is in free fall (parabolic trajectory) until the second burn. Now it is different from the ballistic missile because the rocket has to brake. We assume that both burns are performed with a vertical rocket. In order to calculate the mass of propellant required, the initial and final speed of the parabolic trajectory must be calculated. Firstly, the initial speed is roughly estimated by calculating the initial speed required to reach the proper distance with the angle of 45°. Secondly, the initial speed and the final speed are calculated, as well as the final altitude, with the following equations. The initial speed is modified until an acceptable final altitude is reached (the launching and the landing sites are not at the same altitude).

$$V_{final} = \sqrt{\frac{g_m^2 L^2}{V_{initial}^2 cos(\alpha)^2} - 2gLtan(\alpha) + V_{initial}^2} \tag{7}$$

$$h_{final} = -\frac{g_m L^2}{2V_{initial}^2 cos(\alpha)^2} + Ltan(\alpha) + H \tag{8}$$

where α is the initial angle, 45°, h_{final} is the altitude where the braking starts (it does not take into account the altitude that has been reached during the first burn), and H is the altitude difference between the launching and the landing site. The 0 altitude is taken in the Gusev Crater.

Finally, the mass of propellant is calculated using the rocket equation for the landing and the launch. m_{dry} is the sum of the mass of the capsule, the tank T1 and the tank T2 of Tantalus.

$$V_{final} = -I_{sp} g_e ln(\frac{m_{dry}}{m_{dry} + m_{prop1}}) \tag{9}$$

$$V_{initial} = -I_{sp} g_e ln(\frac{m_{dry} + m_{prop1}}{m_{dry} + m_{prop1} + m_{prop2}}) \tag{10}$$

And finally, the mass of propellant for the hop is :

$$m_{prop} = m_{prop1} + m_{prop2} \tag{11}$$

The figure 1 shows the parabolic trajectory of the rocket (it does not show the two burn). The initial speed is 3 273 m/s and the speed before the second burn is 3 285 m/s. The rocket is going at a really high altitude (850km) but still lower than what a ballistic missile can reach on Earth. The astronaut will endure a maximum acceleration of 6.5g. Since this value is quite high, a proper training is required on Earth in order for the astronauts to sustain such accelerations.

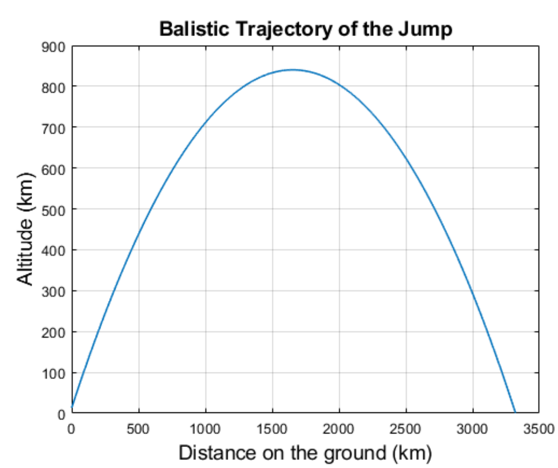

Fig. 1. Ballistic trajectory of the rocket during the hop

D. Refueler design

The fuel m_{prop} needed for the hop has to be brought to Martian surface. It was considered that one of the easiest ways of landing fuel tanks on Mars would be by building these tanks in a manner of actual rockets. Therefore, if accounting for the Structural mass, determined by the value of the structural ratio $\epsilon = 0.05$ we have a final mass m_{refuel} to land on Mars determined as follows:

$$m_{refuel} = m_{prop}\left(1 + \frac{\epsilon}{1-\epsilon}\right) \tag{12}$$

since the structural ratio ϵ defined as

$$\epsilon = \frac{m_s}{m_s + m_p} \tag{13}$$

m_s and m_p being the structural and propellant masses, respectively, of any rocket stage. Unlike the rover and supplies EDL System, there is no need here for such a similar protective extra mass, as the tank itself would possess a shielding around the engines, and the capability of thrusting to perform a retro propulsive landing. However in order to reduce the required propellant, one can consider the use of parachutes to bring the vehicles down to a lower descent velocity. Assuming that we can neglect the mass of parachutes, their area A can be determined by the following equation [18]:

$$A = \frac{2mg}{C_d \rho V^2} \tag{14}$$

where C_d is the parachute drag coefficient (typically 1.75), ρ and g are the density of Mars atmosphere and the gravity of Mars at 20 kilometers altitude, m is the mass to be decelerated and V the equilibrium descent speed.

By posing an arbitrary descent factor K_d multiplying the mass of the fuel tank m_{refuel} to scale it up, one can easily estimate the velocity increment ΔV that can be performed by that new stage with extra propellant, to brake during landing, by using equation 1:

$$\Delta V = I_{sp} g_e \ln \frac{m_{refuel} - (1 - \frac{1}{K_d}) m_{prop}}{m_{refuel}}$$

with m_{prop} the mass of propellant needed for the hop, and m_{refuel} the original total mass of the tank before scaling it up. That being said, equation 14 can be used to dimension the parachutes needed to attain that velocity during descent. In the Results section, a ratio of 1.5 was determined as being reasonable. In the following, the propellant needed for refuel, and the rocket stage housing it, will be referred as Surface Refueler (SR).

E. Sizing the different vehicles

The subsections above addressed the sizing of the key payloads that have to be brought to Mars surface. However, the main vehicle development (for transfer vehicles) still has to be done. From the very beginning, the choice was made to address the two different kind of payloads (Human or mission related, and fuel) separately, i.e. on different transfer vehicles. The vehicles that still need to be sized at this point are: the stages that will carry the Crew Capsule and the Crew Station during all their maneuvers, but also the transfer stages required to bring the rover, supplies and refueler (SR) to LMO.

In order to do that, the different mission phases are studied in a reverse chronological order. Again, one has to keep in mind that all the propelled vehicles are designed to possess the same structural ratio ϵ of 0.05 and specific impulse I_{sp} of 450 s. A Matlab code has been written to do so, and follows the events of the steps bellow:

- Step 1: Bringing the crew back from LMO to the transfer orbit and the ballistic reentry on Earth. Here the code dimensions a transfer stage, that will be called T2, in charge of producing a ΔV that will bring the Crew Capsule and the Crew Station back to Earth. This tank will be located just under the Crew Capsule.
- Step 2: Launching the Crew Capsule and the filled T2 tank from Mars Surface. The code in this section gives us a new sized stage T1 in charge of the Mars launch, positioned under T2. Altogether, these three stages form what is later referred as the Tantalus Crew vehicle.
- Step 3: Determining the amount of propellant needed to land T1, T2 and Crew Capsule (the Tantalus Crew Vehicle) on Mars. This code section determines the amount of fuel needed in the T1 tank to perform the landing on Mars.
- Step 4: Determining the amount of propellant needed to bring the Tantalus Crew Vehicle to LMO. In the results section, one will see that it was decided to include the fuel calculated in Step 3 into an orbital refueler (OR) that will be brought to LMO along with the rover and supply capsule (RSC), in order to limit the amount of propellant computed in Step 4. Therefore this has brought about the need of refueling T1 in LMO before landing with the crew on the surface. This also allows for using the full capacity of existing tank T1 to reach Mars. Being limited by the size of T1, only the amount of propellant needed to perform the second burn LEO and LMO is calculated here.
- Step 5: First Transfer stage for the Tantalus Crew Vehicle. Here, the stage size and amount of propellant needed to perform the fist transfer burn from LEO, and bring the Tantalus Crew Vehicle to LMO, is calculated.
- Step 5: Transfer stage for OR and RSC. Here, the stage size and amount of propellant needed to bring OR and RSC from LEO to LMO is calculated.
- Step 6: Transfer stage for the SR. The transfer stage that is needed to perform the burns between LEO and LMO is sized. Ultimately, it was needed to separate the Surface Refueler into several Surface Refuelers, for feasibility reasons. This separation is discussed under the "Designing a booster to LEO" subsection.

In order to proceed with each of the steps described above, a system of equations has to be solved in Matlab. One can identify two different cases: when the step involves a transfer burn i.e. the burn is collinear with the objects orbit, gravity plays no role in the calculation, and only the following system, derived from 1 has to be solved:

$$M = \exp \frac{-\Delta V}{I_{sp} g_e} \tag{15}$$

$$m_s = m_* \left(\frac{M-1}{1-\frac{M}{\epsilon}} \right) \tag{16}$$

$$m_p = m_s \left(\frac{1-\epsilon}{\epsilon} \right) \tag{17}$$

where ΔV is the velocity increment of the maneuver, given by the Mission-Design group. m_s, m_* and m_p are the structural, payload and propellant masses of the stage under study, respectively. In this case, we can assume that the burnout time is long enough to avoid too great G-forces on the vehicles. In cases where the structural mass of the stage is already known, equation 16 can be disregarded.

On the other hand, when the step involves an ascent or descent manoeuvre on Mars, gravity plays a major role in the calculations. It was decided to dimension the vehicles by using the conservative estimation of a sounding rocket, which implies that the following equations have to be added to the system:

$$\Delta V = \sqrt{\frac{\mu_M}{R_{LMO}}} + g_m t_b \tag{18}$$

$$a_{max} = I_{sp} \frac{\frac{m_p}{t_b}}{m_* + m_s} \tag{19}$$

where R_{LMO} refers to the LMO radius, t_b is the time at burnout, μ_M is Mars standard gravitational parameter and a_{max} is the maximum acceleration desired for the rocket. Equation 18 is derived from the gravity drag and the circular orbit speed. Equation 19 is derived from Newton's second law for propulsion, as shown below:

$$ma = F = \dot{m} I_{sp} g_e \tag{20}$$

with m the mass of the vehicle, F the thrust force, a the acceleration, and $\dot{m}$ the burn rate. Using this equation at burnout allows to fix the final and maximum acceleration.

F. Designing a booster for LEO

In the previous part was described the method to design the vehicles, from LEO to the end of the mission. It is thus needed to size a launcher capable of sending all the vehicles into orbit around Earth. For simplicity reasons, and in a development cost reduction perspective, it was chosen to design a single model booster capable of performing all the launches. Its design was inspired by a Super Heavy booster [19] from SpaceX, with modified engines to match the specific impulse defined in this study, and extrapolated structural ratio, as this parameter has not been published by the company.

An iterative process was carried out using equations 15 to 19, in order to size the second stage to the booster, with the purpose of reaching the maximum payload mass m^*_{max} allowance to LEO. The booster and its second stage will be referred as by Super Heavy Evolved Booster, or SHEB. One could see by running the code the total Surface Refueler mass with its transfer stage exceeded the value of m^*_{max}. Therefore, it was decided to have two smaller Surface Refuelers, which fit with a reasonable margin on top of the SHEB.

G. Validation by simulation

In order to validate the estimations done by the sounding rocket equations during landing and launching phases, one can simulate the trajectories by implementing the following launcher dynamic equations written in the Local Vertical Local Horizontal frame, in Matlab:

$$\dot{V} = \frac{\beta I_{sp} g_e}{m} - \frac{\frac{1}{2}\rho S C_d V^2}{m} - g \sin\gamma \tag{21}$$

$$\dot{X} = V \cos\gamma \tag{22}$$

$$\dot{H} = \sin\gamma \tag{23}$$

$$\dot{m} = -\beta \tag{24}$$

where the derivative of the state vector $[V, X, H, m]$, for speed, ground distance, altitude and mass, forms the left side of the system. Furthermore, β is the mass flow, i.e. $\frac{m_s}{t_b}$, which can be calculated via equation 19, ρ the atmosphere density, S the cross section area of the vehicle, C_d its drag coefficient, and γ the angle of the vehicle, defined by a steering law written as:

$$\gamma(t) = \arctan\left(\tan(\gamma_0) \left[1 - \left(\frac{t}{t_b} \right)^{(1/n)} \right] \right)$$

Here, γ_0 is the initial angle of the simulation (90 deg is horizontal), t_b is the burnout time, and the n parameter can be modified to change the steepness of the trajectory. Typically, values between 15 and 20 were used for the trajectories in the results section.

The system can be solved in Matlab using the ode45 solver [20]. Several time variables have been implemented to consider engine cutoffs, and an iterative process was conducted to optimize them and reach the expected altitude and speed at burnout. The results of these simulations will be exposed in the following section.

III. Results

The results exposed below are the product of the methods described above, coupled to the inputs given by the Human aspects [2], Mars Operations [3] and Mission Design [1] groups. They were obtained in part by running a Matlab script, and needed some manual iterations in order to comply with all the requirements. Although it was intended to generate plausible renders, the physical appearance is not guaranteed to be of scientific value. That being said, the shapes do represent fairly accurately the calculated mass ratios.

A. The Tantalus Crew Station

In Figure 2 one can clearly see the different parts of the Crew Station as explained in the Methods section. And in Figure 3 the cross section of one of the Crew Modules is presented. Note that the central metallic structure can not be

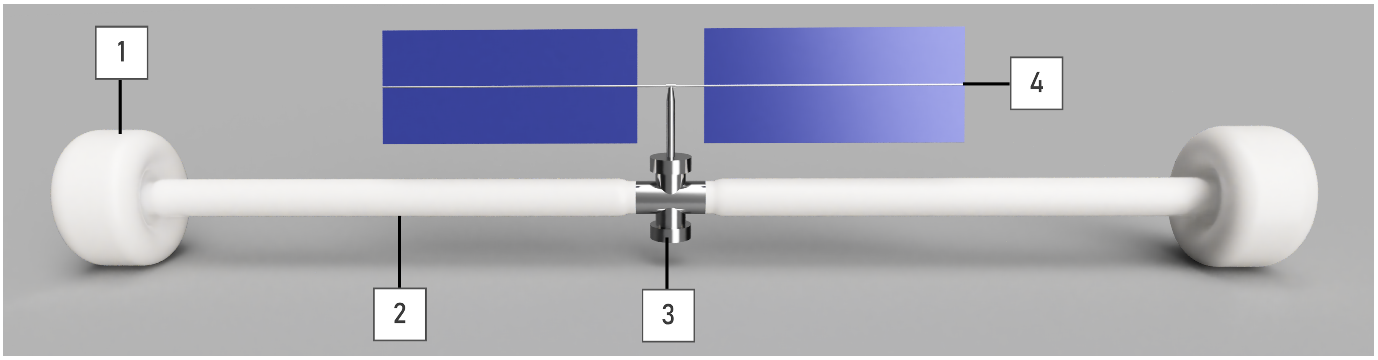

Fig. 2. Tantalus Crew Station. (1) Crew Module. (2) Connecting arms. (3) Central metallic core with de-spun platform. (4) Solar Panels, allocated directly on the platform.

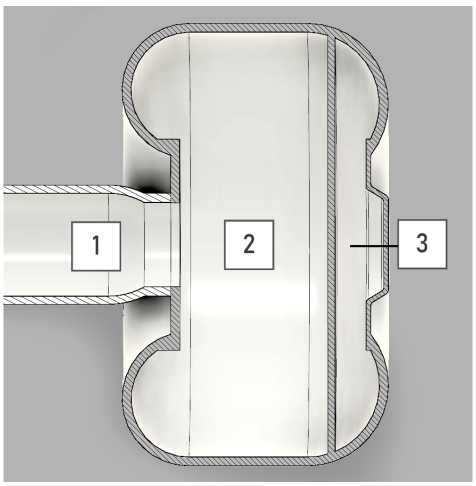

Fig. 3. Cross section view of the Crew Modules. (1) interconnetion between the arms and the modules. (2) Main floor, i.e. the habitat where the astronauts will spend most of their time (3) service floor, i.e. where some vital systems (such as the deployable systems) are allocated.

TABLE I
GENERAL SIZING OF THE CREW STATION

Volumes overview for the Crew Station [m^3]	
Total pressurized volume	150
Total habitable volume	45
Habitable volume per astronaut	15
General Dimensions [m]	
Length of the arms	18
Length of the central metallic module	3
Height of the Crew Module	3
Dimensions of the Crew Modules [m]	
Diameter of the central metallic truss	2
Non-inflated diameter	2
Inflated Diameter	5

seen in this picture. As already explained in Methods section the total length of the Crew Station is 45 m. In the Table I, the main features of the Crew Station in terms of sizing are presented.

The total heights of the Main floor and Service floor are 2.5 m and 0.5 m respectively. The diameter of the central metallic truss for the arm is 1 m and the inflated diameter for the arm is 1.5 m. The central metallic core is expected to have a diameter of 2 m. The Mass breakdown structure can be found in the Table II. The symbol 'HA' means that the value has been provided by the Human Aspect team. The method used to compute the structural mass of the structure has been explained in the Methods section.

TABLE II
MASS BREAKDOWN STRUCTURE

Subsystem	Mass
Structural Mass for the whole Crew Station	6 044 kg
Medical equipment and safety (HA)	42 kg
Food (HA)	1 380 kg
Air (HA)	1 230 kg
Water (HA)	700 kg
Water+Air system (HA)	4 630 kg
Thermal regulation system (HA)	700 kg
Avionics	370 kg
Power	1 710 kg
Total Expected Mass	**18 000 kg**

The pressurized volume value in ISS is 916 m^3 and its power generation is between 75 and 90 kW [21], while the pressurized volume in the Crew Station equals 150 m^3. Therefore the maximum power required by the spacecraft is around 15 kW. A safety margin of 5 kW is taken, Thus the power required being equal to $P_{req} = 20\ kW$. By taking $F_s = 63.2\ MW/m^2$, $R_s = 695600\ km$, $a_M = 249.2.10^6\ km$ and $\eta = 60\%$ in 6 a surface of 67.7 m^2 is required for the solar arrays. The surface mass of the solar arrays $\rho_s = 3\ kg/m^2$ is taken to have a safety margin of 1 kg/m^2 with regards to current values [22], leading to a mass of 203.1 kg for the solar arrays.

B. A brief overview of the vehicles

In Figure 4, one can see the 4 launchers configured by the Space Vehicle team, for the purpose of this mission. The first launcher on the left being the heart of the project, the Tantalus Crew Vehicle. On its right side, one can see the launcher showing the EDL system capsule containing the martian rover and supplies, sitting above the Orbital Refueler (OR) and the transfer stage. Finally, the two last launchers are mounted with the Mars Surface Refuelers (SR), each one on their transfer

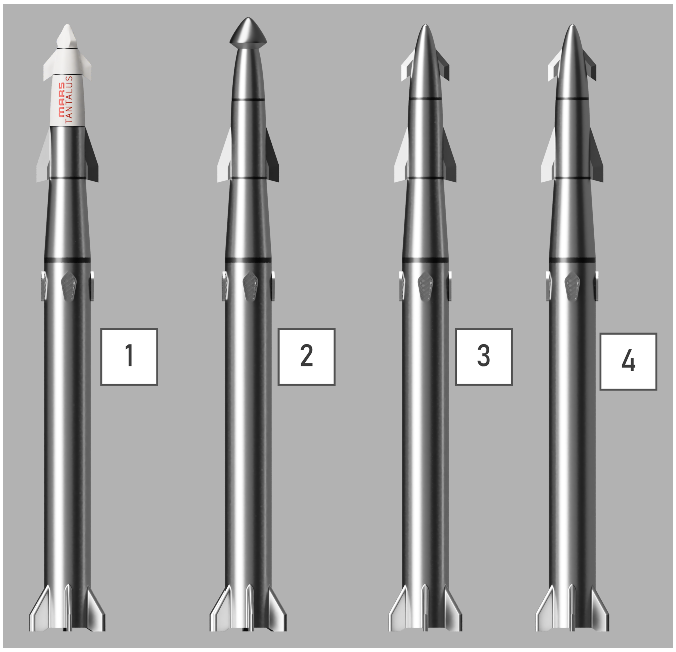

Fig. 4. Tantalus Launchers. From left to right, launchers composed of SHEB mounted with the Tantalus Crew (1), rover and supplies EDL capsule (2) and the two Surface Refuelers (3) and (4).

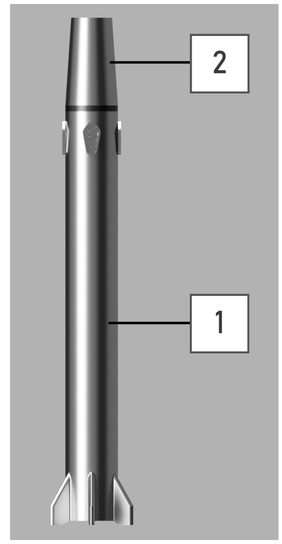

Fig. 5. Super Heavy Evolved Booster, showing the modified Super Heavy booster (1) and its additional second stage (2).

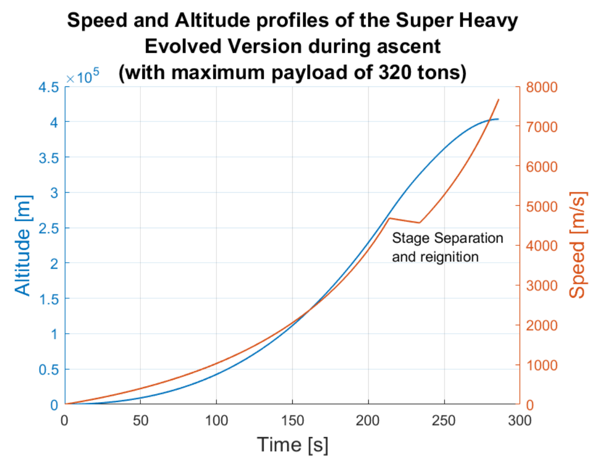

Fig. 6. Launch profile Super Heavy Evolved Booster

stage, that will be land on Mars to refuel the crew vehicle. An extensive description of the mission logistics and timeline is exposed in the Mission-Design report [1].

While there is still the need for a launcher to send the Crew Station to LEO before docking with the Crew Capsule, it is not portrayed in this study. This is because it does not pose a designing challenge in terms of mass capabilities for an existing launchers such as Ariane 5 ECA and 6, Delta IV Heavy, Falcon 9 full Thrust, and much more, all capable of carrying a payload of 20 tons and more [23].

C. The Super Heavy Evolved booster (SHEB)

All the above vehicles share the same launch system of which the design was discussed in the Method section. One can see the vehicles on Figure 5. The calculated technical characteristics and masses, can be seen in Table III. Both stage run on methane/LOX, and are assumed to have an Isp of 450 s and a structural ratio ϵ of 0.05. The maximum payload capacity computed for a launch to LEO orbit, at 400 km altitude, is of 320 tons.

TABLE III
PROPERTIES OF THE SUPER HEAVY EVOLVED BOOSTER.

Stage	Dry mass [kg]	Propellant capacity [kg]
Stage 1	157 850	3 400 000
Stage 2	19 474	370 000

With that maximum payload, the simulated launch profile can be seen in Figure 6. As this study focuses on a human mission, an important aspect of the booster is that it should be human rated. The G-forces profile of the booster is shown in Figure 6. As one can notice, the maximum acceleration is computed to reach around 10g. This due to the need of minimizing the launch duration, to optimize the attainable payload mass. It will be shown later that the crew vehicle consists in a mass of only around 60 % of the maximum 320 tons payload. Therefore, one should expect much lower G-forces during the launch.

Anyhow, it could be argued that such G-forces constitute a less damaging factor for the crew while leaving Earth than after a 6-month-long travel in a reduced gravity field. Furthermore, all the simulations where done with a constant thrust for the booster, far below the maximum thrust allowed by the vehicle according to Space X [19]. One could expect improving the performances by refining the flight profile.

D. The Tantalus Crew vehicle

The obtained final design for the crew vehicle can be seen in Figure 8. Sitting on top of the three tanks (transfer, T1

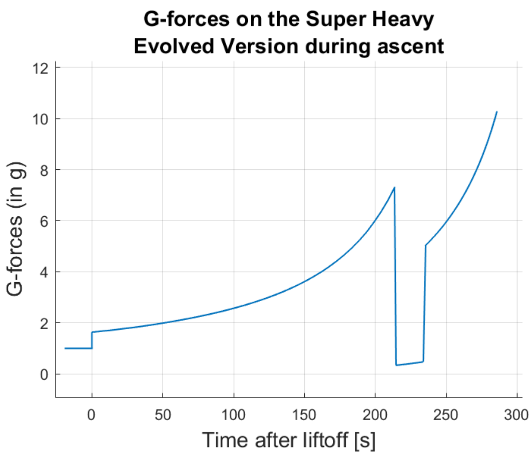

Fig. 7. G-loads during Super Heavy Evolved Booster launch

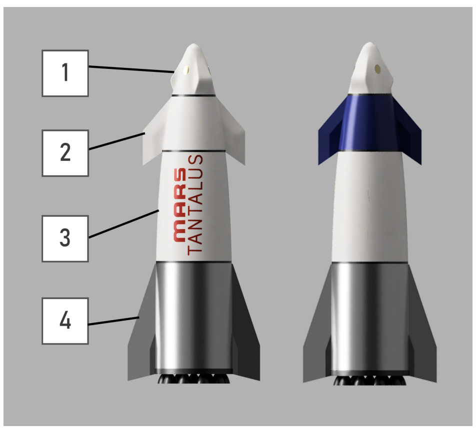

Fig. 8. Tantalus Crew Vehicle front (left) and back (right) faces, on its transfer stage. The Crew Capsule (1) is tacked on the T2 (2), T1 (3) and its transfer stage (4).

and T2) one can find the Crew Capsule, adapted from the Crew Dragon capsule. It possesses a total mass of 13 500 kg. This mass includes the 2 000 kg needed for an average of $10m^2$ of Lithium-Hydride shielding [2]. The total mass of the Crew Dragon being estimated around 13 tons [5], it also accounts for the trunk, which is not present in the current case. Furthermore, one could expect some technical improvements before the year 2038 that would allow for a lower mass. Therefore, all in all, one can consider this approximation as being realistic.

One can also notice the presence of solar panels on the T2 tank. These supply the cabin with power during most of the mission where the cabin is not docked to the Crew Station, leaving place for the cabin batteries during the descent to Earth, after the mission completion. Indeed, the T2 tank is detached before the ballistic reentry. Their surface is estimated as being the same as what is present on the Crew Dragon trunk.

Table IV summarizes the masses of each section of the vehicle throughout the total mission duration, from its first thrust maneuver (transfer from LEO to LMO) to its final one (transfer from LMO to LEO). Again, all the stages are propelled by Methane/LOX and possess an I_{sp} of 450 s and

TABLE IV
TANTALUS CREW VEHICLE CONFIGURATIONS THROUGHOUT THE MISSION

Stage	Dry mass [kg]	Initial propellant mass [kg]	Final propellant mass [kg]
First transfer burn from LEO to LMO (4 005 m/s)			
Transfer Stage	6 678	126 880	0
T1 tank	5 962	39 139	39 139
T2 tank	2 761	0	0
Crew Capsule	13 500	-	-
Crew Station	18 000	-	-
Second transfer burn from LEO to LMO* (3 054 m/s)			
T1 tank	5 962	39 139	0
T2 tank	2 761	0	0
Crew Capsule	13 500	-	-
Crew Station	18 000	-	-
Mars landing**			
T1 tank	5 962	31 705	0
T2tank	2 761	0	0
Crew Capsule	13 500	-	-
Mars hop***			
T1 tank	5 962	100 000	0
T2 tank	2 761	0	0
Crew Capsule	13 500	-	-
Mars launch *****			
T1 tank	5 962	113 280	0
T2 tank	2 761	52 467	52 467
Crew Capsule	13 500	-	-
Return burn (4 121 m/s) *******			
T2 tank	2 761	52 467	0
Crew Capsule	13 500	-	-
Crew Station	18 000	-	-

* Transfer stage is separated.
** T1 tank is refueled by the OR and Crew Station is undocked in LMO.
*** T1 tank is refueled on Mars by the two SR.
**** T1 and T2 are refueled in Gusev Crater before return trip.
*****T1 is separated, and the Crew Station is docked in LMO.

an ϵ of 0.05. The total payload to LEO being estimated to 190 tons, it is far from the limit imposed by the booster, allowing for a lower G-force profile, as mentioned before.

An overview of the simulated trajectory of the vehicle landing on Mars can be seen on Figure 9. This was simulated as a reverse launch, therefore being based on the concept of retro-propulsive landing. The associated flight profiles in terms of speed and altitude in function of the time after liftoff can be seen in Figure 10. One can see that the vehicle was able to reach a null velocity and altitude with some precision, final values being 232 m of altitude, and 42 m/s of final velocity. Considering the precision of all the characterization process, these are acceptable levels of error. Figure 11 shows the G-forces of the vehicle during that landing, that stay under the value of 4.5 g. Indeed, it was decided to limit the acceleration value to an additional 4 g to the local gravity field, in order to preserve the integrity of the crew on Mars.

In Figures 12, 13 and 14, one can see the characteristics of the simulation for the vehicles during the launch from Mars. Again, an important aspect is the reached altitude and speed, and above all, the G-forces that are maintained under 4.5 g.

E. The Tantalus rover + supplies vehicle

With regard to the rover and supplies vehicle design, it is detailed in Figure 15. The characteristics of the different stages are exposed in detail in Table V.

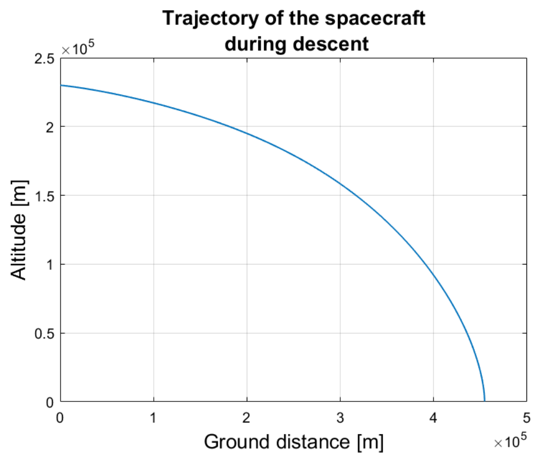

Fig. 9. Landing trajectory of the Tantalus Crew Vehicle

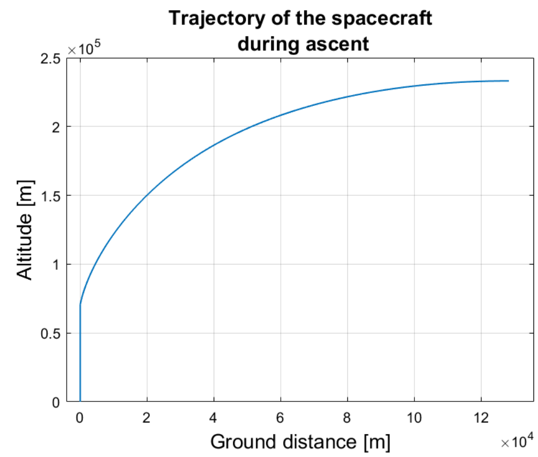

Fig. 12. Launch trajectory of the Tantalus Crew Vehicle

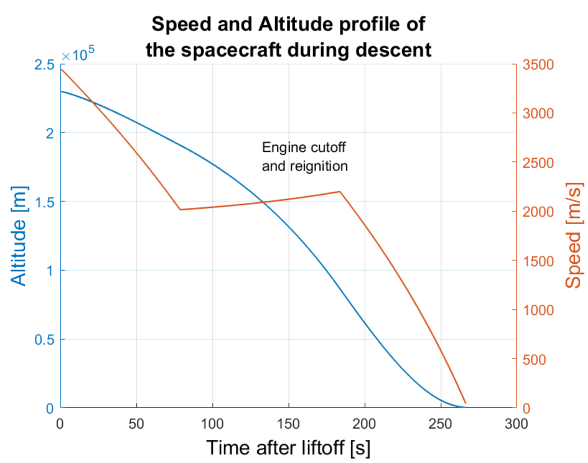

Fig. 10. Landing profile of the Tantalus Crew Vehicle

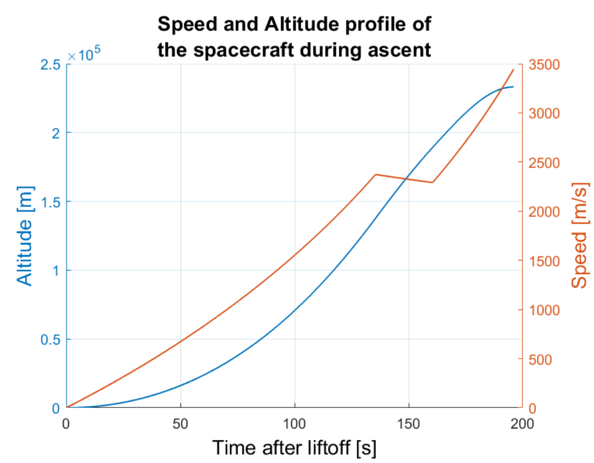

Fig. 13. Launch profile of the Tantalus Crew Vehicle

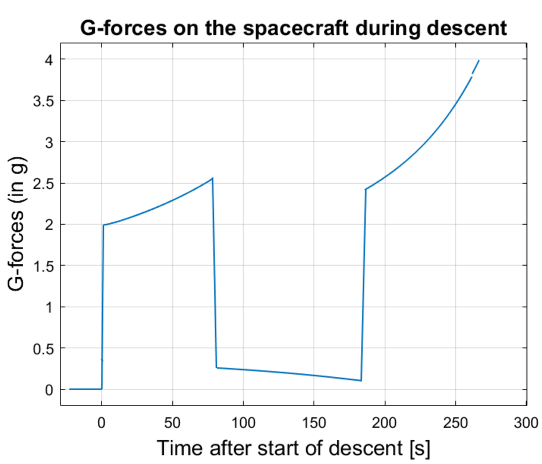

Fig. 11. G-loads during Mars landing

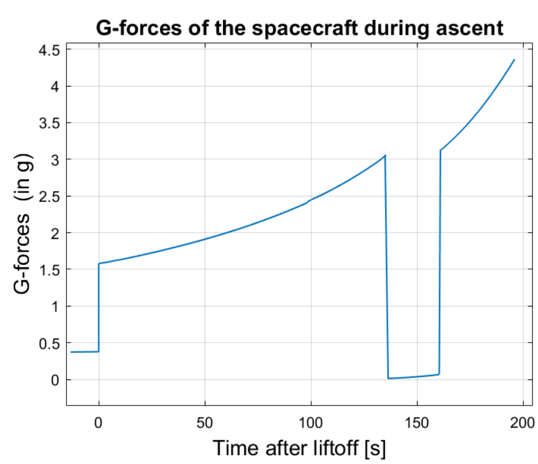

Fig. 14. G-loads during Mars launch

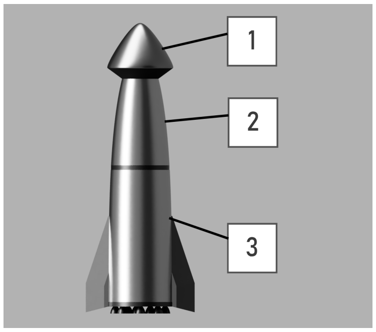

Fig. 15. Tantalus Rover and Supplies Capsule (1) stacked on the Orbital Refueler (2) and its transfer stage (3).

TABLE V
THE TANTALUS ROVER AND SUPPLIES VEHICLE CONFIGURATION

Stage	Dry mass [kg]	Initial propellant mass [kg]	Final propellant mass [kg]
Transfer from LEO to LMO (5 756 m/s)			
Transfer stage	7 796	143 290	0
Orbital refueler	1 668	31 705	31 705
EDL System (rover + supplies)	15 000	-	-

Here, the total mass allowance for the Rover is set to 2 200 kg and the one for supplies to be brought on Mars is 3 800 kg. Together, they form the capsule payload for a total mass of 15 tons. The total mass of the capsule, accounting for the EDL system, is obtained by multiplying the payload mass by a factor 2.5, similar to what it is for the Curiosity Rover [6]. In the same fashion, it therefore includes a heat shield, parachutes, thrusters and a sky crane. No further research was done in the sizing of the capsule, as it was assumed to lead to a reasonable estimation.

F. The Surface Refuelers

Lastly, the Figure 16 shows the final design of the two refuelers. Their detailed mass characteristics during different mission phases can be seen in Table VI. Here, it was assumed that an extra 50 % of propellant would be added in the refuelers (resulting in an increase of structural mass) in order to allow them to land. Therefore, this gives an available $\Delta V_{landing}$ of 1 679 m/s. Note that the total mass of propellant needed to refuel the crew vehicle before the "jump" is of 100

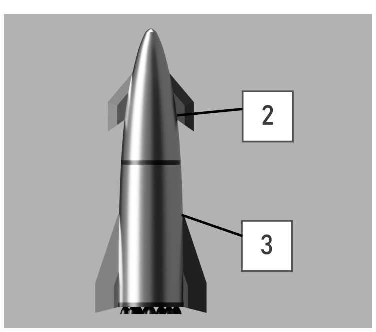

Fig. 16. A Tantalus Mars Refueler (2) on its transfer stage (1).

TABLE VI
THE SURFACE REFUELERS CONFIGURATION THROUGHOUT THE MISSION

Stage	Dry mass [kg]	Initial propellant mass [kg]	Final propellant mass [kg]
Transfer from LEO to LMO (5 756 m/s)			
Transfer stage	12 724	223 920	0
Surface refueler	3 947	75 000	75 000
Mars landing			
Surface refueler	3 947	75 000	50 000

tons. As mentioned in the Method section, they need to be landed next to the crew landing spot, at the Olympus Mons ascension start, at 11 km altitude [3]. Here, each vehicle thus brings 50 tons of propellant as payload as a single vehicle would have represented too great of a mass to be launched to LEO.

The use of these refuelers and the hop is justified when one follows the following reasoning: the crew vehicle has to be refueled, as it emptied its tank T1 during descent, and T2 was already empty. As one can see in table IV with the propellant masses actually in use during the hop, proceeding with two hops with only one refueling is clearly not an option, as it would more than double the needed fuel in T1. This forbids the scenario where the rocket lands at Gusev Crater base for refueling and does a round trip to Olympus Mons. Furthermore, one could argue that the mission could have saved a launch and a landing, by simply leaving Mars from the landing site on Olympus Mons, with the extra propellant from the refuelers. However the numbers do not match. More than 13 tons would have missed in T1 for takeoff and 50 more tons of propellant would have been needed in LMO to refuel T2. The current mass to LEO for the SR vehicles being already 315 tons, more vehicles would have been needed.

Equation 14 allows to calculate the total area of the parachutes needed to slow the vehicle to the $V_{landing}$ speed, assuming an atmosphere density of 4.3×10^{-3} kg/m^{-3} at 15 km altitude, and a parachute drag coefficient C_d of 1.75 [24]. Furthermore, the total mass of the parachute system is assumed to be around 1 ton. This gives an area value of around 30 m^2. The surface seems rather small, however, the speed to reach with it is far from being a landing velocity. The detailed aspects of such a supersonic descent parachute will not be studied further.

IV. DISCUSSIONS

A. Assumptions

One could argue that some of the assumptions used in this study are somehow overly strong. While the maximal theoretical value of I_{sp} for methane is 458.7 s [25], it is unsure whether we might be able to reach a value as high as 450 s in the near future. Furthermore, no difference has been implemented concerning sea level or vacuum values of I_{sp}, which could lead to a loss of capabilities while launching the vehicles from Earth. However, the use of some kind of solid boosters on the main stages could come as a fairly simple solution. Currently, with their current technology on the Raptor engine [26], Space X engineers aim for a 380 s vacuum I_{sp}. Regarding the disregarding of drag on Mars, its significantly lower density [24] makes that claim acceptable, although one must account for the need in thermal protection. That thermal protection was assumed to be included in the structural masses of every stage. Although one might argue that 0.05 is a very low value of structural ratio, this value corresponds to Saturn V main stage, a rocket from 50 years ago, as seen in table 10.3 of Hill, Peterson [27]. Concerning the solar panels 60 % efficiency, some research show promising new technology which can achieve this value [28].

B. Risk estimation and Off-nominal scenarios

A non-exhaustive list of potential off-nominal scenarios is discussed below:

- Failure during a launch or landing : If the launch is crewed, all of the crew vehicle stages are capable of launch abort, in the fashion of the Soyuz abort modes [29]. This could also be used during landing. The main consequences are material and financial, and can delay the mission. If the failure occurs on Mars, it will require a rescue mission for the crew to be launched from Earth as they only have extra supply for a limited time.
- Depressurization of the inflatable modules: a hatch located at the base of each boom would allow to isolate the deteriorated module.
- Failure during docking : very unlikely, as the technology is robust. In the worst case in which this failure occurs in LMO, the best option would be to perform an EVA and load as much supplies as possible from the Station to the Capsule, in order to attempt a return to Earth in the Capsule.
- Failure of the spacecraft during the travel : if there is a problem the crew can not solve during the interplanetary journey, it would mean the failure of the mission and the crew would be lost.

V. Conclusion

The Tantalus space vehicles have been entirely designed to comply with the objectives of the mission, and undergo some of the off-nominal scenarios identified during the mission. The sizing method is based on dynamic rocket equations and basic rocket performance theory. The Tantalus Crew Station, core of the project, has been designed to generate artificial gravity thanks to an inflatable space station assembled in LEO. Every other vehicle stage was dimensioned in a descending way, to carry the payloads above. Four Launchers have been assembled on a modified version of the Super Heavy booster. The three first send propellant tanks and the EDL system carrying supplies and the rover. The last one carries the crew to its journey across the solar system. One can clearly see that the needs in propellant for this mission to and on Mars, and the high payload masses, lead to a skyrocketing mass budget. However, as most aspects have been dealt in the study with no further assumptions on fuel availability apart from the Gusev Crater base, one should not expect any hidden expenses. In further studies, one could also think of an alternative propulsion solution, using the promising low thrust electric *VASIMR* engine, which can vary its I_{sp} between 1 000 and 30 000 s, with a thrust up to 500 N. Some research show that it could one day perform a manned trip to mars in 39 days with a sufficient power source (advanced nuclear reactor) [30].

References

[1] Jingyang Wu, Guillaume Trimoreau, Aleksander Kipiela, Max Bergström, and Antonio D'Anniballe. Mount olympus mons ascension mission - mission design - team red. March 2021.
[2] Louise Fischer, Andrea Mussita, Florian Steiner, and Hanlin GongZhang. Mount olympus mons ascension mission - human aspect - team red. March 2021.
[3] Filip Malmborg, Oscar Andersson, Théo Grimonprez, Benoît Logiou, and Adam Parks. Mount olympus mons ascension mission - mars operation - team red. March 2021.
[4] NASA Glenn Research Center. Rocket mass ratios, 2014.
[5] Wikipedia. Crew dragon.
[6] Wikipedia. Mars science laboratory.
[7] James Engle, Raju Dharmaraj, and Torin K. Clark. Artificial gravity for low earth orbit (iss) and deep space exploration, 2016.
[8] Mark G. Benton, Bernard Kutter, Ruth A. Bamford, Bob Bingham, Tom Todd, and Robin Stafford-Allen. Modular space vehicle architecture for human exploration of mars using artificial gravity and mini-magnetosphere crew radiation shield, 2012.
[9] Alexander Mazarr Samuel Park Phillip A. Williams Nicholas M. Houghton, Joseph Fulton. Utilizing in-space assembly to add artificial gravity capabilities to mars exploration mission vehicles, 2020.
[10] Horacio de la Fuente, Jasen L. Raboin, Gary R. Spexarth, and Gerard D. Valle. Transhab: Nasa's large-scale inflatable spacecraft, 2000.
[11] Nikhil Agarwal, Wahab Alshahin, Eric Anden, Jordan Holquist, Robert Lozar-McDonald, David Slaby, and Ryan Smoot. Conceptual design and architecture for a deep space habitation module, 2012.
[12] Akshay Prasad, Joseph Adinolfi, Brian W. Evans, , and Matthew A. Simon. Conceptual design and analysis of a deep space habitat with a venus flyby, 2020.
[13] Jian-Ming Chang Justin Kanga David L. Akin, Benjamin Nichols and Lemuel Carpenter. Design and testing of an inflatable airlock concept, 2016.
[14] Douglas A. Litteken1. Evaluation of strain measurement devices for inflatable structures, 2017.
[15] Minimum acceptable net habitable volume for long-duration exploration missions, 2020.
[16] Lemuel D. Carpenter and David L. Akin. Vanguard: A common habitable module for future space endeavors, 2017.
[17] Bigelow expandable activity module.
[18] NASA Glenn Research Center. Velocity during recovery.
[19] Space X website. Starship, 2021.
[20] MathWorks website. ode45 solver: Solve nonstiff differential equations - medium order method.
[21] NASA. International space station facts and figures. Online, January 2021. Accessed 16 February 2021.
[22] Spectrolab. *Space Solar Panels*, 2010.
[23] Wikipedia. Comparison of orbital launch systems.
[24] NASA Glenn Research Center. Mars atmosphere model.
[25] Tom Myers. The upper limit of specific impulse for various rocket fuels. 2016.
[26] Wikipedia. Spacex raptor. Online.
[27] Philip Hill and Carl Peterson. *Mechanics and Thermodynamics of Propulsion, second edition*, chapter 10.4. Pearson, 1992.
[28] Daniel C. Lawa, R. R. King, H. Yoon, M. J. Archer, A. Boca, C. M. Fetzer, S. Mesropian, T. Isshiki, M. Haddad, K. M. Edmondson, D. Bhusari, J. Yena, R. A. Sherif, H. A. Atwater, and N. H. Karam. Future technology pathways of terrestrial multijunction solar cells for concentrator photovoltaic systems. *Solar Energy Materials & Solar Cells*, (94):1313–1318, 2010.
[29] Wikipedia. Soyuz abort modes. Online.
[30] Andrew V. Ilin, Leonard D. Cassady, Tim W. Glover, and Franklin R. Chang Diaz. Vasimr® human mission to mars. Ad Astra Rocket Company, 2011.