Hephaestus: Space Vehicle Conceptual Design
Blue team

March 20, 2021

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***Abstract*—A hypothetical contest has been issued to summit Olympus Mons. This report details a conceptual design of a space vehicle intended for that purpose. Functional requirements of the vehicle were defined, and the primary subsystems identified. The SpaceX Starship was chosen to serve as a design baseline and a comparison of suitable launch vehicles was made. Subsystems such as propulsion, reaction control, power generation, thermal management and radiation shielding were considered, as well as interior design of the space vehicle and its protection measures for Mars entry and activities. The consequences of an engine-out scenario were studied.**

The Starship launch system was deemed the most suitable launch vehicle candidate. Six Raptor engines were selected for the spacecraft main propulsion system, with 8 smaller thrusters for reaction control. A radiator mass of 890 kg was estimated for the thermal management system, and lithium metal hydride was determined an adequate radiation shielding material for the crew quarters. A radiation shelter was integrated into the pantry, and layouts of the crew quarters and cargo bay were prepared. Aerobraking and utilization of a heat shield was suggested for Mars atmospheric entry, and propulsive landing on the surface deemed the most feasible for the massive vehicle. An additional engine was found to be a sufficient mitigation strategy for the off-nominal scenario. It was concluded that the design presented met all of the requirements.

***Index Terms*—Mars, Interplanetary, Design, Propulsion, RCS, Radiation, Starship, Ballistic entry, Landing, Crew, Olympus Mons, Mount Olympus, Spacecraft, Space Vehicle**

NOMENCLATURE

β	Ballistic coefficient
ΔV	Velocity increment
$\dot{Q}_{crew}$	Heat emission of crew
ϵ	Surface emissivity factor
γ	Heat capacity ratio
$\mathfrak{R}$	Mass ratio of m_0 and m_f
σ	Stefan-Boltzmann constant
A_r	Radiator surface area
C_d	Drag coefficient
c_p	Specific heat
g_e	Gravitational acceleration at Earth sea level
g_m	Gravitational acceleration at Mars sea level
H	Equivalent Dose
I_{sp}	Specific impulse
m^*	Payload mass
m_0	Initial mass
m_f	Final mass
m_p	Propellant mass
m_s	Structural mass
P	Power
p_{02}	Combustion chamber pressure
p_e	Exhaust gas exit pressure
S	Drag surface
T	Thrust
u_e	Exhaust gas exit velocity
W	Weight
t	Metric tonne

I. INTRODUCTION

In the year 2038 a challenge was issued for a team of pioneers to be the first to reach the peak of Olympus Mons, the highest mountain in the solar system. The reward for the first to accomplish this task is 100 million USD and enough glory to make even Edmund Hillary and Tenzig Norgay envious. The challenge must be completed through human effort and the summit push has to be made on foot. Land vehicles are only allowed up to one km below the peak and flying vehicles are allowed up to 10 km below. To accomplish this task, a mission must be designed that can transport a group of explorers to Mars, land them safely on the surface, provide them with everything they need to reach the peak and their journey back to Earth. The mission can use three established, unmanned bases on Mars. These bases have automated production of water, oxygen and methane, which can be utilized for the mission.

Two teams of aspiring explorers have taken on the challenge, team Red and team Blue. Each team is subdivided into five groups in charge of different aspects of their respective mission. These aspects are Management, Mission design, Human aspects, Space vehicles and Mars operations. Both teams have concluded the initial concept design and presented their proposed Olympus Mons missions. Team Blue has named their mission concept Hephaestus. This report will detail the vehicle design concept developed by the Space vehicles group. The report is divided into the two main *Method* and *Results* part. In *Method* the thought process and reasoning is described for the design of each subsystem and in *Results* the final design is described. This is followed by a study of an off-nominal scenario, discussion section, the conclusion and finally a list of references. The scope of this project is that of a conceptual design study and therefore all results and conclusions made are preliminary. Assumptions made are stated throughout the report.

II. Methods

A. Requirements

For the Hephaestus mission to be successful, a plethora of requirements on the space vehicle must be met. These requirements were primarily a result of careful analysis by the Mission Design, Human Aspects and Mars Operations groups, and they define the necessary resources for the mission and minimum capabilities of the space vehicle. The primary functional requirements of the space vehicle were determined to be the following:

- accommodation designed for a crew of 6;
- storage facility for 2 Mars rovers;
- total payload capacity of 60 metric tons, or 30 tons per vehicle;
- velocity increment, or ΔV, capability of 8.9 km/s to be able to complete all flights required by the mission architecture;
- minimum habitable volume of 25 m^3 per person;
- mission radiation dosage of less than 1 Sv per person;
- high level of robustness and redundancy in mission critical systems.

Other more qualitative requirements that were agreed upon include:

- ability to land on a specified location on Mars;
- ability to perform sub-orbital "hops" on Mars;
- ability to deploy equipment on the surface of Mars;
- main propulsion system that uses methane for fuel.

All relevant components of the space vehicle will be designed such that these requirements are met. The systems and the requirements they fulfill are summarized in Figure 1.

B. Candidate launch systems

With a payload requirement of 30 t per vehicle and a habitat volume minimum requirement of 150 m^3 for the crew of six, it was clear a very large spacecraft design was needed in terms of both mass and volume. Before design work could commence, an Earth launch system had to be selected. The launch system determines the upper bounds of size and mass for the spacecraft. The five different systems that were initially under consideration are NASA's Space Launch System (SLS), Blue Origin's New Glenn, SpaceX' Starship, the Chinese Long March 9 and the SpaceX Falcon Heavy. With the exception of Falcon Heavy, all of the above are under development and thus not flight proven, though the SLS utilizes a lot of flight proven hardware such as the Space Shuttle Main Engine and Solid rocket boosters [1]. Currently flying launchers were, with the exception of Falcon heavy, considered too small for the Hephaestus mission.

With so many heavy launch systems in advance stages of development, it was assumed that a few heavy lift options would be available by 2038. Long March 9 is a rocket developed by the Chinese Space Agency (CSNA) and is intended for crewed lunar missions in the 2030s. It will use kerosene/LOX bi-propellant and has a targeted LEO payload capability of 140 t. Its target date for maiden launch is 2030. [2]. This rocket could potentially be used for the Hephaestus mission but since not many details are currently known and it being in a relative early development stage, the decision was made to exclude it as an option for primary launch system. Falcon Heavy was also excluded due to its relatively low LEO payload capability and small payload fairing size. This left three primary candidates. Table I compares important specifications of these three systems. All three launch systems

Table I: Heavy lift launch systems [1] [3] [4]

	SLS block 2	New Glenn	Starship
Payload mass to LEO	130 t	45 t*	100 t*
Fuel	H_2 + SRB	CH_4 + H_2	CH_4
Oxidizer	LOX	LOX	LOX
Oxidizer:Fuel mass ratio	2.7:1	3.7:1	3.7:1
Number of stages	3	2	2
Fairing diameter	8 m	7 m	9 m
Fairing height	18 m	12 m	18 m
Fairing Volume	900 m^3	460 m^3	1100 m^3
Expected availability	2022	2022	2022
Reusability	No	Yes	Yes
Human rated	Yes	Unknown	Yes

∗ Reusable configuration

are far along in their respective development, with some subsystems complete and hardware tests being conducted. Starship prototypes have completed short test flights and the SLS core stage recently completed a full duration static test fire [5]. A factor that is unknown at this time is the prices of these systems, although SpaceX and Blue Origin have declared intentions of making their launchers reusable in order to greatly reduce the cost per launch [4] [3]. NASA has no such intentions for the SLS.

C. Overall Design

The spacecraft design is a modified version of the SpaceX Starship. Starship is a heavy launch system consisting of two stages currently under development. The first stage is named Super Heavy. It is a large rocket booster powered by liquid Methane/LOX bi-propellant and is intended to be very quickly reusable, perhaps capable of multiple launches per day. The second stage is the spacecraft, which is also named Starship. A schematic of the spacecraft is shown in Figure 2. Both stages use the Raptor engine, also under development by SpaceX. They are intended to have propulsive landing capability, similar to the Falcon 9 first stage and Blue Origin's New Shepard. If completed to current specifications (the figures have been subject to change during development), Starship will be truly massive [3]. With a diameter of 9 m, a stack height of 120 m and a liftoff mass of approximately 4800 t, it will outsize even the Saturn V, which is currently the largest man-made object to have ever flown [7].

The purpose of Starship, as stated by SpaceX, is to be "...a fully reusable transportation system designed to carry both crew and cargo to Earth orbit, the Moon, Mars and beyond"[3]. It has a targeted LEO payload capability of 100 t and the payload section is 18 m long with an internal volume of 1100 m^3. For missions beyond Earth orbit the spacecraft is intended to utilize orbital refueling. By sending tanker versions of Starship with propellant to refuel the outbound

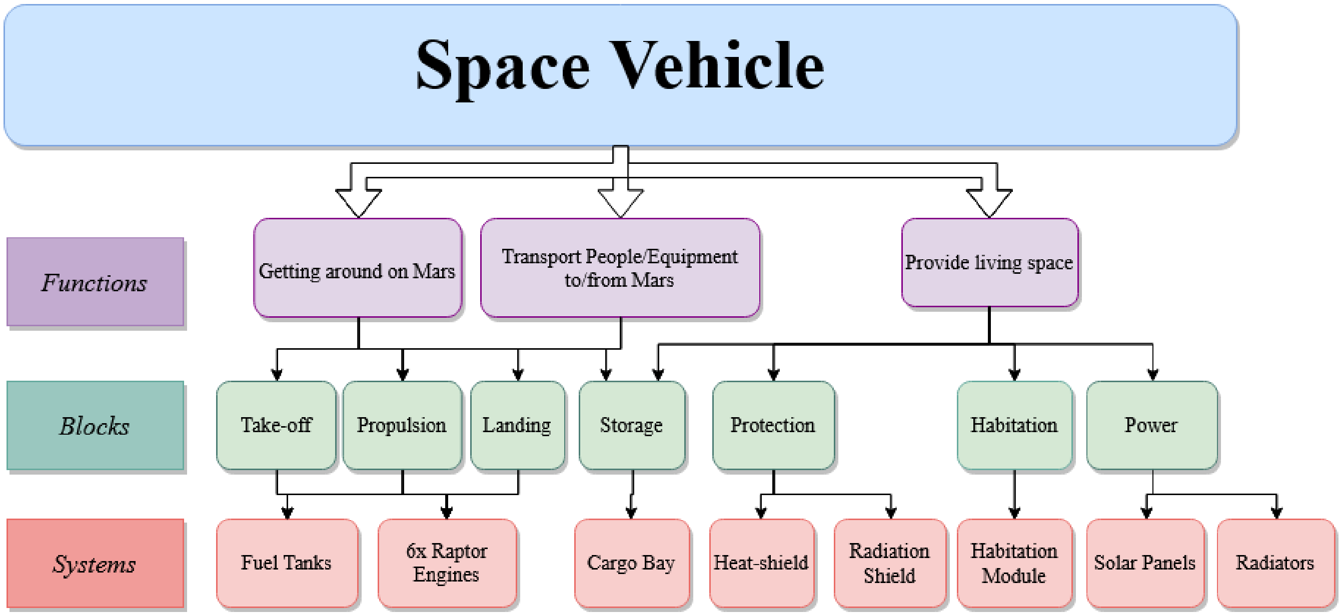

Figure 1: Functional diagram of the vehicle

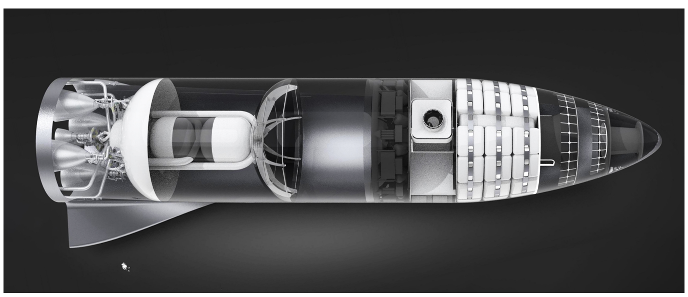

Figure 2: Starship spacecraft schematic of an old design iteration. Propulsion section and propellant tanks are on the left, crew and payload section on the right [6]

spacecraft in orbit, it will have the capability of bringing the 100 t payload to destinations far beyond LEO such as the Moon and Mars [3].

Starship was chosen as a baseline for the Hephaestus vehicle since its primary design purpose is to transport very large and heavy payloads to Mars with the capability of returning to Earth. Developing a completely new vehicle with similar propulsive landing and reuse capability was deemed needlessly risky, complex and expensive. The mission architecture requires the vehicle not only to travel to and from Mars, but also to make suborbital flights on Mars transporting the crew to the slopes of Olympus Mons [8]. Multiple launches and landings will be required of the vehicle and there are minimal possibilities for refurbishment on the martian surface. As of 2021 very few space vehicles with reuse capability have been successfully launched. SpaceX is currently the only entity capable of recovering rocket stages from orbital launches, although they might not remain alone much longer. Several national agencies and companies have expressed intent of developing reusable orbital rocket boosters, such as Blue Origin, Rocket Lab, Arianespace, Roscosmos and CSNA.

1) Main advantages of using the Starship as design baseline:

- very large payload capacity to the martian surface (SpaceX sites 100 t with orbital refueling);
- it is developed and intended for Mars missions other than Hephaestus, meaning it will be tried and tested in real world missions before. NASA is considering using it as a lunar landing system for the Artemis program [9];
- SpaceX has demonstrated expertise in propulsive landing and rocket reuse;
- uses liquid oxygen and methane for propellant which can be sourced from the martian bases. Propellant brought from Earth would be extremely expensive due to the high ΔV required to reach the martian surface.

2) Disadvantages and possible issues with the Starship baseline:

- Starship is currently under development, there is no guarantee it will ever be finished or that it will deliver everything that has been promised;
- extremely ambitious design that seeks to implement a lot of new untested technologies and concepts. The final result may not accomplish everything it was originally intended to do;
- for Starship to be used for Mars missions, orbital refueling and fast reuse must be mastered. These technologies are not currently ready;
- stainless steel design, among other things, results in high structural mass. Even with the modifications the vehicle's structural mass is over 100 t when landing on Mars. The

high dry mass makes the Mars landing more difficult. To date the heaviest thing to successfully land on Mars are the Curiosity and Perseverance rovers at 1 t [10];
- no launch abort system.

3) Two redundant vehicles: Two vehicles will be used for the mission. One vehicle, named Hera, is the primary crew vehicle. The other, named Zeus, will serve as a cargo and backup crew vehicle. Both will have the exact same vehicle design and feature the full compliment of life support systems detailed by the Human aspects group [11]. They only differ in what cargo is loaded onto each ship. Hera will be loaded with all the consumable supplies necessary for the journey to Mars and Zeus will carry consumable supplies for the Mars surface stay and the journey back. The primary purpose of using two vehicles is redundancy. The whole mission could be completed with one of the vehicles. In the event of catastrophic failure of one vehicle the mission can migrate to the other. Since the habitat are inside the vehicles, this also means there is a full backup habitat that can be used during the martian surface stay. If the ships were to fly in a close formation during interplanetary travel, the crew could perhaps even transfer between the two with EVAs.

4) Modification to Starship for Hephaestus: The Hephaestus mission requires roughly 30 t of payload per vehicle, much less than the Starship maximum 100 t payload. The payload section has an internal volume of 1100 m^3, which is also more than what is required for the mission. The primary modification to the baseline design will therefore be a shortening of the payload section by 7 meters. This is done to reduce the structural mass of the vehicle. The propulsion section, meaning the engines and propellant tanks, will remain the same size.

D. Space Vehicle Subsystems

1) Propulsion and Reaction Control Systems: To meet the large ΔV requirement for the interplanetary journey, i.e. 5.8 km/s for planetary transfer [12] and 8.9 km/s for hopping maneuvers (discussed in Mars hops section), the space vehicle must possess an adequate propulsion system. The assumption was made that production and storage facilities for both liquid methane (LCH_4) and water will be in short distance of the landing site at Gusev crater. Utilizing a LCH_4 fed propulsion system has the immense benefit of negating the need to bring propellant for the return journey. The ΔV capability of a rocket is given by the Tsiolkovsky rocket equation

$$\Delta V = I_{sp} g_e \ln \mathfrak{R} \tag{1}$$

where I_{sp} is the specific impulse of the propulsion system, g_e is Earth's gravitational acceleration at sea level and $\mathfrak{R}$ is the mass ratio of the initial mass m_0 and final burnout mass m_f, which are determined from Equations 2 and 3, respectively [13]

$$m_0 = m_p + m_s + m^* \tag{2}$$

$$m_f = m_0 - m_p. \tag{3}$$

Two promising LCH_4 fed engines are currently far in development. Raptor, a full flow staged combustion cycle engine, is in development by SpaceX and has already seen several suborbital test flights. BE-4, a staged combustion cycle engine, is in development by Blue Origin, but has not left the test stand yet. A comparison of important performance and physical characteristics is made in Table II. For given masses

Table II: Raptor and BE-4 characteristics [14][15].

Characteristic	Raptor	BE-4	Unit
Thrust (sea level)	2210	2400	kN
Thrust (vacuum)	≈ 2545	≈ 2764	kN
Chamber pressure	300-330	134	bar
Isp (sea level)	330	>311	s
Isp (vacuum)	380	unknown	s
Length (sea level)	3.1	≈ 4	m
Diameter (sea level)	1.3	≈ 1.8	m
Dry mass	1500	unknown	kg
Longest burn duration	280	>200	s

m_0 and m_f, it is evident from Equation 1 that a high I_{sp} is desirable for a large ΔV capability. The I_{sp} is defined as

$$I_{sp} = \frac{u_e}{g_e} \tag{4}$$

and the exhaust gas velocity u_e, assuming isentropic expansion of a perfect gas working fluid, is

$$u_e = \sqrt{2c_p T_{02}\Big[1 - \Big(\frac{p_e}{p_{02}}\Big)^{(\gamma-1)/\gamma}\Big]}. \tag{5}$$

as derived by Hill and Peterson [16]. A simplified analysis was carried out to compare the rated I_{sp} of the engines. As both the Raptor and BE-4 use methane, the specific heat c_p and heat capacity ratio γ are identical. For some exit pressure p_e, and assuming a similar combustion temperature T_{02}, it follows from Equation 5 that the higher chamber pressure p_{02} of the Raptor engine results in a larger u_e. It therefor achieves a higher I_{sp} according to Equation 4.

Additionally, the total thrust generated by the engines must exceed the vehicle's weight on Mars for it to be able to leave the ground for the return journey, i.e. the thrust-to weight ratio T/W must be greater than unity. This matter will be discussed further in the *off-nominal case* section. A simplified schematic that demonstrates the combustion cycle of Raptor is presented in Figure 3.

It is often the case during spaceflight that attitude maneuvers must be carried out for course corrections to ensure a satisfactory trajectory or orbit. The space vehicle must therefore include some sort of reaction control system (RCS) for it to be able to perform such maneuvers when necessary. It is desirable that the RCS thrusters operate on the same propellant used by the main propulsion system in zero-g, as it reduces mass and improves mission flexibility [17]. A further added benefit of utilizing LCH_4 for the RCS thrusters is simplified propellant loading and storage logistics, due to the highly toxic and corrosive nature of hypergolic fuels.

LCH_4 and liquid oxygen (LOX) fed thrusters intended for reaction control have been developed by Aerojet Rocketdyne and Northrop Grumman, both demonstrating approximately

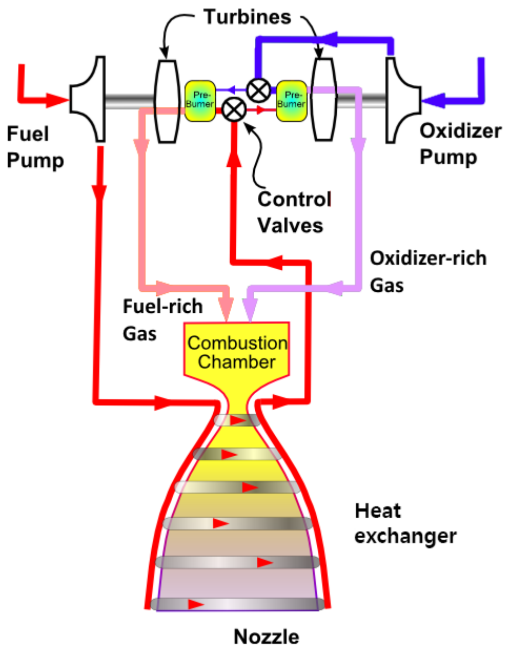

Figure 3: Raptor engine schematic

450N of thrust and a moderately high I_{sp} of 320-330s [18]. To change pitch, roll and yaw, the RCS thrusters must be positioned in such a manner that they can generate a torque around the center of mass of the vehicle. Due to the vehicle's symmetry, an optimal thruster placement scheme would consist of 4 thrusters at either end, at a maximum distance from the center of mass and symmetrically spaced around the circumference.

2) Power Generation: A power supply system for the vehicle is necessary to provide electricity to subsystems. For spacecrafts there are two commonly used options, nuclear and solar power. Solar was chosen for this mission since nuclear power would likely introduce legal complications. It is assumed the vehicle requires roughly the same amount of power as the ISS, since the crew of six is the same size typical for the space station. The ISS currently has a solar array capable of generating 84-120 kW [19].The Hephaestus vehicle has about a third of the ISS pressurized volume (400 m^3 vs. 1200 m^3) [20]. It is assumed 90 kW is sufficient to power the spacecraft. Solar irradiance at Mars is lower than on Earth at 590 W/m^2 [21], so the solar array must be sized to produce 90 kW in Mars' orbit around the sun.

3) Thermal Management: All power consuming equipment on-board the space vehicle will generate heat which combined with solar heating will affect the temperature level on board. This heat must be expelled from the space vehicle in order for its components and crew to function properly at a constant temperature level. This can be achieved using either a passive thermal control system (PTCS), an active thermal control system (ATCS), or a combination of both. The working principle of these thermal management systems is the storage of heat, transport of heat to a radiator, and rejection of heat to a heat sink.

The central component of the PTCS is a multi-layer insulation (MLI) material that functions primarily as a solar radiation reflector. The PTCS also incorporates a grid of copper heat pipes that enable heat transport. In case the PTCS is insufficient, an external ATCS is required. The ATCS usually incorporates two phase flow devices to achieve heat transport, and either mounted or deployable radiators for expelling the thermal energy. The International Space Station (ISS) uses a combination of both systems. For the purpose of mass considerations, the sizing of large components such as the radiator was considered.

In order to size the radiator, the thermal energy from the astronauts and the equipment was estimated. The daily heat emission $\dot{Q}_{\text{crew}}$ is approximately

$$\dot{Q}_{\text{crew}} = \frac{n \cdot E_{\text{intake}}}{24\,\text{h}} \tag{6}$$

where n is the number of crew members and E_{intake} is the average daily caloric intake per person. For a daily caloric intake of 2500 kcal, and assuming most of it is converted to heat, the total daily heat emission from the crew is then determined from Equation 6 to be

$$\dot{Q}_{\text{crew}} = \frac{6 \cdot (2500\,\text{kcal})(1.16\,\frac{\text{Wh}}{\text{kcal}})}{24\,\text{h}} = 725\,\text{W}.$$

The thermal energy emitted from electronic equipment was estimated using ISS data. The daily power consumption of ISS can reach up to 100 kW [22], so assuming a conservative 20% electrical energy to thermal energy conversion factor, some 20 kW are added for a total of approximately 20.7 kW that must be dissipated.

The required area A_r of the radiator can be determined from the following relation

$$P = A_r \epsilon \sigma T^4 \tag{7}$$

where ϵ is the emissivity (reflective power ratio, unity for a black body), σ is the Stefan-Boltzmann constant and T is the temperature difference of the radiator and its surroundings. Assuming a radiator temperature of 313 K, and a surface emissivity of 0.6, the required radiator area was estimated with Equation 7 to be

$$A_r = \frac{20\,700\,\text{W}}{0.6(5.67 \times 10^{-8}\,\text{W/m}^2\text{K}^4)[(313)^4 - (3)^4]} = 63.4\,\text{m}^2.$$

Assuming a radiator area density of 14 kg/m^2 as achieved by Boeing [23], the resulting mass of the radiator is 888 kg.

4) Radiation Shielding: For a long duration mission such as Hephaestus, radiation effects on the human body and other systems are of high importance and must be taken into account when designing the spacecraft. Outside Earth's magnetosphere, the main source of radiation are Galactic Cosmic Rays (GCRs) consisting of high-energy ionized protons, helium nuclei and heavy ions, originating from outside the solar system. Typical energy values range from 10 MeV to tens of GeV per nucleon [24].

Effects on the human body are measured in equivalent dose, H, measured in Sievert (Sv); this represents the absorbed energy per unit mass multiplied by a radiation weighting factor based on the characteristic of the radiation. A lower boundary can be fixed at 1.84 mSv/day for a Mars transfer [25] during Sun's maximum activity. However, during the Sun's minimum activity period, which is the case for Hephaestus mission, GCR flux increases and worsens its effects on human body.

Another radiation related risk are Solar Particle Events (SPEs). These are emissions of extremely high-energy protons, electrons and some alpha particles and heavier ions due to solar flares and coronal mass ejections [24]. These events are of short duration and less common during the Sun's minimum. They are, however, unpredictable and can cause exposure to very high radiation doses in a short period of time, while on the other hand decreasing the GCR flux. To protect the crew from the intense radiation caused by SPEs, a radiation shelter was required inside the spacecraft.

Shielding of the habitat was considered as the most effective solution for this mission. A review of traditional and non-traditional shielding materials was carried out, considering in particular aluminium, high-density polyethylene (HDPE) and metal hydrides.

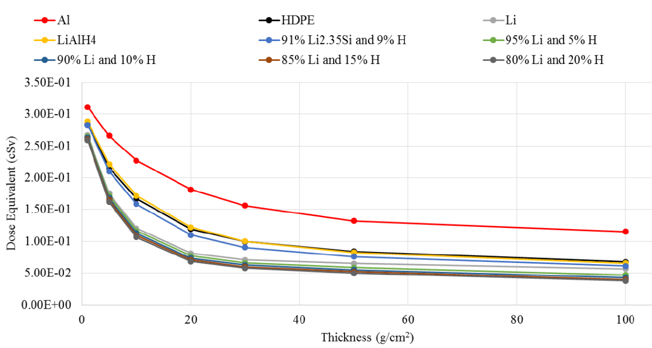

Figure 4: Equivalent Dose vs. Thickness [26]

Figure 4 compares the different materials with respect to the equivalent dose [26]. The study considered a worst-case-scenario of 1977 solar minimum GCR. It can be seen that the higher the percentage of hydrogen in the material, the better the performances. In particular, all metal hydrides outperformed the HDPE and the aluminium. It is also noticeable that after a certain thickness the variation in shielding effectiveness is not relevant anymore, setting an upper limit at about 30 g/cm^2.

Another efficient solution could be using water as shielding material. In fact, since it has a high proportion of hydrogen over mass, it has similar performance to traditional shielding materials such as polyetherimide [27], with the benefit that its mass would have to be carried anyway.

5) Spacecraft Internal Layout: The design of the habitat needs to take into account all of the requirements for a prolonged stay of a 6-persons crew in a very limited volume. A minimum acceptable net habitable volume was set to 25 m^3/person for a Mars mission with the same characteristics as Hephaestus [28] [29]. This was found through an identification of the tasks needed to be carried out inside the habitat, that later defined the space needed for each activity and subsystem. The proposed number is optimized for a crew of 6 people, however, it does not vary linearly with crew size, since functional areas such as the gym and the work space still require the same volume.

Functional areas needed in the spacecraft were considered to be the following:

- private quarters, i.e. sleeping space;
- dining and social activities space that can accommodate the whole crew;
- work space, separated from other areas to avoid cross-contamination;
- exercise area, with training equipment established by Human Aspect group [11];
- bathroom and hygiene area;
- storage for food and water and a cargo bay to accommodate the rovers [8];
- airlocks, necessary for EVAs.

This value for net habitable volume is calculated based on the possibility to exploit the entirety of the volume in a microgravity environment. Due to Hephaestus mission specifications, however, the same habitat must be used on Mars' surface, under gravity conditions. For this specific reason, a higher value must be considered when designing the habitat.

6) Mars Entry: Entry into the Martian atmosphere has usually happened directly from an interplanetary trajectory. This high-velocity entry creates thermal and aerodynamic loads, even in the thin atmosphere. Previous missions have employed blunt-body rigid aeroshells, which act as both heat-shields and, to a certain extent, lifting surfaces. Landers follow a ballistic or low-lift ($L/D \approx 0.2$) trajectory to a velocity of around 450 m/s (Mach 2 on Mars), after which the aeroshell is jettisoned and supersonic parachutes deployed.

The main characteristic of an entering craft is the ballistic coefficient β, given by the formula $\beta = m/(C_d \cdot S)$, where m is the vehicle mass, C_d the drag coefficient, S the drag surface. In general, a low ballistic coefficient is desirable, as hypersonic deceleration concludes at higher altitude, and thermal and dynamic loads are lower. However, decreasing β, while vehicle mass grows, demands ever larger aeroshells, far larger than current launch vehicles allow.

Additionally, uncontrolled ballistic entry penalizes the landing accuracy. It is desirable to maintain some amount of control authority over the vehicle during entry, to make course corrections. Previous missions have employed ejectable masses, in order to shift the vehicle center of mass and provide a fixed lift coefficient. Lifting and yawing the vehicle was accomplished by rolling it, changing the direction of the lift vector. Overall, greater control of lift and yaw can tighten the landing area.

In response to these challenges, the first approach was to use some combination of parachutes and inflatable decelerators. However, due to the spacecraft being a combined lander and hopper vehicle, performing sub-orbital flights, at least 3 atmospheric entries are performed. Reusability of the entry systems was required.

Furthermore, the vehicle requires some protection from aerothermal heating. As the kinetic energy is dissipated via atmospheric friction, a net heat flux heats the surface of the vehicle. While exact heating rates vary with entry velocity, angle, and vehicle shape, a target of 40 W/cm² for a period of 200 s was chosen for Mars entry [30]–[32], yielding a total heat load of 8 kJ/cm². For Earth entry, peak heating reached 480 W/cm² and total heat was 42.6 kJ/cm² for Apollo 4 [33]. Earth entry is therefore the main concern, and the heat-shield will have to withstand these thermal loads.

For both initial landing and the following hops, it was assumed that landing accuracy would follow the current trend, aided by ground sensing navigation, and be accurate enough to land a negligible distance from targets.

7) Mars Hops: While the majority of the Mars operations take place at the landing site of Gusev crater, there is a need for transport to and from the slopes of Olympus Mons. It was decided that this would be accomplished by using the vehicle in a parabolic flight [8]. Therefore, the vehicle must be capable of 2 parabolic flights of approximately 3800 km over the surface of Mars, without refuelling. Furthermore, the vehicle must land at both Gusev crater, located near 2 km below the Martian datum altitude, but also on Olympus Mons, more than 16 km above the datum. Atmospheric conditions vary greatly between the two landing sites, and the landing strategy will need to work accordingly.

E. Off-nominal Scenario

The success of Hephaestus relies on the space vehicle performing nominally for a duration of 985 days, which is plenty of time for something to go awry. Out of the six previously detailed subsystems of the space vehicle, a compromised propulsion system was considered. The malfunction of a single engine during the propulsive Mars hopping activities would result in inadequate generation of thrust and possibly loss of both vehicle and crew. For a configuration of five Raptor engines, the total thrust is

$$T_{\text{tot}} = \sum_{i=1}^{5} T_{\text{engine},i}. \tag{8}$$

For 2 vacuum optimized engines and 3 sea-level optimized engines, for which data was provided in Table II, the total nominal thrust is then

$$T_{\text{tot}} = 3T_{\text{sea}} + 2T_{\text{vacuum}} = 3(2210) + 2(2545) = 11720\,\text{kN}.$$

The total weight of the vehicle on Mars upon takeoff is

$$W = m_0 g_m = (1.346 \times 10^6\,\text{kg})(3.72\,\text{m/s}^2) = 5007\,\text{kN}$$

giving a nominal thrust-to-weight ratio of

$$T/W = \frac{11720}{5007} = 2.341.$$

In case of a single vacuum engine malfunction, it can be shown similarly that the total thrust becomes $9175\,\text{kN}$, for a T/W of 1.83. This also introduces the issue of the line of thrust being altered, resulting in an unstable takeoff flight trajectory. Assuming a gravity-turn trajectory and neglecting Mars atmospheric drag effects, it is desirable to achieve a T/W ratio between 2 and 3 [34].

III. Results

A. Overall Design

A simple CAD model of the space vehicle was made with accurate principal dimensions, from which a standard 3-view engineering drawing was generated as presented in Figure 5. The external geometry draws many similarities to Starship as previously motivated, but the payload section has been tailored to the mission specific internal layout. The outer diameter of the vehicle is $9\,\text{m}$ and its height is $43.2\,\text{m}$. The structural mass was determined to be 106 t, a 12 % reduction of the 120 t Starship baseline. With a payload of 30 t, Raptor engine Vacuum ISP of 380 s and full tanks with 1200 t propellant, the vehicle is capable of 8.5 km/s ΔV according to the Tsiolkovsky rocket equation [13]. For the Mars hops the payload will be 15 tons and the ΔV capability 8.9 km/s.

B. Launch Vehicle Selection

The Starship system was selected as the primary option for launch vehicle. It was chosen because it offers near the same payload capability as the SLS but very likely at a much lower cost. The US Office of Management and Budget had an estimate in 2019 that SLS would cost in excess of 2 billion USD per launch [35]. Also the Starship launch system will integrate well with the Hephaestus Spacecraft since it is based of the Starship second stage. New Glenn has the potential to be competitive on price but its payload capability is far lower than Starship. SLS and New Glenn will be considered as backup options if the Starship development program is unsuccessful. In that case the spacecraft design would need to be altered to suit those launch vehicles.

C. Space Vehicle Subsystems

1) Propulsion and Attitude Control: Due to the higher I_{sp} of Raptor, and superior integration compatibility with the Starship-like space vehicle, it was deemed a more suitable candidate for the main propulsion system. The engine cluster configuration is made clear in the isometric view in Figure 5. The total attainable thrust is $14265\,\text{kN}$ and the overall I_{sp} is $355\,\text{s}$.

The exact location of the center of mass could not be determined considering the conceptual nature of the design, but it is certainly above the vehicle's vertical center for stability reasons. The RCS thrusters will therefore be positioned at the aft end, and 2 independent systems utilized for redundancy for a total of 8 thrusters. No meaningful comparison could however be made for the Rocketdyne and Grumman models due to their similar specifications and lack of mass information.

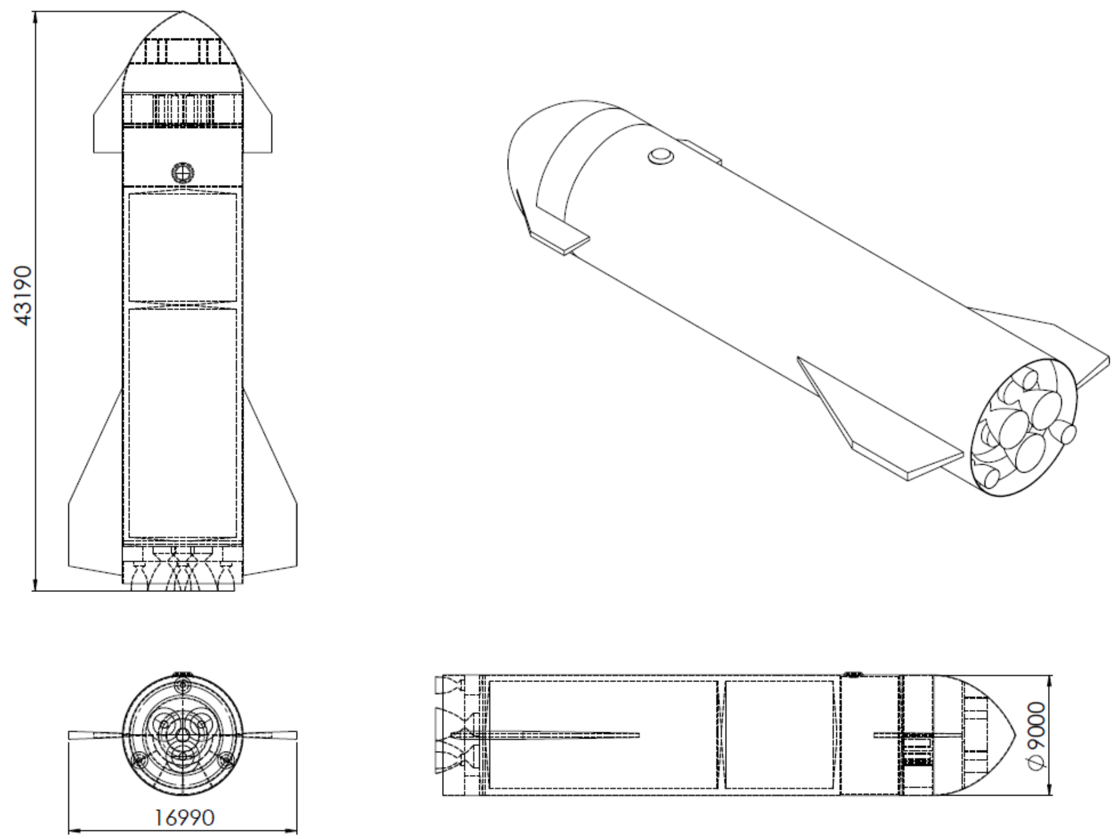

Figure 5: Engineering drawing of the space vehicle

2) Power Generation: In a Jet Propulsion Laboratory report on solar power technologies for future space missions, three currently available solar array systems are compared [36]. The system with the highest electricity generation efficiency was chosen for the Hephaestus vehicle, but all three arrays perform within one percentage point of each other. The selected array is from the company Spectrolab and features an electricity generation efficiency of 30.7%. With solar irradiance of 590 W/m^2 the panels generate 181 W/m^2. To produce 90 kW, 497 m^2 of panels are required. With an area mass density of 0.84 kg/m^2 [36] the solar array system would thus have a mass of about 420 kg.

3) Thermal Management: A combination of PTCS and ATCS to store, transport and expel heat generated on board and reflect incoming solar heating was deemed necessary. The most massive component of these systems was the radiator, which was estimated to have an area of approximately 57 m^3 and mass of 800 kg.

4) Radiation Shielding: As shielding material for the habitation modules, lithium metal hydride, composed by 80% Li and 20% H was chosen, as it outperformed all other materials, both in equivalent dose reduction and in weight. The reason for this can be found in the fact that the secondary radiation production, i.e. an additional radiation flux produced by the interaction between the primary radiation and the material itself, is lower than traditional materials [26]. The shielding covered the entire surface of the habitat, for a total of 12.1 m^3, choosing a 5 cm thickness. The material considered has a density of 0.57 g/cm^3 [26], resulting in a total weight of 6901.6 kg.

The equivalent dose corresponding to this coverage is 2 mSv/day, for the GCR flux of 1977 solar minimum. Considering the 985-day mission [12] planned, the total dose would be around 1.97 Sv. This value is higher that ESA limits for an astronaut's career but represents the limiting case at the worst conditions. It could be further reduced considering the extra protection given by Mars' CO_2 atmosphere during the stay on the planet [37]. It also needs to be noted that most likely new standards for mission outside LEO would be established to take into account the adverse environment, following the ALARA principle (as low as reasonably achievable) [29].

For the radiation shelter, located inside the pantry, the radiation shield is composed of both the food and water surrounding it. The water tanks' thickness is considered safe [27] for extraordinary fluxes of particles when combined with temporarily re-orienting the space vehicle so to interpose the propellant tanks between the crew and the Sun.

5) Spacecraft Internal Layout: The habitat is divided into three floors with different functionalities. In order for the crew to be able to utilize the habitat both in space and on Mars, i.e. with gravity, the main equipment and furniture is placed on the floors. The full volume of the two habitat floors was estimated to be $407\ \mathrm{m}^3$. Even overestimating the volume occupied by all the equipment and furniture to be $50\ \mathrm{m}^3$, the net habitable volume remains over $350\ \mathrm{m}^3$, which is 230% of the minimum requirements of $150\ \mathrm{m}^3$ [28], leaving the crew enough space to live comfortably both in space and on the surface. Each floor is accessible through airlock doors. This is a safety measure in case of faulty depressurization of one of the floors.

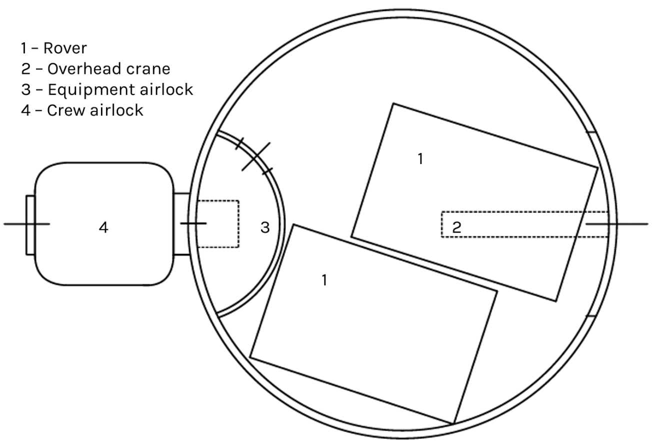

Figure 6: Cargo Bay overview

Figure 6 shows the lower part of the habitat, the cargo bay. Here, the two rovers needed for the mission are stored in an unpressurized environment. This floor offers two different ways out of the spacecraft. One is designed to be used while on Mars' surface and it is equipped with an overhead crane that is able to unload the rovers and the crew. On the other side an equipment airlock and a crew airlock give access to space for EVAs during the orbital transfer.

The crew airlock is design to be inflatable [38]. It can be stored inside the equipment airlock at only 50 cm and expand once in space, reaching 3 m in length and 2.77 m in diameter. The shell of the airlock is composed by several protection layers, including thermal protection, micrometeorite and orbital debris protection and a gas barrier bladder.

Figure 7 is a representation of the ground floor of the actual habitat module. The crew quarters, each with a volume of $10.8\ \mathrm{m}^3$, allow each member of the crew personal space, for sleeping, self-care and recreational activities. For comparison, the ISS crew quarters are $2.1\ \mathrm{m}^3$. A medical facility and bathroom are placed near the sleeping area in case of emergencies during the night. The exercise area is composed by a treadmill, a bike and ARED needed in order to reduce the effects of bone and muscle loss [11].

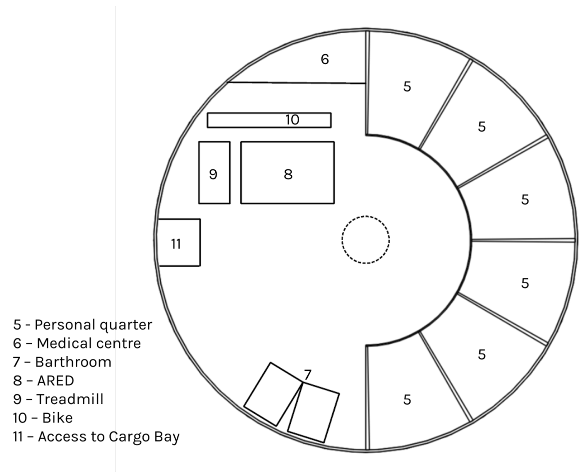

Figure 7: Habitat module - ground floor

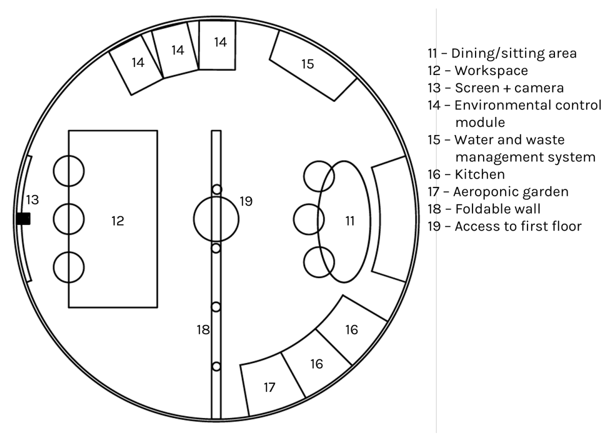

Figure 8: Habitat module - top floor

The top floor of the habitat module, as presented in Figure 8, is dedicated to the work space and the social space. On one side, the work table, a screen and any additional equipment needed to carry out experiments during the journey are placed. To better exploit the volume, machinery for activities that are only carried out during the transfer are placed on the ceiling. The other side of the floor hosts a table and the sitting area, the kitchen and other recreational equipment such as the aeroponic garden, that will provide psychological aid to the crew [11]. The two different areas are divided by a foldable and stowable wall that limits the contamination of the work space. The screen is designed to be multi-functional as it can be used for work and PR activities, but it doubles as virtual window when not in use. The environmental control module and the waste and water management system are also placed on this floor.

6) Mars Entry: Previous Mars landers have employed single-use heat-shields. These ablative covers are jettisoned after the hypersonic phase, exposing the engines and sensors. This strategy can not be employed in this case, as the mission includes multiple atmospheric entries: 3 on Mars, and 1 on

Earth.

As such, the vehicle employs a reusable heat-shield, which suffers negligible degradation after multiple entries. Gogu et al.[39] give several possibilities for material choices. A sandwich of a Nextel aluminosilicate composite top layer, glass fiber filling and aluminium backing, supported by titanium webbing. This is a middle-of-the-road compromise between price and mass. However, this TPS was not sized for Mars entry heat flux, let alone that of Earth entry.

The chosen solution was sized for a heat flux of 4 W/cm², over a period of 1100 s, for a total heating of 4.4 kJ/cm². The goal was to keep the temperature of the backing material below a service temperature of 450 K. In order to extend this performance to Mars-entry conditions, the hollow center portion was thickened by 30%. Earth-entry requires unrealistically thick systems, revolutionary advancements in material science, or a separate system.

The heat-shield was meant to also be a structural part of the spacecraft, reducing the mass of stainless steel body required. The source does not specify a strength for this particular material configuration, and it was assumed that it had the same strength as a steel plate and 75% of its mass. Therefore, 75% of the final heat-shield mass was subtracted from the body mass estimate. The final heat-shield has a mass of 15.3 t, and a rough material cost of at least 13 million USD.

Landing was chosen to be done via retropropulsion, i.e. a rocket-propelled landing. This technique has been widely employed by SpaceX in recent years, and obviates the need for large parachutes or inflatable devices. Instead, one can simply reuse the already-present propulsion system, and the only extra mass cost is propellant.

In terms of landing ΔV, numerical calculations were inconclusive, coming into disagreement with previous studies in terms of velocity and heating. It was decided to extrapolate results from Korzun and Braun [40], where the landing propellant mass fraction decreased linearly with increasing ballistic coefficient. Also, at high ballistic and thrust coefficients, retropropulsion cancels out body drag, and vehicle orientation can be ignored. Therefore, for $\beta \sim 380$ kg/m² at entry, 24% of the mass must be expended for landing. This corresponds to a ΔV of around 1 km/s.

7) Mars Hops: The vehicle was assumed to land back at Gusev crater with no remaining propellant. It would transport the ascent crew, both rovers, and enough food and water for the ascent. The exercise equipment, excess supplies, LSS spare parts and radiation shielding would be left at Gusev crater to save mass. The vehicle dry mass for the hops would therefore be 121 t.

Thanks to calculations done by the Mars Operations group, the ΔV for take-off for each hop was found to be 3.1 km/s to Olympus Mons, and 3.5 km/s back to Gusev crater. The disparity comes from the rotation of the planet. Landing at Gusev crater would be similar to the orbital landing, albeit with lower entry velocity (2.7 km/s at 200 km) and dry mass. As such, the orbital landing capabilities cover this phase, and a landing ΔV of 1 km/s was taken.

Landing on Olympus Mons poses a greater challenge: the thinner atmosphere and high altitude provide less drag to decelerate the vehicle, and less time in which to slow it down. Also, the entry mass is much higher than the return, at over 500 t compared to 160 t. Overall, the vehicle would impact the surface at a velocity approaching 3 km/s if direct landing is attempted. Such velocity can not be negated through pure thrust.

A solution is to drop to a lower altitude, such as 3 km, where the atmosphere is thicker. There, the vehicle can lose velocity through drag, while also using lift to maintain a constant altitude. It would then loft its trajectory, reaching the desired altitude with much lower ΔV required to land safely. Rather conveniently, the terrain between Olympus Mons and Gusev crater is all below 0 km altitude, with the slopes of the mountain rising sharply to 10 km [8]. This particular terrain allows for this glide-and-loft strategy, which has been explored in [32], [40], [41], and which SpaceX plans to use [42], albeit not on Olympus Mons. Aerodynamic control can be implemented through simple, flat aero surfaces. An example of a lofted landing trajectory can be seen in Figure 9. Actual landing ΔV at Olympus Mons could not easily be determined, so was taken to be approximately 1.2 km/s, to be at least that at Gusev.

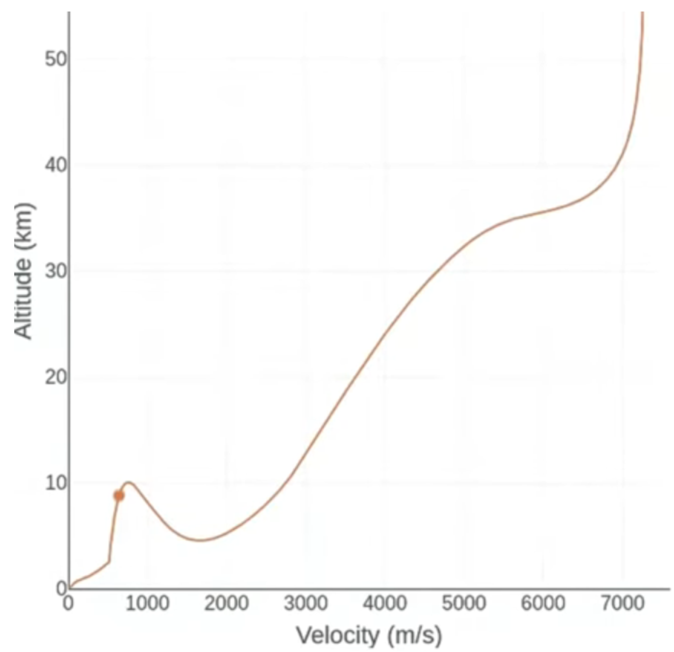

Figure 9: A SpaceX simulation of the landing profile, from [42]

D. Off-nominal Scenario

The decrease of T/W and skewed line of thrust due to the malfunction of a single vacuum-optimized Raptor engine can be compensated for by installing a spare engine for redundancy and automatic gimballing system for thrust vectoring. This will entail increased structural mass of the space vehicle (engine, piping, hydraulics, etc.), which together was estimated to be a total of 3 tons, small in comparison to its total mass. The new $(T/W)_{\text{new}}$ for the same thrust profile, i.e. 5 engines firing in tandem, is

$$(T/W)_{\text{new}} = \frac{11720}{((1.349 \times 10^6\,\text{kg})(3.72\,\text{m/s}^2)) \times 10^{-3}} = 2.335$$

which is almost identical as before, and within the optimal T/W range.

IV. Discussion

The SpaceX Starship was chosen as design baseline because of its many desirable characteristics. However many of these such as orbital refueling, fast reliable reuse, Mars propulsive landing capability and multi-use heat shield are yet to be demonstrated in the real world. The design is a very ambitious one and the final product may not deliver on all the features advertised by SpaceX. The structural mass might have to be significantly higher than planned thus reducing maximum payload mass. The heat shield might not be durable enough for several landings without refurbishment. Propulsive landing is a rather new technology and it might not be reliable enough to safely land human crews. And if something were to befall SpaceX, such as bankruptcy or leadership change, development on Starship might cease.

The currently chosen heat-shield is not capable of assuring hyperbolic atmospheric entry at Earth, nor can it realistically be expanded to do so. Space Shuttle-style carbon-carbon tiles can be explored as an alternative, but at a greater structural and financial cost. Otherwise, a less heat intensive orbital capture can be used, but with higher ΔV requirements. The ΔV for landings and hops are given without safety margins. More accurate simulations would refine these numbers, and give a better estimate of feasibility. The exact landing configuration was not studied, and some deployable leg system was assumed, but no added mass was considered.

The radiation shielding was considered to be removable from the spacecraft to perform the hops and then mounted back after returning to Gusev crater. This design choice was made because of the payload mass requirements for the hops to be around 12 tons. However, this would complicate the shielding system and the layout of the habitat, requiring the crew to disassemble the furniture and remove panels of shielding material and vice versa. This could also mean an increase in mass for the shielding system and further investigation on the feasibility of such solution should be carried out. The mission also requires the ability to fill a spacecraft with thousands of tons of propellant, on the surface of Mars. The facilities for this were assumed to be part of the ISRU bases.

V. Conclusion

This paper presents a conceptual design for a multipurpose spacecraft which, as indicated by the obtained results, is likely to meet the functional requirements as laid out by the Mission Design and Mars Operations teams. Much of the work is based on state-of-the-art or experimental technology, and extrapolations believed to be reasonable within a 20 year time-frame.

VI. Division of Work

A. *Birgir Steinn Hermannsson*

Propulsion and Reaction Control Systems, Thermal Management, Overall Design Results.

B. *Connor Benjamin Cruickshank*

Mars Entry, Off-nominal Scenario, Mars Hops.

C. *Greta Tartaglia*

Radiation Shielding, Spacecraft Internal Layout.

D. *Joel Lundahl*

Launch Systems, Power Generation, Overall Design Methods.

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