Human Spaceflight

First manned Mars mission - The transplanetary vehicle

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Project report

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1. Introduction

The transplanetary vehicle is a vital part of the mission. It is responsible for keeping the astronauts alive and healthy from the moment they leave LEO to the moment they reach Low Mars Orbit (LMO). It is also responsible for carrying the tools to facilitate the astronauts' safe descent and survival on Mars, and their research during the mission. The transplanetary vehicle remains in LMO throughout the duration of the astronauts' mission on the Martian surface. It returns the astronauts from LMO to LEO at the end of their stay on Mars.

One of the key obstacles to a feasible mission is the mass and size of the transplanetary vehicle. We need to carry astronauts, infrastructure for their survival during the transplanetary trek, fuel, as well as research payloads on route. Naturally this drives both the mass and size of the vehicle beyond what is capable now, and even beyond what will be capable in 20 years when we plan to launch the mission.

Our implemented solution to this problem is that everything that is unnecessary for the astronauts' survival and research during the transplanetary stage but is necessary on Mars will be sent on other launches. These launches will be cargo missions and will precede the manned launch to Mars. These cargo launches will carry several components such as a Mars Surface Habitat, Mars Ascent Vehicle, Descent Vehicle, Research payloads, and other equipment required on Mars. The payloads are more thoroughly defined by group 4, who are responsible for survival and research on the Martian surface. Eliminating all equipment that is unnecessary for the manned transplanetary stage significantly reduces both the mass and the size of the manned vehicle required.

The transplanetary vehicle has a 30 tonne empty mass and is therefore launched in one piece to LEO. Spacecraft dock with the manned vehicle in LEO and supply it with its required cargo such as food, oxygen tanks, supplies and spares. They also deliver solar panels that are used as the main power source during the journey. A docking spacecraft may also deliver a robotic arm. This will be attached in LEO and can be used during EVA's to help the astronauts. The astronauts travel from Earth in the Earth Return Vehicle (ERV) and also dock with the manned vehicle in LEO. The ERV remains docked throughout the entire journey and is used to return astronauts back to Earth from LEO at the end of the journey. Once the vehicle has been supplied, the robotic arm assembled and the crew is onboard, the vehicle will be launched from LEO to LMO.

The cargo mission will deposit a Mars Ascent Vehicle (MAV) to the surface, and a Descent Vehicle (MDV) to LMO. Once the manned vehicle enters LMO, the MDV will dock with the manned vehicle. The astronauts will be transferred to the MDV and then travel to the Martian Surface. At the end of their stay on mars, the MAV carries them back to the manned transplanetary vehicle. This avoids the problem of having to launch the transplanetary vehicle from Mars' surface. This increases the possible mass and size of the vehicle, and also reduces the necessity for thermal protection.

The project was started by outlining all the subsystems the spacecraft needs for such a mission. We went on to assign specific subsystems to each group member for further research. One of the first task was a basic mass estimation which was refined during the whole development process. The last task was the specific design of the spacecraft / interior which integrated all the subsystems and modules from each members research. During this process a close communication to group 4 and especially group 2 has been done to fulfill all requirements needed for this mission.

The following report talks about the basic components and technologies required for the astronauts' travel, the size and shape of the vehicle, the interior section of the manned vehicle, and mass estimations for the mission.

2. Communication

The ability to maintain communication with Earth will be crucial, both from the mission operations perspective as well as the psychological aspect for the crew.

The transplanetary vehicle will, instead of using radio waves, use optical technology for maintaining this communication. Due to optical waves having much smaller wavelengths than radio waves (and thus higher frequencies), the propagation effects of the waves are much smaller which leads to the ability of focusing more energy to one point, i.e. increases the ability to carry more information, as well as to longer distances [1][2].

One of the bigger challenges then becomes the ability to focus this signal to a small point on Earth. The beam will only have a diameter of approximately 6 km, which is very small compared to Radio Frequency system [2]. The transmitters on the vehicle will be fitted with reaction wheels which, if in the event the vehicle is disturbed by foreign objects which would cause it to shift its orientation, will compensate for the shift and turn the transmitters always towards the receivers on Earth.

Tests done by NASA has shown that it is possible to have a downlink of 622 Mbps and uplink of 20 Mbps from the Moon to the Earth, which is 6 times higher than the best Ka-band system used today [2]. The mass of the system has been reduced as well to about 300 kg, and the transmitters require only approximately 0.5 W of power [3].

3. Navigation

Knowing where we will be in the solar system and confirming that we are on track will be of most importance, as this is what our mission is about. Hence we must achieve high accuracy in navigation as well. Thus, the vehicle will use pulsars as an approach for maintaining this accuracy.

Pulsars can be seen as interstellar "light houses" which emit very powerful, predictable, beams of light. The vehicle will be outfitted with X-ray telescopes that measures the X-ray photons emitted from these light beams and through algorithms give its position in the solar system [4]. Due to their predictable pulsations the accuracy will rival those of the atomic clocks used in the GPS satellites, and be able to provide accuracy within 500 m anywhere in the solar system [4][5].

4. Thermal protection

The transplanetary vehicle needs thermal protection against the atmosphere of LEO and LMO. It also needs to be able to sustain the temperatures encountered near the Venusian upper atmosphere. Finally, it needs protection from the solar irradiance during the transplanetary journey, which is lowest near Mars and highest near Venus.

Multilayer Insulation (MLI), a form of Passive Thermal Control, can provide effective thermal protection to the transplanetary vehicle. MLI uses radiation barriers to reflect solar radiation [23]. Figure 1 shows the basic cross section of the MLI. It keeps solar radiation out and keeps the bitter cold of space from penetrating the Station's aluminium skin.

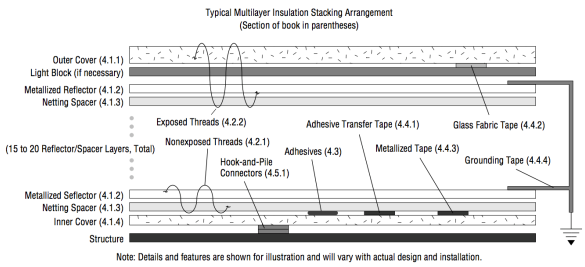

Figure 1. Schematic cross section depicts the key elements of an MLI blanket. Not all elements need be present in every design.

Figure 1: MLI cross section [23]

Each reflector layer shown in Figure 1 is able to reflect 90-99% of sunlight, and the cumulative effect leads to almost total solar reflectance [23].

The Outer Cover in Figure 1 typically needs to be resistant to shedding, flaking and other particulate generation. It is usually opaque to UV radiation. Teflon with Inconel backing layers is a good candidate. It works well for temperatures between $-184° -$

$150°C$. The highest temperature encountered by the transplanetary vehicle would be $125°C$ near the Venusian upper atmosphere. The coldest temperature encountered is most likely $-180°C$ on the Earth's or Mars' shadowed side during orbit [21]. Teflon outer layer has a thickness of 0.013 mm and a mass of $0.028 kg/m^2$. The transplanetary vehicle surface area is $647m^2$, so the outer cover mass will be 18 kg [23].

A major consideration in designing MLI blankets is how many reflector layers to use. Generally 15-20 layers are used. Aluminized Kapton is a good material to use for the reflector layer. The Kapton is aluminized to prevent solar radiation getting through [21]. It has a continuous temperature range of $-250° - 288°C$, a thickness of 0.0076 mm and a mass of $0.011 kg/m^2$. Using 15 reflector layers gives a total reflector mass of 107 kg [23].

Separator layers are placed between each reflector layer. Dacron Netting is generally used. They are placed to prevent heat being conducted between layers. It has a thickness of 0.16 mm and a weight of $0.0063 kg/m^2$. It has a temperature range of $-77° - 177°C$, which is satisfactory in our case. Using 16 layers gives us a total separator mass of 65 kg [23]. The inner cover is adjacent to the underlying hardware. An Aluminized Reinforced Kapton Cover can be used. It has a thickness of 0.013 mm and a mass of 32 kg [23]. The total mass of the MLI works out to be 222 kg.

There are options for threading and adhesives that are not discussed in detail here. Generally Nomex or Nylon threading is used to stitch the layers together. Hook-and-pile fasteners are used to attach MLI joints and assemblies. Silicone based adhesives are used to bond fasteners to the structure. Metallized PSA Adhesive tape is usually used on the reflector layers and the Inner and Outer covers [23].

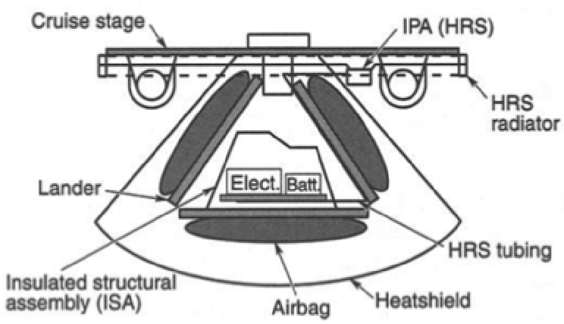

Figure 2: Mars Pathfinder HRS [22]

An active Heat Rejection System (HRS) is needed on the spacecraft to regulate interior temperature. We use a HRS design similar to the Mars Pathfinder mission shown in Figure 2 [22]. It is only employed on the Command Centre, which contains sensitive operational equipment. The objective is to keep this equipment within an allowable temperature range, and is accomplished by a mechanically pumped fluid loop to transfer excess heat from the interior components to an external radiator. The mass of this system is about 35 kg and it requires about 10 W of power. It provides a cooling power of 90 - 180 W and uses Freon as the coolant liquid.

The key components of the HRS are the integrated pump assembly (IPA), Freon-11 working fluid, HRS tubing, electronics and radiator. The electronics are the command centre equipment. The IPA circulates the Freon coolant via the HRS tubing from the electronics to the radiator. Thermal control valves are used to keep the electronics at a fixed temperature within the allowable range for the equipment. If the temperature exceeds the operating temperature, all the flow goes through the external radiator and heat is rejected. If the temperature is too low, all the flow bypasses the radiator and no heat is rejected. For intermediate values, some of the flow is allowed to pass to compensate for the electronics constantly heating up [22].

The Freon-11 working fluid has a freezing point of $-111°C$ and a boiling point of $24°C$. Therefore it can pass through the radiators without freezing as long as it passes quickly. It is also handled on ground without boiling. It has high thermal conductivity and low viscosity for high heat transfer and ease of flowing. [22].

The radiator is a circumferential aluminium strip around the general crew module. It is 0.75 mm thick and connected to the HRS tubing. It is attached to the exterior structure and painted white to maximize heat loss. It is used to reject up to 180 W of heat into space.

Electric heaters are also used in the General Crew module and the Sleeping modules, and are regulated by temperature sensors. Each heater weight about 5 kg. The total weight of the thermal control system adds up to be 267 kg.

5. Micrometeoroid and space debris protection

Micrometeoroids are very small pieces of rock or metal broken off from larger chunks of rock and debris often dating back to the birth of the solar system. Micrometeoroids are extremely common in space. Tiny particles are a major contributor to space weathering processes. Their velocities relative to a spacecraft average 10 kilometers per second (22.500 mph), and resistance to micrometeoroid impact is a significant design challenge for spacecraft and space suit designers. While the tiny sizes of most micrometeoroids limits the damage incurred, the high velocity impacts will constantly degrade the outer casing of spacecraft in a manner analogous to sandblasting . There are 3 primary shielding configurations [7]:

- Whipple shield-is a two layer shield consisting of an outer bumper, usually aluminium, spaced some distance from the module pressure shell wall; the bumper plate is intended to break up, melt or vaporize a particle on impact.

- Stuffed Whipple shield-consists of an outer bumper, an underlying blanket of Nextel ceramic cloth and Kevlar fabric to further disrupt and disperse the impactor, spaced a distance from the module pressure shell.
- Multi-layer shields, consisting of multiple layers of either fabric and/or metallic panels protecting the critical item.

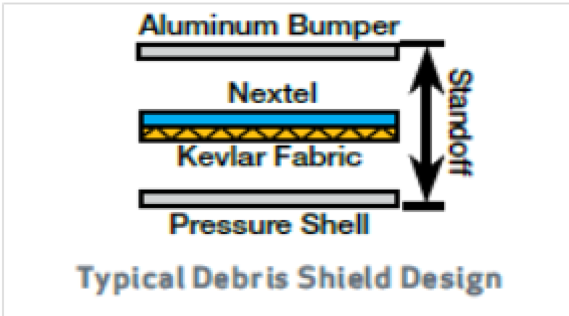

Figure 3: Typical Debris Shield Design

In order to protect the space craft the following double bumper configuration consisting of vectran fibers instead of the Kevlar fibers was selected based on experimental [19] results which show that adoption of this high strength fiber in the bumper materials reinforce the protection capability and reduce the weight drastically . This high strength fiber has been used as airbags with the Mars Path Finder.(aerial density ($10kg/m^2$) is half of that of the previously used shields).

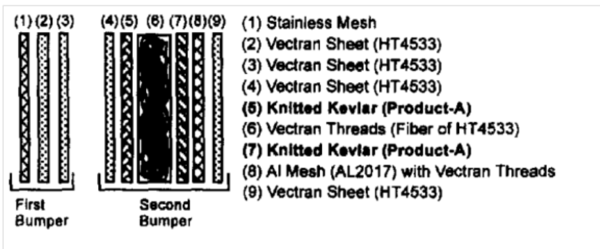

Figure 4: Debris Shield with Vectran fiber

The first bumper is composed of a stainless mesh and two Vectran sheets, and has a role for breaking up space debris into a debris cloud at the beginning of impact. Two Vectran sheets of the first bumper cling with crossing stitch directions. The second bumper composed of two Vectran sheets, a Vectran cloth, a lump of Vectran threads, and an aluminium mesh. This shield features a knitted Vectran cloth with crochet stitch and a sewn aluminium mesh using Vectran threads.

6. Radiation shielding

When conducting long space missions outside the magnetic field of the Earth, it is important to protect the astronauts against radiation exposure. Highly energetic galactic cosmic radiation (GCR), with it origins from supernovas and similar outside the solar system, poses a constant danger to the astronauts. During a solar maximum the level of GCR are lower than during a solar minimum, but there is an increased risk of solar particle events (SPE) during these periods. Solar particle events are very unpredictable and may instantaneously expose astronauts to lethal doses of radiation without sufficient radiation shielding [30].

6.1. Active radiation shielding

The concept of active radiation shielding is to create an artificial magnetosphere to protect the spacecraft and deflect incoming radiation. This is a slightly futuristic concept, since the weight of a system, able to create a magnetic field with sufficient strength to deflect highly energized cosmic particles, would be to heavy to introduce in a spacecraft with most of present day technology. However, research on so called mini- magnetospheric plasma propulsion (M2P2) conducted during the last decade, indicated that the proposed propulsion system could potentially be modified to be used for radiation shielding [11]. The M2P2 system consist of one or multiple magnets whose magnetic fields are expanded using a plasma source. The system was tested in a NASA-funded small scale test around 2001 giving promising results [10], but limited amounts of research in the subject has been conducted since then due to the lack of financial support [25].

Much is still unknown regarding the feasibility of this sort of system, and the required time to develop a functioning product might be too long given the time frame of the mission. Due to the high level of uncertainty regarding the functionality, shape and requirements of an active radiation shield, this kind of system will not be included in the spaceship design of this project.

6.2. Passive radiation shielding

The commonly used option for radiation shielding is to use a material that partially blocks the incoming radiation. A problem when choosing material for the radiation shield is that secondary particles, formed when the energetic particles collide with the radiation shield, may pose additional dangers to the astronauts. For this reason it has been found that materials with high hydrogen content is favorable for radiation shielding. Two shielding materials, with high levels of hydrogen, that can be considered is polyethylene and water. The polyethylene needed for shielding is lighter than the corresponding amount of water, but polyethylene has the disadvantage of being quite flammable, and it has a low melting point [30][17]. For this reason, further material development is needed to enable full utilization of polyethylene as a space ship radiation shield. This is an area of ongoing research. One promising polyethylene material developed by NASA, is the RXF1, a compound that is 2.6 times lighter, but that displays three times higher tensile strength than aluminum [29]. It seems reasonable that a material similar to the RXF1, could be developed and used for radiation shielding, within the time frame of the mission.

According to simulations made by NASA in 1997, using the high-charge-and-energy (HZE) transport computer program HZETRN, a polyethylene shield weighing about $70kg/m^2$, would provide sufficient protection against a GCR environment during a solar maximum. This was based on the GCR level recorded during the solar maximum in 1970 . The shield would also protect the astronauts against a SPE in the same order of magnitude as a highly energetic event recorded in 1972. If the mission were performed during a solar minimum, the corresponding weight of the shields would have to be $300kg/m^2$. This comparison shows the importance of choosing a launch window during a solar maximum, in order to decrease the weight of the spacecraft and thereby reduce the total cost of the mission [17].

6.3. Radiation limits

For long term missions, the most important thing is to reduce the risk that the astronauts die in cancer after radiation exposure in space. The general guideline is that the risk of dying in cancer should not increase more than 3 %. The blood forming organs (BFO) is among the organs that is most sensitive to radiation. The current annual limit for radiation exposure for the BFO is set to 0.5 Sv/year [12], though it should be noted that much is still unknown regarding the effects of long duration radiation exposure on biological tissue.

6.4. Mission solution for radiation shielding

The most reasonable way to handle radiation exposure seems to be to minimize the transit time to Mars as much as possible and to conduct the journey during a solar maximum. A solar maximum suitable for the time frame of this mission has been predicted to occur in year 2034-2035 [14]. To go during a solar maximum significantly reduce the weight of the passive radiation shields, but the shielding material will still be to heavy to cover the entire ship. Therefore, a specifically designed compartment, surrounded by an inner polyethylene structure, will be incorporated in the spacecraft design (see Appendix A). This module will provide an additional shelter for the astronauts during periods of high radiation exposure. Since a large quantity of water still needs to be incorporated in the life support system of the spacecraft, the on-board water supply will also be utilized for radiation protection. This will be done by surrounding the inner structure of the radiation shelter with a number of water-tanks. The radiation shelter has been designed as a 2 meter high cylinder with a 3 meter radius. Given the previously mentioned weight requirements per square meter of the polyethylene-shield, it is estimated that around 21.6 tonnes can be saved if the journey is performed during a solar maximum (See Appendix B).

6.5. Risks and actions against radiation exposure

The most crucial periods, when it's extra important for the astronauts to remain inside the radiation shelter, includes the passage through the Van Allen belts and during solar particle events. Careful monitoring of solar activities, and good communication abilities between the spaceship and the ground control on Earth, are needed during the mission, in order to provide warnings to the astronauts of impending solar flares. The chosen mission trajectory will take the spacecraft in a fly-by around Venus. If a solar particle event should occur during this period, in particular when the spacecraft is behind Venus as seen from the Earth, it will be very difficult if not impossible to convey solar flare warnings to the astronauts in sufficient time. This is made extra difficult because of the communication time delay due to the distance from the ship to the Earth. The heightened risk during this part of the journey, makes it extra important that the astronauts remain inside the radiation shelter for as much time as possible, until the distance to the Sun has increased. For extra protection against the risk of radiation exposure, the crew sleeping quarters will be placed inside the radiation shelter. This way the astronauts will not risk exposure to lethal radiation doses from SPE while they are asleep.

It should be noted that without functioning active radiation shields on the interplanetary vehicle, the astronauts will unavoidably be exposed to higher radiation dosages than on Earth and in LEO. The hull of the spaceship might provide some protection, but since the weight of the passive radiation shields are too heavy to enclose the entire transit habitat, a certain level of risk have to be taken in terms of radiation exposure. This is a risk every participating astronaut must be willing to take. To further reduce the risks of radiation exposure, it is of out-most importance to research the area of active radiation shielding for future interplanetary space travel.

7. Life support system

One of the most crucial systems in a manned spacecraft is the life support system. This system has to keep the crew alive during the whole mission and is not allowed to have any crucial failure since the crew would die then. Thus the life support system has to be very reliable, easy to maintain and the technologies have to be approved for a few years to confirm their ability for long term use as needed in this mission. Because of that the decision was to use mainly currently available or soon available technologies for the life support system to ensure this criteria. Technologies approved in previous missions have also a positive psychological aspect since the astronauts know that a technology is already tested for that kind of long term use instead of being the first persons using that on a long term mission without the possibility of a return to earth when something fails.

The first thing of designing the life support system is to clarify the basic requirements. These basic requirement every life support system has to provide / every person needs are a breathable atmosphere, water, food and waste removal during the whole mission [30]. During the development process further "additional" requirements came up. These are a minimum of 2 levels of redundancy in every part of the system and a closed cycle as possible to save as much weight as possible. The closer the cycle the less supplies need to be carried and in case of an unexpected longer mission time (due to engine failures, missed start window etc) the chances of the crew to survive increase. A last requirement for the life support system set during the development process is that it should not rely on biological systems for life support functions. Despite a lot of research is done especially in biological life support functions this decision was made to provide a higher reliability and maintenance. For every "technical" failure spare parts can be carried in the spacecraft and then the life support system can be repaired. When using biological systems this is not possible. For example when using plants for CO_2 reduction they can't be replaced or repaired when they die. Thus according to the start in 2038 it is saver to stick to "common" technologies for the life support systems. For further missions the biological systems can of course be taken into account when more research / testing data is available.

For the design of a life support system it is necessary to know what the need of one person per day in terms of water, food, oxygen and waste is. According to [30]and [18] this needs are

Input:

- 0.83kg Oxygen
- 0.7kg potable water
- 23 kg hygiene water
- 0.5 kg food (dry mass)
- 2.3 kg water in food

Output:

- 1 kg CO_2 + Trace gases
- 1.6 kg water (air)
- 1.6 kg liquid waste
- 0.1 kg solid waste
- 23 kg hygiene waste water

Considering a crew of 4 persons and a total mission duration of 586 days (304 days outbound, 112 days stay, 160 days inbound) this leads to the basic sizing of the life support system.

According to this needs the life support system is spitted up into 4 different sections: Atmosphere management (AM), water management, waste processing and food supply. The AM is responsible for generating the habitat atmosphere, monitoring the gas composition of the atmosphere and the presence of harmful contaminants and the detection and suppression of fires. Providing potable and hygiene water for the crew is the main outline for the water management, the collecting, processing, storing and dumping of waste is part of the waste processing and finally the food supply system is responsible for providing the necessary food for all crew members.

Due to the size of this report it is not possible to point out every technology used for the life support system with an explicit argumentation why they are used. A list of all applied technologies can be found in the Appendix C as well as a schematic picture of the life support system (Figure 9). For further research about the technologies see [18].

Thus only the main features of the life support systems will be presented here:

- Closed water cycle
- Exhaust gases: NH_3, NO_x and SO_x
- Resupply: N_2, O_2 and food (0.5kg dry mass per person and day)
- Only available or soon available technologies are used

The total weight of the life support system is calculated to approximately 3500 kg (only equipment) plus all resupplies (N_2, O_2, food) and the cryogenic gas tanks required for the life support system. For all these supplies a safety buffer of 112 days is considered in case the first return window from mars will be missed and the crew has to wait for the next one. In addition a 10% safety margin is added to all supplies. This leads to 112.86 kg water per day (including the 10% margin). Due to the closed water cycle there is no additional water needed. Since no loop is completely closed a loss of 1kg water per day is assumed as additional safety buffer which leads to a total water mass of 688.86 kg. The total mass of food is 1289.2 kg. Estimating the weight of the resupply gases is very hard since there is no exact reference of the consumption of the life support systems. According to [18] the weight of this is estimated to approximately 4000 kg.

8. Power supply

For operating the spacecraft it is necessary to provide the maximum needed Power during all flight phases. For this mission the necessary power is estimated to 30kW.

Due to the fact that the spacecraft uses chemical propulsion instead of nuclear propulsion it is decided to use common solar arrays as main power producing system. The main points for choosing this system are the high reliability in space, the low price and the high experience in using this system. To size the solar arrays the worst case scenario has to be considered. This means lowest intensity of the sun and longest time in eclipse where the power can not be provided by the solar arrays. Since the spacecraft will face the sun the whole time during the transit and we never get further away from the sun than in the mars orbit (see group 2) our scenario for sizing the solar arrays is very obvious: Sun intensity in mars orbit and the eclipse time during mars orbit for the secondary power system. The size of the solar arrays depends on the type of power supply used during eclipse since some has to be charged (batteries) and some not (fuel cells). Thus two different options are considered in this report: Solar arrays + batteries and solar arrays + fuel cells. After calculating the specifications and weight of both systems it will then be decided which fits the best for this mission.

The needed power of the solar arrays can be calculated as

$$P_{SA} = \frac{\frac{P_e T_e}{X_e} \frac{P_d T_d}{X_d}}{T_d} \tag{1}$$

where P_e is the Power needed during eclipse, T_e the duration of eclipse and X_e the path efficiency from the power system to the spacecraft during eclipse. The d indicates the same things during "day".

The specific power of a solar array can be calculated to

$$\Psi = \eta \epsilon_{pa} \epsilon_{cos} (1-d)^3 \Phi_{solar} = 103.829 W/m^2 \tag{2}$$

where η is the efficiency of solar cells, ϵ_{pa} the packing factor (loss in efficiency due to spacing of cells, thermal protection etc.), ϵ_{cos} the cosine factor considering the angle of the solar arrays to the sun and d is the degradation per year. Finally Φ_{solar} is the intensity of the solar flux. [20] The material used for the solar arrays is silicon so $\eta = 0.3$, $\epsilon_{pa} = 0.7$, $\epsilon_{cos} = 0.9$ and $d = 0.0375$. The solar flux is $593 W/m^2$ since a mars orbit (longest distance to the sun) is considered as worst case scenario.The maximum angle of the solar arrays to the sun is set to $25°$ since every solar array will have Solar Array Drive Mechanisms (SADM) which turn the array into the direction of the sun.

The final area of the solar arrays is then computed to

$$A_{SA} = \frac{P_{SA}}{\Psi} \tag{3}$$

As pointed out by group 2 the total mars orbit time is 88615.025s with 0.39h eclipse time and 24.22h daytime. For the scenario with fuel cells the eclipse time is set to zero since fuel cells must not be charged by the solar arrays during daytime. This leads to the values for solar array power and area (both cases, fuel cells and batteries) shown in Table 1

	with batteries	with fuel cells
P_{SA}	$46722.17W$	$46153.85W$
Area	$449.99m^2$	$444.52m^2$

Table 1: Properties of solar arrays

It is obvious that the difference in the area of the solar array is very small. Thus the decision for fuel cells or batteries only depends on the weight and reliability / maintenance of this two technologies.

For the batteries the number of batteries which have to be in series / parallel have to be calculated. The number of batteries in series can easily be calculated as $N_{series} = \frac{V_{ba}}{V_{cell}}$ where V_ba is the voltage the whole battery system has to provide (input voltage of the spacecraft) and V_{cell} is the voltage of one battery. The decision is to run the spacecraft with 28V since this is the voltage used on the Russian part of the ISS and thus already proven as a proper choice.
The number of batteries which have to be put in parallel can be calculated to $N_{parallel} = \frac{C_{ba}}{C_{cell}}$ where $C_{ba} = \frac{I_{ba}T_e}{DoD\eta} = 928.57Ah$ where $I_{ba} = \frac{P_{max}}{V_{ba}} = 1071.43A$, $DoD = 0.5$ is the depth of discharge and $\eta = 0.9$ is the transmission loss between batteries and load.

To choose the accurate batteries two different types were analyzed: The SAFT Li-ion VES 180 with 50Ah, 3.7V and 1.11kg per battery [8] which are common batteries for satellites and the future ISS batteries [13] [31] the GS YUASA Li-ion LSE 190 with 190Ah, 3.7V and 4.59kg per battery [32]. The results are shown in Table 2

	SAFT VES 180	GS YUASA LSE 190
N_{series}	8	8
$N_{parallel}$	19	5
N_{total}	152	40
Weight	168.72 kg	183.60 kg

Table 2: Batteries

According to reference missions an equivalent fuel cell system would weigh around 1102kg so it is obvious to choose batteries as additional power supply. The final choice of the battery type are the GS YUASA LSE 190. The weight increase is small and due to the less number of batteries needed they are easier to maintain, failures can be detected and fixed faster. Since they are developed to work on the ISS they will also be reliable and already approved there for a few years before the launch of our mission.

9. Attitude Control

The main propulsion system of the spacecraft is done by group 2 using chemical propulsion systems. But to keep the transplanetary vehicle stable during flight and in orbit and to be able to adjust the attitude for docking maneuvers an attitude control system has to be added. In a preliminary state like the spacecraft is in this project it is very hard to calculate the correct inertias for an estimate of the needed torque of the attitude control system. To give a basic estimate only the major moment of inertia is considered here. The transplanetary vehicle can be assumed to be a hollow cylinder with a radius of $4.5m$ on the outside and $4m$ on the inside. Thus the major moment of inertia (around the roll axis) can be calculated to

$$J = m\frac{r_1^2 + r_2^2}{2} = 45000kg\frac{(4m)^2 + (4.5m)^2}{2} = 815625kgm^2 \tag{4}$$

To perform for example docking maneuvers a roll ratio of $45°$ in a quarter of an orbit (22153.76 s) is assumed as a target. The final angular momentum which has to be provided by the attitude control system is calculated to

$$L = J \cdot \omega = 815625kgm^2 \cdot \frac{\frac{1}{4}\pi}{22153.76s} = 28.92\frac{kgm^2}{s} = 28.92Nm\ s \tag{5}$$

As an attitude control system four reaction wheels will be used (4 due to redundancy). For this mission the Honeywell HR14 is chosen since it provides an angular momentum of 50Nm s which is sufficient for our mission. But the major aspect for choosing this type is the estimated life time of this reaction wheel to over 5 years which ensures that this system will work properly during the entire flight. The weight of one reaction wheel is 8.5kg with a consumption of 105W and a rotation speed up to 6000rpm. The size of the reaction wheel is 159 x 366 mm (height x width). [24]

10. Crew Facilities

In order to keep the astronauts healthy and busy during the transplanetary journey, we have several crew facilities on board the manned vehicle. The transplanetary vehicle is divided into four main modules. Figure 8, in Appendix A, gives a clear picture of the vehicle and these main modules. The biggest module, about $6 \times \varnothing 9m$, is the general crew module. This area contains two toilets, a shower area, a kitchen (also used as the meeting area) and the gym.

We have two toilets in case one fails during the cruise and we don't have the means to fix it. These are zero gravity toilets, similar to ones used on the ISS, space shuttle and previous space stations. There are four basic parts to these toilets: the liquid waste

Figure 5: Zero Gravity Toilet [26]

vacuum tube, vacuum chamber, waste storage drawers and solid waste bags. The liquid waste tube is used for urination and works by suction. It has detachable urine receptacles at its end for male and female astronauts. The vacuum chamber is a big cylinder, with a fan at one end that provides suction and has clips on the rim that are used to attach the solid waste collection bags. The solid waste is stored in a waste storage drawer. The liquid waste will be used for radiation shielding around the sleeping compartments, but is separated from the clean water. [27]

Astronauts generally shower once a week. The shower area doesn't actually contain a shower. Astronauts take sponge baths and also bathe using jets of water. Showers generally don't work very well in Space therefore they are discarded. [9] The kitchen is a large worktable that can be used for crew meetings, having breakfast/lunch/dinner, as well as for the astronauts' individual work or free time. There are a few chairs or stools available but sitting down can be rather uncomfortable in space so the astronauts will most likely prefer to float in the characteristic simian hunch near the table. [15]

The gym contains exercising equipment to prevent excessive muscle atrophy. It has two bikes and two treadmills for the legs. Treadmills tend to retain leg muscles better than the bike does, however they are generally more uncomfortable, therefore the bikes have also been provided. There is also a resistance weightlifting device that is operated using air pockets, similar to the one seen on the ISS. This can also be used to work the legs, as well as the upper body[28]. This is a complicated machine and might be prone to failures. However in the coming twenty years the design could be largely improved and enough spare parts and tools can be provided to repair the machine.

It is hard to determine the exact weight of these facilities; therefore a weight of 2000 kg is estimated for the entire transplanetary vehicle furniture, which is mostly in the General Crew Module and the sleeping module. This estimate is taken from the Mass Semi-Direct mission for a four-person crew [33].

Figure 6: Astronaut sleeping strapped to the wall [6]

The general crew area leads to the sleeping areas via a door that can be latched and sealed in case of a breach in either module. The sleeping area is $2 \times \varnothing 6m$ and is surrounded by polyethylene for radiation shielding. It is just one module and doesn't have individual compartments for the astronauts in order to save space and reduce weight. It has enough space for four astronauts. There are sleeping bags attached to the walls, and the astronauts can strap themselves into the bags when they wish to sleep.This is so they don't float around during sleeping hours and injure themselves or damage any equipment [15].Curtains are provided if they wish to separate themselves from the other astronauts. Also available in the sleeping areas are individual lockers for each astronaut to keep personal items. There are also lamplights next to each sleeping bag so the astronauts can read or use their laptops in the sleeping area during personal times [28]. The astronauts can be given an e-book reader and iPod for entertainment, as they generally prefer to read rather than watch TV or movies. The crew and personal belongings are estimated to weigh 400 kg [33]. There are also oxygen tanks and water supply placed on either side of the sleeping area, as seen in Figure 8.

The sleeping area leads to the Science Lab module, again via a door that can be sealed. The Science Lab is $2 \times \varnothing 9m$ and contains the science payload that is required for the transplanetary cruise research. This is more thoroughly explained by Group 4, as they are responsible for en route research.The science lab leads to the storage area, seen in Figure 8. This is where the astronauts store food, additional oxygen tanks, EVA suits, and other supplies. The storage area is also $2 \times \varnothing 9m$ in size, which is more than enough storage space. The supplies are kept on separate shelves so the astronauts have easy access to food, tools and resupplies should they need them. We store 4000 kg of spares and 6000 kg of consumables at the start of the journey. The EVA suits weigh 400 kg [33].

Figure 7: Space Food stored in Storage Area [26]

There is an additional module attached to the topof the General Crew Module. It has two compartments, as seen in Figure 8. The top compartment is for EVA preparation with an airlock at the end. It is used for EVAs, as a docking module for resupply spacecraft, and also to dock with the MAV and MDV. The bottom compartment is the Command Centre. This is the main operational compartment. It is used by astronauts to check that all modules are functioning correctly and that the vehicle in on the right trajectory. It is also used to communicate with Earth and to receive signals from the Mars Surface Habitat and other docking vehicles. There is another small airlock integrated on the side of the General Crew Module, which is used mainly as a docking point for the ERV. The ERV stays with the transplanetary vehicle during its journey, as explained in the Introduction, and therefore occupies this module permanently.

There can also be a robotic arm that works with the top airlock, and could be used to assist astronauts during EVA's. It could also be used to assist in the docking of the MDV and MAV, as well as the ERV at the start of the journey. This arm would be similar to the European Robotic Arm (ERA). It weighs about 630 kg [16]. It is not necessary and is therefore not included in Figure 8, however it could be implemented if deemed necessary.

11. Off nominal scenario

During a mission in space a lot of things can go wrong. A few examples are pressure leakage, navigation failure, thermal protection failure, failure of the life support system or a power shutdown. It is not possible to address all these aspects in this report and some have already been mentioned in the other parts so just one crucial failure is considered here. We therefore selected a sudden power shutdown in case a meteoroid hits and breaks one of the solar panels. Since there are four solar arrays that would not lead to a complete loss of power however it would limit the power capacity especially if more than one solar array is affected. In this case a few systems can be shut down to save energy. Especially the research facilities which need a lot of power can be shut down so we have enough power for the other systems. In the worst case everything other than the life support system and the attitude control system can be shut down temporarily until the crew has the opportunity to repair the solar panels. Tools and spares for repairing the solar arrays will be carried in the spacecraft.

12. Conclusion

As presented the designed spacecraft is based on already known or soon available technology. A key driving parameter was that all technology is available by the time the spacecraft is built and all crucial technology should be tried and tested for a long term mission. We therefore avoided using too futuristic / unfeasible technology which just exists in research. Our main target was a successful mission and a safe return of all the crew. Thus we relied on a robust but simple design with redundant measures present in case of failures. It was tried to address everything important for a successful mission but of course the design and the mass estimate could be improved with further time and research. But we showed that the basic concept and mission is feasible within the next 20 years.

A. Ship Design

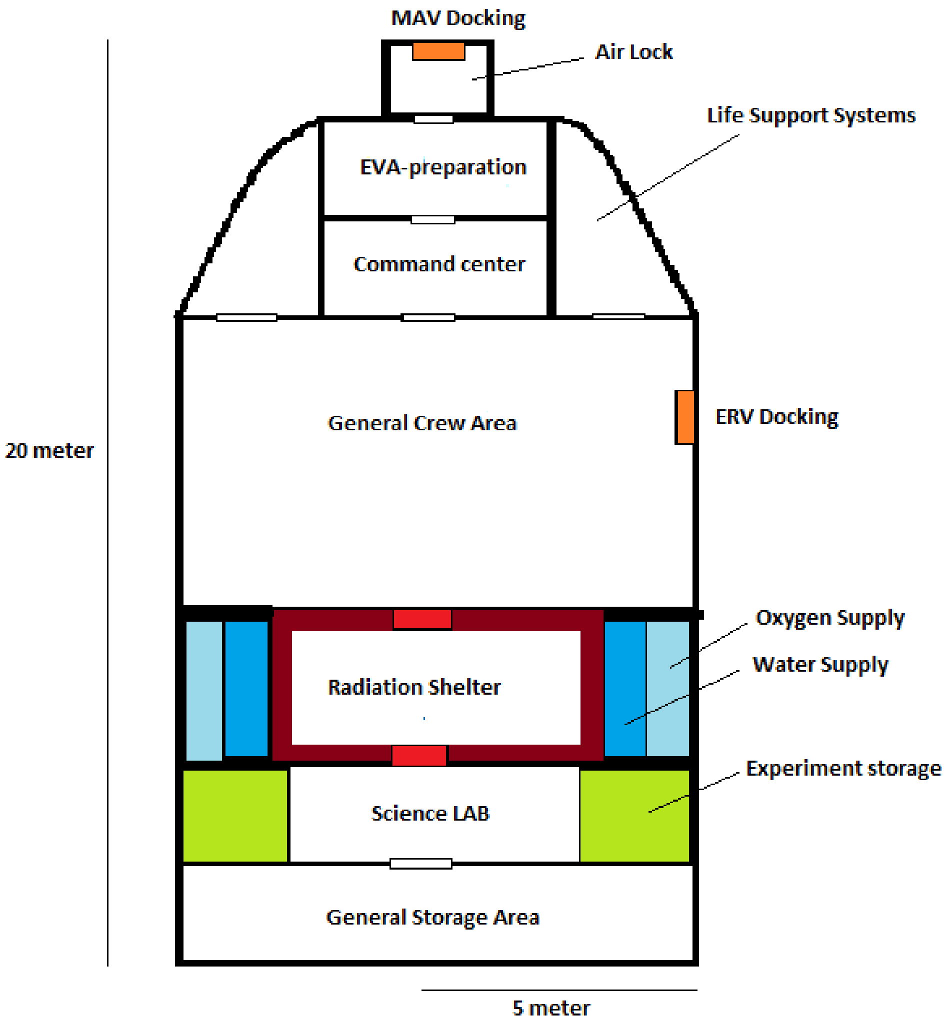

Figure 8: Transit habitat design with main compartments

Figure 8 Displays the general outline of the transit habitat.

The upper section contains a EVA-preparation room and an airlock, which will also be used for docking with the Mars ascent and decent vehicles. There is also a command center, where communication and navigation will be performed, and additional compartments allotted to life support systems.

The middle section contains the general crew area, where there also is a docking port for the Earth return vehicle, which will be attached to the spaceship up until the final parts of the mission. Below the general crew area is a radiation shelter, surrounded by water- and oxygen-tanks.

The lower section contains a science lab and a general storage area.

B. Mass Estimation

This appendix presents the mass estimations made for the transit habitat. Since it is difficult to do an exact mass prediction at this conceptual level, some of the numbers presented here are rough estimates.

Based on the sketch in Appendix A, assuming the modules have a cylindrical form, the surface area of the transit habitat is estimated to 615 m2. With this in mind, the weight of the surrounding pressure and micro meteoroid shields are given in Table 3.

Unit weight per square meter [kg]	20
Pressure and meteoroid shield, total weight [kg]	12300

Table 3: Exterior shielding and thermal control

The required weight of the radiation shelter during solar maximum and solar minimum is presented in Table 4. Only the shields meant for solar maximum will be included in the transit habitat.

Polyethylene weight solar maximum [kg/m^2]	70
Polyethylene weight solar minimum [kg/m^2]	300
Shelter radius [m]	3
Shelter height [m]	2
Shelter, total weight solar minimum [kg]	28274.33
Shelter, total weight solar maximum [kg]	6597.34

Table 4: Radiation shelter

The estimated weight of the power supply system, including solar panels is shown in Table 5. The power supply system is meant to deliver 30 kW, with solar arrays covering an area of approximately $450m^2$.

Solar arrays [kg]	1682
Batteries [kg]	183.6
Radiator [kg]	190
Power supply, total weight [kg]	2055.6

Table 5: Power supply system

An approximated weight of the food and water supply for the inbound and outbound journeys is given in Table 6. The weights are calculated for an outbound journey of 304 days to Mars, and an inbound journey of 160 days back to Earth. Additionally, 112 days are added in case of a missed return window, these numbers have been provided by Group 2.

Number of crew members	4
Outbound journey [days]	304
Inound journey [days]	160
Additional time, missed return window [days]	112
Outbound supply storage time [days]	576
Inbound supply storage time [days]	272
Dry food consumption per person and day [kg]	0.5
Water loss per day [kg]	1
Required water per person and day (drinking & food) [kg]	3
Hygiene water per day (all) [kg]	101.2
Water for cooling system [kg]	500
Total weight of dry food (outbound) [kg]	1267.2
Total weight of dry food (inbound) [kg]	598.4
Total weight of water for drinking (outbound) [kg]	13.2
Total weight of water for drinking (inbound) [kg]	13.2
Total water loss outbound [kg]	576
Total water loss inbound [kg]	272
Total weight, water and food (outbound) [kg]	2457.6
Total weight, water and food (inbound) [kg]	1484.8

Table 6: Water & food supply

Table 7 shows the estimated weight of the ship interior, some of these numbers are quite uncertain.

Furniture [kg]	2000
EVA-suits [kg]	400
Robotic arm [kg]	630
Spares [kg]	4000
Crew + personal belongings [kg]	400
Load bearing structure + interior walls [kg]	4000
Total weight of ship interior [kg]	11430

Table 7: Ship interior

Additional system weights are displayed in Table 8, including life support, thermal protection and communication equipments.

Thermal shielding system [kg]	267
Life support [kg]	3500
Resupply gasses (outbound) [kg]	4000
Resupply gasses (inbound) [kg]	2000
Communication & Navigation [kg]	500
Experimental payload [kg]	1500

Table 8: Additional system weights

Summarizing all the above mentioned weights give the total inbound and outbound weights presented in Table 9.

Empty weight of transit habitat [kg]	29537.95
Cargo weight (outbound) [kg]	15069.6
Total weight (outbound) [kg]	44607.54
Total weight (inbound) [kg]	41634.74

Table 9: Total weight

C. Life Support

In this Appendix the specific systems used for the Life support system are listed. The Atmosphere management is listed in Table 10, the water management in Table 11, the waste processing in Table 12 and food supply in Table 13. Finally a schematic picture of the life support system is presented in Figure 9

CO_2 removal	2-bed molecular sieve (2-BMS)
CO_2 reduction	Sabatier & Solid oxide electrolysis (SOE)
CO_2 provision	Solid polymer water electrolysis (SPWE)
N_2 provision	Cryogenic Tanks
Atnomsphere monitoring & control	Mass spectrometer / gas Chromatograph
Ventilation	Air diffusers and intakes
Temperature and humidity	Condensing heat exchanger
Fire detection and suppression	Scattering smoke detector CO_2 fire suppression system UV/visible IR flame detector

Table 10: AM subsystem technologies [18]

Hygiene and potable water treatment	Reverse osmosis Ultra filtration Milli Q absorbtion beds Regenerable microbial check valve (Iodine)
Urine treatment	Vapor compression distillation
Water monitoring	Electronic nose Ion specific electrodes Total organic carbon Conductivity Test kits (pH, specific compounds)
Water generation	Sabatier reaction

Table 11: Water management technologies [18]

Biologically decomposable solids	Gasification
Non-decomposable solids	Pyrolysis

Table 12: Waste processing technologies [18]

Supply food	Rehydratable food

Table 13: Food supply options [18]

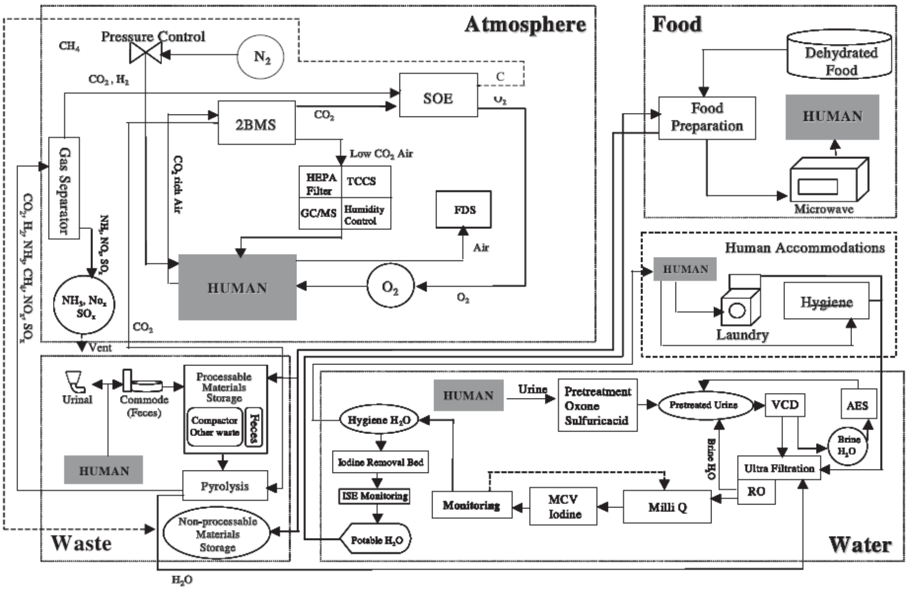

Figure 9: Life support system schematic [18]

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