CRANSPACE

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1. Executive Summary

The CranSpace team have designed a Mars flyby mission suitable for a two-person crew with a proposed launch date before the end of newly 2016 elected presidents prospective second term in office. The focus of the mission was to reduce cost, reduce risk and provide a comfortable environment for the astronauts on-board.

Our designed trajectory software revealed the Delta-V needed for a Mars injection burn for a mission length of under 600 days was too high after the early 2020's. Another issue was after the early 2020's solar maximum is approaching. This would increase the mass of radiation shielding required. These two constraints limited our launch window to the early 2020's. This gives little time for development of new technologies therefore the mission is made entirely of modifications to existing or upcoming technology which will be flight proven before our launch date.

This methodology as well as reducing time of development also reduces cost and risk to due to previous flight heritage of the hardware in question. The spacecraft design will utilize an International Space Station (ISS) module for the main habitat area. With minor internal modifications for radiation and thermal and external modifications for life support by the addition of a service module. The crew will launch into orbit using the Orion module this will also act as their re-entry capsule. The Orion capsule was chosen as it is interplanetary rated and furthest along its development schedule. The Exploration Upper Stage (EUS) of the Space Launch System (SLS) will be used as the on-orbit propulsion system for the Trans Mars Injection (TMI).

The Orion crew capsule and the Habitat module will be launched to Low Earth Orbit (LEO) together using a Falcon heavy launcher with a modified payload bay. The Orion capsule and the habitat module will be arranged so the crew can eject in the Orion safely during a launch abort. In LEO, the Orion capsule will undock from the habitat module and perform a similar manoeuvre to the Apollo lunar lander and re-dock to the habitat module.

The Exploration Upper Stage used for the Trans Mars Injection (TMI) burn will be launched as the upper stage of the SLS launcher, configured to have a very small payload and additional propellant. Once in LEO the AOCS of the EUS will be used to dock with the habitat module and the Orion. After final system checks are carried out, the EUS will reignite, burning this additional propellant to provide the required 4.1 kms^{-1} of Δv.

The trajectory of the spacecraft will involve a Venus flyby as well as the Mars flyby before returning to Earth. The chosen trajectory is longer in duration but the additional mass of consumables is offset by the reduced propellant mass required associated with the lower Delta-V.

To reduce the cost and risk of the mission, especially with the short development time, only existing proven technology will be used for key systems such as communication and life support. That said the mission offers a unique opportunity where by the spacecraft could play host to experimental technologies and systems. Such systems on the mission include biological based bioreactors for life support as well as laser communication. As the technology readiness level is not suitable for these promising new technologies to be relied upon as key primary systems, the mission will allow them to demonstrate their capabilities as possible alternatives for future manned missions.

No person has ever left the near vicinity of the Earth such that they view the Earth as a tiny speck in the sky. As well as technology demonstrations, the mission gives a unique view on the psychological stresses imposed upon the astronauts. The so-called Earth out of sight syndrome will carefully be examined as this mission offers the first glimpse into its effects on the astronauts' daily tasks and morale. Crew wellbeing is an important factor in the design of the mission and resources have been allocated to allow the astronauts to be able to feel reconnected to Earth.

During the mission, the crew will experience solar minimum space weather conditions. The mass of radiation shielding required was therefore reduced, but is still a significant proportion of the overall mass. To reduce the radiation shielding mass further, their sleeping quarters will contain a higher level of radiation shielding acting as radiation vaults, providing additional radiation protection. A normal sleeping pattern is assumed of eight hours a day within the vaults. This time spent in the vaults each day sleeping/relaxing means the astronauts receive an acceptable dosage within the allowance of the 730 mSv requirements. The vaults are also required in the event of a major solar event to provide the additional shielding required.

During the end of the mission the crew will seal themselves and any relevant scientific experiments into the Orion Capsule and undock from the habitat module. The Orion's on-board propellant will allow the crew to decelerate for speeds suitable for an aero capture around Earth. As this is a low cost proving mission, the habitat module and EUS will not be reused at the end of life. A controlled Earth re-entry will eliminate the two sections from orbit.

2. Requirements

The driving force behind the requirements are mainly based on the mission brief provided by the Mars Society [1], such as restrictions on methods of currently available propulsion and crew size. Furthermore, the biggest constraint on the design of the mission was the strict usage of established technology. Additional requirements can be derived from the mission characteristics, for example the capsule having to survive re-entry due to the high-energy return. Another source for mission constraints come from the NASA Advanced Life Support Baseline [2].

Overall the main system requirements are shown as follows:

SYS-REQ-1	The spacecraft shall be compatible with existing launchers.
SYS-REQ-2	The spacecraft shall be able to communicate with the Deep Space Network.

SYS-REQ-3	The spacecraft shall have the capability of surviving hypersonic re-entry.
SYS-REQ-4	The spacecraft shall provide sufficient radiation protection for a maximum exposure of 730 mSv extended-period dose.
SYS-REQ-5	The spacecraft shall provide sufficient life support for a crew of 2, for the entire mission duration.
SYS-REQ-6	The spacecraft shall provide a minimum habitable volume of 5.1 m^3 per person.
SYS-REQ-7	The spacecraft shall have no single point failures in its life support system.
SYS-REQ-8	The spacecraft shall consist only of established technology.
SYS-REQ-9	The spacecraft shall not use any of the following potential propulsive methods: nuclear electric, nuclear thermal, solar thermal, solar sails.
SYS-REQ-10	The mission shall be completed by the end of 2024.

3. Operation and timeline

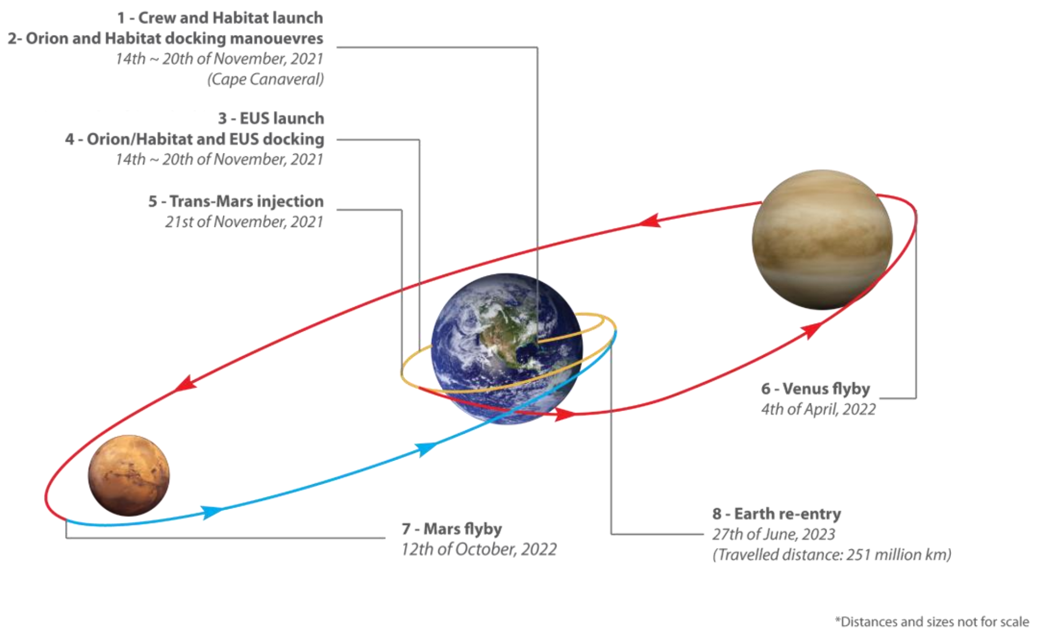


Figure 1 - Operation and timeline

3.1. Mass Breakdown

PPLM				
Variable	**Mass**	**Margin**	**Mass Margin**	
Habitat structure	4082	10%	4490.2	kg
Water	350	10%	385	kg
Life Support*	1343.08	14.89%	1578.1	kg
Exercise	134	5%	140.7	kg
Food	720	5%	756	kg
Radiation Shielding	6900	-	6900	kg
Service Module Structure	450	10%	495	kg
Solar Arrays	50	20%	60	kg
Communication	73.9	5%	77.6	kg
Science	800	25%	1000	kg
Thermal Protection	3000	33%	4000	kg
Misc*	213	18.26%	230	kg
Total	**18116.0**	**-**	**20112.6**	kg
ORION				
Variable	**Mass**	**Margin**	**Mass Margin**	
Crew	140	18%	170	kg
Capsule	10387	5%	10906	kg
Propellant Engine	2000	10%	2200	kg
AOCS Propellant	1500	25%	2000	kg
Service Module	5875	5%	6160	kg
Total	**19902**	**5.41%**	**21436**	**kg**
Propulsion				
Variable	**Mass**	**Margin**	**Mass Margin**	
Inert Mass	13600	-	13600	kg
Propellant	88000	3.90%	91400	kg
Total	**101600**	**-**	**105000**	**kg**
TOTAL On Orbit Mass	139618.0	4.73%	146548.6	kg
TOTAL Dry Mass	51618.0	6.40%	55148.6	kg
TOTAL Payload	38018.0	8.50%	41548.6	kg

*Misc				
Variable	**Mass**	**Margin**	**Mass Margin**	
Medical	80	20%	100	kg
Personal	20	-	20	kg
Clothes	10	-	10	kg
Wipes	13	35%	20	kg
3D Printer	15	25%	20	kg
Heat melt compactor	25	20%	30	kg
Fillament	10	-	10	kg
Sanitary	40	20%	50	kg
Total	**213**	**18.26%**	**260**	**kg**
***Life Support**				
Variable	**Mass**	**Margin**	**Mass Margin**	
water reclaimer	400	15%	460	kg
CO2 Removal assembly	100	20%	120	kg
O2 Production unit	100	20%	120	kg
Algae HArP Reactor	20	20%	24	kg
valves and tanks	50	20%	60	kg
air purifier	4.08	10%	4.5	kg
waste removal	100	10%	110	kg
Fire supression	8	20%	9.6	kg
Atmosphere				kg
Water for O2	416	20%	500	kg
Initial O2	25	0	25	kg
N2	120	20%	145	kg
Total	**1343**	**14.9%**	**1578.1**	**kg**

Table 1 - Mass breakdown

4. Configuration

4.1. External

For our mission, we wanted to reduce costs as well as ensure we met our launch date and reduced risk. This could be achieved by cutting out the development of new technology and used already existing (or already under-development) technology.

4.1.1. Orion Capsule

Two options were available for the crew re-entry vehicle – the Orion or the Dragon v2. Both capsules are projected to be available for our 2021 launch window, and Dragon is projected to have a lower mass than Orion. However, there is significantly more uncertainty about the system specifications of the Dragon capsule, especially in relation to its suitability for long-duration crewed interplanetary flights. The Orion spacecraft also has a more capable service module, as well as larger windows for a better view of Mars and Venus during the flyby. Because of this, the Orion capsule was chosen. The Orion will be re-designed internally to accommodate two crew members, instead of the six it is currently designed to carry. A direct result of this is some weight saving in the Orion, which will be used to carry supplies for the mission such as food and medical equipment.[3]

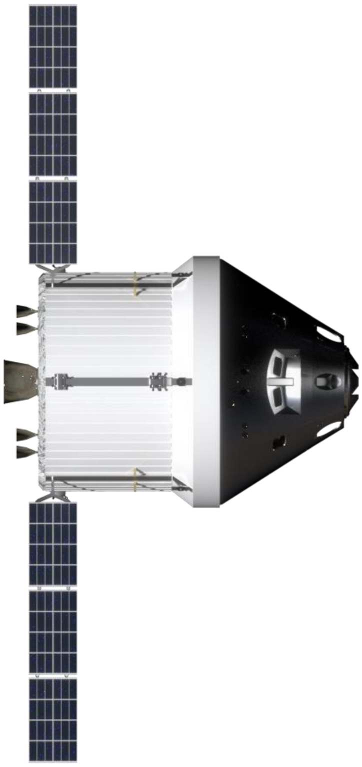


Figure 2 – Orion capsule and service module

The Orion only has a volume of 8.95 m^3 and as such, does not meet the tolerable living space values for long duration missions – 5.1 m^3 per person – as set out by NASA [2]. The Orion will therefore be used for launching the crew and returning them to Earth, but a habitat module will be required to meet this living space requirement.

4.1.2. The Multi-Purpose Logistics Module (MPLM)

While choosing the habitat module, we considered both the MPLM and the Bigelow inflatable modules. We concluded the MPLM was the better choice due to the inflatable module's higher leak rate and relatively low technology readiness level.

The MPLM was chosen as the main habitat module for the mission. With a length of 6.6 m, a width of 4.57 m, and a habitable volume of 31 m^3 after the addition of necessary equipment, it is above the requirements set out by NASA for long duration missions: a minimum of 5.1 m^3 per person[2].

The MPLM was also chosen because of its heritage. With three previously constructed and with 12 flight-proving missions, there is significant experience in the manufacture and operation of the MPLM, thus increasing the confidence in the design and reducing development costs[4].

A further advantage of the MPLM is that it is essentially an empty shell – this reduces complications in redesigning the interior to meet the mission needs. The design of the interior is discussed in further detail in section 4.2.

The MPLM will need to be modified for interplanetary missions due to its lack of adequate radiation and thermal shielding; this is discussed later in section 6. In addition to adding radiation shielding, a service module will also be added due to the fact the service module for the Orion is not capable of supporting the astronauts for the 580-day mission. The service module will be based on a similar design used on the Cygnus/ATV spacecraft.

The MPLM is also equipped with the common berthing mechanism which allows compatible docking with the Orion capsule with the use of a pre-existing adapter for the Orion's NASA Docking System port. As an alternative, the MPLM could be modified to replace its CBM with an NDS port.

Figure 3 – Habitat module and service module

4.1.3. Space Launch System Exploration Upper Stage

No single launch vehicle has the capability to launch 41.6 mT to a C3 of 20.25 km^2/s^2. Two launch options were hence considered. The first option was to dock the spacecraft with multiple smaller upper stages of existing launch vehicles, such as the Delta IV or Falcon 9. To reduce complexity and risk, however, it was decided to use the second strategy of a two-launch approach: one of the Falcon Heavy to launch the spacecraft, and one of the upcoming SLS block 1B to launch the propulsion system used for TMI. It was initially assumed that this would consist of a second EUS (Exploration Upper Stage) launched as the payload, but further work showed that the SLS would be able to fulfil this mission without the extensive modifications to launch procedures and equipment required to add an extra upper stage to the vehicle.

Maximum-payload launches of the SLS block 1B to LEO require the EUS not to be filled to maximum capacity: an EUS with a full load of propellant has a very low acceleration and long burn time, and so would cause significant gravity losses if used for more than the final almost-horizontal portion of the ascent trajectory. Indeed, for a maximum-payload launch to LEO, the baseline trajectory assumed gives that the EUS would typically have approximately 46 mT of useable propellant, compared to its maximum capacity of 125 mT of useable propellant. By instead having a very small payload – the mass budget for modifications to the EUS was set at 1 mT – and loading the EUS propellant tanks to maximum capacity, it was found that enough propellant would remain in the tanks in LEO to be used in the TMI burn.

A launch trajectory simulation by Pietrobon [5] was used to determine the total Δv required for an SLS block 1B launch to LEO. This simulation used a payload mass of 97.1 mT, rather than the SLS's 105 mT advertised maximum payload LEO capacity[6], and hence should give conservative results. By altering the upper stage propellant and payload masses and assuming that the total Δv to orbit remains the same, it was found that this would leave 89.3 tonnes of useable propellant in the EUS, as shown in Table 2. This is sufficient for the TMI burn, with a 118 ms^{-1} margin for gravity losses and underperformance.

Scenario 1: 97.1 mT to LEO				
	Mean Vacuum I_{sp} (s)	**Initial mass (mT)**	**Final mass (mT)**	**Δv (ms^{-1})**
Boosters & Core	298.1	2700.1	1176.7	2429.2
Core Only	452.2	975.0	277.4	5575.3
EUS (45.8 mT propellant)	462.0	157.4	111.6	1557.3
Total Δv				**9561.8**

Scenario 2: 1 mT payload + 89.3 mT unused propellant to LEO				
	Mean Vacuum I_{sp} (s)	**Initial mass (mT)**	**Final mass (mT)**	**Δv (ms^{-1})**
Boosters & Core	298.1	2700.1	1160.0	2452.8
Core Only	452.2	958.2	260.7	5774.0
EUS (45.8 mT propellant)	462.0	140.7	104.8	1335.0
Total Δv				**9561.8**

Table 2 - Maximum-payload LEO launch scenario compared with low-payload additional-propellant launch schenario.

It may be possible to increase this propellant margin: since 12 mT of launch mass margin is available on the Falcon Heavy, it would be possible to launch the spacecraft into an elliptical parking orbit with a higher apogee. It was found that a TMI burn at the perigee of this elliptical orbit would require less propellant, and that the propellant savings would outweigh the additional propellant burned to place the EUS in this higher orbit. Further analysis would be required to trade these advantages off against the additional challenges of rendezvous in an elliptical orbit and the risks of additional radiation exposure due to repeated passes through the van Allen belts.

4.1.4. Launch Configuration

The Orion capsule, habitat module, and EUS will be assembled into their flight configuration in LEO. The first launch, using a Falcon Heavy with a custom fairing as shown in Figure 4, will launch the Orion module, MPLM and the additional service module into orbit. This will take place a significant time, assumed to be two weeks, before the TMI burn window, to allow system checks to be carried out on the spacecraft.

Figure 4 - Orion and MPLM atop Falcon Heavy in custom fairing

The second launch will take place as close as possible to the TMI burn window in order to minimise propellant boiloff, and will consist of an SLS block 1B with a “payload” of additional propellant and modifications and additions to the EUS. These modifications include a docking structure to interface with the MPLM’s service module, along with additional insulation to reduce propellant boiloff and any other additional hardware such as extra batteries required to extend the EUS’s on-orbit lifetime.

A brief analysis was also conducted to estimate propellant boiloff, potentially a significant issue for LOX/LH2 stages such as the EUS. While a detailed thermal analysis of propellant boiloff is beyond the scope of this report, a preliminary analysis of the impact of adding two deployable conical “sunshades” of multi-layer insulation suggested that propellant boiloff could be reduced to perhaps 0.4% per day – approximately 15 kg per hour, or roughly 180 kg in total given the assumptions on docking procedure duration – with a sunshade mass of approximately 300 kg. This is, however, one of the largest areas of uncertainty of the launch architecture, as propellant margins are already tight, and further analysis here would be worthwhile for any follow-up studies.

4.1.5. Docking

For launch the Orion will be connected to the MPLM with the heatshield face down. When in LEO, the Orion module will undock with the MPLM, re-orientate itself and re-dock with the MPLM to allow the astronauts to enter the MPLM habitat module.

Once the Orion has re-docked with the MPLM, the Orion and MPLM will then dock with the EUS as shown in Figure 5. Considering the docking procedures used by the ISS it is assumed from launch to completion of the entire docking procedure will take 8 hours, after which system checks will be completed before the EUS is used for the TMI.

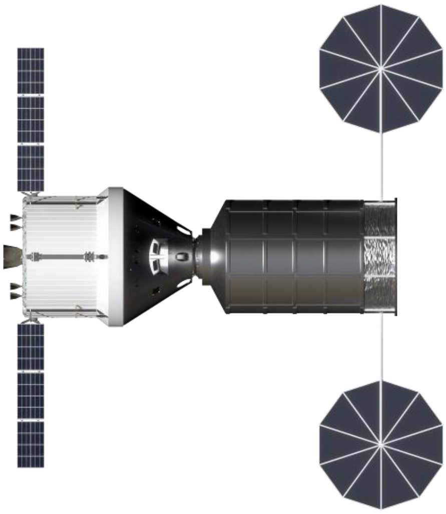


Figure 5 – Orion docked with the habitat module in on-orbit configuration.

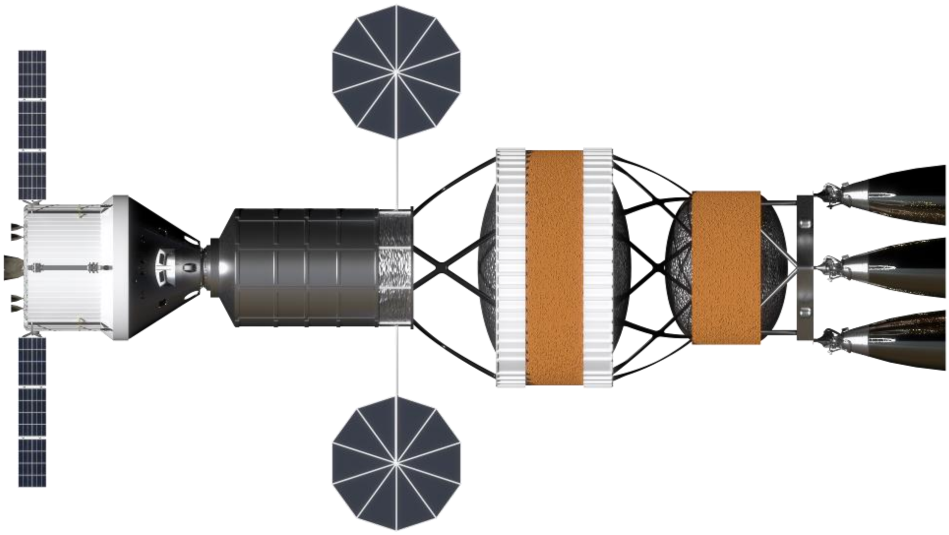


Figure 6–Complete on-orbit spacecraft configuration.

4.2. Internal Configuration

As discussed earlier the MPLM is ideal as the interior can be easily redesigned. The habitat will need to be able to support the crew for a 580-day mission. After consideration of human spaceflight requirements, the interior layout has been based on the design for the Russian Zvezda module of the ISS which is a self-contained module that includes everything required for the crew to survive.[7]

The interior of the MPLM interior will include:

- Work compartment which includes the sleeping quarters/radiation vaults as well as the kitchen
- Toilet and hygiene facilities
- Fitness area
- Storage
- Refrigerator and freezer
- Guidance and control segment
- Elektron system to condense the humidity and waste water to provide oxygen
- A “Veggie” growth unit
- A biological life support experiment

5. Trajectory

The objective of this section is to research the optimal launch window for the mission. The optimum launch window will be a compromise between the Δv and the travelling time. Reducing both implies less complexity, risk and costs. There are two scenarios that have been studied for the mission, a direct fly-by to Mars, or a gravity assist around Venus with a subsequent fly-by of Mars. An algorithm was created to compute the minimum Δv for a direct fly-by either within or outside the sphere of influence (SOI) of Mars.

The algorithm uses 3D patched conics and the Lambert method to solve the astrodynamics patched conic equations and compute the overall Δv. It also uses Lagrange multipliers to find a minimum Delta-V that satisfies the restricted constraints.

The constraints of the algorithm are as follows:

- The arrival date to Mars should be earlier then the arrival date to Earth
- The maximum length of the trip is 600 days
- The arrival dates should be later than the launching date.

The algorithm computes the best orbit for a given launch date from Earth, where a loop computes all the best orbits from 2020-01-01 to 2024-12-31, and the orbits with minimum Delta-V consumption chosen as possible launch windows. A flowchart of how the algorithm works is shown in Figure 7.

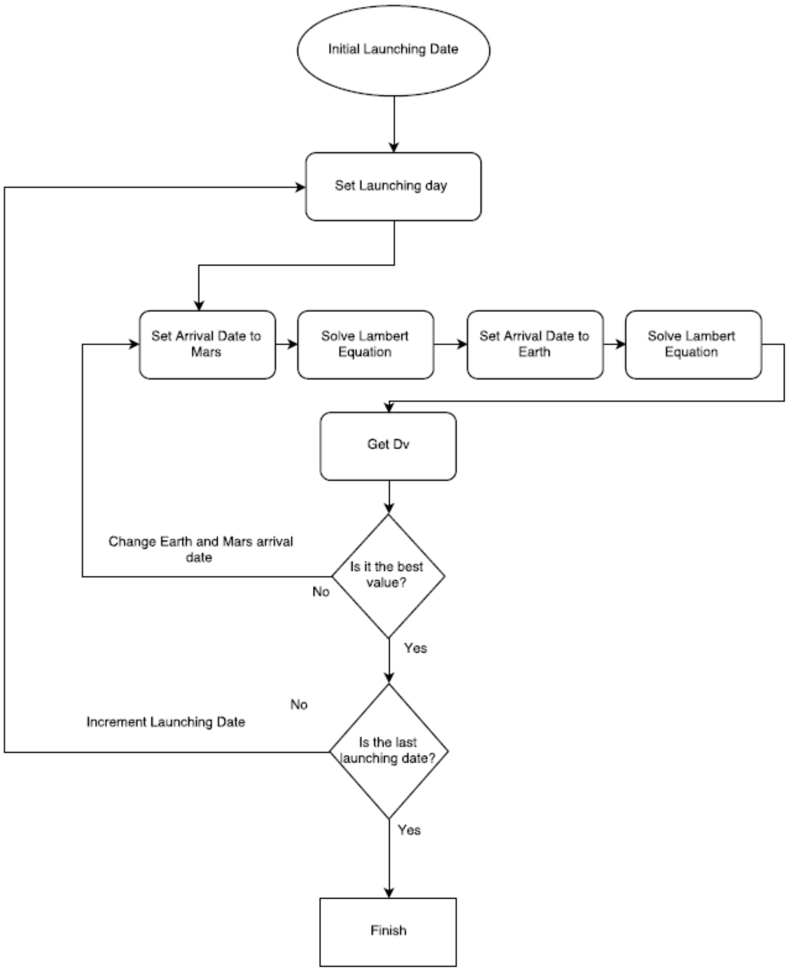


Figure 7 - Launch window algorithm

The Delta-V computed is referred to as required from the parking orbit. The parking orbit is a circular LEO orbit at 250km of altitude. The orbital velocity is 7.755 km/s.

The launch window found with the minimum Delta-V departs Earth on 2020-01-07, and will be a free return trajectory arriving at Mars on 2020-09-16, reaching Earth on 2021-07-08. The Δv is computed from the LEO; therefore, the Δv of the launcher is not considered. The best results from the algorithm are presented in table 3. Figure 7 demonstrates a schematic of the best direct orbital transfer to Mars found which has a total of 548 travelling days.

Departure Date	Arrival Mars	Arrival Earth	DV (km/s)	Travelling Days
2020-01-07	2020-09-16	2021-07-08	6.17	548
2021-12-08	2022-07-30	2022-12-08	8.56	365
2021-12-29	2022-08-02	2022-12-31	9.75	367
2022-03-17	2022-11-12	2023-09-17	8.44	549

Table 3 - Lowest-Δv results computed by algorithm

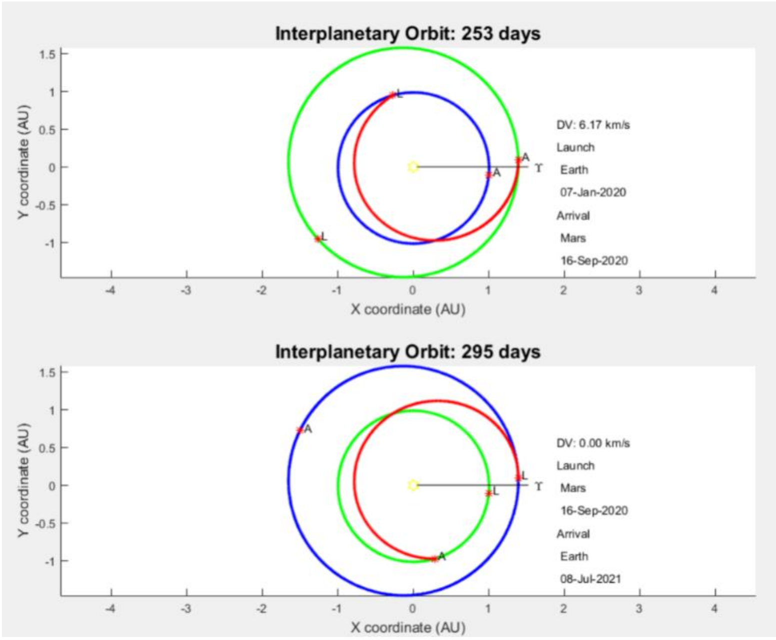


Figure 8 - Direct Mars Fly-by orbit diagram

This trajectory requires a very small burn at Mars, and the flyby is done outside the SOI of Mars.

The second option is performing a gravity assist manoeuvre at Venus prior to Mars. During 2021 the relative planetary positions of Earth, Venus and Mars will be suitable for performing this manoeuvre at low Δv costs[8].

Launch Date (mm/dd/yyyy)	$V_{\infty,Launch}$ (km/s)	C3 (km^2/s^2)	TOF (days)	$V_{\infty,Arrival}$ (km/s)	V_{Entry} (km/s)
11/22/2021	4.50	20.25	582	6.53	12.85
12/03/2021	5.10	26.01	579	6.95	13.07
12/04/2021	5.20	27.04	569	6.46	12.82
12/08/2021	5.50	30.25	566	6.55	12.87
12/14/2021	6.00	36.00	564	6.81	13.00
12/15/2021	6.10	37.21	563	6.78	12.99
12/16/2021	6.20	38.44	561	6.77	12.98
12/17/2021	6.30	39.69	560	6.78	12.99
12/19/2021	6.50	42.25	559	6.84	13.02

Table 4 - Best opportunities for Venus gravity assist before Mars flyby [8]

The payload figures in Table 5 assume that 89.3 mT of useable propellant is available in the EUS, as discussed in Section 4.1.3.

Immediate issues with the direct flyby is that although the 2020 launch date offers the lowest Δv, it is too early for our design, since the scheduled development timeline of the SLS, EUS, and Orion are scheduled for completion in early 2021. These three components are crucial to the low cost and risk design of the mission. The direct trajectory's Delta-V after 2020 increase significantly by over 3 km/s in one case. Table 5 compares the only viable direct trajectory with the Venus flyby. It is shown that the mass of propellant required to perform the TMI is over the maximum 89.3 mT propellant allowance. The mission including a Venus flyby, however, requires less propellant and is within the mission budget. Therefore, the optimal TMI burn will take place on November 22nd 2021.

	V_∞ (m/s)	Δv (m/s)	Δv margin	Maximum Payload Mass (mT)	Margin (mT)	Payload Mass Margin
2021-11-22	4500	4100.8	2%	42.4	0.8	2%
2021-12-03	5100	4341.2	2%	37.4	-4.2	-10%

Table 5 - Venus fly-by compared with direct trajectory

5.1. AOCS

Requirements:

- The attitude and orbital control subsystem shall maintain the spacecraft attitude required for thermal control, power generation, and scientific observations of Mars and Venus during flybys.
- It shall compensate for disturbances to maintain the spacecraft's trajectory.

For the majority of its flight, the spacecraft will be travelling through interplanetary space, so the only substantial perturbation affecting it will be solar radiation pressure[9]. As a rough upper bound, assuming the centre of solar radiation pressure is 5 m away from the centre of mass, the resulting torque at perihelion is approximately 4×10^{-3} Nm. This gives 200 kNs as an upper bound for the angular impulse over the entire mission, which (assuming the RCS thrusters are also approximately 5 m from the centre of mass) will require approximately 15 kg of propellant to compensate for.

The largest rotational control manoeuvres will take place during the Venus and Mars flybys, to maintain the correct attitude for scientific observations and crew viewing. However, due to the low rotational rates required, the propellant requirements are also assumed to be small.

Small control moment gyros may also be included for fine attitude control and compensating (with periodic desaturation) for small torques such as the SRP torque above. Given the small torques required due to the low agility requirements of the spacecraft, a set of small CMGs (whose mass and power consumption is assumed to be equal to the CMG 15-45S[10], being 72 kg and 100 W respectively) is likely to suffice.

A reasonable Delta-V budget for mid-flight course corrections is 100 m/s. Given the mass of the spacecraft and the 2650 m/s specific impulse of Orion's RCS thrusters[11],the spacecraft requires 2000 kg of propellant for mid-flight course corrections.

Additional AOCS propellant will be required to dock with the Exploration Upper Stage. However, as this is burned before the TMI burn, this is not included in the mass breakdown for the attitude control subsystem.

5.2. Re-Entry

The final challenge of the mission is to have to crew return safely to Earth. Re-entry conditions are one of the most extreme conditions on the Earth with very high velocity, temperature, and deceleration etc. Moreover, this mission will have the fastest manned re-entry initial speed ever, 12.85 km/s[8] (previously held by Apollo 10 with 11.8 km/s).To survive these extreme conditions, the Orion re-entry module employs an ablative heat shield, made with composite materials.

The re-entry velocity is fixed by the chosen trajectory, but we must choose the entry flight-path angle (FPA) to reduce the load applied to the capsule and the crew.

Parameter	Value
Mass	10000 kg
Initial entry velocity	12.85 km/s
Drag coefficient	2.2
L/D ratio	0.2
Aerodynamic surface	19.6 m2
Altitude	150 km

Table 6 – parameters for re-entry simulation

The simulation will be done for several FPA for optimization, but keeping the trajectory into the entry corridor. The chosen trajectory uses the skip-entry technique in order to reduce the heat load and the deceleration [12]. The Orion is planned to be able to perform skip-entry. This method can be done with a small set of FPA between the ballistic entry (lower FPA) and no-capture trajectories (higher FPA).

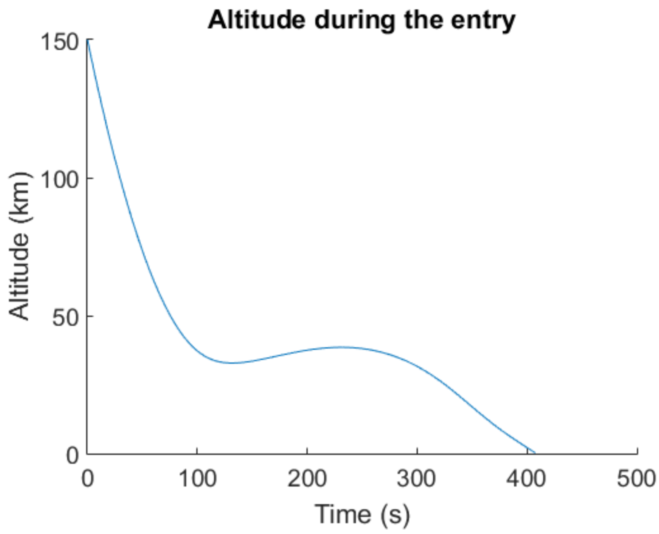


Figure 9 - Evolution of altitude

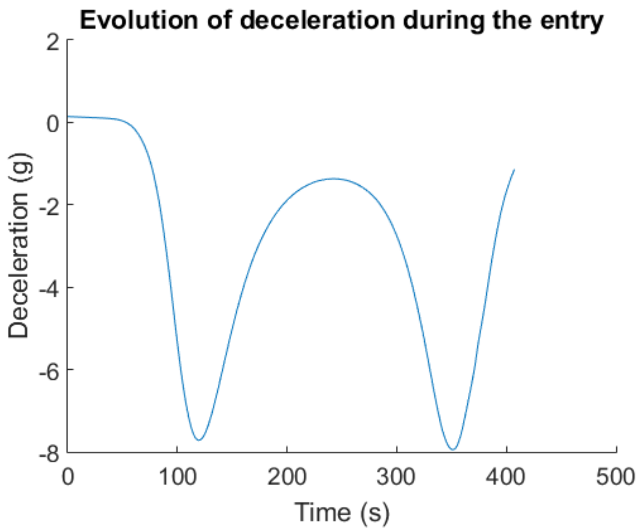


Figure 10 - Maximum deceleration

The maximum deceleration undergone by the crew is 8 g; deceleration above 5 g is limited to two times 50 seconds. The maximum heat transfer occurs during the first part of the entry [13]–[15].
The crew will finally land safely in the Orion Crew Module 7 minutes after the beginning of the entry.

6. Radiation and Thermal Protection

6.1. Radiation properties

One of the key aspects of manned space mission is protection against space radiation. This radiation is mostly composed of high energy particles, originating from Solar Particle Events and Galactic Cosmic Rays. The radiation caused by the Van Allen belts are not considered because of the very short time spent in LEO.

6.1.1. Solar Particle Events

The SPE (Solar Particle Event) flux is composed of high energy particles (hundreds of MeV) which are mostly protons. There is a high dependency on the solar cycle: during the maximum solar activity, the flux of particles is much stronger than during solar minimum.

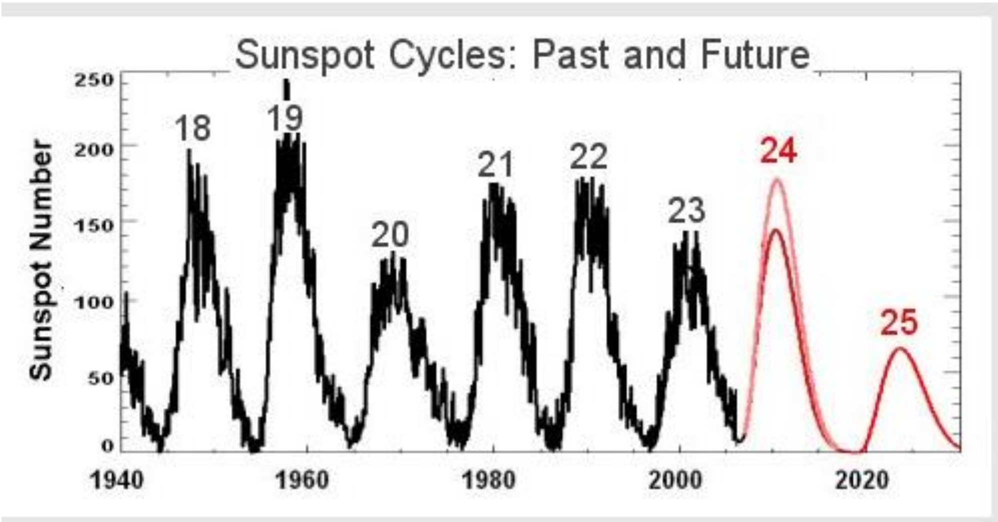


Figure 10 - Solar cycle evolution and prediction (NASA)

The mission will take place during the increase of solar activity during cycle 25 and will return to the Earth near the solar maximum. An efficient protection against solar events should be implemented into the spacecraft, as without the protection of the Earth's magnetic field, the crew is highly vulnerable to solar storms. Unfortunately, the direction of the incoming solar particles is not predictable, so the orientation of the spacecraft can't be used for complete protection against SPE.

6.1.2. Galactic Cosmic Rays

GCR (Galactic Cosmic Rays) are also composed of very high energy particles, but with higher energy and lower flux than SPE. Because of their high energy, those particles are more difficult to stop. The shielding can reduce the radiation but cannot completely block it. The distribution of the incoming particles is isotropic, so the whole surface of the habitat should be shielded against GCR.

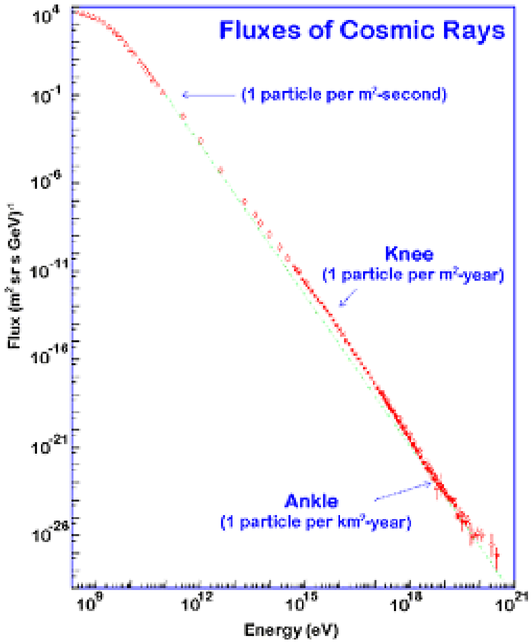


Figure 11 - Galactic Cosmic Rays spectrum (NASA)

There is a strong interaction between the solar activity and the GCR. During the maximum solar activity, GCR is significantly reduced.

6.2. Design of the shielding

The requirement is a maximum dose of 730 mSv per astronaut for the entire mission. This dose is calculated as an increase of 1.2% risk of fatal cancer during the crew's lifetime, and is within NASA and ESA career limits for astronaut radiation dosage. With a 580-day mission, the maximum dose acceptable is 1.22 mSv/day.

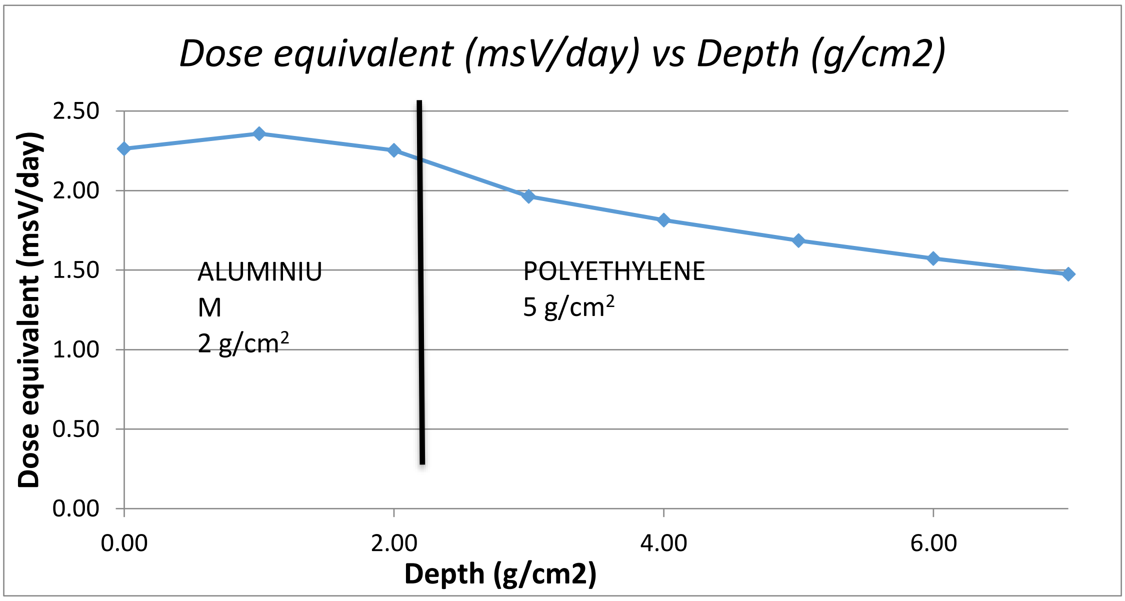


Figure 12 - Radiation absorption by the main shielding (Dose Equivalent mSv/day Vs Depth g/cm²)

The outer wall of the MPLM has some shielding effect, but it is only adequate protection LEO and not for a long-duration interplanetary mission. So, additional shielding must be added to the spacecraft. The radiation shielding design was validated with OLTARIS, the NASA On-Line Tool for the Assessment of Radiation In Space.

The aluminum MPLM walls have a shielding depth of 2 g/cm². An additional 5 g/cm² of polyethylene is added to the inner surface, for a total thickness of 7 g/cm². Polyethylene is the best choice for interplanetary missions because of its excellent radiation absorption properties[16], [17]. The polyethylene will be modified to be flame retardant. Water shielding was considered during the design process, but was not chosen due to configuration and structural issues. With this shielding, the dose equivalent in the habitat is 1.49 mSv/day.

To reduce this value, a small radiation vault is built around the crew's sleeping area to provide better protection during the time spent sleeping (assumed to be 8 hours per day). The vault is a cylinder of 0.65 m in radius and 2.2 m high. It is made of 14 g/cm² of polyethylene.

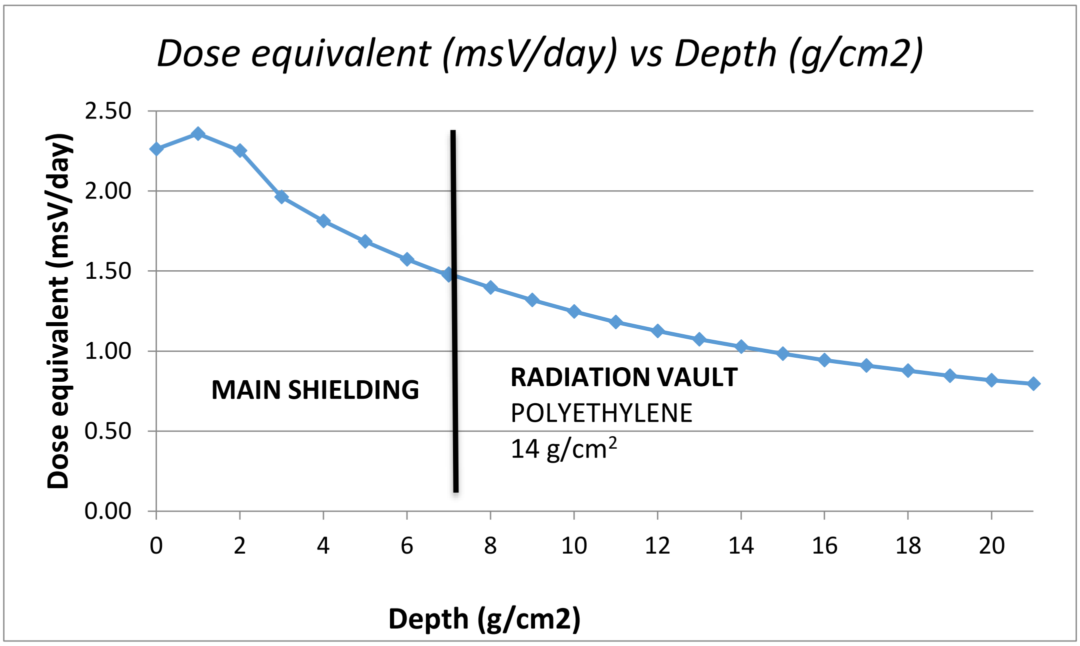


Figure 13 - Radiation absorption by main shielding and the vault (Dose Equivalent mSv/day Vs Depth g/cm²)

The dose equivalent inside the vault is 0.80 mSv/day. If the crew spend a third of the day inside the vault, the mean dose equivalent is 1.22 mSv/day, which meets the requirement for the mission. The radiation vault should also be used to protect the crew in case of major solar event. Once warned by Earth, or by built-in radiation sensor, the crew takes refuge in the vault until the end of the solar event.

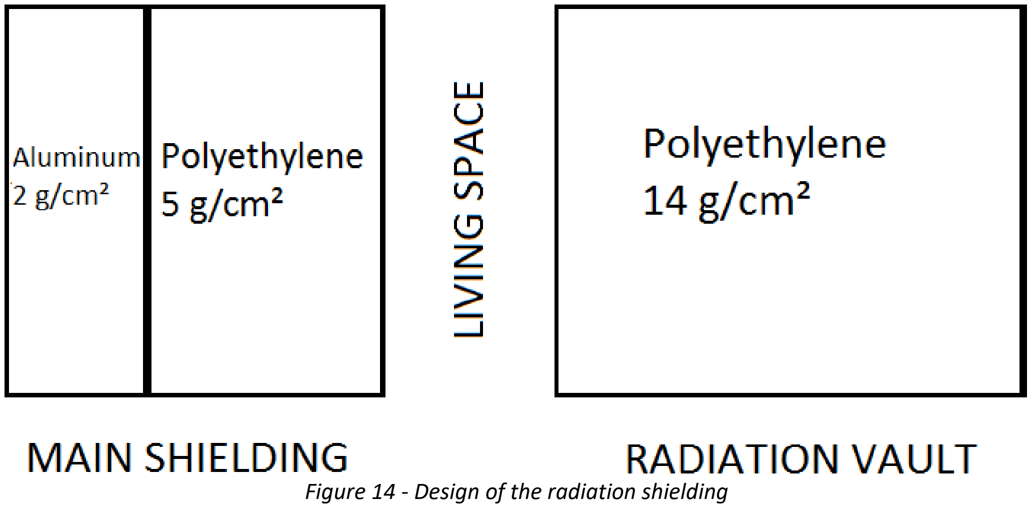


Figure 14 - Design of the radiation shielding

Main shielding (excluding aluminum layer)	5280 kg
Radiation vault	1630 kg
TOTAL radiation protection	6910 kg

Table 7 - Radiation shielding mass budget

In addition to this passive protection against radiation, the crew will wear passive dosimeters to monitor their radiation absorption during the whole mission. These dosimeters should have the capability to give results at any time during the mission, to enable real-time monitoring of the radiation dose during the mission. If the dose goes above the upper limit, the time spent in the vault should be increased.

6.3. Thermal Protection

Requirements:

- The spacecraft shall be able to maintain a human rated temperature within its pressurised volume.
- The spacecraft shall be able to keep all components within their safe operating temperatures.

While thermal control of a spacecraft is quite challenging, the main complication for this mission is the varied thermal environment due to encounters with two other planets. Venus is only at a mean distance of 0.72 AU to the sun, whereas Mars is 1.52 AU away [9]. The spacecraft must be able to cope with a wide range of temperature swings and therefore active control will be mandatory. Neglecting heat emitted from the visiting body itself and assuming the spacecraft skin will have constant absorptivity, the maximum and minimum thermal energy input on the spacecraft is shown in table 8.

Position	Solar flux at position [18]	Area exposed to solar flux	Total Solar energy
Venus Sphere of Influence	2643 W/m^2	76.2 m^2	201.4 kW
Mars Sphere of influence	593 W/m^2	76.2 m^2	45.19 kW

Table 8 - Maximum and minimum heat energy imparted from the Sun

Table 8 shows that the solar energy input varies wildly, with the maximum near Venus being more than four times as much as the energy absorbed near Mars. This poses great challenges for the spacecraft design and adequate measures of passive as well as active thermal control will need to be implemented.

6.3.1. Passive Thermal Control

The primary way of passive thermal control is the spacecraft's coating. Radiation is the main contributing factor, as convection and conduction are negligible in the vacuum of space. However, areas that are exposed to sunlight heat up very quickly and they will distribute the heat throughout the spacecraft by convection and conduction. If the spacecraft's coating has a low absorptivity whilst keeping a high emissivity (a so called selective surface), solar energy input can be minimised while maximising heat output through the infrared spectrum. When using this kind of coating in a multi-layer setup, which provides space between the layers to reduce conduction between layers, the spacecraft can be protected more efficiently.

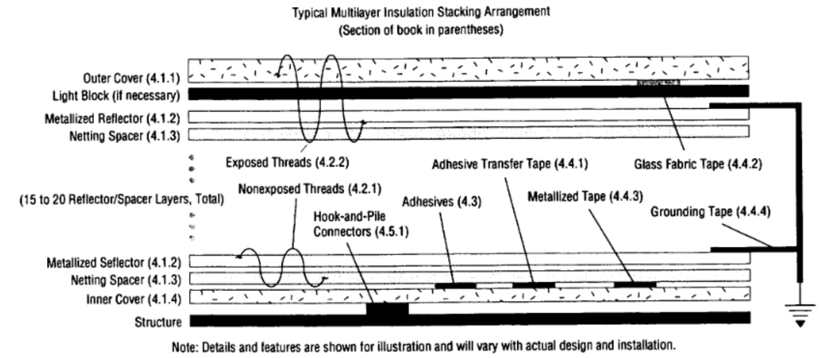
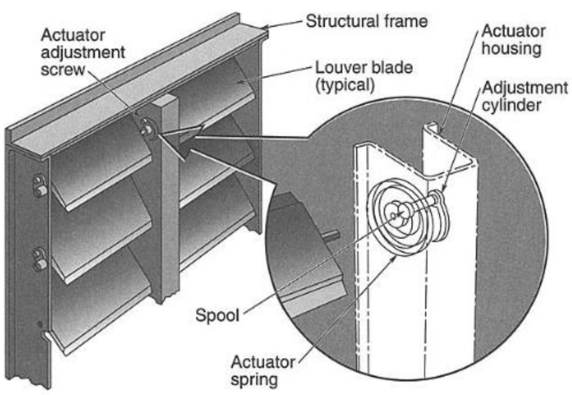


Figure 15 - Cross-section of multilayer insulation [58]

6.3.2. Active Thermal Control

For the duration of close proximity to the sun, which is between the Earth's and Venus' orbit several measures have to be taken to keep the spacecraft cool. Firstly, the spacecraft shall be orientated in such a manner that the EUS is facing the sun at all times. Due to the large diameter of the fuel tank, it will offer some shade for the pressurised volume of the spacecraft, making the thermal environment more manageable. Depending on the structural design of the Exploration Upper Stage and its thermal operating range, it might be necessary to use heat pipes to allow some of the heat to flow to the cold, shadowed areas where it can be dispersed using louvers and other space radiators. Heat pipes use the circulation of a fluid to transport heat energy. The fluid evaporates in the heated areas, continues travelling along the pipe and condenses at the cold end, after which it returns to the heated areas. This is a very efficient and relatively simple way of transporting heat.



The heat can then be dispersed into space using louvers, which are non-powered radiators capable of emitting large amounts of heat. The most common type of louver is the so-called "venetian blind", which consists of mechanically actuated blades, mounted on a frame. They can rotate and in the open position radiate six times as much as it does when it is closed. The more sophisticated versions do not use hydraulic systems to actuate the blades and instead use the temperature of the actuator spring itself to rotate the blade.

When the spacecraft is closer to Mars and its orbit around the sun, the spacecraft shall have its side facing the sun to increase exposure to sunlight and maintain an acceptable temperature. Pending further analysis, it might be necessary to include simple electric

Figure 16 - Schematic of Venetian Blind louver [59]

heaters in the spacecraft, especially in the non-pressurised service modules, to maintain a safe operating temperature for the equipment.

Thermal control of a spacecraft is also highly dependent on the heat dissipation of the subsystems within the spacecraft, this is further complicated by the addition of crew. Human heat dissipation varies per the form of activity and provides a much smaller range of acceptable operating temperatures. As detailed heat dissipation of the spacecraft's subsystems is not within the scope of this report, more emphasis was placed on selecting appropriate measures of energy dissipation or production. The scale and power of these active control devices then largely depends on the output of the other subsystems outlined in this report.

7. Power

7.1. Power Generation and Storage

Requirements:

- The power generation subsystem shall supply sufficient power for all spacecraft subsystems at all points of the mission.
- The power storage subsystem shall store sufficient energy to power all necessary spacecraft subsystems during periods of eclipse in Earth parking orbit or during planetary flybys.

The estimated power consumption of the spacecraft subsystems is shown in table 9.

Subsystem	Power (W)	With 20% Margin (W)	Comment
WPS (running) [7.1]	367	403.7	Scaled down from equivalent on ISS
WPS (standby) [7.1]	148.5	163.4	
OGA (running) [7.2]	1100	1210	
OGA (standby) [7.2]	120	132	
Air purifier [7.2]	60	66	
CDRA (running)	1100	1210	Assumed to be equal to the OGA.
CDRA (standby)	120	132	
Algae bioreactor	76.8	92.2	30% margin due to low TRL. Power consumed by LEDs.
Coolant loop pumps [7.3]	275	330	
Heat Melt Compactor	500	600	From HMC requirements [7.4]
Avionics	100	110	

Other electronics	250	275	
Miscellaneous pumps	300	360	
Communication	165.5	182.1	
Science	1000	1100	
AOCS	100	110	May be unnecessary.
Mode 1 Total		**4704.8**	WPS and OGA: running CDRA and HMC: standby
Mode 1 with 20% margin		**5645.8**	System-level margin added.
Mode 2 total		**5042.6**	WPS and OGA: standby CDRA and HMC: running
Mode 2 with 20% margin		**6051.2**	System-level margin added.

Table 9 – Spacecraft subsystem power breakdown

Certain subsystems – in particular, the Oxygen Generation System (OGA), Carbon Dioxide Reduction Assembly (CDRA), Water Processing Subsystem (WPS), and Heat Melt Compactor (HMC) – are designed not to run continuously, and will typically run for a duty cycle of less than 12 hours in any 24-hour period. This allows for two modes, as shown in table 9: mode 1, in which the WPS and the OGA are running and the CDRA and the HMC are in standby, and mode 2, in which the CDRA and HMC are running and the WPS and OGA are in standby. This will reduce peak power consumption. Further work to trade between cabin air quality and mechanical lifetime is required to determine the length of this cycle, which is currently assumed to be 24 hours.

Due to its trajectory, the spacecraft will experience a wide range of solar flux levels – from 2635 W/m^2 at perihelion (0.72 AU) to 591 W/m^2 at aphelion (1.52 AU). In addition, there are two main periods of eclipse. The first of these is while the spacecraft is in a parking orbit around Earth: at an altitude of 250km, the spacecraft will be in eclipse for at most 37 minutes of its 90-minute orbit[9]. The second eclipse period will occur 334 days into the mission, when the spacecraft flies by Mars and passes through its shadow for less than 1 hour. Unlike the previous case, the spacecraft will be in full sunlight for several months before and after this eclipse. Thus, while the spacecraft will be reliant on battery power during this eclipse, it can safely be assumed that batteries are fully charged before entering eclipse, and that batteries can be recharged slowly – perhaps over the course of several weeks – after the eclipse, significantly reducing the power required to recharge batteries before and after.

The Orion service module is equipped with four solar arrays, capable of generating 11.1 kW of power at 1 AU from the Sun. However, this will not supply enough power at aphelion, so the power subsystem also includes a pair of lightweight 4.9-metre-diameter ATK UltraFlex solar panels for an additional 6 kW of power at 1 AU. While these are somewhat larger than the flight-qualified panels used on the Cygnus cargo vehicle [19], and hence new arrays will need to be developed for this mission, UltraFlex arrays of a similar or larger size had reached TRL 6 by 2015[20], so this is not believed to pose a significant development challenge. Further work may reveal that the overall power consumption may be sufficiently low to use the flight-qualified 3.7 m solar arrays, as the current solar array size allows an additional 19% system margin (taking into account losses due to array pointing accuracy and a typical 3% degradation for GaAs solar cells after this time[9]) on top of the 20% system-level margin detailed in table 9. Given the 150 W/kg (1 AU) specific power for these arrays and the 300 W/kg specific power typical for large Power Conditioning and Distribution Units, the total mass of this additional array is estimated at 60 kg.

Operating Point	Peak Power Required /kW	Power to Charge Batteries /kW	Max Power Available /kW	System Margin
Earth parking orbit	6.05	6.31 (70% efficiency)	17.1	38.3%
Aphelion	6.05	0	7.2	19.0%
Perihelion	6.05	0	32.0	428.9%

Table 10 – Power demands during orbit

Note that around perihelion, the solar arrays would be placed at an approximately 75-degree angle to the Sun to reduce the power output to reasonable levels and reduce thermal loads on the arrays. It is also necessary to ensure that the solar arrays' support structures are long enough to avoid being shaded by the EUS when the spacecraft is near Venus.

The Orion command module is equipped with six lithium-ion batteries with a total storage capacity of 21.6 kWh[21]. As that the largest total energy drain in eclipse is, at most, 6.3 kWh (considering a typical 95% discharge efficiency) these batteries will provide sufficient capacity with a low depth of discharge.

7.2. Communication

Requirements:

- No Single point failures.
- Short upload and download times.

To simplify the communication constraints with the spacecraft it has been decided that the loss of communication with Earth is an acceptable risk due to a human crew being trained to handle situations in emergencies. Moreover, the chosen trajectory does not appear to take the spacecraft behind the Sun, so the only communications blackout will be a brief loss of signal when the spacecraft passes behind Mars.

The spacecraft will have an array of standard communication antennas for different frequencies including X and Ka-bands. This allows for a variety of data rates to be utilized for different purposes such as TT&C and data dumps. Each frequency will feature two transceivers and amplifiers for redundancy. In addition, an experimental new form of communication using lasers, tested on the Lunar Laser Communications Demonstration (LLCD) mission will be utilised. The Laser Communications Relay Demonstration (LCRD) is the next stage of testing and is scheduled for 2017. This technology will be tested on the flyby mission. The laser communication system will utilize the ground stations built by NASA for the 2017 test flight therefore no additional infrastructure will be required[22]. Given the already sufficient data rates of Ka-band RF communications, if the laser system can be proven reliable, then the crew would have the use of the optical system for even higher gain communications with Earth allowing for larger data dumps with a quicker upload and download time; benefits include higher resolution videos and less compression of scientific data collected. The system has a high TRL, and is compact and lightweight[22]. As the optical system has no heritage for interplanetary flight it was decided that the laser communication payload would not be the primary high-gain communication system, but only an experiment to assess its viability for possible future manned flights. This would involve incremental tests of data rates at different distances, associated noise and received signal strength.

Unit	Qty	Total Power (W)	Total Mass (kg)	Notes
X-Band Transponder	2	26	6	
X-Band Travelling Wave Tube Amp (TWTA)	2	50	5	RF output 25 W
X-Band Diplexer	2	0	1.2	
X-Band Switching Network	2	0	1	
X-Band Cables	All	0	5	18 RF Cables
X-Band Low Gain Antenna	2	0	1.4	Waveguide horns 70 deg beamwidth (3dB)
X-Band Medium Gain Antenna	1	0	1.5	0.3 m waveguide Horn 5 deg beamwidth (3dB)
X-Band High Gain Antenna	1	0	6	1.5 m diameter parabolic reflector offer
Ka-Band Exciter	2	3	0.6	Incremental increases to X-Band transponder
Ka-Band TWTA	2	81	5.6	RF output 40 W
Ka-Band Waveguide	All	0	3	
Ka-Band High Gain Antenna	1	0	5	3 m diameter parabolic reflector
Ultrastable Oscillator	2	5	2.6	
Optical Transceiver	1	0.5	30	Mass includes complete assembly and wiring
Total:	-	165.5	73.9	

Table 11 – communication instrument list [9]

	X-Band (High Gain)	Ka-Band (High Gain)	Laser
Frequency	32 GHz	8.35 GHz	Infrared
Power Consumption	100 W	35 W	0.5 W
Coding	1/6 turbo coding	1/6 turbo coding	-
Data Rates (before coding)	0.5 Mbps	2.6 Mbps	
With Coding	1.5 Mbps	6 Mbps	-
Required E_b/N_o	1 dB	1 dB	-

Table 12 – Frequency and data rates [22], [23]

The Deep Space Network (DSN) will be used for uplink and downlink of high data content to and from the crew. A summarised view of the different options available and their relative estimated costs for fiscal year 2021 is shown in table 13. As constant high gain contact is not required option 3 will be considered as it offers the time required for large amounts of data to be uploaded and downloaded from the mission at a reasonable cost. Scheduled communication windows will take place each day to give communication order. This is especially important due to the time delay between messages. Communication windows will increase during important manoeuvres and way points. Option 3 covers enough time of each day for emergency high gain communication if required.

Option	DSN Antenna	Contact Hrs/Day (outbound)	Contact Hrs/Day (return)	Total Cost FY2021 ($M)
1	34 m	5	6	15.18
2	34 m	10	12	29.808
3	34 m	15	18	34.776
4	34 m	20	24	37.153
5	70 m	5	6	62.928

6	70 m	10	12	97.704
7	70 m	15	18	134.136
8	70 m	20	24	172.181
9	34 m (75% time) & 70 m (25% time)	20	24	94.392
10	34 m (50% time) & 70 m (50% time)	20	24	119.232
11	34 m (25% time) & 70 m (75% time)	20	24	149.04
12	34 m & 70 m (same time)	5	6	119.232
13	34 m & 70 m (same time)	10	12	205.344
14	34 m & 70 m (same time)	15	18	288.144

Table 13 – DSN costs with usage[23], [24]

8. Life Support

8.1. Atmosphere

Requirements:

- Oxygen shall be provided in the most mass efficient manner possible.
- Oxygen shall be provided such that there is enough for 2 astronauts over a 580-day mission (plus margin).
- Where available, oxygen shall be recycled at as high an efficiency as possible from carbon dioxide and waste water.
- Carbon dioxide shall be removed from the environment quickly enough such that the astronauts do not suffocate through lack of oxygen.
- Air shall be supplied at such a rate that provides a high enough pressure atmosphere such that the astronauts are comfortable, the oxygen levels are appropriate, and does not impede normal functions for equipment.
- Emergency oxygen tanks shall be provided in case of failure of oxygen generation system.

The air management system is a crucial part of the life support system. As well as supplying breathable oxygen, the carbon dioxide breathed out by the crewmembers must also be removed, as well as contaminant particles needing to be filtered out. The best systems that currently exist are operational on the ISS, and so for the mission it is essential to look at the current technology on-board the space station and consider the modifications needed for the Mars flyby mission.

8.1.1. Oxygen Generation System

The Oxygen Generation System (OGS) that is on-board the ISS consists of the OGA, the CDRA and a Power Supply Module (PSM). Oxygen on the ISS is generated through the OGA, which produces oxygen through the electrolysis of water. By processing feed-water by-product from the Water Processor Assembly, this can save the need to carry the necessary oxygen. Carbon dioxide produced by the crew is converted back into water through the CDRA using the hydrogen produced from the electrolysis reaction earlier through the Sabatier reaction, as shown in Equation 1 below, with this water fed back into the WPA to create oxygen again. Current estimates put the Sabatier reaction efficiency at approximately 50% [25], but currently advancing technologies could put this to 75% by the mission's launch date with only minor modifications of the system required[25], [26]. Testing of CO_2 scrubbing technology is important if considering a Mars landing mission, as this technology could be used to convert the carbon dioxide in the Martian atmosphere to breathable oxygen. At approximately 356 kg [27]and nominally operational to produce 5.4 kg of oxygen per day, the OGA currently used on board the ISS far exceeds the required oxygen conversion necessary for the mission. Considering the modifications by Takada et al[28], as well as appropriately scaling for the crew size, it would be safe to assume that an OGA developed for the mission could be an approximate mass of 100 kg.

$$CO_2 + 4H_2 \rightarrow CH_4 + 2H_2O \quad (1)$$

Nitrogen and oxygen can be stored using the Nitrogen/Oxygen Recharge System (NORS) currently utilised for the ISS to supplement the oxygen and nitrogen losses that occur. The gases are stored at a high pressure of approximately 40 MPa in composite tanks which are robust enough to handle the high pressure [29]. The NORS system will primarily be used for the inert nitrogen gas used to keep atmospheric pressure, however it would be sensible to keep a supply of oxygen in case the electrolysis system fails. Since there are no EVAs planned, it is not necessary to consider additional air to supplement EVA air losses.

Current estimates give the nominal rate of oxygen required per crewmember-day as 0.835 kg [2]. To produce this amount of oxygen for a crew of two, 1.879 kg of water is required per day. Using the 75% recycle rate from carbon dioxide as previously discussed, and with an average rate of CO_2 production from humans of 0.998 kg per crewmember-day, this gives 0.544 kg per crewmember-day of recovered O_2, meaning 0.291 kg per crewmember-day needs to be supplemented. It is also important to consider leakage of air from the spacecraft. A good approximation from the early days of the ISS gives a leakage rate of 0.05% per day[30]. Given that the ambient pressure of the vessel is 101.3 kPa, this gives the total oxygen leakage over the course of the mission as 7.261 kg. Including the initialisation oxygen required, the total water required to supply oxygen for the whole trip, with a 20% margin, is 500 kg.

To keep the cabin pressure at the ambient pressure desired, the ratio of nitrogen gas to oxygen should be kept at approximately the same level as at sea-level – that is, 78% nitrogen to 21% oxygen, with 1% for other inert gases. This is small enough to be negligible in the calculations. With the previous assumption of the leakage rate, the total nitrogen required, with a 20% margin, is 145 kg. 93.9 kg is required at any one time, with 25.3 kg of oxygen, to give the correct atmospheric pressure.

8.1.2. Atmosphere Pressurisation

It is possible to use lower than sea-level atmospheric pressure for the cabin. Using a lower pressure means that the cabin walls do not need to be as strong to cope with the pressure difference, reducing the spacecraft mass, as well as lower leakage and loss rates. There are also many disadvantages to using a lower pressure. The cooling efficiency of electronic components is reduced since less is air flowing over, meaning fans need to run faster and make higher noise; communicating vocally between crewmembers becomes more difficult; the risk of fire is higher due to the higher partial pressure of oxygen; and conditions for experiments are less-standardised compared to Earth [31]. It therefore seems more feasible and safer to use the sea-level pressure of 101 kPa for the cabin's atmosphere.

8.1.3. Atmosphere Revitalisation and Monitoring

The Atmosphere Revitalisation (AR) subsystem currently adopted on-board the ISS can be modified and adapted for the mission. Air purifiers, like the Airocide filter [32]which uses NASA technology, can be utilised for the removal of trace contaminants and organic volatiles. As well as the removal of volatiles, the AR can be used to monitor the atmospheric content, ensuring that the cabin has the right balance of oxygen, nitrogen and carbon dioxide, as well as other constituents like water, to ensure the right humidity is utilised on board.

8.1.4. Fire Detection and Suppression

Currently, the ISS uses a Fire Detection and Suppression (FDS) subsystem[33]which can be utilised in an identical manner for the mission – this includes smoke detectors, fire isolators, extinguishers and a fire recovery system. In case of fire in the habitat, the fire should be immediately extinguished using one of the nearby portable fire extinguishers – currently the ISS utilises carbon dioxide and water mist extinguishers. The astronauts shall be trained to utilise the most appropriate extinguisher.

8.2. Biological Based Life Support

Due to the length of the mission, alternatives to standard chemical atmosphere revitalisation systems have been examined. The obvious choice was the use of photosynthetic organisms to remove carbon dioxide and replenish oxygen with an added benefit of a fresh food supply. Plants and cyanobacteria (bacterial micro-algae, referred to as algae from now on) were examined. The use of higher plants for oxygen production was found to be infeasible, as the volume of foliage required to sustain the oxygen requirements of a single crew member was greater than the habitable volume of the habitat module. Algae, however, needed a thousandth of the volume required by higher plants, depending on the concentration. This is because oxygen production relies on illuminated surface area of algae rather than volume. On top of this improved surface-area-to-volume ratio, algae are single-celled organisms, so its process of photosynthesis is far more efficient. As well as the benefits of atmosphere revitalisation, some species of algae are edible and are praised for their high protein and oil content, making them a good source of nutrients for the astronauts.

Algae has been shown to adapt well to the microgravity environment and the radiation environment after being exposed to ionizing gamma and neutron radiation equivalent to varying periods of time on-board the ISS[34]. This, as well as its simple culture requirement, makes it an excellent candidate for a long duration mission to Mars.

Some initial problems are that the illuminated area required to produce sufficient oxygen is large, and as the algae concentration increases, light cannot permeate as far into the solution, creating dead zones[35]. A possible solution is to use thin sheets of algae allowing for higher concentrations, which can be stacked, folded or in tubes creating the high surface area required in a confined space. This allows for better illumination with well positioned

LED grow lights, thus eliminating dead zones and making it an ideal method for low volume environments such as the MPLM habitat module. Another important aspect is suitable gas exchange within the solution, required to feed the algae with carbon dioxide rich waste air and to remove the produced oxygen for redistribution into the spacecraft. Gas permeable membranes are advantageous in this application because of the large available surface area and the short depth of the algae solution promoting efficient gas exchange whilst minimising bubble formation. The two gasses can easily be separated due to their large silicone permeability coefficient ratio, making it an excellent candidate for silicone membrane separation [36].

Requirements for an Algae Photo Bioreactor:

- Produce 0.835 kg of oxygen per day[2].
- Remove 0.998 kg of carbon dioxide per day[2](Efficient gas exchange).
- Shall fit within an International Standard Payload Rack.
- Mass shall be no more than 150 kg per human.
- Power demands shall not exceed 100 W.

A high area photo bioreactor (HArP Bioreactor) was designed for the Gemini Mars mission which is compact, power efficient and almost closed loop for oxygen revitalisation. The HArP bioreactor offers efficient gas exchange due to the tubing design seen in Figure 17. The design promotes gas exchange in and out of the gas permeable inner tube which holds the algae flow. This is due to the narrow distance between the outer gas flow tube and the inner algae tube as well as the long distance of the tubing. The inner membrane tube has an inner diameter of 1.5mm allowing for significantly higher concentrations with respect to depth, to be used due to the shorter path length through the algae culture solution. The higher concentration means less solution is required reducing the weight of the overall system. The tubes are folded into sheets and illuminated from each side using LED grow panels, shown in purple in Figure 19. LEDs are efficient at producing light so thermal issues are negligible compared to other light sources. Moreover, they are compact, low cost, low power and have a lifetime far exceeding the mission length. Each LED panel will contain 32 LED clusters of five (16 in each direction) which draw 0.3 A, so eight panels gives a total power of 76.8 W.

The produced oxygen and non-absorbed carbon dioxide will be separated with a silicone membrane filter, with the excess carbon dioxide being fed back into the system and the oxygen recovered.

The physical characteristics of the HArP Bioreactor are outlined in table 14 directly compared to the Boeing large algae experiment conducted in 1959[37]. Both Bioreactors have the capability to sustain the atmosphere demands of one human in a closed environment.

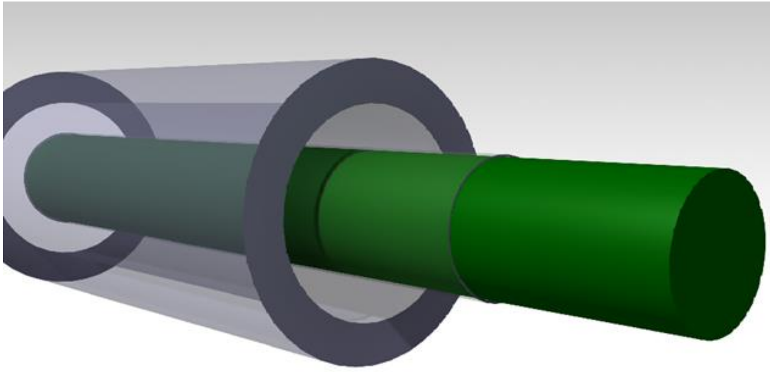


Figure 17 - CAD model of section of tubing

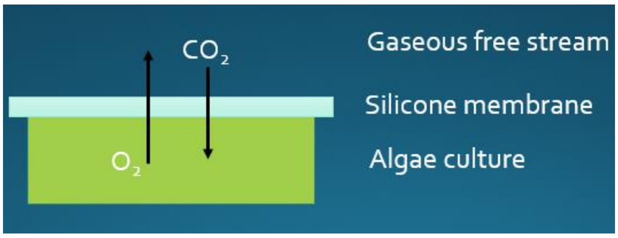


Figure 18 -– Gas exchange process within the design

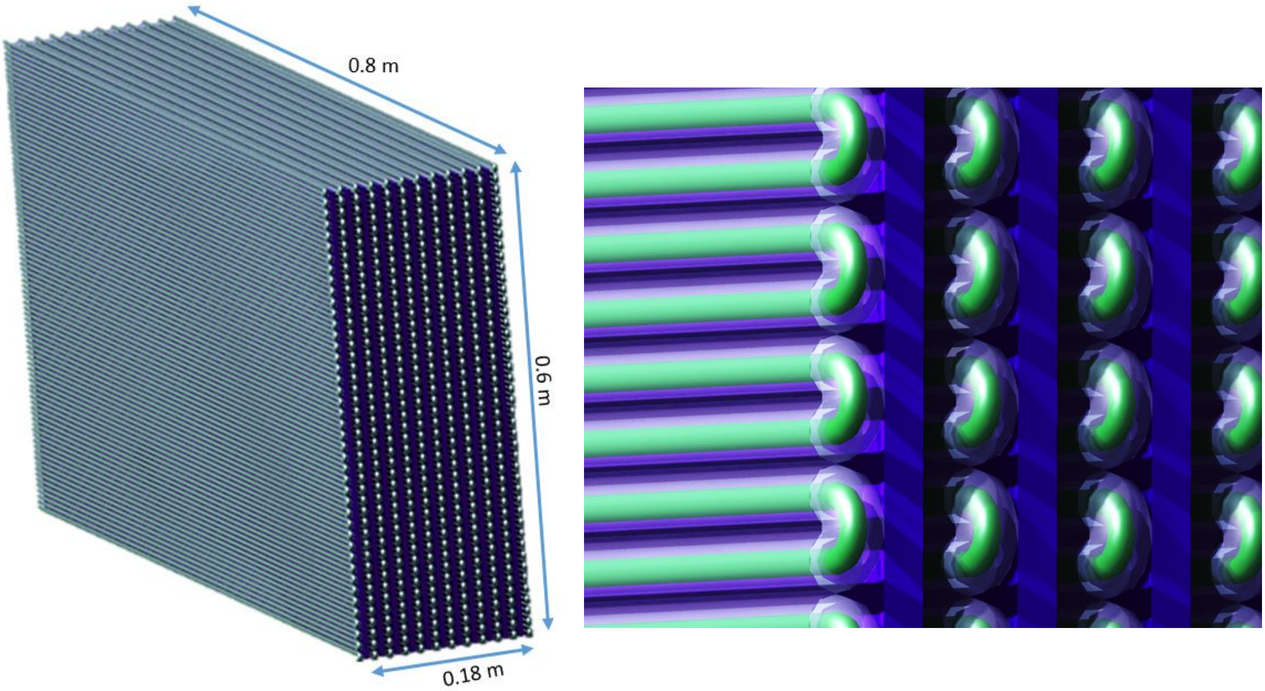


Figure 19 – tubing and light panels (a) dimensions (b) zoomed view

	HArP Bioreactor	Boeing Bioreactor
Algae concentration	8%	0.1%
Algae Water Solution	4.7 Litres	380 Litres
Illuminated area	5.7 m^2	22.3 m^2
Culture containment	605m of 3mm tubing	8 (3 x 1.5 x 0.038 m) Tanks
lighting	8 LED light panels (1280 LEDs)	300 florescent light tubes
Power	76.8 W	9000 W
Mass	21 kg (tubing and LEDs)	-

Table 14 - comparison of HArP and Boeing bioreactors [37]

The HArP Bioreactor having a low technology readiness level will work alongside the chemical based recycling system as a redundant system. This flight will provide crucial scientific data on the viability of biological based life support systems such as the effect of algae reproduction. The data collected will be able to confirm whether they are reliable enough to be an independent system. As the HArP bioreactor can only sustain one person, two will be included in the flight. This will allow the chemical system to be switched off and allow the HArP Bioreactors to run independently, while still having the chemical system as a fallback option. The gas exchange will be measured along with other processes to further refine the design for future missions.

8.3. Food

By assuming the two astronauts have an average metabolic rate per day of 2,677 kcal each and a respiration quotient of 0.87 then the dried food mass required per astronaut per day is 0.62 kg. With mission duration of 580 days that gives a dried food mass of 720 kg. Assuming a margin of 5% this takes the dried food mass to a total of 755 kg [38].

The 755 kg of dried food required for 580 days will be taken instead of relying on growing food on the journey as this will mitigate risk. Any food produced on the journey will be extra calories. The 5% of margin also includes food that is not specifically 'calorie efficient', such as foods that are not consumed for calories and nutrition, but also for morale. The margin also includes foods that were not dehydrated, as dehydrating the food reduces the mass.

One way to ensure the astronauts receive their required nutrients is through a soluble nutrient pill which can be added to their drinks, as this method is compact and effective.

8.3.1. Vegetable Production

The vegetable production system (Veggie) currently in place on the ISS will be used on-board to grow lettuce and flowers. Its compact fold out design allows it to be easily stowed when not in use, an important feature due to the low habitable volume[39]. The Veggie gives a much-needed variety and freshness to the pre-packaged pre-cooked food available. Lettuce will be grown due to its adaptability to the microgravity environment and its simplistic growing conditions, no need for pollination. Flowers will be for the crew's personal enjoyment boosting self-esteem and reconnecting them to Earth.

8.4. Water

Requirements:

- The water processing system shall provide potable water to the crew for drinking and food preparation. The quality of the potable water shall meet water quality requirements for the Water Reclamation System on the ISS[40].
- The total mass of the water processing system and any consumables required shall be less than the total mass of water required if no water reclamation was used.

The design of the water reclamation system for this spacecraft is based on the ISS Water Reclamation System. This consists of the Urine Processor Assembly and the Water Processor Assembly. The UPA uses vapour compression distillation to distil water from crew urine. This water is then combined with other waste water condensed from the cabin air, and passes through a series of multi-filtration beds to remove contaminants[41].

The UPA also produces a concentrated brine which cannot be further processed without causing soluble components to crystallise, so the water component of this brine is considered lost. Initially, the UPA was designed to reclaim 85% of the water from crew urine, but due to unexpectedly high levels of calcium sulphate due to bone calcium loss in microgravity, its efficiency is currently limited to 75%[42], so this value is used for the system design. Recent work suggests that this efficiency can be improved to above 85% by replacing sulphuric acid with phosphoric acid in the urine pre-treatment formula.

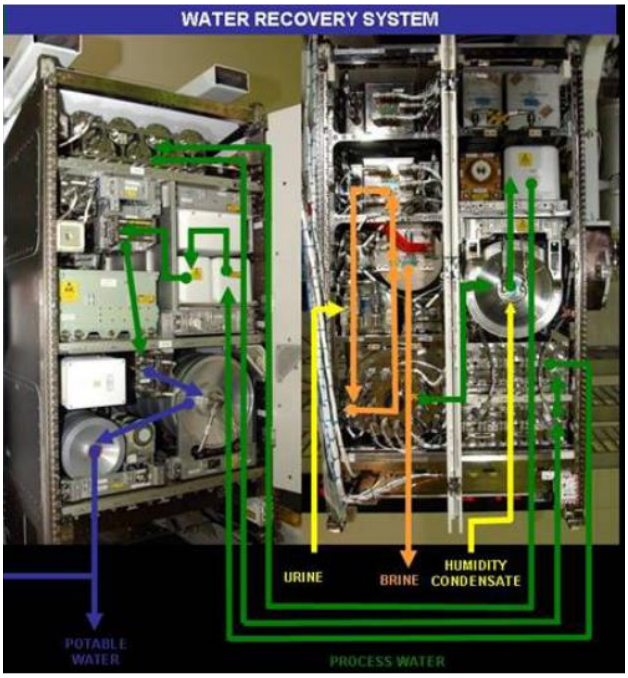


Figure 20 – Diagram of the WRS from Carter et al [26]

The ISS water reclamation system has a mass of at most 660 kg and takes up two International Standard Payload Racks for a total of 3.14m^3 of pressurised volume [40]. However, as this is designed for a crew of 6 on the ISS, the water reclamation system for this mission may be smaller. While mass, power and volume may not scale linearly with throughput, it seems reasonable to assume that the mass and volume may be reduced by 30% and the power consumption by 50%.

The total water input and output from the water reclamation system is estimated as follows[38]. Note that the most significant losses are due to the UPA and unrecoverable water lost in faeces, but an upper bound of 3% is allowed for other water systems to account for smaller-scale losses, such as humidity lost due to atmospheric leakage. The 0.5 kg/CM-d of washing water is a generous estimate for the water required for a “sponge bath”, but as much of this water is recycled, the results are not very sensitive to errors in this estimate. Due to the use of dehydrated food, it is assumed that the food water input is

Water use	Mass (2 crew) (kg/day)	Output Item	Mass (2 crew) (kg/day)	Recycling efficiency	Potable water reclaimed (kg/day)
Food water	0	Respiration and perspiration water (condensate)	4.56	97%	4.42
Food preparation water	3.82	Urine water	3.0	75%	2.25
Drinking water	3.24	Faeces water	0.18	0%	0
Oral Hygiene	0.74	Greywater (from hygiene water)	1.74	97% (assumed to evaporate and condense)	1.67
Washing	1.0				
Total	**8.8**				**8.34**

Table 15 – Estimated water mass balance

zero, and thus that the potable water used as food preparation water must be increased to compensate for this.

The total daily water loss is hence 0.46 kg per day. Allowing a 20% margin, this means 320 kg of water must be brought along for the mission. Even with the total mass of the water reclamation system, this is still significantly less than the mass required for bringing the water required without recycling.

Over the past three years, the ISS has averaged approximately 1 kg of consumable mass required in the form of consumables and spare parts for each 10 kg of water output [42]. Much of this mass appears to be due to spare parts, so with further maturation of the design and increasing component lifespan it seems reasonable to assume that this ratio may be improved by a factor of two. This would mean approximately 250 kg of consumables and spare parts is required for the mission.

8.5. Human Waste

Requirements:

- Solid and liquid human waste shall be collected and stored with minimal leakage.
- The waste management subsystem shall minimise odours and contamination of the crew and cabin by waste materials.

The Universal Waste Management System (UWMS)[43] is a concept for an updated commode, designed for improved cleanliness and reduced mass and volume compared to prior systems. Faecal matter is collected in a hydrophobic bag, which is then transferred to a canister and compacted using a crew-operated lever mechanism. This reduces volume and mass, and drastically reduces power consumption compared to previously-proposed motorised compactors. The sealed canisters of faeces are then stored, and may be used as additional radiation shielding.

Urine is entrained in airflow through a funnel and hose and carried to a centrifugal separator, which separates the urine from the air and transfers it to the Urine Processing Assembly.

To reduce odours, air from the commode is passed through an activated charcoal filter. This has a lifetime of 210 days and so will need to be replaced multiple times. It is assumed that the total mass of these replacements is 20 kg. The mass of the system is assumed to be equal to that of the Extended Duration Orbiter Waste Collection System [44] used on the

Space Shuttle, for a mass of 111 kg. This is a conservative estimate, as one of the principal design goals of the UWMS is to have a lower mass than previous-generation systems.

8.6. Other Waste

During the mission, other solid waste is generated, consisting largely of food packaging and other plastic wastes, as well as any adhered food remains. Over the course of a long-duration mission, this takes up significant pressurised volume, causes residual water in food wastes to be lost, and may allow microorganisms to grow. One proposed solution is the Heat Melt Compactor currently under development for long-duration spaceflight. It uses mechanical pressure and heat to compact packaging and residual food waste into dry sterile tiles, as well as evaporating residual water from food waste so that it can be reclaimed and recycled into potable water [45]. For this mission's crew of 2 and duration of 580 days, it is estimated that the HMC would save approximately 6.6 m^3 of pressurised volume by the end of the mission simply by compacting trash [46] – a significant gain given the low pressurised volume available.

It may be possible to use the HMC to recover the remaining water from crew urine brine [46]. This would further increase the efficiency of the water recovery subsystem, making the water subsystem a closed loop, and perhaps even able to produce excess water to supply the OGA. As this application of the HMC has not yet been demonstrated, it is not included in the current baseline, but further investigation here is certainly worthwhile.

9. Mission Science

This mission will be the first long-duration and deep-space manned mission, and so a huge amount of scientific and technologic return can be expected. The knowledge that can be gained from the mission can have huge consequences both for science and for space technology.

9.1. Human Science

The primary scientific payload is the crew itself. Unfortunately, we have very little information about the effects of long-term exposure to the deep-space environment. The only deep-space mission is the Apollo program, which accumulated only 83.8 days of manned spaceflight over several flights. Further long-term missions and eventual Mars colonization require better knowledge of the impact of space environment, micro-gravity and deep-space radiation, on human health. This mission will increase the cumulative time spent beyond LEO by a factor of 14.

The crew will wear biometric equipment which can monitor heart rate, temperature and blood pressure. In addition to this constant monitoring, the astronauts will have regular blood test and ultrasonic examination. Vision, motor and orientation tests should also be regularly done. All these results, including blood sample, will be analyzed after the mission to have a better understanding of human physiology in space.

Another key aspect of long-duration space-flight is the psychology of the astronauts during a long-term journey in the solar system. This is discussed later in section 11, however psychological science is an integral part of human science. The ISS astronauts' studies are the only indication scientists have about the behavior of human crew during those long-term stays in space. However, further missions require better knowledge.

9.2. Deep-Space Science

The access to beyond-LEO space is quite restricted for scientific mission because of the obvious reason of cost. The Mars fly-by is a great opportunity to conduct very deep-space experiments (effect of micro-gravity, hard vacuum and deep-space radiation). The experiments can be placed either inside the spacecraft or outside, per the purpose of the experiment. The inside ones can be manipulated by the crew, giving a huge flexibility while the outside ones are much exposed to space environment. Moreover, thanks to the long duration of the mission – more than one year – and the range of distances from the Sun (0.72 to 1.52 AU), a large range of experiments can be done. Those experiments can lead, for example, to a better knowledge of deep-space radiation, of biological and material resistance to deep-space environment. Like human science, the discoveries will be used for further manned missions to deep-space.

The mission can also increase our knowledge about the survival of organisms during exposure to the deep-space environment. To avoid contamination of results during Mars-ground mission, probes are now decontaminated before launch, which increase the cost of the mission. A measurement of the survival rates of bacteria and other contaminants during the mission can determine if decontamination of probes before the launch is necessary, or if the space environment can do it by itself it during the journey.

9.3. Planetary Science

The mission consisting of two fly-bys, of Venus and Mars, is a good opportunity to learn more about those planets. Because the spacecraft is not landing, only remote scientific sensors can be used, like imagers, spectroscopes or radars.

As the Mars periapsis is quite low, only 363 km, multiple surface penetrators can be considered. They can carry scientific payload underground to provide information about geology or regarding the presence of water. Surface penetrators can reach areas which were never investigated by surface survey. Results can then be useful for determination of the landing site for further missions.

But, this mission is also an amazing opportunity to promote science to the public, to show high definition picture, raising interest in by science and space exploration.

9.4. Technology Demonstrator

This mission is also a demonstrator for technologies which will be used for later manned missions. These technologies will not be used as an operational subsystem of the spacecraft, as they are not flight proven. For example, new life-support technologies, at a lower scale, can be installed on the spacecraft to test them in real flight conditions without risk for the crew.

10. Psychological Health

Taking an almost two-year trip constrained to a single spacecraft will require exceptional psychological fortitude of the astronauts.

10.1. Earth Communication and Outreach

Communications with Earth will have positive effects on the crew. Having a complete mission team keeping continuously up-to-date status reports will bring some peace-of-mind to the astronauts, knowing that there are several people looking after them at any one time. As well as psychiatric care, having communication sessions with friends and family will provide ease and enjoyment to the astronauts whilst they are on-board. Of course, as the crew travels further from Earth, live video becomes impossible, but video messages to and from Mission Control should still suffice for communicating with the people on the ground. The astronauts may also benefit psychologically from social media and outreach programmes, where they can have near-immediate discussions with the general public, and share insight into their lives. This will allow them to feel a greater sense of purpose on their journey, inspiring others with their mission.

10.2. Environmental Factors

Living in the limited space provided will cause serious distress for the astronauts, and so it would be sensible to optimise the space available. On the ISS, one of the favourite relaxing past-times is to watch the Earth through the windows. There will be windows provided in the capsule, however it will not be possible to observe planets outside for very long periods of time. It may be feasible to provide a telescope to look at the planets from afar and to stargaze, which may also help quench any feelings of monotony the astronauts may be feeling. It is also suggested to utilise a large television screen or projector if possible to display natural environments, providing the crew with a sense of a larger space as well as contributing to a more diverse set of colors. Colors may also be used for orientation requirements around the spacecraft[47]– gradients of colors along the walls may help aid in defining directions for the crew.

Lighting conditions are important for both physiological and psychological factors. Humans follow a circadian rhythm – a cycle that biological processes follow which repeats every 24-hours, driven by the environment, i.e. the Earth day/night cycle. To maintain the astronaut's circadian cycle, it is required to have access to up to 2,500 lux of illumination to emulate a typical sunny day, however currently the ISS only has access to around 100-500 lux – approximately equivalent to office lighting[47]. This affects the sleeping pattern of the astronauts, and in turn, stress levels, alertness and cognitive ability. A brighter LED lighting system is to be tested on the ISS later this year which will be tested to improve the circadian rhythm in astronauts [48]. This system would also reduce the necessary mass for the lighting system, as well as reduce power requirements and the generation of heat from light sources in comparison to the typical fluorescent light on-board the ISS.

Spending time in natural environments is also known to reduce stress, as well as increase poor health recovery time[47]. To this end, the hydroponic system that will be in place will be of use. As well as providing a greener, more natural setting for the astronauts, tending to the plants also helps with overall wellbeing – helping the plants grow shall provide a satisfactory, rewarding experience which can counteract the monotony of working aboard the module.

10.3. Astronaut Selection

It has been noted that astronauts that make a return trip to the ISS adjust much more easily than their colleagues going up for their first expedition [49]. Thus, it would be sensible to first consider seasoned astronauts for the long-haul trip to Mars. Carefully selecting two candidates who are compatible with each other may play the biggest factor psychosocially. It has been noted on previous ISS missions that astronauts with culturally different backgrounds may cause an increase in stress levels between crew mates. A case could be made for a married couple to be flown, meaning the astronauts will have already demonstrated a high compatibility rate together. With that being said, it would still be important for each crewmember to have access to private crew quarters, as research has shown that having access to privacy is a key habitability factor[47].

Astronauts who already display the traits of resiliency are likely candidates for the mission – the ability to adapt to difficult scenarios appropriately. Often when seeking candidates for spaceflight, the term “The right stuff" has been traditionally used to describe a candidate who can adapt to complex situations, has the desire to achieve goals and has a high social competence which allows them to work well with the rest of their crew.

10.4. Autonomy and monotony

Monotony plays a major role in the near two-year trip. In such a restricted volume with a limited amount of supplies, it’s highly likely that boredom will strike the crew at some point. Therefore, it is important to give the astronauts meaningful work and as much leisure activities as possible. The key psychological factor in such a long-duration mission is to give the astronauts useful work to do, giving a sense of purpose to their work [50].With that said, what gives purpose to one astronaut may not be the same as another. Thus, it is important to analyse the crew and give tasks per their psychological traits – one crewmember may be more hands on, and like to perform more maintenance on the spacecraft, whereas another may see more purpose in performing research. Other activities like exercise, watching movies, reading books and playing video games will also help break the monotonous journey.

It is also important to allow the crewmembers to work as autonomously as possible. This will achieve certain goals, namely it should stop the crew being overworked, and should go far to stop any dissention having a daily schedule micromanaged for nearly two years may cause the crewmembers to feel resentful towards ground control. By setting their own schedules, the astronauts shall feel in more control and should stop any feelings that they are only there as a science project.

10.5. Overall Morale

In order to upkeep overall morale throughout the journey, it is important to ensure that the astronauts adjust to their new life accordingly and appropriately – "unsuccessful psychosocial adaptation can lead to adjustment disorders characterised by decrements in performance" [51]– bad adaptation early in the journey may manifest itself later on at key phases for the astronaut, with lower performance rates, which will lead to lower morale and may possibly be life-threatening. This can be counteracted with good psychological support in the early phase of the mission, when communication with Earth is still near-instantaneous. Alongside this, the astronauts will also be allotted 10kg each of personal mass – this may be used for candy bars, comfortable clothing, personal photos or care packages from family that may be opened at certain milestones. Little reminders from home will give an overall morale boost to the crewmembers.

11. Physiological Health

Requirements:

- The spacecraft shall be equipped to handle most medical emergencies if the need arises.
- The spacecraft shall have fitness equipment which is utilised to counteract the problems that arise from microgravity.
- The spacecraft shall have a supply of vitamins, drugs and other tablets that can be used to offset the effects of microgravity.
- Measures shall be taken upon return to Earth to rehabilitate the astronauts to a gravitational environment accordingly.

Upkeep of the physiological state of the crew members is vital. Current data regarding long-duration spaceflight is limited, with a typical expedition to the ISS lasting 6 months. The longest single-duration flight is held by Valeri Polyakov with 437 days during the Mir space station mission, with the "Year in Space" missions with Mikhail Korniyenko and Scott Kelly recently ending in March 2016. The data from these missions will help understand the effects of microgravity in space, and therefore help construct future missions to Mars.

11.1. Muscular Atrophy

Prolonged exposure to a microgravity environment results in muscular atrophy. Both muscle mass and muscle strength is lost. Studies have shown that after approximately 4 months of exposure to low gravity, the muscle mass and strength reaches a new steady state (Williams, 2009); it is unknown if this steady state could persist for a 580-day mission. The muscle mass loss rate is rapid; in the first two weeks of microgravity there is a 20% loss in mass. Upon re-entry in a gravity field, this muscle loss can be so high that it can take weeks of rehabilitation before being able to properly stand upright and walk. To counteract this muscle loss, it is important to condition the body optimally before flight by constructing a suitable exercise program, and during the mission to perform a variety of different exercises using on-board fitness tools. Both upper body and lower body exercise equipment should be supplied, and so modified versions of the Shuttle's treadmill and rowing machine shall be provided on-board. This equipment is relatively easily storable and require low-to-no power to be operated. Supplementing the crew's diet with amino acids shall also help counteract the risk of muscle loss.

11.2. Bone Demineralization

As well as muscle loss, astronauts also experience a loss of bone density as well as a rapid increase in the loss of calcium. It has been noted that after a 6-month stay on the ISS, bone density decreases by as much as 12% [52]. This can be catastrophic upon return to a high-gravity environment, as the bones may not be strong enough to carry the weight of the person, and thus the risk of bone fracture highly increases. It is noted that it may take up to three years for most astronauts to regain pre-flight levels of bone density. It is important to take as many counteractive measures as possible during flight. Supplementing the diet with calcium and vitamins D and K, as well as other minerals and drugs like bisphosphonates, will help counteract the decrease in bone strength.

11.3. Fluid Redistribution

Body fluids redistribute towards the upper half of the body during microgravity. Indeed, there is around a 10% decrease in the total volume of blood in the legs through the first 24 hours of the mission [52] due to the body adapting to a higher central blood volume set point, and a total reduction of 10% over the course of the expedition. Aerobic exercising can be used to counteract the effects of fluid redistribution. Another method to simulate the gravitational gradient that the fluid system experiences is to use a lower body suction, or negative pressure, suit[53]. This pressure gradient helps regulate the fluid distribution and can help readjust before returning to a gravitational environment. This is important because upon return to Earth, many astronauts cannot stand up continuously due to light-headedness and heart palpitations [52].

11.4. Neurovestibular Effects (Space Motion Sickness)

Space motion sickness (SMS) is another problem that astronauts face, in particular during the first few days of flight. Around two-thirds of Space Shuttle astronauts suffered from SMS [53]. Symptoms of SMS are nausea, vomiting, pallor and cold sweats[53], as well as relatively minor symptoms like headaches and dizziness. SMS can cause the crewmember to feel drowsy, lethargic and apathetic – all affecting the cognitive and physical abilities of the astronauts. It is imperative to condition the crew to this effect pre-flight with appropriate simulations – parabolic flights and virtual reality training can help the astronaut gain experience in the motions of microgravity[52]. The effects of SMS can also be offset by drugs – promethazine is used to combat motion sickness, with dextroamphetamine utilised to offset the promethazine's sedation effects. Astronauts may also feel motion sickness and vertigo upon the return to a gravitational field, in particular those returning from a long-duration flight, and so it is important to have countermeasures ready on return to Earth.

11.5. Immune Dysregulation

The immune system of astronauts also tends to suffer from entering a microgravity environment. The development of pathogenic viral and bacterial infections is a serious threat to the whole crew, so an understanding of how immunity develops during spaceflight is vital. Whilst not much study has been performed on immune systems during flight, post-flight information is much more accessible. Numerous changes to the immune system occur, with the likely cause of impairment from both physical and psychological stresses experienced by astronauts [52]. A study of the crewmembers' immune systems and if there are any latent infections pre-flight should be undertaken, and it is suggested that a quarantine system be in place in case of infection of a crewmember – it may be possible to keep emergency supplies within the Orion capsule, where the infected astronaut may reside for several days until the risk of infection reduces. A range of antibiotics should be taken on board, with the crew developing an understanding of general symptoms to know how to deal with certain common diseases – a doctor on the ground can also help in these situations.

11.6. General Medical Care

As well as counteracting the effects of microgravity, general medical care must also be performed on board. Directives issued by NASA say that a space module should have access to the equivalent of the current best medical practices, with the equipment to prevent, diagnose and treat any medical problems that may arise [31]. The equipment to best communicate the patient's symptoms with NASA's medical doctors should be supplied. Other tools like heart monitors, blood pressure pumps, etc. should be on-board for general health check-ups, as well as tools for dental care. Whilst it is unfair to assume that the astronauts shall be trained to the same level as medical doctors, suitable training and study should be performed before flight to be able to assess any medical problems that may arise. Anaesthetics, intravenous fluids, pharmaceuticals and hyperbaric equipment should also be supplied, as well as advanced life support systems, diagnostic imagers and surgery tools in case it becomes necessary to operate. It may even be possible to utilise a robotic surgeon for the mission, with advancing research into humanoid robots like Robonaut 2 currently on-board the ISS [54]. This would save the concerns of having a non-surgeon crewmember operating. Portable breathing kits much like what is currently on the ISS should also be considered, in case of hypoxia or leakage of other chemicals like ammonia.

12. Risks

The three main components selected for the mission are based on systems that are flight proven or under development.

The launch segment, composed of the Falcon Heavy and SLS, is expected to be operational by the time of launch. Falcon Heavy is scheduled to have its first flight in the second half of 2016 and the SLS has an initial flight planned for 2021. In addition, the propulsion system for the cruise is based on the EUS, which is included in the SLS program. The presence of mission authorities on design reviews, especially Flight Readiness Review (FRR), will assure the scheduling for the final delivery of each launcher for the mission.

The Orion capsule is under advanced campaign of tests and qualification not representing a major risk for the mission. Its design complies with the requirements for deep space manned exploration regarding radiation protection and Earth atmospheric re-entry. Minor modifications for mass relief are possible but not necessarily expected, not representing a risk to the mission. Modifications are expected for the Habitat module regarding radiation shielding and internal configuration. However, much of the design is based on flight proven hardware developed and used on the ISS.

The concept of operation follows similar level of complexity of Apollo missions regarding modules' manoeuvres and docking. Considering this, extensive training for such procedures is expected for the crew. Aspects of long duration flight have been approached on ISS missions and are an object of experiment for the mission. The lessons learned will be put in practice with the crew monitoring regarding their physiological and psychological health.

The main development activities are related to the systems' interfaces considered single point of failure. The standardization and heritage of the interfaces in use at the ISS for docking are the main background, not representing the design of a completely new component. As a main mitigation measure, a dedicated test and qualification campaign shall be performed up to the delivery for the complete system integration.

The high-level risks, the consequences and the mitigation measures are presented in the table 16.

	Risk	Consequence	Likelihood	Severity	Mitigation
Development	Crew vehicle readiness/delivery on time	Unable to launch	1	5	Delivery plan with systems providers
	Habitat module readiness/delivery on time	Unable to launch	1	5	Delivery plan with systems providers
	Launch vehicle readiness/delivery on time	Unable to launch	1	5	Delivery plan with systems providers
Launch	Systems launch delay	Postponed operations	2	2	Operations in advanced for launch window
	Systems launch failure	Loss of major systems	1	5	Use of qualified or flight proved design
	Crew launch delay	Postponed operations	2	2	Operations in advanced for launch window
	Crew launch failure	Loss of the crew	1	5	Use of qualified or flight proved design Flight abort system
Operation	Crew and systems dock failure	Unable to transit	1	4	Crew training and interfaces tests
	Trajectory failure	Unable to transit	1	4	General system check prior burn
	Solar radiation peaks	Threat to crew and systems	1	3	Shielding modification for the habitat Crew emergency procedures and individual radiation shielded cubicles Mission schedule considering solar cycles
	Habitat failure	Threat to crew and systems	1	4	Use of flight proved design Mechanical/electrical ground tests of interfaces Orion used as "lifeboat"
	Habitat modification failure	Crew contamination	2	4	Use of qualified or flight proved design Mechanical/electrical ground tests of interfaces
	Interface capsule/habitat failure	Loss of crew	1	5	Use of qualified or flight proved design Mechanical/electrical ground tests of interfaces
	Interface habitat/upper stage failure	Loss of the propulsion Loss of the crew	1	5	Use of qualified or flight proved design Mechanical/electrical ground tests of interfaces
	Communication system failure	Loss of direct communication	2	3	Communication periods apart from operational critical events Use of alternative/redundant system
	Power system failure	Systems partially or totally dead	1	4	Ground tests of batteries and solar panels Orion/habitat redundancy
	Life support system failure	Threat to crew	1	4	Use of qualified or flight proved design Orion/habitat redundancy Algae bioreactor
Crew	Waste management failure	Limited crew operations	2	3	Use of qualified or flight proved design Ground cycle tests
	Crew illness	Unable to perform mission activities	3	3	Crew training and monitoring Medical supplies
	Crew "backwards acceleration"	Physical injuries to the crew	2	2	Acceleration profile Crew monitoring
	Internal repairs/tools	Unable to perform mission activities	3	1	3D printer, filament and digital repository of printable items

Table 16 – High Level Risks

13. Costing

13.1. Advanced Mission Cost Model

The Johnson Space Centre developed the Advanced Mission Cost Model (AMCM). The model is formed by the input of a database of more than 260 programs which includes top-level cost, system mass and multiple factors that helps to fit the model. The model relates six different variables to the total system cost in USD millions of fiscal year 1999 (FY1999$).

$$Cost\ (99\$) = \alpha Q^{\beta} M^{\Xi} \delta^{S} \epsilon^{\frac{1}{IOC-1900}} B^{\phi} \gamma^{D} \tag{13-1}$$

The variable parameters of the equation are:

- Quantity (Q): This variable includes the number of identical spacecraft that will be made. For the Inspiration Mars mission, only one spacecraft will be made.
- Dry Mass in lb (M): This variable is the dry mass of the spacecraft which was determined.
- Specification (S): This parameter determines the type of mission. For a human re-entry mission, the value is 2.17.
- Initial Operational Capability (IOC): This variable is the first year that the systems need to be operative. The Mars Gemini mission intends to launch in 2021, hence IOC is set to 2021.
- Block (B): The block number represents the level of design inheritance in the system. If the system is a new design, then the block number is 1. If the estimate represents a modification to an existing design, then the number increases per the inheritance experience. In our case, the design is done with already existing modules that has slight modifications, a block number of 3 is assumed.
- Difficulty (D): The difficulty factor represents the level of programmatic and technical difficulty anticipated for the new system. This difficulty should be assessed relative to other similar systems that have been developed in the past. This difficulty value is in the range of - 2.5 (extremely easy) to 2.5 (extremely difficult). This is a manned mission which is inherently complex. Nevertheless, the complexity relative to the Apollo mission or ISS is considerably low, therefore a value of -0.5 is set in the model.

Constants		Variables	
Parameter	**Value**	**Parameter**	**Value**
$\boldsymbol{\alpha}$	$5.65x10^{-4}$	Q	1
$\boldsymbol{\beta}$	0.5941	M	$106400\ lbs$
$\boldsymbol{\Xi}$	0.6604	S	2.17
$\boldsymbol{\delta}$	80.599	IOC	2021
$\boldsymbol{\epsilon}$	$3.8085x10^{-55}$	B	3
$\boldsymbol{\phi}$	-0.3553	D	-0.5
$\boldsymbol{\gamma}$	1.5691		

Table 17 - Constant Parameters and variables for AMCM model

The AMCM provides the cost in 1999$, hence the value should be corrected by the inflation to adapt the cost to the mission year, in this case 2021. The inflation conversion factor from 1999 to 2021 is 1.422.

The corrected estimated cost of the AMCM given by the equation (13-1) is USD$ 4400 Million.

13.2. CERs and Cost Breakdown

As some of the cost are already known, a more accurate estimation of the cost can be obtained by combining estimation models and already known values. The cost estimations have been made by CERs estimations and wrapping factors. [9].

13.2.1. Wrap factors

The wrap factor estimates the cost to a space project that are not directly related to the development of the spacecraft itself. This cost is usually represented as a percentage of the total cost of the spacecraft. CERs have been used to compute those costs [9].

WBS Element	Factor	Cost in $M
Annual Operations and Support for Ground Station	1%	$17M
Project Systems Engineering	15%	$255M
Project Management	10%	$170M
Systems, integration and test	12%	$204M
Product Assurance	3%	$51M
Configuration Management	4%	$68M
Contractor and subcontractor Fee	10%	$170M
Development Support Facility	1%	$17M
Hardware/Software Integration	13%	$221M
Integrated Logistics	6%	$102M
Safety and Mission Assurance	7%	$119M
Site Activation	1%	$17M

Table 18 - Common Wrap factors for space projects [9]

13.2.2. Reserve Factor

The reserve factor considers unplanned adverse events and the cost of management. A reserve estimation is useful in case a contractor overruns, a system test was not successful, or a technology development does not meet the required delivery date. The reserve of the Mars Gemini mission is based on the NASA Headquarters Reserve Model, where a risk factor can be applied on a scale from 0 to 16 (no risk to very high). The risk factors are presented in table 19.

Risk Factor Description	Risk Factor	Weight	Product
Planning definition	10	0.3	3
Design Heritage	8	0.2	1.6
Hardware/Software complexity	15	0.1	1.5
Difficulties in integration	8	0.2	1.6
Organization Complexity	7	0.1	0.7

Requirements for simultaneous development	8	0.05	0.4
Experience base	6	0.05	0.3
Total Scoring			9.1

Table 19 - Risk factor table for reserve computation

From the guide to reserve factors, a score of 9.15 corresponds to a reserve factor of 45% which is used in the total cost estimation.

Phase	Description	Total in FY21	Percentage	Source
A-D	Development Support Facility	$17M	1%	Table 17
A-D	Project Management	$170M	5%	Table 17
B	Project Systems Engineering	$255M	8%	Table 17
B	Safety and Mission Assurance	$119M	4%	Table 17
B-D	Configuration Management	$68M	2%	Table 17
C/D	Orion Module	$1000M	31%	[55]
C/D	SLS	$516M	16%	[56]
C/D	PPLM	$185M	6%	[57]
C/D	Contractor and subcontractor Fee	$170M	5%	Table 17
C/D	Hardware/Software Integration	$221M	7%	Table 17
C/D	Integrated Logistics	$102M	3%	Table 17
C/D	Systems, integration and test	$204M	6%	Table 17
E	Product Assurance	$51M	2%	Table 17
E	Site Activation	$17M	1%	Table 17
E	Launch Vehicles & Services	$135M	4%	[9]
E	Annual Operations and Support for Ground Station	$16M	1%	Table 17
Total		$3248M	100%	
	Reserves	$1462M	45%	Reserve Factor, p45
Total		$4709M		

Table 20 - Cost Breakdown

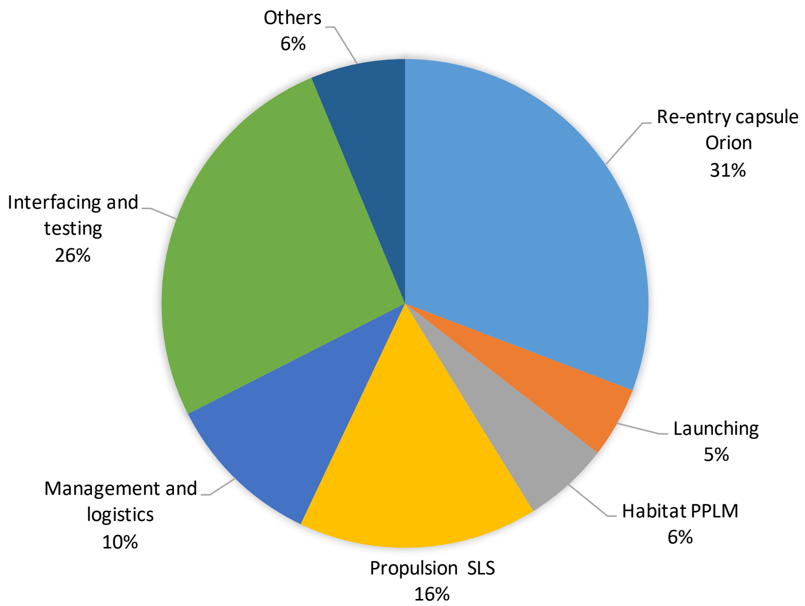


Figure 21 - Cost distribution by WP elements

13.3. Cost distribution along the project

A cost estimate phasing model was developed by Burges (2004) using a Weibull distribution function [9]. The Weibull parameters are estimated from other space projects. The computation of these parameters can be very complex, but it can be simplified to the model shown in (13-2).

$$E(t) = d\left[Rt + 1 - e^{-\alpha t^{\beta}}\right] \tag{13-2}$$
$$d = \frac{Total\ Cost}{[R + 1 - e^{-\alpha}]}$$
$$R = 0.00148 \cdot duration(months)$$
$$\alpha = -0.414 + 0.0729 \cdot units + 0.0488 \cdot months + 0.0145$$
$$\beta = 1.71$$
$$t = time/months$$

The results of the cost distribution can be found in figure 21 and the investment requirements in figure 22.

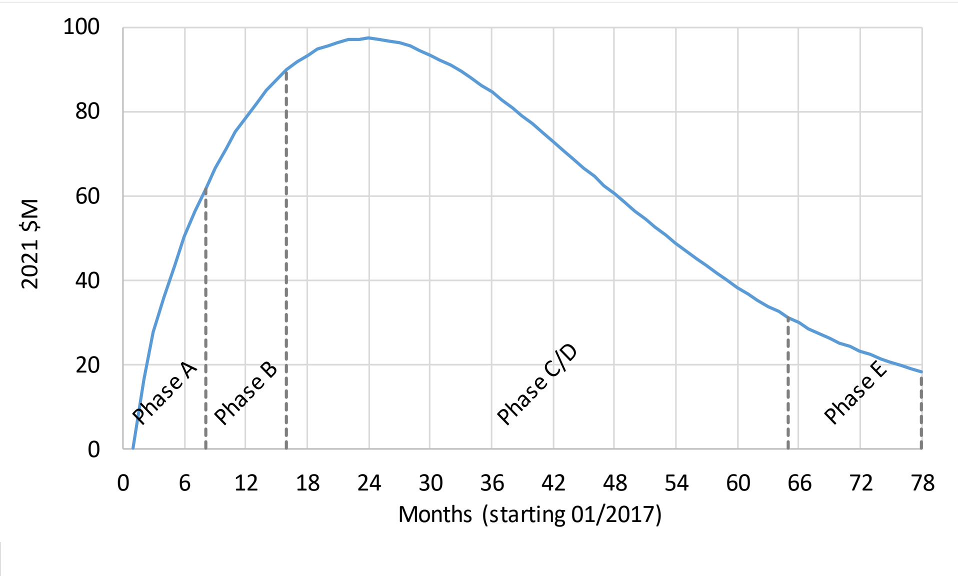


Figure 22 - Investment Requirements chart along the project

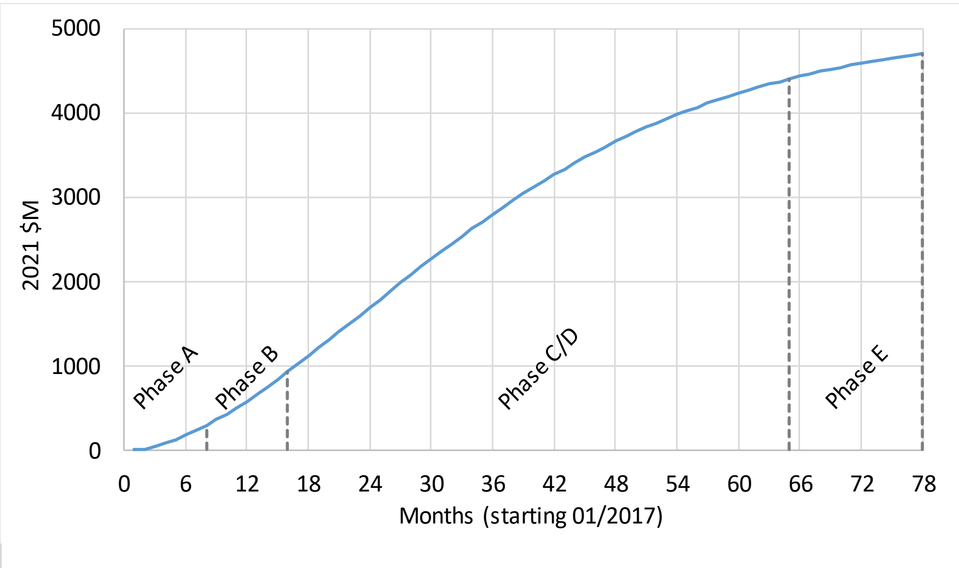


Figure 23 – Cumulative cost chart through over each project phase

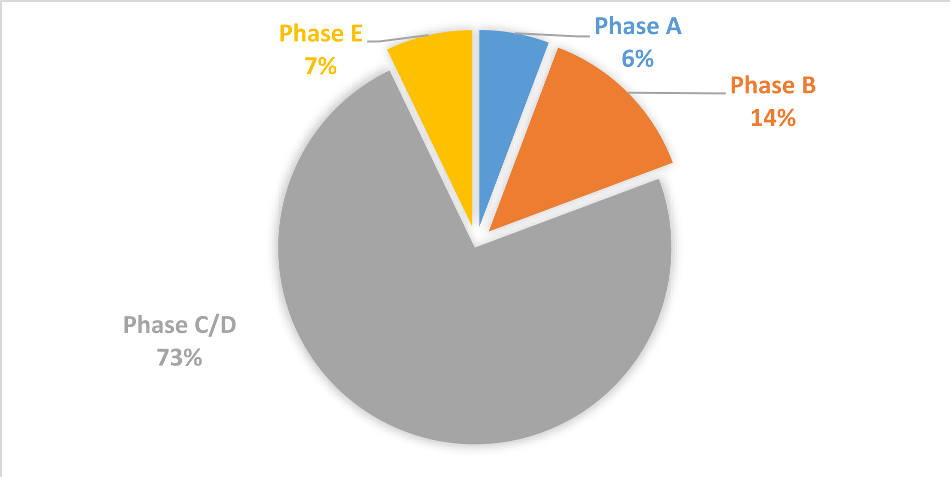


Figure 24 - Phase cost pie chart

As seen, both the AMCM and the breakdown with CERs give similar results around $4500M. Phase C/D is the most critical part of the project in cost terms, and hence, it would require more attention in cost managing during this phase.

14. Conclusion

A manned Mars flyby was designed to accommodate two astronauts for a 580-day journey. The mission meets the requirements outlined in Section 2. The use of pre-existing technology allowed for a significant reduction in development costs as well as reducing time between design and launch. The risks of the mission were kept low by using components and systems with long and successful heritages from previous manned spacecraft. This methodology made up much of the primary systems on-board. To reduce radiation exposure, it was decided to launch in the early years of 2020 during solar minimum. Passive radiation shielding modifications to the habitat module meets the 730 mSv maximum allowable dosage for the entire journey. The choice of pre-existing spacecraft for the configuration of the propulsion system, Habitat module and launch and re-entry capsule were based upon, ease of modification, ability to fit within existing launchers and heritage. The mission offers a unique opportunity to test a variety of new technologies in an interplanetary environment. These range from biological based life support to laser communication. As well as technology, the study of human psychology for such a mission has never occurred before and the information learned will be invaluable to future missions.

The goal is challenging but with the correct design considerations outlined in this report, the mission is feasible and will enable the human exploration of Mars and beyond.

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