Low Mass Human Mission to Mars
Blue Team - Logistics Report

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Submitted March 20, 2022

***Abstract*—Pythom Space is planning to bring the first human to Mars and back. The mountaineers Tina and Tom, founders of Pythom Space are planning to set their feet on the Red planet before any other person. The C.R.I.M.S.O.N mission is planning to make this possible, the goal was to design a low-mass human mission to Mars which would be launched in the next six years. A larger transfer vehicle shall be assembled in LEO in order to bring the two astronauts safely onto the red planet and back. In this paper, the In-space assembly of this transfer vehicle and a preliminary assembly plan is discussed. Individual subsystems such as power, communication, and thermal control were sized and are elaborated as well. Diverse technology was investigated and numerous trade-off studies had to be performed to arrive at the final design presented in this paper. Finally, diverse off-nominal scenarios were investigated and potential mitigation solutions are discussed.**

***Index Terms*—Interplanetary, Human Spaceflight, Mars, Martian expedition, Logistics, Low Mass, In Space Assembly**

I. Introduction

With an increasing interest in human spaceflight, space tourism and exploration of other celestial bodies in recent years, there has been an increasing number of private actors joining the market of human spaceflight. Rapidly growing companies such as SpaceX and Blue Origin have increased the accessibility of space for humans and the probability for successfully landing humans on Mars only increase with time. Even smaller companies have shown interest in these kinds of missions. The start-up Pythomspace visualise making a successful mission to Mars within the next six years. The aim of this project is to conceptually design this mission to Mars by making it as simple and low mass as possible but still with an adequate chance of success. Pythomspace is planning on using their own launcher called Kang, which is currently being developed, capable of bringing 3 000 kg into LEO.

This paper covers the logistics of the mission, more specifically the In-space assembly of the transfer vehicle, the assembly plan, power supply, communication and thermal control. Furthermore, it includes a trade-off study of a robotic arm and possible solutions for maintenance during the mission.

II. In Space Assembly

With the rapid development of human space flight technology, the traditional launch vehicle approach does not meet the current requirements anymore. Therefore, in-space assembly (ISA) of larger structures and vehicles is becoming an essential part of the future of space exploration. ISA enables to tailor spacecraft performances to their given mission and reduce operating costs. [1]

One of the main tasks of the logistics team was to determine a way to assemble the transfer vehicle in LEO. For that, a literature review was performed to determine what possible technologies exist, and which of them would be suitable for the application of the Mars mission.

A. Classification of In-Space Assembly

There exist 4 main ISA technology types [1]:

First, there is the mating between two elements, seen in Figure 1. This approach consists of having two or more independent modules dock into a larger structure. This can be done by docking or berthing, depending on the equipment, the attitude and orbital control system (AOCS), and the launch vehicle. This represents the most basic and simplest assembly technology. It has already been done numerous times and is proven to be a feasible method that is easily automated compared to the other approaches. The main parts of the ISS were built using mating between two elements.

Then there is the modular assembly approach, Figure 1. By docking two elements and assembling them, a larger structure is formed. This can include some sort of welding or other connection methods. For modular assemblies commonly a robotic arm is used, assisted by extravehicular activities (EVAs). By doing so larger and slightly more complex structures can be assembled and installed. This is more complex to be fully automated. On the ISS the solar array wings can be considered as modular assemblies.

Third, there is the complex assembly, Figure 1. It is defined as the assembly of several components using an assembly sequence with numerous operations. This has also been done in the past by EVAs with the assistance of robotic arms. However, this is unlikely to be fully automated nowadays.

Finally, the assembly from parts consists of the complex joining of individual components which are then assembled to a larger structure which is visualized in Figure 1. An example would be building a computer from the CPU, heat sink, and other parts. This is the most complex form of the ISA and with current technology is unfeasible to be fully automated.

B. In-Space Assembly Technology Selection

As the mission design team [2] planned to launch from LEO in 2026, it was agreed with them that the assembly of the spacecraft (SC) shall start in 2024. Therefore, it is important to select technologies with a high technology readiness level (TRL). This assures the feasibility of the ISA and mitigates the risk of delays or assembly issues. Furthermore, by using

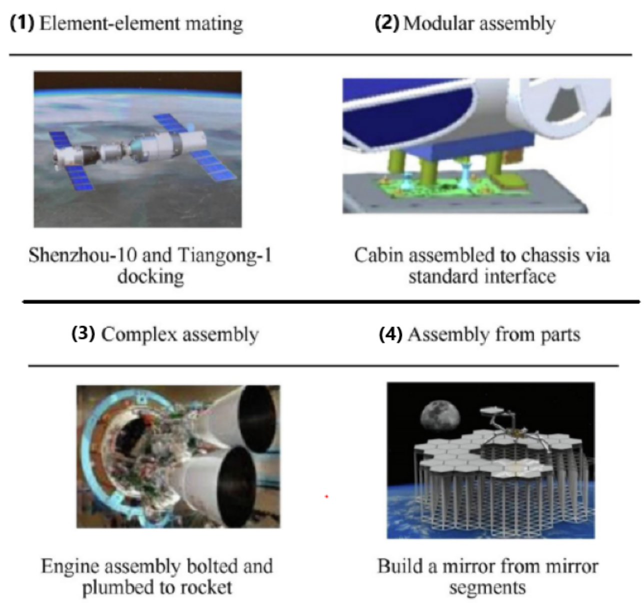


Fig. 1. In-space assembly technologies [1]

high TRL levels, the possibility to find suitable manufacturers and off-the-shelf components is increased.

Since Tina and Tom said that they wanted to minimize the work and maintenance they have to perform on their way to Mars. It was decided that the assembly of the SC shall be done fully automatically and/or controlled from Earth. Therefore, for the initial assembly, EVAs are not an option and were only considered as a repair approach for off-nominal cases.

Thus it was clear that the complex assembly and assembly from parts are too complicated to automate to be considered a feasible ISA approach. This means the major assembly trade-off was held between the element-element mating and the modular assembly.

The core difference between these two approaches is that one uses standard docking systems which connect modules together with integrated power and data connection. Whereas the assembly uses other mating ways and possibly secures the connection with other techniques. Often additional power and data connections need to be added.

The major benefit of the modular assembly is that it gives more freedom in the design as no major connection nodes are necessary. The mating of elements in comparison will always require a heavy and voluminous docking system. The drawback of the modular assembly is that individual components will require a robotic arm to assure the precise assembly of the system. Furthermore, the technology of these alternative connection systems is not as developed as the standard docking technology. It is unclear if this sort of assembly procedure would be able to be fully automated in such a short period of time.

In conclusion, the docking method would be the safer but less performing option, and the modular assembly would be slightly more complex. However, it would significantly reduce the mass and increase the performance of the SC, especially for more complex and ramified designs. A robotic arm can be included for both approaches depending on the number of modules and other factors.

In order to select the adequate ISA technology, close co-operation with the transfer team was required. Several design options were discussed with the transfer team and the assembly possibilities were debated in order to arrive at a final design that is both feasible to assemble and assures all necessary capacities. It was a dynamic process as new challenges kept appearing and previously solved problems were questioned again which changed the ISA approach several times.

However, the main driver in the decision of the ISA technology was the final design of the vehicle. As it was a simple geometry that mostly consists of a long tube, it was clear that the simpler docking approach would be easier to be implemented and assures no delays in the assembly process.

C. Automatic Docking or Berthing Assembly

With the ISA process decided the final trade-off was if the modules shall be automatically docked, or if an automatic berthing mechanism shall assemble the transfer vehicle.

The automatic docking method implies that each module has its own AOCS system and communication system in order to navigate to the assembly structure and dock safely. Furthermore, automatic docking takes longer than berthing as more precision is required and the slightest offset can have severe consequences.

The berthing on the other side uses the assistance of a robotic arm to capture the incoming module and safely guide it to the existing structure. This requires an additional robotic arm which would have to be fully automated or controlled from a ground station. Developing a new robotic arm would be unfeasible, and therefore a potential existing model was chosen.

The European robotic arm (ERA) was chosen to be a suitable option for the berthing. It was discussed if the ISS could assist the assembly. However, due to the long duration of the assembly and the limited utilization time of the ISS this proposal was dismissed. Each module could have an individual connection port for the robotic arm so it can make its way along with the spacecraft. This would enable easy assembly of incoming modules.

The robotic arms, which are found on the International Space Station (ISS) or NASA's Space Shuttles, were considered as part of the ISA. A potential idea was that the robotic arm assists the assembly process. After the assembly is completed the robotic arm shall be undocked from the transfer vehicle to economize mass. The arm would then be sold to potential companies like Axiom Space which could implement the robotic arm into their commercial space station. [3] As the robotic arm has its advantages and drawbacks an adequate trade-off study was performed. Here the key points are briefly summarised:

Advantages of the robotic arm

- *Assist docking of modules*: During the assembly process, the robotic arm can be used to dock the modules to

each other. These would reduce the required fuel for the AOCS. Furthermore, docking failures are mitigated.
- *Could be used for modular assembly*: With a robotic arm, the assembly can be more complex and allows more design freedom. The arm could switch places in between the modules for better support.
- *Assist during Off-Nomical cases*: During off-nominal cases, the robotic arm could support an initial EVA before it is undocked.

Drawbacks of the robotic arm

- *High price*: The original Canadarm1, which was used on Space Shuttle, had a price which was over 100 million dollars [4]. This represents a significant part of the budget which would be spent on a sub-system that is not used after ISA.
- *Additional mass*: The Canadarm2 has an approximate mass of 1500 kg [5]. The ERA arm, on the other side, has a mass of 630 kg due to its reduced size [6]. An additional subsystem would most likely require an additional launch vehicle. Furthermore, adequate connection ports on several modules are required to accommodate the robotic arm. These connection ports will increase the transfer vehicle's mass and occupy useful space.
- *AOCS system*: Even if the robotic arm is a great asset for the assembly process. Each module will still require an AOCS system.
- *More complex assembly*: Adding a robotic arm to the assembly will ultimately lead to a more complex system, even if the goal is to have a low mass and safe mission to Mars.

The berthing approach strongly depends on the selected launch vehicle. As the Khan rocket is still being designed it is unclear what kind of AOCS system the launcher has and if it could perform a rendezvous maneuver with the assembly site. As the launcher is supposed to be an economical option to bring a variety of cargo to space, it was assumed that the launcher would not be able to navigate into the reach of the robotic arm. [7] Thus it was assumed that the modules would require their own AOCS system either way; a rough estimate is presented in Section VI. This meant additional mass due to the robotic arm and the AOCS system.

In order to come to a final decision, one major factor was the number of modules that have to be assembled. This number remained unknown until close to the end of the project.

Finally, the decision was made to not include the robotic arm and use the automatic docking process. Thus the modules will carry slightly more fuel for the AOCS system. The major reasons that led to this decision were the high price of the robotic arm and the uncertainty if the ESA would sell such a cutting-edge piece of technology. Neither is it clear if a new robotic arm could be manufactured in such a small time window. The AOCS system, on the other hand, is a common off-the-shelf product that is used in a variety of satellites and other research projects. Large manufacturers like MT Aerospace or Ariane Group can provide the required hardware. [8]

III. POWER SUPPLY

In order for the mission to be successful, it requires a reliable and robust power system. The challenge was to design an according system based on the spacecraft configuration determined by the vehicle design team and the previously chosen assembly system. [9]
The power system addresses several aspects: power generation, storage and distribution. In the past, human space flight relied on solar arrays as a continuous power source and on batteries during eclipses. This was a feasible and simple solution as humans have never travelled further than the Moon. However, Mars is 200 times further away than the Moon, this brings up some interesting challenges. [10]

A. During the transfer to Mars

To determine the most suitable power supply an initial literature review was performed. In the book Spacecraft System Engineering [11] a useful sizing tool, Figure 2, gave a general idea of which power systems could be relevant for the Mars mission.

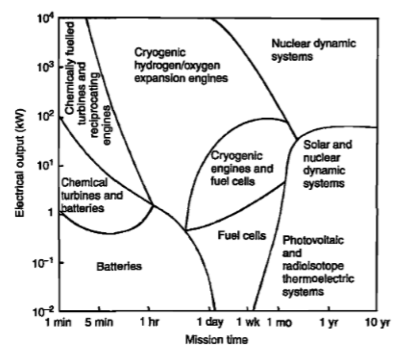


Fig. 2. Relationship between energy source and appropriate operational scenario [11]

In order to use the sizing tool correctly, a power budget with adequate margins was required. Since it is only a concept design, individual teams weren't able to provide a power budget. Therefore, a conservative power budget was determined based on the ISS. [12]

$$P_{SC} = \frac{P_{ISS} * n_{SC}}{2 * n_{ISS}} = 11.4kW \quad (1)$$

Where n_{SC} represents the number of astronauts on the SC to Mars and n_{ISS} the number of astronauts on the ISS. Furthermore, the normalised power consumption was halved as there are a significant number of science experiments on the ISS which consume a lot of power. The determined power consumption of $11.4kW$ is quite high for a two-person SC. However, this high estimate will compensate for any deployment or maintenance issues. The power consumption value can be revised during the detailed design phase of the mission.

Solar arrays provide the best specific power supply $26 - 100W/kg$ while close to the sun. A table that summarises the statistic of the individual power sources can be seen in appendix A12. Further away from the Sun a nuclear reactor could be a favourable alternative. However, due to the high risk and radiation issues, this alternative was disregarded. Other options such as radioisotope thermoelectric generators and fuel cells cannot compete with the long-lasting high specific power of the solar arrays. Solar dynamic generators, which according to Figure 2 seem to be a reasonable alternative. However, their deployment is quite complex. The mission design team scheduled the beginning of the in-orbit assembly for 2024 and the launch from LEO in 2026. Therefore, a complex deployment mechanism and an innovative power supply was estimated to not be the right choice. Therefore the primary power supply during the transfer to Mars was decided to be a set of solar arrays. [13]

Based on power consumption, the required solar array area was determined. In Space Mission Analysis and Design (Figure 2) the following formula is given to size solar arrays:

$$ArrayArea = \frac{P_d}{S * L_p * \eta_{cell} * \eta_{packing} * (1 - D)} \quad (2)$$

Where P_d is the required power, L_p is solar incidence angle which is usually around 0° to 4°, η_{cell} is the efficiency of the solar cells which was assumed to be 33%, $\eta_{packing}$ is the packing efficiency of the cells, which was taken as 0.85 and D is the radiation degradation factor over the SC lifetime which was set to 0.9 according to the books recommendation. Finally, the solar flux is described by $S = \frac{1\,370}{R^2}$ where R is the distance to the sun and measured in AU. As the challenge is to provide power to the SC in Mars orbit the solar flux at Mars was determined. The solar flux around Mars is only 43% of that around Earth. These values were used to calculate the required array area which came out to be $688m^2$.

Once the array area was known the next challenge was to determine a suitable deployment mechanism. As it was made clear in the previous section the SC will not perform any modular assembly nor have a robotic arm to deploy the solar panels. Thus the solar arrays will have to deploy automatically and without any assistance. Therefore a specific power module was originated, that can be seen in Figure 12. The module consists of a structural core with the pre-loaded deployment mechanisms on the side. The power module is supposed to be a core part of the vehicle as this will reduce the need for primary batteries during the initial vehicle assembly.

The Solar module was inspired by the Roll-out Solar Arrays (ROSA) attached to the ISS. [14] These consists of solar arrays which are stored in a rolled configuration and once installed, they deploy by using controlled metallic tape springs. On the ISS the ROSA systems were installed by EVAs in order to be deployed above the existing solar array wings (SAWs). For the Mars mission however, the idea is to have 4 ROSA systems of dimensions 5×34.4m^2, 2 on each side of the module as seen in Figure 12. These will be attached to a mechanical hinge which will allow the solar arrays to point towards the sun. By using high precision sun sensors a quasi-orthogonal incidence angle is assured for optimal power production.

A core benefit of this deployment mechanism is that it power is also supplied while the arrays are only partially

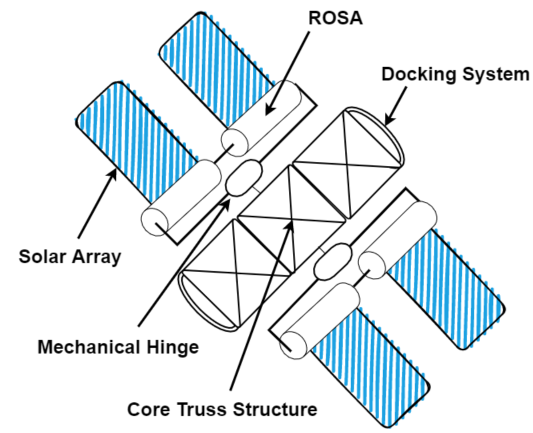


Fig. 3. Concept Design of the Power module (not to scale)

deployed. As it was seen earlier the solar flux around Mars is much weaker than around Earth. This means that while the SC is being assembled in LEO, the SC is only required to deploy $297m^2$ of its solar arrays to provide power to the entire vehicle. This avoids potential damage due to space debris or micometeoroids. Furthermore, it reduces the radiation degradation of a significant area of the solar arrays. This will assure better array efficiency around Mars and on its way back. Finally, in case two of the ROSA mechanisms jam during the initial deployment, there is redundancy. As the other two mechanisms can fully deploy to compensate for the deficient ROSAs. Once the crew is on-board an EVA can be performed to fix the jammed mechanisms.

Batteries are an essential secondary power source. A secondary power system provides power when the primary one is unavailable or during peak loading. For the photovoltaic cells this is the case when the SC is going through an eclipse. By eclipse, it is meant that the planet, in this case, either Mars or the Earth is blocking the Sun from the space vehicle. During this period the spacecraft is fully powered by batteries. Once out of the eclipse the batteries need to be recharged. Batteries in LEO experience around 5 to 6 thousand charge/discharge cycles per year. [10]

In order to size the batteries, it was necessary to first determine how long the SC would be in the eclipse of the respective planet. The derivation of the time in an eclipse of both the elliptical Mars and circular Earth orbit can be seen in the appendix A.

The time in eclipse for Earth is $T_E = 35.1$min and for Mars $T_M = 2.56$min. This means that the limiting eclipse time is the one in orbit around Earth.

A common characteristic for secondary batteries in LEO is that they encounter a lot of short eclipses which only require a lower depth-of-discharge. In contrast in GEO an eclipse can last up to several hours and even days. There high DoDs are required to assure power for the satellite. [11]

The battery type was selected to be a nickel-cadmium (NiCd) battery based on the number and duration of eclipse cycles in LEO. The NiCd provides a high cycle life and a small DoD of 20%. Furthermore, the batteries can be fast-charged and trickle charged. A significant benefit is that they can be left for a long time and still recover, this is useful as the orbit period around Mars is relatively long. In the Appendix, a brief overview of the main secondary batteries used in space flight is given, figure 13.

To determine an acceptable capacity following formula was used from the book Space Mission Analysis and Design [13]:

$$C_r = \frac{P_e T_e}{DoD * N + \eta} \tag{3}$$

Where P_e is the average required eclipse load, in this case 11.4 kW, the T_e is the time in eclipse in hours, DoD represents the depth-of-discharge, N is the number of batteries and η is the efficiency of the power path, which was taken as 85%.

In order to have a reasonable battery capacity, in between $200 - 600$ Whrs, the number of batteries was adjusted accordingly. A final number of six batteries was determined with a capacity of 452 Whrs. This didn't include any backup batteries. As the power system is crucial for the survival of the astronauts, 3 additional batteries were added for redundancy. This adds up to a total of 9 NiCd batteries.

The power distribution system of a space vehicle depends on the load requirements, the different subsystems and the source characteristics. The space shuttle for example uses a 28VDC power bus as the fuel cells provide a low voltage and the power demand is intermediate. The ISS in comparison has a power bus that uses 124 VDC this is due to the high source voltage of the solar arrays. Furthermore, a lot of scientific experiments require high voltage. [12]

For the Mars mission, the solar arrays will also have a comparable high source voltage. Furthermore, radar and navigation systems all require high voltages between 5-270 VDC. The life support systems all consume a significant amount of power, having a high voltage will help to keep the current down. Therefore a similar 124 VDC power bus was determined to be the preferred choice for the SC. Adequate voltage dividers were planned be used to regulate the output for individual subsystems, as well as power inverters shall convert DC to AC current if needed. Finally, the main advantage of using a high voltage power bus is that it reduces the size and mass of the required wires and machinery, as a lower current is used.

B. *During Mars operations*

The power supply on Mars presents other challenges in comparison with the power supply during the transfer to Mars. Since the operation on Mars is limited to 23 days, the durability of the power source, which is a considerable selling point of the solar panels, is of less importance than during the transfer. Fuel cells are a compact and highly efficient source of power, which is suitable to the limited volume and mass budget of the Mars descent vehicles. In fact, fuel cells can convert chemical energy directly to electrical energy with an efficiency capable of exceeding 60% while producing zero emissions. Hydrogen-powered fuel cells only emit water and heat which means that the crew on Mars does not necessarily have to spend time deploying this system outside. [15] Choosing solar panels for this part of the mission implies needing batteries to supply power during nighttime, whereas the fuel cells are a reliable source of power as long as fuel is fed through the system. Solar panels also require the crew to spend an entire EVA, thus at least one entire day, to deploy the solar panels. Therefore, fuel cells save the crew valuable time on the Mars surface during the short stay.

Fuel cells were already used on the Space Shuttle and since then the technology has only improved. In recent years, the US Department of Energy along with the private industry has made significant advances in the development of PEM (Proton Exchange Membrane) fuel cells. These use hydrogen and oxygen for fuel and oxidants. NASA is currently building upon this technology to create compact, reliable, and high-energy renewable power sources for aerospace applications. The targeted power output for these space systems is 1 to 10 kW, which is significantly higher than what is required for the Mars operation. [16]

Resulting from this research of power supply during the Mars operations, the PEM fuel cells were chosen as the power supply system for this part of the mission.

To determine the size of the fuel cells and how much hydrogen is needed to fuel the power system for the entire stay on the Martian surface, some decisions had to be made in terms of the desired performance. From the Mars operations team, the required power output was obtained. They demanded a continuous power output of 300 W for the entire stay, which corresponds to 165.6 kWh. [17] In addition to the power required, the output voltage had to be chosen to determine the size of the cell stack. It was set to be the same output voltage as the ISS and the transfer vehicle, 124 V. The sizing of the fuel cells and how much fuel is needed was made with adequate estimates and equations from Keith [18].

Starting with the current and the number of fuel cells needed to produce the desired amount of voltage. The current was calculated as shown in equation (5).

$$I = \frac{P}{U} \tag{4}$$

Where the power was taken as $P = 300$ W and the voltage $U = 124$ V, this led to a current of $I = 2.42$ A. A rule of thumb when determining the voltage of a single fuel cell is to assume 0.7 V. [18] Hence 178 fuel cells are needed in the fuel cell stack to provide the correct voltage. To obtain a reasonable estimate of the mass of each fuel cell, the Toyota Mirai was considered since it uses the same kind of titanium PEM fuel cells which have been proven to work. The mass of each fuel cell is 102 g which makes the total mass of the cell stack to be 18.2 kg. [19]

To determine the amount of fuel needed for the entire stay on Mars, again equations from [18] were used. The hydrogen consumption is given by:

$$\xi_{H2} = \frac{IN}{zF} \tag{5}$$

Where I is the current in Ampere, N is the number of cells, z is the number of electrons produced per mole of fuel and F is Faraday's constant. With $I = 2.42$ A, $N = 178$,

$z = 2\frac{mol_e}{mol_{H_2}}$ and $F = 96\ 485\frac{C}{mol_e}$, the hydrogen consumption is $\xi_{H2} = 0.0045\frac{g}{s}$. Thus, assuming an efficiency of 50%, a total of 17.9 kg of hydrogen was determined to be needed for 23 days of continuous power supply.

To minimize the volume and mass of the fuel tank, the hydrogen has to be stored in liquid form in a cryogenic tank. Assuming a cryogenic tank with 8.5 bar of pressure, a tank with a capacity of 30 liters is needed, given that the density of liquid hydrogen is $\rho_H = 70.9\frac{kg}{m^3}$. To estimate the mass of the empty tank, again the Toyota Mirai was used as a reference. Scaling down the size of that hydrogen tank gave an approximate mass of 21 kg for a 30-liter tank. However, since the tank is only a high-pressure tank and not cryogenic, an additional 5 kg was added to the tank mass.

The total mass for the power supply system was determined to be $18.2 + 17.9 + 26 = 62.1$ kg.

IV. COMMUNICATIONS

Communications from Mars to the Earth and vice-versa usually take from 5 to 20 minutes depending on the relative distance between the two planets that can be as small as 56 million kilometers or as large as 401 million kilometers. The biggest complex of antennas on Earth, that can be used to communicate with interplanetary missions, is the Deep Space Network (DSN) owned by NASA. So permission to use the Network has to be asked before designing the communication systems.

In case of emergencies or malfunction of the DSN, messages could also be sent to the ESA's Estrack. This complex is made up of a core of antennas present in French Guiana, Belgium, Sweden, and Portugal, and the Deep Space Antennas complex present in Australia, Spain, and Argentina. Together with additional stations, the European complex is fully capable of tracking and communicating, with S- and X- band, signals to the spacecrafts in the Solar System. [20]

The ranging of the spacecraft is done in collaboration with the DSN sending lines of code to the vehicle, sending it back immediately, and measuring the time elapsed. Taking into account the delay time from the electronics, it's possible to measure the position with a precision of 5 to 10 meters up to 200 million kilometers away. [21]

A. Parabolic antenna

The main antenna would be very similar to the one onboard ESA's ExoMars Trace Gas Orbiters (TGO) which is a 2.2-meter parabolic High-Gain Antenna that uses an X-Band system. [22] The data transmission rates can be as low as 0.5 Mbit/s or high as 4 Mbit/s depending on the position of the two planets. In case of problems with direct communication, it could be attempted to transmit the data to the orbiters around Mars and use them as repeaters. These would be subjected to the delayed time for the long distances of space. Currently, around Mars, there are NASA's Mars Reconnaissance Orbiter, MAVEN, 2001 Mars Odyssey, ESA's ExoMars Trace Gas Orbiter and Mars Express. They also can communicate with the DSN at a maximum speed of 6 Mbit/s in the best case scenario [23].

B. Electra UHF Radio

As done with the ExoMars TGO and all NASA's missions to Mars, also the transfer vehicle would be retrofitted with an Electra UHF Radio provided by NASA (Figure 4). This now has become the standard system to communicate with rovers on the surface. It will be essential if communication between the landers and the transfer vehicle is required. The terminal consists of dual string UHF Transceivers, dual string Ultrastable Oscillators for precision navigation and surface positioning, and a low gain UHF antenna pointing nadir. [22] The small dimensions and especially the low mass (4.9 kg) make it a valuable instrument on-board. It can reach a max transmission rate of 2 Mbit/s. This communication system will also be adopted by the lander and all the ground segments of the mission. [17]

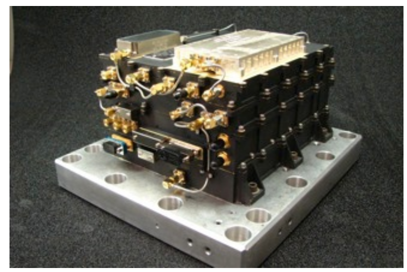


Fig. 4. NASA's Electra UHF Radio [22]

V. THERMAL CONTROL

A. General description

The thermal environment of the spacecraft varies significantly, especially one in orbit. For some periods, the vehicle is in direct sunlight and temperatures can reach over 200 C°, whilst in the eclipse of Earth and Mars, temperatures can reach -200 degrees [24]. The vacuum of space further complicates matters as two of the three types of heat transfer, convection and conduction, are almost non-existent. Parts of the vehicle, such as the outer structure may tolerate several hundred degrees, whilst systems such as the batteries and humans operate best at a narrow span around 20 C°. Heat is generated internally by the crew, mechanical devices, and electronics. Externally, the heat mainly comes from the sun, and albedo and black body radiation from the earth or Mars [11]. The thermal system of the space vehicle aims to provide a temperature balance, both locally and globally in the spacecraft. The thermal control consists of both active and passive components.

The passive system is comprised of three methods. Some of the heat can be reflected by coating the spacecraft in a heat reflective coating, such as a white paint ($\alpha = 0.15$) or reflective metal($\alpha = 0.08 - 25$) [11]. Moreover, the spacecraft will use cold plates that act as heat sinks and are placed on components with high thermal output. These cold plates draw heat away from the device close to the spacecraft and radiate it away into space. Lastly, the insulation of the cabin will enable a stable thermal environment.

The largest part of heat rejection comes from the active thermal control system, which consists of radiators and pumps which circulate a refrigerant throughout the spacecraft. The refrigerant was chosen to be glycol which has good thermal characteristics and is safe for human contact. Another common working fluid is anhydrous ammonia, it has slightly better characteristics but was rejected due to the danger it poses for the crew during a potential leak [25]. The glycol is pumped through the spacecraft absorbing heat energy, via heat exchangers and cold plates throughout the various systems and structures onboard. The heated glycol then passes through an array of radiators which radiate the heat before the glycol is once again pumped through the system.

Size estimate

Two main methods were used to calculate the size and weight of the radiator array. A rough estimate was calculated by simplifying and approximating the spacecraft and finding a thermal equilibrium temperature in orbit. Based on the mass of the vehicle the thermal energy which needs to be dissipated was estimated and finally, the radiator area was determined. The other method consisted of looking at the calculated power budget and scaling it similarly to the ISS and space shuttle. Due to the many unknowns of the spacecraft design, the latter was chosen. Based on this it was calculated that $12Kw$ of thermal energy needs to be dissipated this results in $16m^2$ of radiators with a total mass of $674kg$.

VI. Attitude and Orbital Control System

General description

The attitude and orbital control system (AOCS) is required to stabilize and orient the SC during its mission, as well as to change its orbit and control docking maneuvers. Thus it is important to be able to determine the attitude of the SC. Therefore each module shall be equipped with two inertial measurement units (IMU) which consist of high accuracy gyros and accelerometers in a tetrahedral constellation. Furthermore, to determine the precise orbit around the earth a horizon sensor shall be implemented for every module. These are lightweight but still offer a high precision of 0.1°. [13] By knowing the orbit and the attitude of the module precisely, a safe automatic docking procedure can be assured. To navigate safely through deep space, on its way to mars the vehicle shall be equipped with two-star trackers. These compare the location of the stars to existing star maps and therefore allow precise navigation through most of the known deep space.

The second part of the AOCS system consists of actuators. These correct the attitude of the SC after disturbances or avoid them from occurring in the first place. One of the most common ways to avoid the vehicle being disturbed is reaction wheels. These consist of brass discs spinning at relatively high speed. Commonly 4 reaction wheels are built up in a pyramidal way in order to stabilize all three axes as seen in figure 5. Due to the inertia the of the spinning discs, the SC is stabilised. Additionally, by speeding up or slowing down individual discs a corrective toque can be applied to the vehicle. These are usually used for small perturbations and overall stabilisation.

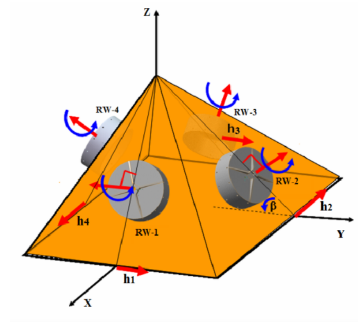


Fig. 5. Reaction Wheel Pyramidal Configuration [26]

In order to perform docking maneuvers and make large changes to the SC attitude, the reaction wheels are not powerful enough. Thus a 3-axis thruster system is required. This consists of 12 or 24 small thrusters which are commonly operated with hypergolic propellants such as liquid hydrazine, as seen in figure 6. This system can provide torque as well as direct forces in and around each axis by firing an adequate combination of thrusters. In order to determine a suitable thruster system, a case study was made based on the crew dragon to get a general overview. It was decided to use a similar approach. 12 Draco thrusters each providing up to 400 N of thrust shall be used to orientate the modules and assure safe docking. [27]

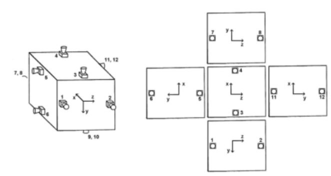


Fig. 6. 12 Thruster Configuration [28]

Attitude and Orbital Control System Mass Estimation

A rough estimation was done based on the AOCS mass of well-known capsules and spacecrafts, with respect to the total mass (wet mass) of the vehicle. The AOCS mass was normalized to the mass of individual modules. The total mass to AOCS mass ratio of the SpaceX Cargo Dragon was taken as a reference values: $\frac{total\ mass}{AOCS\ mass} = \frac{10\ tonnes}{1.3\ tonnes}$. [29] This helps to scale up the mass of the AOCS system based on the module size. Larger modules will require more fuel to be controlled. The calculations were performed iteratively and yielded the following results:

- *KANG rocket*: For each launch, a maximum mass of 3 tonnes was assumed. This estimation includes the payload mass as well as the fuel mass for the AOCS. The final "dry" payload was determined to be 2.6 tonnes while the fuel of the AOCS is 0.39 tonnes.
- *reusable Falcon 9*: The Mars descent and return vehicles have a 9 tonnes dry mass, with the AOCS fuel included the launcher will weigh 10.3 tonnes.
- *expendable Falcon 9*: The Habitat and the consumables were estimated to have a mass of 21 tonnes combined according to the Mars operation team. The required AOCS

fuel was determined to be 3.1 tonnes, thus exceeding the capabilities of the Falcon 9. But as the falcon 9 has had an increase in payload mass in these past years, it is reasonable to assume a steady increase till 2025/2026.

- *reusable Falcon Heavy*: Each of the Falcon Heavys needs to bring 44 tonnes of fuel and tanks as payload to the assembly orbit. The AOCS fuel was determined to have a mass of 6.6 tonnes, which brought the total mass up to 61 tonnes to LEO per Falcon Heavy launch.

the

VII. Docking Method

The older ISS's docking mechanisms are designed to handle large, 100-tonne craft as the Space Shuttle. The mechanism catches a spacecraft as it pushes into the station. This approach is not feasible anymore as the next generation of spacecraft will be smaller and lighter. A 10-tonne vessel would just bounce off these docking systems as it hasn't got enough momentum to engage the locking rings. [30]

A. The International Docking System Standards

To provide a new docking mechanism (axial view in Figure 7) for lighter space vehicles, a new international standard was defined and it's identical for both crafts. The connection is an international standard while the mechanism behind the docking ring can be designed by the individual space agency or company. The International Docking System Standard (IDSS) is built on the heritage of the Russian APAS system (Androgynous Peripheral Attachment System). [31]

Vehicles using this interface may include light vehicles in the range of 5-8 tonnes, and medium vehicles in the range of 8-25 tonnes. Once the rendezvous is completed and the active navigating vehicle has aligned itself with the passive component, the docking which is made up of two phases can begin. Firstly the Soft Capture System (SCS) is performed when the active docking mechanism's SCS aligns with and latches to the passive docking mechanism, then stabilizes the newly joined spacecraft relative to each other. The soft capture system then pulls the docking spacecraft together. Secondly, the Hard Capture System (HCS) is activated, latching and sealing the docking interface. This creates a pressurized transfer tunnel for crew and cargo operation. [32]

B. The International Berthing Docking Mechanism

The International Berthing Docking Mechanism (IBDM), is currently being developed by ESA, NASA, and Sierra Nevada Corporation. It is an androgynous, contact force-sensing, low-impact docking system, capable of docking and berthing large and small vehicles. It is fully compatible with the International Docking System Standard (IDSS). The dual active control loop improves on existing technology by reducing the docking forces on the space infrastructure and by enabling the capture and docking of a wide range of spacecraft mass and flight envelopes. [33] Thus it would be perfect to dock the modules to the transfer vehicle in construction around the Earth. As for the IDSS, the docking sequence is made of the Soft and the Hard Capture System. The estimated mass should be lower than 325 kg once completed.

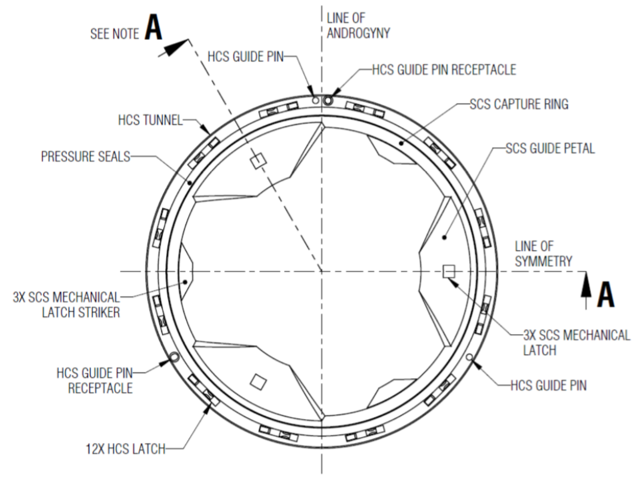


Fig. 7. Androgynous docking interface - Axial View [32]

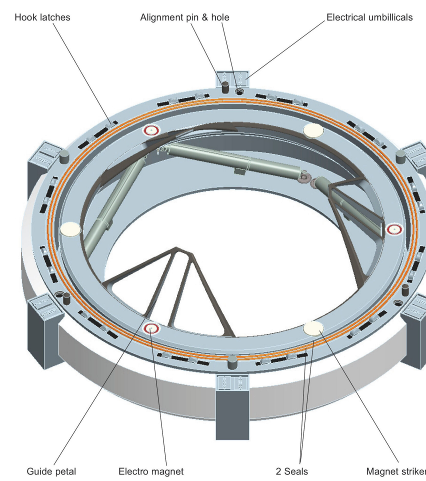


Fig. 8. International Berthing Docking Mechanism [33]

VIII. Number of Launches

The determination of the number of launches and which launcher to use was done in close collaboration with the Transfer Vehicle group. The mass of each individual segment and its estimated volume was provided by the transfer vehicle team. A "break down" of the mass was done and allocated to a single or combination of modules which shall be launched together in order to minimize the total number of launches and the total cost of the mission. A requirement to the mission was to use as many KANG launchers as possible, a new still-under-design launcher from PythomSpace. It's supposed to be able to carry up to 3 tonnes to Low Earth Orbit. For Falcon 9 and Falcon Heavy, the capabilities taken into consideration are presented in Table I.

TABLE I
SPACEX LAUNCHERS' CAPABILITIES

	Falcon 9	Falcon Heavy
reusable:	17 tonnes ($50M)	57 tonnes ($90M)
expandable:	23 tonnes ($60M)	64 tonnes ($150M)

Before computing the number of necessary launches also the fuel of the attitude & orbit control system (AOCS) has to be taken into account. This is done in Section VI. Finally, the allocation of the total mass of the transfer vehicle is shown in Table II.

TABLE II
DISTRIBUTION OF MASS ONBOARD THE ROCKETS

	Mass	Payload	N°	Notes
Falcon 9 (reusable)	9 tonnes	9 tonnes	1	2 MAVs
Falcon 9 (expandable)	23 tonnes	21 tonnes	1	Habitat and consumables
Falcon Heavy (reusable)	51 tonnes	178 tonnes	4	All fuel set into 4 tanks
Kang	3 tonnes (2.6 payload)	16 tonnes	6	With AOCS' fuel

IX. ASSEMBLY PLAN

The assembly of the spacecraft will have five different sub-assemblies. Having different phases where the sub-assemblies will be assembled and attached to the main assembly is mandatory as it was stated in section VI that around 12 launches are necessary to deliver all required items, modules, and equipment to the orbit. The assembly plan order illustration is in Figure 9.

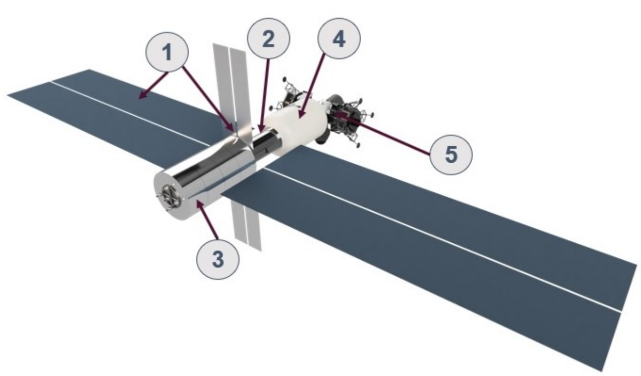


Fig. 9. The assembly plan order and sub-assembly illustration of the spacecraft [9]

The order of assembly is predetermined in a way that was deemed to be convenient regarding the assembly of the spacecraft. This will take into account the maximum chance of the operation being successful. The first sub-assembly is made of the power supply and communications systems. The rest of the sub-assemblies are built around the power supply module.

A "Dynamic assembly plan" was also designed to make the assembly plan modular, see Figure 10, and to avoid delays in the schedule in the case of off-nominal cases. These might be, for example, delays in the module's manufacturing, delivery, or malfunctions in the docking systems of the modules.

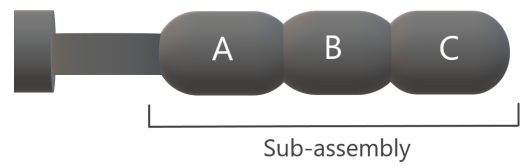


Fig. 10. Inside the sub-assembly system, order of modules A, B and C could be changed if it is necessary

Idea is that inside the sub-assembly, the order of the modules could be changed. If assembly would have three modules, A, B, and C with assembly order A-B-C. Flexibility in the dynamic assembly plan would enable the possibility of assembling in a different order, for example in B-A-C, thus allowing to stay in the schedule in case of off-nominal cases.

A. *First phase*

As the robotic arm is not going to be part of the mission and assembly plan, the first assembly phase is centered around bringing the power supply and the communication system modules. They're followed by the solar arrays.

B. *Second phase*

In the second phase, the service module is brought to orbit and attached to the first already into orbit subsystems. Adding a service module in the early phase is required as it is directly linked to power systems and communication systems.

C. *Third phase*

The third phase includes the sub-assembly of the fuel tanks and the five Asterix engines [9]. The engines and fuel tanks should be brought to orbit right after the second phase as they could be attached only after completing the first two phases. Assembling fuel tanks and engines in third phase enables to have spare time to solve cases like delays in deliveries or component malfunctions in the engine. Also, the fourth phase could be started even if the third phase would not be completed yet. This could be implemented due to the fact that third phase would be assembled towards the aft of the transfer vehicle.

D. *Fourth phase*

The fourth phase includes the habitats and the resources like consumables, tools, equipment, and spare parts. Here the inflatable habitats for the Mars-mission crew should be inflated and made operational.

E. Fifth phase

The main modules will be the two Mars landers/reentry vehicles. They make the final sub-assembly components before the crew arrives.

F. Sixth phase

The sixth and last phase of the assembly plan is to send the crew for the mission for Mars. All the sub-assemblies and modules should already be part of the main assembly and everything should be prepared and working in full function. Some small items could be brought in with the crew if necessary.

X. MAINTENANCE

By investigating previous long-duration space missions such as Skylab, Mir, and the International Space Station, one can get an idea of the maintenance required to facilitate human space flight. Even though flight hardware and systems are designed with an extreme emphasis on safety and reliability, things have and will break, and accidents may occur. It is therefore important to be able to mitigate failures and handle them once they arise.

A. NASA's approach to maintenance

For the ISS, NASA has categorized four types of on-orbit maintenance:

- *Preventative*: Cleaning, inspection, small part replacement;
- *Corrective*: Pre-planned maintenance to restore an item to its original condition
- *In-situ*: Repairs performed at the hardware site (spacewalks)
- *Contingency*: Maintenance to restore vital functions for crew and vehicle survival

Furthermore, NASA state in their paper regarding on-orbit maintenance that they've transitioned from a *"if it ain't broke, don't fix it"* mindset to a more preventative approach. This has been done to reduce equipment failure onboard and thus reduce hours spent fixing equipment. On the ISS, on average 2536 crew hours are spent on maintenance each year. This equates to almost a full year of work, assuming an 8-hour workday [34]. To enable ongoing maintenance, the ISS houses a suite of tools, both imperial and metric. As well as some standard parts and spare components for critical systems. In addition to this, NASA and its partners have a large library of parts and replacements, ready to ship to the station on cargo craft, should they be needed.

B. Low mass mission maintenance

For a low-mass mission to Mars, the approach taken differs from that of NASA and its partners. The space station can be quite flexible with its maintenance due to its proximity to earth, its size and resources dedicated to it. If a problem arises, alarms will alert the crew available 24/7 to solve the issue and coordinate a response. For a mission to Mars, the crew will have to be self-sufficient for almost 1000 days. The crew will therefore have to bring all the equipment, parts, and tools deemed necessary for mission success. What to bring and how much, is a complicated trade-off with the weight limit of the vehicle. In regards to equipment and spare parts, the crew will have to bring items necessary for the maintenance and repair of critical components such as oxygen regeneration and CO_2 removal, pressure containment, water, and power generation. If any of these systems fail during transit, the crew will without a doubt pass away.

During the vehicle's time in space, it will be subjected to large amounts of radiation. Modern electronics are susceptible to damage in this type of environment and can wreak havoc on flight computers. A common workaround for this is to have multiple redundant systems which all crosscheck with each other and correct themselves when an error occurs. To further ensure mission success and reduce development time it was aimed to use a high technology readiness level.

Several EVAs might be required to keep the spacecraft operational due to both scheduled maintenance and emergency situations. Together with the Mars Operations group, it was decided to implement a spacesuit that would fulfill both the requirements of an EVA suit and the exploration on the Red Planet. Further discussion of the H.E.R.O. (Hybrid Exploration and Repair Operation) suit is in [17].

XI. OFF-NOMINAL CASES

In the case of off-nominal cases, plans to solve them if they occur are crucial for the continuity and completion of the mission successfully or at least save the crew. There are two main areas where could be problems which are related to logistics. The first problem area is within the assembly and the second problem area is within communications.

A. Problems in Assembly

In assembly, docking fails and problems cases related to modules of the spacecraft are possible. For example malfunction of the docking port could be neglected with flipping the module or by dynamic assembly plan. This is allowed by the fact that the International Berthing Docking Mechanism described in Section VII-B is androgynous allowing it to be both the active and passive component in a classical docking sequence. Also for supply delays or manufacturing issues of the modules, the dynamic assembly plan fixes those issues that other modules could be sent first instead of the module which has issues. Correction with AOCS could be also used to

B. Problems with Communication

In case of malfunction in the main communication system where connection to Deep Space Network is lost, ESA's Estrack could be used as a spare communication system. In case of physical damages or breaks to the communication system, spare parts and equipment for the antennas and other components are brought onboard the spacecraft. Training the crew on how to use spare parts and components is provided before the flight.

If there will be a total blackout and all communication systems to earth are lost, a flight plan without any communication to earth will be followed. In the case voyage to Mars is still in its early stages, the immediate return voyage is executed. If communication is lost closer to Mars, a flight plan is to go to Mars orbit and then return to earth when the flight path will be as short and safe as possible. All unnecessary modules are abandoned. The crew will be trained for these scenarios and they will also keep track of their location, location of Mars, and location of Earth frequently during the mission so they could navigate in space without connection to the Earth and create a flight path which could enable their return to earth with the amount of fuel they have.

XII. Conclusions

A mission like this presents many challenges from a logistical point of view. Key systems such as the power supply, communication, thermal control, and docking systems have been developed with sufficient redundancy. The In-Space Assembly and the different assembly procedures were carefully considered to ensure a successful mission. Furthermore, all the sizing of the systems and planning of the assembly required a lot of coordination with the other sub-teams. The final part of the project included reviewing some possible off-nominal scenarios since a mission without any problems cannot be assumed. Potential problems with the assembly and the communication system have been considered and are presented with possible mitigation strategies.

XIII. Workload Breakdown

A. Giacomo Dal Toso

Giacomo worked on the communication systems of the transfer vehicle, on the docking method, on the estimation of the fuel necessary for the AOCS, and on the number of launches and mass allocation.

B. Juho Salminen

Juho worked on the assembly plan by designing sub-system assemblies and creating the assembly plan order, the robotic arm and also worked with the off-nominal cases by inventing some solutions for the cases.

C. Léon Messmer

Léon was the overall team representative. He worked on the Power system, notably the choice of power system, the sizing of the solar arrays, the batteries, and the power distribution system. Furthermore, he has worked on the In-space assembly, the robotic arm, AOCS, and related literature reviews and trade-off studies.

D. Simon Stenberg

Simon worked on the thermal control, such as system design, sizing, and estimation of the system. Moreover, he worked on the in-space maintenance and looked at the feasibility therein.

E. William Josefsson Rudberg

William worked on the power supply, with a focus on the power supply during Mars operations choosing the type of system, sizing of the fuel cells, and estimations of the amount of fuel needed. He was also involved in deciding the number of launches and which vehicles to be used.

XIV. Appendix A

A. Power

Fig. 11. Matrix Comparing Most Common Space Crafts Power Systems [13]

EPS Design Parameters	Solar Photovoltaic	Solar Thermal Dynamic	Radio-isotope	Nuclear Reactor	Fuel Cell
Power Range (kW)	0.2–300	5–300	0.2–10	5–300	0.2–50
Specific Power (W/kg)	25–200	9–15	5–20	2–40	275
Specific Cost ($/W)	800–3,000	1,000–2,000	16K–200K	400K–700K	Insufficient Data
Hardness – Natural Radiation – Nuclear Threat – Laser Threat – Pellets	 Low–Medium Medium Medium Low	 High High High Medium	 Very high Very high Very high Very high	 Very high Very high Very high Very high	 High High High Medium
Stability and Maneuverability	Low	Medium	High	High	High
Low-orbit Drag	High	High	Low	Medium (due to radiator)	Low
Degradation Over Life	Medium	Medium	Low	Low	Low
Storage Required for Solar Eclipse	Yes	Yes	No	No	No
Sensitivity to Sun Angle	Medium	High	None	None	None
Sensitivity to Spacecraft Shadowing	Low (with bypass diodes)	High	None	None	None
Obstruction of Spacecraft Viewing	High	High	Low	Medium (due to radiator)	None
Fuel Availability	Unlimited	Unlimited	Very low	Very low	Medium
Safety Analysis Reporting	Minimal	Minimal	Routine	Extensive	Routine
IR Signature	Low	Medium	Medium	High	Medium
Principal Applications	Earth-orbiting spacecraft	Interplanetary, Earth-orbiting spacecraft	Inter-planetary	Inter-planetary	Inter-planetary

1) Eclipse Time Derivation:

The following section of the appendix discribes how the time in eclipse of the transfer vehicle was determined, both around Earth and Mars. See in table III the main orbit information.

TABLE III
Main Data For Eclipse Calculation

Planet:	Earth	Mars
Radius(km)	6 371	3 389.5
Orbit Type	Circular	Elliptic
Apogee (km)	400+Re	77 028 000
Perigee (km)	400+Re	6 509 500
Orbit Period	90 min	72h

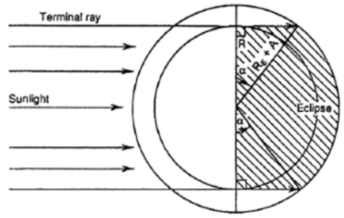
In order to determine the eclipse time of a circular orbit following formula was used based from [13] :

Applying these formula to the orbit in LEO:

$$\alpha = arccos\left(\frac{R_E}{R_E + 400}\right) = 19.79deg \tag{6}$$

$$TimeinEclipse = T_E = 90 * \frac{180 - 2 * \alpha}{360} = 35.1min \tag{7}$$

For the elliptical orbit the time derivation is a bit more complicated. By applying Kepler's Second law to the orbit, it is know that the area swept during a certain time remains constant. In order to find that value, the area of the entire ellipse is determined as followed:



$$\alpha = arccos\left(\frac{R_E}{R_E+h}\right)$$

$$Time\ in\ eclipse = T_e = Period\ of\ orbit * \frac{180-2\alpha}{360}$$

Fig. 12. Eclipse Time for Circular Obrits [13]

$$A = Apogee + Perigee \tag{8}$$

$$B = \sqrt{Apogee * Perigee} \tag{9}$$

$$Area = pi * A * B \tag{10}$$

$$DA = \frac{Area}{OrbitTime} \tag{11}$$

The next thing that has to be determined is the angle of the eclipse:

$$\alpha_M = arccos\left(\frac{R_M}{Perigee_M}\right) = 89.97deg \tag{12}$$

Based on this angle the arc area of the eclipse can be calculated as followed:

$$F(\theta) = \frac{AB}{2}\left[\theta - tan^{-1}\left(\frac{(b-a)*sin(2*\theta)}{b+a+(b-a)cos(2*\theta)}\right)\right] \tag{13}$$

$$Area_{eclipse} = F(\theta_1) - F(\theta_2) = 1\,744\,000\,000\,000km^2 \tag{14}$$

Finally, the time of the Space Craft in eclipse around Mars can be determined:

$$T_{eclipseMars} = 2.56minutes \tag{15}$$

2) Batteries: Figure 13 compares the three most common battery options for space flight.

Fig. 13. Matrix Comparing the main Battery Types [35]

Secondary battery types

Battery type	Cycle life	DoD %	Abusability	Memory effect?
NiCd	20000	10-20	High –can be fast charged/ discharged, left for long and still recover. Trickle charging poss.	Yes
NiMH	20000	40-60	Med – last a long time, can be left for long, but cannot be trickle charged.	Low
Lithium-Ion	3000	80	Low – Charge/discharge fairly fast, but must not overcharge or go below 3V when discharging, risk of explosion or fire.	No

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