Instituto Superior Técnico
Integrated Master in Aerospace Engineering
Aerospace Design

CAMOES

Commissioned Attempt at a Mars Odyssey and Exploration of Space

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Nomenclature

Greek symbols

α Absorptance.

Δ_v Change in velocity.

ϵ Emittance.

σ Stefan-Boltzmann constant.

Roman symbols

A Radiator's area.

a Planet's albedo coefficient.

A_a Albedo's projected area.

A_p Planet's projected area.

A_s Spacecraft's area exposed to Sun.

C_3 Characteristic energy.

C_r Consumption rate.

d Distance between the Sun and the spacecraft.

E Total energy.

E_0/N Signal-to-noise ratio

F Visibility factor.

g Gravitational acceleration.

h Altitude.

I_{sp} Specific impulse.

J_a Albedo's total energy flux.

J_p Planetary radiation intensity reaching the spacecraft.

J_s Solar energy flux.

L_d Life degradation.

m_d Dry mass.

m_f Fuel mass.

$M_{reactants}$ Fuel required by cells.

p Average planetary emission.

P_s Sun's total power output.

Q_{albedo} Albedo's heat.

$Q_{external}$ External heat.

$Q_{internal}$ Internal heat.

Q_{planet} Planet's heat.

$Q_{radiated}$ Radiated heat.

$Q_{radiator}$ Radiator's heat.

Q_{solar} Solar heat.

$Q_{surface}$ Surface's heat.

R Planet's radius.

r Radius.

$R_{\oplus}$ Earth's radius.

$R_{altitude}$ Spacecraft's altitude from planet's center.

R_{rad} Planet's effective radiating surface radius.

T Radiator's working temperature.

V Velocity.

V_{∞} Hyperbolic excess velocity.

Subscripts

$()_{arr}$ Arrival.

$()_{BOL}$ Beginning of life.

$()_{dep}$ Departure.

$()_{eff}$ Effective.

$()_{EOL}$ End of life.

$()_{ext}$ External.

$()_{in}$ Internal.

$()_{radiator}$ Radiator.

$()_{rad}$ Radiated.

$()_{surface}$ Surface.

Acronyms and Abbreviations

ACT Active Thermal Control.

AHP Analytic Hierarchy Process.

ALS Advanced Life Support.

approx. Approximation.

APS Active Pixel Sensor.

ARED Advanced Resistive Exercise Device.

Arg. Argument.

AU Astronomical Units.

AutoNav Autonomous Optical Navigation.

Avg. Average.

BER Bit Error Rate.

BOL Beginning of Life.

BPSK Binary Phase Shift Keying.

BWG Beam Waveguide Antenna.

C&DH Command and Data Handling.

CAD Computer-aided Design.

CAMOES Commissioned Attempt at a Mars Odyssey and Exploration of Space.

CERs Cost Estimation Relationships.

CHCS Crew Health Care System.

CM Crew members.

CMS Countermeasures System.

Dep. Departure.

DPC Daily Planning Conference.

DS1 Deep Space 1.

DSA Deep Space Antennas.

DSH Deep Space Habitat.

DSN Deep Space Network.

Ecc. Eccentricity.

ECLSS Environmental Control and Life Support System.

EHS Environmental Health System.

EOL End of Life.

EPO Earth Parking Orbit.

EPS Electrical Power System.

ESA European Space Agency.

ESDM Environmental Control and Life Support System Design Model.

FFD Functional Flow Diagram.

Freq. Frequency.

FY Fiscal Year.

GCMS Gas Chromatograph and Mass Spectrometer.

GMAT General Mission Analysis Tool.

GNC Guidance Navigation and Control.

HD High Definition.

HGA High Gain Antenna.

HMS Health Maintenance System.

I/O Input/Output.

IMU Inertial Measurements Unit.

Inc. Inclination.

IR Infrared.

IRED Interim Resistive Exercise Device.

ISS International Space Station.

JPL Jet Propulsion Laboratory.

LED Light-Emitting Diode.

LEO Low Earth Orbit.

LOS Line-of-sight.

MANS Microcosm Autonomous Navigation System.

MICAS Miniature Integrated Camera and Spectrometer.

MIT Massachusetts Institute of Technology.

MMSEV Multi-Mission Space Exploration Vehicle.

MOI Mars Orbit Insertion.

MPLM Multi-Purpose Logistics Module.

MPO Mars Parking Orbit.

NA Non-available.

NASA National Aeronautics and Space Administration.

OGS Oxygen Generation System.

PDU Power Distribution Unit.

PFL Pumped Fluid Loops.

PICA Phenolic Impregnated Carbon Ablator.

RAAN Right Ascension of the Ascending Node.

RCS Reaction Control System.

S/C Spacecraft.

SLS Space Launch System.

SMA Semi-major axis.

SOA State of the Art.

SpaceX Space Exploration Technologies Corporation.

SWIR Shortwave Infrared.

TCM Transfer Correction Maneuver.

TCS Thermal Control System.

TDRS Tracking and Data Relay Satellite.

TEI Trans-Earth Injection.

TMI Trans-Mars Injection.

ToF Time of Flight.

TRL Technology Readiness Level.

TT&C Telemetry Tracking and Communications.

UV Ultraviolet.

VDMLI Variable Density Multiple Layer Insulation.

VPCAR Vapor Phase Catalytic Ammonia Removal.

1 Introduction

The allure of exploring the red planet has never been greater and more readily achievable than in the coming decades, boosted by the growing demand for commercial space exploration and the rise of public and private partnerships in the space industry.

Mars has been an object of fascination for humanity since ancient times and as been identified as the next step in human space exploration. Up until this day, there have been numerous notable milestones, from the early telescopic observations of Galileo and Huygens, to the modern era of spacecraft-based exploration. The precedents set by the initial Apollo moon flyby missions, prior to the first Moon landing, illustrate the value of performing a Mars orbital mission, reducing future risks as well as igniting the public's excitement for a new generation of space exploration. Now, in the 21st century, a new era of spaceflight has been set upon mankind, as it gets its first chance to become a multiplanetary species.

In order to achieve this, an end-to-end Mars orbital mission for a crew of four will be designed. This mission is to be as safe, simple and cost effective as possible. Due to its nature, all aspects from launch to the Earth re-entry need to be considered. The mission's launch date is May 2033 and will take 882 days in total. The crew will have to endure launch, in orbit assembly, Trans-Mars Injection, a Martian orbit phase, the return to Earth and Earth re-entry. During the mission, the safety and well-being of the crew is therefore our number one priority. In order to ensure the crew members' safety, all systems must be designed with safety and reliability in mind. Also, simplicity is taken into consideration since it influences cost and reliability, factors that need to be addressed. To achieve this, off-the-shelf components are preferred. Finally, the design must comply with all the mission requirements.

2 Project Overview

The CAMOES mission is simple in nature but very extensive. Therefore, the Concept of Operations was schematized and a Functional Flow Diagram was made. A preliminary systems analysis was also performed using quality function deployment to rank the engineering specifications needed to meet the requirements of this mission.

Table 1: Engineering specifications needed to accomplish the mission

Rank	Hows
1	Hardware reliability
2	Vehicle dry mass
3	Number of launches
4	Radiation/Thermal shielding
5	Habitable volume
6	Re-entry velocity

During the entire development, the CAMOES team followed four principles: simplicity, safety, low cost and feasibility. Furthermore, appropriate measures were taken to make CAMOES adhere to the sustainable requirements of today.

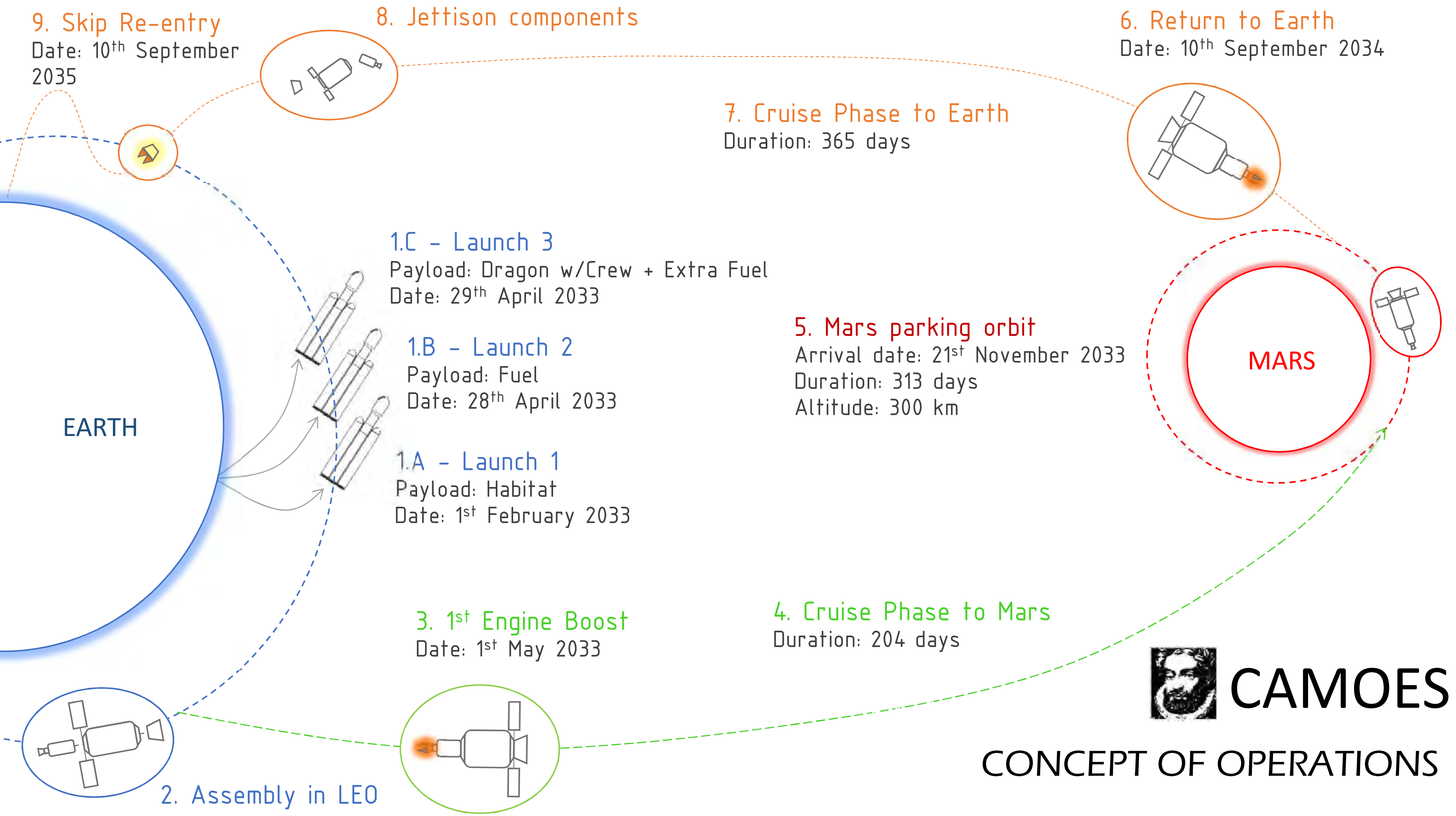
The system consists of modified versions of the Bigelow Aerospace B-330 and the SpaceX Dragon, sent to space in 3 Falcon Heavy launchers. Both modules have already been tested in their basic configuration and are currently under further development.

The total mission duration from TMI is 882 days which was reached with a trajectory that represents a compromise between mission duration and required Δv. A section was dedicated to the crew's safety and health. Keeping the crew in good spirits is of great importance, since the crew should perform experiments and document as much as possible during their journey. Thus, the endeavor will result in the maximum scientific benefit for future missions and possible spin-offs.

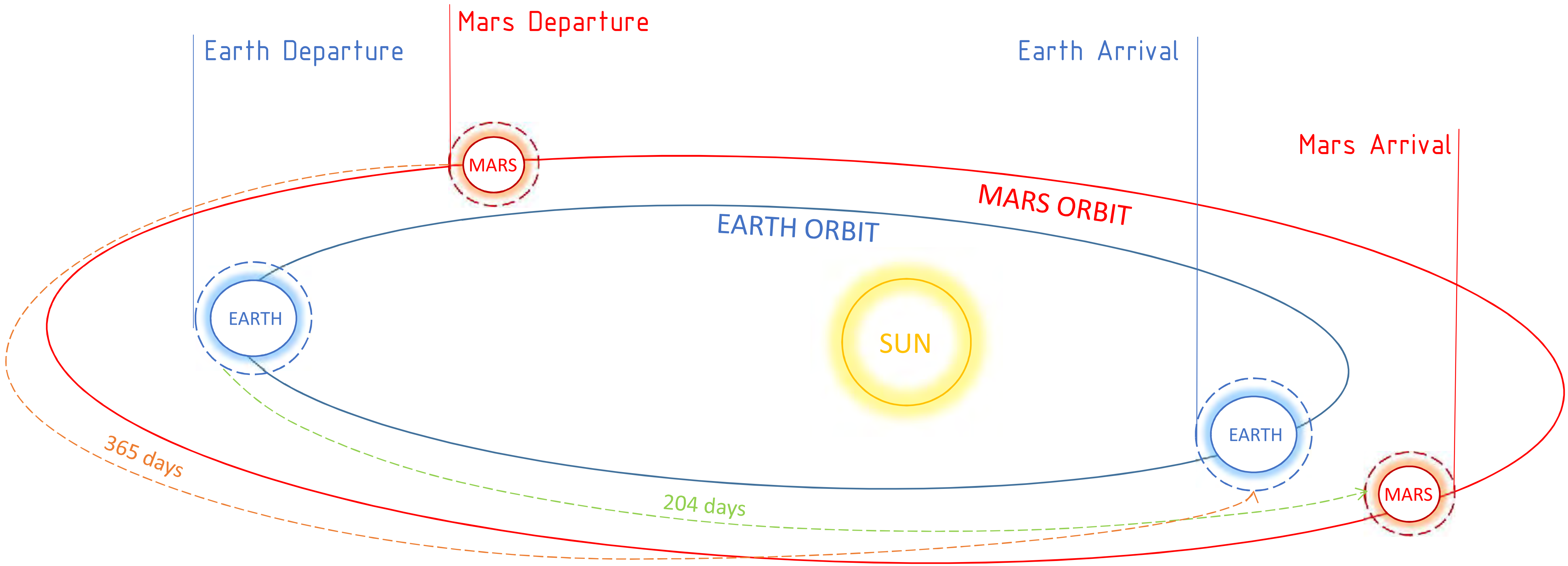
The total cost of such a mission is estimated using current prices, heritage data and cost models. Finally, a simple risk analysis was performed focused not only the dangers stemming from technological aspects but also programmatic issues are presented and how they might be mitigated.

2.1 Concept of Operations

The concept of operations is presented below.



9. Skip Re-entry
Date: 10th September 2035
8. Jettison components
6. Return to Earth
Date: 10th September 2034
7. Cruise Phase to Earth
Duration: 365 days
1.C - Launch 3
Payload: Dragon w/Crew + Extra Fuel
Date: 29th April 2033
1.B - Launch 2
Payload: Fuel
Date: 28th April 2033
1.A - Launch 1
Payload: Habitat
Date: 1st February 2033
EARTH
5. Mars parking orbit
Arrival date: 21st November 2033
Duration: 313 days
Altitude: 300 km
MARS
3. 1st Engine Boost
Date: 1st May 2033
4. Cruise Phase to Mars
Duration: 204 days
2. Assembly in LEO
CAMOES
CONCEPT OF OPERATIONS



CAMOES

CONCEPT OF OPERATIONS

2.2 Functional Flow Diagram

The Functional Flow Diagram (FFD), presented in Figure 26, appendix A, divides the mission in nine phases, each having multiple functions that are required to occur in a specific order. As it can be seen, each phase is identified along with the corresponding functions. During a critical phase, the functions may be further sub divided. These functions in turn, determine the required procedures and systems, which means that if these functions are met from start to end, the mission will be a success.

2.3 Long and Short Term Strategies

The sustainability strategies are composed of two types: long term and short term. The long term ones aim at completing sustainability goals which have an effect on the environment over a longer period of time. Therefore, these strategies are applied to CAMOES, so that future Mars missions can benefit from them. The long term strategies are stated below:

- The scientific payload will provide data for future studies;
- Develop reusable components that cause less damage to the environment;
- Minimize environmental impact during the development phase, by using off-the-shelf components;
- Develop an approach with the intent of decreasing the use of fossil fuels and reducing green house gas emissions;

Short term strategies, on the other hand, are sustainable approaches that have a direct effect on the environment. Listed below are the short term strategies which are going to be used during the mission's design. The short term strategies are stated below:

- Solar panels will generate all the needed energy;
- The water will be recycled during the trip;
- The use of toxic materials will be minimized in the development and final design;
- All jettisoned spacecraft components will be sent into a graveyard orbit to reduce space debris;
- The jettisoned space-bus at the end of the mission will be used as a deep space measuring satellite;
- An energy efficient orbit is used;
- Requirements are analyzed to reduce the use of resources;
- Left-over components, like a re-entry capsule, are put on display in a history museum.

3 Trajectory Design

The purpose of the trajectory design is to find a feasible trajectory both from Earth to Mars and from Mars to Earth. The opportunities for such trajectories must be investigated, which is done by solving the Lambert's problem, considering astrodynamics and interplanetary spaceflight.

In order to compare various launch opportunities, a trade study of exiting launch windows between the years of 2024 and 2036 was made. This time frame was chosen to account for the time needed to execute all the steps of the mission planning.

For the validation and verification of the final trajectory and for the prediction/estimation of certain factors, *GMAT* was used. *GMAT* is an open-source space mission analysis tool provided by NASA that enables the simulation of gravitational forces of all celestial bodies in the solar system.

3.1 Interplanetary Trajectory Trade-study

For most unmanned missions, it is desirable to minimize the total mission Δv, either to lower the cost or to maximize the science return. For those cases, the choice always falls on the trajectory with the lowest Δv, selected from each opportunity (or 'porkchop'). However, for human spaceflight endeavors a mission designer will want to lower the mission duration, most of the times, at the cost of extra Δv.

Therefore, before the beginning of the trade study, the constraints of the mission feasibility must be defined based on the characteristic energy $C_3 \left(= V_\infty^2\right)$:

- C_{3dep}: Determines the launch feasibility. In a report from Jet Propulsion Laboratory (JPL), a feasible launch assumed that C_{3dep} is less than 25 km^2/s^2 [1]. Considering envisioned advances in technology, we use a C_{3dep} of 30 km^2/s^2 as a launch feasibility criterion.

- C_{3arr}: For direct entry or orbit insertion at arrival, propulsive capture generally requires a minimum arrival velocity, while aerocapture tolerates higher arrival velocities. Given an 8 km/s limit on Mars entry velocity, C_{3arr} at Mars up to 40 km^2/s^2 is acceptable. On the other hand, from an analysis that was carried out during the Draper/MIT CE&R project we know that Earth entry velocity up to 13 km/s is tolerable from a gravity load and heating perspective, which would rule out all trajectories with C_{3arr} at Earth over 45 km^2/s^2.[2]

$$C_{3dep} \leq 30 \text{ [km}^2\text{/s}^2\text{]}$$
$$C_{3arr} \leq \begin{cases} 40 \text{ [km}^2\text{/s}^2\text{]} & \text{(Mars arrival)} \\ 45 \text{ [km}^2\text{/s}^2\text{]} & \text{(Earth arrival)} \end{cases}$$

Based on these constrains, an analyze of the feasible launch windows was made, through a creation of a porkchop. Figure 1 shows an example of the interplanetary transfer from Earth to Mars with the contours of the total V_∞, i.e. the $(V_{\infty Departure} + V_{\infty Arrival})$ in km/s.

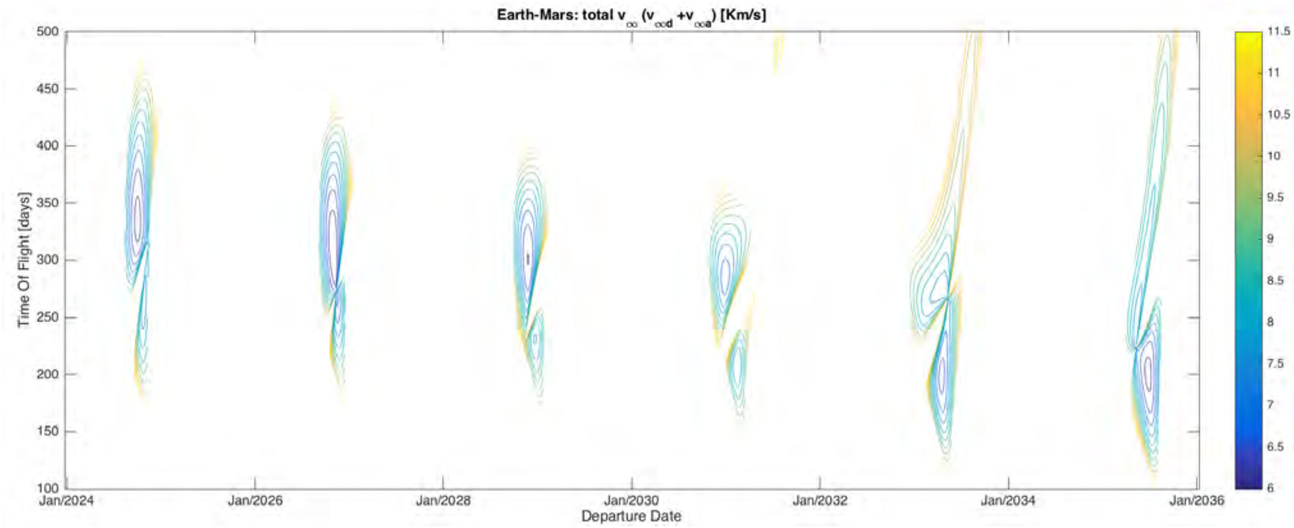


Figure 1: Porkchop 2024-2036 Earth-Mars, V_∞ contours

Studying the porkchop graphics for both transfers (Earth to Mars and Mars to Earth), the best trajectory opportunities throughout the years 2024-2036 were computed and can be seen in Table 2.

Table 2: Mission profile

Earth to Mars				**Mars to Earth**			
Departure	**Arrival**	**ToF [days]**	**V_∞ total [km/s]**	**Departure**	**Arrival**	**ToF [days]**	**V_∞ total [km/s]**
02/10/24	01/09/25	334	5.8029	03/08/26	19/06/27	320	5.7619
26/10/24	15/07/25	262	7.2161	11/08/26	13/05/27	275	7.0579
01/11/26	07/09/27	310	5.6128	06/09/28	11/08/29	339	6.9050
13/11/26	11/08/27	271	6.1621	11/09/28	03/06/29	265	8.1607
24/11/28	21/09/29	301	5.9867	08/11/30	23/09/31	319	7.7748
21/12/28	09/08/29	231	7.4473	10/11/30	24/09/31	318	7.8240
31/10/30	30/08/31	303	8.9350	30/09/32	19/09/33	354	7.7181
29/12/30	10/10/31	285	6.7402	31/12/32	02/09/33	245	6.9254
01/01/31	13/10/31	285	6.6566	16/02/33	20/09/33	216	6.2306
13/02/31	09/09/31	208	7.5662	01/04/33	01/10/33	183	7.1837
01/03/31	26/09/31	209	7.9489	30/09/34	30/09/35	365	9.0894
31/12/32	27/09/33	270	8.8084	31/12/34	06/11/35	310	7.9155
28/02/33	21/11/33	266	7.3379	31/03/35	14/11/35	228	6.5729
16/04/33	01/11/33	199	6.1215	14/05/35	26/11/35	196	6.2816
01/05/33	21/11/33	204	6.4782	01/07/35	28/12/35	180	7.8795

With the available trajectory data for both cases, a single chart was created, which comprises the complete itinerary of the mission.

To do so, in the first place, V_∞ had to be converted into Δv, which is the Δv required for all the main maneuvers: TMI, MOI and TEI. This conversion process is highly dependent on the parking orbits around the planets, as it will be explained in the Section 3.2, where the parking orbits to be used will also be described .

The most promising itineraries for our mission are shown in Table 3. These results were obtained assuming an Earth's circular parking orbit with an altitude of 1500 km and a Mars' elliptical capture orbit with a periapsis of 300 km and apoapsis of 33764.36 km.

Table 3: Itinerary of the entire mission

					from porkchop							
Trajectory Itinerary		Departure	Arrival	ToF	$V_{\infty\,dep}$	$V_{\infty\,arr}$	$C3_{dep}$	Δv_{TMI} / Δv_{TEI}	$C3_{arr}$	Δv_{MOI} -	TOF_{total} (days)	Δv_{tot} (Km/s)
Optimal TOF	Earth-Mars	01/03/31	26/09/31	209	4.2365	3.7773	17.9479	3.8021	14.2677	1.5280	438	9.3145
	Mars-Earth	01/10/31	12/05/32	224	7.0968	12.3507	50.3641	3.9844	152.5387			
Optimal ΔV	Earth-Mars	02/10/24	01/09/25	334	3.3629	2.4532	11.3090	3.4936	6.0183	0.8120	982	5.2266
	Mars-Earth	30/07/26	11/06/27	316	2.6847	3.0219	7.2078	0.9210	9.1320			
#1	Earth-Mars	01/05/33	21/11/33	204	3.2446	3.4095	10.5276	3.4567	11.6249	1.3081	927	5.9463
	Mars-Earth	31/03/35	14/11/35	228	3.1856	3.0923	10.1478	1.1815	9.5624			
#2	Earth-Mars	01/05/33	21/11/33	204	3.2446	3.4095	10.5276	3.4567	11.6249	1.3081	882	6.7058
	Mars-Earth	30/09/34	30/09/35	365	4.4152	4.5310	19.4938	1.9411	20.5302			
#3	Earth-Mars	01/07/33	01/07/34	365	4.4779	5.1189	20.0518	3.8980	26.2028	2.4359	866	7.5154
	Mars-Earth	31/03/35	14/11/35	228	3.1856	3.0923	10.1478	1.1815	9.5624			
#4	Earth-Mars	01/07/33	01/07/34	365	4.4779	5.1189	20.0518	3.8980	26.2028	2.4359	821	8.2749
	Mars-Earth	30/09/34	30/09/35	365	4.4152	4.5310	19.4938	1.9411	20.5302			

3.1.1 Trajectory Selected

As it was stated before, the lowest mission duration is desired, even if it is accomplish at the cost of extra Δv. Analyzing the results in Table 3, it can be concluded that the second option is the one with the better relation between ToF and Δv.

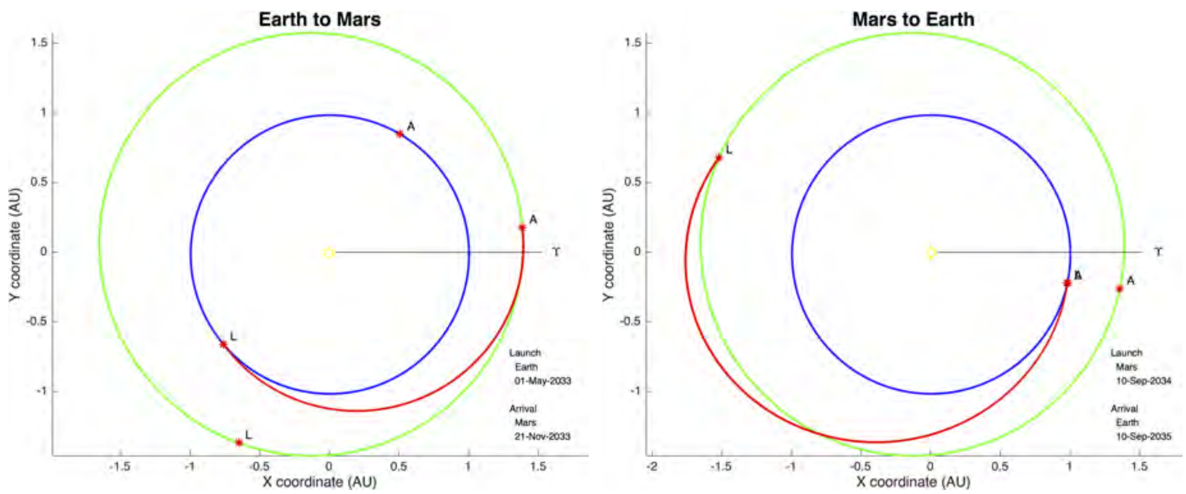


Figure 2: Interplanetary mission trajectory

Table 4: Orbital elements of the Interplanetary trajectories

	SMA [AU]	Ecc.	Inc.[^{o}]	Arg. Perigee [^{o}]	RAAN [^{o}]	True Anomaly [^{o}]
Earth-Mars	1.19222	0.173	2.251	148.85	40.57	31.06
Mars-Earth	1.37213	0.271	4.46	7.272	6.19	150.59

3.1.2 Flyby Trajectories Opportunities

An alternative to the orbit mentioned in the Section 3.1.1, would be a flyby Mars mission. The advantage of such type of orbit would be a lower Δv, when using a free return option, or the shorter duration of the mission, when using powered flyby maneuver. The disadvantage, however, is that the scientific missions meant to be performed on Mars wouldn't be possible, since the spacecraft would only be a few hours close to the planet due to the high velocities involved in this maneuver.

A quick analysis of flyby opportunities between the years of 2024 to 2036 can be seen in Figure 3. These results were computed under the assumption that injection occurs at a 200 km altitude. As it is possible to see in Table 5, the duration can be highly reduced when performing a powered flyby maneuver, instead of a free-return flyby.

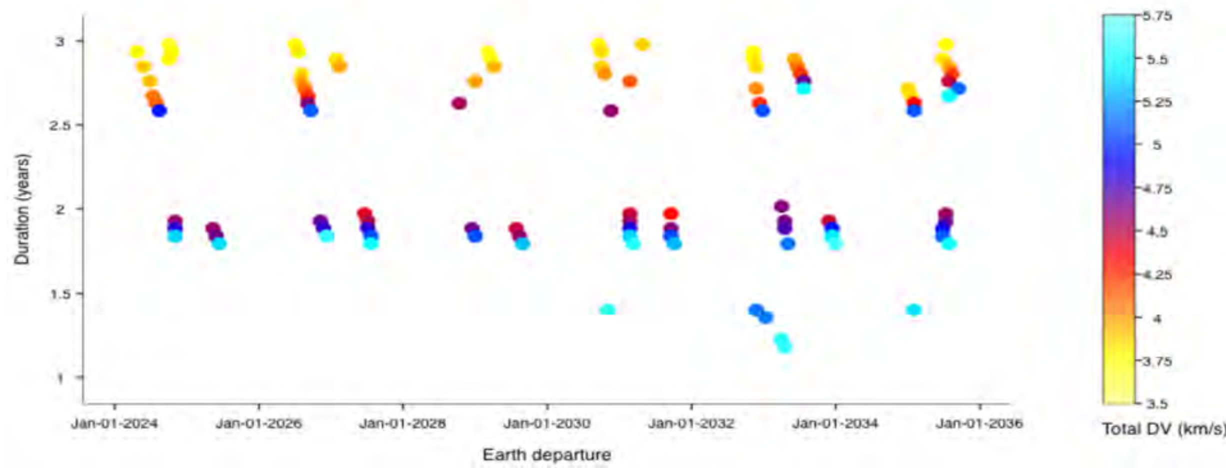


Figure 3: Flyby opportunities [3]

Table 5: Best flyby trajectory opportunities

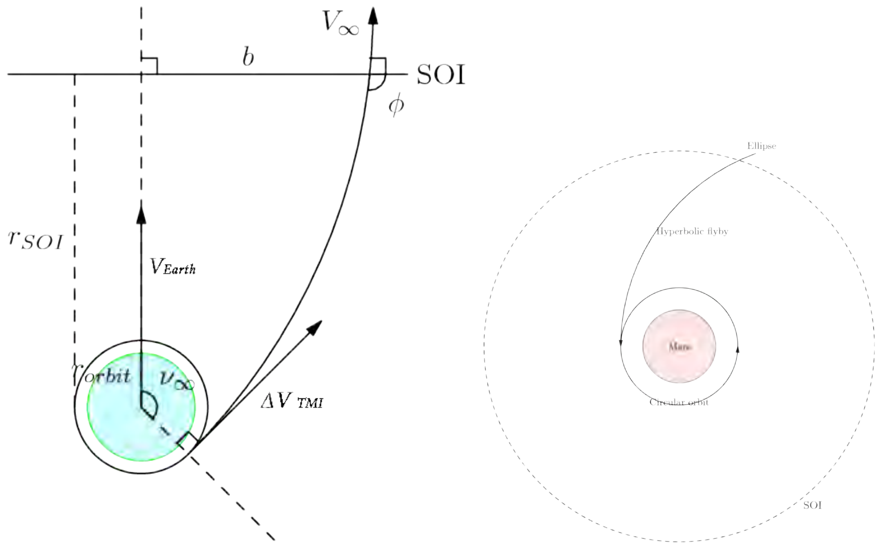
Trajectory	Earth Dep.	Flyby	Earth Return	ToF [years]	Δv total [km/s]
Free Return	Oct-01-2024	Sep-02-2025	Sep-24-2027	2.98	3.69
	Mar-05-2029	Apr-09-2030	Feb-10-2032	2.93	3.71
	Nov-02-2024	Mar-26-2025	Oct-07-2026	1.93	3.57
	Jun-20-2027	Jan-16-2029	Jun-09-2029	1.97	4.41
	Aug-07-2027	Jan-16-2029	May-08-2029	1.75	6
Flyby Burn	Jun-04-2033	Oct-10-2033	Jun-23-2034	1.05	6.65
	Mar-11-2031	Sep-03-2031	Apr-30-2032	1.14	6.38
	Dec-31-2028	Aug-28-2029	Apr-09-2030	1.27	6.98
	Dec-26-2026	Apr-17-2027	Aug-25-2028	1.66	6.67
	Jun-14-2025	Dec-10-2026	Apr-01-2027	1.80	5.39

Comparing these trajectories with the one chosen before, it can be seen that for about the same Δv we could reduce the ToF from 882 days to 384 days. This reduction, while limiting the research that could be made during the proximity of Mars, would be highly beneficial for the safety and health of the crew members.

3.2 Hyperbolic Injections Trade-study

In order to escape the gravitational force of a planet, the spacecraft must travel in an hyperbolic trajectory relatively to the planet, arriving at its sphere of influence with a relative velocity V_∞ (hyperbolic excess velocity) greater than zero. The same idea is applied when capture by a planet's gravitational field is desired, based on the relative velocity V_∞ with which the spacecraft arrives at the planet.

These V_∞ were already calculated in Section 3.1, so now the study of the parking orbit is necessary, to determine the Δv (TMI, MOI and TEI) required for the mission.



(a) TMI (b) MOI

Figure 4: Hyperbolic trajectory [4]

3.2.1 Parking Orbits

- Earth Parking Orbit:

  Firstly, the choice was limited to a circular orbit, to simplify the rendezvous and docking maneuvers that will have to be performed.

  A simple study of the mechanical equations shows that the higher the circular parking orbit, the lower the energy necessary for the escape hyperbolic trajectory. Therefore, an orbit as high as possible was preferred. On the other hand, the launcher capacity limits our option, since the higher it needs to launch, the lower payload it can take. In the end, a parking LEO with 1500 km altitude was chosen.

- Mars Parking Orbit:

  From the several different alternatives for a MPO, we narrowed them to four possible orbits that will be studied here. All of these four options have different interesting properties. The 1st orbit was chosen with the intention of having a close proximity to the planet, for research purposes. An altitude of 300 km was decided for this case, which is a safe altitude to avoid the drag effect of the atmosphere. The 2nd is a circular areosynchronous orbit, which could be very useful to research very specific locations on Mars. The 3rd and 4th orbits are elliptical with 300 km altitude at periapsis and a n-period around Mars equivalent to the sideral rotation of the planet.

Assuming the Δv_∞ at Mars from the chosen trajectory in Section 3.1 ($\Delta v_{\infty\ arr}$ = 3.4095 km and $\Delta v_{\infty\ dep}$ = 4.4152 km) we computed the converted Δv for the TMI and TEI maneuvers:

Table 6: MPO simple trade study

	Periapsis-apoapsis [km]	Δv_{MOI} **[km/s]**	Δv_{TEI} **[km/s]**
Circular	300 - 300	2.4951	3.1281
Circular aerosynchronous orbit	17032 - 17032	2.5292	3.4189
Elliptical (24 hour Martian period)	300 - 33764	1.3081	1.9411
Elliptical (48 hour Martian period)	300 - 57765	1.2243	1.8573

From the Table 6 it is clear that using an elliptical orbit instead of the circular ones saves a considerable amount of fuel, due to the differences of up to 2.5 km/s of $\Delta v_{MOI+TEI}$. Therefore, an elliptical orbit with 24 hour martian period is chosen. Although the second elliptical orbit requires slightly less Δv, the 24 hour period orbit was preferred because, when the time of returning to Earth comes, this will grant one opportunity per day to perform the TEI, while the other orbit would only guarantee one opportunity every two days.

- Selected parking orbits:

Table 7: Parking orbits parameters

Earth Parking Orbit		**Mars Parking Orbit**	
Periapsis altitude	1500 km	Periapsis altitude	300 km
Apoapsis altitude	1500 km	Apoapsis altitude	33764.36 km
Period	1.933 hours	Period	24.6229 hours
Semi-major axis	7878 km	Semi-major axis	20429 km
Eccentricity	0	Eccentricity	0.819
Inclination	28.5^{o}	Inclination	89.649^{o}

3.2.2 Aeroassist Alternatives

To perform the MOI it was assumed the use of propellant burn, however, there are other alternatives, such as aerocapture or aerobraking.

Aerocapture technology was evaluated for use in manned Mars missions and it was found it offers significant mass benefits. Nonetheless, for the success of this maneuver, the trajectory must be constrained to avoid excessive deceleration loads on the crew. Although there are similar constraints on trajectories for unmanned missions, the human limits are typically more stringent, specially in light of the effects of prolonged microgravity in acceleration tolerances.

Aerobraking is another aeroassist maneuver that also uses the celestial body's atmosphere to slow the aircraft down into the desired orbit. While aerocapture only uses one pass through the atmosphere to reduce its velocity, aerobraking needs from 100 to 400 passes to achieve the desired velocity reduction.

Table 8: Aeroassist maneuvers comparison

Aerocapture	**Aerobraking**
Rapid process (hours to days)	Gradual process (weeks to months)
Descent into a relatively dense mid-atmosphere	Descent into sparse outer atmosphere
Requires a heavy heat shield due to rapid deceleration resulting in high g-forces	Small reductions in spacecraft velocity per pass thus no additional mass for a heat shield is necessary

One of the main advantages of using an aerocapture technique over an aerobraking technique, is that it enables mission concepts for human spaceflight, due to the rapid process of transitioning to the desired orbit. On the other hand, aerocapture requires the use of a heavy heat shield which was not meant to be included in the spacecraft model, thus excluding this option.

3.2.3 Phobos or Deimos Gravity Assist

There is a possibility to use the martian moons to perform a maneuver that might require less Δv to put the spacecraft orbiting around Mars.

Both Phobos and Deimos are essentially in the equatorial plane of Mars, with nearly circular orbits at 9378 km and 23459 km, respectively. Phobos travels around the planet three times a day, zipping across the martian sky approximately once every four hours. Deimos, on the contrary, takes about 30 hours, a little over a martian day, to complete a revolution.

Through a simulation with the *GMAT*, it was concluded that this case is an actual possibility, since it was predicted that, at the time of arrival to Mars, the velocity of the moon would be almost in the opposing direction, decreasing the relative velocity of the spacecraft.

Since using this method would probably lead to a different parking orbit than the predicted one, because it would need a different inclination to encounter the moon, it was also discarded.

3.3 Transfer Correction Maneuver

During the interplanetary flight, the spacecraft trajectory will suffer slight adjustments to make sure it arrives at the targeted location. These will be divided into four trajectory corrections maneuvers along the transfer from Earth to Mars.

The first, TCM-1, is planned to occur about 10 days after the beginning of the TMI and will be the largest correction, to remove the launch vehicle injection errors. The TCM-2 will occur after half of the TMI, with the intent of removing the TCM-1 execution errors and others that might be accumulated during flight. The final ones, TCM-3 and TCM-4, will happen closer to the planet Mars and are meant to direct the spacecraft to its final target point for Mars orbit insertion. The last TCM will only be done 10 days prior to MOI.

Accurate predictions for the required TCM burns are very hard to obtain since they are totally dependent on the trajectory errors input throughout the mission.

For insurance purposes, an estimated value is obtained based on the simulation ran with *GMAT* software. It was chosen a value 10 times bigger than the result obtained with *GMAT*, to allow a good margin for errors. This estimation is about 79.84 m/s for the total TCMs from Earth to Mars.

The same process of TCMs is used for the return trajectory to Earth. These will be TCM-5 to TCM-8 and, once again based on the *GMAT* simulation, an estimated value of 66.00 m/s for the total TCMs was determined.

Table 9: TCMs study

	GMAT Δv [m/s]	**Estimated for needed Δv [m/s]**
$\sum_1^4$TCM-i	7.984	79.84
$\sum_5^8$TCM-i	6.600	66.00
$\sum$ TCMs	14.584	145.84

4 Re-entry Design

This is not only the last stage of the mission but also the most critical one, since a very precise trajectory and entry angle are required, to avoid missing the entry or crashing. Also, during the process of entering Earth's atmospheric environment, significant energies are generated, namely thermal and kinetic energies. Finally, the total heat flux is also a concern, as long duration reentries can lead to damage in the internal components that are not designed for high temperature operation.

The re-entry speed in this case would be about 12 km/s, nearly 20 times faster than a supersonic jet can travel. This means that during the re-entry the spacecraft's heat shield must protect both the vehicle and the crew from external temperatures that can reach up to 2760 K.

To protect the capsule through the extreme temperatures of re-entry when it returns to Earth, the Dragon capsule uses a version of NASA's Phenolic Impregnated Carbon Ablator heat shield, the PICA-X, which SpaceX developed alongside NASA. According to SpaceX, "It can potentially be used hundreds of times for Earth orbit re-entry with only minor degradation each time [. . .] and can even withstand the much higher heat of a moon or Mars velocity re-entry."

4.1 Direct Re-entry

During re-entry the amount of g forces to be sustain by the crew must be below 8 g's, with less than 6 g's being the desirable case. However there's another factor that requires even lower g's during re-entry, the Dragon capsule that has a limit of 3.5 g's that should not be surpassed in order to maintain the propulsion system functional.

There are two types of entry which influence the design of manned vehicles for hypersonic reentries into the Earth's atmosphere: ballistic and skip reentries.

4.1.1 Ballistic Re-entry

For the ballistic re-entry, the maximum deceleration g-force acting on the capsule would be around 42.5 g's, over 12 times higher than the maximum acceptable limit of 3.5 g's. Temperature generation would not be a limiting factor in this scenario, as the Dragon capsule has already been tested with temperatures of 2770 K at a velocity of 12.9 km/s.

The acceptable re-entry velocity to produce 3.5 g's happens at 3.45 km/s, meaning that a burn of Δv = 8.55 km/s would have to be performed, which is quite a lot for current capabilities.

4.1.2 Skip Re-entry

The skip re-entry model reduces the g-forces felt by extending the total duration of this maneuver. The deceleration forces will be felt for a longer time but in a lower magnitude. Table 10 presents the g-forces that would result from this trajectory.

Table 10: G-forces for skip re-entry, from the Air Force Institute of Technology

Velocity [km/s]	**g forces 1st skip**	**g forces 2nd skip**
12.2	3.22	N/A
12.1	3.09	N/A
12	3.7	N/A
11.9	4.07	3.06
11.8	5.43	3.72

The N/A values in Table 10 mean that the capsule continues past Earth after one skip, entering into a highly elliptical orbit (aerocapture) or missing the Earth's orbit entirely. Aerocapture would extend the re-entry timeline, placing further straining on life support and crew, being therefore avoided. Note these scenarios do not account for pilot input: with major pilot input, a skip re-entry of 12 km/s would be possible.

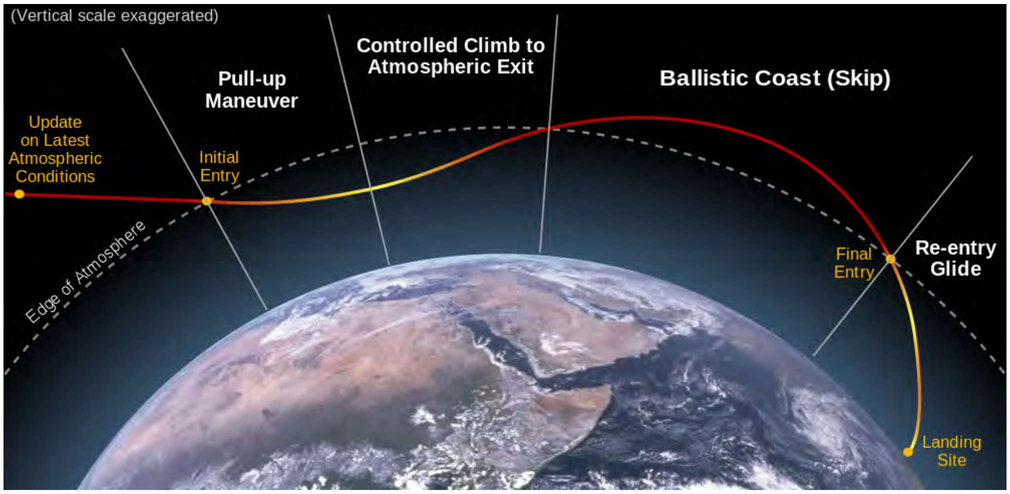


Figure 5: Skip re-entry trajectory [5]

4.2 Selected Re-entry Method

In a skip re-entry trajectory, flying out of the atmosphere has a large effect on cooling the vehicle down and getting it ready for the next dip with lower kinetic energy. Therefore, a skip re-entry was considered as the best option in a trade-off decision for re-entry, due to its beneficial effects and not very complicated nature. Figure 5 represents a good visual expression of the skip entry trajectory. In order to match the limit of 3.5 g's of the Dragon capsule, a small increment of the velocity will be performed, raising it from 11.92 km/s to 12.1 km/s. This maneuver will result in the reduction of g's felt by the crew and capsule from 4.07 to 3.09. In order to avoid missing the 2^{nd} skipping re-entry in this process, the pilot must perform some adjustments. By the end of each skip, the pilot would roll the capsule in such a way that the lift vector would cause the capsule to reenter, performing another skip, until the velocity of the capsule is able to safely approach the Earth's surface.

4.3 Flyby Moon

As an alternative to a direct entry, we could use the lunar gravity assist to either slowdown the velocity of the spacecraft or to capture it into an orbit around the Earth, as seen in Figure 6.

If the spacecraft was placed in a captured orbit, a separated mission to that orbit would be required, in order to retrieve the crew members and bring them back to the Earth. That retrieve mission could be made using the International Space Station as an intermediary.

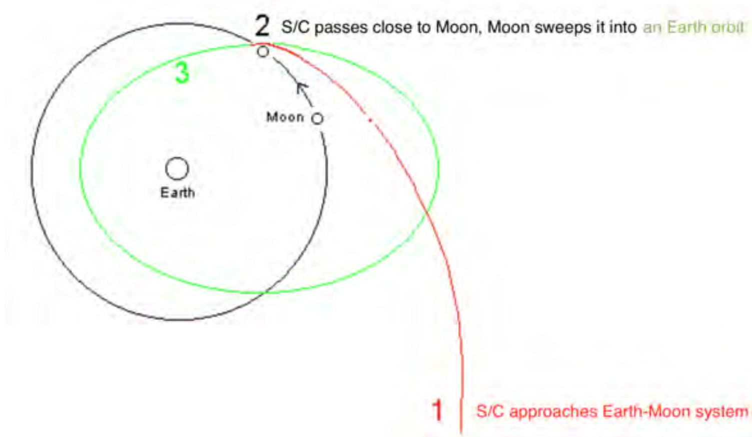


Figure 6: Gravity assist to capture the S/C into an orbit around Earth. [6]

5 Spacecraft Subsystems Design

5.1 ECLSS

Any deep space manned mission is governed by its ability to sustain the crew members over the long mission period. The Environmental Control and Life Support System have witnessed immense technological developments since the formation of the ISS. Re-usability and recycling of resources is the only way to tackle the mass budget as well as to meet the requirements of the crew.

A greater efficiency in the ECLSS design can be achieved by using a mix of State of the Art equipment and Advanced Life Support systems with an improved Technology Readiness Level (TRL). This can help in sizing down the base requirements of the crew.

5.1.1 Base Requirements

A crew comprising a man and a woman with a nominal metabolic rate of 11.82 MJ/d will be used as reference over the 882 days period. Then, the results will be adapted for a crew of four people. The Table 11 describes the metabolic interface values of the crew [7].

Table 11: Crew metabolic requirements

Interface	Average Single Crew Member [kg/CM-day]		Total 4CM 882 days [kg]
Overall Body Mass	70		280
Respiratory Quotient	0.869		N/A
	Input	**Output**	
Air			
Carbon Dioxide Produced		0.998	3 590.944
Oxygen Consumed	0.835		2 945.88
Water			
Potable Water Consumed	3.909		13 790.952
Fecal Water		0.091	321.048
Respiration and Perspiration Water		2.277	8 033.256
Urine Water		1.886	6 653.808
Metabolically Produced Water	0.345		1 217.16
Food			
Dry Food Consumed	0.617		2 176.776
Waste			
Fecal Solid Waste		0.032	112.896
Perspiration Solid Waste		0.018	63.504
Urine Solid Waste		0.059	208.152

It is seen that the major constituents, with respect to mass, are air and water. These quantities can be used and replenished again by means of efficient life support systems, which is discussed in detail in Section 5.1.2. Table 12 provides details regarding the water requirement for hygiene purposes. The values follow a Water Lean Approach which eliminates the requirement of daily shower and other water rich consumption approaches [7].

Table 12: Crew water consumption

	Mass Required [kg/CM-day]	Total 4CM 882 days
Body Wash	0.91	3 210.48
Urine Flush	0.30	309.6
Total	1.21	3 520.08

5.1.2 Selecting an ECLSS

Different options were considered when selecting a suitable ECLSS. Firstly, since there will not be any form of re-supplying, a closed loop system is used. Within the closed loop system, two options are available: the Bio-regenerative and the Physicochemical system. The Bio-regenerative system uses living organisms to produce and break down molecules in the ECLSS, whereas the Physicochemical system uses physical, chemical and mechanical devices [8].

Table 13: Trade study between bio-regenerative systems and physicochemical systems

System type	Advantages	Disadvantages
Bio-regenerative	Able to produce O_2	Requires higher power
	Able to remove CO_2	Requires higher volume
	High sustainability	Requires more crew input
Physicochemical	Simple maintenance	Low sustainability
	High reliability	Limited duration

From the trade study, the conclusion is drawn that the Physicochemical life-support system is the best option for this mission. This conclusion is mainly based on the criteria of volume and power.

5.1.3 Life Support Systems

In this section, the sizing and the requirements for basic life support components like air, water, food and other provisions are discussed. Optimum sizing with respect to mass and volume, along with operational simplicity are desired in a deep space mission lasting for more than a year, so air and water subsystems should act together in order to establish a balanced mass budget.

Air Management

The atmosphere is mainly composed by nitrogen and oxygen. Volume wise, nitrogen occupies 78.084% and oxygen occupies about 20.95% at 1 atm pressure. It is assumed that the spacecraft cabin is pressurized with O_2, N and other trace elements before launch. Since O_2 is constantly consumed by the crew, it has to be generated on-board during the mission. This is achieved with electrolysis, in which water is converted to O_2 and N_2 by applying an electric current.

The Oxygen Generation System (OGS) is a scaled down version of the ISS system, which has the capacity to provide oxygen to 11 crew members. It consists of 26 electrolytic cells in total, which can provide up to 9,25 kg of O_2 per day [9]. But our requirement is only 0.84 kg of O_2 per crew member per day, so the OGS will only use 10 cells, reducing the mass of the OGS to around 524 kg. These 10 cells require 500 A of current in order to produce 3.58 kg of O_2 per day. There is an excess of oxygen production which can help compensating leakage from the pressurized cabin. This process requires 4.030 kg of H_2O per day, producing 0.448 kg of H_2 per day as byproduct [9].

Another important aspect of the the Air Management System is the recycling of CO_2. The Sabatier Equipment used in the ISS helps convert CO_2 to H_2O and CH_4, requiring H_2 as a reactant which can be partially supplied by the byproduct of the electrolysis. The crew exhales 3.992 kg of CO_2 per day and this can be removed by the atmosphere continuously by means of a 4-bed molecular sieve (4BMS). The Sabatier equipment uses the CO_2 captured by the 4BMS. The molar fraction of CO_2 and H_2 required for the reaction is 1:4 at 350-400ºC. To carry out this process, 0.726 kg of H_2 per day are needed. The remaining 0.278 kg of H_2 are supplied by an external storage tank, which for the duration of this mission must store 245.752 kg of H_2. The result of the Sabatier equipment is the production of 3.268 kg of H_2O along with 1.452 kg of CH_4 every day.

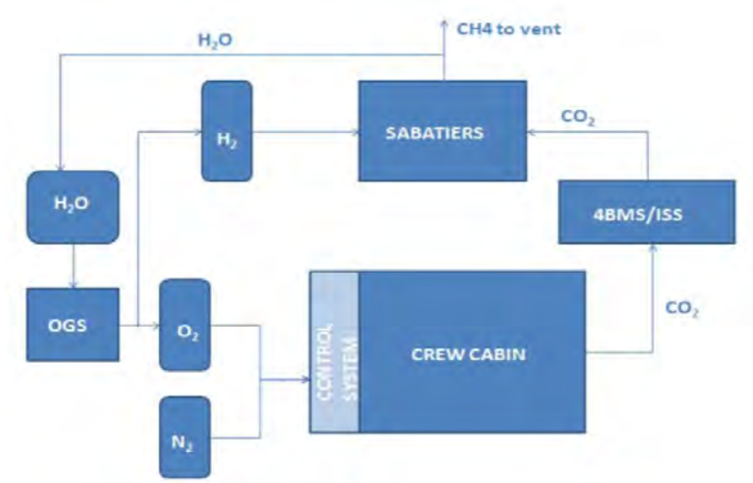


Figure 7: Air Management System layout

The water produced is recycled back to the Water Management System, while toxic methane gas is vented out into space. The Air Management System also comprises the Atmospheric Pressure Control System and the Fire Detection and Suppression system which are an integral part of the Dragon Spacecraft.

The leakage of both nitrogen and oxygen from the cabin means that they have to be replenished during the course of the mission, which depends on the free space available to the crew members. An additional 1 338.80 kg of N_2 and 357.92 kg of O_2 are therefore carried to compensate possible leaks, assumed to be in the order of 5% per day [10]. These values also include the tanks' mass. The properties of the Fire Detection and Suppression system and the Atmosphere Control System are inbuilt, being the mass and power requirements part of the overall requirements of the SpaceX Dragon dry mass and power system [11].

Table 14: Air Management System equipment list

Subsystem	Technology	Mass [kg]	Volume [m^3]	Power [W]
Atmospheric Pressure Control	Inbuilt	N/A	N/A	N/A
Carbon Dioxide Removal	ISS	179.14	0.42	536.06
Carbon Dioxide reduction	Sabatier	143.53	0.19	148.59
Oxygen Generation System	ISS	262	0.70	369.00
Gaseous-Trace Contaminant Control	ISS	85.81	0.40	194.35
Atmosphere Composition Assembly	ISS	54.30	0.09	103.50
Sample Delivery System	ISS	35.11	0.04	0.00
Nitrogen Storage	High Pressure	1 338.80	3.368	0.00
Oxygen Storage	High Pressure	357.92	0.904	0.00
Hydrogen Storage	Cryogenic	245.752	3.428	32.00
Fire Detection and Suppression	Inbuilt	N/A	N/A	N/A
Total		2 699.362	9.54	1 383.5

Water Management System

The Water Management System requires equipment from the Advanced Life Support system since water represents most of the ECLSS's mass. TRL of 6 was deemed reasonable considering the developmental schedule until 2033. As mentioned before, 4.030 kg of H_2O per day are required by the OGS, whereas only 3.268 kg of H_2O per day are returned by the Sabatier equipment. This represents a deficit of 0.762 kg of H_2O per day.

From Tables 11 and 12, one can see that the total water input required for metabolic and hygiene purposes for the crew is 20.476 kg per day. It is possible to retrieve the majority of the total input as potable water through the combination of the Vapor Phase Catalytic Ammonia Removal (VPCAR) and Lyophilizing equipment [12]. Urine and gray water are stored in a waste water tank that is coupled to the VPCAR, which in turn processes the waste water and sends it to the Lyophilizing Unit. Here, the brine is separated and stored along with the solid waste storage tank. After Lyophilizing, the processed liquid is passed again to the VPCAR until pure water is detected by the Water Quality management system and is then transferred to the potable water tank.

The humidity is assumed to be completely recovered whereas the grey water, urine and flush can be recovered at 98% of the input value. As a result we are able to reclaim 20.07 kg of H_2O per day, leading to another deficit of 0.41 kg of H_2O per day and a total of 1.63 kg/day when including the Air Management System. This means there is a deficit of 1 440.83 kg of water, which has to be included in the potable water tank. Adding an estimated 15% mass for the tanks, we have a total mass of 1 656.95 kg.

The Waste Water Tank, which stores urine, gray water and brine water, has to be sized according to the processing time of the Water Management System. The tanks required for storing waste water before recycling can be sized based on a 30-hour storage rate for our four crew members' output. These are also sized assuming a non-recyclable quantity,

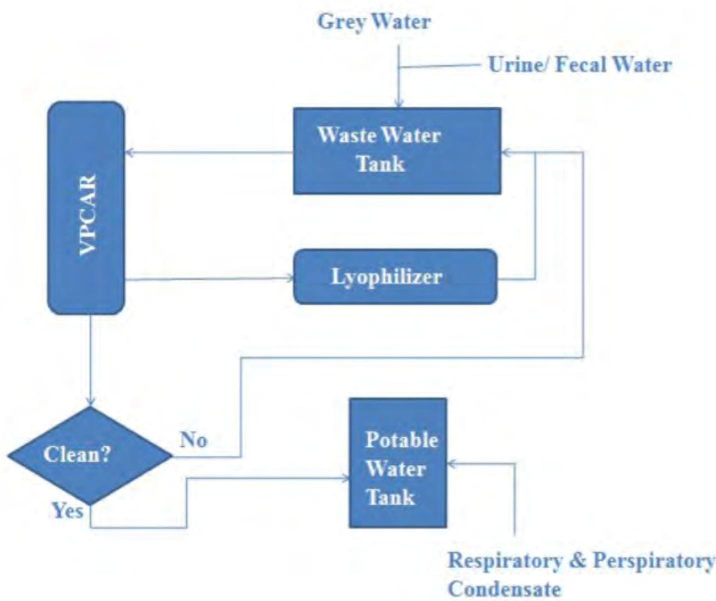


Figure 8: Water Management System layout

which arises due to the 2% efficiency drop in the water recycling system. Figure 8 represents the Water Management System.

Table 15: Water Management equipment list

Subsystem	Technology	Mass [kg]	Volume [m^3]	Power [W]
Waste Water Collection Systems	ISS	4.55	0.02	4.00
Water Treatment Process	VPCAR, Lyophilization	582.5	2.02	2 831.45
Waste Water Tanks		80.96	0.20	7.19
Microbial Control Valve		8.67	0.02	0.00
Process Controller		63.00	0.00	180.00
Water Quality Monitoring		14.07	0.04	4.72
Product Water Delivery System		71.28	0.17	4.58
Potable Water Storage		1 656.95	1.23	60.00
Total		2 481.9	3.70	3 091.94

Waste Management System

The Waste Management System includes a urinal and a commode for expelling biological waste. The urinal transfers all the waste water to the waste water tank to be later processed. The commode collects the feces and transfers them to the Solid Waste Storage tank. Perspiration and urine solid waste are also stored in the Solid Waste Storage tank. It was estimated, according to Table 11, that a total of 384.552 kg of solid biological waste will be produced throughout the mission. The Table 16 lists the important elements in the Waste Management System [12].

Table 16: Waste Management System equipment list

Subsystem	**Technology**	**Mass [kg]**	**Volume [m^3]**	**Power [W]**
Solid Waste Collection	ESDM	36.36	0.13	14.00
Solid Waste Treatment and Storage		450.01	12.57	0.00
Total		486.37	12.70	14.00

Human Accommodation

The crew accommodations subsystem is responsible for maintaining the health of the crew, providing activities for them, and integrating human factors into the spacecraft. Namely, it is important to have appropriate mass and volume reserved for crew accommodation supplies.

Table 17: Crew accommodation items

Subsystem	**Mass [kg]**	**Volume [m^3]**	**Power [W]**
Food Storage	2 401.776	18.64	672.00
EVA Suits + Clothing	886.10	0.54	0.00
Hygiene	85.00	1.50	0.00
3D-Printer	56.00	0.18	443.33
Housekeeping	63.00	1.60	0.00
Operational Supplies	90.00	0.50	0.00
Video Photography	120.00	0.50	200.00
Medical Supplies	160.00	0.83	0.00
Exercise Equipment	140.00	0.19	0.00
Portable Life Support System	96.16	0.19	0.00
Miscellaneous Items	80,00	1.00	0,00
Total	4 178.04	25.67	1315.33

As on the ISS, clothing will not be washed. Instead, enough long lasting bacteria resistant clothing for the full duration of the transit will be carried. Cleaning supplies will be included and used frequently to maintain the habitat. Equipment and consumables for patching the spacecraft will be made by a 3D-Printer. Simple and robust exercise equipment is essential to the health of the crew and can provide power to non-essential systems if necessary.

5.2 Habitat

The habitat is the place where the crew will spend most of its time, which means it's going to play the role of a new house for the astronauts. That being said, it is very important that it guarantees both comfort and a practical work environment.

In this section, a habitat vehicle is going to be chosen from already existing concepts. In order to make the best decision, the requirements are first defined and then compared to the concepts' capabilities. In the end, a parametric study is made to compare the different habitats and to choose one.

5.2.1 Requirements

From [13], the minimum admissible volume, per crew member, is 20 m^3, so for a crew of four, the habitat needs at least 80 m^3 of habitable volume. Another aspect to take into account when discussing the astronauts and their lives in space, is their health, both physical and psychological. As it was mentioned before, to maintain good physical health, the crew needs to exercise regularly. On the other hand, the psychological health is more difficult to preserve and is accomplished using a variety of small details, as eating together or having a private space. Thus, along with exercise facilities, the habitat should also include an eating area big enough to accommodate all the astronauts at the same time and private quarters for each crew member. Other important health risk is the radiation present in space which must be deviated away from the crew. Additionally, the habitat is where all the sub-systems are located so it must also ensure space for them.

Considering the total volume required for the crew, plus the volume required for the consumables, storage and other ECLSS components, which from Tables 14, 15, 16 and 17 is of at least 51,61 m^3, we need a total pressurized volume of about 132 m^3.

Finally, the structure of the habitat defines the final shape of the spacecraft and its integrity is crucial for the performance and success of the entire mission.

Therefore, the habitat has to fulfill the following requirements:

- The vehicle needs to guarantee an habitable volume of 100 m^3;
- The total pressurized volume should be at least 152 m^3;
- The vehicle configuration should include a common area for astronauts to eat or do other activities together;
- That configuration should also include private quarters for each crew member to allow them privacy moments;
- Ballistic protection as well as radiation protection must be provided;
- Resistance to the maximum and ultimate structural loads must be ensured.

5.2.2 Existing Concepts

There are four different concepts being studied for a Mars missions: Deep Space Habitat (DSH) from NASA, B330 from Bigelow Aerospace, and Destiny Module and the Multi-Purpose Logistics Module (MPLM) both from ISS. The comparison between them is presented in Table 18.

Table 18: First trade-study of concepts

Concept	Mass [kg]	Volume [m^3]	TRL (average)	Cost [USD]
Deep Space Habitat	14 116	193	7.7	-
B330	12 100	330	7	125M
Destiny Module	14 520	106	9	1.2B
MPLM	13 154	31	9	-

Based on the volume requirements, the MPLM and the Destiny Module are immediately discarded. So a further study about the Deep Space Habitat and the B330 was done, in order to make a decision between these two.

5.2.3 Deep Space Habitat

This habitat is based on the ISS systems and is a version of the ISS Lab Module. Currently, it is being already ground tested by NASA and the idea consists of adding a MPLM to the main module to separate living and working environments. With the MPLM included, this configuration offers 90 m^3 of habitable volume, which meets the requirements.

Regarding the crew, this concept offers them private and large crew quarters with no through traffic, in the quiet end of module, acoustic insulation, personal control over temperature/air flow, adjustable lightening, data/power access and private communications. It also provides weightless restraints and radiation protection. For food preparation, it has a small galley with a microwave and a refrigerator and then there's an area that accommodates all the 4 elements of the crew so they can have at least one meal together per day. To exercise, the astronauts have an open area with adjustable air flow and easily cleaned. As for the waste management, it provides larger enclosure than ISS, also easily cleaned. Finally, for personal hygiene, the astronauts will have an enclosed area that allows for whole body cleansing, hand wash, tooth brushing and personal grooming.

During their free time the astronauts can choose between spending time at the window, exercising, in their quarters or in the common area. There is also the working area which consists in a flight operations workstation and small laboratory areas for carrying out geology and life science research, as well as storage space for samples and a suiting-up area.

Although this concept has higher mass, it uses designs and systems with high TRL so the cost, production and flight readiness schedule may be reduced. A mass budget estimation for a 500 days mission is presented in Table 19. [14]

Table 19: Mass budget for a 500 days mission using the DSH

Category	Mass [kg]
Structures	14 116
Propulsion	17 908
Dry Mass	32 022
Stowed Provisions and Consumable Fluids	8 953
Non-Propellant Fluids	457
RCS Propellant	
DSH Wet Mass	41 430
Margin (10%)	4 143
Total Wet Mass	45 573

Using a cryogenic propulsion stage, the Orion capsule and a MMSEV (Multi-Mission Space Exploration Vehicle), the final configuration would be:

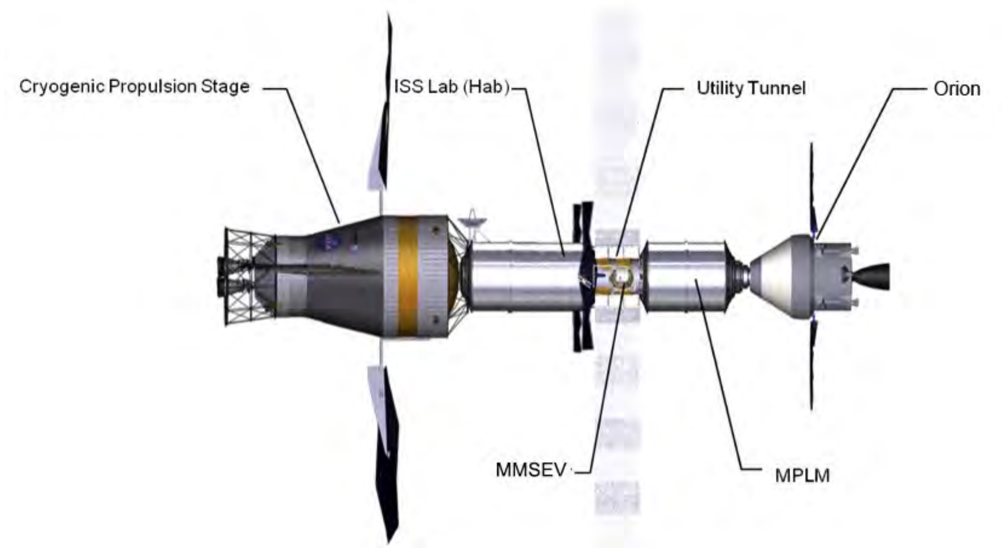


Figure 9: Deep Space Habitat spacecraft configuration

5.2.4 Bigelow Aerospace B330

Bigelow Aerospace is a private company that is currently developing inflatable modules for future space exploration, with B330 being one of those modules. Its name is derived from the fact that it offers 330 m^3 of volume, more than what we require.

This habitat, contrary to the other concepts, does not have a metallic structure. Instead, its structure is made of several layers of composite materials (using Kevlar and Vectran), which means it is much lighter than the typical aluminum ones. In fact, when compared to an ISS Destiny Module, for the same volume, B330 would weight about 5 tons less, which represents a considerable save in the mass budget.

Due to the materials used, this concept offers radiation protection equivalent to the one present on the ISS and a superior ballistic protection as well as thermal protection.[15] The design specifications are the following:

- Two solar arrays and two thermal radiator arrays for heat dissipation;
- Life Support Systems to sustain a crew of up to six astronauts, including a zero-g toilet with solid and liquid waste collection;
- Semi-private crew quarters and a person hygiene station;
- Exercise equipment, food storage and a preparation station;
- Lighting;
- The exterior will feature four large windows coated with an UV protection film.

There are some expected attributes of this concept:

- Robust Environmental Control and Life Support and communication suite;
- Large storage volumes (for food, mechanical parts or medical supplies);
- Real-time visual command and observe capability for crew;
- Low crew irradiation;
- Semi-autonomous integration of multiple mission specific propulsion units.

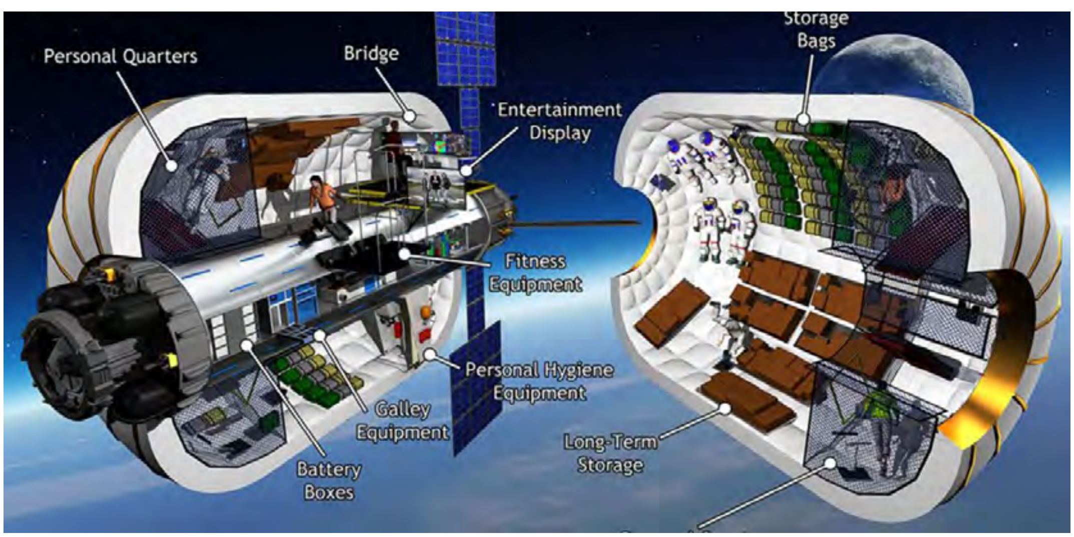


Figure 10: B330 interior schematics [15]

In Figure 11 is presented how the total spacecraft would look, combined with a propulsion module and a crew capsule.

Figure 11: B330 spacecraft configuration - Copyright Bigelow Aerospace

5.2.5 B330-adapted

Because the previous concept offers a volume almost three times bigger than what we need, we decided to do a third study about an adapted B330 which would have about 120 m^3. Since there is a concept also made by Bigelow Aerospace that has 180 m^3, we decided to use the specifications of this module to do the study. Therefore, this third concept would have the same properties of the B330 module, but would have the following dimensions:

- Pressurized Volume: 180 m^3;
- Diameter: 6.3 m;
- Length: 8.7 m;
- Structures Mass: 8700 kg.

5.2.6 Decision Matrix

Mass

Observing the mission requirements, all of this concepts could be used to successfully complete it. However, looking at the mission constraints, we can see that for each launch needed the cost rapidly increases, i.e., the more mass our spacecraft has, more launches will be required and the cost budget will be exceeded. Besides this, the launchers have a maximum mass that they can carry to a LEO (where our parking orbit will be). Based on the Table 18, we can see that the structural mass of the DSH is about 14 tons, for the B330 is about 12 tons and for the B330-adapted is about 8 tons. Because this parameter has a great influence on the launcher, and therefore on the cost, a weight of 40% was assigned.

Volume

Comparing the three concepts, we can promptly conclude that the B330 is oversized, offering a lot more volume than needed which can be a downside of this concept, for example, when it comes to accommodate it in the launcher. The other two concepts meet the volume requirements and consequently are tied in this matter. Bearing this in mind, this aspect won't be too important in the decision so a weight of 10% was decided.

TRL

The Deep Space Habitat has the best TRL and it has been already calculated by the team responsible for the concept. At the same time, the B330 and therefore the B330-adapted are based on other concepts that have been and are still proving to be very successful in space missions. So, in spite of the TRL of this two concepts being just a rough estimation, those previous missions support the readiness and availability of this concepts' technology.
The TRL can be very important not only to speed up the production of all the components of the mission but also to reduce their cost. However, in this case, it will not play a big part in our decision because there's still a lot of available time to study and test technologies that are not yet completely proven. A weight of 20% was therefore chosen for the TRL.

Cost

Finally, the DSH claims that it won't be very expensive since the technology is the same used in ISS so it's already available and no further studies have to be made. However, the fact that it's heavier will lead to multiple launches and in the end we expect the total cost to be much bigger than the total cost using the other concepts. On the other hand, Bigelow Aerospace has shown some promising numbers regarding the inflatable costs, although they are not specific on what that cost involves.
Because the total cost of the mission is one of its biggest constraints, this parameter was given a weight of 30%.

Decision

With the weights assigned to each parameter, an analytic hierarchy process can be made to compare all three concepts. An AHP tree was then built with the concepts being graded from 1 to 3, with 1 being the lowest score and 3 the highest, as presented in Figure 12. The results are shown in Table 20, along with a final trade-study.

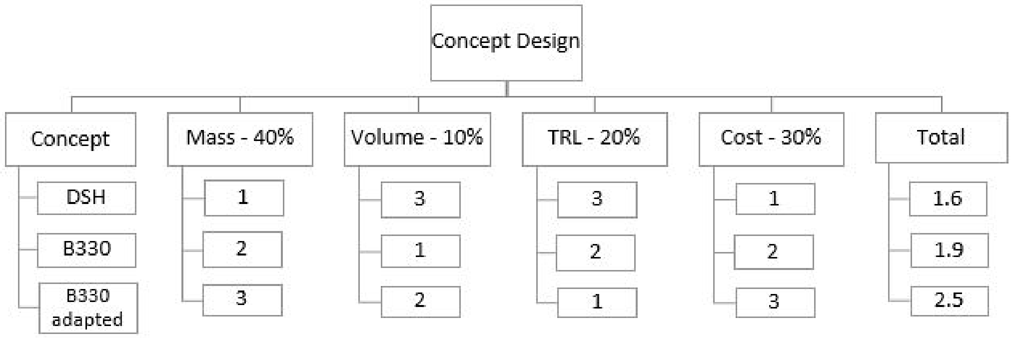


Figure 12: AHP Tree

Table 20: Concept trade-study

Concept	Deep Space Habitat	B330	B330 – adapted
Structures Mass [kg]	14116	12100	8700
Volume [m^3]	193	330	180
TRL (average)	7.7	7	7
Cost [USD]	-	125M	100M
AHP Results	1.6	1.9	2.5
Final Results	Too heavy	Oversized	Fits best the requirements

Looking at the AHP results and at the discussion made for each parameter, the concept 3 – the B330 adapted - was chosen because it represents a significant save in mass, allowing for the habitat to be sent to orbit in only one lunch, which in turn represents a saving in money. A CAD model using *SolidEdge* software was made and is presented in Figure 13.

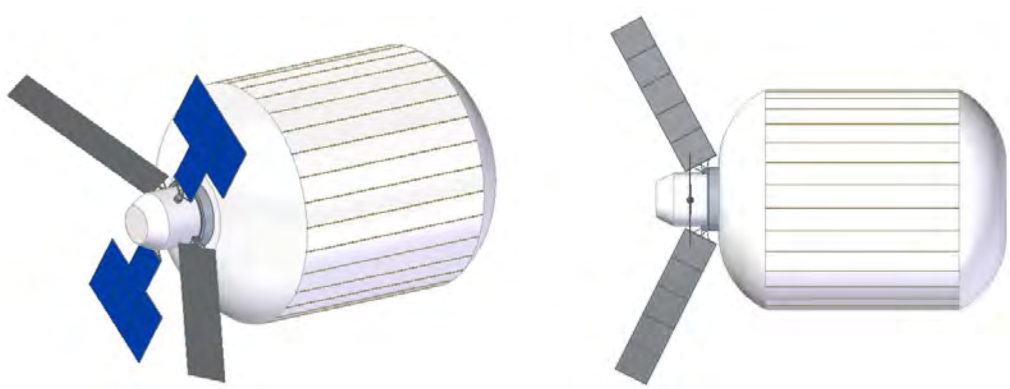


Figure 13: Habitat CAD model

5.3 Command Module

The command module is the section of the spacecraft responsible for the re-entry, which is the final and the most delicate part of the mission, since a minor error in precision could lead to bouncing off the atmosphere back into space or heating up too quickly. It is also the last component to be launched, taking the crew to the habitat.

5.3.1 Requirements

In order to achieve a successful re-entry, the following three requirements should be carefully balanced.

- Deceleration;
- Heating;
- Accuracy of landing or impact;

As it was mentioned before, the peak g-load the human body can withstand is equal to 12g's if just for a short time, or that 8g's if sustained longer. So, with the above requirements, a re-entry vehicle must walk a tightrope between being squashed and skipping out, creating a three-dimensional re-entry corridor that can be seen in Figure 14. [16]

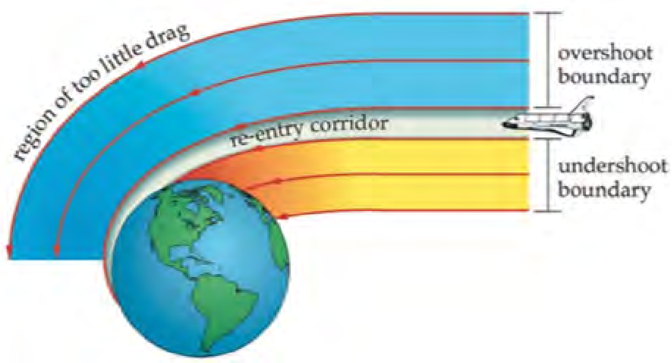


Figure 14: Re-entry corridor [16]

5.3.2 Capsule Selection

To choose the capsule, a trade study was made to compare some command modules available or in production. In Table 21 are listed the capsules considered in this study.

Table 21: Command module's Trade Study

Capsule	Manufacturer	Max. crew	Dry mass [kg]	g's	Endurance	TRL
Orion	Lockheed Martin	4	23 281	16	6 months	8
Dragon V2	SpaceX	7	6 000	3.5	2 years	8
CST-100	Boeing	7	13 000	3	210 days	7
Dream Chaser	Sierra Nevada Corporation	7	9 000	1.5	210 days	7
Federation	Roscosmos	6	19 000	3	200 days	4

In Figure 15 it is presented an AHP tree, which will help in the selection of the most adequate command module for this mission.

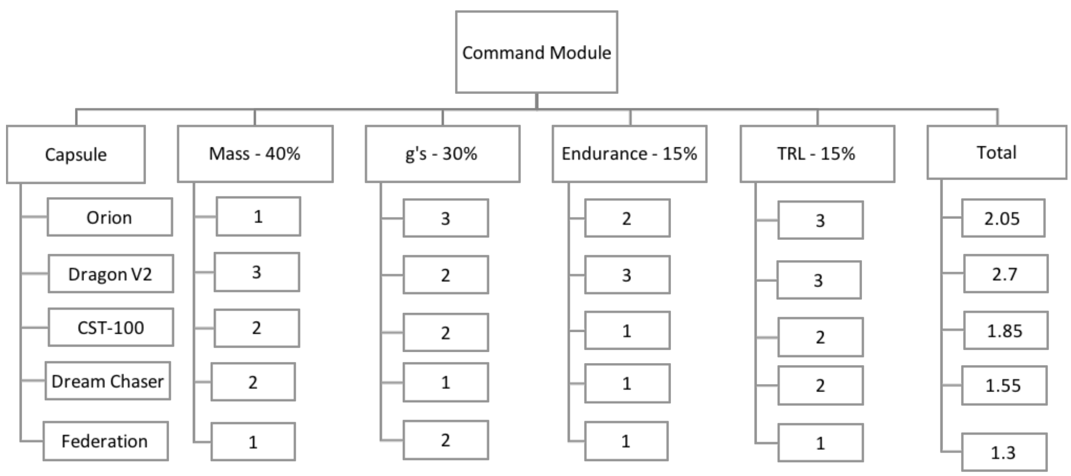


Figure 15: Command module AHP tree

After analyzing the different options and the results obtained, the Dragon capsule was selected for a number of reasons, next presented.

The dry mass value is a very important characteristic in this specific mission since it is a launch constraint: three launchers will be used, so a careful planning of the payload has to be made, in order to stay within the limits. The Dragon V2 capsule is from far the lightest of all the concepts. Also, since it is built to carry 7 crew members and we are only taking 4, we can adjust the rest of the available volume to carry payload or simply to have more space for each person.

Furthermore, the Dragon V2 has an updated third-generation PICA-X heat shield that protects the spacecraft from temperatures of more than 1650ºC [17]. This heat shield has already proven its ability in NASA's Stardust sample mission, setting a record for the fastest re-entry speed of a spacecraft into Earth's atmosphere [17].

Finally, the Dragon supports both propulsive-landing, with a very high accuracy, using four extendable landing legs, and backup parachutes, in case of propulsion failure.

Thus, the Dragon V2 spacecraft is able to fulfill all the requirements previously stated, making it the chosen capsule.

5.3.3 Characteristics

In the figure 27, on appendix B, a schematic of the Dragon capsule with the main characteristics highlighted is presented.

5.4 Propulsion

In this section the propulsion system of the spacecraft will be designed. Firstly, the requirements of this subsystem are listed. Then, the launch vehicle is described and its performance is calculated, including the propellant masses needed to deliver the required Δv. Finally, the in-space propulsion system is described.

5.4.1 Requirements

- The launcher internal payload fairing shall have an internal diameter with a minimum of 3.6 m. This is the diameter of the Dragon capsule;
- The launcher shall be able to provide a Δv of 8.4 km/s to get the payload from the Earth to a circular LEO parking orbit at 200 km;
- The propulsion system shall be able to provide a Δv of 6.7 + 5% = 7 km/s to bring the payload into the required trajectory from the parking orbit and to allow the trajectory corrections;
- The in-space propulsion system shall have enough redundant thrusters to account for failure.

5.4.2 Launcher

The selected launcher for this mission is the Falcon Heavy. The choice was based mainly in the cost per kg and the TRL. Atlas V and Delta IV, despite being proven technologies, were discarded due to their prohibitive cost. The SLS Block I, although it has a relative high cost, would be an option if it didn't have such a low TRL. The Falcon 9, from the point of view of the TRL, would be the best launcher. However, the number of launches, around 10 would be needed, would represent an increased risk. Apart from the risk, the boil-off rates of the propellant, of about 2% per day, would mean even more launches.

Table 22: Launchers market research

Launcher	Cost per launch M [$]	Payload to LEO [kg]	Cost per kg [$]	TRL
Delta IV Heavy	290	28 790	10 070	9
Falcon Heavy	135	53 000	2 500	8
Falcon 9	56.5	13 150	4 300	9
Atlas V 552	250	20 520	12 180	9
SLS Block I	500	70 000	7 100	5

The Falcon Heavy is the only option left. This launcher has its first flight scheduled for 2017, but it is based on the Falcon 9, which has been launched successfully multiple times. It is assumed that at the time of the mission this launcher will be a mature technology. The performance figures for both launchers can be found in Table 23.

Table 23: Falcon 9 and Falcon Heavy data [18] [19]

	Falcon 9 v1.1	**Falcon Heavy**
Height [m]	68,4	
Diameter [m]	3.7	11.6
Mass [kg]	505 846	1 462 836
Oxidizer/Fuel	Liquid Oxygen/ RP-1	
Stage 1		
Number of engines	9	27
Burn time [s]	180	
Thrust at sea level [kN]	5 885	17 615
Thrust in vacuum [kN]	6 672	20 017
Stage 2		
Burn time [s]	375	
Thrust [kN]	801	

5.4.3 Propulsion System Selection

Taking into consideration the critical nature of this mission, a big emphasis was put in reliability and fuel efficiency.

Due to low thrust, weight ratios and high power consumption, electrical engines were ruled out. The increased complexity and lack of flight heritage of hybrid chemical engines, and the impossibility of turning off solid propellant engines, meant they were not suitable for this mission. A nuclear option wasn't an option as well due to radiation concerns.

This led to the choice of a chemical propulsion subsystem, namely bipropellant engines due to their higher I_{sp}. Several other engines are available and a comparison between them can be found in [23].

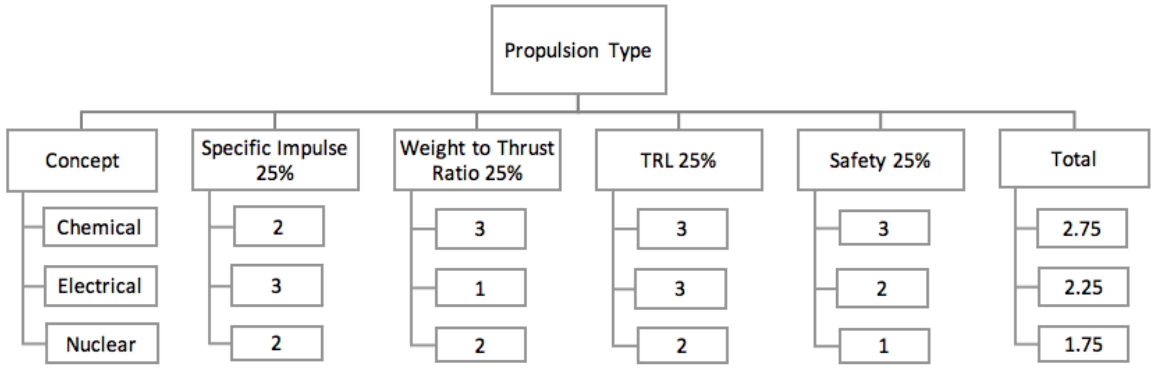


Figure 16: Propulsion AHP

Knowing the dry mass of the spacecraft, m_d, the Δv required for the whole mission, and the specific impulse of the engine selected, it is possible to calculate the total fuel mass, m_f, as shown in Equation 1. In Table 24, different cryogenic engines are assessed.

$$m_f = m_d \left(e^{\frac{\Delta v}{g_0 I_{sp}}} - 1 \right) \tag{1}$$

Table 24: Engines fuel and weight comparison for a Δv of 7 km/s

Engines	**I_{sp} [s]**	**Weight [kg]**	**Fuel required [kg]**
RL-10B-2	462	277	110 791
CE-7.5	454	445	117 737
LE-5B	447	269	118 288
HM7B	446	165	116 711
YF-75	438	550	127 346

From Table 24 it can be seen that the RL-10B-2 engine is the best option due to its better fuel consumption (higher I_{sp}). This engine has been flight proven so its failure risk is quite low. Also, its performance can be further improved by using another propellant, the $H_2 : Be$49:51 [22]. The increased exhaust velocity of this propellant represents a saving of 28 263 kg of fuel. Thus, the total fuel mass is 82 528 kg.

The RL-10B2 rocket engine is a bipropellant engine using liquid oxygen and RP-1. Bipropellants have the best specific impulse of the existing chemical rockets, and therefore a lower propellant mass. The main disadvantage is that liquid oxygen, being a cryogenic propellant, is hard to store in space. For launch vehicles cryogenic storage is usually not a problem, but if they have to loiter in space the propellant (LOX) will boil-off.

In order to face this issue, a strategy was devised using a Integrated Common Evolved Storage system developed by Lockheed Martin. This was designed for an upper stage that uses the very same RL-10B2 engines so, with minor modifications, this system can be implemented. This system uses Variable Density Multiple Layer Insulation (VDMLI), optimizing radiation insulation capability relative to standard MLI by having a gradient of spacing in between each of the layers of insulation. The drawback of this technique is the slightly larger volume due to the larger spacing but boil-off rates can be reduced from 2% per day to 0.7-1% per day [20] which is more important than volume savings.

5.4.4 In-space Propulsion System

Following the requirements, the spacecraft has to be able to perform a Δv of 300 m/s during the trajectory, for evasive maneuvers, spacecraft orientation, unexpected maneuvers that may be needed, as well as the thrust needed for the GNC system.

The Dragon capsule has 18 Super Draco thrusters and a fuel tank capacity of 1 388 kg. [21] From the propulsion requirement, using Tsiolkovski's rocket equation, a propellant mass of 8 200 kg is found. It was assumed that the fuel that is already available in the Dragon capsule (1 230 kg) is needed for the re-entry. Therefore, a total of 9 430 kg of propellant will be needed. For 18 Super Draco thrusters, the total amount of propellant that can be carried is 24 984 kg of fuel, which is more than needed. Therefore, this type and amount of thrusters is sufficient for the mission. Since this is inbuilt in the Dragon and performs accordingly to our requirements, these thrusters were chosen over other similar ones that have never been tested with this capsule, hence they would mean an increased risk. Also, adapting existing thrusters to the Dragon wouldn't be cost efficient for eventual marginal fuel efficiency gains.

5.5 Guidance Navigation and Control

The guidance, navigation and control subsystem is essential in all phases of the mission because it controls the orientation of the spacecraft as well as its position. This is critical for the spacecraft to be able to perform the correct maneuvers and to keep the solar panels pointed to the Sun and the communication's antennas pointed to the Earth.

In order to work properly, this subsystem uses several types of sensors whose output data is transmitted to various control devices (thrusters or actuators) that will correct any deviation from the original path of the spacecraft.

There are four main phases where the GNC system must be active: in the parking orbit, during the cruise phase (interplanetary trajectory), during the parking orbit in Mars and finally in the re-entry, descent and landing phases.

In the first phase, the parking orbit around Earth, the system has to guarantee safe rendezvous and docking procedures. In the end, it's also responsible for the insertion trajectory which needs to be accurate.

In the cruise phase, both going to Mars and returning to Earth, the GNC ensures that the spacecraft is following the right trajectory and that it is correctly pointed to the Sun.

While the spacecraft is orbiting Mars, the system continues to track the vehicle position in its orbit. It can also be used to point any cameras or sensors to Mars surface, with high accuracy.

Finally, the last phases (re-entry, descending and landing) are accomplished with the Dragon Capsule, which has its own GNC system, so this phases can be neglected when designing the spacecraft's subsystem.

5.5.1 Requirements

The requirements for the GNC are dependent of the mission and other subsystem's requirements. The first major requirement is maintaining the right altitude of the parking orbits. In the Earth's case, the velocity needed to leave the Earth's sphere of influence, and therefore the fuel needed, increases when altitude decreases, so it is important to keep the orbit at the defined altitude. Also, in both cases if the altitude gets too low, the planet's atmosphere, and the derived drag and thermal loads, can be harmful to the spacecraft.

The pointing stability is the second aspect that needs to be considered at all times, since it is important for the efficiency of both the communications and power system. The pointing accuracy also plays a big part in maintaining the thermal balance mentioned in Section 5.8, since it was assumed that only a part of the spacecraft receives solar light. Finally, accurate attitude determination and control in all the different phases must be ensured.
Based on similar missions, the normal requirements for this kind of missions are [39]:

- Altitude accuracy – minimum of 1 km;
- Attitude determination accuracy – 0.004º;
- Attitude control accuracy – 0.012º;
- Pointing stability – 0.1º.

5.5.2 Onboard vs Ground Systems

Until now the orbit maintenance and control were executed from the ground, using standard radiometric tracking data and sometimes onboad optical data (images taken by the spacecraft's cameras). This arrangement has proven to work well, providing accurate navigation in all the missions where it was used. However, this means that orbit adjustments have to be computed in the ground, which leads to the major drawback of this type of system: the delay caused by the time it takes for a signal to be transmitted between the spacecraft and the Earth. In addition, there is also the time that the ground station needs to process the information received before it can transmit the right commands to the spacecraft and execute them. Besides this limitation, the current Deep Space Network, which is the ground station that provides the navigation for various interplanetary missions, will eventually get overloaded and other navigation methods will be needed.

To respond to this need, autonomous navigation systems were created, enabling some or all of the navigation functions to be performed onboard the spacecraft. With this, a spacecraft can determine its orbit and attitude, point its solar arrays at the Sun and the antennas at the Earth without outside intervention [24]. Connecting this system with a ground station will enable the last to also know where the spacecraft is at all times, even if they are not directly communicating.

This type of GNC system has some advantages, with the main one being the elimination of the delay, not only the one due to the transmissions but also the human-related delay. Thus, the time required to react to late-breaking navigation information could theoretically be reduced from days to minutes [25]. Consequently, this will result in smaller Δv maneuvers, less fuel needed, increased mission performance and decreased mission cost. Also, as it was already mentioned, using an autonomous navigation system will relieve the DSN of operational burden and eliminate the need for continuous planning/replanning cycles since the position of the spacecraft is always being updated. Finally, it also enables the vehicle to have smaller and lighter weight thrusters and attitude control components, because the thruster burns are much smaller than for traditional orbit maintenance.

On the other hand, it also has some disadvantages. For example, the navigation systems will be heavier because they require onboard hardware to analyze the collected information and execute the right commands. This increase in hardware will in turn lead to an increase in power requirements.

Bearing this in mind, an automatic navigation system was chosen to be used in this mission. However, in the rendezvous and docking phases, the automatic system will not be sufficient on its own so it will be used combined with an Earth-based tracking station (near Earth the time delay is negligible).

Currently there are two systems that could be used: the Microcosm Autonomous Navigation System and the Autonomous Optical Navigation. Since the MANS has only been used in low Earth orbits and the AutoNav has already been successfully used in deep space flights, and because our choices always fall on proven technology, the AutoNav was selected.

5.5.3 AutoNav

The basis of the AutoNav is to take images of natural bodies, like asteroids or planets, either from a distance or a close-up, in order to provide one or more line-of-sight vectors to that body or to locations on that body. A succession of these images provide multiple LOS vectors that are input into a filter and then used to estimate the spacecraft's position and velocity. The accuracy of this method is dependent of many factors, from the camera used and its systems parameters, to the distance to the target bodies or the knowledge of the ephemerids of these targets.

Originally, the AutoNav code was developed for the Deep Space 1 mission in 1998 and since then it has proven to be versatile and has worked well in five missions. However, it is now considered old, so a new version is being developed based on the experience gained from this current version. This new version will have many enhancements, the major one being the addition of attitude guidance and control merged with the current translational navigation capabilities, and will be called AutoGNC. In face of this new version, it is expected that by the time this mission is set, this system will have achieved better capacities than the ones achieved until now. In the DS1 mission a Miniature Integrated Camera and Spectrometer was used, which consists of one ultraviolet spectrometer (UV Imaging spectrometer), two high-resolution imagers (APS and VISCCD imagers) and one infrared spectrometer (SWIR Image spectrometer) [39]. During the cruise phase of the mission, the images are taken by the UV Imaging Spectrometer, SWIR Image spectrometer and the VISCCD imager. Hours before the approach to Mars the high resolution camera will be switched to the APS imager, which uses landmarks on the surface of Mars to determine its position and velocity. The images provided by this device can also be used later to study possible locations for future landing missions.

5.5.4 Budgets

For the attitude determination and control there are two important factors that need to be taken into account: the disturbance torques and the constant pointing of the same spacecraft's area towards the Sun (thermal requirement). It's these two factors that will influence the sizing of the hardware needed for the mission. The Table 25 presents the components needed for the GNC system, including the ones responsible for the attitude determination. [20].

Table 25: GNC components

Component	**Weight [kg]**	**Power [W]**
Reaction Wheels	10	60
Camera	-	-
Landmark Tracking	4	10
Star Sensing	5	20
Margin of 15%	19	90
Total (approx.)	22	104

5.6 Telemetry Tracking and Communications

The communications subsystem presents itself as an integral part of the spacecraft, since it represents the only data link between the spacecraft and the Earth. For a mission so far away from the Earth, a robust and reliable connection is necessary. This system, besides its main function, will also be useful to help maintain the psychological health of the astronauts, since it allows them to communicate with their family and friends, helping them to cope with the distance. This way, the subsystem indirectly contributes to a good working environment.

5.6.1 Requirements

The main functions of this subsystem and its requirements will be next defined.

- **Telemetry** - This is the function responsible for monitoring voltages, temperatures and accelerations, which is important to determine if the satellite is in a good health state or, in case of failure, to find where it occurred. Usually, in deep space missions, a rate of about 8 kbps is used for this type of data[20]. However, in this case, it had to be multiplied by a factor of 32, because the mission is designed for humans, meaning there are a lot more parameters that have to be constantly monitored. A maximum data-rate of 256 kbps was therefore determined, so it meets the monitoring requirement. A multiplexer is also needed, to combine all telemetry data collected into a single bit stream, reducing the data rate.

- **Command** - This function is determined by the user, that sends the command data through the ground segment to provide instructions about the spacecraft's performance. For safety purposes, these commands are first stored, verified and then, only after a final input, executed. Since this type of data is generally of the same order as the one used in telemetry, a data-rate of 256 kbps was chosen once again.

- **Tracking** -As the name indicates, this is the part charged with tracking the position of the spacecraft, at each moment. Therefore, the data gathered from the sensors has to be sent to the ground station, which will then compute the position, range and the range rate of the vehicle. Again, a maximum data-rate of 256 kbps was determined.

- **Data collection** - The data collection is made using radars, sensors or cameras. The kind of data collected by this devices is therefore much larger than the previous ones, since it's composed of pictures and videos, for example. For this reason, it is necessary onboard data handling, i.e., the data must be processed and selected onboard so that only the important parts are sent to the ground segment. Acquiring this data is part of the secondary goals, so the largest volume possible of transmitted information, as well as the fastest transmission are desired. The current technologies can provide a maximum data-rate of 6 Mbps, which in turn makes it possible to achieve at least 3Mbps in critical situations. This way, an HD video can be sent almost in real time: volume data for one minute of HD video is around 200 Mbit so data-rate = volume/60s = 3.3 Mbps.

5.6.2 Ground Segment

The best support on the Earth to communicate with the spacecraft was identified as the Deep Space Network of NASA. This network also includes a Tracking and Data Relay Satellite, to compensate the periods in which the spacecraft is not in line of sight with any antenna on the Earth . The other option would be the ESA ground station DSA, which has more stations and better spread on the Earth surface. However, it doesn't have a relay satellite as good as the NASA one. This way, the reduction of darkness periods was preferred to the quality of the link and the DSN was chosen.

Some technical features of the DSN and TDRS will be presented next.

- DSN currently consists of three deep-space antennas placed approximately 120 degrees apart around the Earth, in Goldstone/CA USA, Madrid/Spain and Canberra/Australia. They work with S-band, X-band or Ka-band (micro-waves) and require a large parabolic dish and an huge amount of electric power.

In the Table 26 some technical data of the Goldstone site is presented.[27]

Table 26: Goldstone antennas technical details

Antenna type	**Site**	**Uplink freq. [MHz]**	**EIRP [dBW]**	**Downlink freq. [MHz]**	**Gain [dB] and G/T[1] [dB/K]**
34m BWG	Goldstone	X 7145-7235	89.5-109.5	X 8400-8500	65.2/54.2
	Goldstone	Ka 34315-34415	97.8-103.8	Ka 31800-32300	78.4/61.1
34m HEF	Goldstone	X 7145-7190	89.9-109.8	X 8400-8500	68.3/53.2
	Goldstone	-	-	-	-
70m	Goldstone	X 7145-7190	95.6-115.8	X 8400-8500	74.5/61.5
	Goldstone	-	-	-	-

[1]Gain-to-noise-temperature ratio

- TDRS is a constellation of nine working satellites (Figure 17), located in a geosynchronous orbit, of which only six are operational. This system was designed to support multiple missions and once more each satellite supports S-band, Ku-band and Ka-band. Although the X-band represents a good compromise between the quantity and speed of the data, the Ka-band would be even better. However, it is more affected by the atmosphere. For this reason, our spacecraft will be equipped with both X-band and Ka-band, with the last one being used only to link with the TDRS, since it doesn't support the X-band.

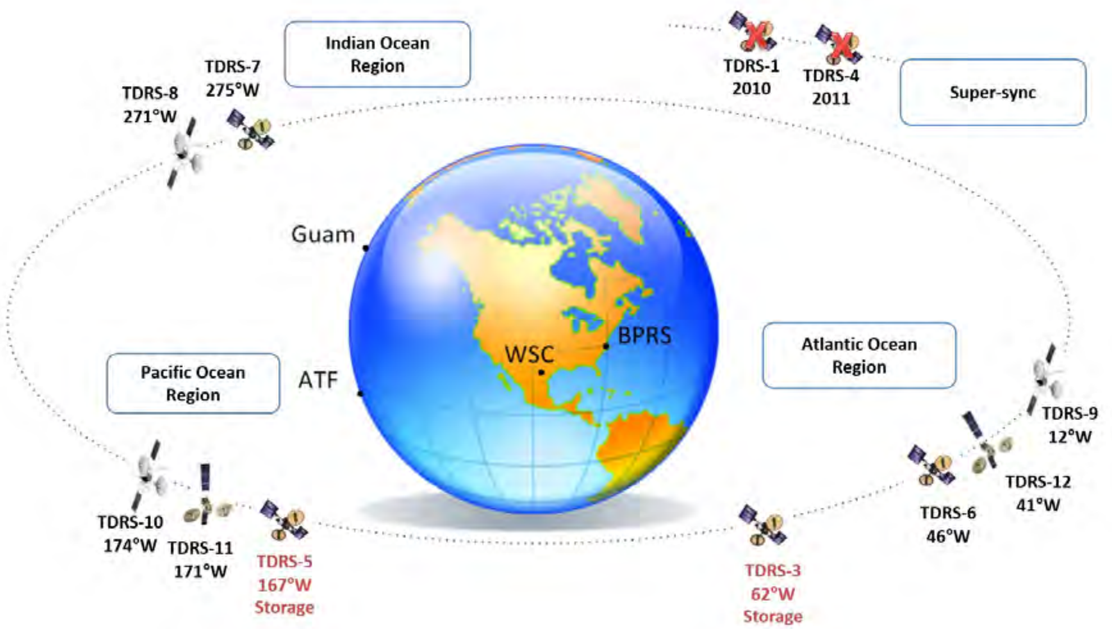


Figure 17: Tracking and Data Relay Satellite [26]

5.6.3 Main Equipment

To be able to properly communicate with the ground segment, the spacecraft was equipped with the following devices:

- An High Gain Antenna [29], that receives and sends to the Earth all the data during normal phases of the mission. It is a parabolic one, highly directive, with a 3m diameter dish and high data rate, supporting both X-band and Ka-band. The purpose of the antenna is to collect the incident power signal, so the bigger dish the better will be the reception. One of its advantages is the compatibility with the main dimensions of the spacecraft. Also, since a similar antenna was already produced, manufacturing problems are not expected, which means availability and reliability should be guaranteed. Due to the dimensions and to the extreme pointing precision required (highly directive antenna) [28], a heavy gimbal with drive motors is necessary.

- To improve the reliability of the subsystem, there will be two Low Gain Antennas pointed differently, so that it is possible to use them for short periods or during emergencies. They will have a smaller data-rate and will be omnidirectional antennas as it is fundamental to communicate without pointing, for example, in case of loss in attitude.

- Hidden in the structure there will be several amplifiers, transponders and multiplexers. To increase the reliability of the link, two amplifiers of 100W of nominal power will be mounted onboard and the transponders will be redundant as well, despite only one working at each time. Also, similar amplifiers were already used in a deep space mission and they are a good compromise between power and weight.

These equipment is summarized in Table 27.

Table 27: Feature of devices

Device	Mass [kg]	Volume [dm^3]	S/C power input [W]
X-band trasponders (Ka-)	5.8 (x2)	3.4	16
X-band amplifiers (x2)	14.8	3.6	172
Ka-band amplifiers (x2)		3.6	81
High gain antenna (HGA)	19.1	-	-
Low Gain Antenna (x2)	0.9 (x2)	-	-
HGA gimbals	45	-	14
Total	**92.3**	-	**283**

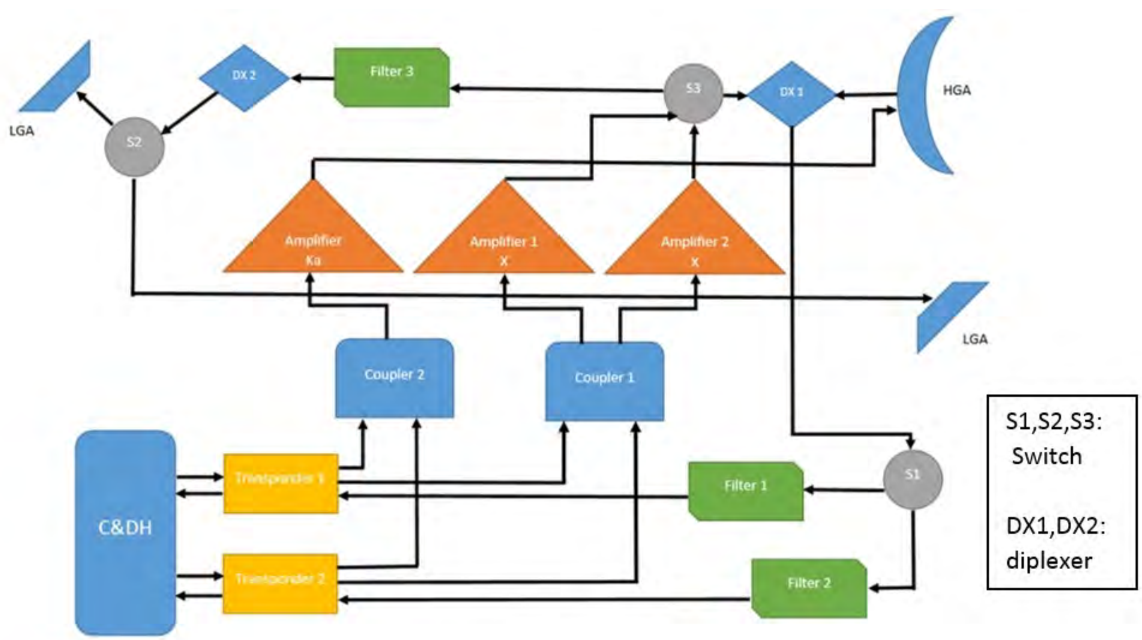


Figure 18: Block diagram of communication system

5.6.4 Communication Budget

It is necessary to do a link budget to check if our link is close, i.e., working properly, or if there is any waste that should be reduced. The connection was designed to work in the worst case scenario. In Table 28 communications schedule for all the mission phases is presented.

Table 28: Communications schedule

Mission phase	Sub-network requested	Hours per track	Tracks per week	Weeks required
Launch	34m BWG	continuous	-	-
LEO insertion	34m BWG	continuous	-	-
Maneuver	70m	continuous	-	-
Interplanetary cruise	34m BWG	4	4	28
Mars orbit insertion	70m	continuous	-	-
Observation	34m BWG	11	7	44
Cruise back	34m BWG	4	4	52
Landing	70m	continuous	-	-

The specifics about the scheduled contacts derive, first of all, from the risk that some phases involve, like the launch or the interplanetary maneuvers. On the other hand, during simpler phases, such as cruises, it was decided to hold the communications service costs.

A different guide line will be followed during the Mars orbit phase, due to the relative positions of the planets and the spacecraft. This position grants a closed efficient link for 10-11 hours per day, compared to theoretical 16 hours. If the relative position was different, during this scientific phase a continuous contact would be better.

Finally, the most critical phase will be the observation: since the spacecraft will be around Mars, so far away from the Earth, all of the devices must be operational so the scientific data collected is sent. Therefore, to define the worst case scenario, the maximum distance between the spacecraft and the Earth was considered, as can be seen in Figure 19.

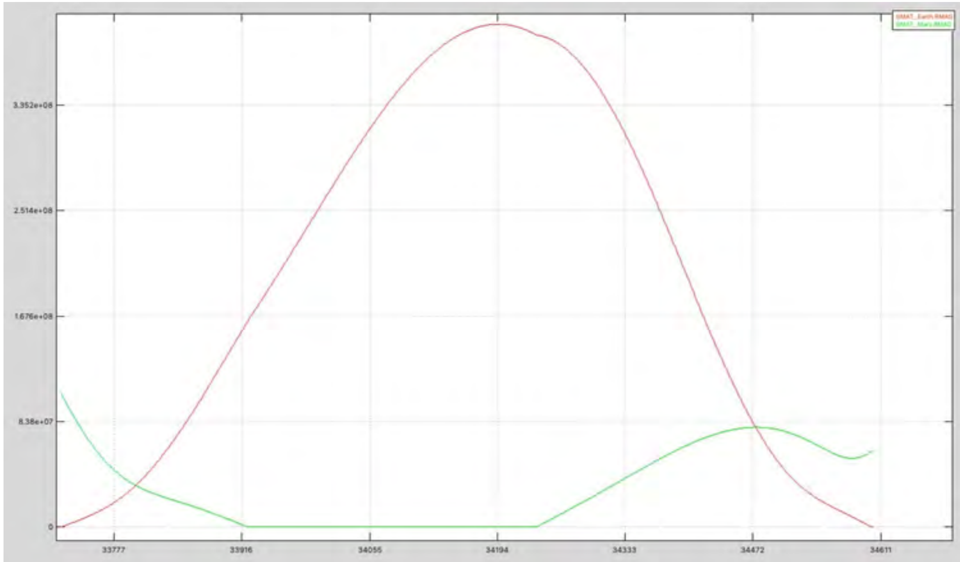


Figure 19: Distance between the Earth and the spacecraft during the entire mission

The maximum distance will be 4×10^8 km. In Table 29 the uplink and downlink between the HGA and one antenna of 34m BWG[2] located on Earth is shown. Note that this choice was dictated by the greater presence of this kind of antenna around the Earth.

Table 29: Link margin

Antennas	**34m BWG**	**3m HGA**
Link	uplink	downlink
Centre frequency [MHz]	7150	8450
Transmit power	20k	100
Space loss [dB]	281.6	283.2
Max data-rate [bit per second]	6M	6M
Gain antenna[dB]	68.2	46.7
Beamwidth [deg]	0.07	1.38
E_0/N	10.2	3
Bit Error Rate	10^{-5}	10^{-5}
Link margin	7.48	0.3

A very low link margin in downlink was detected, this is the most critical phase.

The modulation used was the BPSK because it has a low value of E_0/N (signal-to-noise ratio) required (2.7 dB at BER=10-5) and it leads to the best performance of the BER, at the expense of higher complexity. The maximum data-rate value for the LGA antennas must be at least enough for telemetry and command data, because these antennas

[2]BWG-Beam Waveguide Antenna is a large steerable parabolic antenna in which the radio waves are transported in a beam between movable dish.

are used specially in emergency phases and these kind of data is necessary at all times.

As it was already mentioned, the main system will be redundant to withstand the occurrence of certain failures of its components. If it still doesn't work, the LGAs could be used even if they are limited to basic functions.

In Earth, each "big" antenna of the ground station covers a huge surface, so it is impossible to shift the link to another antenna because the nearest is still too far. One solution to this problem could be the use of a new network for deep space, from other companies, like the DSA from ESA. However, this would involve incompatible regulations and excessive cost increases. To avoid these problems, it would be preferable to transmit the signal to another antenna of the DSN through the relay satellites (TDRS), during the time of repairing the failure.

5.6.5 Computer System

The communication system is closely connected with the computer system, with the main features of this system being:

- It is an embedded system, i. e., there is a built-in processor, providing real time control of the components of a larger system, for example, handling or processing information at the time of events.
- The operating system manages the computer's resources: I/O device, memory and scheduling. The capability to store information is very important for our mission, since all of the spacecraft control procedures must be stored. Solid State Recorders [30] based on Flash technology with a capacity up to 20 Tbit were chosen, in order to store all the mission data.

These two systems work with two kinds of software: one relating to the management of all housekeeping data, that is not directly related to payload, and the other being the application software, required by the user, that works with mission data. The latter was commissioned to three different companies, not connected to each other, in order to avoid the same mistakes and obtain a software as reliable as possible.

The heart of this subsystem can be seen like a black box, called command and data handling (C&DH): it receives, validates, decodes and distributes commands to the peripherals and moreover it gathers, processes and formats spacecraft's housekeeping and mission data for downlink. Therefore, having a computer system that is capable of performing all of this, an efficient mean for autonomous control of the spacecraft is guaranteed.

5.7 Power

Electrical Power System (EPS) is necessary both for life support and for most subsystems of the spacecraft. EPS is responsible for the generation, regulation and distribution of power to the vehicle. Therefore, to achieve an optimal design, it is necessary to define the EPS's requirements, choose a suitable primary and secondary power system and finally, a power control and distribution network.

5.7.1 Requirements

When designing the electric power system of a mission, the first step is to identify the power budget. The total power budget is shown in Table 30, taking into account the average power consumption of each sub-system.

Table 30: Total power budget

Sub-System	Avg. Power [W]
ECLSS	5 805
TT&C	283
GNC	107
Thermal	808
Scientific Payload	123
Sub-Total	7 126
10% Margin	713
Total	7 840
50% Margin	3 920
Peak Power	11 760

A 10% safety margin was considered as a precaution and a 50% margin was defined as a power peak approximation. The following requirements of the EPS were identified:

- The EPS must provide an average of 7.8 kW;
- The EPS must be able to provide 11.8 kW during power peaks, during 2 hours a day;
- The EPS must be able to sustain the spacecraft for 882 days;
- The EPS must be able to adjust the power produced with varying solar intensities.

5.7.2 Primary Power System

The primary power system converts fuel into electrical power. Usually, solar arrays are used as the primary energy source, meaning the solar radiant energy is converted to electrical energy via the photovoltaic effect. On short duration missions, fuel cells can also be considered for the primary energy source. These cells perform a controlled chemical reaction in order to produce electrical energy. For longer duration missions, like CAMOES, a combination of solar arrays and fuel cells can be adopted. Nuclear power systems were avoided due to their inherent risk.

Table 31 compares solar photovoltaic cells and fuel cells in different parameters [20].

Table 31: Solar photovoltaic cells vs fuel cells

Design Parameter	**Solar Photovoltaic**	**Fuel Cell**
Power Range [kW]	0.2 - 300	0.2 - 50
Specific Power [W/kg]	25 - 200	275
Maneuverability	Low	High
LEO Drag	High	Low
Degradation	Medium	Low
Fuel Availability	Unlimited	Medium

From Table 31, it appears as if the use of fuel cells would be a major advantage over solar photovoltaic cells. In fact, fuel cells give more flexibility to the spacecraft, as they provide power both during sunlit and eclipse periods. It also has a high energy density [34]. However, this power source requires fuel. After some preliminary calculations, it was concluded that the use of fuel cells for periods longer than one month becomes unfeasible.

A combination of both power sources was studied and proved to be the most satisfactory design solution.

Fuel Cells will be used for LEO maneuvers, such as *rendezvous* and docking. As shown in Table 31, these cells are not influenced by the atmospheric drag, and are less affected by degradation, such as spacial debris (a factor to consider, specially in LEO). This way, the deployment of the photovoltaic panels will be postponed, since they are very sensitive to degradation and can easily deteriorate due to particle collision. Fuel cells will also be used as a complementary power system during peaks of power and they might be used as an emergency power source as well, e.g., if an eclipse period is longer than predicted.

In LEO, the spacecraft will require an average power of 7840 W. It was assumed that, during this period, there won't be any power-peaks. Assuming that the fuel cells used have a specific power of 245 W/kg (as in the Space Shuttle [34]), a combined weight of 32 kg is required. The fuel required by these cells is given by Equation 2:

$$M_{reactants} = E \cdot C_r \tag{2}$$

where E represents the total energy, and C_r the consumption rate, assumed to be of 0.50 kg/kWh. So, considering a 3 days period in LEO, it is necessary $M_{reactants} = E \cdot C_r = (7.84 \times 3 \times 24) \times 0.50 = 282.24$ kg.

For the solar arrays, the UltraFlex Solar Array System will be used. This technology has 100% flight success and uses Triple-Junction solar cells, which have a solar cell efficiency of 30% and a specific power of 103 W/kg [31].

Assuming a degradation of 2.75% per year, for a mission of 882 days (approximately):

$$L_d = \left(1 - \frac{2.75}{100}\right)^{\frac{882}{365}} \qquad P_{EOL} = P_{BOL} \cdot L_d \tag{3}$$

where L_d represents the life degradation and P_{EOL} and P_{BOL} represent the power in the EOL (End Of Life) and BOL (Beginning Of Life) phase, respectively. The solar arrays must provide 8.4 kW, even in the EOL phase, so now P_{BOL}

can be computed.

$$P_{BOL} = \frac{7840}{0.9348} \approx 8400 \text{ W}$$

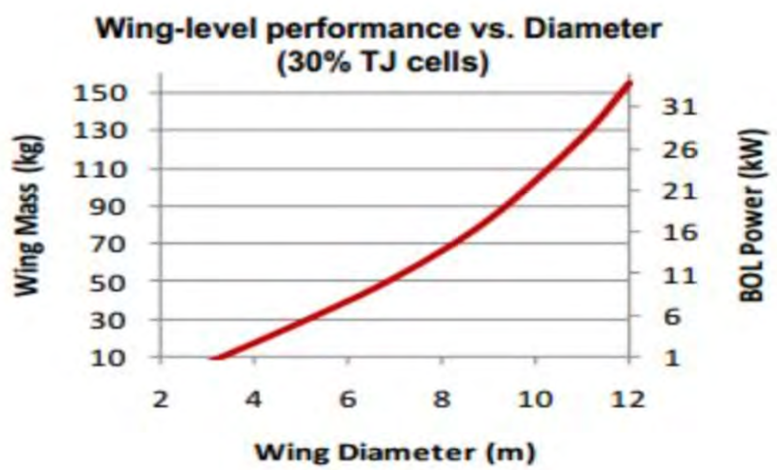


Figure 20: ATK ultraflex circular array wing performance

Figure 20 shows the performance of the solar arrays chosen for the spacecraft. To meet the power requirements, 2 solar arrays are necessary, each of them with a wing diameter of 6 m, with a combined weight of 80 kg.

Finally, it is assumed that the spacecraft experiences power-peaks periodically, around 2h a day. So, in a 882 days mission, the spacecraft requires 11.76 kW during 74 full days. Therefore, it is necessary to design an additional power source. Fuel cells represent an interesting option, considering that they are already installed in the spacecraft. But, as referred before, for periods longer than one month, the required fuel mass becomes unfeasible.

An alternative to fuel cells is to expand the solar arrays area. Given the power necessary for peak periods, it is estimated P_{BOL} = 12 580 W. So, for the CAMOES mission, 2 solar arrays, each of them with a 8 m wing diameter and with a combined weight of 120 kg are required.

The Dragon capsule has its own electrical power system, which was designed to ensure power production, management and distribution, at least during the Stage I (launch) and Stage IV (re-entry). Its default configuration has 2 sets of solar arrays, each one with approximately 23m^2, capable of producing 2.0 kW in LEO. If necessary, the solar arrays can be augmented, thus providing more power.

According to the studies performed, this configuration is enough to meet the power energy requirements.

5.7.3 Secondary Power System

Batteries have been used extensively as the secondary power system, as they provide power when the primary system is not enough. This secondary system works as a back-up for the solar arrays, providing power during eclipses and/or any technical issues. In the presence of sunlight, the batteries are recharged.

GMAT estimates that the spacecraft will be in the shadow area of Mars during 80 days, for 15% of each day. This means that, for more than 3.5 hours, the batteries must provide all the power needed by the orbiter.

According to [32], lithium ion cell batteries should be used in deep space missions, where the period in the shadow is not considerable. The type of batteries selected is therefore Cobalt Li-Ion batteries, as they have the highest energy density at 160 Wh/kg. Although the manganese and phosphate types of lithium-ion batteries are more stable over time, the low number of cycles required (less than a 100) for the batteries system justifies the use of Cobalt Li-Ion batteries.

A total energy of $7\ 840 \times 3.5 = 27\ 440$ Wh is required, thus $\frac{27\ 440}{160} \approx 172$ kg of batteries must be integrated in the spacecraft.

It is important to note that the Dragon Capsule also has its own battery system integrated.

5.7.4 Power Distribution and Control

The power distribution and control network is required to deliver appropriate voltage-current levels to all spacecraft loads when required. Both primary and secondary power system characteristics will change during the mission (the primary power source always degrades during the mission, for example), leading to a requirement for voltage and/or current regulation.

The chosen power distribution unit (PDU) is capable of distributing 2400 W, in eclipse mode, and weighs 16.3 kg [33]. Since 4 PDUs are needed, to control the power output required by the spacecraft, the complete power distribution and control unit has a combined weight of 65.2 kg.

Figure 21 shows the power management of the EPS and the connection between all of its systems.

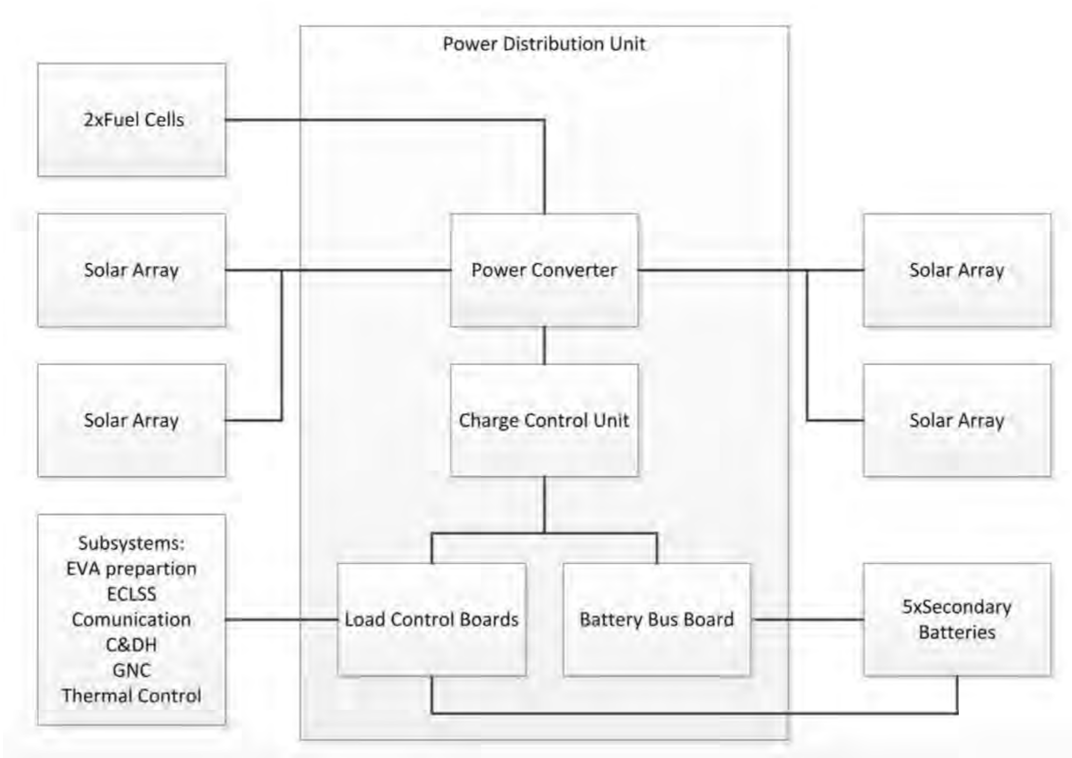


Figure 21: Electrical Power System architecture

5.7.5 Budgets

The EPS designed will be composed of the following elements:

- Dragon capsule EPS (solar arrays, Li-Polymer Batteries, and PDU);
- Fuel Cells;
- Ultra-Flex Solar Arrays;
- Cobalt Li-Ion batteries;
- Power Distribution Unit;

The final mission power budget is presented in Table 32 and the total EPS mass budget is presented in Table 33.

Table 32: Final power budget

Subsystem	Power [W]
ECLSS	5 805
Habitat	0
Command Module	inbuilt
Propulsion	0
GNC	107
TT&C	283
Thermal	808
Scientific Payload	123
10% Margin	713
Average Power Total	7 840
Peak-Power Total	11 760

Table 33: EPS mass budget

System	Mass [kg]
Fuel Cells	32
Fuel Cells Reactants	282.2
Solar Arrays	120
Batteries	172
PDU	65.2
Total	671.4

5.8 Thermal

The Thermal Control System (TCS) is the control of the spacecraft equipment and structural temperature, to ensure that all the components are in their specific operating temperature range during the mission. Besides keeping the temperature within these ranges, the goal is also to minimize the temperature gradients, i.e., keeping a constant temperature.

There are two main reasons to why this subsystem is required, one being the fact that usually the electronic and mechanical equipment only operate efficiently and reliably if they are within relatively narrow temperature ranges. The other is concerned with the thermal expansion coefficients of most material being different from zero, which means that temperature gradients can result in thermal distortion. Another important aspect to take into consideration is that developing and testing new equipment is much easier and cheaper at a room temperature and that most components used in the spacecraft were originally designed for terrestrial use [34].

Based on this, the thermal control is no more than a process of energy management in which the environment around the spacecraft has a critical role. The major forms of environmental heating, directly contributing to the thermal balance, are the direct sunlight, the sunlight reflected off the planet around which the spacecraft is orbiting (planet's albedo), and the infrared energy emitted from the planet (IR). During launch or in exceptionally low orbits, there is also a free molecular heating effect caused by friction in the rarefied upper atmosphere [36]. Typical TCS elements include surface finishes, insulation blankets, heaters, and refrigerators.

5.8.1 Requirements

This subsystem's requirements are directly related to the temperature requirements of the spacecraft's components and are summarized in Table 34. [20]

Table 34: Temperature requirements

	Component	**Typical Temperature Ranges [^{o}C]**	
		Operational	**Survival**
Crew	Crew quarters	18 – 26	15 – 30
	Surface temperature	4 – 40	-1 – 45
	Food Storage	-20 – 4	-25 – 10
Electrical Power	Batteries	0 – 15	-10 – 25
	Power Box Baseplates	-10 – 50	-20 – 60
	Solar Panels	-150 – 110	-200 – 130
Attitude Control	Reaction Wheels	-10 – 40	-20 – 50
	Gyros/IMUs	0 – 40	-10 – 50
	Star Trackers	0 – 30	-10 – 40
Communicaton and Computer	C&DH Box Baseplates	-20 – 60	-40 – 75
	Telemetry & Command units	-10-50	-15-55
Propulsion	Tanks	15 – 40	5 – 50
	Rl-10B2 Engines	-50-150	-55-155
Antennas	Antenna Gimbals	-40 – 80	-50 – 90
	Antennas	-100 – 100	-120 – 120
	Onboard computers	-10 – 50	-15 – 55

5.8.2 Passive Thermal Control

This type of thermal control makes use of materials, coatings, or surface finishes (such as blankets or second surface mirrors) to maintain temperature limits. Because it's much simpler than the active control, the thermal design should be as passive as possible. This means choosing the right materials and coating so that the internal heat can be dissipated and low heat is absorbed from the external environment. Therefore, the habitat should have a highly reflective surface, specially the sun-facing part. Since we don't need to worry about the radioactive protection, because the habitat structures are already radiation proof, we just need a coating that provides a low solar absorptivity (α) and a high emissivity (ε). Accordingly to Kirchhoff's rule, for the same wavelength, α and ε are the same, so it would be impossible to have different values of α and ε. However, in this case, the energy sources are different and so is their wavelength. In thermal control design, α means the absorptance of a surface to solar radiation and is therefore often referred to as the "solar absorptance". In the other hand, ε means the emittance of a surface radiating in the infrared region so it is often referred to as the "infrared emittance" or "thermal emittance". [34]

According to [37], using AZ-70-WIST - Inorganic white, nonspecular thermal control paint/coating, an α = 0.10 $\pm$ 0.02 and a ε = 0.91 $\pm$ 0.02 are obtained. From tests already made, it can be considered that no deterioration of this values will happen. A protective coating should also be used so AZO-5000-PF was chosen. This way, if exposed to atomic oxygen in Low Earth Orbit, this overcoat is etched away and leaves the primary coating intact. Because the inorganic coating may not be compatible with the composite structure, a primer will also be applied before the main coat, in this case a RF primer: MLP-100-AZ Epoxy-based primer paint/coating.

5.8.3 Active Thermal Control

As it was shown in Section 5.8.1, the thermal requirements consist in very narrow temperature ranges, so a complete passive thermal control is not enough to guarantee their fulfillment. ACT techniques can be executed through heaters, louvers, heat pipes, thermoelectric coolers, cryogenic cooler and pumped fluid loops.

Pumped Fluid Loops

PFLs are the most used systems because they can provide efficient transfer of large amounts of thermal energy between two points using liquid convective cooling. As shown in Figure 22, a PFL consists of a pumping device, a heat exchanger and a space radiator. This heat transfer can be accomplished with a coolant, being used as a thermal energy transport agent: the coolant absorbs the dissipated thermal energy from a component, through a heat exchanger, and transfers it to the radiator. The final heat-rejection process depends on whether the coolant is expendable or non-expendable: if the coolant is expendable the working fluid is rejected from the space vehicle once it has accomplished its mission, otherwise the working fluid is recirculated within the system once the thermal energy has been rejected to space by a radiator [36].

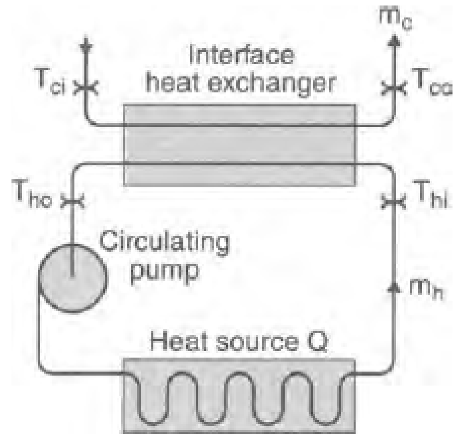


Figure 22: Schematics of a fluid loop [36]

Orbitec, which was subcontracted by Bigelow to design the TCS, proposed an active thermal control achieved through a single fluid loop that utilizes a human-safe fluid, with no heat exchangers. This kind of configuration has advantages as:

- Flexibility in locating heat dissipating equipment inside the spacecraft as well as in the radiator deployment methods: since the thermal loops will be embedded in the vehicle, the placement of radiators will be independent of the internal system's locations;
- Ability of accepting and rejecting heat at multiple locations, resulting from the previous point;
- Allowing for late design changes in the s/c to be easily incorporated;
- Providing easy scalability to meet changes in power dissipation requirements;
- Flexibility of the working fluid based on the required thermal environment.

For efficiency purposes, fluids with high specific heat and low viscosity are desirable because they reduce the required pump power to circulate them. Water was already used in various missions, since it is non toxic, which is important in the case of manned missions, and has one of the highest values of specific heat. However, to keep the water from freezing, electrical heaters must be added. In the periods when the spacecraft is behind Mars, i.e., it doesn't receive any direct sunlight, the fluid velocity can be reduced and the heat pump switched off as long as a thermostat is included. This enables the control of the amount of heat being rejected, which in turn allows the temperature maintenance [39].

Heaters

In this case, the heaters' main function is to keep water from freezing (above 0^{o}C). This components may be in the form of compact metal-mounted resistors, metal co-axial cables in which the heating element forms the core of the cable (thermo-coax) or patch heaters. The patch heaters are the most common and consist of an electrical-resistance element sandwiched between two sheets of flexible electrically insulating material, such as Kapton [36]. For redundancy, multiple circuits can be included in the heaters (so if one fails the rest of them still work) or more than one heater can be used.

Pumps

The pumps are used to ensure the circulation of the fluid in the loop. The mass, power and volume can be estimated using the following relations [38]:

- Mass (kg): 4.8 $\times$ loop capacity in kW;
- Power (W): 23 $\times$ loop capacity in kW;
- Volume (m^3): 0.017$\times$ loop capacity in kW.

In addition to the pumps, the loop also includes plumbing and valves whose mass, power and volume must be estimated as well. For the plumbing and valves an estimated addition of 15% to the total active system mass must be considered. Finally, the fluids represent about 5% of the total mass and must also be taken into account. This same relations can be used to estimate the volume of this components while their power is negligible.

Radiators

Most unmanned spacecrafts have power levels that allow the rejection of heat to be made using a small area of radiators. When this happens, structural-panel or body-mounted radiators can be used. However, as power requirements increase, which results in the increase of waste heat, it becomes more difficult to integrate all the radiator area needed in the spacecraft's structure.

For this mission, not only the power required is very high but also the structure is made of composites and is meant to be inflated in space which complicates even more the incorporation of the radiators in the structure. Because of this, deployable radiators were chosen. From [36], the average weight per m^2 of a deployable radiator can be up to 12 kg, including the support structure.

5.8.4 Thermal Equations

As it was mentioned before, there are four main contributions to the spacecraft's received energy. In this section, the contribution of each one of them is going to be described.

Direct Sunlight

The energy flux received from the Sun in W/m^2, at a certain position, is given by the Equation 4. [34]

$$J_s = \frac{P_s}{4\pi d^2} \tag{4}$$

With P_s being the Sun's total power output (3.856×10^{26} W) and d the distance between the Sun and the spacecraft.

Table 35: Energy flux received from the Sun

	d [km]	$\boldsymbol{J_s}$ **[W/m²]**
Earth	147.14×10^6	1.4173×10^3
Mars	206.72×10^6	7.1807×10^2

Albedo radiation

A planet's albedo coefficient, a, is defined as the fraction of incident solar radiation which is reflected from the planet. This parameter is highly variable so average values will be used. For the Earth an average value of 0.37 is going to be used while for Mars it will be 0.29 [20]. The total energy flux due to the albedo is given by Equation 5.

$$J_a = J_s a F \tag{5}$$

Where J_s is the solar energy flux, a is the albedo coefficient and F is the visibility factor given by Equation 6.

$$F = \frac{1}{2}\left(1 - \frac{\sqrt{H^2 - 2H}}{1 + H}\right) \tag{6}$$

In this case, H is the ratio R/h, with R being the planet's radius and h the altitude of the spacecraft from the planet's surface.

Table 36: Albedo radiation received from the planets

	a	$\boldsymbol{J_a}$ **[W/m²]**
Earth	0.37	1.08×10^2
Mars	0.29	71.52

Planetary Radiation

As mentioned before, the planetary radiation it's the radiation emitted by the planet, usually in infrared. This happens because not all the incident sunlight is reflected as albedo, some being absorbed and eventually re-emitted as IR energy. Also, since the planets have non-zero temperatures, they radiate heat. Due to its relatively low temperature, the Earth radiates all of its heat at infrared wavelengths so, for this reason, this radiation is often referred to as thermal radiation. The intensity of the planetary radiation reaching the spacecraft can be calculated using Equation 7.

$$J_p = p\left(\frac{R_{rad}}{R_{altitude}}\right) \tag{7}$$

Where R_{rad} is the radius of the planet's effective radiating surface (assumed to be the same as the radius of the planet), $R_{altitude}$ is the altitude of the spacecraft from the given planet's center, and p is the average planetary emission (p is equal to 231 W/m^2 for Earth [36] and 141 for Mars [20]).

Table 37: Heat flux due to planetary radiation

	$\boldsymbol{J_{p_{average}}}$	$\boldsymbol{J_p}$ [W/m^2]
Earth	231	186.98
Mars	141	133,91

Internal Heat Production

There are two main sources of heat inside the s/c: the electric systems and the crew's metabolic system, with the current standards for handling metabolic heat production with normal activity being 100 W per person when at rest and about 200 W per person when performing activities [40]. So, since the crew has 4 people, we have a maximum of 800 W of human heat production. From Section 5.7.5, we have that the maximum power is 11 760 W while the average value is 7 840 W.

Thermal Balance

The thermal balance is achieved when the total heat received is equal to the total heat radiated to the exterior of the aircraft.

$$Q_{external} + Q_{internal} = Q_{radiated} \tag{8}$$

With $Q_{external}$ being the contribution from the Sun and the planet, $Q_{internal}$ being the heat produced inside the spacecraft (electronics and metabolism) and $Q_{radiated}$ being the radiated heat. Because the spacecraft is not a black body, it absorbs only a fraction of incident energy and emits a fraction of the radiation of a black body at the same temperature. Due to the coating used in PTC, we have an absorptance, α, of 0.10 and an emittance, ε, of 0.91. The solar contribution is therefore given by Equation 9.

$$Q_{solar} = J_s \alpha_{eff} A_s \tag{9}$$

In this case, since the coating has proven that a minimum degradation of alpha occurs, it will be considered constant meaning that $\alpha = \alpha_{effective}$. A_s is the area of the spacecraft that faces the sun. The planet's albedo contribution can be determined in the same way, using the projected area that receives the albedo radiation:

$$Q_{albedo} = J_a \alpha A_a \tag{10}$$

It was explained before that for the same wavelength, $\alpha = \varepsilon$, so since the planets radiate in infrared, the ε (infrared emittance) will replace α in the Equation 11.

$$Q_{planet} = J_p \varepsilon A_p \tag{11}$$

So we have that $Q_{external} = Q_{solar} + Q_{albedo} + Q_{planet}$. From the Stefan-Boltzmann equation, the radiated power is:

$$Q_{radiated} = \varepsilon \sigma A T^4 \tag{12}$$

With ε being the emittance of the radiator, σ the Stefan-Boltzmann Constant, A the area of the radiator and T the working temperature of the radiator (to be effective this temperature should be much higher that the environment temperature: knowing that the radiator temperature can be increased until 50ºC by the heat pump, a value of 40ºC was used to account for possible inefficiency). Finally, knowing that the radiated power has two components, the power rejected by the radiators and the power rejected by the surface of the spacecraft ($Q_{rad} = Q_{radiator} + Q_{surface}$) and rearranging the equations, we can calculate the needed radiator area.

$$A_{radiator} = \left(Q_{ext} + Q_{in} - T^4_{surface}.\sigma.\varepsilon_{eff}.A_{surface}\right) \frac{1}{\varepsilon.\sigma.T^4_{radiator}} \tag{13}$$

With ε_{eff} being the effective emittance of the surface which is related to its thickness and composition. An estimated value of 0.1 (the same as the absorptance) was used.

5.8.5 Worst Hot Case Scenario

The main task of the thermal control system is to assure that equipment stays within certain temperature limits, as it was seen before. In order to do this, the worst case conditions must be defined. This conditions are typically the orbits with maximum and minimum sunlight combined with extreme spacecraft attitudes and operational modes. Usually, the thermal system is designed for the worst hot case, which means the radiators are dimensioned to reject the maximum power dissipated by the equipment under maximum external loading.

For this mission, the worst hot case scenario is when the aircraft is closer to the sun, i.e., orbiting the Earth, and all the systems are on (peak power). The following data was input in a script meant to compute the expressions described in Section 5.8.4. The results are shown in Table 39.

Table 38: Constants used to determine the radiator area

Constant	**Value**	**Constant**	**Value**
$R_{\oplus}$[km]	6 371	A_s[m^2]	31.17
h [km]	1 500	A_p[m^2]	54.81
α	0.1	A_a[m^2]	54.81
ε	0.91	$A_{surface}$[m^2]	4.22×10^2
σ[W/m^{2}K^4]	5.67×10^{-8}	$T_{surface}$[K]	293.15

Table 39: Worst hot case results

Q_{solar}[W]	4.42×10^3
Q_{albedo}[W]	5.93×10^2
Q_{planet}[W]	9.33×10^3
$Q_{internal}$[W]	12 560
$A_{radiator}$[m^2]	17.97

Considering an efficiency of 80%, the radiators area will be increased by 20% so in the end the total radiator area will be approximately 22 m^2. This area will be divided by the two deployable radiators meaning each one of them will have about 11 m^2.

5.8.6 Worst Cold Case Scenario

In this mission, the worst cold case scenario is when the aircraft is orbiting Mars and an average power is being produced by the internal systems. It will also be considered that the crew is resting. Since we get to Mars in a point of its orbit very far from the Sun, the solar radiation, that was the biggest contribution in the hot case, is very small, as are the other heat fluxes. With this data, we get a radiator area smaller than 1 m^2 which means that if the full area of radiators is used, the temperature inside the spacecraft will be very low, because the radiator will be dissipating all the heat being produced inside.

To solve this problem, the radiators can be painted black on one side so that, when in cold situations, they can be rotated, making that side face the sun and absorb more solar heat. This could be specially useful in the parts of the orbit behind Mars, where J_s is 0. This solution can be used together with the thermostat that regulates the fluid loop.

5.8.7 Budgets

Using a radiating area of about 22.4 m^2 and their average weight, the radiators mass can be calculated. Using a thickness of 0.06 m, we get their volume.

Then, using the total power that we need to dissipate, we can determine the pumps mass. And adding the two results, and using the percentages previously defined in Section 5.8.3, we get the mass and volume for valves and fluids.

Table 40: Mass, volume and power budget for the TCS

Component	**Mass [kg]**	**Volume [m^3]**	**Power [W]**
Radiator	258.80	1.08	-
Pumps	127.54	0.45	611.12
Plumbing and Valves	444.29	1.76	-
Fluids	405.66	1.61	-
Total	1 236.3	4.9	611.12

6 Scientific Experiments

Until now, the maximum period an astronaut has spent in space was approximately one year. Therefore, the effects of a long-term, deep-space mission are still unknown, so this mission will represent the first opportunity to study them. Furthermore, the proximity to the red planet will allow us to map a possible landing site for future missions and to analyze Mars' atmosphere.

6.1 Health Effects of Deep Space Missions

In order to study each crew members' health during the mission, a Crew Health Care System (CHCS) is brought on board [41]. The CHCS is divided into the following three subsystems:

- Countermeasures System (CMS)

  The CMS provides the equipment and protocols for the performance of daily and alternative regimes to mitigate the effects of living in a microgravity environment. The CMS also monitors and evaluates crew members during exercise regimes. [41]

- Environmental Health System (EHS)

  The EHS monitors the atmosphere for gaseous contaminants, microbial contaminants, water quality, acoustics, and radiation levels. [41]

- Health Maintenance System (HMS)

  The HMS provides life support and resuscitation, medical care and health monitoring capabilities. The Medical Equipment Computer not only downlinks data from the medical equipment but also contains physiological monitoring software, an electronic medical record and medical reference software, thus being the platform for the computer-based medical training. [41]

6.1.1 Bone Loss

In a microgravity environment the bones are no longer subjected to the amount of weight and stresses that they are used to on the Earth. Over time, the calcium stored in the bones is broken down and released into the bloodstream, leading to the decrease in bone density [42]. This loss in bone mass weakens the bones, making them unable to support the body's weight and movement when returning to Earth, thus increasing the risk of fracture. The rate of density loss is approximately 1-2% per month, while, by comparison, the rate of bone loss for elderly men and women on Earth is from 1-1.5% per year [49]. Bisphosphonates drugs have shown to be effective in preventing bone loss so they will be used during the mission [43].

6.1.2 Muscle Loss

It is well known that muscle mass decreases during space flight due to the lack of demanding activities and from the unloading of postural muscles. Besides affecting the astronauts' performance, muscle atrophy also increases the risk of severe muscle injury upon the return to Earth. Without exercise, on long duration missions, astronauts could lose up to 50% of their muscle mass, some of it permanently. [44] The Countermeasures System is composed by the following exercise equipment: [45]

1. Cycle Ergometer - A bicycle where the main activity is pedaling. It is used to measure fitness in space because it's easy to check heart rate and how much work is being done.

2. Treadmill - Walking is the most important way to keep bones and muscles healthy. With the lack of gravity, harnesses are attached to the astronauts to hold them to the walking surface.

3. Interim Resistive Exercise Device (IRED) - The IRED looks like weight-lifting machines. To use it, astronauts pull and twist stretchy rubber-band-like cords attached to pulleys. The IRED can be used for a total body workout. From squats and bending exercises for the legs, to arm exercises and heel raises, astronauts can do them all on the IRED.

Along with the above equipment, the Advanced Resistive Exercise Device (ARED) will also be aboard the spacecraft. The ARED uses a piston and flywheel system to simulate free-weight exercises in normal gravity, to work all the major muscle groups through squats, dead lifts and calf raises. Investigation regarding this equipment suggests that it could be an effective countermeasure against loss of conditioning during spaceflight. [46]

6.1.3 Isolation & Confinement

When people are isolated with others in a small space over a long period of time, the emergence of behavioral issues among them is inevitable. Sleep disorders may be developed due to an unbalanced circadian rhythm, the noisy environment or the stress of prolonged isolation and confinement. Depression can also occur and fatigue is inevitable, since there will be times with heavy workload and shifting schedules. Also, periods of monotony may lead to boredom. Misunderstandings and impaired communications with other team members might impact performance and mission success. Finally, the lack of fresh food and variety, or deficiency in nutrition, may contribute to physiological and cognitive decrements. [49]

In order to tackle some of these issues, the spacecraft will carry devices like actigraphy, that helps astronauts to assess and improve their sleep and alertness, by recording how much they move and how much ambient light is around. LED technology will also be used to help the alignment of the circadian rhythms which will improve sleep, alertness and performance. [49]

6.1.4 Radiation

Radiation exposure increases cancer risk and it can damage the central nervous system, with both acute effects and lifelong consequences. The symptoms range from altered cognitive function to reduced motor function. Space radiation can also cause radiation sickness that results in nausea, vomiting, anorexia, and fatigue. The crew members could also develop degenerative tissue diseases such as cataracts, cardiac and circulatory diseases. The food available and the medicine the astronauts take must be safe and retain their nutrient and pharmaceutical value, even while being bombarded with space radiation. Thus, shielding, monitoring and operational procedures is all it can be done to control the radiation risks within acceptable levels, to keep them safe. [49]

6.2 Mars Science Mission

In this mission the main scientific goals while on Mars are mapping its surface, searching for possibly landing sites for future missions and analyze the planet's atmosphere. In order to successfully accomplish these objectives a number of instruments is carried onboard.

1. Two narrow angle cameras and a wide angle camera will be carried. The two narrow angle cameras will provide extreme close-up images of Mars' surface, whereas the wide angle system will image Mars' surface at seven different wavelengths, to characterize the distribution of resources. [50]

2. A system constituted by a dual-lens camera head and a digital electronics assembly. The camera head will acquire the images while the digital electronics will buffer and compress the images for transmission to Earth. The system acquires images at 2 visible wavelengths and 2 ultraviolet wavelengths. The images will be used to document the weather on Mars, by observing the variations of dust storms, dust devils, polar frost and clouds of water vapor, water ice and carbon dioxide crystals. The ultraviolet observations will map the distribution of water vapor and ozone in the atmosphere. In addition, these images will provide data on the growth and retreat of the polar caps.

3. A Thermal Emission Spectrometer, that will acquire thermal infrared observations of Mars. These will be used to map mineral occurrences, infer surface physical properties and examine atmosphere/weather conditions. [52]

4. A laser altimeter to calculate spacecraft's altitude above the local terrain. [53] As the spacecraft flies above hills, valleys, craters, and other surface features, its altitude above the ground constantly changes. A combination of the laser altimeter data with images from the cameras will allow to construct a detailed topographical atlas of the planet.

5. A Gas Chromatograph and Mass Spectrometer (GCMS) that will measure the chemical composition of Mars' atmosphere and determine the isotope ratios of the major gaseous constituents. [54]

Table 41: Scientific payload mass and power

Scientific instrument	Mass [kg]	Power [W]
Narrow angle camera	22 ($\times$2)	14 ($\times$2)
Wide angle camera	13.3	14
Wide angle spectral camera	17.7	14.5
Thermal Emission Spectrometer	14.4	14.5
Laser altimeter	23.8	34.2
Gas Chromatograph and Mass Spectrometer	12.2	18
Total	**152.4**	**123.2**

7 Crew Safety & Health

7.1 Crew Selection

In order to select the right crew for the mission, some requirements must be fulfilled. The astronaut candidates must have a bachelor's degree from an accredited institution in engineering, biological science, physical science or mathematics. This degree must be followed by at least three years of professional experience or, in case of military candidates, at least 1000 hours of pilot-in-command time in jet aircrafts. [55] The candidates also have to pass a thorough physical examination, with the following conditions:

- Distant and near visual acuity must be correctable to 20/20 in each eye, with the use of glasses being acceptable as well as refractive surgical procedures of the eye;
- The blood pressure must not exceed 140/90, measured in a sitting position;
- The applicant must be free from any disease and any dependency;
- Normal range of motion and functionality in all joints;
- Free from any psychiatric disorders;
- Be healthy, with an age and gender adequate fitness level;
- The candidates must have a standing height between 157 and 190 cm. [55][56]

Since all crew members will fly aboard the spacecraft vehicle and perform Extravehicular Activities (space walks), applicants must meet the anthropometric requirements for both the vehicle and the extravehicular activity mobility unit (space suit).

The candidates will undergo a training and evaluation period lasting approximately 2 years. As part of the training program, the candidates are required to complete military water survival before beginning their flying syllabus, and become scuba qualified as preparation for spacewalk training.

Candidates are also exposed to the hyperbaric and hypobaric environments in altitude chambers in order to learn how to deal with emergencies associated with these conditions. In addition, they will experience microgravity during flights in a modified jet aircraft, as it performs parabolic maneuvers that produce periods of weightlessness of about 20 seconds. The aircraft then returns to the original altitude and the sequence is repeated up to 40 times in a day. The candidates must also successfully complete the following tasks: Spacecraft systems training, Extravehicular Activity skills training, Robotics skills training and aircraft flight readiness training. [55]

Astronauts are also required to have a detailed knowledge of the spacecraft systems which add more 2 to 3 years of training beyond the initial training and evaluation period.

Besides the physiological testing, a personality assessment has to be done as well. The primary personal attributes of a successful astronaut are emotional and psychological stability, supported by personal drive and motivation. Important traits include curiosity, adaptability, resourcefulness and resilience, among others [56].

7.2 Crew Activities

Isolation has serious effects on both psychological and physiological terms. This has consequences on the crew such as stress, hormone regulation, sleep quality, mood and the effectiveness of dietary supplements. Therefore, evaluations and monitoring activities are of the utmost importance, and can be divided in the following categories.

- Medical/Clinical Evaluations

  Periodic inflight medical evaluations include crew conferences with ground specialists. Inflight examinations are performed by the designated Crew Medical Officer and include radiation monitoring and physical fitness tests. [41]

- Occupational Monitoring

  Occupational monitoring of the space flight environment is performed on a regular basis and includes monitoring radiation, air and water quality. Biodosimetry data is collected before and after flight to determine whether or not aberrations in chromosomes have occurred. [41]

- Physical Fitness Evaluations

  A submaximal, incremental load test is performed periodically using the cycle ergometer to assess aerobic capacity. Strength and conditioning of the crew are also monitored in association with inflight crew exercise protocols.[41]

- Nutritional Assessment

  Preflight and postflight nutritional assessment includes determination of typical dietary intake using a questionnaire, blood and urine analysis, as well as body mass and composition measurements. Inflight, dietary intake is monitored and body mass measurements and blood chemistry data are obtained on a periodic basis. [41]

- Psychological/Behavioral Health Status

Assessment of crew members behavioral health status is primarily done by interviews with a psychiatrist or psychologist. During the mission, there are regularly scheduled private psychological conferences, monitoring of mood and evaluation of work/rest schedules. [41]

The crew must have a restricted schedule in order to be occupied sufficiently to keep them motivated and to avoid depression. A typical daily timeline is presented in Table 42.

Table 42: Standard crew day [57]

Time	Activities
06:00	Crew wake, clean up, eat, read news and messages uplinked overnight
07:30	Morning Daily Planning Conference
07:55	Work prep
08:15	Crew available work time
13:00	Lunch
14:00	Crew available work time
18:15	Evening work prep
19:05	Evening Daily Planning Conference
19:30	Dinner and relax time
21:30	Crew sleep

During the Daily Planning Conference the astronauts sync up with the ground base before executing their day. The reviewing of procedures to support the day's activities is done during the work prep time. During the available work time the crew performs the scientific experiments, preventive and corrective maintenance, stowage operations, environment sampling and medical analyses.One of the most important activitiesis the two and a half hours of exercise, which help fighting muscle and bone mass loss. During the evening the crew can review procedures and the timeline for the following day.

8 Cost Analysis

Cost estimation is always a complex task. The different parts and stages of the design are sometimes developed to specific missions and their cost is difficult to determine, specially for a first of a kind mission. However, the widespread use of off-the-shelf components, which have listed prices, facilitates a preliminary cost estimation.

There are different models to estimate the total mission cost. In this project, the Combined Method Cost Estimation will be presented, followed by the presentation of previous similar missions. Finally, a market research will be made, including the costs of other missions to Mars.

8.1 Combined Method Cost Estimation

The Combined Method Cost Estimation is a combination of the Cost Estimation Relationships (CERs) [20], and of already available cost systems, like the Falcon Heavy launch.

The CERs used are derived from historical data, including satellites, statistical frameworks and error models. These include the research, development, testing and evaluation phases for each subsystem, being the cost estimation in dollars for the FY2000$.

Table 43: Inflation factors relative to 2000 [20] and dollar to euro conversion at 31/12/2016

Fiscal Year (FY)	**Inflation Factor to Base Year 2000**
2000	1.000
2016	1.396

US$	**€**
1.00	0.948

Using the conversion data in Table 43, the total cost can be estimated. The results are presented in Table 44 and Figure 23 presents the distribution of cost.

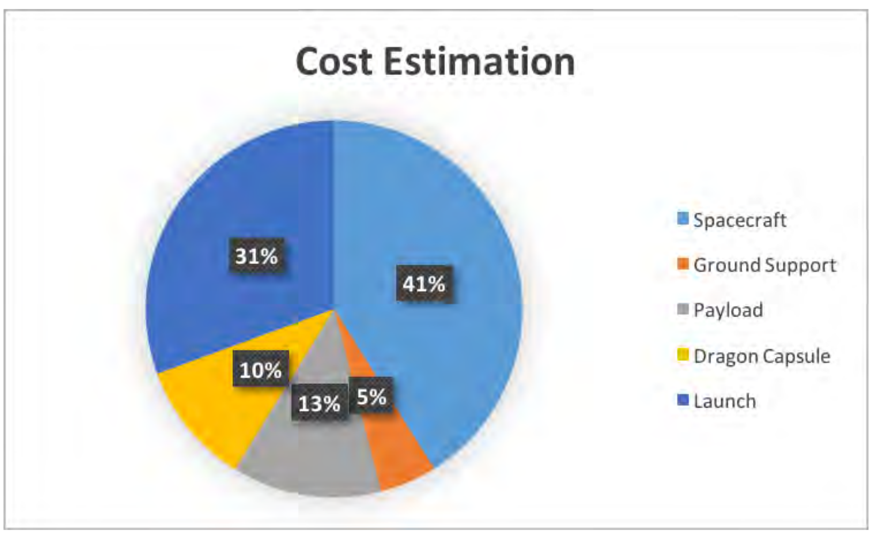


Figure 23: Pie chart of cost estimation

Table 44: CAMOES' mission cost estimation

Mission Segment	FY2000 [M$]	FY2016 [M$]	FY2016 [M€]
Spacecraft	-	544.11	515.82
Structure	113.97	159.10	150.83
ECLSS	224	312.70	296.44
Thermal	3.76	5.25	4.98
Power	9.18	12.82	12.15
GNC	3.00	4.20	3.98
Communications	12.92	18.04	17.10
Propulsion	-	32	30.34
Payload	-	170	161.2
Scientific Equipment	-	170	161.2
Dragon Capsule	-	140	132.7
Launch (Propellant + Service + Fees)	-	405	383.94
Launch 1: Habitat + ECLSS	-	135	127.98
Launch 2: Fuel + Fuel Tanks	-	135	127.98
Launch 3: Engines + Capsule + Crew	-	135	127.98
Ground Support	46.52	64.94	61.56
Total Estimated Cost	-	1 324.05	1 255.18

8.2 Market Research

In this section, a brief description of previous Mars missions will be presented.

- Mars Direct

  Mars Direct is a sustainable human-to-Mars mission, that thrives on a maximum results/minimum investment principle. This concept achieves its low cost objective in two ways: by using proven technology, adapted for the specifics of a Mars mission, and by generating rocket fuel on the surface of Mars, for the return mission. The design drastically lowers the amount of material to be launched from Earth to Mars, thus reducing the mission cost.

  Mars Direct would cost roughly $30 billion.

- Mars Science Laboratory

  Mars Science Laboratory is a mission designed by NASA, in which a large and mobile laboratory, the rover Curiosity, was sent. Using precision landing technology, it allowed the rover to land on the most intriguing regions of Mars for the first time. The rover was safely delivered to the surface of Mars and immediately began sending back stunning images and science data.

Mars Curiosity Landing costed $2.5 billion.

The differences in cost estimates are mainly due to the amount of new equipment and technology, which must be developed specifically for a given purpose. Other Mars exploration programs have had very high costs, usually associated with the development of new technologies and infrastructures such as assembling a large spacecraft on space or using advanced propulsion systems.

8.3 Cost Estimation Comparison

The combined method cost estimation uses a combination of bottom-up CERs approach with already priced parts or stages. A detailed bottom-up method was used to determine the costs of most subsystems presented in Table 44. Theoretically, a combined method cost estimations is more precise, because the cost of each sub-system is determined and then added to the cost of the others.

In this case, the total estimated cost is about 1 110 M$, which is well below the 5 000M$ initial budget.

When comparing the CAMOES mission with the Mars Direct mission, some similarities can be noted. First of all, both missions aim at a manned spaceflight to Mars and a return back to Earth after a period of time. Also, both use proven technology, as a way of reducing costs. The Mars Direct mission cost is higher because it includes a Mars' landing, which directly affects the mission cost as new equipment and technologies are required.

While CAMOES mission focuses on using proven technologies, the Mars Science Laboratory relied on new innovative technologies, specially for the landing. This fact made the Curiosity Landing a more costly mission because it was the first time that such technologies were sent to space.

The total estimated cost might change during the mission design. It can either decrease, from removing unnecessary margins, or increase, due to unforeseen elements of the mission.

9 Mission Analysis

In this section, the mission final budgets, Δv and mass, will be presented. The margins given are also going to be discussed, since they are important to guarantee the mission success and safety.

9.1 Δv Budget

The total Δv is presented in Table 45, as this value will influence not only the design of the mission but also the mass budget.

Table 45: Final Δv budget

Stage	**Δv [km/s]**
TMI	3.467
TCM - to Mars	0.07984
MOI	1.3081
TEI	1.9411
TCM - to Earth	0.066
TCM - Re-entry (Dragon)	0.400
Total Δv	7.262

9.2 Mass Budget

In Table 46, the subsystems mass is presented, as well as the fuel mass and the total result. Also, a percentage of the dry mass is given to the power wiring. This value can vary between 1% and 4%.[20] In this case, a value of 4% was chosen, taking into account the duration of the mission, which will lead to possible degradation with time, and the vulnerability of this components to radiation and other perturbations. For the fuel mass calculation, a margin of 10% was given both to the Δv needed and to the dry mass.

Table 46: Final mass budget

Subsystem	**Mass [kg]**
ECLSS	9 846
Habitat	8 700
Command Module	6 000
Propulsion	1 108
GNC	12
TT&C	66
Power	671
Thermal	1 236.3
Scientific Payload	152
Dry Mass	27 793
Wiring (4%)	1 111.7
Total Dry Mass	28 905
Fuel Mass	87 467
Total Wet Mass	116 372

9.3 Timeline

Figure 24 represents the timeline of the mission, from the first launch to Earth's re-entry, to clarify the order of events.

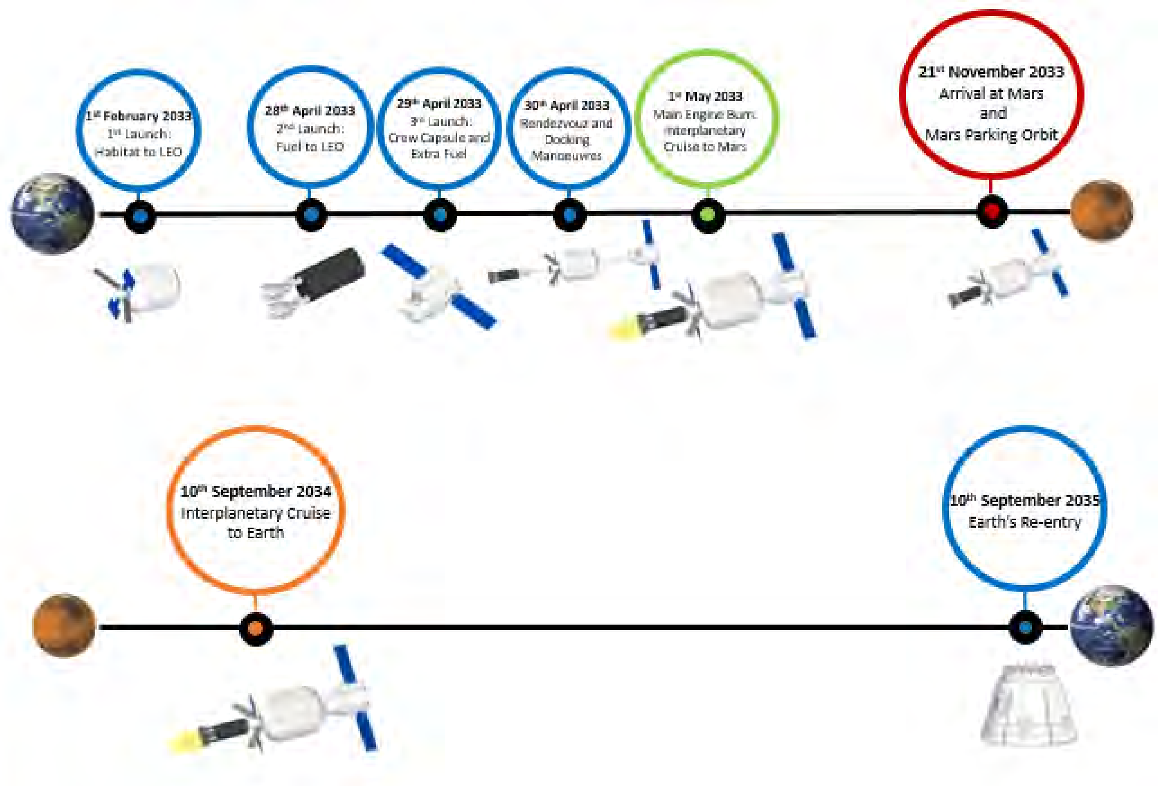


Figure 24: Timeline

10 Risk Assessment

10.1 Safety and Mission Success

The first step in the risk assessment process is to establish what defines the mission success and to set the safety goals of the mission:

- Mission success: Send a crew of 4 members to a Mars orbit and return them safely to Earth.
- Safety goals: Identify all possible safety hazards and eliminate/control them to an acceptable level during all phases of the mission.
- Probabilistic goals (overall risks): Human space flight statistics show a 5% risk of losing the crew. Any next-generation system for transporting astronauts to Mars will probably be designed to a risk requirement much lower than that, e.g. 0.5%.

10.2 Mission Factors/Issues:

Throughout the mission design the following factors are important:

- Mission abort and rescue capabilities: Acceptable risk can be achieved if abort options are included in the mission;
- Greater reliability and/or redundancy of systems, e.g., Common Mode/Common Cause failures;
- Preventive and/or corrective maintenance strategy, e.g., robotics, spares, aged equipment control, caution and warning systems;
- Capability to monitor, detect and assess effects of slow events such as: fatigue, crack formation, dust, corrosion and rust;
- Cabin atmosphere toxicity, contaminant and hazardous substances concentrations are potential toxic threats to all the subsystems of the spacecraft, for example, for the water and air recycling systems or for the insulation of wires.
- Protection against space radiation, effects of gravity force changes and physiological/psychological risks of extended confinement, as well as pathologies and relevant medical care is a concern.

10.3 Technical Risk Assessment Scope

Within the risk assessment process, available risk information is produced and structured, facilitating risk communication and management decisions. The results of this assessment and the residual risks are then communicated to the project team.

Since this is a very preliminary top-level analysis, aimed at identifying first-risk trends, the following factors will not be assessed:

- Earth operations and software risks;
- Legal and programmatic risks;
- Human errors.

Assessment Process

The process must follow the presented steps:

1. What can go wrong? - Identification of hazardous and failure conditions;
2. When can it go wrong? - Identification of failure scenarios and their consequences;
3. What if... - Categorization of the scenarios according to their consequence;

4. How likely? - Analysis of likelihood and uncertainties of risks;
5. Identification and ranking of risk contribution of individual scenarios.

10.4 Abort Possibilities

Thorough investigations of martian mission risks have not yet been performed. As it was mentioned before, acceptable risks can be achieved if abort options are included for all mission phases. The abort option requirement eliminates mission profiles involving very fast and energetic trajectories, as shown in Table 47.

Table 47: Abort possibilities

Phase	Abort Options
Earth departure	Return to Earth possible
Early part of transfer to Mars	Quick return to Earth usually possible for about the first 30-75 days
Later part of transfer to Mars	Mars swing-by (gravity assist) to return to Earth via opposition-like trajectory*
Mars Orbit	Early: Return to Earth opposition-like trajectory Later: Wait for normal Earth return opportunity**
Trans-Earth injection	No practical abort if main propulsion fails
Transfer to Earth	Continue normal return to Earth

* In some cases, propulsion is required at Mars to reach the Earth return trajectory, a deep-space burn may be required.
** Providing the mid-flight maneuvers to return to Earth on an opposition-like trajectory causes a severe initial mass penalty for some Mars launch windows.

Table 48: Technical risk assessment

Risk	Result	Likelihood	Mitigation
Interior equipment fire	Potentially catastrophic equipment damage, decrease in air quality	2	Fire suppression, insulate heat generating components
Habitat failure	De-pressurization, catastrophic failure	1	The habitat has built in safeguards
Reaction thruster failure	Loss in one thrust vector	2	Can be compensated by combination of 17 other thrusters
Engine out	Functionality unhindered	3	SpaceX standard procedure
Launch delay	Launch 1 or 2: Intervals account for minor delay; Launch 3 Possible missed launch window, mission scrubbed	2	Take advantage of first possible launch 3 window; develop contingency launch plans
Orbit/trajectory insertion failure	Mission scrubbed	2	None
TMI booster ignition failure	Mission scrubbed	2	None
Solar panel failure due to debris	Power loss, potential mission failure	1	Rank vital systems to distribute remaining power
Power delivery failure in one solar panel wing	Reduced power generation capability	2	Solar generation from one "wing" can be routed through the other, adding redundancy
Emergency medical event	Crew death/mission failure	3	Emergency handbook, tools and surgical enclosure; extensive training
Life support failure	Equipment damage/loss, personnel damage/loss, total mission loss	2	More Redundancy in Mechanical Components; Spare Parts; Routine Maintenance and Inspection During Transit

10.5 Risk Analysis

The purpose of this phase is to analyze the acceptability of risks and the available risk reduction options, according to the risk management policy, thus choosing the appropriate risk reduction strategy.

The results of the preliminary technical risk assessment indicate where the first risk reduction efforts should be made. Main risk contributors at this stage are shown in Figure 25, with risks related to the crew being the maximum concern. Human factors are therefore extremely important for the mission success. Large uncertainties exist in this context regarding physiology and psychology of the crew due to the lack of previous experiences. However, with the availability of more information and the optimization of the vehicles design, particularly with reference to failure detection, warning, caution and the recovery systems definition, this process is expected to be improved.

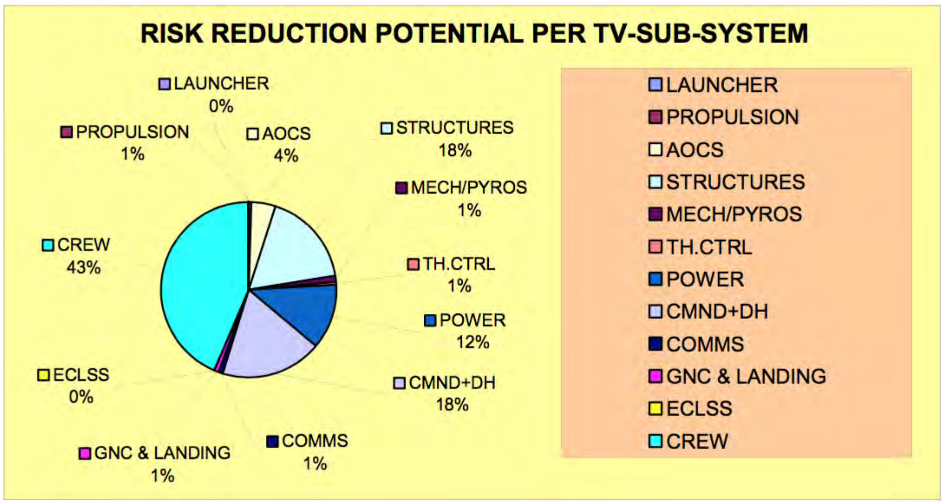


Figure 25: Risk reduction potential

11 Conclusion

CAMOES will be the next step towards setting a footprint on Mars. The mission uses a combination of existing technologies to lower the cost, while at the same time complying with the requirements. To increase the sustainability of the mission and to also provide activities for the crew, scientific payload is taken on the mission. These aspects will all contribute to completing the end-to-end orbital mission to Mars.

A feasible trajectory was designed and optimized, which resulted in a departure on the 1^{st} May 2033. The mission will take a total of 882 days to be performed, with arrival at Mars on the 21^{st} November 2033 and return to Earth on the 10^{th} September 2034, re-entering on the 10^{th} September 2035. The mission will require three launches: the first will take the habitat and its subsystems into the parking orbit, the second one will carry the fuel and fuel depots and the third launch will take the crew, inside the dragon capsule, along with extra fuel to that same orbit. Then, it's in this parking orbit that the Dragon capsule will rendezvous with both the fuel depots and the already inflated habitat. CAMOES will therefore consist of a Dragon re-entry capsule, an inflatable habitat and the propulsion module. Then, the RL-10B2 engines will be ignited to execute a Trans-Mars Injection.

The Environmental Control and Life Support System of CAMOES consists of an advanced water recycling system which is highly efficient in reducing the payload mass. The water tank will be directed towards the sun at all times to further shield the crew from Solar Particle Events. To achieve this pointing accuracy, the autonomous GNC system will use a combination of thrusters and reaction wheels. The propulsion system designed for the spacecraft consists of 4 RL-10B2 engines and 30 thrusters inbuilt in the Dragon. The Telecommunication system will use X-band and Ka-band to be able to communicate both with the ground segment and with the TDRS. These systems will be operated by the Command and Data Handling system.

To provide enough power, the Electrical Power Subsystem consists of fuel cells (mainly used during Low Earth Orbit), solar arrays, secondary batteries (used during Interplanetary Trajectory) and a power distribution and control unit. The Thermal Control System consists of passive and active systems to keep the temperature of CAMOES within the required values.

During the mission, scientific payload will be used to increase CAMOES's return on investment, mapping a possible landing site for future missions and analyzing Mars' atmosphere.

Finally, it was determined that a skip re-entry in the Earth's atmosphere will be performed using the Dragon capsule. After the crew's landing on the surface, further medical tests will be made as well as a further analyze to the data collected during the mission, thus continuing the mission return, even after it is completed. An increase in knowledge about the red planet, as well as about human health, is therefore expected and specially desirable for the future of human space exploration.

Appendices

A Functional Flow Diagram

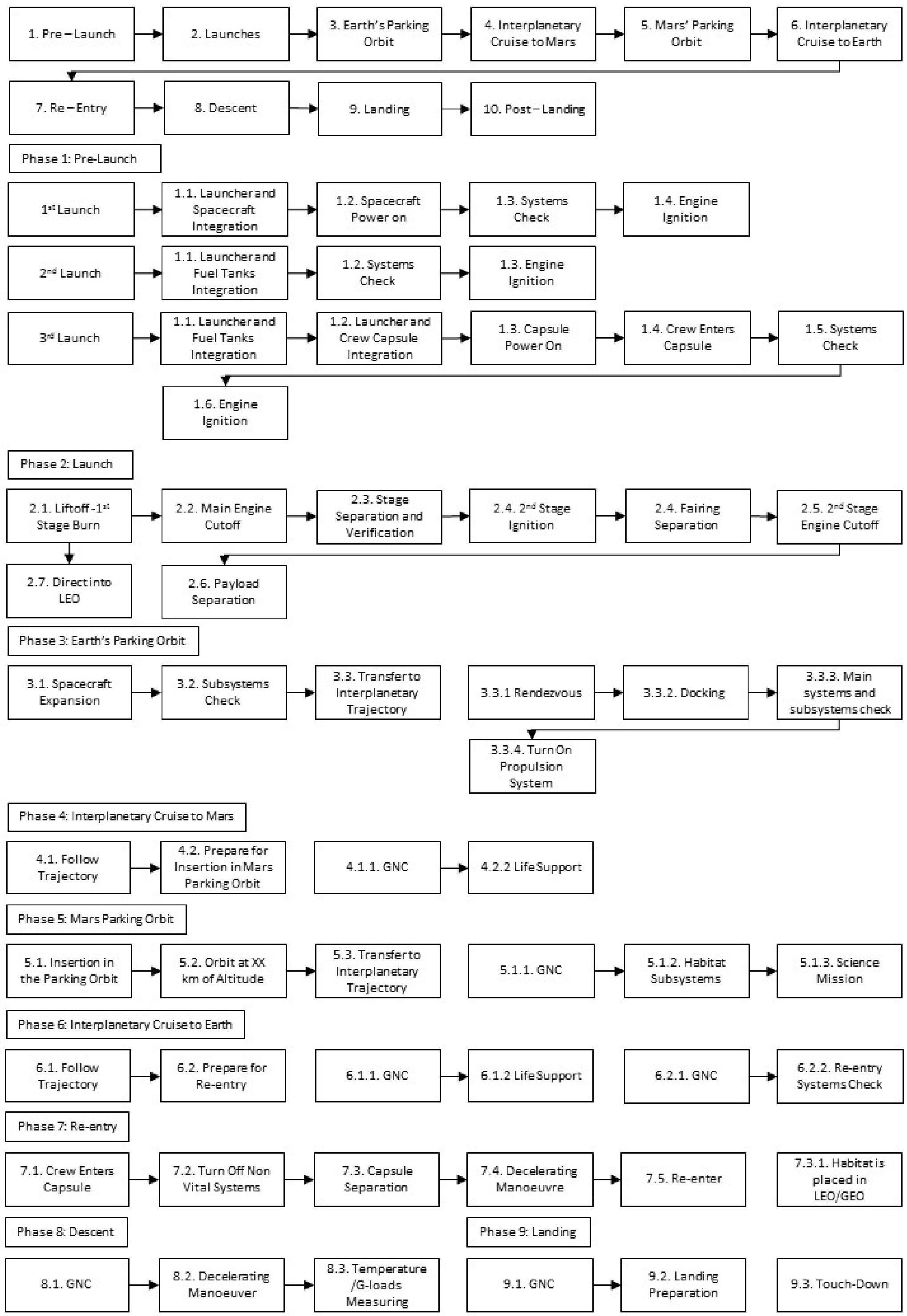


Figure 26: CAMOES Functional Flow Diagram

B Command Module Characteristics

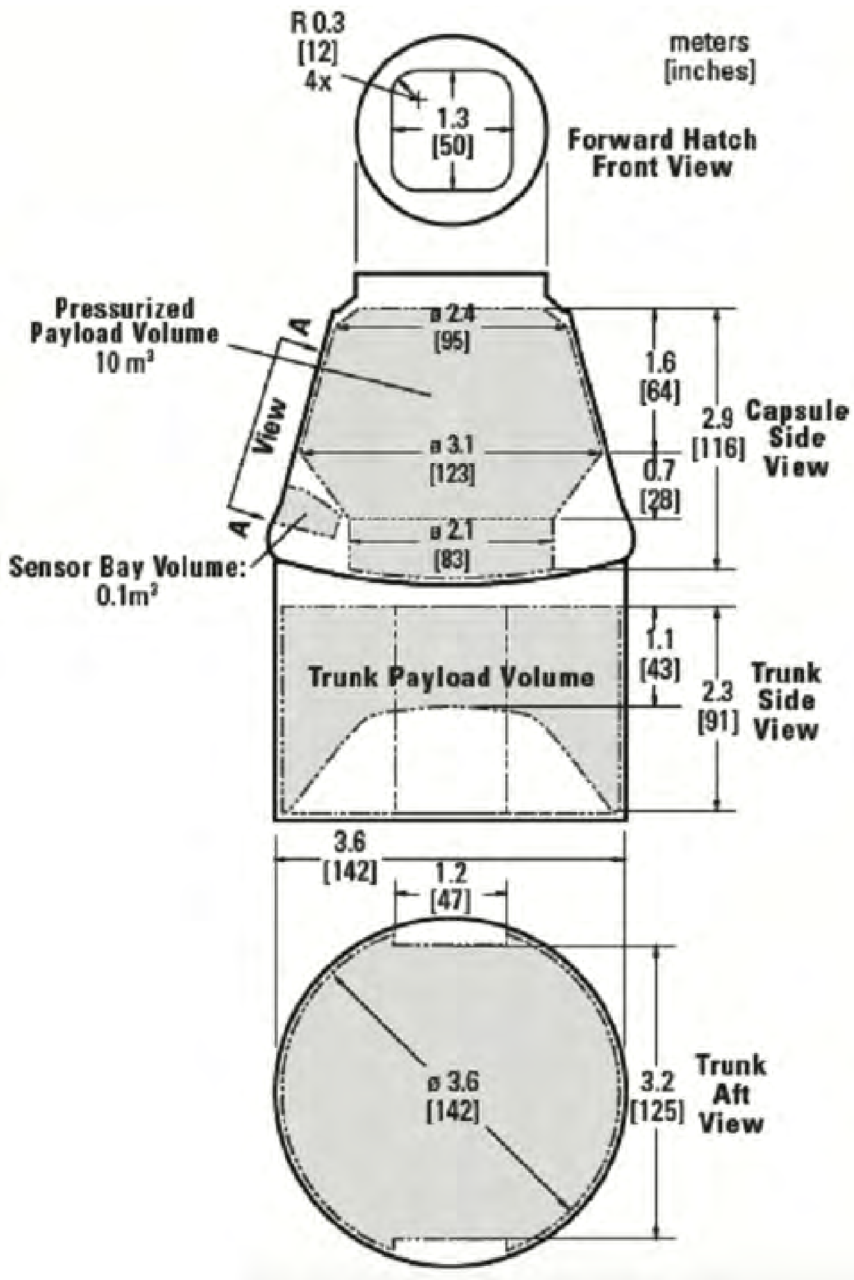
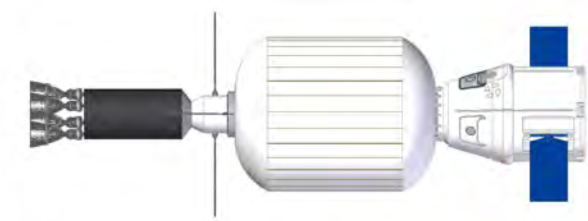


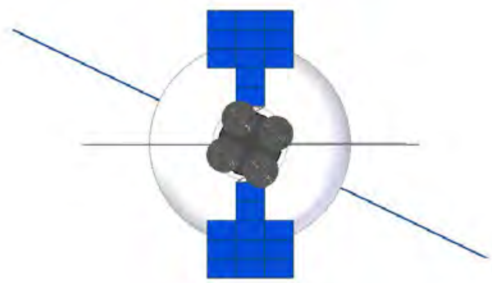
Figure 27: Dragon capsule properties[60]

C 3D CAD Model of the Spacecraft

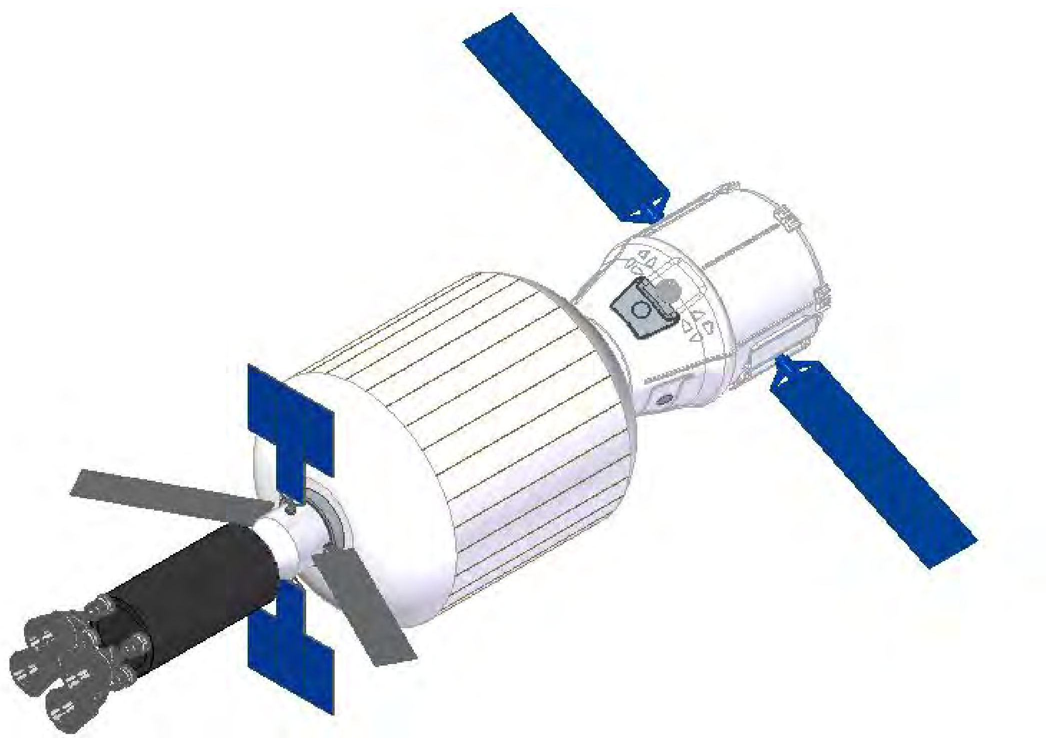
The 3D CAD model of the entire spacecraft are presented in Figure 28. Note that both the Dragon Capsules and the engine models were obtained from *GrabCad*. [61] [62]



(a) Side view



(b) Back view



(c) Isometric view

Figure 28: Different views of the Spacecraft 3D model

D Rendering of the Spacecraft

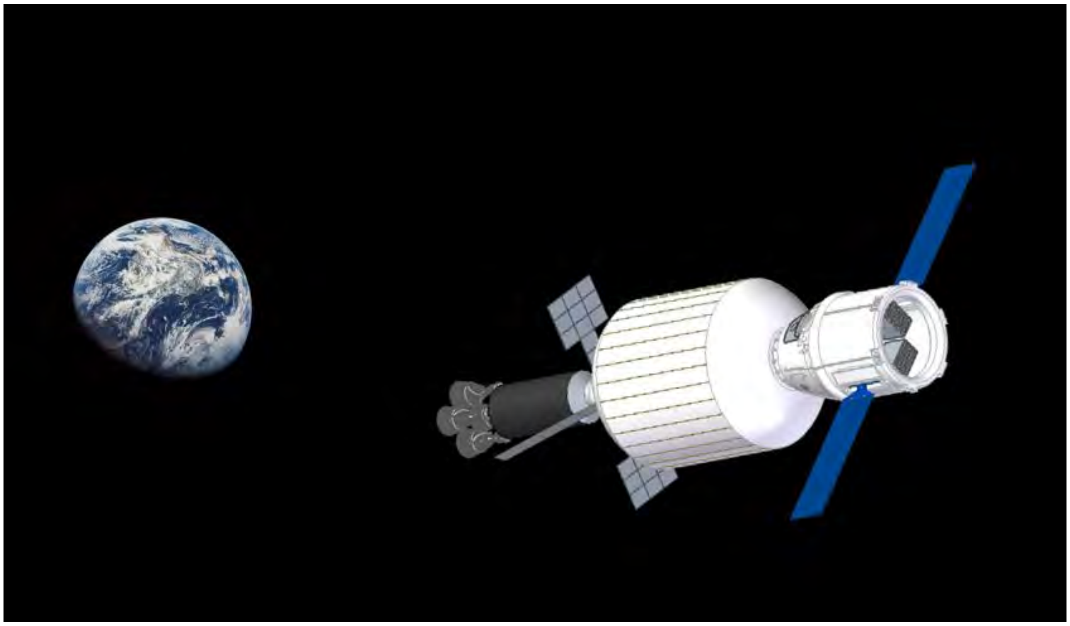


Figure 29: Rendering of the spacecraft after the departure (Earth picture from NASA)

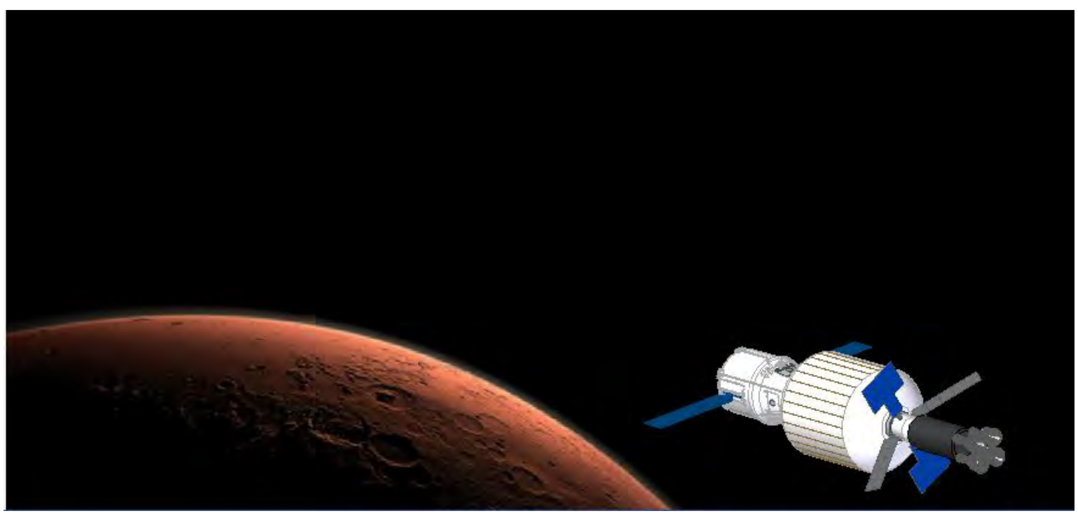


Figure 30: Rendering of the spacecraft's approach to Mars (Mars picture from REUTERS/NASA/JPL-Caltech/Handout)

E Requirements

Table 49: Requirements structure document

REQUIREMENTS	PLACEMENT
Operations concept Macro description of the mission	**Subsection 2.1** pag. 6
Select vehicle architecture	**Chapter 5** pag. 22
Science mission objectives	**Subsection 6.1** pag. 72
Trade study of mission Perform studies at architecture level	**Chapter 5** pag. 22
Planned science approach Planned observation and collection, instruments	**Chapter 6** pag. 71
ECLSS Design and define	**Section 5.1 - 5.2** pag. 22
Communication system Space and ground segment architecture	**Section 5.6** pag. 48
Design mission operation Define each step of the whole mission	**Chapter 3 - 4** pag. 10
Define ground segment	**Subsection 5.6.2** pag. 49
Safety and reliability considerations	**Chapter 7 - 10** pag. 75, pag. 84
Cost Entire mission, including launch	**Chapter 8** pag. 78

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