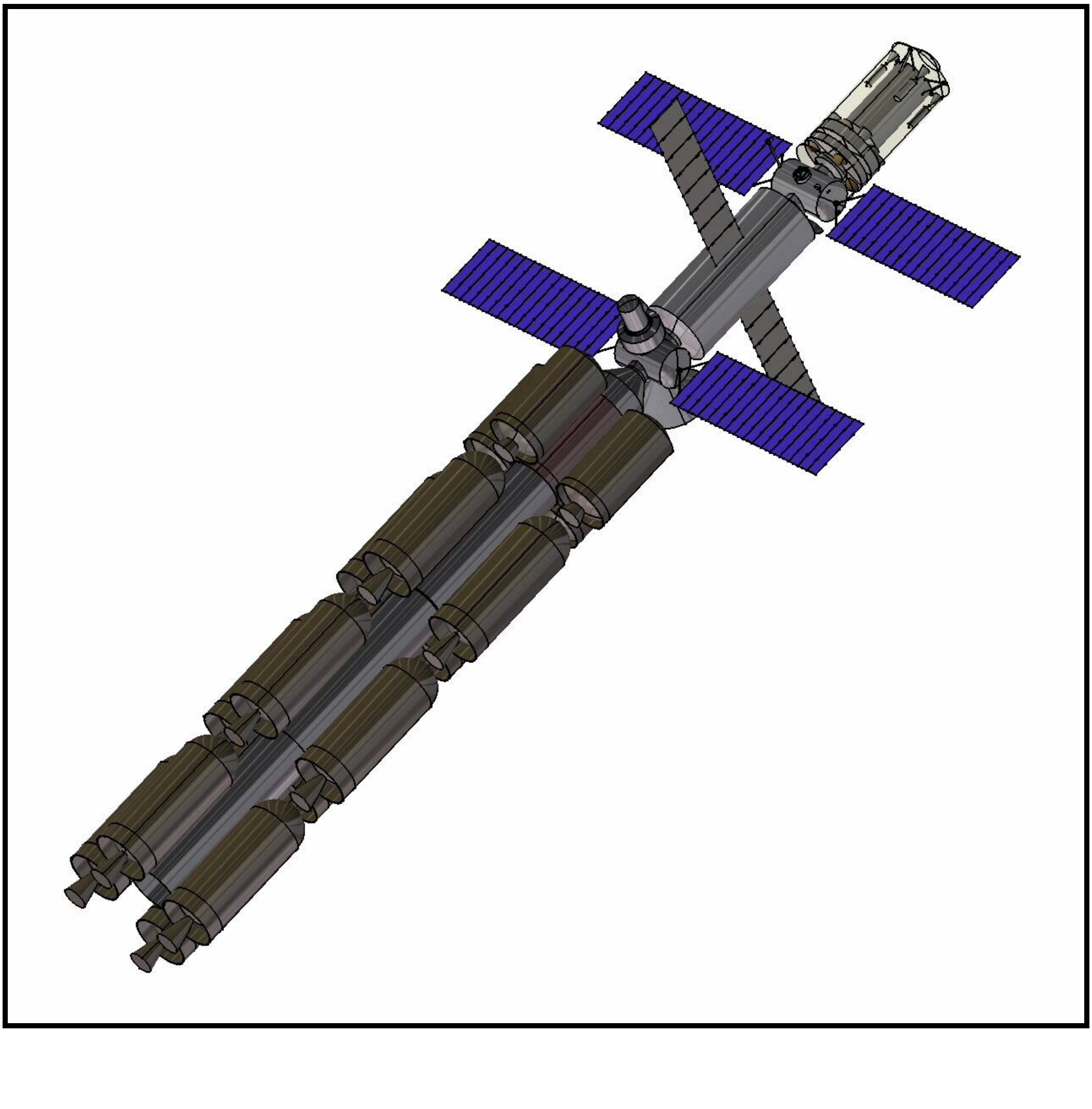
CDF Study Report

Human Missions to Mars

Overall Architecture Assessment



FRONT COVER

(The front cover shows schematic of the complete vehicle for a human mission to Mars)

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1 INTRODUCTION

1.1 Background

In November 2001, the ESA Council at Ministerial level approved the Aurora Programme dedicated to the human and robotic exploration of the Moon, Mars and asteroids. The ultimate goal of the Aurora Programme is human exploration of Mars foreseen in 2025-2030 time frame.

A first ESA activity performed with industrial support concerned high level trades and defined some of the mission boundaries and constraints.

Subsequently, the Aurora Programme asked for further internal activities to refine and detail the above trades.

The main objective of the study was *not* to define an ESA “reference human mission to Mars” but rather to start an iteration cycle which should lead to the definition of the exploration strategy and the associated missions and the set-up of requirements for further mission design and further feedback to the exploration plan. The mission shown in the report is a design case, and does not address prior technology demonstration missions. These are covered separately by the Aurora Long Term Plan.
Peer reviews of the design case have been performed in ESTEC and reports are available as separate documents.

1.2 CDF study

Within the time frame September 2003-February 2004, in two phases, for a total of 23 sessions, the ESTEC Concurrent Design Facility (CDF) performed an assessment study of a Human Mission to Mars, known as the HMM study.
The results of this study are reported in this document.

1.3 Document structure

This document is structured so that the background to the study is described first, followed by an executive summary that gives an overview of the mission. This is followed by the chapters introducing the mission analysis and the mission architecture trade-offs.
The actual subsystem designs that were performed in the study were split into two main separate vehicles: the Transfer Vehicle (TV) and the Mars Excursion Vehicle (MEV). In turn, these two vehicles are split into their main subvehicles/components.
The document includes the chapters relevant to some other overall disciplines such as operations, cost, risk, programmatics and simulation.
Due to the different distribution requirements, only the cost assumptions (excluding figures) are given in this report. The costing information is published in a separate document.

2 GENERAL ARCHITECTURE

2.1 Study objectives

A human mission to Mars is possibly the most ambitious space mission to undertake. Even more so, when it needs to be linked to an overall planet exploration programme that may involve several expeditions and long permanence on the surface.
A consistent long-term plan needs to be elaborated considering all the technological, programmatic and cost aspects.
Preparation of such long-term plan and associated missions requires deep understanding of the technical and programmatic issues relevant to human missions to Mars.
For this reason, rather than formulating an overall general plan and deducing from it the requirements for the associated missions, it has been decided to follow a bottom-up approach for the sake of the present study.

- It has been assumed that the ultimate goal is the establishment of a permanent outpost on the surface of Mars and that this will require several missions for the set-up and several missions for the routine exploitation of the outpost.
- These vehicles have been designed referring to a selected mission “case”. This case was chosen so as to reduce the design effort, though remains representative enough of the main technical issues associated to a human mission to Mars.
- The performed design is to be used to identify and recommend further investigation in potentially promising mission options and scenarios.
- The performed design will support the definition of the guidelines and the required technologies for the exploration plan
- The common “building blocks” (basically vehicles) required to comply with most of the above possible missions have been identified.

The “building blocks” investigated in this study are:

- The Transfer Habitation Module (THM), defined as the vehicle that hosts the crew in its trip from Earth orbit to Mars orbit and back towards Earth and during the orbital phase around Mars. Though several configurations are possible depending on the type of technology used for the transfers and the orbit insertion, many subsystems are common to all cases. Mastering the design and technologies for such a vehicle will be fundamental to perform any human mission to Mars, or long duration missions within the solar system.
- The Mars Excursion Vehicle (MEV), defined as the vehicle that performs the entry descent and landing onto the Martian surface, hosts the crew during the Mars stay, lift-offs to Mars orbit at the end of the surface mission and performs the rendezvous with the THM before departing back to Earth. This vehicle is present in all the mission scenarios and it is most critical. In particular, entry, descent and landing represents a challenge.

In addition to the above vehicle designs, the objective of the study was to tackle the main technical issues relevant to a human mission to Mars and address the following list of general mission architecture questions:

1. How far can safety requirements be fulfilled? What is the risk level that can be accepted without hampering the mission feasibility?
2. What is the impact of radiation protection on the mission?
3. Which is the most appropriate approach to counter microgravity effects on the human body? Is there any showstopper linked to microgravity exposure over a long period of time?
4. What is the optimal compromise between mission duration (impact in e.g. consumable and radiation protection mass) and ΔV (impact in e.g propellant mass)?
5. What is the optimal time-sharing between time spent around Mars and time spent in interplanetary transfer, and what is the impact on mass?
6. What are the mass critical components and what are the design possibilities to reduce the impact of these components on the mission feasibility?
7. Which are technologies worth investigating for this mission, meaning that their implementation will result in significant benefits when compared to the “mission case”?
8. What is the best assembly strategy for the vehicle in LEO?

The mission “case” has been defined according to the following criteria:

- Capability to perform quantitative assessments in a reliable way. This leads to select technologies known and relatively mature even if not the most mass effective for the overall mission.
- Mass effectiveness sought in trajectory and architecture definition (e.g. type of trajectory, surface stay duration, number of vehicles, etc.). Therefore, an effort to “optimise” the mission remaining within the limits of existing technologies has been done.
- Possibility to extend the results to more generic/advanced missions.

2.2 Overall mission requirements and constraints

The requirements for the mission “case” have also been defined taking into account the most general set possible with emphasis on physiology requirements, safety requirements and planetary protection (common to all missions).
Specific requirements on functions and operations to be performed on the Martian surface and during the whole mission have been reduced to a minimum to have a simple first design point as a basis for future sensitivity analysis.

2.2.1 Mission Objectives

The following mission objectives for the design case to be analysed in this study were agreed by a group of planetary exploration experts at the second Aurora Working Meeting:

- Land a crew of humans on Mars around 2030 and return them safely, ensuring planetary protection for both Earth and Mars
- Demonstrate human capabilities needed to support a human presence on Mars
- Perform exploration and expand scientific knowledge taking maximum advantage of human presence including sample selection
- Assess suitability of Mars for longterm human presence (habitability, resources availability, engineering constraints)

These highlevel requirements have the following consequences for the mission case definition:

- Landing on the Martian surface is required. Missions limited to Mars orbit or fly-bys are not acceptable as design case
- Duration of the stay on the Martian surface shall allow for some excursion (surface EVA) and sample collection capability (no flag footprint only, as first Apollo mission)
- Mission case shall take into account all the constraints coming from human requirements (physiology, radiation, habitability, etc.)

From the high-level requirements, a set of more specific technical requirements has been derived:

2.2.2 Mission-general requirements

- Number of crew: six in total, with three landing on the surface of Mars
- Mars sample collection with EVA required (up to 100 kg) in close proximity of the landing site
- Launch dates: between 2025 and 2040
- Maximum assembly time in LEO: 6 years (comparable to nominal ISS building time) but preferably of the order of 2 years to catch up with the opportunity windows to Mars

2.2.3 Safety requirements

The overall safety requirement is a “cultural” choice and depends on the probability of loss of life the public opinion is ready to accept. A high-level decision between the “pioneer” (high-risk) approach and the “clerk” (low-risk) approach has to be taken.
For human space flight (ISS, Space Shuttle) the maximum number of acceptable failures leading to life loss over total number of missions is generally about or equal to 1/200. However, at this stage this number cannot be verified by analysis, therefore it has been taken only as reference.
For the design only the following requirements have been set:

- All the systems shall be made fail safe and fail operational. Whenever this is not possible abort scenarios shall be built-in,
- Communication gaps need to be minimised. One week maximum duration is acceptable.

2.2.4 Physiology requirements:

2.2.4.1 Habitability

The habitability requirements can be expressed in terms of “required pressurised volume”, which, as shown in Figure 2-1, is a function of the mission duration for times less than 100 days. For longer missions the required volume stays almost constant.

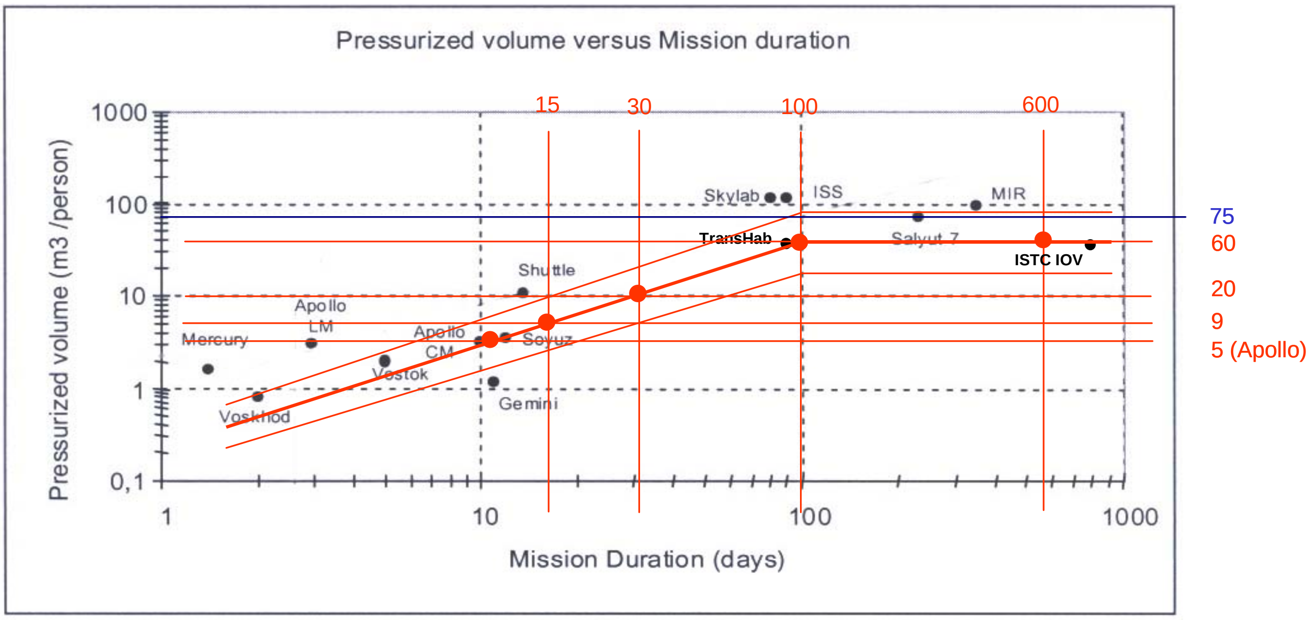
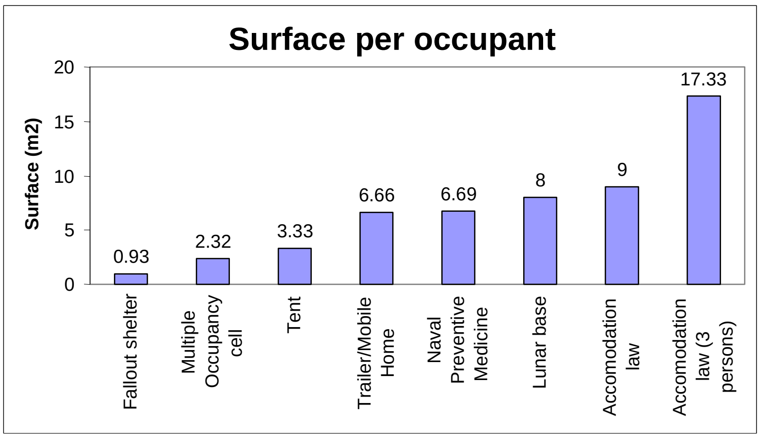


Figure 2-1: Required pressurised volume as a function of the mission duration (historical data)

Based on historical data, the required pressurised volume for the THM was determined to be 75 m^3/person. Out of this 25 m^3/person was determined to be equipment-free space.

In the case of the SHM, a gravity of 1/3 g will be affecting the astronauts, therefore the requirement changes to free surface. A survey has been carried out to assess the required surface:



Naval Medicine Standards: 6.69 m^2 per occupant or 20 m^2 for a crew of 3

Figure 2-2: Surface available in the different earth systems

For a crew of three people at least 20 m^2 are required with a minimum altitude of 2.5 metres. This makes a volume of 50 m^3, plus 25 m^3 for storage and 4 m^3 for the airlock. Total pressurised volume for the SHM is 79 m^3.

2.2.4.2 Accelerations

The requirements for maximum g-loads vary depending on the body axes considered (see Figure 2-3 for body axes). Crew seats are oriented so that the g-loads during the critical phases (i.e. launch and landing. etc) are along the +Gx direction, direction in which higher loads can be sustained. The maximum allowable loads along that axis are shown in Figure 2-4.

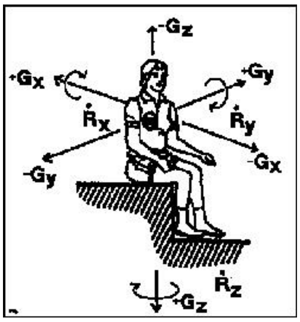


Figure 2-3: Body axes directions

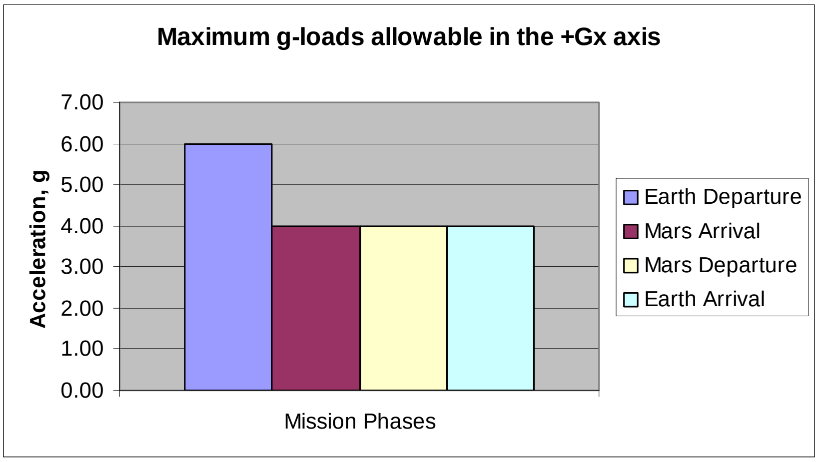


Figure 2-4: Maximum allowable g-loads in the +Gx direction

Maximum allowable g-loads at Mars arrival, Mars departure and Earth return are lower to account for crew deconditioning after experiencing microgravity for several months. Due to the lack of data for very long durations during microgravity, after a certain threshold time the maximum g-loads have been assumed as constant (optimistic approach).

The requirements for sustained g-loads are not only a function of the direction but also depend on the time of exposure. As shown in Figure 2-5, long-duration g-load limits depend linearly on the time of exposure. The +Gx axis direction has the highest allowable loads and –Gz the one with the lowest. Figure 2-5 data do not consider microgravity exposure.

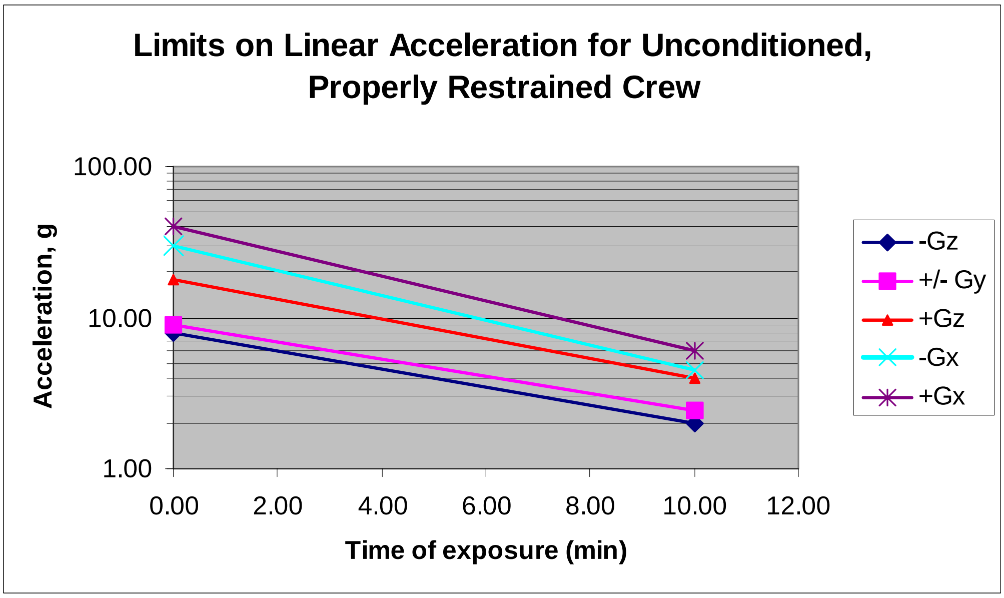


Figure 2-5: Requirements for sustained g-loads

The requirements on the impact g-loads (dynamic) depend again on the time of exposure to the impact. Figure 2-6 represents the tolerance to short duration –Gz accelerations (worst case).

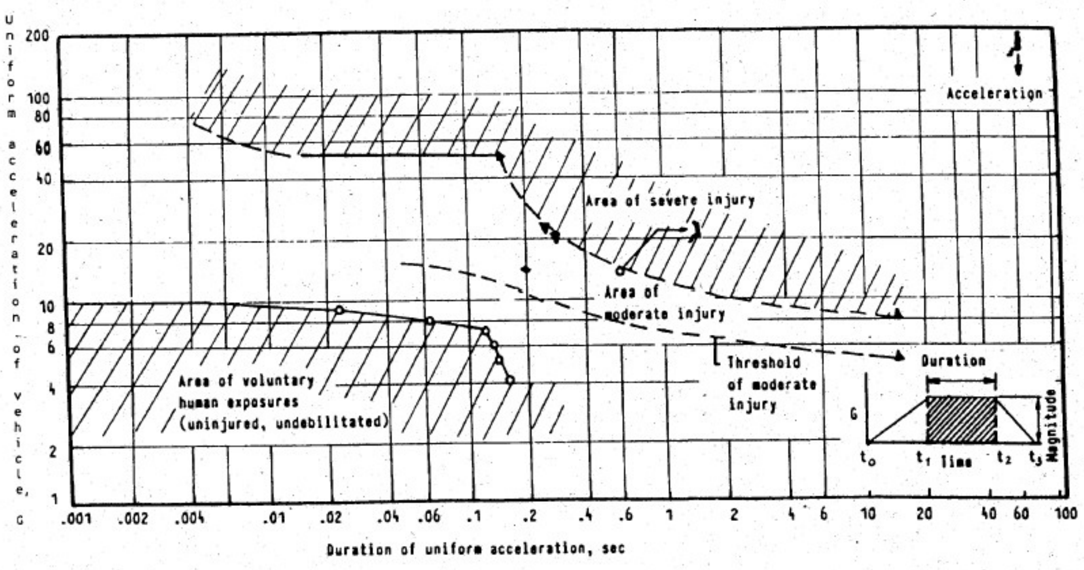


Figure 2-6: Tolerance to short-duration –Gz accelerations

2.2.4.3 Noise

The requirements for noise levels were established to be the following:

- Maximum noise exposure for 8 hours: 84dB (NASA-STD-3000)
- Maximum noise exposure for impulse sound: 140dB (NASA-STD-3000)
- Maximum ambient noise level during daytime: 60dB
- Maximum ambient noise level during night time: 55dB

2.2.4.4 Temperature and relative humidity

The temperature and relative humidity requirements were determined in compliance with Figure 2-7. As a consequence, the temperature in the habitable environment shall be between 18 and 27 °C at all times, for all the habitable volumes (THM, SHM, MAV, ERC).

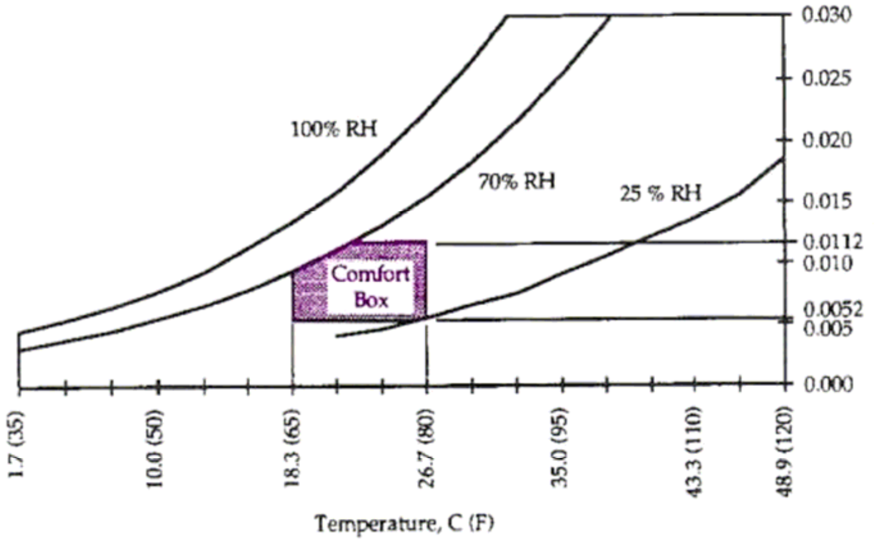


Figure 2-7: Temperature and Humidity levels

2.2.4.5 Radiation

The requirements for radiation doses depend on various factors such as the anatomical part exposed, age and gender of the astronauts. The limits taken as a reference for the different anatomical parts are listed in Table 2-1:

	NCRP Report No. 98 (Sv)	
Anatomical Location	**30 day limit**	**1 year limit**
Eye	1.00	2.00
Skin	1.50	3.00
BFO	0.25	0.50

Table 2-1: Radiation dose limits (BFO: Blood Forming Organs)

Figure 2-8 shows the maximum radiation dose levels allowable depending on the crew age and gender, for the BFO. The selection of the age of the astronauts plays a very important role in the dose tolerance and, therefore, in shielding mass. According to the calculations, no extra shielding is required for the case of a 55-year-old male while 13 g/cm^2 are required for a 25-year-old woman.

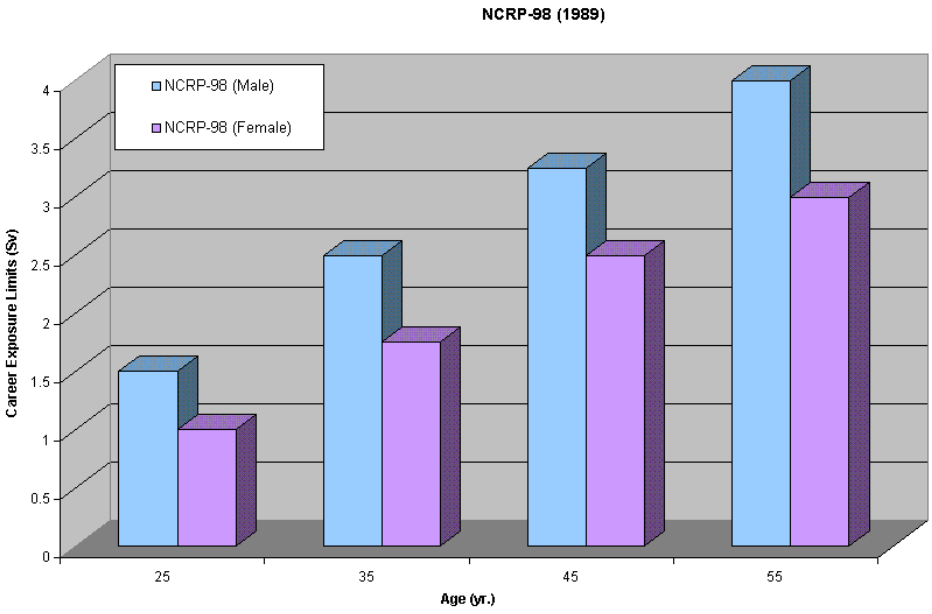


Figure 2-8: Allowable career Radiation Levels based on gender and age of the crew

2.2.4.6 Effect of microgravity

Physiological systems have to adapt to the Martian gravity environment after the long exposure to microgravity, and the reconditioning time varies for the different systems. The re-adaptation of these systems is shown in Figure 2-9, in which a tentative limit has been established to determine if some light physical work (walking on the surface of Mars for example) could be performed without risk. This limit has been set after extrapolation from today's knowledge about the return to a 1-g environment. It can be seen in the graph that, with this extrapolation, the physical limitation would be present during the first 7 days after return to the gravity environment of Mars.

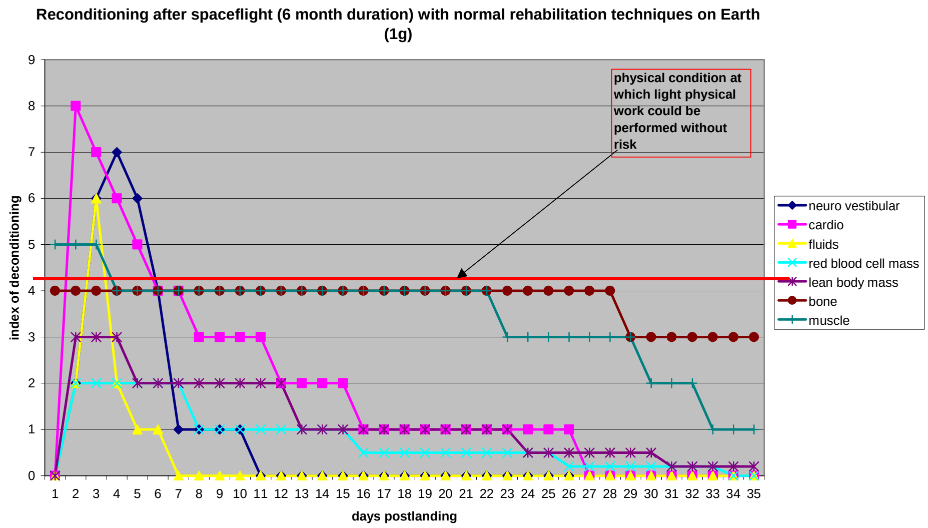


Figure 2-9: Assumed reconditioning time to Martian gravity environment

The most critical systems are the cardiovascular system (fainting), the neuro-vestibular system (dizziness, disorientation after quick turns), and the possibility of blood anemia (weakness). The muscles would be very weak during the first 3-4 days but would rapidly adapt.

Regular exercising during transit and during the first days on the surface of Mars would have to be mandatory for fast adaptation. If no continuous artificial gravity is implemented, the following exercise equipment and countermeasures have been identified as necessary on-board:

- Human centrifuge as a basis combined with a vibration platform as a possible countermeasure for bone and muscle loss (about 2 h/day/crew member as reference).
- Endurance exercise: treadmill and cycle ergometer
- Strength training: resistive exercise
- Pharmacological countermeasures: bone loss, radiation, space anaemia, fluid loss

2.2.4.7 Health

The autonomous medical capabilities required on-board have been identified as the following:

- Periodic evaluation of health status
- Preventive care
- Countermeasures (drugs for effect of the environment)
- Trauma care
- Analysis and Diagnostics (blood, urine, air, water, imaging systems)
- Medications
- Reanimation & First Aid
- Anaesthesia
- Surgery and Intensive Care ("home doctor surgery")
- Hyperbaric treatment
- Radiation dosimetry, protection
- Information technology: on-board expert system

2.2.4.8 Psychology

Psychological requirements other than the ones implicitly taken into account in the volume definition have not been defined. No crew composition requirement has been considered.

2.2.5 Planetary protection requirements

No specific planetary protection regulations for human missions to Mars exist at the moment but there is currently a working group addressing this issue. For the vehicles involved in the mission and for the samples collected it is assumed that the COSPAR 2002 regulations apply.

As regards Humans the assumptions taken for this study were the following:

- Any contamination of the habitat has to be avoided to prevent any backward contamination to Earth. Therefore, elements that have "seen" Mars should not be introduced in the surface habitation module.
- Strict isolation of the crew during and upon Earth return is not foreseen, as it would not be practicable and very difficult to implement.

As regards Vehicles and Samples, the following applies:

2.2.5.1 Mars bio-contamination

- Any spacecraft intended to land on the surface of Mars shall satisfy Cat. IV (bioload on exposed surfaces of less than Viking pre-sterilization levels).
- Spacecraft, or parts of spacecrafts, not intended to land on the surface of Mars shall have a probability of impact on Martian surface of less than 10^{-4} or bioload of less than $5 \cdot 10^{-5}$ spores on the whole spacecraft. This requirement is also valid for parts that re-enter the Martian atmosphere.
- Orbiting spacecraft are classified as category III and shall have a probability of impact for the first 20 years of $<10^{-2}$ and for the following 30 years of $<5 \cdot 10^{-2}$. Any Mars orbiting spacecraft is exempt from these orbital lifetime requirements if the bioload on the entire spacecraft is less than $5 \cdot 10^{-5}$ spores.

2.2.5.2 Earth backward bio-contamination

- Return samples classified as category V, restricted Earth return. As such, they have to be enclosed in a biological containment for all mission phases until they are inside the Mars Sample receiving Facility back on Earth. Verification of sample container sealing required before entering in the Earth-Moon system.
- Contaminated vehicles returning from Mars shall not enter the Earth-Moon system unless their external surface is sterilized by the high temperature during entry. The probability of contaminated material to be exposed to the terrestrial biosphere shall be less than 10^{-6}.

2.2.6 Constraints

Besides the above requirements, the following constraints were specified for the mission case definition:

- Avoid the Martian dust storm season for the landing and the surface operations
- No critical operation allowed during superior conjunction
- Development of a new on-purpose launcher is excluded
- No previous cargo mission with surface infrastructure or consumables shall be assumed

Technology constraints have a large impact on the mission case definition. They consist of the elimination of possible “advanced” technologies from the mission options. Including such technologies will contrast with the need for obtaining a clear understanding of the technical issue at this stage.

- Nuclear power either for cruise and Martian surface shall not be considered
- Nuclear propulsion shall not be considered
- Electric propulsion shall not be considered
- In-Situ Resources Utilisation shall not be considered, either for propellant or for food
- Food production (e.g. greenhouse) shall not be considered
- Inflatable structure technology for the Habitation Module shall not be considered

No specific requirement on the Readiness level for the technologies has been applied to the mission. A high technology readiness level (TRL) is preferred to obtain more robust assessments.

2.3 Background

The only comparable project to date is the Apollo programme but:

- Overall mission ΔV was half
- Crew consisted of only three people
- Mission duration was much shorter
- Distance from the Earth was much shorter

The ISS experience is applicable in some areas but:

- It is a LEO infrastructure with regular logistics
- It is research-oriented
- It is protected by the Earth magnetosphere
- The possibility of fast return to Earth always exists

During the past 50 years several studies have been carried out by national space agencies, industry and academia. The documentation available has been taken as reference, e.g.:

- NASA Reference Mission 1.0 and 3.0
- ISTC projects 1172 and 2120
- Humex study
- Primes Support
- European Mars mission architecture study, S51
- Future power systems for space exploration, S54
- Aroma study, S56

2.4 Mission analysis

2.4.1 Requirements and design drivers

The requirement for mission analysis design was to define the overall trajectory design required for a human mission to Mars. This involves:

- LEO around Earth during in orbit assembly phase
- Staged Earth escape
- Transfer to Mars
- Mars orbit insertion
- Mars target orbit acquisition
- Mars orbit phase
- Mars escape
- Transfer to Earth
- Earth arrival conditions

Further significant mission analysis contributions were in the following areas:

- Abort capabilities
- Aerobraking
- MEV entry analysis

The mission analysis discussions on these additional topics are included in the chapters for the respective subsystems.

2.4.2 Assumptions and trade-offs

As a first step, an analysis of the characteristics of all mission opportunities from 2028 to 2043 was required. Section 2.4.5 shows different mission durations arising from different launch dates, “Mission 2033” with launch in 2033 and return in 2035 was chosen as baseline. Three-week windows are assumed for launch and return.

2.4.3 Baseline design

2.4.3.1 Basic trajectory design issues

Figure 2-10 shows an overview of the trajectories for transfer to (red) and from (purple) Mars at the starts of the launch and return windows, respectively. Mars arrival is during the global dust storm season, which precludes an immediate landing on the surface. The TV must wait in orbit until the global dust storms (if any) have subsided. The return is safely before the start of the next global dust storm season.
Figure 2-11 shows the Earth-Sun-spacecraft geometry for the transfer to Mars. The maximum Earth range is 0.9 AU, the maximum Sun range 1.4 AU. No superior conjunctions occur.

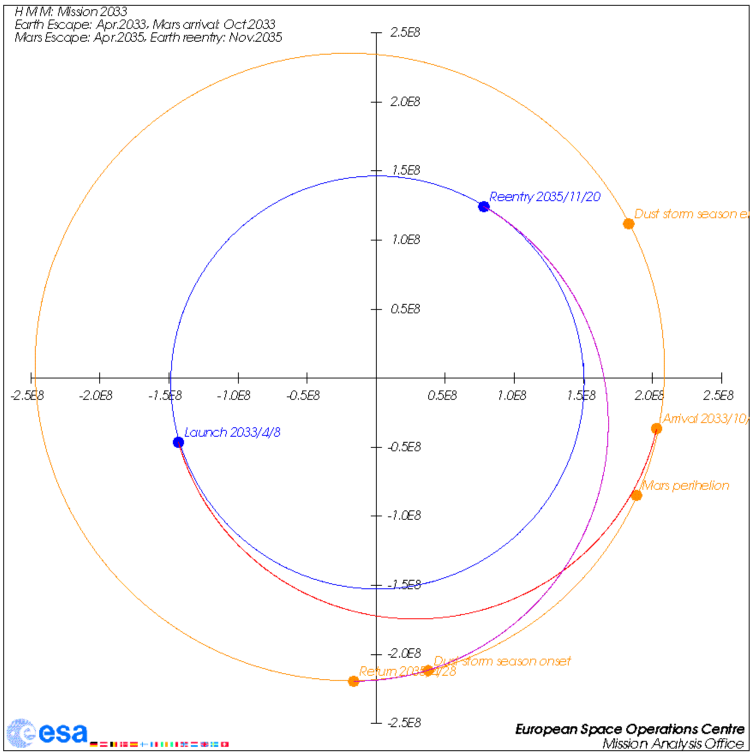


Figure 2-10: Trajectory Overview for Mission 2033

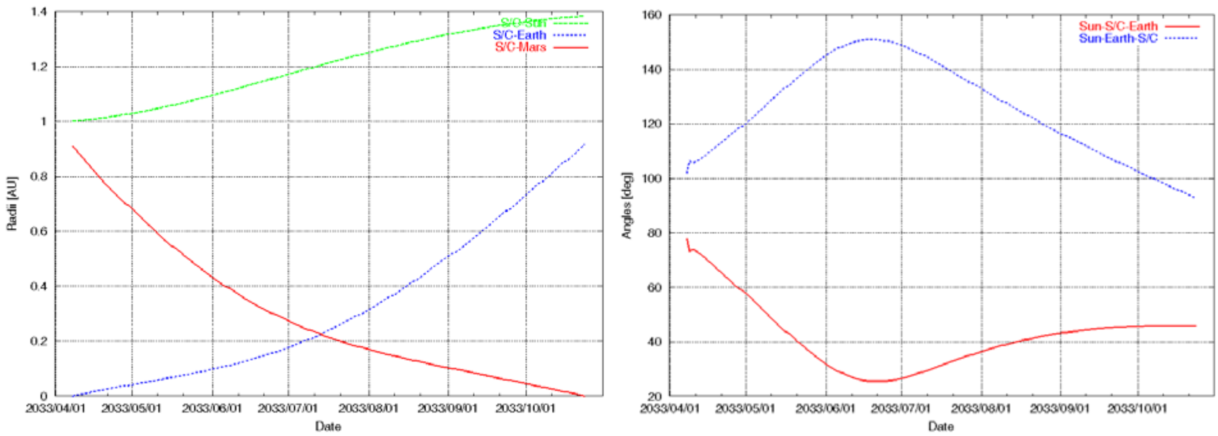


Figure 2-11: Geometry for Earth-Mars Transfer

Figure 2-12 shows the Earth-Sun-Mars geometry for the Mars phase. The Earth range rises to almost 2.7 AU, the Sun range to 1.65 AU, as Mars passes its aphelion in late summer 2034. There is a superior conjunction in August 2034. During this time, communications with the Earth will be impeded and no terrestrial measurements are available for orbit determination. This may impose constraints on operations and preclude critical actions during this time.

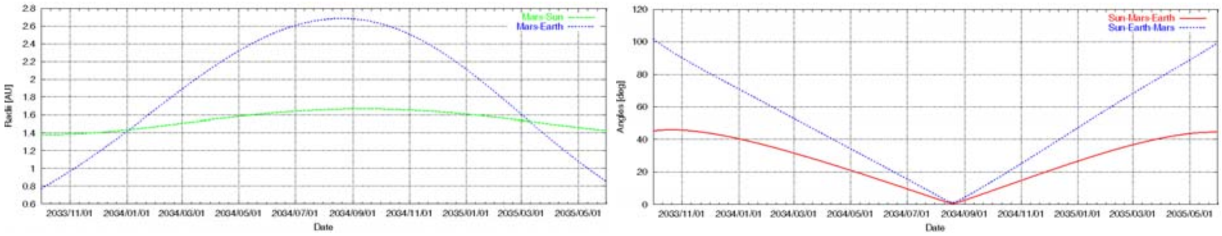


Figure 2-12: Geometry During Mars Phase

Figure 2-13 shows the Earth-Sun-spacecraft geometry for the transfer to Earth. Here, there are no superior conjunctions.

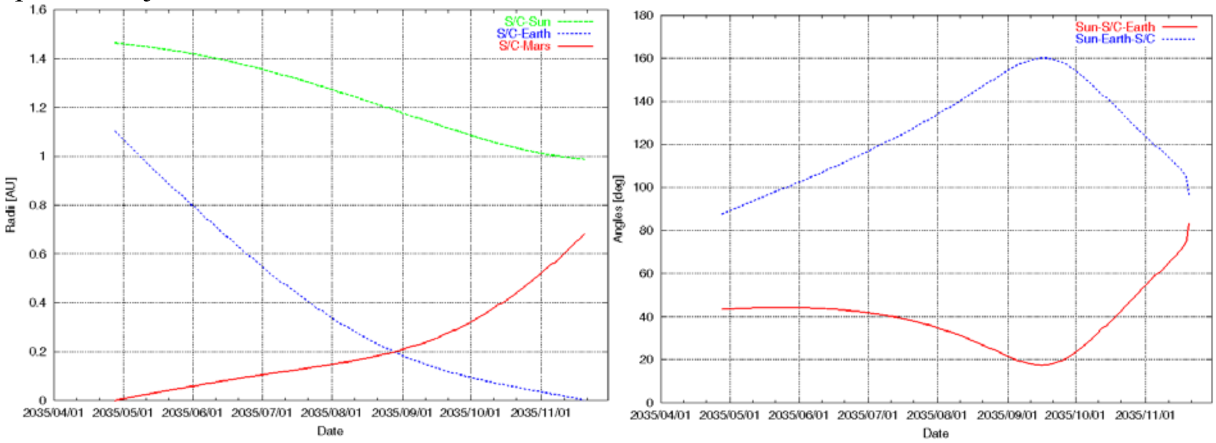


Figure 2-13: Geometry During Mars-Earth Transfer

2.4.3.2 Earth escape and planetary protection

The TV will be assembled in LEO. To minimise gravity losses, allow a staged solution and enhance operational safety, the TMI burn is split into three parts. The first two insert into eccentric orbits, the third and final one into the hyperbolic escape. After each burn, the spent stages are jettisoned and perform a controlled reentry. Otherwise they would reenter at some undetermined time and could endanger inhabited regions. The required Δv for de-orbiting is budgeted.

The final, escape burn must be biased such that the spent third stage does not hit Mars. After separation, the TV is retargeted on the course to Mars. This adds to the TV manoeuvre budget.

2.4.3.3 MOI and planetary protection

The assumption is that MOI inserts into a 4-sol orbit (approx 500x96 000 km). The MOI stage is jettisoned and remains in this orbit, the TV continues to a circular 500 km orbit. It was shown that the orbit of the MOI stage can be expected to decay within 10 years. A lifetime of over 50 years can be achieved by raising the pericentre altitude to 3000 km. This adds to the MOI stage budget.

2.4.3.4 The Mars orbit phase

After MOI, the orbit is circularized. The Mars orbit phase is spent in a 500 km circular orbit. This orbit will remain stable for over 50 years. The operations taking place during this phase are discussed in separate chapters.

2.4.3.5 TEI and planetary protection

If TEI is broken into phases to allow for a staged approach, then it must be ensured that the jettisoned first stage does not impact the surface of Mars within 50 years. The escape manoeuvre must be biased such that the escape stage does not hit the Earth. The THM must then be redirected to a trajectory closer to the Earth. This adds to the THM budget.

2.4.3.6 Final retargeting and Earth arrival

Designing this phase implies striking a balance between maximizing protection of the Earth and maximizing crew safety. The strategy assumed for the return trajectory is shown in Figure 2-14. The TEI dispersions are corrected with the retargeting manoeuvre seven days after Mars escape. The trajectory still is fail-safe and Earth-avoiding. Any failure at this time will lead the THM past the Earth.
Sixty days prior to the nominal arrival, at a distance of 18 million km, if the spacecraft is still functional, a retargeting is performed. Optionally, at this time the longitude and latitude of the touchdown point can be changed. One day prior to arrival, the crew moves to an Earth entry capsule and separates from the THM.

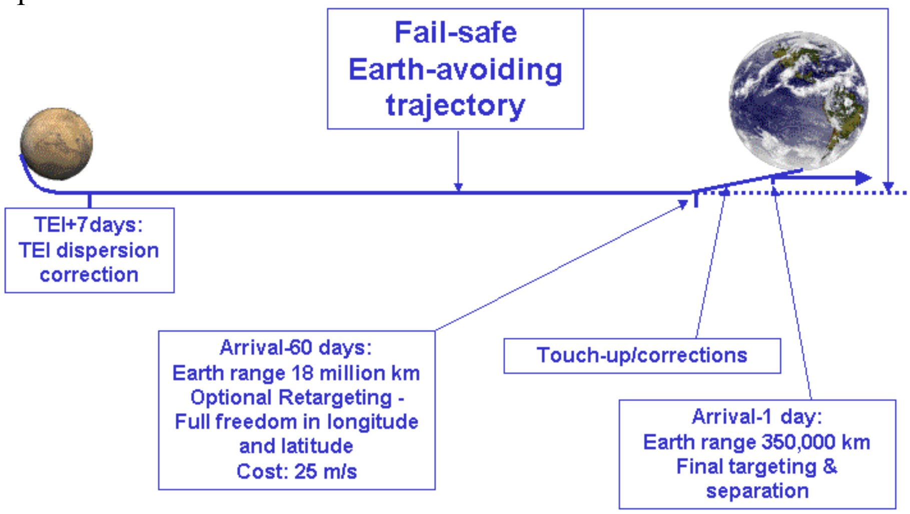


Figure 2-14: Earth Return Trajectory Strategy

There are two options for the final phase:

- At the retargeting 60 days before arrival, the THM can be targeted for an entry orbit. Then, after the entry capsule is separated one day before entry, the THM must be moved to a trajectory with a slightly higher perigee so that is passes by the Earth
- Alternatively, the retargeting can send the THM on a trajectory that passes close to the Earth without entering the atmosphere. Then, after the entry capsule is separated, it must navigate itself to an entry trajectory

2.4.4 Budgets

2.4.4.1 Earth escape window

- Escape window: 08/04/2033 – 28/04/2033
- Hyperbolic escape velocity: 3.022 – 3.200 km/s

- Impulsive TMI: 3.590 – 3.639 km/s from 400 km LEO
- De-orbit burns for 1st and 2nd stages: approx 95 m/s and 41 m/s, depending on staging assumptions
- Maximum sensitivity of escape manoeuvre: 150 000 km position error at Mars per m/s error in manoeuvre
- De-biasing and correction of escape manoeuvre dispersion: Max. 6 times the 3sigma-dispersion of the final escape manoeuvre, if performed 7 days after final escape
- Declination of escape hyperbola outgoing asymptote with respect to Earth equator: -47º – -62º. This leads to the inclination of the initial LEO, which must also be at least 62º

2.4.4.2 Mars arrival window

- Arrival period: 24 October 2033 – 11 November 2033
- Hyperbolic arrival velocity: 3.315 – 3.413 km/s
- Impulsive MOI: 1.143 – 1.201 km/s, assuming an initial orbit of 500 x 96000 km (period 4 sols)
- Additional Δv to raise pericentre of jettisoned stage to 3000 km: 48 m/s
- Final Orbit Acquisition to 500 km orbit: 1.285 km/s (impulsive)

2.4.4.3 Mars escape window

- Escape window: 28 April 2035 – 18 May 2035
- Hyperbolic escape velocity: 2.960 – 2.990 km/s
- Impulsive TEI from 500 km orbit: 2.229 – 2.245 km/s
- De-biasing and correction of escape manoeuvre dispersion: Maximum 4.6 times the 3 sigma-dispersion of the Mars escape manoeuvre, if performed seven days after Mars escape
- Declination with respect to Mars equator: -29º – -32º. This defines the target inclination with respect to the Mars equator, which must be at least 32º.

2.4.4.4 Earth arrival window

- Arrival period: 20 October 2035 – 27 November 2035
- Retargeting manoeuvre 60 days before arrival: maximum 25 m/s for change of arrival time by +/- 12 hours
- Perigee altitude change from 1000 to 100 m/s (or vice versa, depending on final strategy): 17 m/s. This is applied to ERC or THM, depending on which body needs to be moved
- Hyperbolic arrival velocity: 2.999 – 3.052 km/s
- Speed at 100 km altitude (not taking into account Earth's rotation): 11.506 km/s**Options**

All mission opportunities in the 15-year cycle from 2028 through 2043 were studied. The characteristics are listed in Table 2-2. The following assumptions apply for the data in the table:

- All cited manoeuvres are impulsive
- The maximum TMI, MOI and TEI values are provided for every launch opportunity. This maximum MOI includes only the orbit insertion. The column "Total Δv" cites the maximum of the subs of all three throughout the launch and return windows. This maximum total is *not* necessarily equal to the sum of the maximum TMI, MOI and TEI
- The column titled "Max. Varr" cites the maximum hyperbolic velocity at the Earth for the respective opportunity
- All given durations cite the maximum possible value occurring for a transfer or Mars phase. They do not necessarily add up to the maximum total mission duration given in the last column

Mission	Max. TMI [km/s]	Max. MOI [km/s]	E→M cost [km/s]	Max. TEI [km/s]	**Total Δv [km/s]**	Max. V_{arr} [km/s]	E→M dur. [d]	Mars dur. [d]	M→E dur. [d]	**Tot. dur. [d]**
2028	3.637	0.975	4.585	1.977	**6.562**	5.793	306	423	256	**971**
2031	3.717	1.310	4.946	1.969	**6.915**	4.255	285	490	225	**996**
2033	3.639	1147	4.786	2.254	**7.040**	3.052	200	571	207	**962**
2035	3.696	0.794	4.450	2.595	**7.045**	3.736	204	560	271	**1033**
2037	4.005	0.918	4.797	2.366	**7.163**	2.926	360	390	287	**997**
2039	3.810	0.745	4.492	2.159	**6.651**	3.091	342	362	306	**992**
2041	3.658	0.707	4.365	2.053	**6.418**	4.151	323	372	334	**1022**
2043	3.619	0.877	4.495	1.955	**6.450**	5.525	311	405	337	**1041**

Table 2-2: Comparison of Characteristics for Opportunities 2028 Through 2043

2.5 Cruise and surface operations

2.5.1 Planetary protection

Robotic precursor missions have to verify that life is at least not widespread on Mars. As such, it has to be shown that life is not part of the global dust-cycle. However, even if life is not widespread on Mars, planetary protection regulations for forward and backward contamination have to be applied because astronauts will investigate sites of potential extant or extinct biological activity.

A human mission to Mars will contaminate the planet to a certain extent. This forward contamination can be minimised by using appropriate procedures. The philosophy for backward contamination is to *break the chain between Mars and Earth*. Any contamination of the habitat has to be avoided as this contamination would be transferred to the crew, and hence the Earth. Strict isolation of the crew upon return is not practicable and would be difficult to implement.

2.5.2 Human surface operations on Mars

2.5.2.1 Introduction

The long-term objective of the first missions to Mars is to show that humans can go to Mars and develop a continuous presence. As part of this mission objective, exploration of the Martian surface (subsequently described by the term “field exploration”) is essential to assess the habitability of Mars, and to evaluate the potential use of Martian resources as part of a long-term strategy to *live off the land*.
The term “surface exploration” is used to describe activities that are related to the more engineering part of a mission (e.g. securing the landing site, deployment of communication equipment, test of the EVA equipment, etc.), and to the more explorative part of a mission (e.g. to understand the terrain of the landing site, to evaluate the location of potential resources, to asses potential hazards, etc.). In practical terms, the distinction between engineering and scientific knowledge required to assess the habitability of Mars is not a strict one. For the time being, the description of the human surface operations on Mars will focus on activities closely related to the more explorative part of a mission because it is essentially the driver for resources required for the surface operations on Mars (number of crew members on the surface party, mobility, amount of samples returned to Earth, etc.).

2.5.2.2 Description of surface activities

There are three different kinds of EVA activities that have to be performed on the surface of Mars:

1. Upon landing, securing the landing site and deployment of equipment in the vicinity of the MEV. This will require one or two dedicated EVAs.
2. Field exploration EVAs to explore the more distant environment and to collect samples.
3. A dedicated pre-launch EVA to select the samples to be taken back to Earth, perform the required activities to prepare the samples for the flight back to Earth (containment and transport to the MEV), and to collect deployed equipment.

The following description will focus on field exploration EVAs because they determine the required resources for the surface operations on Mars.

2.5.2.3 EVA duration

The maximum EVA duration on Apollo was 7 to 8 hours. However, EVA activities on Mars will be more demanding because of the higher gravity (to carry the EVA suit and additional equipment) and the more difficult terrain. In addition, the astronauts on Mars may have to perform more EVAs compared to the individual Apollo missions. This puts an additional physical strain on the astronauts. Therefore, it is foreseen to limit the time for a nominal EVA to 6 hours.

2.5.2.4 Maximum distance from the Mars Excursion Vehicle

There is a strong requirement to give astronauts enough time to perform field exploration on the surface of Mars, and therefore to limit the time necessary for travel to 20% of the total EVA time. Assuming the speed of a surface rover is 9 km/h for rover-assisted traverses, and for walking-traverses as 1-3 km/h, the maximum distance of EVAs from the Mars Excursion Vehicle (MEV) would be 5 km for rover-assisted EVAs, and 1 km for walking-traverses. These distances are in agreement with the requirement to be able to walk back to the MEV from any location during an EVA if the greatest distance from the MEV is reached at the beginning of an EVA activity.

2.5.2.5 Number of crew-members required for surface exploration

A typical EVA-day leaves no room for any spacecraft operations and preparation for upcoming EVAs for the crew members that are performing the EVA. With the requirement of a *buddy-system* for EVAs this means that EVAs can only be performed every second day. At the same time, having only two crew-members on the surface is a single point failure for EVAs in case one of them becomes incapacitated. The Apollo experience has revealed that EVA activities are by themselves very demanding (with pulse rates of up to 140 per minute), and produced fatigue and injuries especially to the fingers, which reduced the performance to an extent of not being able to remove the spacesuit without help. This, and other potential threats to the performance of astronauts on the surface of Mars, implies that a third crew-member is highly recommended. Comparing the timeline of a three crew-member team shows that even with this extra member it will not be possible to have more than one EVA every second day.

2.5.3 Conclusions

The following recommendations have been issued for the design:

1. The nominal duration of each EVA is 6 hours.
2. Upon landing, 1-2 dedicated EVAs are required for securing the landing site.
3. The maximum distance for walking-traverses (if any) from the MEV is 1 km.
4. The maximum distance for rover-assisted traverses from the MEV is 5 km.
5. The minimum number of field exploration EVAs is two (to revisit a site).
6. The minimum number of crew-members for surface operation is three.
7. The total sample mass returned to Earth in the first human mission to Mars shall be up to 100 kg.
8. One dedicated EVA is required to select the samples to be taken back to Earth, perform the required activities to prepare the samples for the flight back to Earth, and to collect deployed equipment.
9. The sample return part of the mission is classified as Planetary Protection Category V, restricted Earth return.

2.6 Radiation environment

Energetic charged particles with energies in the MeV range are encountered throughout the Earth's magnetosphere, in interplanetary space and in the magnetospheres of other planets. While solar proton events can provide very high particle fluxes over a short period of time, the radiation belts and cosmic ray fluxes provide a more continuous source of radiation. The energies of particles from these sources also varies, with the cosmic ray energies in excess of 1 GeV/nucleon and trapped particle energies limited to the MeV range.

2.6.1 Trapped particle belts

Energetic electrons and ions are magnetically trapped around the Earth, forming the radiation belts. The radiation belts consist principally of protons of up to several hundred MeV energy and electrons of up to a few MeV energy. The inner belt principally contains protons, extends to about 4 Earth radii, and is reasonably stable in time. The outer belt consists principally of electrons and extends to about 10 Earth radii and is highly dynamic: being subject to storms and injection events that follow solar-terrestrial disturbances. The radiation belts are of principal concern during the low-Earth orbit assembly phase, the Earth escape phase and the Earth return phase. Mars, lacking a strong magnetic field, is not expected to provide any significant trapped radiation belts.

2.6.2 Solar proton events

Energetic solar eruptions (Solar Particle Events, SPEs) produce large fluxes of Solar Energetic Particles (SEPs), which are encountered in interplanetary space and close to the Earth. These events are rare, occurring primarily during periods of solar maximum activity, which commences 2.5 years before Sun spot maximum and lasting for seven years. The duration of such events is usually of the order of days, with larger events lasting a week or more. The large fluxes of energetic particles can contribute a large, even lethal dose over a short period of time and the mission will be exposed throughout its duration.

Two aspects of the solar proton dose contribution must be considered: the short term and long-term effects. To ensure that a short-term limit, e.g. the 30-day limit, is not likely to be exceeded, a storm shelter can be provided that sufficiently shields the astronauts over the duration of the event. Considering the largest event measured to date, in August 1972, at least 20 g/cm^2 of shielding would be required to remain below the 30-day limit.

However, to calculate the radiation dose budget for the entire mission, it is more appropriate to use a statistical model to produce a radiation level based on a confidence level. The ECSS standard model is the JPL-1991 solar proton model and the confidence level of 90% for a 3-year mission is used.

2.6.3 Galactic cosmic rays

Galactic Cosmic Rays (GCR) provide a continuous flux of energetic ions from hydrogen to uranium. Although the flux is low (a few particles per cm^2 per s), GCRs include energetic ions, which can deposit significant amounts of energy in a small volume and are particularly damaging to biological materials, e.g. DNA. Because of the high energies of these particles, it is very difficult to shield against them.

2.6.4 Requirements and design drivers

Radiation limits set by ESA are shown in Table 2-3. Each exposure interval must be addressed in the mission planning and shielding design. The limits are selected based on a probability of increased risk to the subject, leading to the career NCRP BFO results ranging from 1 to 4 depending on age and gender, e.g. an older male is less likely to develop cancer at 1 Sv than a 24-year-old female.

Ionising Radiation Exposure Limits Organ Specific Equivalent Dose Limits (Sv)			
Exposure Interval	**Blood Forming Organ**	**Eye**	**Skin**
30 days (ESA)	0.25	0.5	1.5
Annual (ESA)	0.5	1	3
Career (NCRP-98)	1-4[a]	4	6
[a] varies with gender and age at initial exposure			

Table 2-3: Radiation exposure limits set by ESA and the NCRP career limit

The ECSS-E-10-04 space environment standard provides additional limitations and recommendations.

2.6.4.1 30 Day limit

The radiation sources that can contribute to exceeding the 30-day limit are from passages of the trapped particle belts; particularly during a prolonged Earth escape phase and solar proton events. To ensure the dose remains below such a level, it will be necessary to provide a storm shelter in which the astronauts can take refuge. Most events last less than 2 days, however, it is expected that the worst events can last up to a week or more, as shown in Figure 2-15.

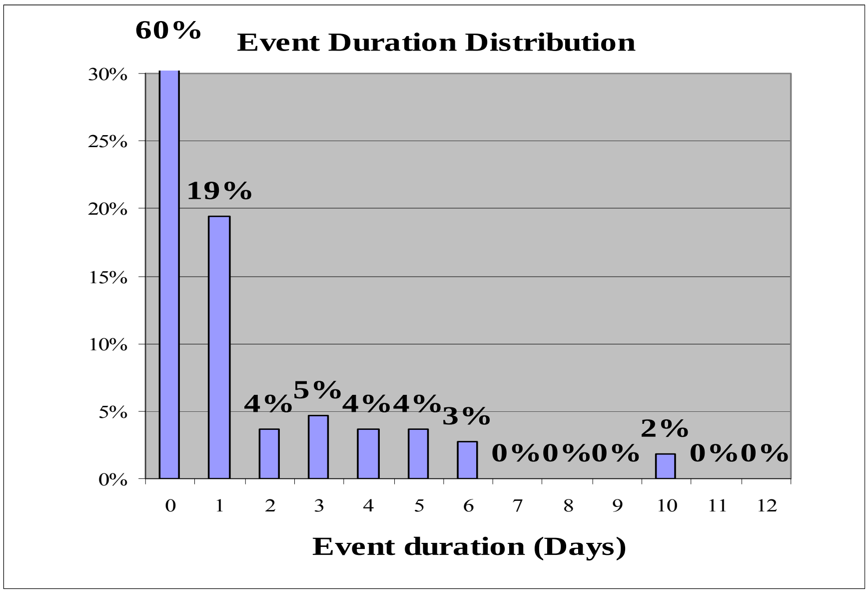


Figure 2-15: Distribution of the event duration where the 10 MeV flux exceeds 2 protons/cm^2/s/ster.

2.6.4.2 Yearly limit

The radiation sources that can contribute to exceeding the yearly limit include solar proton events and galactic cosmic rays. Due to the variation of the cosmic rays with the solar cycle, the yearly limit is not expected to be exceeded during solar maximum by the GCRs alone, but the limit would be exceeded at solar minimum if inadequate shielding is provided, as shown in Figure 2-16.

The skin and ocular dose from solar protons, using the ECSS-E-10-04 recommended 95% confidence level of the dose not being exceeded, with 9 g/cm^2 of shielding is 2 Sv/year. However, inside a storm shelter providing 25 g/cm^2 of shielding a yearly dose of 0.12 Sv could be expected in the eyes and skin tissues, assuming a semi-infinite planar shielding geometry. For the Blood Forming Organs (BFOs) the doses are 5.92 Sv and 0.532 Sv, respectively, assuming a spherical shell shielding geometry and the muscular tissues providing an additional 5 g/cm^2 of shielding in addition to the 25 g/cm^2 of the storm shelter.

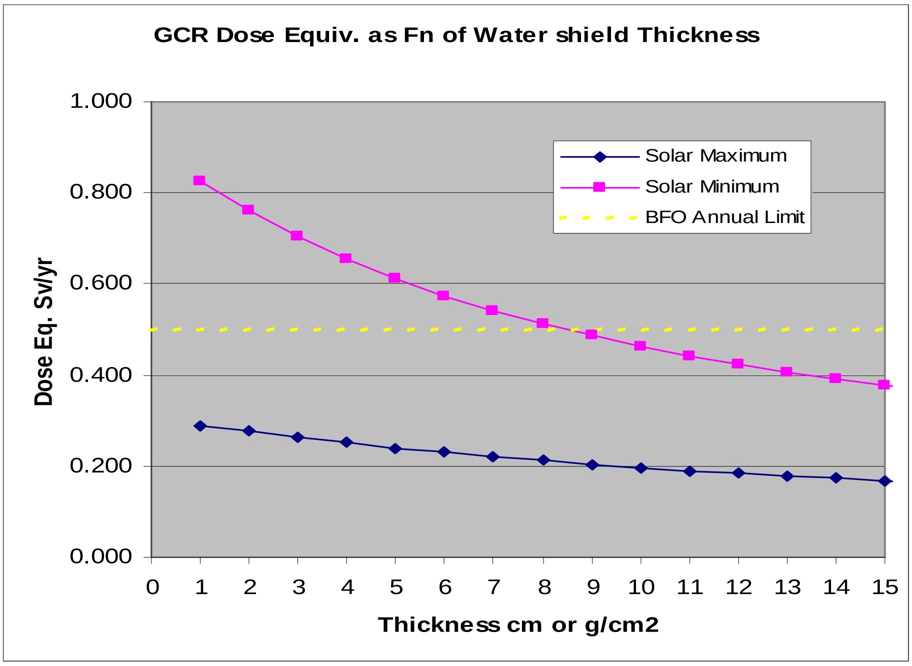


Figure 2-16: GCR dose equivalent (Sv/year) as a function of shielding thickness

2.6.4.3 Career limit

All radiation sources contribute to the career limit dose calculation, although the majority of the dose accrued throughout the mission will be from solar proton events and galactic cosmic rays.

2.6.5 Assumptions and trade-offs

- 500 km orbit about Mars – leads to reduction of about 25% in GCR flux due to planetary “bulk” shielding.
- A minimum shielding of 16 g/cm^2 to be provided by the Martian atmosphere during Martian surface activities.
- 25 g/cm^2 of shielding provided by the storm shelter to ensure remaining below the 30 day and 1 year limits.

- 9 g/cm^2 of shielding provided in habitation module to ensure remaining below the yearly limit and career limits from GCRs.
- The BFO doses for solar proton doses have been calculated assuming that an additional 5 g/cm^2 of shielding is provided by surrounding tissue, e.g. muscles, bones, etc. It is possible that this can result in significant errors in the estimation of the solar proton dose, as the calculation is at the limits of both the solar proton models and the radiation transport models used. It is expected that the proton fluence at high energies will be less than currently predicted by the JPL model and so result in an over prediction of the dose estimate. However, the errors in the shielding transport models are likely to result in the under prediction of the dose estimate for the solar proton dose. *A more comprehensive review of the models and analysis should be performed in the future.*

2.6.6 Budgets

Skin and Eye radiation Doses are shown in Table 2-4:

Phase	Duration (day)	Phase start	Sv/day	Total Sv	%	Year
LEO	2	2033/04/06	0.00077	0.002	0.040	
HEO	2	2033/04/08	0.03825	0.077	1.974	
Earth->Mars Transfer	215	2033/04/10	0.00085	0.184	4.736	
Mars Orbit	100	2033/11/11	0.00064	0.064	1.643	
Surface Activities	30	2034/02/19	0.00016	0.005	0.126	1.454
Mars Orbit	403	2034/03/21	0.00064	0.257	6.620	1.555
Mars->Earth Transfer	213	2035/04/28	0.00085	0.182	4.692	0.868
Solar Proton Dose	965			3.108	80.170	
Total	**965**	**2035/11/27**		**3.877**		

Table 2-4: Skin and eye radiation doses

Blood Forming Organ (BFO) Doses are shown in Table 2-5:

Phase	Duration (day)	Phase start	Sv/day	Total Sv	%	Year
LEO	2	2033/04/06	0.00077	0.002	0.040	
HEO	2	2033/04/08	0.03825	0.077	1.974	
Earth->Mars Transfer	215	2033/04/10	0.00064	0.138	3.550	
Mars Orbit	100	2033/11/11	0.00048	0.048	1.231	
Surface Activities	30	2034/02/19	0.00016	0.005	0.126	0.329
Mars Orbit	403	2034/03/21	0.00048	0.192	4.961	0.263
Mars->Earth Transfer	213	2035/04/28	0.00064	0.136	3.517	0.174
Solar Proton Dose	965			0.834	4.357	
Total	**965**	**2035/11/27**		**1.430**		

Table 2-5: Blood forming organs radiation doses

2.6.7 Conclusions

To ensure that the astronauts do not receive a 30-day dose in excess of the limits, it will be necessary to include a storm shelter with at least 25 g/cm^2 of shielding to protect against solar proton events. It will also be necessary to provide a minimum of 9 g/cm^2 of shielding throughout the habitation module to ensure the yearly limit and career dose limits are not exceeded by the galactic cosmic ray radiation.

The BFO career radiation limits vary with the age and sex of the astronaut. The lowest career radiation limit of 1 Sv is set for young females and will be exceeded during the mission. However, by selecting crew by age and sex, it should be possible to ensure a minimum crew dose limit that is above the 1.4 Sv dose to be accrued on the mission.

Finally, the solar proton particle fluence calculation should be reviewed and the uncertainties from the high-energy solar proton fluence resolved. Additionally, the radiation transport dose calculation should be revisited with greater accuracy models.

2.7 Mission architecture

2.7.1 Mission case

Many design options are possible for a human mission to Mars, but only a limited set has been considered according to the criteria defined for the case selection:

- Capability to perform quantitative assessments in a reliable way (e.g. selected technologies known and relatively mature even if not the most mass effective)
- Mass effectiveness sought in trajectory and architecture definition (e.g. type of trajectory, surface stay duration, number of vehicles, etc.)
- Possibility to extend the results to more generic/advanced missions

2.7.2 Mission case basic architecture

The main goal of the human mission to Mars is to land a crew of astronauts on the Martian surface and bring them back to Earth. The minimum elements required (are shown in Figure 2-17):

- Propulsion system to perform the cruise to Mars and back, Propulsion Module (PM)
- Habitation module to support the astronauts during the trip, Transfer Habitation Module (THM)
- Descent/Ascent module to land onto Mars and get back to Mars orbit, Mars Excursion Module (MEV)
- Entry capsule to allow astronauts to reenter the Earth’s atmosphere on return, Earth Reentry Capsule (ERC)

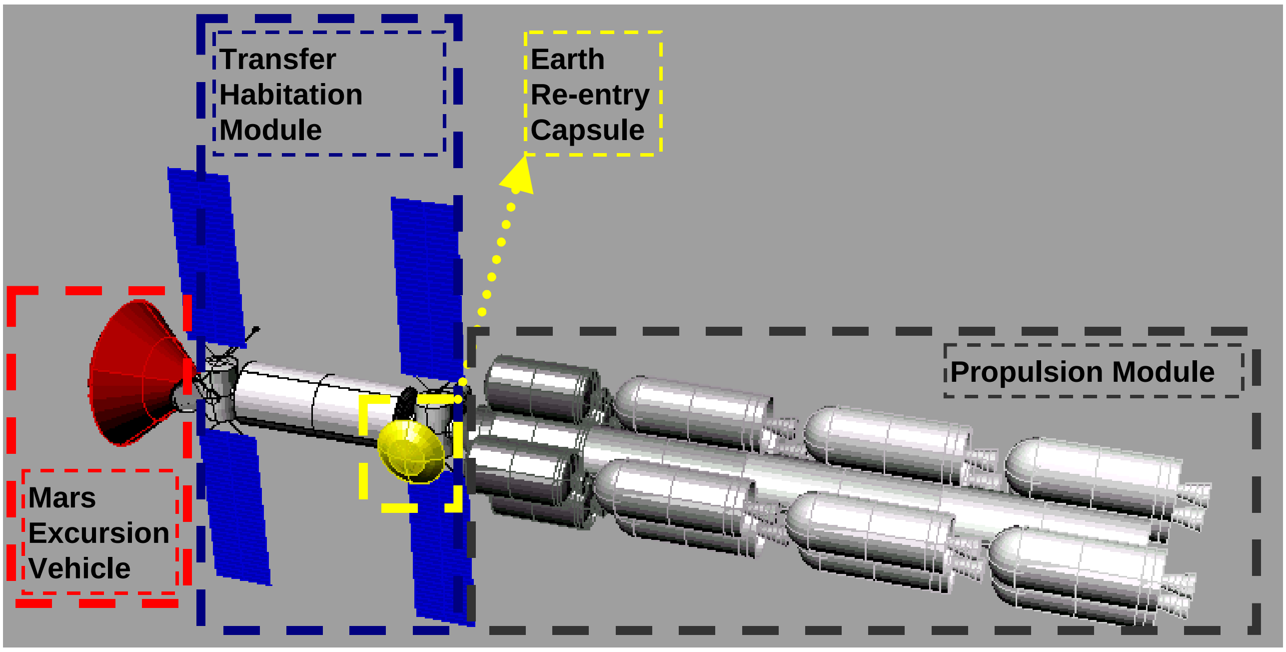


Figure 2-17: Mission elements for a human mission to Mars

If no extra infrastructure is available in LEO on return, an ERC is required to return the crew to Earth surface, either performing a direct entry from interplanetary trajectory or by parking the TV in LEO. In the case of an existing infrastructure, the TV could rendezvous and dock to it to allow the transfer of the astronauts. This last option has not been considered in the study.

2.7.3 Key parameters

To perform the selection of the mission case, the following parameters have been analysed:

- Overall mass into LEO
- Overall mission duration
- Assembly time
- Time availability for Martian surface operations

2.7.4 Mission case architecture options

A preliminary set of trade-offs and options was established at the beginning of the study. The trade space has been already reduced by taking into account all the technological and programmatic constraints, as well as the criteria defined for the mission case study. The main trade-offs and options studied are shown in Table 2-6.

Trade-offs	Options
Trajectory	Conjunction Opposition Venus swing-by Low thrust
Surface stay duration	Long stay Short stay
Propulsion	Chemical Storable Cryogenic NTP SEP NEP
Return approach	THM and ERC inserted around Earth THM discarded, ERC inserted THM discarded, ERC direct entry
Orbit insertion around Mars	Propulsive Aerocapture Aerobraking
Orbit around Mars	Circular High elliptical orbit
MEV release	From circular orbit From high elliptical orbit
Split / All up	Split scenario All up scenario
Microgravity countermeasures	Spinning spacecraft Centrifuge
Crew number	3 to 6

Table 2-6: Trade-offs and options for the study case

2.7.5 Basic assumptions for trade-offs

The architecture trade-offs were performed along the study in parallel to the evolution of the design. However, an initial screening of the options was done to reduce the options. For this activity, a starting vehicle design point was required. The numbers used at the beginning for this purpose are the following:

2.7.5.1 Mission elements dry masses

Mission Element	Mass (tonnes)
THM	55.4 (dry)
MEV	29 (wet)
ERC	10.2 (wet)

Table 2-7: Mission Elements masses

These figures are derived mainly from literature or from preliminary simplified computations, and just represent a starting point.

2.7.5.2 Life support system for the THM

The levels of closure assumed for the life support system are the following:

Element	Level of closure (%)
Oxygen	95
Potable water	95
Grey water (condensate, used hygiene water)	95
Yellow water (water in contact with urine)	95
Black water (water in contact with faeces)	20
Solid organic waste to food	20
Solid inorganic waste	0
Packaging reuse	0

Table 2-8: Life support system level of closure

Taking these levels of closure into account and typical mission duration of 950 days, the consumables required for a crew of six for the whole mission are 10.2 tonnes.

2.7.5.3 Propulsion system

A modular design for the propulsion module has been assumed, that is, separate propulsion systems are used for each main propulsive manoeuvre. This approach allows the jettisoning of each propulsion module after its usage. Within each main propulsive manoeuvre, a staging approach is also followed, so that the manoeuvre is split into several stages to increase the efficiency of the system. This approach allows you to get rid of the stages once they have been used and also reduces the gravity losses as the time required for each burn is lower. Therefore, the system is assumed to be as follows:

- TMI module (3 stages)
- MOI module (2 stages)
- TEI module (1 stage)

In general, each propulsion stage will be bigger than the launcher capabilities in terms of mass, so, each stage will have to be split into submodules, called stacks. With this approach it is expected to reduce the cost of the system, as the same design will be used for all the stacks.
Regarding the propulsion technologies used, the values assumed are as shown in Table 2-9:

Technology	Isp (s)	Structural index (%)
Storable	345	8
Cryogenic	450	11

Table 2-9: Propulsion technologies characteristics

Note that actual storable technology for such thrust levels provide an Isp of 325 s. Here an optimistic approach has been taken.

2.7.5.4 Waste management

As the consumables are used, waste is generated. The waste is defined as the goods or material that once used, cannot be recycled. The amount of waste produced is directly linked to the level of closure of the life support system. For a mission of six crew members with a duration of around 1000 days and the levels of closure presented before, the total waste generated is 5.4 tonnes.
The assumption is that the waste produced up to the MOI is discarded so the payload mass for this propulsive manoeuvre is reduced. The same applies to the TEI manoeuvre, the waste produced from the MOI till the TEI is also discarded.

2.7.5.5 Assembly in orbit

For the assembly of the composite in Earth orbit it has been assumed a circular orbit of 400 km altitude. The inclination depends on each trip opportunity to Mars, i.e. 62 degress for the 2033 opportunity.

2.7.5.6 Parking orbit around Mars

As parking orbit around Mars it has been assumed a circular orbit of 500 km altitude with the required inclination for the return to Earth trajectory, which also depends on the trip opportunity, i.e. 32 degrees is the one for the 2033 opportunity.

2.7.5.7 Launcher

The biggest launcher selected for the analysis is the Russian Energia. The performances assumed for this launcher are shown in Table 2-10:

Energia performances	
Mass to 400x400 (tonnes)	80
Fairing diameter (m)	6
Fairing length (m)	35

Table 2-10: Energia assumed performances

Other launchers potentially usable are:

- Ariane-5 (EC-B version)
- Proton (K version)
- Soyuz
- Space Shuttle (only if crewed missions are required during the assembly in LEO)

2.7.5.8 MEV release

The MEV is released from the parking orbit at Mars, 500x500 km.

2.7.5.9 Strategy on Earth return

The ERC is assumed to perform a direct entry on Earth return. The THM is separated and put on and Earth avoidance trajectory.

The following table gives a summary of the design point for the trade-offs:

Design Point	
THM dry mass	55.4 tonnes
MEV dry mass	29 tonnes
ERC dry mass	10.2 tonnes
ECLSS level of closure: **Oxygen** **Potable water** **Grey water(condensate, used hygiene water)** **Yellow water(water in contact with urine)** **Black water(water in contact with faces)** **Solid organic waste to food** **Solid inorganic waste** **Packaging reuse**	% 95 95 95 95 20 20 0 0
Consumable mass for the THM	10.2 tonnes
Waste mass	5.4 tonnes
Waste discarded prior to	MOI and TEI
Cryogenic propulsion Isp	450 sec
Cryogenic propulsion structural index	11%
Storable propulsion Isp	345 sec
Storable propulsion structural index	8%
In orbit assembly orbit	400 X 400 km, 62 degrees
Energia performances: **Mass to LEO** **Fairing diameter** **Fairing length**	 80 tonnes 6 m 35 m
Other launchers used	Ariane-5, Proton, Soyuz, Space Shuttle
MEV released	From parking orbit 500 x 500, 32 degrees
THM	Discarded on Earth arrival
Strategy for Earth return	Direct entry of ERC

Table 2-11: Design point and assumptions for trade-offs

2.7.6 Trade-offs

2.7.6.1 Trajectories

The general problem is to find a trajectory to go to Mars and back. Only a few solutions are possible for this problem, and that establishes a direct link between the mission duration and the energy required (ΔV):

- short mission – high ΔV
- long mission – lower ΔV

The ΔV required for the mission has a exponential impact in the amount of propellant required for the mission. On the other hand, the mission duration has a linear impact on the wet mass of the THM. The THM mass can vary depending on the following factors:

- THM structures remain the same as the required volume is constant for missions over 100 days
- Life support equipment for the level of closure selected will remain the same (slight modification due to the storage facilities for the consumables)
- Consumables vary linearly with time
- Radiation shielding for the GCR, no impact for missions below 1000 days, as the required protection is already provided by the structure and internal equipment
- Storm shelter could be slightly reduced if the duration is shortened, but still required for single events

The existing solutions to the trajectories problem for the period 2028 – 2043 have been provided by Mission Analysis, 2.4.

Low-thrust trajectories have not been studied, since the mission case shall be simple.

The typical solutions are the opposition class trajectory (long duration, low ΔV), the conjunction class mission (short duration, high ΔV) and missions including swing-bys at Venus (medium duration, medium ΔV).

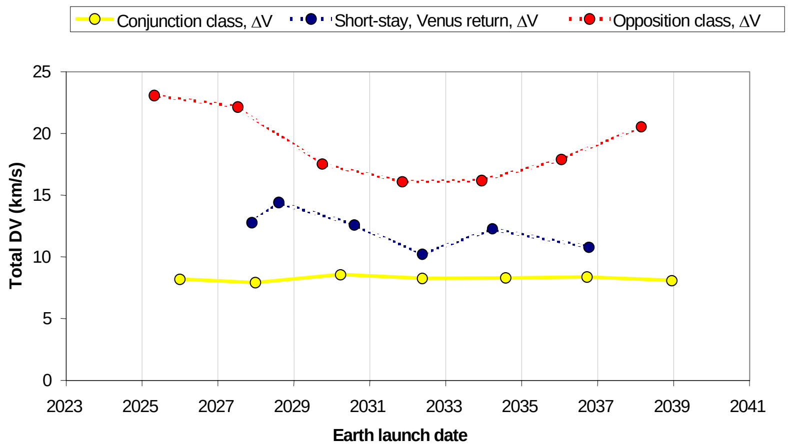


Figure 2-18: ΔV required for the different solutions

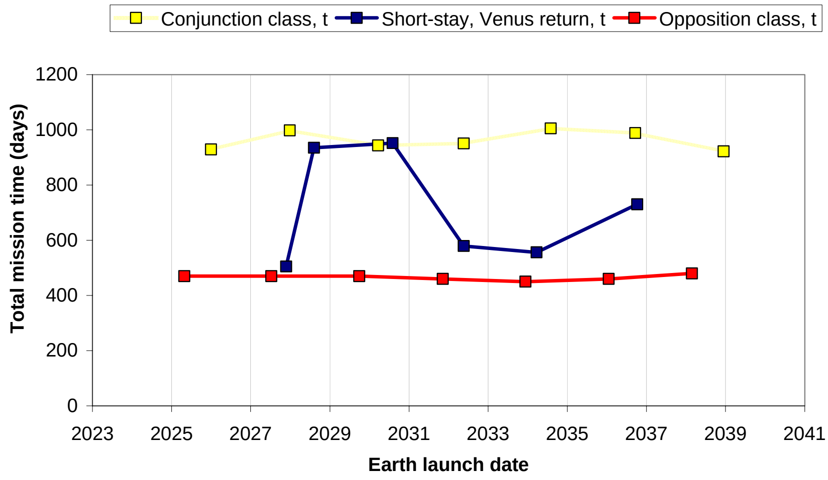


Figure 2-19: Mission duration for the different solutions

The mass efficiency of the mission has been traded against the three types of possible mission for the 2033 opportunity. This is considered to be a representative opportunity for all the mission classes. The trade-off is based on the analysis of the following parameters: total ΔV, possibility for surface permanence (time), radiation dose, and consumable mass. For the trade-off the dry mass of the THM has not been modified.

The results of this trade-off are shown in Table 2-12:

	Conjunction	Opposition	Venus Swing-by
Total Mission Duration (days)	963	376	579
Possible surface Duration (days)	533	30	28
ΔV (m/s)	8368	15120	10230
Radiation Dose (GCR,Sv,BFO)	1.087	0.496	0.756
Consumables (tonnes)	10.2	4.2	6.4
Mass to LEO (tonnes)	1336	45938	2481

Table 2-12: Trade-off for different trajectories class

As shown in Table 2-12, for the opposition and Venus swing by class missions, the stay duration around Mars is only 30 days, making it quite difficult to perform a landing, from an operational point of view.

In terms of total mass to LEO, the opposition and the Venus swing-by class missions present the highest masses.

In terms of radiation, the GCR environment was analysed. The radiation dose increases with the mission duration, but it is still under the limits for all the missions, although the conjunction class mission receives the higher dose.

Taking into account the mass efficiency and the surface opportunities, the conjunction class mission was selected.

All trip opportunities from 2028 to 2043 have been analysed (the opportunities recur on a 15 year cycle basis). For the mission case one of them has to be selected. (The relevant data can be seen in Table 2-2). The 2033 date has been taken as reference for the mission case.

Depending on the opportunity, the mission duration and its split in between trip duration and time around Mars varies, as shown in the following examples.

T total maximum = 1031 days in 2043
T total minimum = 956 days in 2033
Difference = 75 days
Deep Space maximum = 655 days in 2041
Deep Space minimum = 413 days in 2033
Difference = 242 days
Around Mars maximum = 553 days in 2033
Around Mars minimum = 342 days in 2039
Difference = 211 days

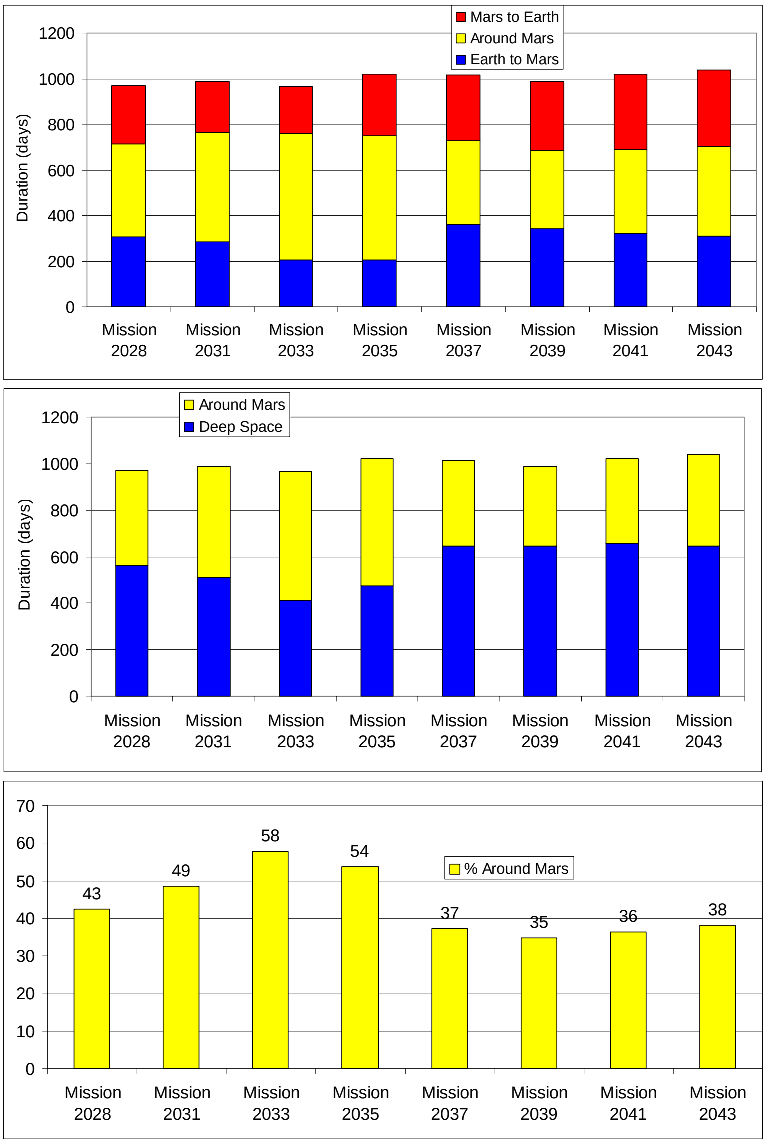


Figure 2-20: Total mission duration, trip duration and stay time around Mars and ratio

The total mission duration varies only slightly during the whole cycle, but the trip durations and the stay duration around Mars do have significant variations, particulary the ratio time around Mars over trip duration. From this point of view, the 2033 opportunity offers the longest stay

duration around Mars, and also the highest ratio, increasing the exploration opportunities. This opportunity also minimises the time spent in deep space.
The following shows an example of difference in energy requirements for the different dates:

TMI+MOI+TEI max = 8593 m/s in 2037
TMI+MOI+TEI min = 7721 m/s in 2041
Difference = 872 m/s
TMI max = 4005 m/s in 2037
TMI min = 3619 m/s in 2041
Difference = 386 m/s
MOI max = 2613 m/s in 2031
MOI min = 2010 m/s in 2041
Difference = 603 m/s
TEI max = 2595 m/s in 2035
TEI min = 1955 m/s in 2043
Difference = 640 m/s

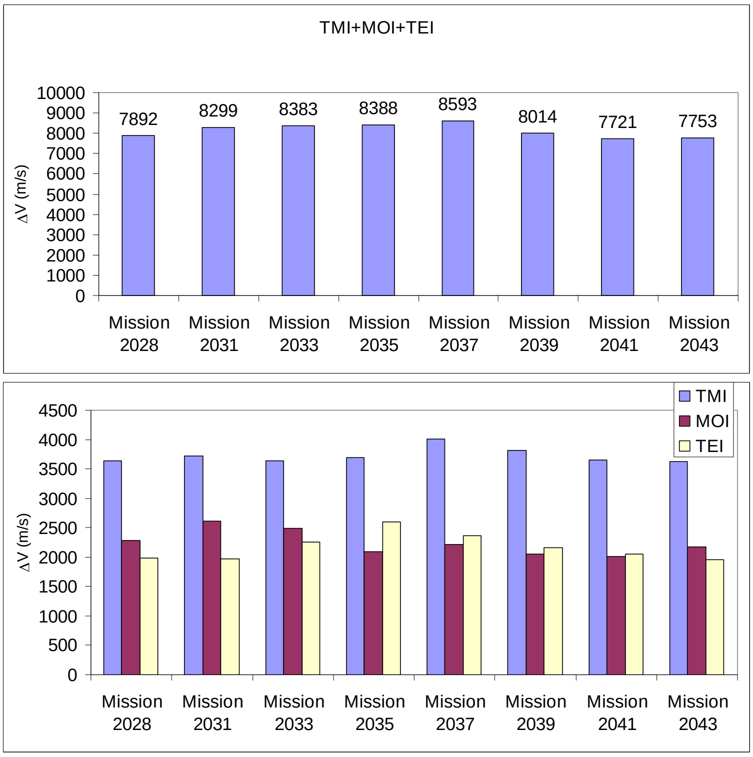


Figure 2-21: Energy requirements

The difference during the cycle is slight, around 10%. Therefore, any of the trip opportunities is representative enough for the mission case.
Concerning the entry velocity at Earth arrival the differences are:
V entry max = 12515 m/s in 2028
V entry min = 11472 m/s in 2037
Difference = 1043 m/s
Apollo Entry Velocity = 11200 m/s

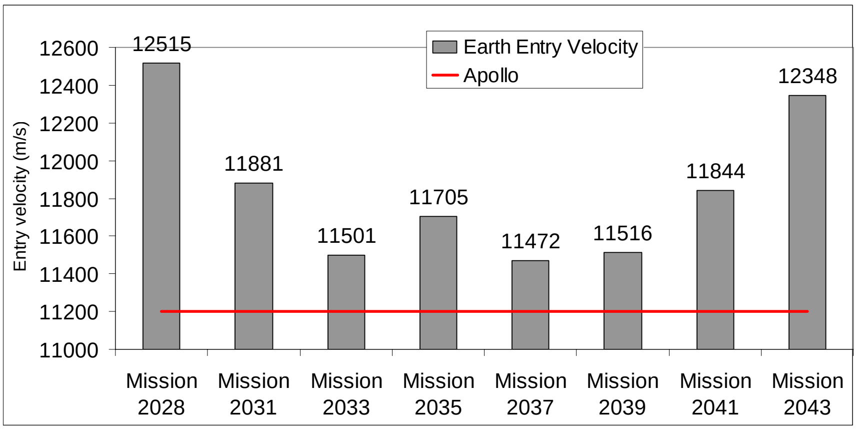


Figure 2-22: Entry velocity at Earth

The entry velocity does not vary greatly either, it ranges from 11.4 km/s to 12.5 km/s (atmosphere rotation not taken into account). 12.5 km/s is taken as design point for the ERC so the design will fit for any mission opportunity.
A summary of the mission data for the reference case is shown in Table 2-13:

Phase	Duration (days)
Departure	08 April 2033
Earth departure window	21
Earth to Mars	207
Mars arrival	11 November 2033
Around Mars	553
Mars departure	28 April 2035
Mars departure window	21
Mars to Earth	206
Earth arrival	27 November 2035
TOTAL in space	413
TOTAL mission	963
% around Mars	58
Kick	**ΔV (m/s)**
TMI	3639
Hyperbolic Earth escape velocity	3200
MOI	2484
Hyperbolic Mars arrival velocity	3413
HEO insertion	1187
TEI	2245
Hyperbolic Mars escape velocity	2990
EOI	3598
Hyperbolic Earth arrival velocity	3052
Earth Atmosphere Entry Velocity	11505
TMI+MOI+TEI	5884
TMI+TEI	8383
TOTAL	11966

Table 2-13: Mission 2003 opportunity relevant data

This opportunity minimises the total mission duration as well as the time spent in deep-space (inbound and outbound trips) and maximizes the time spent around Mars. Therefore, it maximizes also the ratio time around Mars / time in deep-space. Finally, it offers one of the lowest entry velocities on return to Earth, although the approach followed for the ERC design reduces the influence of this parameter.

2.7.6.2 Surface stay duration

One of the objectives of the study is to select the simplest mission case.
A long stay duration on the surface would imply a much higher complexity of the mission, as more resources and infrastructure would be required to support the astronauts while on the surface, typically more complex life support systems, more consumables, more habitable volume and higher power demands in general.
This increment in the mass of equipment required for a long stay would imply the definition of a cargo mission to take all the extra infrastructure. This would lead to the new requirement for the lander, of high precision landing, as the astronauts will have to rendezvous with the infrastructure on the surface.
To avoid this complication, a short stay duration on the surface of Mars has been selected.
After landing, one week is required by the astronauts to recover from the deconditioning, and it is assumed another week for the launch preparation. Taking into account the recommendations for the surface operations, seven EVAs are required as minimum. A surface stay of 30 days is therefore a minimum reasonable time.

2.7.6.3 Propulsion

The propulsion technologies for a human mission to Mars can be reduced to the following:

- Chemical propulsion
    - Cryogenic
    - Storable
- Solar electric propulsion
- Nuclear electric propulsion
- Nuclear thermal propulsion

According to the criteria defined for the mission case selection, electric propulsion has not been studied, to keep the complexity low. For the same reason and because it is not a mature technology (reduced knowledge which leads to not reliable estimations) nuclear propulsion has been also discarded.
Therefore the choice remains between cryogenic and storable propulsion systems, as they are well known technologies and therefore offer a good starting point for analysis.
The benefits of cryogenic propulsion is high Isp, which allows a significant reduction in the propellant mass required for a given ΔV and payload. But the drawbacks are the boil-off of the cryogenic propellants and the volume of the tanks due to the low density of the propellants.
The propulsion module design chosen follows a modular approach, combining both the cryogenic and the storable system in the same mission. Three cases have been defined; all storable, all cryogenic, and cryogenic for the first manoeuvre (TMI) and storable for the remaining two (MOI and TEI).

	Storable (Isp 345)	Cryogenic (Isp 450)	Cryo + Storable
Mass to LEO (tonnes)	1728	969	1336

Table 2-14: Mass to LEO for different propulsion technologies

As shown in Table 3-15, the use of cryogenic propellant for all the mission can reduce the initial mass to less than 1000 tonnes, but the problem of boil-off remains. An intermediate solution was adopted: cryo propulsion system for the first propulsive manoeuvre (TMI) and storable for the other two (MOI and TEI). This approach will allow a mass reduction in LEO and enables a possible analysis of the two types of propulsion technologies in the framework of a mission to Mars.

Boil-off represents the main problem in cryogenic systems. To reduce the volume of the propellant (LOX and LH2), you must store them in liquid phase. For that purpose there are only two solutions: keep them at cryogenic temperatures, or compress them to high pressure. The second option has a big disadvantage in terms of mass, as the tanks have to support the internal pressure. Therefore the best solution is to keep the propellants at low temperatures.

To completely isolate the propellants at cryo temperatures from any source of heat is practically impossible, therefore the propellants will undergo some phase change from liquid to gas. This propellant in gaseous form has to be depleted to avoid an increase in the internal pressure of the tank.

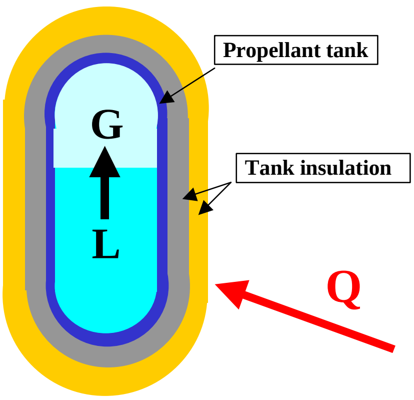


Figure 2-23: Boil-off process

During assembly of the vehicle in LEO, propellant mass is lost, and the ΔV capabilities of the vehicle are reduced. Loss of propellant can be compensated by launching more propellant before the composite departs. There is a point at which launching more stacks does not compensate the boil-off effect, as the new propellant launched is less than the mass lost by all the other stacks that are already there.

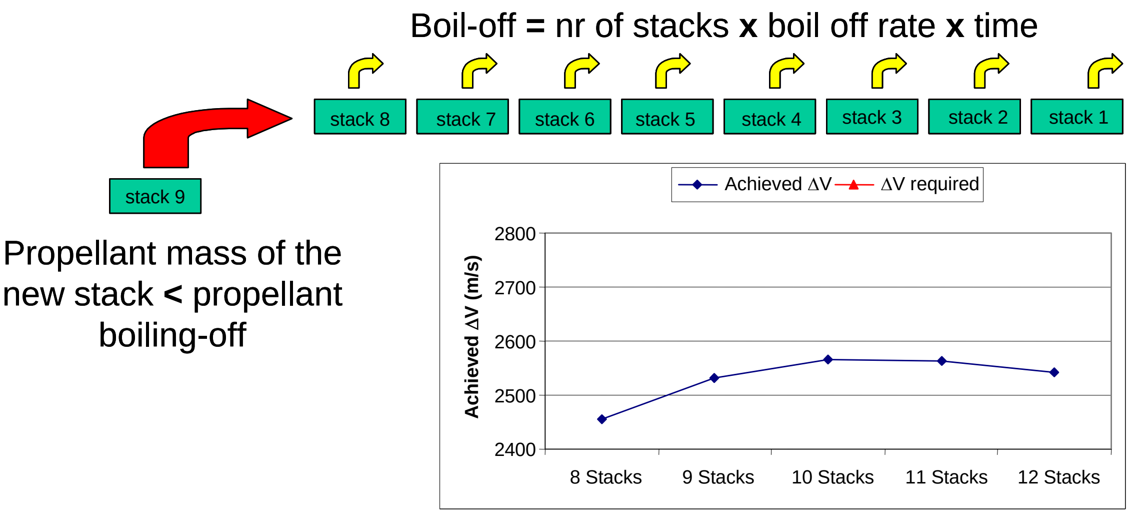


Figure 2-24: Boil-off effects and ΔV capability loss

The main parameters playing a role in the boil-off process are:

- Boil-off rate, namely the mass of propellant lost per unit of time, which depends on the design of the system
- Time prior to the usage of the propulsion stage, which mainly depends on the assembly time and therefore, on the launcher rate and the commissioning time

An analysis was carried out to assess the influence of all the parameters involved in the boil-off process. The main contributor to the DV capabilities losses was the boil-off rate, but the launch rate and the commissioning time before departure also had an impact.

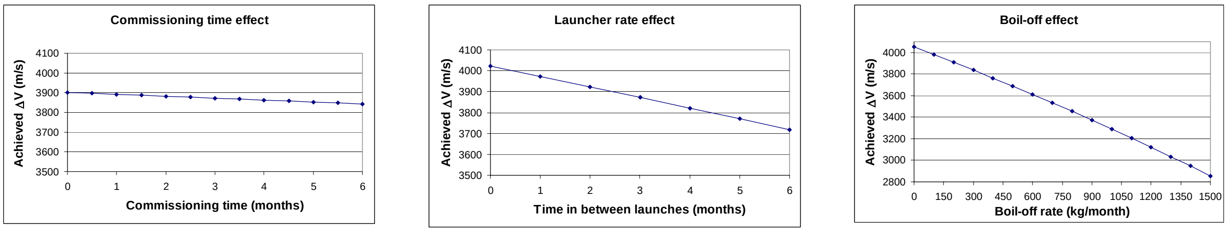


Figure 2-25: Effect of different parameters on the boil-off (11 stacks with a payload of 470 tonnes)

Two scenarios have been analysed in more detail, fixing the commissioning time to three months and varying the time between launches between 1 and 3 months. The results have been compared with the storable propellant scenario, which provides the limit in terms of mass efficiency.

Figure 2-26 shows the influence of the boil-off rate. The figure of merit represented is the number of propulsion modules (80-tonne cryogenic propulsion stacks) required to insert the payload into its trajectory to Mars. In the case of 3 months in between launches, a boil-off rate higher than 800 kg per month makes the mission not feasible with cryogenic propulsion. On the other hand, if only storable propellant is used, the number of stacks required is 16.

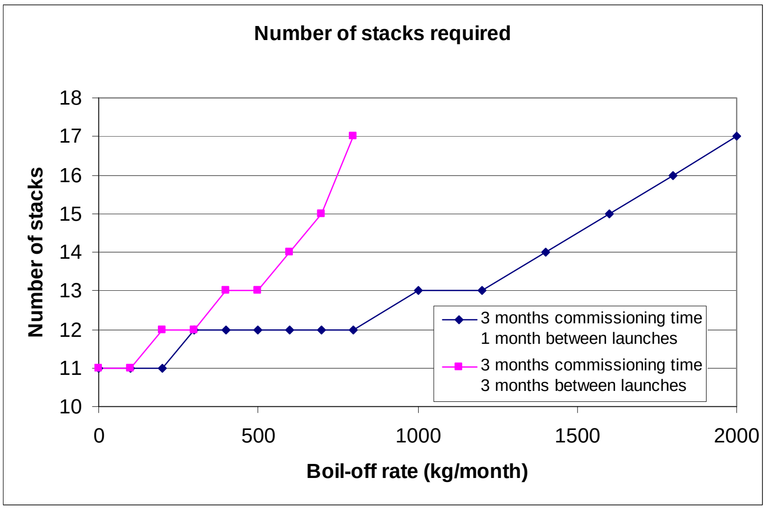


Figure 2-26: Number of stacks required as a function of launcher rate and boil-off rate

The analysis led to a boil-off requirement rate of 500 kg/month. Figure 3-31 shows that the maximum number of stacks in the worst case is 13.
Cryogenic propulsion systems could be extended to other mission phases, i.e. MOI, if we take into account that the time from departure to the MOI is around 6 months and the assembly time is already more than 54 months, and also that the interplanetary environment is more benign than the LEO environment from the thermal point of view. However, for the design case, the conservative case of only cryogenic TMI has been taken.

2.7.6.4 Return approach

Several strategies can be adopted for the return to Earth. The crew will return from Mars in the THM and ingress in the ERC to perform the atmospheric entry descent and landing. You could insert the THM into Earth orbit so it can be reused in a next mission. If not, it has to be discarded so that it does not impact the Earth. Besides, the ERC with the crew inside can also be inserted into an Earth orbit allowing the crew transfer to an orbital space station (e.g. ISS) before landing, or it can perform a direct entry. Table 3-16 shows a preliminary analysis of these possibilities:

	THM and ERC inserted	THM discarded, ERC inserted	THM discarded, ERC direct entry
Mass to LEO (tonne)	4892	1813	1336

Table 2-15: Mass to LEO for the different strategies on Earth return

As shown in Table 3-16, parking the THM and/ or ERC in Earth orbit is too expensive in terms of mass to LEO, as the propellant to perform the manoeuvre has to be taken from Earth to Mars and back (other techniques as aerocapture/aerobraking have not been analysed for this manoeuvre).
Planetary protection issues imply, no contaminated vehicle should remain in Earth orbit, and this is the case of the THM, as its exterior will be contaminated by the Mars Ascent Vehicle (MAV). In the case of the ERC, which will be also contaminated, the problem is overcome by the sterilization arising from high temperatures during reentry.

Therefore, direct entry of the ERC is selected as it presents a lower complexity and scientifically reduces the mass to LEO.

2.7.6.5 Orbit insertion and acquisition around Mars

Several technologies or strategies can be used for the orbit insertion around Mars. Preliminary analysis have been performed for these cases:

- Propulsive manoeuvre, using storable bipropellant technology.
- Aerobraking, no configuration changes assumed, orbit insertion performed by means of a propulsive manoeuvre, storable technology, 120 m/s added for orbit corrections
- Aerocapture, no configuration changes assumed, no heat shield included, 120 m/s added for orbit corrections

	Propulsive	Aerobraking	Aerocapture
Orbit insertion duration (days)	<15	100 - 2920	< 10
Time available for landing	Always	Only in some cases*	Always
Mass to LEO (tonnes)	1336	943	599

Table 2-16: Orbit insertion around Mars (see Surface opportunities)

Aerocapture could allow large reductions in mass, but requires dedicated complex analyses and has deep implications for the configuration of the vehicle (lift required). Detailed studies are required to further analyse this option. It has not been considered suitable at this stage of the mission.

Aerobraking can also provide reductions in mass, but requires long times for orbit acquisition and reduces the opportunities for surface operations. In some cases the landing was not even possible.

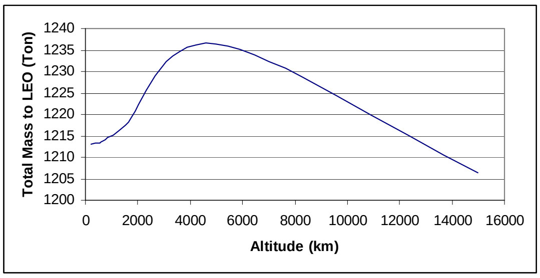
Propulsive braking is the most expensive in terms of mass, but is a technique that is well known. Therefore, it has been selected as the preferred option for the mission case.

2.7.6.6 Orbit around Mars

No detailed trade-off has been performed. Two main options can be considered: an elliptical orbit and a circular orbit.

An elliptical orbit could provide (unassessed) reductions in mass, as the final circularisation does not need to be done and the energy required to depart to Earth is lower. However it would imply a rendezvous and docking with the MAV in an elliptical orbit, which will increase the complexity of the mission. It will also require a bigger and more complex MAV to achieve the parking orbit. Finally, the radiation shielding that the planet can provide to the vehicle while in orbit around Mars would be reduced.

To reduce the complexity, a circular orbit around Mars has been selected. As regards orbit altitude, a preliminary analysis has been performed analysing the radiation dose (computed for nominal shielding for the overall mission), communications (both with Earth and Martian surface), MEV preliminary design, MAV propellant requirements for the ascent, propellant requirements for TV insertion and departure from different orbits and orbit decay. Orbits ranging from altitudes of 200 to 1500 km have been studied.



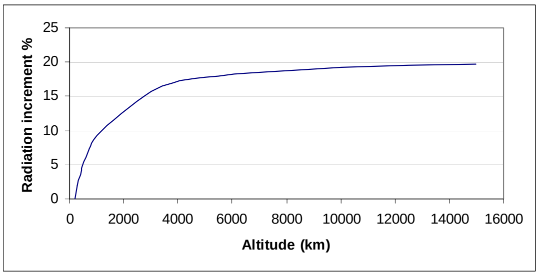


Figure 2-27: Orbit around Mars

Figure 3-32 shows on the left that the variation in the total mass to LEO versus altitude is negligible. The dose received by the astronauts above the low Mars orbit, as the effectiveness of the planet shielding is reduced. Orbit decay is not a problem above 500 km.
The baseline of 500 km circular has therefore been chosen and the inclination of the final orbit has been taken as the optimal one for departure.

2.7.6.7 MEV release

There are two possibilities for releasing of the MEV: while still in high elliptical orbit before the final orbit acquisition, or once the orbit has been circularized. In principle, the first option and reduce mass, as the mass to be put into the circular orbit is lower.

	MEV released from HEO	MEV released from LMO
Mass to LEO (tonne)	1312	1336

Table 2-17: Mass to LEO as function of the MEV release strategy

Release from high elliptical orbit provides very little mass advantage, which is not compensated by the increment in complexity. Furthermore, the TV would have little time (the surface stay time) to reach the final rendezvous orbit with the MAV from release, which significantly raises the risk of the mission. Finally, release may not be possible at arrival because of dust storms, so the TV would have to wait in HEO until the landing is feasible.
A MEV release from the final circular orbit has been therefore selected.

2.7.6.8 All-up / Split scenarios

An all-up scenario is defined as one in which the mission composite is sent to Mars in one vehicle, with all the required vehicles and infrastructures in one go. The split scenario is the one in which some of the infrastructures and/or vehicles are sent in a different composite vehicle. Two possible split scenarios have been considered:

1. Two identical vehicles with three crew members each, one carrying a MEV and both carrying an ERC.
2. One vehicle including THM+ERC and one only the MEV, which will have to rendezvous with the THM+ERC in Mars orbit before descent.

The means to come back and consumables have to be included in every crewed vehicle. In this way the crew can survive if any of the rendezvous manoeuvres fails.

	All up	Split case 1		Split case 2	
		Crew of 3	Crew of 3 + MEV	Crew	MEV cargo
Mass to LEO (tonne)	1336	837	984	1189	236
Total (tonne)	1336	1821		1425	

Table 2-18: Masses for all-up and split scenarios

Split scenario 1:
The overall mass is larger than in the all-up case, and both composites will have to be assembled in orbit at the same time, introducing more complexity in operations.
Split scenario 2:
The mass of the THM composite would be lower than in the split case but the overall mass in LEO is higher (more launches). The only advantage would be the reduction in assembly time of the THM composite which will reduce boil-off losses, but the overall assembly time will be increased.
For the mission case the all-up scenario has been selected, as it looks the most simple and effective.

2.7.6.9 Artificial gravity / centrifuge

The overall mission duration is around 1000 days, therefore some means will have to be provided to reduce the effects of such a long exposure to microgravity.
To provide these countermeasures, artificial gravity can be generated by two methods: either spinning the whole spacecraft or providing it with a centrifuge for crew exercise.
The spinning spacecraft option was found to lead to a very complex configuration with limited benefits and was therefore discarded for the mission case.

2.7.6.10 Crew number

The number of crew is one of the mission drivers, but it has not been traded off during the present study. A crew of six people has been adopted since the beginning of the mission following recommendations from the customer and due to similarities with other, comparable studies.

2.7.6.11 Summary

Table 3-20 lists the concluded trade-offs and options:

Trade-offs	Options
Trajectory	Conjunction Opposition Venus swing-by Low thrust
Surface stay duration	Long stay Short stay
Propulsion	Chemical Storable Cryogenic NTP SEP NEP
Return approach	THM and ERC inserted around Earth THM discarded, ERC inserted THM discarded, ECR direct entry
Orbit insertion around Mars	Propulsive Aerocapture Aerobraking
Orbit around Mars	Circular High elliptical orbit
MEV release	From circular orbit From high elliptical orbit
Split / All up	Split Scenario All-up Scenario
Distributed/localised artificial gravity	Spinning spacecraft Centrifuge
Crew number	Six

Table 2-19: Trade-off conclusions

2.7.7 Vehicle architecture

2.7.7.1 Mission elements

The main mission elements are:
Transfer Vehicle (TV): This element includes the Transfer Habitation Module (THM), which will provide accommodation and life support to the crew during the whole mission, transfer to Mars, orbital operations around Mars and transfer from Mars, and the propulsion modules which will provide the required ΔV for the mission: TMI, MOI and TEI.

Mars Excursion Vehicle (MEV) : This element includes the Mars Ascent Vehicle (MAV), which transport its the crew from the surface of Mars to an orbit, where it will rendezvous with the TV, the Surface Habitation Module (SHM) and the Descent Module (DM), which includes the entry, descent and landing system.

Earth Reentry Capsule (ERC): This is the element that will return the crew from its interplanetary return trajectory (after crew transfer from the TV to the ERC) to the Earth's surface.

2.7.7.1.1 Transfer Habitation Module

This mission element is the core of the mission because it is the one that will provide the habitable volume for the astronauts during most of the duration of the mission. It is composed of a central cylinder, which houses most of the facilities and equipment, and two nodes that act as connection points with the rest of the mission elements and also provide extra volume for the crew.

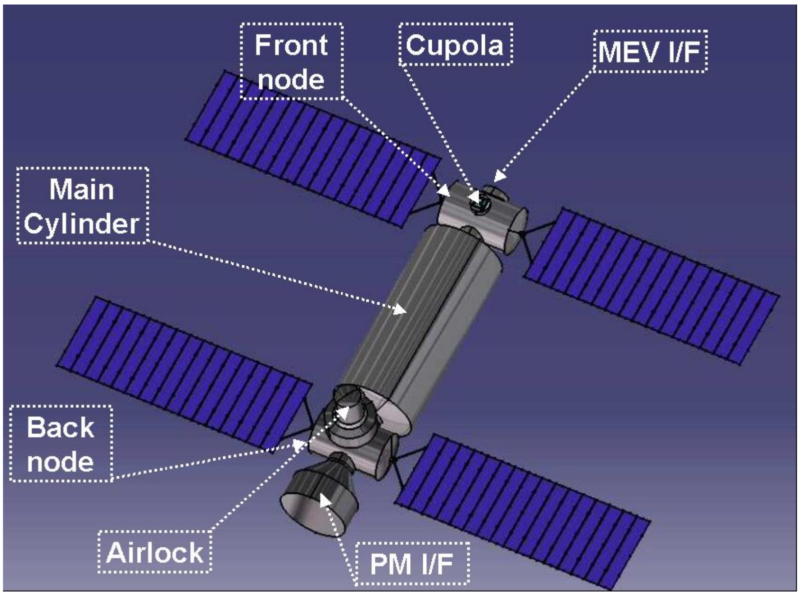


Figure 2-28: Transfer Habitation Module

The interior volume is split into the crew quarters, command module, hygiene facilities, training facilities, laboratory and social area, providing storage volume all along the vehicle.
It also provides interfaces with the MEV and ERC as well as an airlock to allow EVAs and spare docking port. Mechanical interfaces with the propulsion modules are also provided.
The main characteristics and dimensions are shown in Table 2-20:

Characteristic	Value
Overall mass (tonnes)	66.7
Consumables mass (tonnes)	10.2
Total pressurised volume (m^3)	480
Overall length (m)	about 20
Main cylinder diameter (m)	6
Nodes diameter (m)	3.5
Nodes length (m)	5.2
Solar arrays (m *x* m)	5.1 x 15

Table 2-20: THM properties

For more detailed information, see section 3.2.1 Transfer Habitation Module.

2.7.7.1.2 Propulsion module

The propulsion module is split into three submodules, one for each main propulsive manoeuvre, as shown in Figure 2-29.

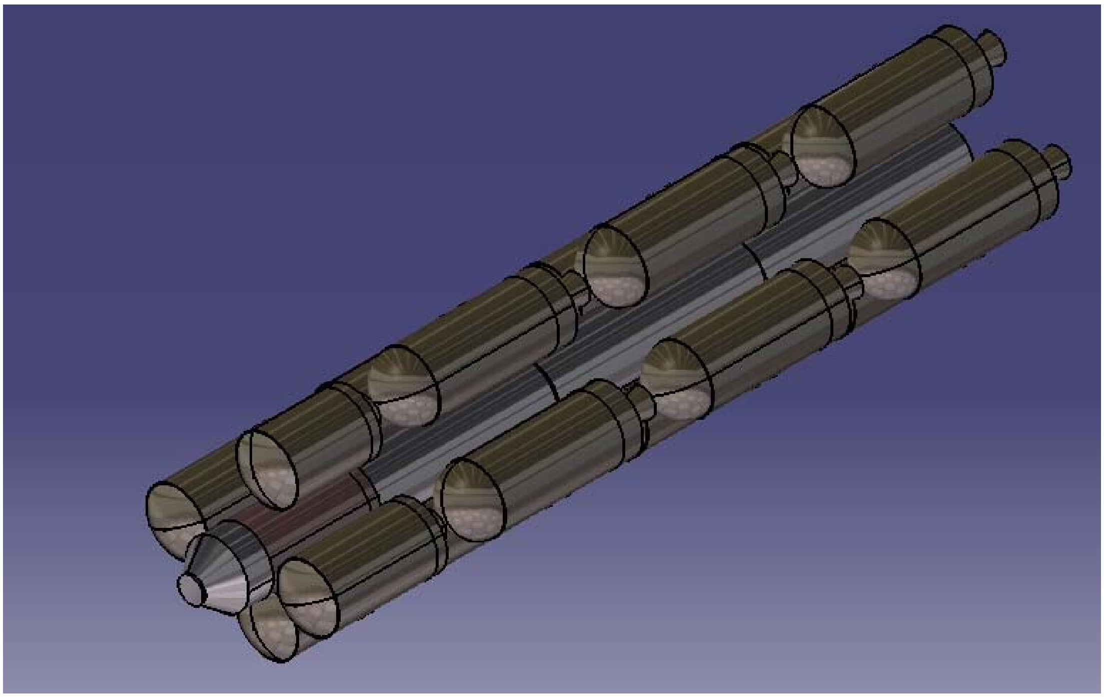


Figure 2-29: Propulsion module

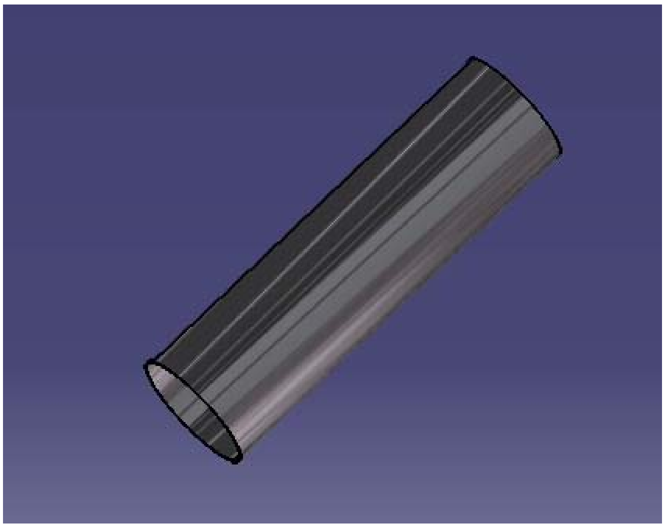
2.7.7.1.2.1 TMI

It provides the required ΔV for the trans Mars injection manoeuvre. It is composed of three serial stages, each of them split into four identical stacks and a supporting structure. Cryogenic propulsion technology has been selected for this module.
Each stack has a wet mass of 80 tonne to fit into Energia, and it is equipped with a single engine and the corresponding oxygen and hydrogen tanks.

Stack characteristic	Value
Overall mass (tonnes)	80
Propellant mass (tonnes)	70.7
Engine	Vulcain 2
Thrust (kN)	1300
Diameter (m)	5
Length (m)	13.6

Figure 2-30: TMI stack characteristics

The core supporting structure has a cylindrical shape, as shown in Figure 2-31:



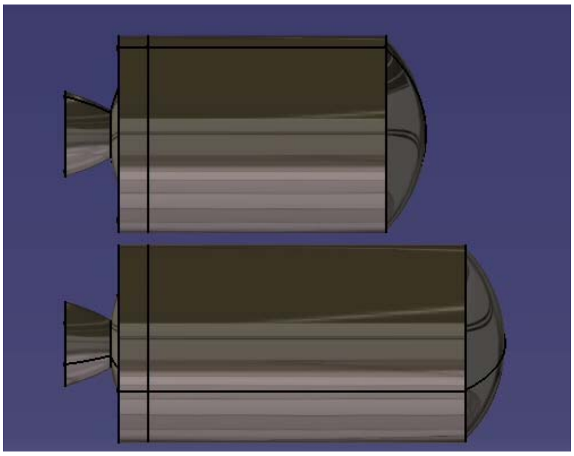
Supporting structure characteristics	Value
Overall mass (tonnes)	5.2
Diameter (m)	5
Length (m)	16.8

Figure 2-31: TMI supporting structure characteristics

The four stacks and the associated supporting structure are discarded once the burn is performed. The used stage will be provided by a small propulsion system that will allow the required manoeuvres for a controlled entry in the case of the first and second stage, or put it into a trajectory the does not impact Mars or the TV in the case of the third stage.

2.7.7.1.2.2 MOI

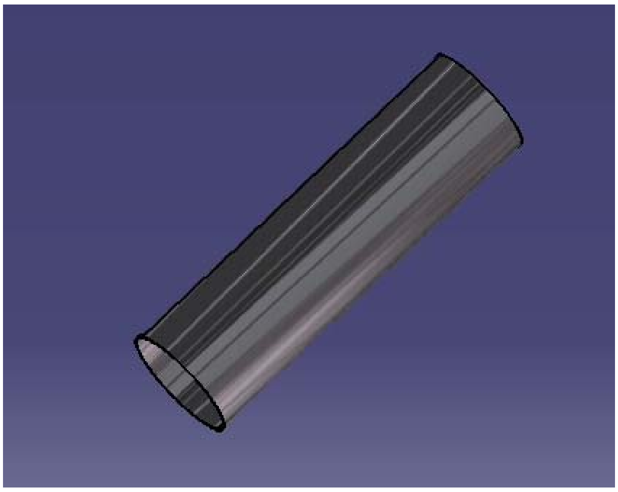
It provides the required ΔV for the Mars orbit injection and final orbit acquisition manoeuvres. It is composed of two stages, each of them split into two identical stacks, and a supporting structure. Storable propulsion technology has been selected for this module.
The first stage (orbit insertion) is composed of two stacks of 80 tonnes, while in the case of the second stage (final orbit acquisition, it is composed of two stacks of 50 tonnes each.



Stack characteristics	Value
Overall mass (tonnes)	80
Propellant mass (tonnes)	76.3
Diameter (m)	5
Length (m)	about 9.5
Overall mass (tonnes)	50
Propellant mass (tonnes)	47.7
Diameter (m)	5
Length (m)	about 8
Engine	RD-0212
Thrust (kN)	612

Figure 2-32: MOI stack characteristics

The four stacks are attached to a supporting structure with a cylindrical shape.



Supporting structure characteristic	Value
Overall mass (tonnes)	3.6
Diameter (m)	5
Length (m)	9

Figure 2-33: MOI supporting structure characteristics

Once used, the stacks of the first stage are jettisoned. Once the stacks of the second stage are used, they and the supporting structure are jettisoned. The used stacks of the first stage will be provided with a small propulsion system that will allow the required manoeuvres to avoid the collision with Mars (raise the pericentre).

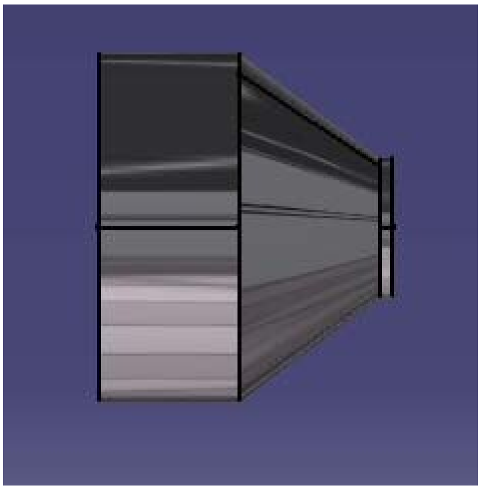
2.7.7.1.2.3 TEI

It provides the required ΔV for the trans Earth injection manoeuvre. It is composed of one single stage, composed of a 80-tonne stack and a supporting structure. Storable propulsion technology has been selected for this module. The design of the stack is identical to the one of the first stage of the MOI.

Stack characteristics	Value
Overall mass (tonnes)	80
Propellant mass (tonnes)	76.3
Engine	RD-0212
Thrust (kN)	612
Diameter (m)	5
Length (m)	about 9.5

Figure 2-34: TEI stack characteristics

The stack will be placed inside the supporting structure of the MOI, attached to a conical adapter to the THM.



Supporting structure characteristics	Value
Overall mass (tonnes)	5.2
Initial diameter (m)	about 2
Final diameter (m)	about 2
Length (m)	5

Figure 2-35: TMI supporting structure characteristics

2.7.7.1.3 Mars excursion module

This mission element is the one that allows a crew of three astronauts to land on the surface of Mars and take off after 30 days to rendezvous and dock with the TV. It is composed of three main elements:

- Descent Module, mainly composed of the deorbit propulsion system, inflatable heat shield, back cover and parachutes for the entry and descent.
- Surface Habitation Module, a cylindrical module that will house the astronauts during their stage on the surface providing life-support systems and EVA equipment to perform the exploration. The landing systems (retrorockets and landing legs) are located in this module. It will also provide the interfaces with the MAV.
- Mars Ascent Vehicle, the ascent vehicle in which the astronauts will return to orbit, mainly composed of a capsule (in which the astronauts will be placed also during the descent) and a propulsion module split into two stages. It will provide life support for 5 days.

The MEV will be attached to one of the extremities of the TV, in a docking port on the longitudinal axis of the TV.

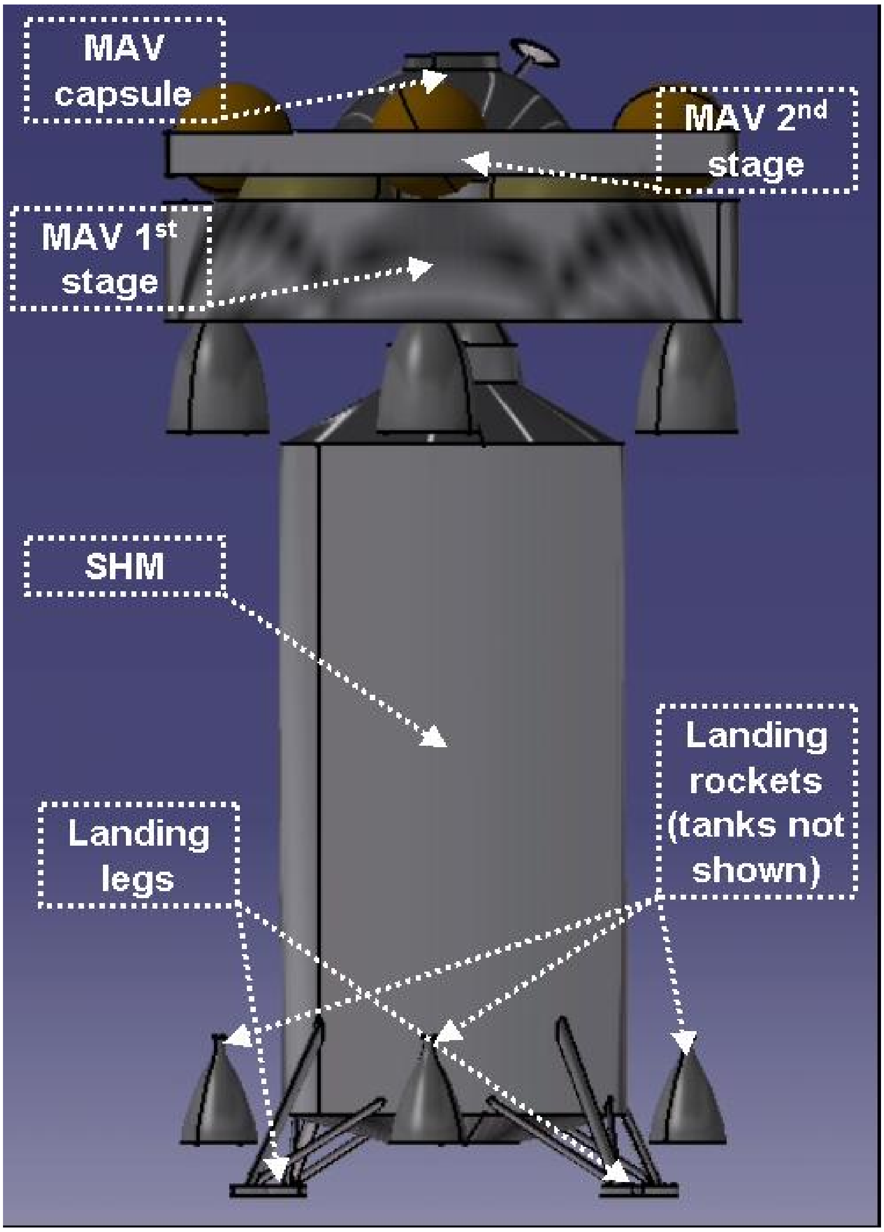


Figure 2-36: Mars Excursion Vehicle

The main characteristics and dimensions of the MEV are in Table 2-21.

Characteristics	Value
Overall mass (tonnes)	46.5
Consumables mass (tonnes)	0.3
Propellant mass (tonnes)	20.5
Total pressurised volume (m^3)	80
Overall length (m)	12.1
Overall diameter (m)	6

Table 2-21: MEV properties

2.7.7.1.4 Earth return capsule

This mission element will house the whole crew for the entry descent and landing on Earth return. It has not been designed within this study. A reference mass of 11.2 tonnes has been taken into account for the analysis. The figure came from a scaling up of an Apollo capsule.

2.7.7.1.5 Additional mission elements

For the mission, more infrastructure is also required:

- Mars Stationary Satellite for data relay with the surface.
- Workbenches for the assembly of the composite in LEO.

2.7.8 Launchers

To be able to run the analyses, it has been decided to use existing launchers for the mission case, as information can be gathered about them, instead of defining a new “theoretical” launcher being able to launch hundreds of tonnes to LEO.

2.7.8.1 Energia

The Russian launcher Energia has been selected as the main launcher for the mission. Although it is no longer in production, it was assumed that the effort to make it operational is smaller than the one to develop a launcher of such performances or even higher from scratch.
The Energia launcher was designed in the 1970s and made only two flights at the end of the 1980s, one with an external cargo (1987) and a second one with the Russian shuttle, Buran (1988). It is a two-stage launcher, consisting of central core and booster, with a lateral configuration for the payload.

No user manual for the launcher is available, but performances have been taken from international launchers guides. The performances and characteristics assumed are shown in Table 2-22:

Characteristics	Value
Overall mass (tonnes)	2400
Payload mass (tonnes)	80
Success rate (%)	100
Status	Out of production
Fairing length (m)	35
Fairing diameter (m)	6

Table 2-22: Energia launcher

The payload is inserted into a low orbit of 200 x 200 km from where it is pushed up to the final orbit by an RCS stage. This stage is assumed to have attitude control capabilities and will be used to perform the rendezvous and docking of the payload with the composite to proceed to the assembly.

2.7.8.2 Other launchers

Smaller launchers may be also used for the assembly. The launchers used and their main characteristics are shown in Table 2-23:

Characteristics	Values
Proton	
Mass to LEO (400 km)	20 tonne
Status	Operational
Ariane-5 (envisaged)	
Mass to LEO (400 km)	20 tonne
Status	Development
Soyuz	
Crew capability	three
Status	Operational
Space Shuttle	
Crew capability	seven
Status	Operational

Table 2-23: Launchers used

2.7.9 Mission phases

The mission phases for the study case are shown in Table 2-24 and Figure 2-37.

Mission Phase	Description	Event start	Event end	Duration (days)	Tasks performed	Modules involved TV	MEV	ERC	Assumptions
Launch and LEO Operations	Assembly takes place in orbit at 400 km altitude	1st Launch	All elements assembled and ready	About 1600	Assembly in LEO	**X**	**X**	**X**	
Check-out	Check out of the assembled vehicle by the commissioning crew (3) before final crew transfer	Arrival of commissioning crew	Return to Earth of commissioning crew	20 - 90	Checking of all the subsystems	**X**	**X**	**X**	
Crew transfer		Final crew launch into orbit	Final crew boarding	3		**X**	X	X	MEV, ERC hibernating
Crew training		Final crew boarding	Orbit insertion manoeuvre	30	Training on all subsystems, Exercise / Centrifuge, Communications, Maintenance & housekeeping	**X**	**X**	**X**	
Trans-Mars Injection		Orbit insertion manoeuvre	Transfer to Mars Orbit Acquisition	Less than 2	Stowage of the solar arrays TMI Burn	**X**	X	X	TMI burn split into three stages, discarded after its use MEV, ERC hibernating
Earth-Mars cruise	TV's attitude such that solar arrays are always illuminated	Transfer to Mars Orbit Acquisition	Entry into Mars sphere of influence	207	Deployment of solar arrays, Science Exp / Observation, Maintenance & housekeeping, Communications, Exercise / Centrifuge (about 2 hr/day), Navigation: Mid course corrections / manoeuvres	**X**	X	X	MEV, ERC hibernating

Mission Phase	Description	Event start	Event end	Duration (days)	Tasks performed	Modules involved			Assumptions
						TV	MEV	ERC	
Capture into Martian orbit	Initial orbit inmediately after MOI is high elliptic and has an inclination of 32 degrees	Entry into Mars sphere of influence	Mars orbit insertion	Less than 2	Stowage of the solar arrays Capture Burn	**X**	X	X	1.2 tonnes of waste dropped before the manoeuvre, first stage of MOI used and discarded MEV, ERC hibernating
Final orbit acquisition	The final circular orbit has an inclination of 32 degrees and it is at an altitude of 500km	Orbit insertion manoeuvre	Final orbit acquisition	< 2	Stowage of the solar panels Capture Burn	**X**	X	X	Second stage of MOI used and discarded MEV, ERC hibernating
Orbital operations		Final orbit acquisition	Earth return manoeuvre	553	Deployment of the solar arrays, Communications Exercise/Centrifuge Maintenance & housekeeping	**X**	**X**	X	ERC hibernating
Undocking, entry, descent and landing		Undocking of the MEV from TV	MEV Landing	about 0.125	Communications, Navigation	**X**	**X**	X	ERC hibernating
Surface operations		MEV Landing	MAV Take off operations	30	Communications, Exploration		**X**		Sample collection, handling and transported into the MAV (planetary protection)
Ascent		MAV Take off	Orbit insertion for Rendezvous	about 4	Communications, Navigation		**X**		SHM sealed and left in the surface
Rendezvous and docking		Orbit insertion for rendezvous	Docking with TV		Communications, Navigation	**X**	**X**	X	Sample transfer from MAV to TV, MAV discarded after docking, ERC hibernating
Trans-Earth Injection		Orbit insertion manoeuvre	Trans-Earth Orbit Acquisition	hours	Stowage of solar panels, TEI Burn, Communications, Navigation	**X**		X	3 tonnes of waste dropped before the manoeuvre, TEI stage used and discarded ERC hibernating

Mission Phase	Description	Event start	Event end	Duration (days)	Tasks performed	Modules involved TV	MEV	ERC	Assumptions
Mars-Earth cruise	TV's attitude such that solar arrays are always illuminated	Trans-Earth Orbit Acquisition	Earth orbit/trajectory acquisition manoeuvre	206	Deployment of solar arrays, Science Exp / Observation, Maintenance & housekeeping, Communications, Exercise / Centrifuge (about 2 hr/day) Navigation: Mid course corrections / manoeuvres	**X**		X	ERC hibernating
Earth orbit acquisition		Orbit insertion manoeuvre	Final Earth orbit / trajectory acquisition	hours	Communications, Navigation	**X**		X	Final Earth pointing, ERC hibernating
Undocking, descent and landing		Undocking of the ERC from TV	Earth landing	about 2	Communications, Navigation	**X**		**X**	TV discarded before entry and left a non-collision trajectory

Table 2-24: Mission Phases

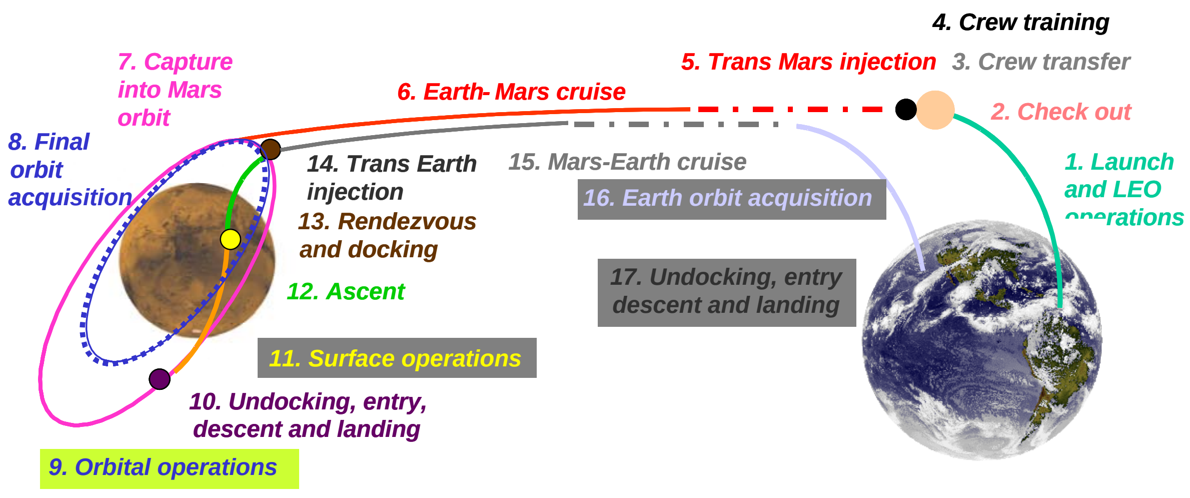


Figure 2-37: Mission phases

The mass evolution of the composite is shown in Table 2-25:

Phases	Initial Mass (tonnes)	Final Mass (tonnes)	Comments
1. Launch and LEO operations	0	1355	All the modules are launched and assembled in LEO
2. Check-out	1355	1355	All the systems are checked before departure
3. Crew transfer	1355	1355	
4. Crew training phase	1355	1355	
5. Trans-Mars Injection	1355	485	Three stages, discarded after its use, T/m from 3.8 to 9.8
6. Earth-Mars cruise	485	470	Cruise correction manoeuvres performed, waste discarded before MOI manoeuvre
7. Capture into Martian orbit	470	310	Stacks discarded after their use, thrust over mass from 2.6 to 3.8
8. Final orbit acquisition	310	204	Stage discarded after its use, thrust over mass from 4 to 5.6
9. Orbital operations	204	157	MEV discarded, waste discarded prior to TEI
10. Undocking, entry descent and landing	204	157	MEV detached from TV
11. Surface operations	45	23	Exploration, SHM discarded prior to ascent
12. Ascent	23	6	Propellant burnt, first stage discarded
13. Rendezvous and docking	6 157	157	MAV capsule discarded after its use
14. Trans-Earth Injection	157	74	Stage discarded after its use, thrust over mass from 4 to 7.7
15. Mars-Earth cruise	74	72	Cruise correction manoeuvres performed
16. Earth orbit acquisition	72	11.2	TV discarded
17. Undocking, descent and landing	11.2	11.2	

Table 2-25: Mass evolution

2.7.10 Mission performance

Table 2-26 shows a summary of the mission case:

Characteristics	Value
Crew	
Total number of crew	six
Number of crew landed	three
Masses	
THM mass (tonnes)	66.7 (wet) 56.5 (dry)
MEV mass (tonnes)	46.5
ERC mass (tonnes)	11.2
Consumables (tonnes)	10.2
Propellant (tonnes)	1083
Propulsion systems (tonnes)	130
Supporting structures (tonnes)	19.7
Total mass at Earth departure (tonnes)	1357
Sampled collected (kg)	65
Trajectories	
Earth departure	8 April 2033
Mars arrival	11 November 2033
Mars departure	28 April 2035
Earth arrival	27 November 2035
Duration of stay on the surface (days)	30
TMI ΔV (m/s)	3639
MOI ΔV (m/s)	2484
TEI (m/s)	2245
Earth atmospheric entry velocity (m/s)	11505
Launches	
Total number of launches	28
Total mass launched (tonnes)	1541
Assembly time (years)	4.6

Table 2-26: Mission case summary

The total mission duration since the launch of the first element is 7.2 years, of which only 2.6 correspond to the trip to Mars and back. From the 1541 tonnes that will be launched, only 1355 will depart to Mars, the rest being supporting structures, workbenches and tools for the assembly, and propellant boiled off. 89% of the mission mass corresponds to propulsion systems.

2.7.11 Surface opportunities

The mission architecture takes into account the periods of time during which it is possible to carry out an excursion to the Martian surface. A surface opportunity is considered to be any period of 40 days that does not coincide with:

- The *global dust storm season* on Mars. This occurs for approximately 3 months on either side of the Martian perihelion passage.
- *Superior conjunction* (i.e. the Earth and Mars are on opposite sides of the Sun). In this situation there is a communications difficulty that can last for up to 55 days (an angle of 10° either side of the Sun).
- *Martian winter* at the landing site *if power is supplied by solar arrays*. The power subsystem provides a cut-off associated with the size of the solar arrays. In winter the solar flux reaching the Martian surface is reduced so the lander would have insufficient power. The exact length of the excluded time period depends on the landing site latitude. There is a trade-off to ensure that the size of the solar arrays does not impact too greatly on other subsystems by providing more opportunities for surface stays. In the case of a fuel cell power source, the landing site and hence the Martian season are immaterial.

In addition there are several mission operations during which the surface stay cannot take place. These are:

- Aerobraking
- Final Martian orbit acquisition
- System check. These are assumed to last 1 week after arrival in the final Martian orbit and 2 weeks before departure on the return journey to Earth.
- MAV-TV rendezvous. This can take several days at the end of the surface stay.

The analysis took into account:

- Selection of the power source used, whether solar arrays or fuel cells.
- Selection of one of four landing site latitudes: 21°N, 12.5°N, 12.5°S and 22.5°S. However, in the case of using fuel cells, the landing site is immaterial.
- Selection of whether or not to include dust deposition on the solar arrays when considering the power constraints.

Figure 2-38 shows how the opportunities vary when the landing site and the aerobraking manoeuvre are taken into account. Note that the fuel cell case is the same as the best case of each of the solar cell launch window scenarios. Aerobraking manoeuvres are considered to last six months in this case also.

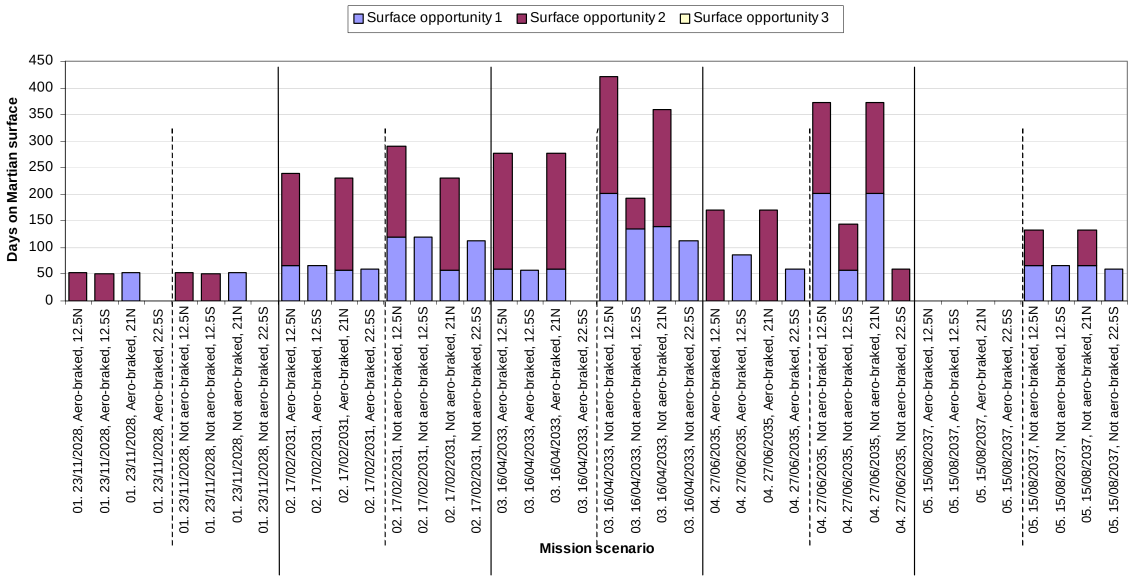


Figure 2-38: 40-day-plus surface opportunities, using a solar cell power system

Figure 2-39 shows the timeline for the architecture selected for the study case; 2033 opportunity, no aerobraking, and fuel cells as power generation on the surface. There are two surface opportunities greater than 40 days in length: 202 and 218 days.

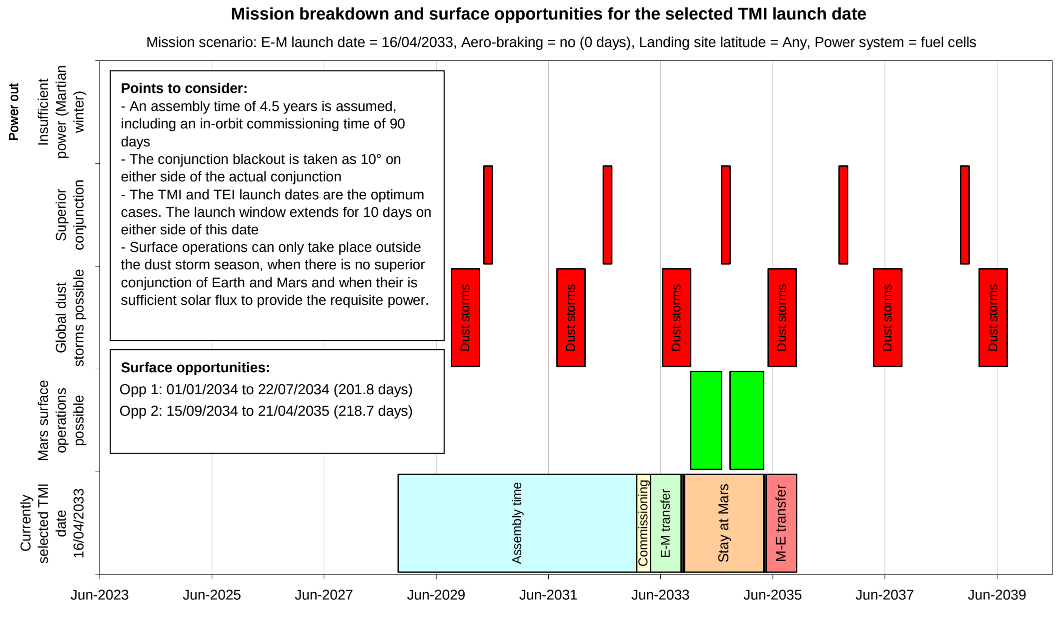


Figure 2-39: Mission timeline

Several conclusions can be drawn from the surface opportunities assessment:

- Generally, aerobraking is not recommended. If it is required it should be aimed to reduce the duration to the order of six months. For longer durations the landing opportunities are reduced, even non existing.
- When solar arrays are used as power generators on the surface, the opportunities are reduced and constrained by the landing latitude. This implies that some latitudes will not

be allowed. In this case northern latitudes are preferred as the dust storms coincide with the winter.

2.7.12 Abort Options

In the event of a failure or emergency that precludes completion as planned, the mission will have to be aborted. Two categories of failures have been considered:

- Failures in equipment/subsystems
- Failures in manoeuvres/operations

The first category is dealt with at subsystem design level by making the design fail safe. For the second category, mission-abort scenarios have been considered.
In the event of a mission abort, the original mission objectives shift to a safe return of the crew, as soon as possible and within the constraints dictated by the system design. Any defined abort scenario must be consistent with all budgets imposed for the various subsystems, e.g., propellant, thermal, structural, power or ECLSS consumables.
Firstly, the abort options for each phase of the mission require study. Then the cost (in terms of the budget of each relevant subsystem) must be quantified for each identified option. The outcome is to know what can conceivably be done to save the crew at each point of the mission.
The options for a mission abort strongly depend on the selected scenario. Here, the study is limited to the baseline scenario as studied for the 2033 launch opportunity.
First, the options per mission phase were analysed. Then, the abort cost for each option was quantified and listed.

2.7.12.1 Phase-by-phase analysis

The mission phases were studied chronologically, starting from Earth escape and ending with the return trajectory from Mars to Earth.

2.7.12.1.1 Abort during Earth escape

The Earth escape sequence is split into three manoeuvres, each of which further raises the apogee until a hyperbolic orbit with the required orbital parameters is achieved. Any of these burns can be prematurely terminated. For over 90% of the total manoeuvre duration, abort leaves the spacecraft still in a wide elliptical orbit around the Earth. The crew will have to wait until the next perigee to be able to then board the entry capsule and return to Earth's surface.

2.7.12.1.2 Fast abort during Mars transfer

The fast option requires a large DSM to insert the spacecraft into a trajectory that intersects the Earth's orbit. The nominal mission parameters are shown for comparison with the abort transfer characteristics in Table 2-27:

- Earth escape date: 8 April 2033
- Nominal Mars arrival date: 24 October 2033
- Nominal Earth arrival date: October - November 2035
- Nominal mission duration (maximum): 962 days
- Nominal Earth arrival velocity (maximum): 3.052 km/s

Abort decision date	ΔV [km/s]	Earth arrival date	Earth arrival velocity [km/s]	Duration [d]
1 April 2033	2.335	8 April 2034	1.9	365
1 May 2033	2.447	13 April 2034	2.8	370
1 June 2033	2.695	27 April 2034	4.6	384
1 July 2033	2.910	13 May 2034	6.4	400
1 August 2033	3.117	29 May 2034	8.1	416
1 September 2033	3.323	19 June 2034	9.5	431
1 October 2033	3.523	27 June 2034	10.9	445

Table 2-27: Fast abort transfer characteristics

The fast-abort transfer characteristics are shown in Table 2-27. The abort manoeuvre cost, despite rising for late abort, is always within the mission budget, as all propellant reserved for MOI, orbit circularization and TEI can be used for this purpose. The remaining ΔV capability after TMI is 5.2 km/s, once the MEV is discarded.

The duration is counted starting from Earth escape and increase from 1 year to 1.5 years if the abort decision is delayed. In terms of life support systems, there would be no problem as the system is designed for a nominal lifetime that is longer than the duration of the abort mission in any case. The time spent in deep-space can exceed the nominal duration, but in no case should the radiation limits be exceeded.

The shaded area indicates the potential problem with a late fast abort, the Earth arrival velocity is significantly larger than that in the nominal mission. This situation occurs 75 days after departure. One option here is to use the remaining ΔV capabilities to reduce the arrival velocity at Earth. If not, the ERC should be redesigned to cope with entry velocities as high as 17 km/s. This will require further study.

2.7.12.1.3 Slow abort during Mars transfer

The slow option retargets for a modified Mars swing-by. After that, another deep-space manoeuvre is required for retargeting to Earth arrival.

Abort decision date	DSM 1	ΔV -1 [km/s]	Mars swing-by	DSM 2	ΔV-2 [km/s]	Total ΔV [km/s]	Earth arrival date	Arrival hyperbolic arrival velocity [km/s]	Duration [d]
1 May 2033	22 October 2033	0.899	25 October 2033	21 January 2035	2.110	3.010	28 September 2035	5.280	904
1 August 2033	23 October 2033	0.895			2.111	3.005		5.284	
1 October 2033	24 October 2033	0.894			2.110	3.004		5.287	

Table 2-28: Slow abort transfer dependencies

Table 2-28 summarizes the dependencies of the slow abort. Note that the abort cost, which is within the mission manoeuvre budget, remains independent of the abort decision date, because the abort manoeuvre is best performed close to the arrival at Mars. A significant improvement of the efficiency could be achieved by allowing a powered swing-by. This is however not regarded here for the sake of obtaining conservative results. At any rate, abort could probably be performed even shortly prior to the arrival at Mars.

The total transfer duration, with 904 days, remains within the nominal mission duration. Nevertheless, the time spent in deep-space increases, and so does the radiation dose due to the GCR, although the limits are not exceeded.

The problem again is the hyperbolic arrival velocity, which is about 2 km/s higher than the nominal value. This translates into an increase of the entry velocity of the order of 0.8 km/s for the mission opportunity of 2033. The heat shield of the crew entry capsule was designed for an entry velocity of 12.5 km/s, which is not exceeded in this case.

2.7.12.1.4 Recovery from failed MOI

If MOI fails completely due to a malfunction in the propulsion system, the spacecraft will perform a swing-by at Mars and end up in a heliocentric orbit much different from that of the Earth-Mars transfer. This orbit does not intersect that of the Earth. A natural close encounter with the Earth occurs in 2038. A correction manoeuvre of 2.5 km/s would enable a arrival in June 2038, five years after departure from the Earth.

To fulfil the requirements of this abort scenario, major modifications would be required in the design of the THM. This analysis has not been carried out.

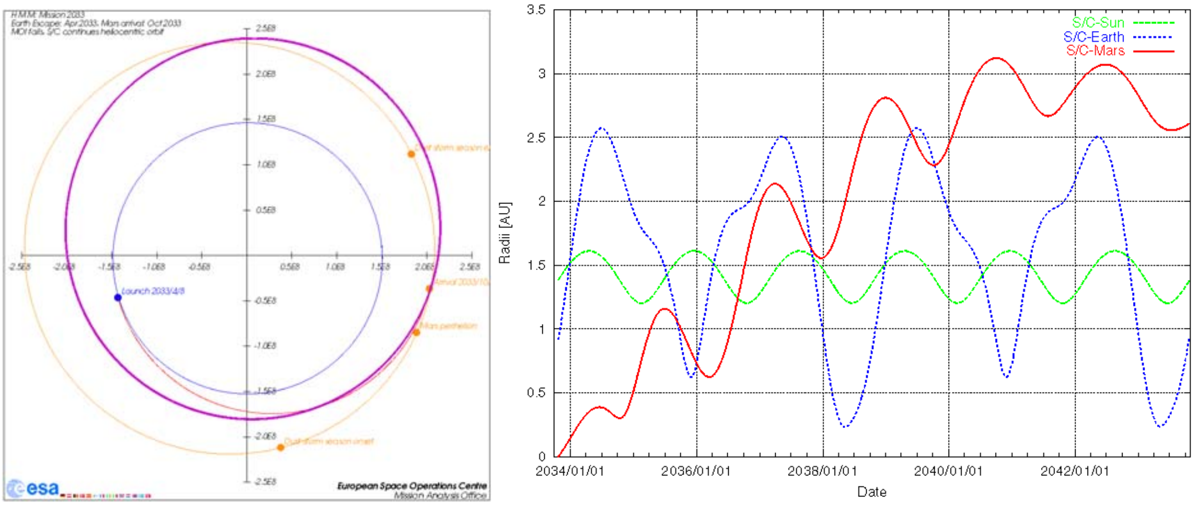


Figure 2-40: Conditions in case of failed MOI

Forcing an Earth encounter in June 2035 would require a correction manoeuvre of 5 km/s (available ΔV capability is 5.2 km/s). Arrival would occur at a hyperbolic velocity of 4.9 km/s, considerably faster than the nominal value. In this case the arrival velocity is not a problem, as the ERC is designed to cope with velocities up to 5.8 km/s.

2.7.12.1.5 Abort after MOI

There is a theoretical option for a return to Earth via a Venus swing-by in December 2033 (i.e., a few weeks after nominal arrival). The scenario involves the following characteristics and cost:

- Mars escape: 3 December 2033
- Hyperbolic escape velocity: 5.94 km/s
- TVI from 500 km orbit: 4.25 km/s, from HEO at least 3 km/s
- Mars-Venus-transfer: 179 days
- Venus swing-by minimum altitude: 6600 km
- Venus-Earth transfer: 188 days
- Earth arrival date: 5 December 2034 (around one year earlier than nominal)
- Total transfer duration: 367 days
- Earth arrival hyperbolic velocity: 4.74 km/s
- No deterministic midcourse manoeuvres

The total transfer duration is only one year, so arrival would occur one year earlier than for the nominal mission. The hyperbolic Earth arrival velocity is larger than in the nominal case but smaller than for the slow abort during Mars transfer (see above).

The problem is the high escape velocity required from Mars. Insertion into the Venus transfer from the 500 km final orbit around Mars would cost 4.25 km/s, from the initial HEO at least 3 km/s. This value applies only if the HEO is oriented exactly as required for the escape, which is not the case. The escape manoeuvre from HEO would therefore incur a further large penalty.

In the case of LMO, to provide the system with the required ΔV for the TVI, more propulsion modules should be added to the TEI, increasing the total mass to LEO by 1000 tonnes. In the case of HEO, the required ΔV can be provided by the second stage of the MOI plus the TEI without any mass penalty.

The only other option appears to be to cancel the Mars landing and wait in orbit around Mars for the nominal return window in May 2035.

2.7.12.1.6 Abort after TEI

No option was found for abort after TEI.

2.7.12.2 Options

In addition to the regarded cases, there is a huge variety of other scenarios for which abort and recovery strategies need to be analysed. These relate primarily to the cases where critical manoeuvres are not fully executed due to a failure. Some examples are:

- Incomplete execution of the TMI burn. A distinction must be made between two cases:
    - Failure while still in Earth orbit: This implies that the burn completed so far was not yet sufficient to inject into hyperbolic escape. A decision on whether to proceed with the burn at the next perigee pass, whether to abort and return the crew to Earth or to proceed in any other way must be made based on the gravity of the failure, the chances for recovery and the orbit achieved so far.
    - Failure after achieving escape: This implies that a hyperbolic orbit was achieved. Depending on the time of failure, the resulting heliocentric orbit will range from very close to the Earth orbit to very close to the nominal Mars transfer. Again, the abort/recovery options depend on the spacecraft and orbital conditions achieved.
- Incomplete execution of MOI: Again a case distinction is necessary:
    - Failure after reaching a bound orbit: The burn removed enough orbital energy to insert into a possibly quite wide elliptical orbit around Mars. The reaction must again depend on the case and could consist of immediate stabilization of the orbit

and further apoapsis lowering manoeuvres at the next pericentre pass. Alternatively, it might be an option to force a low-velocity escape. Failure before reaching a bound orbit: The manoeuvre fails while the spacecraft is still in a hyperbolic orbit with respect to Mars. In the case regarded in section 2.7.12.1, only the case of a complete failure was regarded. A partial failure would result in a trajectory closer to the orbit of Mars.

- Incomplete execution of TEI: Here the case distinction made above also applies.

2.7.12.3 Conclusions

An abort cannot be always guaranteed with no further consequences:

- During the MOI and TEI manoeuvres abort is not possible
- During the first part of the transfer to Mars, abort is always possible without mission mass increase
- During the second part of the transfer to Mars, abort is always possible but mission mass increase is needed (either propulsion system or ERC)
- From low Mars orbit, there are two possibilities: return via Venus swing-by but mass increase is needed or waiting for next return window

2.7.13 Aerobraking

Aerobraking is a proven technique to remove energy from an orbit, e.g., when transferring from a highly eccentric orbit to one of low eccentricity, with minimal propellant consumption. Aerobraking involves lowering the pericentre of the initial orbit so that it grazes the upper atmosphere. At every perigee pass, the spacecraft loses some orbital energy to atmospheric friction. This lowers the apocentre radius. After a number of passes, during which the pericentre altitude must be observed and repeatedly corrected so that it does not descend too deeply into the atmosphere, the apocentre will have reached the required altitude. Then, a manoeuvre at the apocentre raises the pericentre and the aerobraking phase is terminated.

The use of aerobraking rather than propulsive manoeuvres for final orbit acquisition can lead theoretically to significant savings in propellant mass. (see Mission architecture)
For this reason, a preliminary estimation was performed in this study.

2.7.13.1 Requirements and design drivers

Aerobraking is a lengthy process but it is relatively safe. The structural and thermal loads imposed on spacecraft components are low compared to other techniques involving atmospheric flight such as aerocapture and entry/landing. However, with the present spacecraft there were design concerns for some of the subsystems, in particular the solar arrays. If left deployed during aerobraking, they would provide the large surface area required to maximize the deceleration and minimise the manoeuvre duration but they would also be particularly vulnerable to the increased structural and thermal loads. Therefore it was necessary to perform a trade-off between the manoeuvre duration and the solar array restrictions. The constraints are summarized in Table 2-29:

Maximum manoeuvre duration	6 months (about 180 days)
Maximum dynamic pressure • Solar arrays facing flow • Solar arrays parallel to flow	 0.2 N/m^2 13 N/m^2
Maximum heat flux (Q) • Solar arrays facing flow • Solar arrays parallel to flow	 10 kW/m^2 Uncertain

Table 2-29: Aerobraking constraints

The dynamic pressure constraint comes from the structural limitations of the solar array structure. For the hinge the maximum allowable bending moment is 185 Nm and the maximum allowable shear load is 25 N and for the support beam the maximum allowable bending moment is 300 Nm (from the solar array design specifications). The force acting on the panels was evaluated from the dynamic pressure as:

$$F = \frac{1}{2} P_{dyn} C_D S$$

where C_D is the drag coefficient of the structure (assumed to be that for a flat plate for which C_D=2.0) and S is the surface area facing the flow. For a solar array area of 95 m^2 and a thickness of 0.1 m (including the thickness of the support beam), the maximum allowable dynamic pressure loads given above were derived.

2.7.13.2 Assumptions and trade-offs

A trade-off was required between the aerobraking manoeuvre duration and the structural and thermal loads on the solar arrays. Three solar array configurations were considered in the trade-off:

1. Solar arrays facing flow.
2. Solar arrays turned parallel to flow (to avoid stowage requirements).
3. Solar arrays stowed.

The results of the various analyses are summarized in Table 3-31 below. The highlighted areas show values that violated the constraints outlined above.

	Option	Q_{max} [kW/m^2]	$P_{dyn,max}$ [N/m^2]	Duration	Operational Issues
1	Solar arrays deployed facing flow	45.0	11.0	3 months	
2	Solar arrays deployed facing flow	23.0	5.5	6 months	
3	Solar arrays deployed facing flow	low	0.2	about 8 yrs	
4	Solar arrays deployed parallel to flow	60.0	13.0	About 16 months	Turning of arrays
5	Solar arrays stowed	630.0	145.0	3 months	Retraction and deployment of arrays
6	Solar arrays stowed	315.0	72.0	6 months	Retraction and deployment of arrays

Table 2-30: Results of aerobraking analyses

2.7.13.3 Baseline design

From the analyses presented above, it is evident that it is not possible to meet all of the aerobraking constraints with the current vehicle. If the duration is constrained the loads are unacceptably high. Conversely, if the loads are constrained, the duration is so long that the additional ΔV that would be required to reduce it to meet the constraint would make the aerobraking mass savings negligible. Therefore, an aerobraking manoeuvre was not chosen for the baseline design of this mission case.

2.7.13.4 Manoeuvre budget

A typical manoeuvre budget for an aerobraking phase that reduces the apocentre altitude from initially 96 000 km to 500 km is 115 m/s, 15 m/s for pericentre control (initial lowering and subsequent adjustment manoeuvres) and 100 m/s for the perigee raise from the final altitude

2.7.13.5 Options

In addition to the more conventional aerobraking manoeuvre considered, “deep aerobraking” is a possible option. This would involve deploying an inflatable heat shield (or using an ablative heat shield), storing the solar arrays, and going deep in the atmosphere to shed the orbital energy in a limited number, of passes. This option has not been analysed in this study.

2.7.14 Artificial gravity

One of the biggest problems that must be overcome is the harmful effects of weightlessness on the human body. These effects include loss of bone and muscle mass, loss of red blood cells, fluid shifting from the lower to the upper body, cardiovascular and neurosensory deconditioning, and changes in the immune system. The physiological systems start to change immediately upon launch into microgravity and the time courses of change is different for each of them. For example, the fluid shift and cardiovascular system start immediately within hours while the muscle and bone need some time to adjust to microgravity; deconditioning starts after days (muscle) or weeks (bone). Body fluid and cardiovascular system adapt to new environments in less than 2 weeks. However, cardiac arrhythmia might be a problem in-flight for some individuals during elevated workloads. Muscle loss (muscle volume and power) is maximum within the first 4 weeks, but afterwards the loss rate is reduced (strongly dependent on in-flight countermeasures). Bone loss, however, continues progressively (1% per month) throughout the mission in free space. Figure 2-41 summarizes the reactions to microgravity of each of the physiological systems:

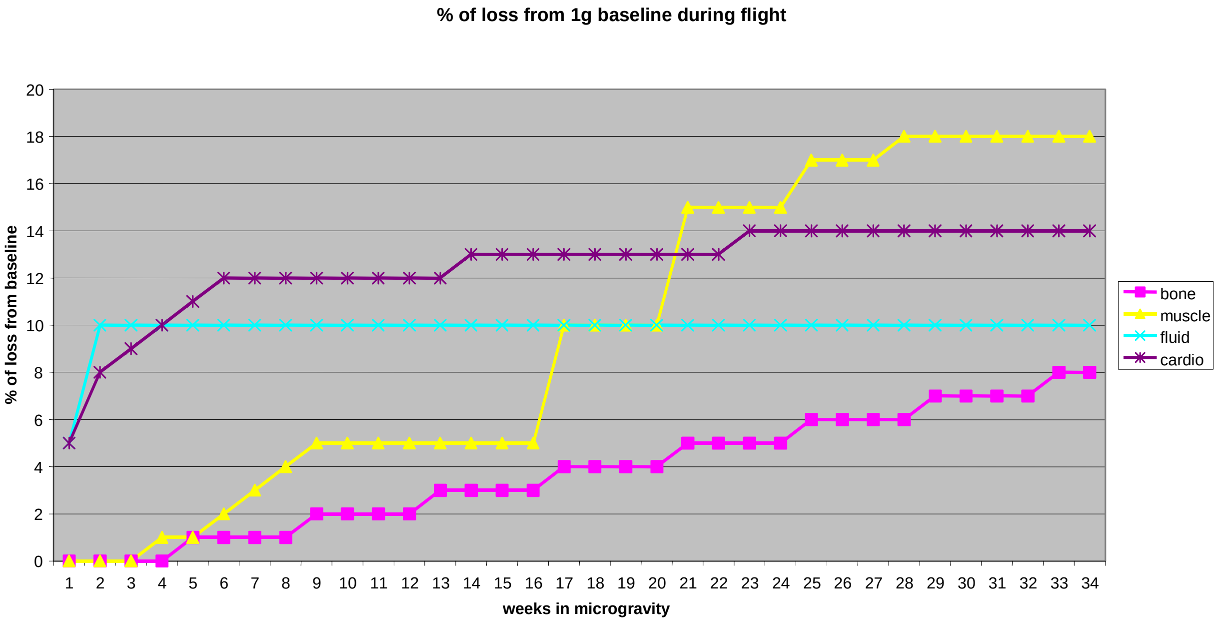


Figure 2-41: Effect of microgravity on physiological systems

Currently, some countermeasures for weightlessness such as physical exercise, lower body negative pressure and drugs are used in human spaceflight. However, these countermeasures prevail for the permanent ISS crew but have not proven to be sufficient for longer duration missions on-board MIR. Moreover, these countermeasures currently adopted focus only on stimulating a particular physiological system.
Artificial gravity, however, represents a different approach to the problem of microgravity effects because it simulates our natural 1g environment.

2.7.14.1 Trade-offs

There are two options for how to approach the implementation of artificial gravity as a way to fight against the negative effects of long exposures to weightlessness. One is by providing artificial gravity through continuous rotation of the entire spacecraft with a large radius of rotation and low angular velocities.

The second option is intermittent exposure to artificial gravity enough to overcome the damaging effects of microgravity, using on-board centrifuges with a small radius of rotation and high angular velocities to simulate gravity for certain amounts of time.

2.7.14.1.1 Continuous artificial gravity

This option requires that the comfort level for the astronauts is respected (see Figure 2-42 from NASA – Habitability data handbook, Volume 1, MSC-03909, 1971). It implies that the minimum rotation radius must be 17 m, achieving 0.3 g at the maximum spin rate of 4 rpm.

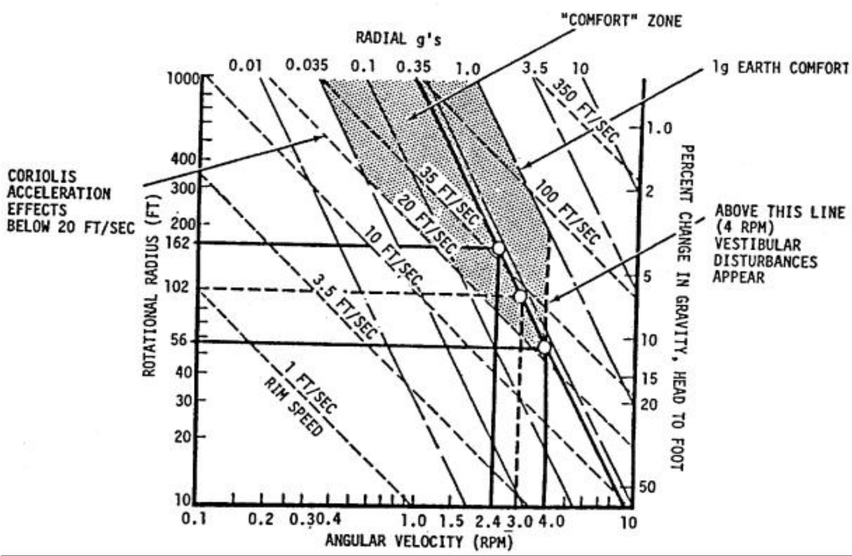


Figure 2-42: Radial acceleration depending on rotational radius and angular velocity

Two options are considered:

Option 1: Rotation of two end masses at the extremities of a long structure. Additional structure is required, as well a propulsion system to spin the vehicle, power, avionics, and tether or truss deployment mechanisms. The increase in the dry mass of the spacecraft is estimated to be 20%.

Option 2: Take advantage of the existing design of the entire vehicle (THM, Propulsion Stages and MEV) and its configuration and rotate it. With the current design, however, the comfort zone cannot be achieved.

In both cases the complexity of the configuration is increased and some operational constraints are introduced, because no EVA is possible while the spacecraft is spinning. Therefore this option was rejected.

2.7.14.1.2 Shor- arm centrifuge

The short-arm centrifuge option results more realistic, simple and effective. From a practical perspective, it is very likely that humans do not need gravity (or a fraction of it) 24 hours a day to remain healthy. If intermittent gravity is proven to be sufficient, an on-board centrifuge presents a realistic near-term opportunity for providing artificial gravity.

For the purpose of this study, a 4.5-metre diameter, two-man balanced centrifuge was considered. It incorporates an exercise cycle or treadmill and has a spin rate of 21 rpm for 1g at the feet and 13 rpm for 0.38g. This centrifuge would have a mass of about 450 kg and would be used in combination with 2 flywheel crew exercise devices to minimise muscle deconditionning.

2.7.15 Sensitivity analysis

A sensitivity analysis on several of the mission parameters was performed during the trade-offs. The exercise has been repeated over the final design for the study case, confirming the same trends.

2.7.15.1 Influence of the mass of the THM

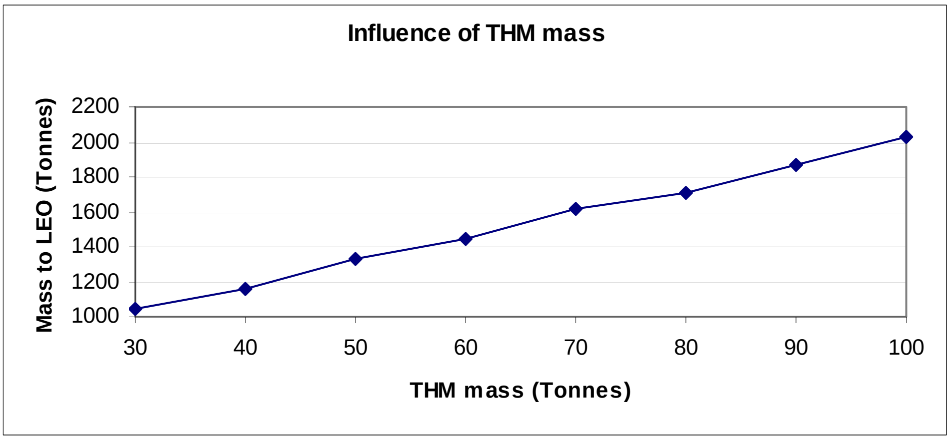


Figure 2-43: Influence of the THM mass

The THM is one of the mission elements that is carried along all the mission phases, so, propellant has to be provided to give it the required ΔVs for all the main propulsive manoeuvres. The dry mass has a huge influence on the overall mass to LEO. Its mass has to be therefore kept as low as possible.

2.7.15.2 Influence of the closure level in the ECLSS system

Table 2-31 shows an overview of the sensitivity analysis performed concerning ECLSS. The maximum level of closure signifies 95% O_2, 95% water (all but black), 20% black water and organic waste and 0% inorganic waste.

	0% level of closure	Maximum level of closure
ECLSS System hardware mass (kg)	7620	9670
Storage hardware mass (kg)	39888	8434
Mass of consumables (kg)	53663	8826
Mass of packaging (kg)	1146	1393
Total mass (kg)	102317	28323
ECLSS system power (W)	3757	4033
Storage system power (W)	3470	3054
Total power (W)	7227	7087

Table 2-31: Influence of the level of closure in the ECLSS for a 1000-day mission

In the case of zero level of closure all the consumables have to be brought from the beginning. The study considered only dry food.
In terms of power requirements, only a small increase is observed in the system when a high level of recycling is used, although the power requirements for storage in this case are smaller. Dry food is considered somewhat inert and requires less power for storage than fresh food. In-situ food either needs to be processed or needs higher storage power (refrigerator, freezer). Therefore, the reduction is only 400W in this analysis as no extensive processing and fresh food storage were assumed.
Open-loop life support systems for such a long duration mission with no resupply capabilities represents a prohibitive mass and volume for the system. The overall mass required is four times bigger. In terms of power requirements there is no great difference between the closed and the open loop systems. This mission therefore requires a high level of closure life support system..

2.7.15.3 Influence of the mass of the ERC

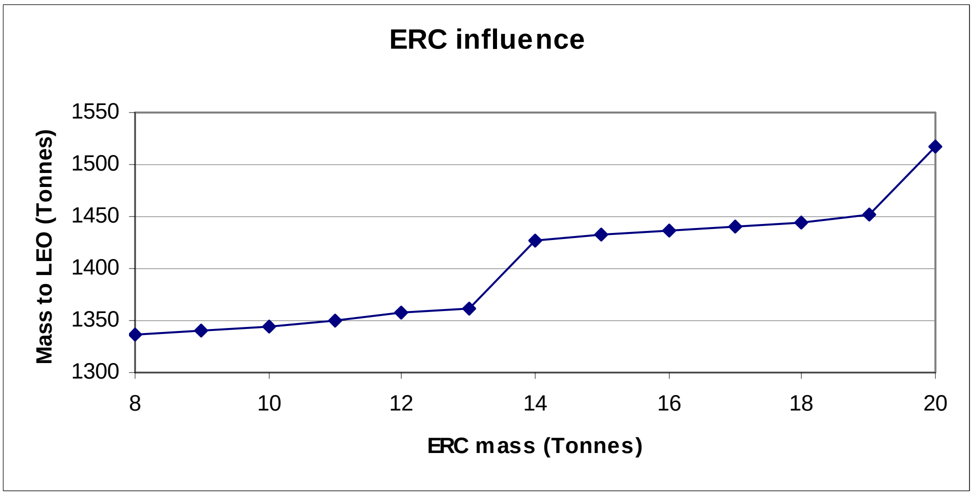


Figure 2-44: Influence of the ERC mass

The ERC mass has also a large influence on the initial mass for the same reason as the THM, although the mass of the ERC itself is lower and such is its effect compared with the one of the THM.
The sudden increases shown in Figure 2-44 are due to the approach followed for the TMI propulsion module, which is composed of 12, 13 and 14 stacks respectively.

2.7.15.4 Influence of the mass of the MEV

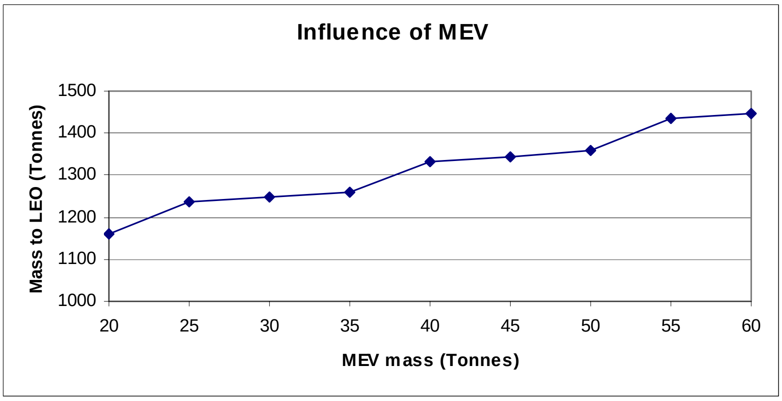


Figure 2-45: Influence of the MEV mass

As shown in Figure 2-45, the influence of the MEV mass on overall mass is lower than that of the THM, as it is discarded during Martian orbit.

2.8 In orbit assembly

2.8.1 Assumptions and trade-offs

Note that when considering the mission architecture:

- A chemical mission results in a high mass into LEO and a configuration with several elements
- The parameter to assess the feasibility of a mission architecture is the *time-to-assemble* such elements in LEO (or the selected orbit)
- A long assembly time would result in:
    - Aging of elements even before the mission starts (potentially, unacceptable aging).
    - Unacceptable boil-off losses for the case of cryogenic-propulsion.
    - Delay of the following mission within a chosen exploration strategy.

Several orbits are candidates for the assembly operations:

- LEO
- Nuclear safe orbit
- MEO
- GEO
- HEO

Note that when choosing the orbit for the assembly operations:

- All the elements have to be inserted into the assembly orbit and perform a rendezvous & berthing or docking with the already existing composite.
- Human interaction may be required to complete any given addition to the composite.
- Nuclear safe orbits will only be considered if nuclear devices are present in the composite, currently none are baselined.
- Elliptical orbits are discarded due to the complexity of the RVD manoeuvre.
- GEO and HEO are discarded due to the loss of performance of the launchers and the (potentially) more hostile environment.

- LEO is the preferred assembly orbit. An altitude of 400 km (similar to that of the ISS) has been selected.

Figure 2-46 shows that the assembly activities potentially establish more than 50% of the total 'life' of (some of) the hardware for a given mission timeline. Thus the assembly sequence and the time spent assembling the composite becomes an important component of the mission timeline.

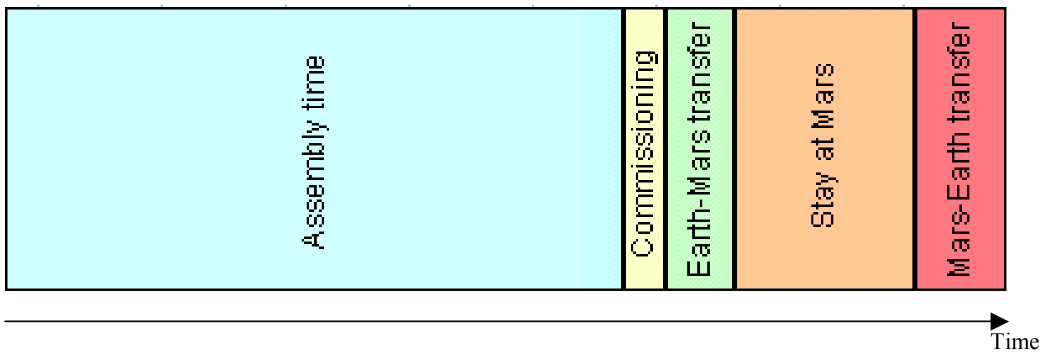


Figure 2-46: Mission timeline

- To estimate the composite in-orbit assembly time, the following parameters are critical:
- Launcher type and frequency of launch
- Element availability rate (on-ground production assumption)
- EVA or Automated assembly
- Cargo or 'Manned launch' or Resupply
- Equipment Check-out Times
- Launch failure assumptions

The following assumptions have been made for the In-orbit assembly analysis:

Assumptions	Qualifying statement
Cargo Launcher = Energia	
Launch Frequency o Min: every 3 months o Max: every month	Launcher production infrastructure in place to be able to launch 1 per month Operational support o Up to eight teams working in parallel at ESOC- 1 for orbital vehicle, 1 each for launcher/mission in prep
LEO Workbench/platforms present to support LEO operations & power requirements (LEO crew Habitat.)	LEO workbench provides power, docking & resupply, AOCS and boost, autonomous Robotic Arm to support the composite and facilitate Capture & Berthing operations
Equipment Check-out times o Habitation modules 3 months o Propulsion modules 1 month	
Minimise number of manned flights o Crew support using short term shuttle visits	
(Minimise the use of 'flight systems')	
Minimise Cryogenic Fuel Boil-off	
Minimal reconfiguration of vehicle in-orbit after initial capture & berthing or docking operations	

Table 2-32: Assumptions for assembly

The following are identified as open points whose influence is not considered within this analysis.

- LEO Power Requirements
- The service platforms are required to fully support the composite at all stages during the assembly operations.
- TV Thermal Control in LEO
- Specifically more important for the (Cryogenic-) propulsion stages.
- TV susceptibility to ATOX-
- Due to the LEO. Of more specific concern for the Solar Arrays. The Flight Arrays shall be launched as late as possible.
- Crewed or Un-crewed LEO Operations
- The baseline selected is for autonomous LEO operations in as far as possible.
- Docking Constraints
- Launch Failure Modes
- On-orbit Cryogenic Stage Refuelling
- Disposal/Handling of upper launch vehicle stages for each individual module.

An Excel-based model has been developed to estimate the in-orbit assembly time based on a number of parameters.

The following parameters are set per element:

- Connecting two elements in orbit → 2 phases:
  1) Docking
     I. Docking only.
     II. Capture & Berthing → Need robotic arm from service platform
  2) Connection of cables (power, life support, etc)
     I. By Berthing/Docking
     II. External cabling → Need EVA/Robotics
     III. Internal cabling → Need Crew/Robotics
- Assembly/Docking strategy
  – If EVA is required: Need either Shuttle launch each time or a manned assembly station
- Boil-off Rate
  – The LH_2 Boil-off Rate is assumed to be 70 kg per month on average
- Production Assumptions
  – Availability rate of the elements
- Operation Centre
  – What is the minimum time between one element is docked to the composite until the next element can be launched?
- Other factors
  – LEOP and commissioning duration (+ GS)

Within the model, the following is applicable:

- Production & Testing of the elements is taken into account only as an availability rate (relative readiness).

- No failures assumed.
- As a result of boil-off a refueling flight could be made but is not considered a baseline. Boil-off is considerable, but acceptable in terms of loss of performance if kept below 70 kg/month of liquid hydrogen
- No reboosting of composite is included.

The following parameters in Table 2-33 have been used in the model given the assumptions above:

2) Launch Vehicles, Operations & Production Assumptions

Launch Vehicles

Assumption: Launch window independent between vehicles

Number	Name	Nr launches /year	Launch rate	Performance to LEO (kg)	Fairing dim1 (m)	Fairing dim2 (m)
1	ENERGIA	12	31	80,000	35	6
2	A5	8	46	20,000	10	4.5
3	Proton	8	46	20,000		
4	Soyuz	12	31	8,000		
5	Shuttle	4	92			

Operations

Composite Operations time (non-manned modules)	30	days	After Berthing
Composite Operations time (manned modules)	90	days	After dock & assembly
LEOP	2	days	
Commissioning Duration	8	days	Should be about 30 days for launch of the first element in order to check the Ground Segment
On-orbit assembly	3	days	

Production Assumption

Development/Integration	8	years/element	
Element Availability rate	30	days	I.e. time btw one element finished with respect to the previous one

Margin & Commissioning

Assembly margin for each element	0	days
Overall assembly margin	180	days
Commissioning duration	60	days
Commissioning duration/Crew Training	30	days

Table 2-33: Assembly parameters

2.8.2 Sensitivity analysis

Apart from the following baseline analysis, the effect of certain parameters on the resulting assembly time and total boil-off have been examined.

1. Effect of Energia launch on assembly time

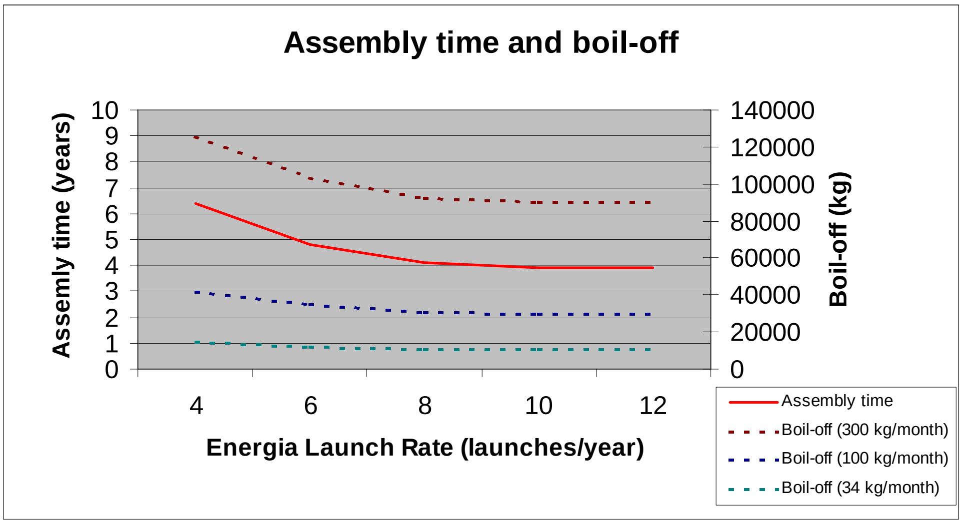


Figure 2-47: Assembly time and boil-off as a function of launch rate

Parameters used:

- No Shuttle
- Element Availability rate: 30 days
- Margin: 180 days, Commissioning: 90 days
- Check-out time: 30 days

I. Effect of Element Check-out time and Energia launch rate

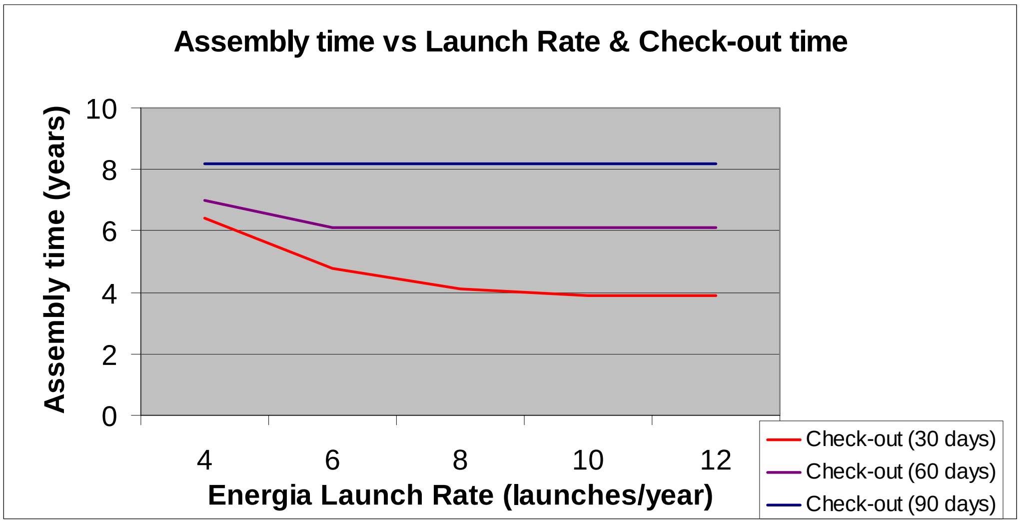


Figure 2-48: Assembly time as a function of launch rate and check-out

Parameters used:

- No Shuttle
- Element Availability rate: 30 days
- Margin: 180 days, Commissioning: 90 days
- Check-out time: 30, 60, 90 days

2.8.3 Baseline design

The baseline assembly sequence is given below, with each step supplemented by qualifying statements.

In the sequence, dedicated spacecraft called "service platforms" are identified for vehicle power, attitude control and robotic assembly. These auxiliary vehicles are launched at different times of the sequence and are jettisoned before departure.

2.8.3.1 Launch 1- Habitation module

- AOCS & LEO Service Platform (SP1)
  - LEO AOCS
  - LEO Power
  - Rendezvous Booster- TBD
  - EVA Airlock capability
  - LEO Crew Habitation

- ATV LEO reboost capability
- Two on-axis ports available
  - One for further assembly operations
  - One for on-orbit resupply on service platform (Soyuz or ATV typical)
  - Alternative port required if Shuttle used for crew transfer

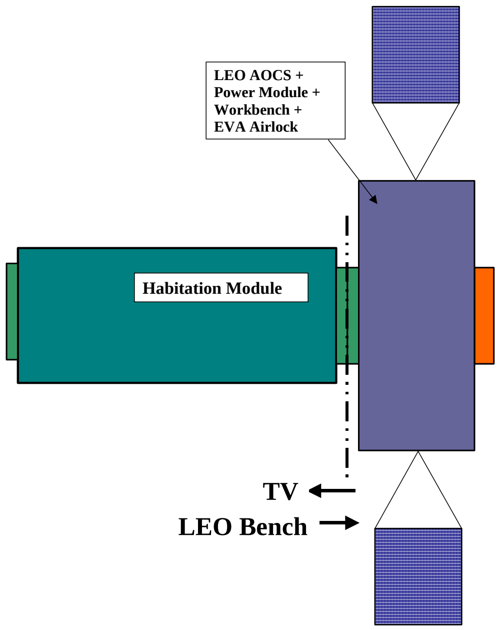


Figure 2-49: THM central cylinder and SP1

2.8.3.2 Launch 2- Back node

- Addition of Back Node
- Addition of mission Flight EVA Airlock
- Addition of Service Platform 2 (SP2)
  - LEO AOCS
  - LEO Power
  - Berthing Arm
- EVA Operations required to complete assembly

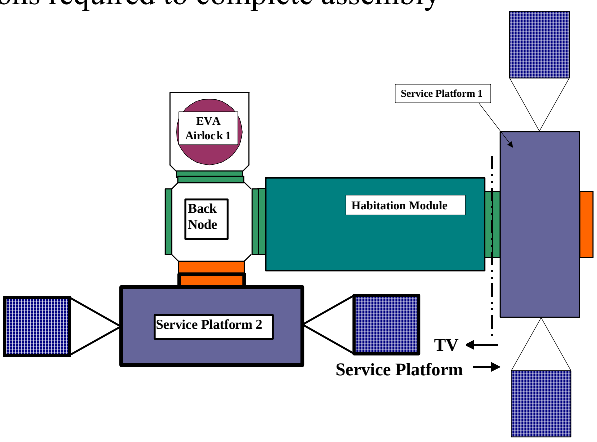


Figure 2-50: Launch of Back Node and SP2

2.8.3.3 Launch 3- Front node

- Addition of Front Node
  - With Cupola
- Disposal or Relocation of SP1
- Addition of SP3 (if no relocation of SP1)
  - LEO AOCS
  - LEO Power
  - Rendezvous Booster- TBD
  - EVA Airlock capability
  - LEO Crew Habitation
  - ATV LEO re-boost capability
- EVA Operations required to complete assembly

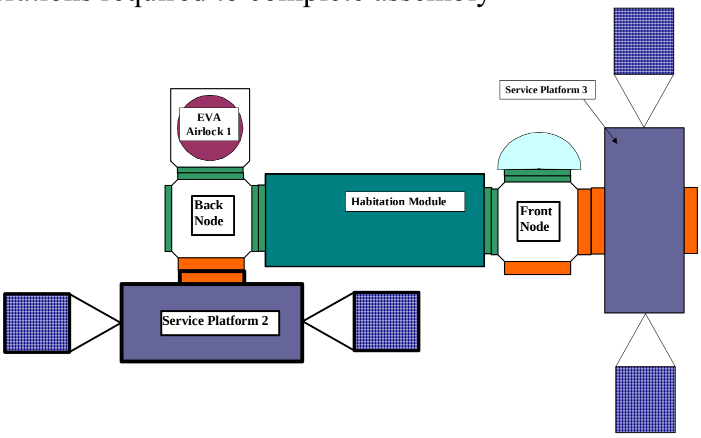


Figure 2-51: Launch of Front Node

2.8.3.4 Launches 4-8- TEI & MOI assembly

- Addition of TEI & MOI Propulsion stages
  - Assume TEI stage is the central structure or TEI within central back-bone structure
  - Each propulsion stack is captured & berthed by Robotic arm from service platform
  - Each propulsion stack will require a Launcher upper stage to provide rendezvous capability

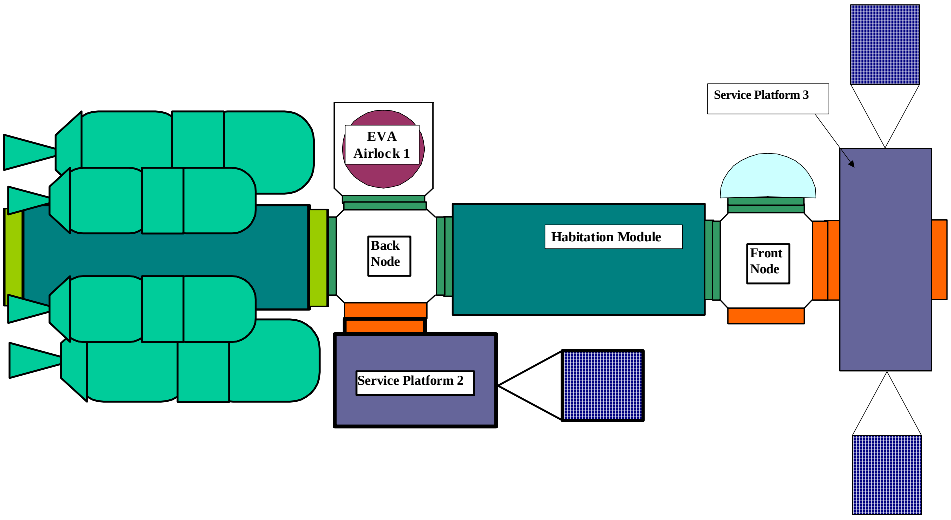


Figure 2-52: Launch of TEI and MOI

2.8.3.5 Launches 9-17- TMI stages 2 & 3 assembly

- Addition of TMI stage 2 & 3 Propulsion Stages
  - Launch 9- Central structure + SP4
  - Each propulsion stack is captured & berthed by Robotic arm from service platform
  - Each propulsion stack will require a Launcher upper stage to provide rendezvous capability.

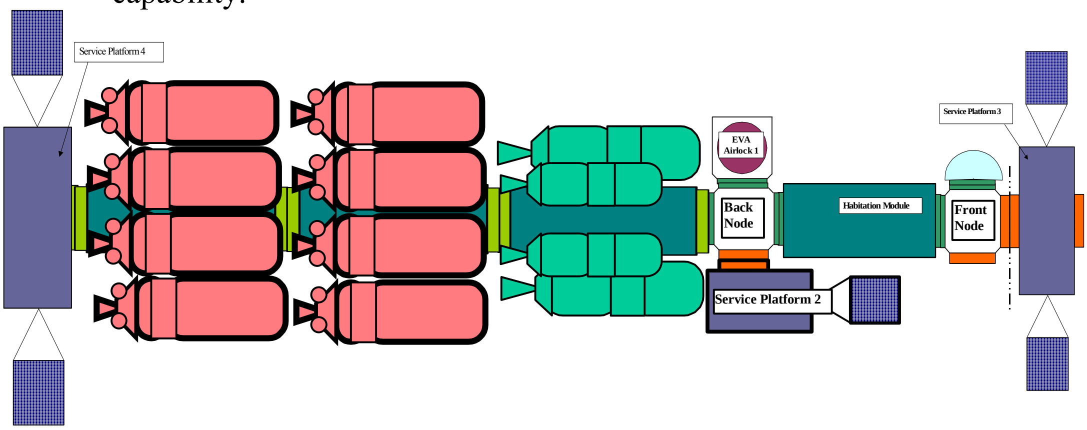


Figure 2-53: Launch of 2nd and 3rd stage of TMI

2.8.3.6 Launches 18-22- TMI stage 1 assembly

- Addition of TMI stage 1 Propulsion Stages –
  - Launch 18- Central structure + SP5
    - Disposal of SP4 prior to berthing
  - Launch 19-22- Propulsion stacks
  - Each propulsion stack is captured & berthed by Robotic arm from service platform

– Each propulsion stack will require a Launcher upper stage to provide rendezvous capability.

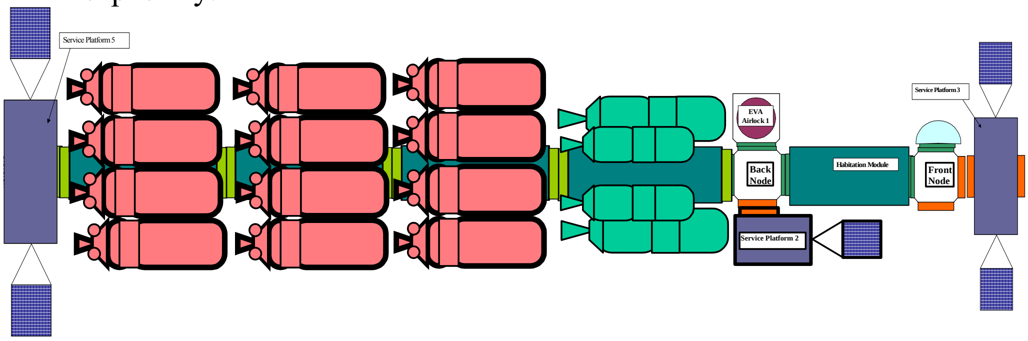


Figure 2-54: Launch of TMI stage 1

2.8.3.7 Launch 23- ERC and 'appendage' assembly

- Addition of ERC and Appendages
  - ERC
  - Four Solar Arrays (EVA-assisted Assembly)
  - Two Antennas (EVA-assisted Assembly)
- Disposal of SP2 prior to ERC docking

  - Launch Mass (Estimated)
  - ERC will require a launcher upper stage to provide rendezvous and docking capability.
  - Appendages will require disposable support structure (possible component of disposable upper stage) and will require EVA-assisted assembly by robotic arm from service platform.

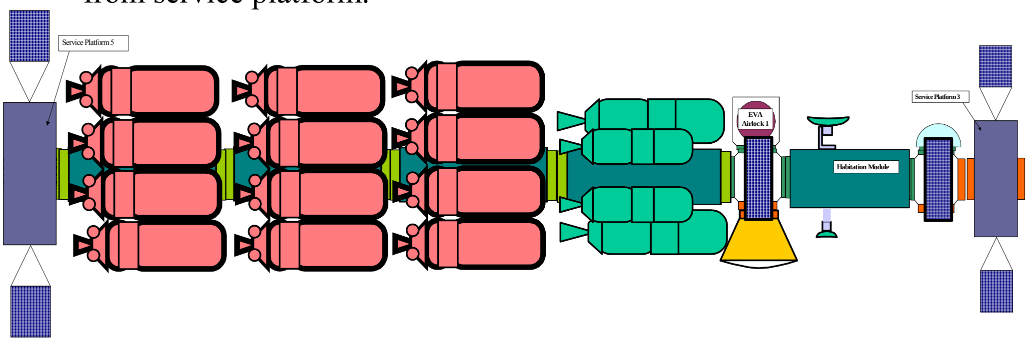


Figure 2-55: Launch of ERC and appendage

2.8.3.8 Launches 24-25- Composite completion

- Resupply, Crew Transfer, Re-fuel & Addition of MEV prior to service platform disposal
  - Launch 24 - Consumables (Re) Supply
  - Service platform disposal to free port for MEV
  - Post service platform disposal
    - Launch 25 – MEV

- EVA-supported completion of MEV structural I/F
- MEV Check-out, Commissioning Phase & Crew orientation commence at MEV delivery

Once the MEV is added to the composite, the 90-day commissioning phase begins. If an in-orbit crew is required, dedicated commissioning crew shall be present for the first 60 days.

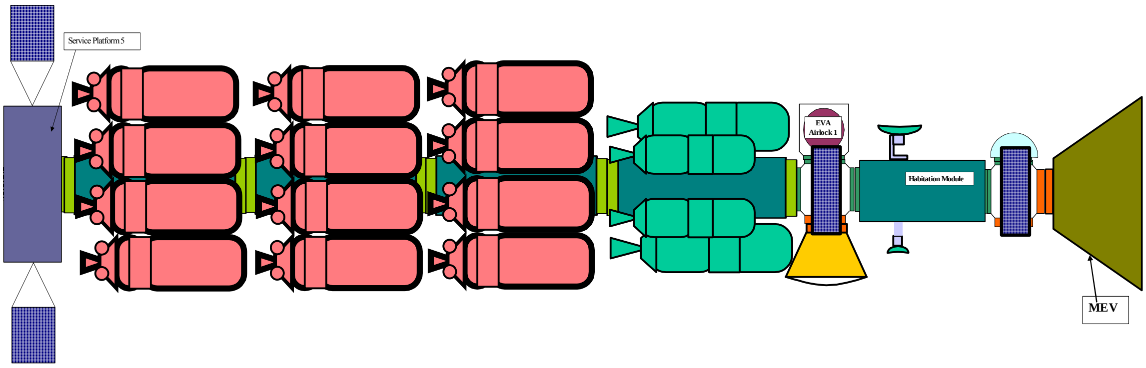


Figure 2-56: Launch of MEV

2.8.3.9 Launches 26-27- Crew

- Launch 26 & 27 - Mars Crew delivery 30 days before departure
  - assumption, two launches of three crewmembers by Soyuz
  - alternative, single launch by shuttle

2.8.3.10 Launch 28- Cryogenic stage refueling (if required)

- Refueling of Cryogenic stages performed during commissioning phase

Note that the above assembly approach satisfies the constraint to minimise the Cryogenic fuel boil-off. However the assembly of the habitation modules at the beginning of the composite assembly sequence *maximizes* the life requirement of the equipment.

Table 2-34 show a summary of the sequence defined above:

Element Name	Mass	Cryo Stage?	Launch Sequence	Checkout time of composite	Launcher	Docking		Connections	
						Type	Arm required?	Type	Robotics or EVA required?
Habitation module + Service Platform 1	50,000	No	1	90	ENERGIA	N/A	No	N/A	No
Node 1 + Airlock + Service Platform 2	20,000	No	2	90	A5	Capture & Berth	Yes	External cable(s)	Yes
Node 2 + Cupola + Service Platform 3	20,000	No	3	90	Proton	Capture & Berth	Yes	External cable(s)	Yes
PS TEI	80,000	No	4	30	ENERGIA	Capture & Berth	Yes	Berthing/Docking	No
PS MOI 1	80,000	No	5	30	ENERGIA	Capture & Berth	Yes	Berthing/Docking	No
PS MOI 2	80,000	No	6	30	ENERGIA	Capture & Berth	Yes	Berthing/Docking	No
PS MOI 3	50,000	No	7	30	ENERGIA	Capture & Berth	Yes	Berthing/Docking	No
PS MOI 4	50,000	No	8	30	ENERGIA	Capture & Berth	Yes	Berthing/Docking	No
PM Central Structure 1 and 2 + Service Platform 4	20,000	No	9	30	ENERGIA	Capture & Berth	Yes	Berthing/Docking	No
PS TMI 1	80,000	Yes	10	30	ENERGIA	Capture & Berth	Yes	Berthing/Docking	No
PS TMI 2	80,000	Yes	11	30	ENERGIA	Capture & Berth	Yes	Berthing/Docking	No
PS TMI 3	80,000	Yes	12	30	ENERGIA	Capture & Berth	Yes	Berthing/Docking	No
PS TMI 4	80,000	Yes	13	30	ENERGIA	Capture & Berth	Yes	Berthing/Docking	No
PS TMI 5	80,000	Yes	14	30	ENERGIA	Capture & Berth	Yes	Berthing/Docking	No
PS TMI 6	80,000	Yes	15	30	ENERGIA	Capture & Berth	Yes	Berthing/Docking	No
PS TMI 7	80,000	Yes	16	30	ENERGIA	Capture & Berth	Yes	Berthing/Docking	No
PS TMI 8	80,000	Yes	17	30	ENERGIA	Capture & Berth	Yes	Berthing/Docking	No
PM Central Structure 3 + Service Platform 5	20,000	No	18	30	ENERGIA	Capture & Berth	Yes	Berthing/Docking	No
PS TMI 9	80,000	Yes	19	30	ENERGIA	Capture & Berth	Yes	Berthing/Docking	No
PS TMI 10	80,000	Yes	20	30	ENERGIA	Capture & Berth	Yes	Berthing/Docking	No
PS TMI 11	80,000	Yes	21	30	ENERGIA	Capture & Berth	Yes	Berthing/Docking	No
PS TMI 12	80,000	Yes	22	30	ENERGIA	Capture & Berth	Yes	Berthing/Docking	No
ERC + SA + Antennas	15,000	No	23	90	A5	Capture & Berth	Yes	External cable(s)	Yes
TV Supply	20,000	No	24	30	A5	Dock only	No	N/A	No
MEV	46,500	No	25	0	ENERGIA	Dock only	No	Berthing/Docking	No
	1,511,500		25						

Table 2-34: Launch sequence

Main Outputs				
Description	Days since start	Years since start	Date	Remarks
Phase A start of first element	-2920	-8.0	24/12/2020	Assuming development/integration of 8 years/element
First launch	0	0.0	22/12/2028	
Last element launched	1293	3.5	07/07/2032	
Last EVA/Shuttle launch	235	0.6	14/08/2029	
End of assembly (no margin)	1306	3.6	20/07/2032	
End of assembly	1486	4.1	16/01/2033	Includes Overall margin of 180 days
Crew Launch	1546	4.2	17/03/2033	LV: 1 Shuttle
Start of commissioning	1486	4.1	16/01/2033	
Departure	1576	4.3	16/04/2033	Commissioning duration of 60 days

Description	Value	Unit	Remark
Total mass launched	1,511,500	kg	Includes Service Platforms
Prop mass loss due to boil-off	20,358	kg	Mass lost until launch window and is launched as last PS
Nr of launches	25		Includes launch of all elements and relaunch due to boil-o
Nr of launches for EVA/robotics	2		Shuttle launches for EVA/arm
Nr of Crew launches	0		1 Shuttle (Could also be 2 Souyz)
Total number of launches	27		

Table 2-35: Assembly simulation results

For the baseline assembly scenario, the effect on the in-orbit assembly time and cryogenic boil-off due to a longer availability rate has been assessed and is given in Figure 2-57.

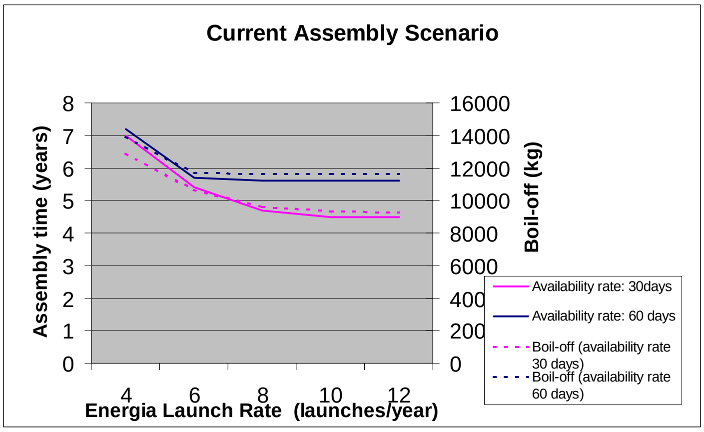


Figure 2-57: Assembly scenario

Figure 2-57 shows that:

- With the present mission configuration the minimum assembly time found would be *4.6 years*
- This is strongly dependent on the launcher selection and the associated launch rate. If the launch rate of Energia is 4 times per year the assembly time is longer than 6 years
- The need for manned operations in the assembly sequence and their associated launches increases significantly the assembly time, reliant on the shuttle availability
- A launch for the refuelling of the cryogenic stage may be required

2.8.4 Options

The above scenario minimises the cryogenic boil-off but has the disadvantage of maximizing the flight life of the habitation volumes because these modules are launched at the beginning of the sequence.

The assembly sequence can be modified so that the assembly of the habitation modules occurs last in the sequence. The resulting sequence and the effect on the result are shown in Table 2-36:

Element Name	Mass	Cryo Stage?	Launch Sequence	Checkout time of composite	Launcher	Docking		Connections	
						Type	Arm required?	Type	Robotics or EVA required?
PS TEI + Service Platform 1	80,000	No	1	30	ENERGIA	N/A	No	N/A	No
PS MOI 1	80,000	No	2	30	ENERGIA	Capture & Berthing	Yes	Berthing/Docking	No
PS MOI 2	80,000	No	3	30	ENERGIA	Capture & Berthing	Yes	Berthing/Docking	No
PS MOI 3	50,000	No	4	30	ENERGIA	Capture & Berthing	Yes	Berthing/Docking	No
PS MOI 4	50,000	No	5	30	ENERGIA	Capture & Berthing	Yes	Berthing/Docking	No
PM Central Structure 1 and 2	20,000	No	6	30	Proton	Capture & Berthing	Yes	Berthing/Docking	No
PM Central Structure 3 + Service Platform 2	20,000	No	7	30	A5	Capture & Berthing	Yes	Berthing/Docking	No
PS TMI 1	80,000	Yes	8	30	ENERGIA	Capture & Berthing	Yes	Berthing/Docking	No
PS TMI 2	80,000	Yes	9	30	ENERGIA	Capture & Berthing	Yes	Berthing/Docking	No
PS TMI 3	80,000	Yes	10	30	ENERGIA	Capture & Berthing	Yes	Berthing/Docking	No
PS TMI 4	80,000	Yes	11	30	ENERGIA	Capture & Berthing	Yes	Berthing/Docking	No
PS TMI 5	80,000	Yes	12	30	ENERGIA	Capture & Berthing	Yes	Berthing/Docking	No
PS TMI 6	80,000	Yes	13	30	ENERGIA	Capture & Berthing	Yes	Berthing/Docking	No
PS TMI 7	80,000	Yes	14	30	ENERGIA	Capture & Berthing	Yes	Berthing/Docking	No
PS TMI 8	80,000	Yes	15	30	ENERGIA	Capture & Berthing	Yes	Berthing/Docking	No
PS TMI 9	80,000	Yes	16	30	ENERGIA	Capture & Berthing	Yes	Berthing/Docking	No
PS TMI 10	80,000	Yes	17	30	ENERGIA	Capture & Berthing	Yes	Berthing/Docking	No
PS TMI 11	80,000	Yes	18	30	ENERGIA	Capture & Berthing	Yes	Berthing/Docking	No
PS TMI 12	80,000	Yes	19	30	ENERGIA	Capture & Berthing	Yes	Berthing/Docking	No
Node 1 + Airlock + Service Platform 3	20,000	No	20	90	Proton	Capture & Berthing	Yes	External cable(s)	Yes
Habitation module	50,000	No	21	90	ENERGIA	Capture & Berthing	Yes	External cable(s)	Yes
Node 2 + Cupola + Servive Platform 4	20,000	No	22	90	Proton	Capture & Berthing	Yes	External cable(s)	Yes
ERC + SA + Antennas	15,000	No	23	90	A5	Capture & Berthing	Yes	External cable(s)	Yes
TV Supply	20,000	No	24	30	A5	Dock only	No	N/A	No
MEV	46,500	No	25	0	ENERGIA	Dock only	No	External cable(s)	Yes
	1,511,500		25						

Table 2-36: Modified assembly sequence

The resulting assembly time becomes:

Main Outputs				
Description	Days since start	Years since start	Date	Remarks
Phase A start of first element	-2920	-8.0	24/12/2020	Assuming development/integration of 8 years/element
First launch	0	0.0	22/12/2028	
Last element launched	1293	3.5	07/07/2032	
Last EVA/Shuttle launch	1155	3.2	20/02/2032	
End of assembly (no margin)	1306	3.6	20/07/2032	
End of assembly	**1486**	**4.1**	**16/01/2033**	Includes Overall margin of 180 days
Crew Launch	1546	4.2	17/03/2033	LV: 1 Shuttle
Start of commissioning	1486	4.1	16/01/2033	
Departure	1576	4.3	16/04/2033	Commissioning duration of 60 days

Description	Value	Unit	Remark
Total mass launched	1,511,500	kg	Includes service Platforms
Prop mass loss due to boil-off	28,210	kg	Mass lost until launch window and is launched as last PS
Nr of launches	25		Includes launch of all elements and relaunch due to boil-off
Nr of launches for EVA/robotics	2		Shuttle launches for EVA/arm
Nr of Crew launches	0		1 Shuttle (Could also be 2 Souyz)
Total number of launches	27		

Table 2-37: Modified assembly results

Table 2-37 shows that the overall assembly time does not alter. The change in the assembly sequence only affects the amount of cryogenic boil-off. The effect of delaying the assembly of all habitable modules until the end of the sequence, increases the boil-off by about 7.85 tonnes.

When considering the in-orbit assembly sequence, the determining factors will therefore be the ‘operational life’ of the (habitable volume) equipment for the mission as a trade-off against the amount of fuel boil-off considered acceptable (mainly in the cyrogenic tanks launched first in the sequence) and also whether a refuelling launch shall be considered.

2.9 Safety/risk assessment

2.9.1 Mission-specific characteristics

The driving characteristics of this mission are:

- It falls within the category of Human Space Flight, including EVA activities.
- It is an inter-planetary mission with sample return, therefore the Interplanetary Protection Rules and the UN treaties are applicable.

2.9.2 Definition of “safety and mission success”

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

- Mission success: to bring a crew of 6 members to Mars and return them safely to Earth.
- Safety goal: to identify all possible safety hazards, to eliminate/control them to an acceptable level during all the phases of the mission.
- Probabilistic goals (overall safety & mission success risks): Human Spaceflight statistics show a 5% risk of losing the crew. Any next-generation system for transporting astronauts to Mars will be probably designed to a risk requirement much lower than that, e.g 0.5%.

2.9.3 Safety requirements:

- Double & Single Failure/Fault/Operator error tolerance to catastrophic & critical events; safety margins
- Fail safe: This is the capacity of the system to remain in a safe condition when a failure occurs or to skip directly into another safe condition

2.9.4 Mission factors/issues:

Throughout the mission design the following factors are important:

- Mission abort/ rescue capabilities. (Acceptable risks can be achieved if abort options are designed into the mission for all phases except for those for which it is impossible
- 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 system)
- Capability to monitor/ detect and assess effects of slow events such as: metal fatigue, cracks; dust, corrosion and rust

- Cabin atmosphere toxicity, contaminant and hazardous substance concentrations are potential toxic threats in the recycling of breathable habitat atmospheres, water recycling systems, and solid waste handling and recycling systems; bio-hazards; deterioration of electrical insulation of wires; thermal insulation; seal deterioration; food spoilage, potable water contamination…etc
- Protection against space radiation hazards; several effects of changes in gravity forces and physiological/psychological risks of extended confinement and hazardous operations. EVA safety. Design of a safe haven. Pathologies to be considered and relevant medical care are a main concern too.

2.9.5 Technical risk assessment scope

Within the risk assessment process, available risk information is produced and structured, facilitating risk communication and management decision making. The results of risk assessment and reduction and the residual risks are communicated to the project team for information and follow up.
This is a very preliminary top-level analysis, aimed at identifying first-risk trends:

- Earth Operations and software risks are *not* assessed.
- Legal & Programmatic risks are *not* assessed.
- Human errors are *not* assessed.

2.9.5.1 Assessment process

Step 1. Identification of hazardous/failure conditions. (what can go wrong…)
Step 2. Identification of failure scenarios and their consequences. (when…)
Step 3. Categorisation of the scenarios according to their consequence.(what if…)
Step 4. Analysis of likelihood and uncertainties of risks. (how likely…)
Step 5. Identification and ranking of risk contribution of individual scenarios.

Safety hazards	Examples
I. Contamination/ corrosion	Moisture, oxidation…
II. Electrical Discharge/ shock	Static discharge, short, corona…
III. Environmental/weather	Fog, vacuum, sand/dust, temperature extremes…
IV. Fire/explosion	Chemical change, high heat source…
V. Impact/collision	Meteoroids, rotating equipment…
VI. Loss of habitable environment	Contamination, toxicity…
VII.Pathological/ physiological/ psychological	Illness, excessive workload…
VIII. Radiation	Electromagnetic, radioactive element…
IX. Temperature extremes	High/low, variations…

Table 2-38: Technical risk assessment

2.9.6 Abort possibilities

Thorough investigations of Martian mission risks have not yet been performed. Acceptable risks can be achieved if abort options are designed into the mission for all phases. The abort option requirement eliminates mission profiles involving very fast and energetic trajectories, as shown in Table 2-39.

Phases	Options
Earth Departure	Return to Earth possible
Early Part of Transfer to Mars	Quick return to Earth usually possible for about the first 75 days
Later Part of Transfer to Mars	Mars swing-by (gravity assist) return to Earth via opposition-like trajectory
Mars Orbit	• Early: Return to Earth opposition-like trajectory • Later: Wait for normal Earth return opportunity
Mars Descent (not present in the present mission, but recommended)	Separate ascent stage and crew module; abort to Mars orbit
Surface Operations	Use Mars ascent stage; if it is inoperable there is no abort
Mars Ascent	No practical abort scheme
Trans-Earth Injection	No practical abort if main propulsion fails
Transfer to Earth	Continue normal return to Earth

Table 2-39: Abort possibilities

2.9.7 Risk acceptability

The purpose of this is to analyse the acceptability of risks and risk reduction options according to the risk management policy and to determine 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 2-58, but the maximum concern is risks to the crew. Human factors are 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 experience and information available and the preliminary definition of the design. This is expected to improve with the availability of more information and optimisation of the vehicles design, particularly with reference to the failure detection, warning, caution and recovery systems definition. The public safety of people on Earth is also of concern.

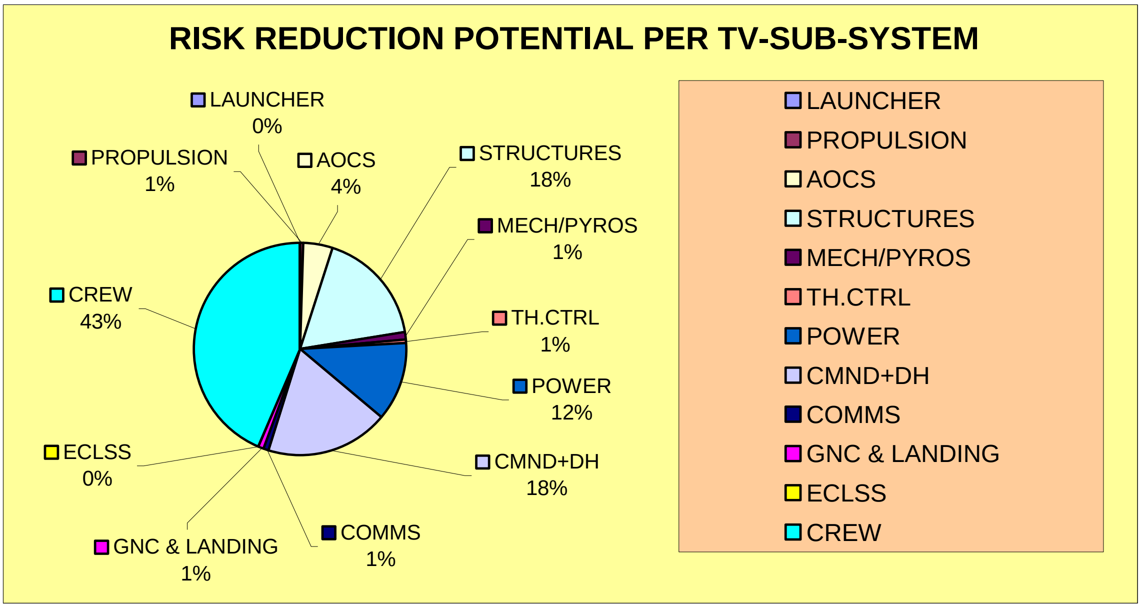


Figure 2-58: Risk reduction potential

Note that the results must be interpreted as a first trend of the technical risk status, therefore of a purely indicative nature. The risk analysis should be further developed during the project definition to analyse all the system, refine the risk identification and classification, and provide evidence that all the risks have been effectively controlled.

The colour codes in Table 2-40 represent (ref: ECSS-M-00-03A risk management):
Red: maximum or high risk. Proposed actions: implement new team process or change baseline and seek project management attention at appropriate high management level.
Yellow: Medium or low risk. Proposed actions: control, monitor and seek work package management attention.
Green: Minimum that is, acceptable risk.

Risk acceptability	Risk domain & scenario (indicative nature)	Reason-status
Unacceptable	1. *Maximum* likelihood with *catastrophic* consequences: 1. Human factors inadequate to mission. 2. EVA Suits inadequate to environment. 3. Inadequacy to radiation environment. 4. Landing on Mars failure. 5. ECLSS failure. 2. Maximum likelihood with critical consequences. Failures during AIV activities	3. Numerous critical areas with uncertain environment definition. 4. Research level only. 5. New project beyond the status of the art. 6. High level of autonomy required for operations. Highly complex Program.
Acceptable if reduction impossible	7. Medium likelihood with critical consequences. Communications loss.	8. Qualified technologies but never applied in projects. Numerous modifications of qualified product.
Acceptable	Others	Defined environmental conditions, qualified products, existing processes & facilities.

Table 2-40: Risk acceptability

2.9.8 Risk assessment process example

Figure 2-59 shows how the analysis is done, following the sequence of steps already mentioned at section 2.9.5.

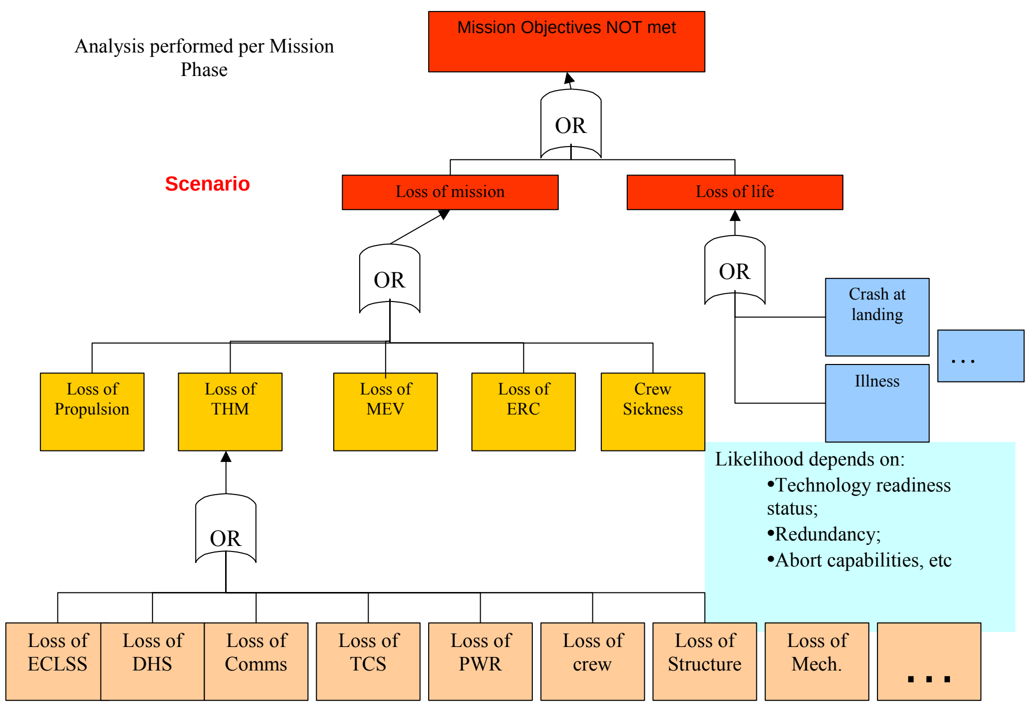


Figure 2-59: Risk assessment process

2.10 Ground segment and operations

2.10.1 Mission operations concept

The ground operations shall be able to support the following operational tasks:

- In-orbit assembly
- Operation of Life Support System
- Operation of subsystems
- Support crew work
- Support crew daily life
- Interplanetary transfer
- Mars orbit insertion
- Landing on Mars
- Ascent from Martian surface
- Rendezvous and docking in Mars orbit
- Insertion into Earth trajectory from Mars
- Earth reentry

2.10.1.1 On-board autonomy

The following autonomy concept is assumed:

- On-board autonomy (cruise and at Mars)
    - Short term for all situations
    - Medium term for nominal operations/situations by:
        - Crew to handle higher level decisions
        - On-board hierarchical operations concept with automatisms on lowest level

- Earth Orbit Assembly Phase under real-time ground control.

- Critical Phases Operations (cruise and at Mars)
    - Real time and lower level decisions by automatisms
    - Crew autonomy for decisions that do not allow for ground feedback
    - Ground control responsibility for high-level decisions as far as commensurable with ground feedback time
- Mission Planning (long term) on ground
- Monitoring on ground
- Permanent Presence of basic ground control
- Think Tank on ground on call to solve problem situations

The above autonomy concept is based on the following rationale:

On-board autonomy implications

Long feedback loops make ground real time control impossible. System is very complex, complete automation would be too risky and costly. High effort to make system operable for on-board crew

Earth orbit assembly phase

Near real-time ground control is feasible. Crew presence and involvement to be minimised. Assembly contains complex one of a kind steps. Higher level on-board autonomy is unfeasible and unnecessary.

Critical phases operations Nominal operations to be preplanned because they contain highly complex operations steps. Failure scenarios to be worked out with on-board automatic, crew and on-ground decision distribution as appropriate considering the feedback loop times.

Mission planning (long term) on ground

Natural task for the ground is to set the long-term aims taking into account complex mission parameters. Uplink of medium-term master time line.

Monitoring on ground

Highly complex (prototype) system requires detailed monitoring.

Think tank on ground to solve problem situations

Full understanding of the system is feasible only through on-ground sources.

Permanent presence of ground control

- Time for build up of critical situations shall be minimised
- Within reason smaller degradations shall be analysed to keep crew comfort at a high level, crew may also delegate lower level tasks to ground
- Crew has to be given the feeling of being taken care of by ground

Permanent Presence shall be provided on a best effort basis. Compared to the other items above it has a lower priority.

On-board survivability autonomy for credible failures without ground support, at least: *1 week*.

Time is driven by unavoidable gaps in communications, activation times of full expert ground support, and time to draw up recovery procedures.

(In contrast to unmanned satellites a safe mode with a deactivated satellite that buys time to solve the problem is not feasible for Human Mission to Mars. A basic functionality has to be always guaranteed.)

- On-board degradations management autonomy for 1 week supported by:
- Automatic redundancy management on lowest level
- Design for graceful degradations
- Crew interaction possibility on higher level to restore
- Ground control first-level support to crew available round the clock
- Ground control second-level support available on call

2.10.1.2 On-board operations

The following factors have been considered:

- Low-risk safe mode concept
- Graceful degradation of subsystems/components

- Higher robustness level required than with today's satellites (e.g. higher number of sensors, fuzzy logic connection between sensors, distributed intelligence connected by a net)
- Redundancy switching on lowest level (intelligent units required with self analysis capability) Active management of redundancy restoration on higher level
- Out of limit checks also on-board
- On-board capacity to repair
- On-board capacity to patch software
- Tools (e.g. simulators) required on-board (capacity for off-line testing required)
- Ground – orbit cooperation to fix things despite long ground feedback time
- More robust systems may make use of advanced technologies such as:
- Intelligent sensors
- Fuzzy logic
- Artificial sensors (on the basis of intrinsic redundancies)
- Intelligent consistency checks (on the basis of intrinsic redundancies)
- Intelligent fault tolerance and recovery to enable for early warning/reconfiguration and optimum use of on-board systems
- Optimisation of processes may make use of:
- Adaptive control
- Data fusion

2.10.1.3 Ground operations features

The following ground features have been considered:

- 24-hour-manned (small core team) monitoring of Martian TV and MEV
- On-call availability of full operational team
- Engineering capacity and real mock up on ground (requirement to track all configuration changes and actions on-board and to implement them on ground, increases down traffic for astronaut TCs)
- Operations processes to be updated to reflect manned safety
- Long feedback loops for deep-space limit the ground operation activities to planning off line monitoring and support
- Cooperation with crew for tasks requiring short feedback
- Continuous training of crew
- High degree of crew autonomy requires concept to track actions of crew
- Ground control would have to keep track of the changes and still keep an inventory of items and their state of usability

Concerning crew communication, the following shall be implemented:

- Crew to be enabled for email communications and internet access for personal, private, health, religion and work-related interests
- On-board personal server to be available to the crew on-board that mirrors internet sites of personal interest to be available to the crew
- E.g. a scientist working in his field of expertise. A three-year mission disrupts astronaut from contact to Earth scientific community. Astronauts should be able to contact and work with the Earth community via email and internet. High uplink rate expected. Safe firewall required

2.10.1.4 Communications

2.10.1.4.1 Earth orbit

S-band ground stations of the ESA LEOP network are used for LEOP and emergency. An additional station is set up to fill the Pacific gap.

The routine communications in assembly orbit are via a TDRSS. This structure is assumed to be still existing.

2.10.1.4.2 Mars orbit relay

Communications from surface of Mars to Mars orbiting transfer vehicle would be limited to short slots. 24-hour communications require a Mars stationary satellite.

Redundancy (approximately 10 h coverage/day) by direct link to Earth.

Relay satellite is also used to support rendezvous and docking communication and navigation.

2.10.1.4.3 Solar flare warning infrastructure

Early warning for solar flares is required, e.g. to reschedule EVAs. Potentially Sun-orbiting warning spacecraft(s) are required. Near 24-hour coverage for warning message is required.

2.10.1.4.4 Link requirements (cruise and at Mars)

- Near 24-hour/day communications capability required for all links

- Crew communications requires video:
- Currently available RF links not sufficient at far distances
- Link should be available at crew working hours
- Availability should exceed 90%

- Crew requires internet capabilities:
- High performance (off line) uplink (with on-board server)
- Link should be available round the clock
- Availability should exceed 90%

- Crew requires permanent presence communications link:
- Near real time video, voice, and e-mail at a medium level performance
- Link should be available round the clock (small gaps allowable)
- Availability should exceed 95%
- Availability should exceed 99% for 18 hours within a full day

- Spacecraft is one to two orders more complex than today’s planetary spacecrafts, software complexity and diversity may be even higher

- housekeeping, commanding and software maintenance require up to two orders of magnitude more link capability
- as for satellites one contact/day with feedback possibility is sufficient but also necessary
- availability should exceed 99% for 8 hours/day contact

- Essential link is required that guarantees communications in all conceivable contingency situations. Degradations with respect to the links above are acceptable.

2.10.1.4.5 Communications link concept (cruise and at Mars)

Essential link
(operating under all failure scenarios, near real time, near 24-hour/day):
X-band 70 m ground stations, MGA on-board

- Crew – ground communications: e-mail
- Essential housekeeping data downlink
- Essential commanding capability to support on-board autonomy

Basic RF link
(near real time, near 24-hour/day, maximum outage 24-hour TBC):
Downlink
Based on a scaled up Ka-band implementation (70 m ground station, 4 m on-board terminal)
Uplink
Based on Ka-band with major improvements in the uplink data rate (by a combination of coded uplink, ground station high power amplifier power improvements, reduced losses, and low on orbit antenna and system temperatures.)

- Crew – ground communications: (basic) video, voice, internet and e-mail
- Housekeeping data, basic science data and software image downlink
- Commanding, master plan uplink and software update

High-performance optical down link
(near real time, 24-hour/day capability (12 to 18 hours/day duty cycle), best effort):

- Crew – ground communications: state of the art video, voice, and e-mail
- Housekeeping data, science data and software image downlink
- High-performance communications are vulnerable (e.g. Ka-band and optical links). Redundancy on a lower performance level is required. Communications shall be organised hierarchically. In case of loss of the high performance link, essential data still have to reach the ground (either along another path or in the same path with increased signal to noise ratio).

2.10.1.4.6 Tracking

- The infrastructure shall be able to track orbits with the following methods:
- Doppler
- Ranging
- GPS or Galileo in Earth orbit

- Delta DOR
- Same Beam Interferometry
- Satellite to satellite Doppler and Ranging
- The accuracy of the above methods shall be at least the state of the art projected for 2010 (deltaDOR: 5 nrad, Doppler and Ranging as BepiColombo Radioscience (0.6*10-15 Allan variance and 10 cm ranging ground station contribution)
- Navigation on Martian surface implemented locally (i.e. not via ground)

2.10.2 Ground segment concept

The HMM is unique in several aspects:

- It has a number of technologically driving aspects, e.g. large elements rendezvous and docking, optical communications etc. These have to be prepared by studies and precursor missions.
- Its large size requires a special build up phase to create the pool of people with the respective expertise. This expertise cannot be drawn from the limited number of people involved in conventional missions.
- The staffing cannot easily be adapted to changing workloads, because there are no other missions that large to absorb such a large workforce.
- Existing centre infrastructures cannot just be adapted. New centre facilities must be created.

2.10.2.1 Central versus distributed control centre concept

The HMM flight operations can be subdivided into major tasks:

- Launches
- LEO assembly
- Habitation module mission
- Martian surface mission

Each respective major task should be operated centrally. It is, however, conceivable, that there are different dedicated Control Centres for each major task.

No clear preference is given at this stage to a central or distributed approach for the overall HMM. A control infrastructure of this size is not available and has to be built up. A central structure makes sense if it is reused for similar endeavours. On the other hand, smaller (but still large) control infrastructures for the individual major tasks may be easier to adapt if smaller missions are conducted before and thereafter.

As far as the communications network is concerned there is clear preference for a centralized approach, i.e. there should be one centre in charge of network operations.

Except for the launches, each major task is assumed to be organised hierarchically with a system entity on top supported by entities responsible for the subsystems. The subsystem teams and tools are shared between the major tasks as applicable.

It is not excluded that special components/instruments are controlled within boundaries set by the major tasks via special centres (called National Control Centres in Figure 2-60).

2.10.2.2 Major mission tasks

2.10.2.2.1 Launches

Ariane

It is assumed that the Ariane launches are based on a derivative of the ATV launches. The operation of the ATV is assumed to be performed by the CNES ATV operations centre. The approach and associated recurrent costs are assumed to be similar to currently planned ATV operations at the ISS.

Because of the complexity of the cargo and its sophisticated subsystems a purely passive launch is not assumed, so the cargo operations therefore require an operational effort. This is assumed to be provided by the LEO Assembly task.

Considerable nonrecurrent effort is required to develop the launch, rendezvous and docking concept under consideration of the safety and complexity of the to-be-assembled large composite. This effort is shared between the Ariane launch task and the LEO Assembly task.

Proton

It is assumed that the Proton launches are based on a derivative of a Russian upper stage (e.g. Fregat-based TBC). The operation of the vehicle is assumed to be performed in Russia. The approach and associated recurrent costs for the upper stage operation are assumed to be similar to currently planned Soyuz flights to the ISS. The effort is assumed to be included in the launch costs.

Because of the complexity of the cargo and its sophisticated subsystems a purely passive launch is not assumed TBC, the cargo operations thus require a TBD operational effort. This is assumed to be provided by the LEO Assembly task.

Considerable nonrecurrent effort is required to develop the launch, rendezvous and docking concept under consideration of the safety and complexity of the to-be-assembled large composite. This effort is shared between the Proton launch task and the LEO Assembly task.

Soyuz

The Soyuz missions are assumed to resemble those to the ISS. The missions are bought in Russia.

Energia

For the delivery of the composite elements to the assembly composite it is assumed that the Energia's upper stage is used. This may introduce some design changes in the actual design of the RCS.

As far as the Energia launch to low circular orbit is concerned, this is bought including operations in Russia.

The upper stage operations are assumed to be directed from a dedicated operations centre.

2.10.2.2.2 LEO assembly

The LEO Assembly task is in charge of all operations of the spacecraft composite until end of assembly/start of commissioning.

The LEO assembly is concerned with integrating a new element into the transfer spacecraft composite structure every 50 days over a timespan of 3.5 years. Each type of composite component needs its dedicated team of specialists to look after. In addition there are highly complex robotic operations to be conducted with the service platforms. To be able to do the assembly in a minimum time, shift work is assumed during the commissioning periods.

The spacecraft composite does not support an early manned capability. The early EVAs (when needed) are performed by the visiting Shuttle crew. The responsibility for these EVAs is assumed to rest with the Shuttle operator.

The concept is to establish a dedicated operations infrastructure specialized in the assembly of the transfer spacecraft. A hierarchical control infrastructure is proposed for the overall spacecraft and the composite components geared to the changing characteristics of the built-up spacecraft composite.

Operations Architecture:

LEO Assembly Flight Operations Team, consisting of:
- System Team
- Mission Planning Team (in control of launches and resources)
- Component Operations Teams (one team for each type of component)
- Workbench Operations Team

Industrial Support Team for LEO Assembly with system and subsystem experts (DHS, thermal, power, mechanisms, AOCS etc.), responsible for the following teams:
- System Team
- Propulsion Stages MOI
- Propulsion Stages TMI
- Service platforms
- SM
- Airlock and Cupola
- PM Central Structure Items
- Habitation Module
- Launcher Interface

Mission Control System, consisting of:
- System mission control system
- Component mission control systems

Simulators, consisting of:
- System simulator (being able to represent the different stages of assembly)

- Component simulators

Flight Dynamics, consisting of:
- Spacecraft composite control
- Rendezvous and docking control

2.10.2.2.3 Habitation module mission

The Habitation Module Mission task is in charge of composite operations from start of commissioning up to return to Earth.

Although the Habitation Module Mission task has taken over the overall responsibility, the teams working for LEO Assembly stay intact up to commissioning after Mars trajectory injection.

The operational effort is governed by two requirements:
- to provide a permanent operational presence for the crew over 2.6 years
- to provide on-call expertise for the whole spacecraft over 2.6 years

Operations Architecture:

Habitation Module Mission Team, consisting of:
- System Team
- Mission Planning Team
- Crew Interface Team
- Component Operations Teams (one team for each type of component and habitation module subsytem)
- Industrial Support Team on call
- Habitation Module ground mock up

Mission Control System, consisting of:
- System mission control system
- Component mission control systems

Simulators, consisting of:
- System simulator (being able to represent the different stages of assembly)
- Component and habitation module subsystems simulators
- Ground reference simulator

Flight Dynamics, consisting of:
- Spacecraft composite control
- Manoeuvre control
- Rendezvous and docking control

Data Dissemination (TBD)

2.10.2.2.4 Martian surface mission operations

The Martian Surface Mission Operations task is in charge of the descent, landing, surface operations and ascent from departure from the habitation spacecraft up to docking again.

Although the mission cumulates in a short duration surface presence, the respective operations preparation and tools development require a large operations infrastructure to be set up.

Operations Architecture:

Martian Surface Mission Operations System Team, responsible of:

- MEV operations
- SHM operations
- EVA operations
- MAV operations

Each element team has a dedicated mission control system and simulator and has to cover the operation of all subsystems. The MEV and MAV operations in addition have flight dynamics support.

2.10.2.2.5 Mars relay satellite

The Mars Relay satellite is seen as a mission in itself. It can be operated from a separate Mission Control Centre.

The operational effort is estimated to be of a similar order of magnitude as a current-day Mars observation mission.

2.10.2.3 Timeframe

Table 2-41 shows the timeframe for the mission:

HMM Timeframe			
Start of Activity	Time to Departure [years]	Operations Duration [years]	Activity
Project/Mission Activities			
31 October 2020	-12.5		Phase A start of first element
29 October 2028	-4.5		First launch
29 December 2032	-0.3		End of assembly
16 January 2033	-0.2		Start of commissioning
16 April 2033	0.0		Departure from Earth orbit
11 November 2033	0.6		Mars arrival
28 April 2035	2.0		Mars departure
27 November 2035	2.6		Earth arrival
LEO Assembly Operations Task			
1 May 2017	-16.0	5.0	Operations support to system study
1 May 2022	-11.0	3.0	Assembly operations definition

HMM Timeframe			
Start of Activity	Time to Departure [years]	Operations Duration [years]	Activity
1 May 2025	-8.0	3.0	Assembly operations preparation
29 April 2028	-5.0	3.6	Test and Validation (0.5 years per individual launch)
29 October 2028	-4.5	3.5	Assembly Operations
16 January 2033	-0.2	0.2	Support to Habitation Module Mission Task
Habitation Module Mission Operations Task			
1 February 2019	-14.2	5.0	Operations support to system study
1 February 2024	-9.2	4.0	Operations definition
1 February 2028	-5.2	4.0	Operations preparation
16 January 2032	-1.2	1.0	Test and Validation
16 January 2033	-0.2	0.0	Take over of S/C composite operations
16 April 2033	0.0	0.5	Transfer to Mars
16 April 2033	0.0	0.1	Commissioning in transfer orbit
11 October 2033	0.5	1.5	Mars operations
28 May 2035	2.1	0.5	Transfer to Earth
Martian surface Mission Operations Task			
1 February 2017	-16.2	5.0	Operations support to system study
1 February 2022	-11.2	5.0	Mars mission operations definition
1 February 2027	-6.2	5.0	Mars mission preparation
16 January 2032	-1.2	1.0	Mars mission test and validation
16 January 2033	-0.2	0.7	Standby Period
11 October 2033	0.5	0.2	Martian surface Operations (no sand storms assumed)
Ground Station and Communications Network			
1 April 2021	-12.0	3.0	Operations infrastructure technology studies
1 November 2024	-6.0	3.0	Operations infrastructure build up
29 October 2027	-3.0	1.0	Test and Validation (0.5 years each element)

Table 2-41: Mission Timeframe

2.10.2.4 Ground station network

TDRSS services
The TDRSS service is rented from NASA, only the communication lines to NASA have to be set up.

LEOP network
An additional 15-m LEOP station is set up in the Pacific.

Essential link
The essential link is performed with the existing deep-space X-band stations plus a X-band capability at (at least two) of the 70-m Ka-band stations.

Basic RF link
Four 70-m Ka-band stations with more than 20 kW RF uplink power are set up.

Four Ka-band stations are taken, because the Ka-band link is very vulnerable at low elevations. For the required availability four stations distributed over the Earth are required.

The technology is not yet available, neither for the precision pointing nor for the uplink power, but is expected to be available in 2030.

High performance optical down link
Six 10-m optical terminals are set up.

The telescopes are assumed to be of photon bucket design. A photon bucket has a large photon gathering area but only a limited optical quality. (The signal reception is limited by Poisson statistics because of the limited number of photons received.) These telescopes are only used for data reception. Modulation will be based on pulse length. Coherent modulation schemes may be not feasible.

Choosing good sites (such as the Tenerife mountaintop site), means availabilities of 84% are achievable for a single station. To achieve above 90% availability, at least two telescopes have to be visible from the spacecraft at any time. The telescopes have to be at distances of more than 2000 km from each other to be in areas of different weather patterns.

Weather is permanently monitored. The spacecraft switches beam pointing to a redundant station if weather conditions are bad at one station (beamwidth on Earth is only 350 km to 1250 km).

The performance at low Sun-Earth-S/C angles is unclear. Buffers of for example 5 to 10 times the telescope diameter seem impractical. Heating of primary mirror will require design similar to Sun observation telescopes with forced cooling. (Night operations only is not acceptable.)

The technology at this scale is unproven. Current ESTEC technology studies reveal technical performance limits. One of the problems is the on-board pointing. The pointing device has to be decoupled from microvibrations and astronaut movements.

2.10.2.5 Communication network

A dedicated worldwide communications network (see Figure 2-60) needs to be set up. It serves all the parties involved in the ground and flight operations of the mission.

The network consists of two separated networks: the Ground Station Network and the Operational Network. Both are connected via the Mars Operations Control Centre.

Only a single control centre (MARS OCC) is shown in Figure 2-60. In the case of a distributed concept, control centres dedicated to a major task can be introduced into the network concept.

An essential feature of the network are the firewalls to enable the astronauts to communicate with the world.

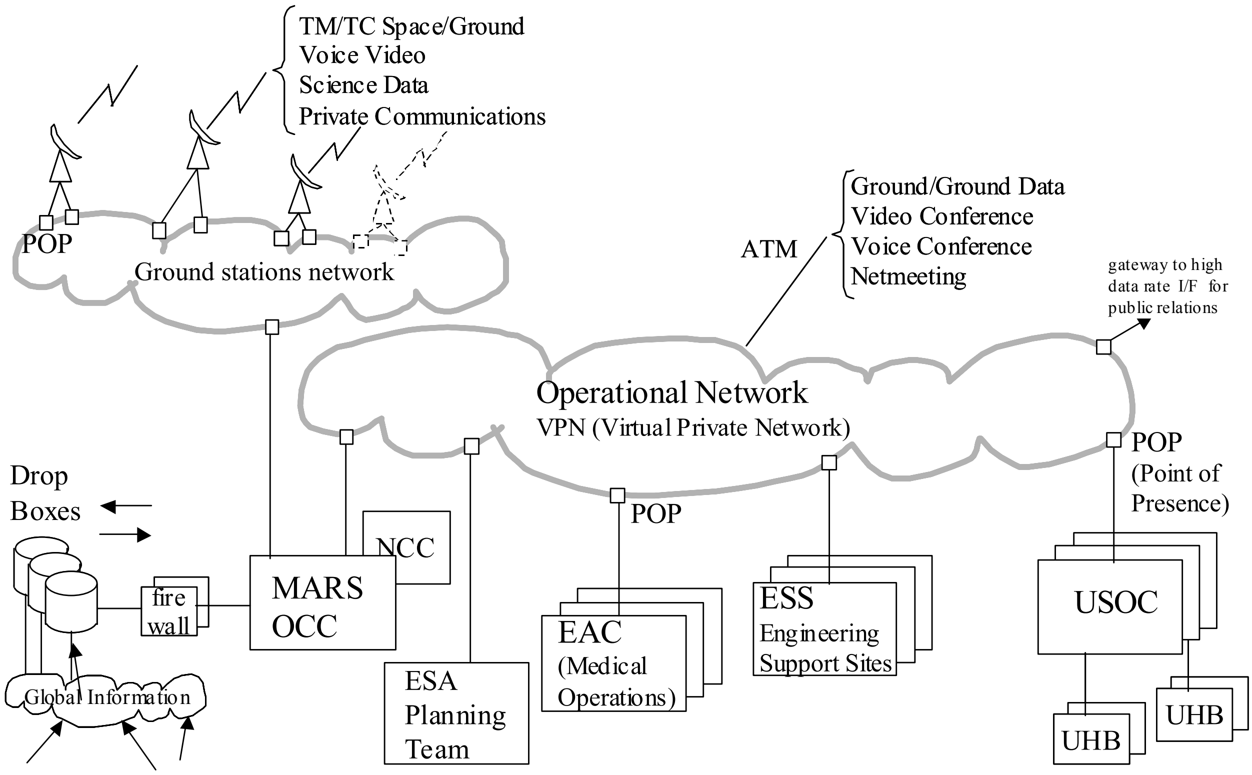


Figure 2-60: HMM Communication Network

2.11 Simulation

The result of the simulation activity for the HMM study are movies in which the astronauts' relative size with respect to the transfer vehicle and astronaut mobility are shown. The objective of the simulation activity is to show that the astronauts have enough space and that there are no obstacles to their movement around the vehicle.

The current tools used for simulation were not suitable, so a new tool had to be identified. The selected tool had to facilitate the creation of astronaut characters and their manipulation. Also it should be able to import the CAD data generated by the configuration engineers.

2.11.1 Simulation results

Two suitable software tools were considered: Maya 5 and Poser 5.
Maya is a solid tool used for the cinema industry for 3D animation, it had been used before for the Exomars study. It therefore complies with the CAD import requirement. It also provides a powerful dynamic engine, but unfortunately it is not very accurate; therefore it is not recommended. The tools allows to import human characters, but a skeleton and skin deformers had to be added to it manually later, which is time consuming.
Poser is a tool meant to generate human still poses with good quality and photo-realism. It also provides a basic key framing animation tool to add dynamism to the characters. The main

advantage of Poser is that it comes with predefined human characters with a high level of detail. The manipulation of their limbs and body is also very straightforward. Also it provides a simple library of clothes and hair.

A scene from the simulation produced, with a cutaway showing six astronauts is shown Figure 2-61:

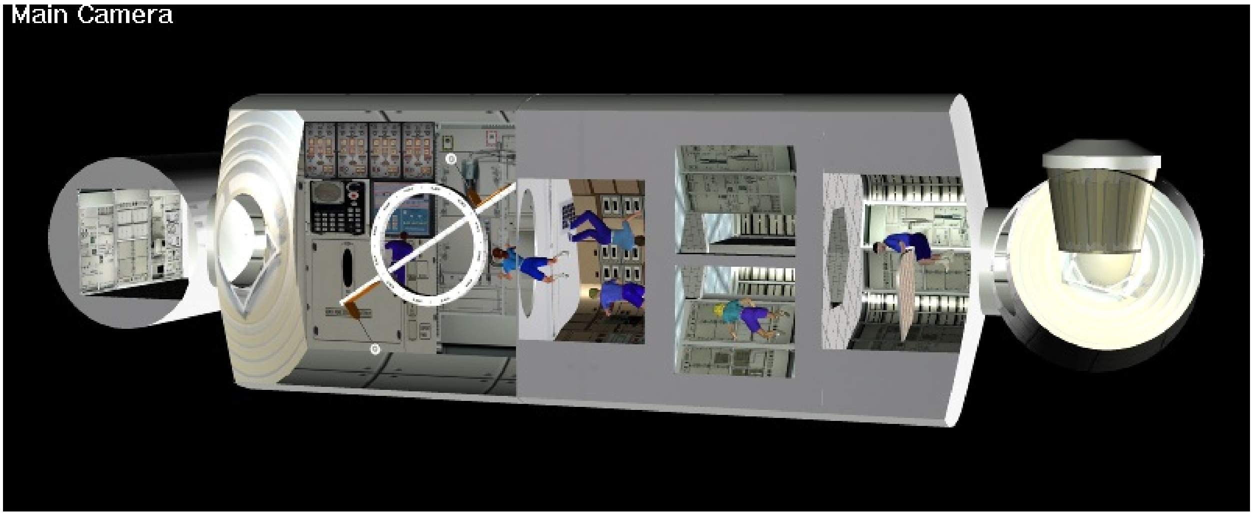


Figure 2-61: Transfer Vehicle cutaway

Figure 2-62 shows a close up of an astronaut. This is useful for collision detection and to evaluate distances:

Figure 2-62: Astronaut posing view

2.11.2 Simulation tool selection

Poser 5, the selected tool, provides some functionality for human factor studies but it is limited in animation.
Problems identified are:

- Camera manipulation problems, particularly when the scene became more complex.
- Photo-realistic render engine produces lower quality than the real-time engine due to lack of control over the Z-buffer resolution.
- The tool does not take advantage of the hardware-accelerated anti-aliasing and therefore it is very slow producing anti-aliased movies.
- The tool is not multiprocessor optimised.
- Object manipulation is not easy.

However, the simulation produced within this study has shown that the internal configuration design is sound and complies with the basic Human Factor requirements.

2.12 Programmatics

2.12.1 Requirements and programmatic drivers

The main requirements for the study, as used in the programmatic assessment, were to:

- Design a system able to support the journey of a crew of six members to Mars orbit, to land three of them on Martian surface, to provide crew shelter and base of EVA operations on the Martian surface, to safely return the Mars excursion crew to the orbital vehicle, and to return to Earth
- Consider the mission requirements, namely, overall mission duration (from TMI until Earth landing) of about 1000 days with a Mars excursion time of about 30 days
- Design the system taking into consideration as much as possible available technology
- Consider that no specific launchers can be developed for this mission, therefore nearly available launchers only.
- Consider that the in-orbit assembly time should not last more than 6 years, with a goal of 2 years.

2.12.2 Assumptions and trade-offs

With the selected mission scenario, the mission opportunity (injection into Earth – Mars Transfer orbit) is every 2 years.
The launch rate will be limited by the availability rate of launchers (mostly Energia), the launch campaign constraints, and the delivery rate of the vehicle modules.
Planetary protection rules have to be applied.
Because of the complexity of the in-orbit assembly phase, orbital infrastructures are needed to support the integration of the space vehicle elements. The design of these infrastructures is not exploited in this study.
Design drivers for these support systems are the required availability of a robotic arm to enable handling and berthing of the vehicle elements during the assembly phase, their required capability to actively cool down the cryogenic propulsion tanks, their required capability to provide attitude control to the spacecraft modules during assembly, and their man-tended capability.
In the case of the propulsion stages, given that they are built with a central backbone structure around which the propulsion modules are assembled, a possible trade-off is to evaluate the way

of designing the backbone structure as initial assembly support structure, providing it with manoeuvre capability and robotic arm(s).
The prolonged exposure to 0-g conditions is negative for crew health and planetary surface operations capability. The implementation of micro gravity countermeasures for the crew is considered necessary. A trade-off was performed, and a system was described, able to provide to the crewmembers artificial gravity (centrifuge) during their sleeping hours. The effects of this partial compensation are not well known yet, so they should be studied with a precursor experiment e.g. on the ISS.

2.12.3 Model philosophy and qualification

Due to system complexity, Qualification Models (QMs) are required for the Mars Excursion Vehicle (complete), and Earth Return Capsule (ERC) elements.
Some QM elements will be tested together on ground.
For schedule reasons the flight elements will not be tested together on the ground. Interface reference models have to be produced for ground testing of the system elements. These models have to be based on the design at the QM maturity stage. In principle a QM element will be used in combination with an interfacing FM element to perform system interface tests. Later on, the QM is kept as reference model for the whole mission.
The capability to load the cryopropellant in-orbit has to be significantly improved, to support the required boil-off compensation before TMI. The loading system and its relevant operations have to be qualified with a dedicated flight mission. This could be accomplished after one of the first launches of the propulsion elements, to verify the loading on a reduced configuration of the system.
For a program of a similar time span is the obsolescence of the components is a problem. Considering the high rate of innovation in the field of electronic components such as processors, memory banks and computer boards, it is guaranteed that from the time of design until the mission exploitation, parts will rapidly evolve and new generations will replace the old. Design of avionic subsystems and units would quickly become obsolete. It is therefore necessary to implement some mitigating factors to this process:

- Applying open design to facilitate the implementation of more modern (space qualified) parts along the development phase, as soon as they become available.
- Selecting components in the field of military or commercial aviation, where their usage is planned to last decades, so the production lines are kept alive accordingly.

2.12.3.1 Qualification flight

A qualification flight is required. There is no way to assess the system’s real capability to perform its mission and to verify its actual reliability without performing a flight test.
In the best scenario, a scaled model of the manned vehicle should be built, and launched. It should be piloted in fully automatic mode. It should perform a complete mission sequence that includes:

- Injection into transfer orbit to Mars
- Capture and injection into Low Mars Orbit
- Descent, landing and deployment of a Surface Element (full scale) on the same landing site selected for the manned mission
- Launch and ascent of a Mars Ascent Vehicle (scaled)

- Rendezvous and docking to the Orbital Vehicle (scaled)
- Injection in a transfer orbit to Earth of the Earth Return Vehicle (scaled)
- Landing of the Earth return capsule (it could bring back Martian samples, if deemed suitable)
- Exploitation of the Planetary Protection procedures, as far as their unmanned parts are concerned.

One advantage of this model is that it could be possible to fully verify the system and gain vital mission data for the safer performance of the next manned mission. In addition, the landing and deployment of a Martian surface Element could provide additional back up to the next mission, making available to the astronauts another fully equipped habitable module. The Surface Element could be set in a quiescent state after the post-landing activation and checks, with just telemetry of housekeeping data, surrounding environment data and pictures.

An alternative qualification flight concept could be exploited in Earth orbit, in a reduced configuration of the final spaceship (partial assembly) that could enable the exercise of flight manoeuvres:

- Exercise separation, rendezvous and docking (a dedicated reduced flight model of the MEV should be built. It would be expendable)
- Exercise partial thrust activation and propulsion stages separation (dedicated reduced flight models of the TMI and TEI should be built. They would be expendable)
- Verify the flight vehicle on-board subsystems
- Assess system's true reliability
- Crew should be on-board and issuing commands

Even if not a complete mission, this flight test would dramatically improve the knowledge of the spaceship behaviour in the real environment conditions. However, a second cycle of assembly operations would still have to be performed on the inhabited section of the spaceship, to mate it for the propulsion stages.

2.12.3.2 Descent and landing system qualification

From a programmatic point of view it is clear that whichever the choice, the qualification of a full vehicle will have a much higher impact (time and cost) than qualifying an aero shell only.
The qualification of the landing system requires the build-up of a dedicated facility. A support test facility is required, to verify first a development model, then a qualification model, the landing system controls, end to end. The test facility shall include remote control and direct (from the lander model) control by the astronauts. It will be a training facility for the pilot astronauts too. It is a reasonable assumption to rely on previous development and experience from previous lunar manned missions.

2.12.4 Baseline operations

A realistic delivery rate for the propulsion modules of the various stages is 2 months and it would fit with a reasonable assumption of 2 months for the launch rate of the modules, i.e. with the availability rate of the launchers (mainly Energia).
The current assembly sequence defines a serial integration of the spacecraft modules/elements.
As regards safety, it would be reasonable to assume that the assembly of the propulsion stages would be performed at a safe distance from the habitable elements of the spaceship. This would

imply that either the propulsion is assembled first, and the habitable elements later, or that the two parts, manned and unmanned, are integrated in parallel but at different orbital stations. The means to mate the two parts would include an orbital manoeuvre that has not yet been studied.
The propulsion stages need 18 launches for in-orbit assembly. Assuming a launch rate of 2 months, and one month of in-orbit assembly and commissioning, about 36 months (3 years) are necessary to complete the propulsion system build-up. All launches are with Energia.

2.12.5 Options

Besides the assembly option mentioned above, a further time saving would come from a power-up of the Energia launch capability. If this launcher could be made ready for launch every month, possibly using two separated launch processes and launchpads, and provided that the in-orbit commissioning can actually be accomplished in one month each propulsion module, it would be possible to save between 1 and 1.5 years along the in-orbit assembly process. The investment and the relevant benefits of doing this should be assessed with a dedicated analysis in the next phase.

3 TRANSFER VEHICLE

3.1 Systems

The Transfer Vehicle is the vehicle that hosts the crew for most of the mission. It acts as transfer vehicle both in the outward and the inward cruise and must provide an adequate habitat for the astronauts as well as the propulsion means required.

The transfer vehicle is composed of the Transfer Habitation Module (THM), where the astronauts will live, and the Propulsion Module (PM), that will provide the propulsion system for the main propulsive manoeuvres.

3.1.1 System requirements

At the beginning of the study the following requirements were established:

	Max	Min	Ideal	Units
Study Objectives				
Study a minimal achievable first mission with capability of extending to more advanced subsequent missions				
Architecture design for all elements and phases of the mission				
System and S/S conceptual design for key elements and phases				
Assembly, operations and logistics assessment				
Safety and Technical risk assessment				
System Requirements				
Land a crew of humans on Mars by 2030 and return them safely				
Deliver 6 crew members Mars			6	
Land 3 astronauts on Mars			3	
Bring back to Earth at least 100 kg of Martian samples		100	100	kg
Astronauts shall perform exploration on Mars surface				
Maximise the reusability of the mission elements / design				
Crew non survival shall be lower than 1 in 200	0.01		0.01	
Mission constraints				
Operational date	2040	2025	2030	
Launcher			Energia	
Reference First Mission Requirements				
Departure date			2033	
Trip time E-M			207	days
Time around Mars			553	days
Low altitude circular orbit around Mars			500	km
Time on Mars surface			30	days
Trip time M-E			206	days
Minimum ΔV for TMI		3639	3639	m/s
Minimum ΔV for MOI		2490	2490	m/s
Minimum ΔV for TMI		2254	2254	m/s
Direct entry at Earth arrival				

The TV is composed of the THM and the propulsion stages, TMI, MOI, TEI					
THM lifetime shall be longer than 6 years			6		Years
Life time of propulsion stages shall be long enough to cover the assembly in orbit and mission phases for which they are designed		2	5		
Each propulsion module shall be discarded in a safe way after its usage					
Probability of Mars impact shall be lower than10^-4				10^-4	
THM shall be safely discarded, avoiding the Earth Moon system					
Science and exploration shall be performed on board during transfer and orbiting around Mars phases			600		kg
THM shall provide a storm shelter to protect the crew in case of a Solar Particle Event					
THM shall be able to suport life during at least the mission duration and TBD during assembly in orbit		6	3		
THM shall provide communications with the Earth and the MEV					
THM shall provide EVA capabilities:					
* shall provide an airlock					
* shall provide EVA suits					
TMI shall provide the required impulse to put the TV on its orbit towards Mars					
MOI shall provide the required impulse to put the TV on its orbit around Mars					
TEI shall provide the required impulse to put the TV on its orbit towards Earth					
THM shall provide the capability of manoeuvring to skip the Earth Mars system at Earth return					
Mission Constraints					
Assembly in LEO				400.00	km
Assembly shall take no longer than 2 years		2.00		2.00	Years
Safety Requirements					
Rescue of the crew and/or abort of mission shall be possible during phases: TBD				TBD	
Single failure/fault/operator error tolerance for critical hazards.					
Two failure/fault/operator tolerance for catastrophic hazards.					
Failure detection, isolation and recovery means shall be provided (automatic and manual)					
TV shall provide automatic detection means for at least the following hazards:					
* Fire					
* Depressurisation					
* Biohazards					
* Atmosphere degradation conditions					
* Radiation					
* Temperature					
* Food spoilage and water contamination					
The TV shall provide a Caution and Warning System (C&W, this system must be able to receive system data, inform the crew of off-nominal events, and provide sufficient information to direct the crew to the correct response)					
TV shall be one failure tolerant to prevent loss of an EVA crewmember due to inadvertent separation from TV					
The TV shall have a 0.81 (minimum) combined probability of no penetration (PNP) of meteorite/orbital debris critical items during the mission			0.81		
Physiology Requirements					
g-loads should be lower than (in the +Gx axis)					
* Earth Departure		6.00			g
* Mars Arrival		4.00			g
* Mars Departure		4.00			g
* Earth Arrival		4.00			g
Habitable volume per crew member shall be:		30.00	20.00	25.00	m^3
THM shall provide appropiate public and private areas to sustaint optimal living and working conditions					
Radiation Organ Specific Equivalent dose Limits (BFO)					
Accute event		0.15		0.15	Sv
30 days		0.25		0.25	Sv
Year		0.50		0.50	Sv
Career		1 to 4	1 to 4	1.00	Sv
THM shall provide equipment in order to minimise the deconditioning of the crew, exercise and artificial gravity shall be considered					
THM shall provide medical equipment for the crew					
See Human Factors for more details					

Operational Requirements						
It shall be possible to command the TV in an automatic way						
TV shall be controllable from Earth						
TV shall be controllable by the crew						
Crew shall be able to override the automatic control						
EVA operations shall be kept to a minimum during transfer and orbiting around Mars phases				0.00		
Capability of inspecting the the external and internal parts of the vehicle shall be provided						
TV shall provide means for corrective and preventive maintenance						
Crew time dedicated to maintenance shall be minimised						
On-board training capabilities shall be provided						
Assembly in orbit						
Assembly in orbit shall be as automated as possible						
Assembly shall be performed in LEO				400.00	km	
Capability for verification on orbit shall be provided						
* all connections shall be verified						
* all functions shall be verified						
Replace/Repair capability of system modules shall be provided during the assembly phase to ensure full functionality and redundancy prior to committing to departure						
Assembly phase shall kept as short as possible		2.00		2.00	Year	
Planetary Protection Requirements						
For the forward contamination:						
1.) The probability of an impact on Mars by any part of the launch vehicle (also launch from LEO in that case) shall not exceed 1E-4						
2.) If the Mars orbiting spacecraft does not meet the Viking pre-sterilization bioburden levels the probability of an impact on Mars shall not exceed 1E-2 for the first 20 years of the mission, and shall not eceed 5E-2 for the time between 20 and 50						
For backward contamination:						
1.) All sample material returned from Mars shall be contained, and containment shall be verified before entering the Earth-Moon system						
2.) It shall be possible to isolate the surface crew from the rest of the crew on the habitation module for a TBD period of time after returning from the surface of Mars.						
3.) Contamination of the THM shall be avoided during all mission phases (in that respect, docking the ascent vehicle and transfer of crew and material from there to the habitation module is critical)						
Interface requirements						
Interfaces between assembly elements shall be kept to a minimum to simplify the assembly						
THM shall be as independent as possible from the Propulsion Transfer Stages						
THM shall provide interfaces with the ERC, allowing the crew to pass from THM to ERC and back						
THM shall provide interfaces with the MEV, allowing the crew to pass from THM to MEV and back						
THM shall provide interface with the Propulsion Transfer Stages to transmit the loads and commands						
THM shall act as data relay between the MEV and Earth						
THM shall provide housekeeping functions to the rest of the mission elements (ERC, MEV) when they are in stand by						
Interfaces shall be standarised						
Propulsion						
All propulsion stages shall be designed to provided the required trajectory changes at each mission phase						
Staging shall be considered within each main propulsive manoeuvre (TMI, MOI, TEI)						
Cryogenic stages shall be considered for TMI						

Table 3-1: Transfer Vehicle high level requirements

After the realisation of the study, it was discovered that some of these requirements cannot be fulfilled with the proposed design. For example:

- Mission success requirement cannot be evaluated at this stage
- Assembly in orbit cannot be performed in less than 2 years even in the most optimistic scenario
- Lifetime of the different modules will have to be set after the assembly in-orbit sequence is defined
- Abort is not possible during all the mission phases

3.1.2 System design drivers

The THM main design drivers are:

- The habitability requirements, which determine the size of the vehicle. A minimum of 25 m^3 has to be provided for each astronaut. It leads to a total pressurised volume of 450 m^3
- It is the main payload for the propulsion module, so its mass has to be as low as possible
- High level of closure of the life support system to reduce the total mass
- Launcher constraints in terms of dimensions

- Safety, so a fail-safe design is mandatory
- Has to provide interfaces with all the other elements, because it is the 'backbone'

The PM main design drivers are:

- ΔVs required for the mission are of the order of 3 km/s for each manoeuvre
- Large payloads, specially the THM which has to be sent to Mars and brought back
- Launcher constraints in terms of mass and dimensions
- Only cryogenic and storable technologies to be used
- Boil-off rate
- Usage of existing engines

3.1.3 Mass budget

The mass budget for the THM is shown in Table 3-2:

Transfer Habitation Module		
	Mass (kg)	Margin applied (%)
Total Mass with Margin	**66764**	
Total Dry Mass with Margin	56545	
System Margin Applied	9424	20
Structure	12468	7.8
Thermal Control	3920	8.7
Mechanisms	5445	14.5
Communications	162	13.4
Data Handling	660	20
GNC	1800	0
Propulsion	0	0
Power	4483	20
Harness	2000	0
Lifesupport	15592	14.4
Consumables:	10219	
Potable Water	1009	
Hygiene Water	324	
Dry Food	3831	
Oxygen	394	
Packaging	1392	
Inorganic Material (excluding packaging)	3178	
Nitrogen	91	
Waste generated (from consumables)	5375	
Payload	0	0
Astronauts	591	0
Total Propellant Mass	0	0

Table 3-2: Mass budget for the THM

The mass budget for the propulsion modules is shown in Table 3-3:

	Trans Mars Injection Module	Mars Orbit Insertion Module	Trans Earth Injection Module
Number of stages	3	2	1
Number of stacks	12	4	1
First stage	4	2	1
Second stage	4	2	
Third stage	4		
Number of supporting structure	3	1	1
Stack design			
Total mass with margin (kg)	80000	80000	80000
Total dry mass with margin (kg)	9214	3676	3676
Structure with margin (kg)	5702	2416	2416
Thermal control with margin (kg)	843	95	95
Mechanisms with margin (kg)	33	36	36
Propulsion with margin (kg)	2637	1130	1130
Total propellant mass (kg)	70786	76324	76324
Supporting structure design			
Total mass with margin (kg)	5178	3572	683
Structure with margin (kg)	3776	2533	523
Mechanisms with margin (kg)	540	443	46
System margin (%)	20	20	20

Table 3-3: Mass budget for the PM

Therefore, the total mass to be launched into LEO is:

Element	Mass per unit (tonnes)	Number	Total mass (tonnes)
THM	66.7	1	66.7
TMI stacks	80	12	960
TMI supporting structure	5.2	3	15.6
MOI stacks first stage	80	2	160
MOI stacks second stage	50	2	100
MOI supporting structure	3.6	1	3.6
TEI stacks	80	1	80
TEI supporting structure	0.7	1	0.7
TOTAL			**1386.6**

Table 3-4: Total mass to be launched into LEO

Note that the mass presented above is not the mass prior to the departure due to the losses due to the boil-off.

3.2 Configuration

3.2.1 Requirements and design drivers

- The Transfer Vehicle (TV) is composed of the Transfer Habitation Module (THM) and the Propulsion Module (PM). The PM consists of three stages: Transfer Mars Injection stage (TMI), Mars Orbit Insertion stage (MOI) and Transfer Earth Injection stage (TEI).
- All main components of the TV shall fit inside the fairing of the Energia launcher, a cylinder with a diameter of 6 m and a length of 35 m.
- Interfaces between assembly elements shall be kept to a minimum to simplify the assembly.
- THM shall be as independent as possible from the PM.
- THM shall provide an interface with the Earth Return Capsule (ERC) and Mars Excursion Vehicle (MEV) allowing crew to move from one to the other.
- Required pressurised volume of the THM should be 450 m^3.
- THM shall provide an airlock for Extra Vehicular Activities.
- Each propulsion module shall be safely discardable after its usage.

3.2.2 Assumptions and trade-offs

For the configuration of the THM, three options were considered:

- One-module configuration (Figure 3-1).
- Parallel configuration (Figure 3-2).
- Cross configuration (Figure 3-3).

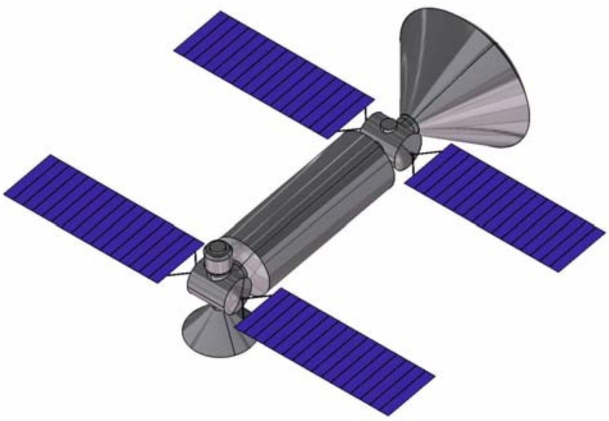


Figure 3-1: One-module configuration

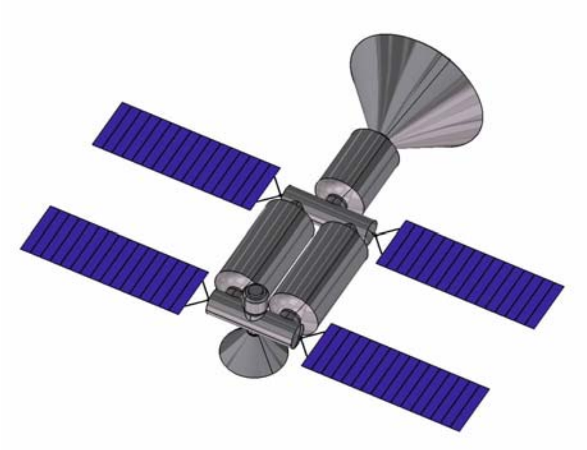


Figure 3-2: Parallel configuration

Figure 3-3: Cross configuration

The one-module configuration (option 1) was chosen in the trade-off shown in Table 3-5:

	Option 1	Option 2	Option 3
Habitable volume	+	+	+
Redundancy, isolation of depressurised part	+	+	+
Passable I/F with MEV, ERC and Airlock	+	+	+
Connection to Propulsion Module	+	+	+
Structural simplicity	+	-	-
Minimum number of interfaces (mass, leakage risk)	+	-	-
Mass radiation and debris shielding	+	-	-
Minimum number of launches	+	-	-
Privacy crew	-	+	+
Internal layout simplicity	-	+	+

Table 3-5: One model configuration (option 1)

The parallel and cross configuration (option 2 and 3) were rejected because of their structural complexity and mass impact.

3.2.3 Baseline design

Figure 3-4 shows the whole vehicle. The overall dimensions are shown in Figure 3-5 and Figure 3-6.

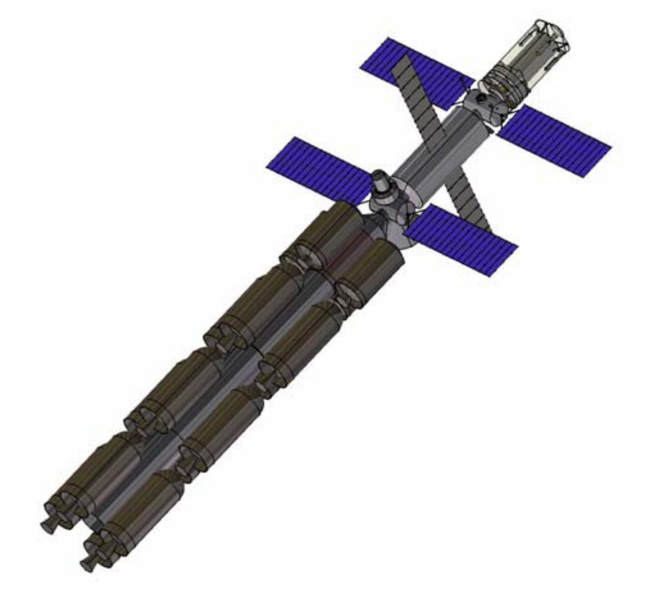


Figure 3-4: Complete Vehicle

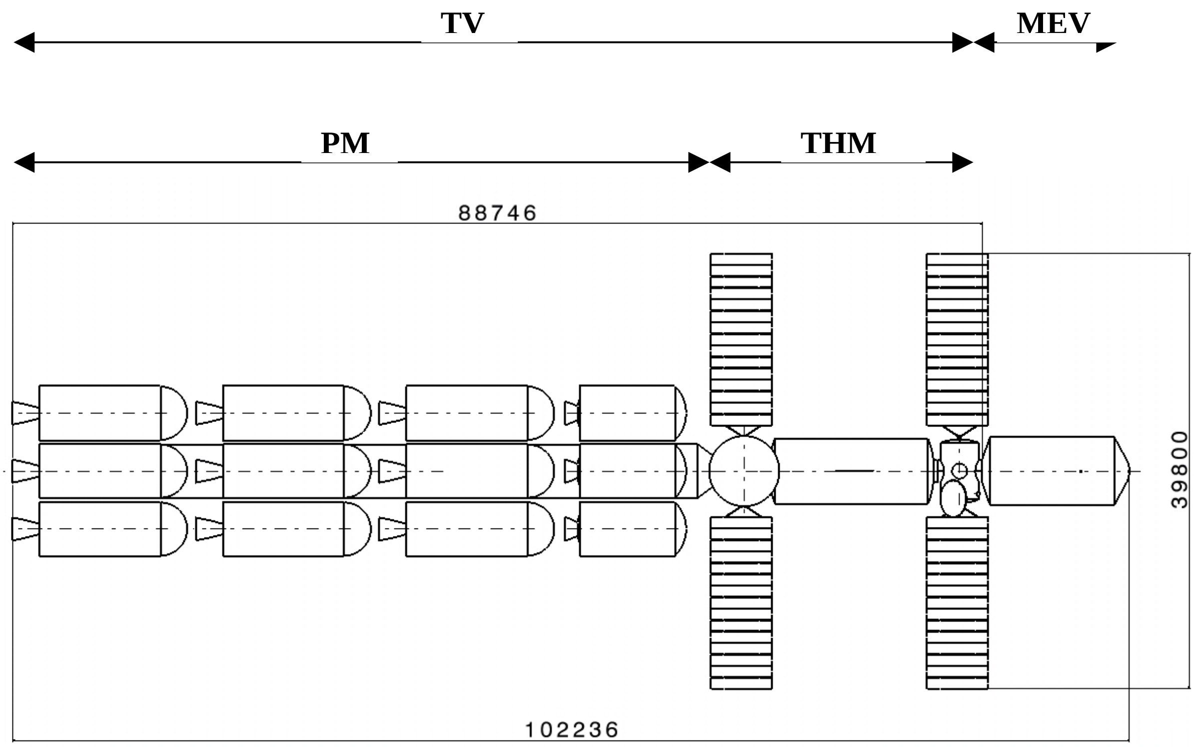


Figure 3-5: Global dimensions complete vehicle top view (dimensions in mm)

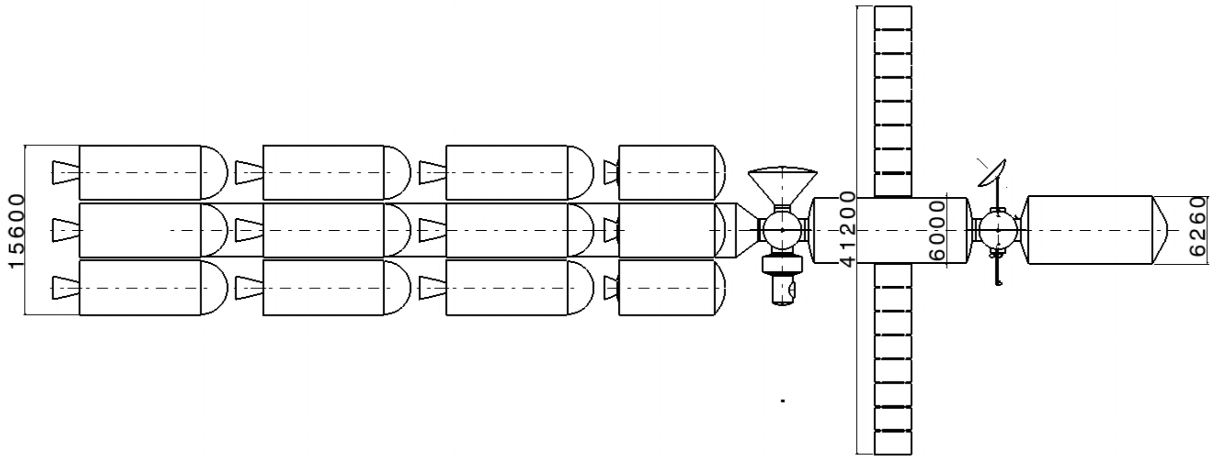


Figure 3-6: Global dimensions complete vehicle side view (dimensions in mm)

3.2.3.1 Transfer Habitation Module

The THM is the pressurised part of the TV and consists of a main cylinder and two nodes: a back and a front node. Each of these three parts can be sealed hermetically in case of depressurisation. If the main cylinder depressurises, the crew has to be evacuated to the front or back node for a couple of days until the leakage has been repaired.

Table 3-6 the dimensions of the main cylinder and the nodes. The pressurised volume fulfils the requird 450 m^3 .

	Length [m]	Internal Diameter [m]	Volume [m^3]
Main Cylinder	14	5.8	369.9
Back Node	5.2	3.4	47.2
Front Node	5.2	3.4	47.2
Total Pressurised Volume			464.3

Table 3-6: Dimensions of main cylinder and nodes

3.2.3.1.1 Main cylinder

This is the main part of the THM in which the crew will stay during the cruise to and from Mars. In a dedicated paragraph the interior of this main cylinder is presented. Two large deployable radiators are attached to the main cylinder (see Figure 3-7).

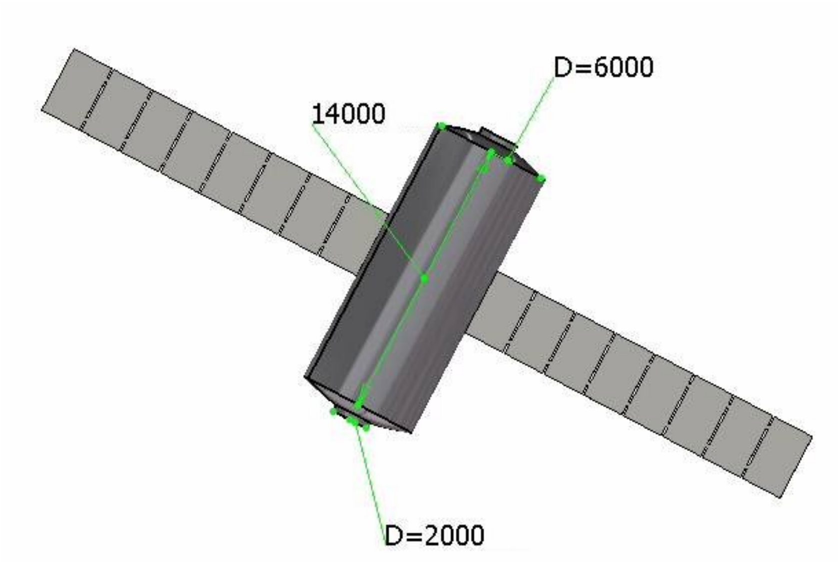


Figure 3-7: Main cylinder (dimensions in mm)

3.2.3.1.2 Back and front nodes

These nodes are the interfaces between the main parts: PM, THM, ERC, EVA Airlock and MEV. Two solar arrays are connected to the left- and right-hand side of both nodes. The power batteries are accommodated inside the nodes, but if they leak they will not bring danger to the astronauts. The main cylinder is positioned between both nodes.

The PM, the ERC and the airlock are attached to the back node (see Figure 3-8). The MEV and the cupola are attached to the front node (see Figure 3-9). The cupola allows the crew to have a 2.5D view outside and monitor Extra Vehicular Activities. Also all the communication antennas and equipment is attached to the front node. There is a spare docking point at the front node. The diameter of each node is 3.5 m and the length is 5.2 m.

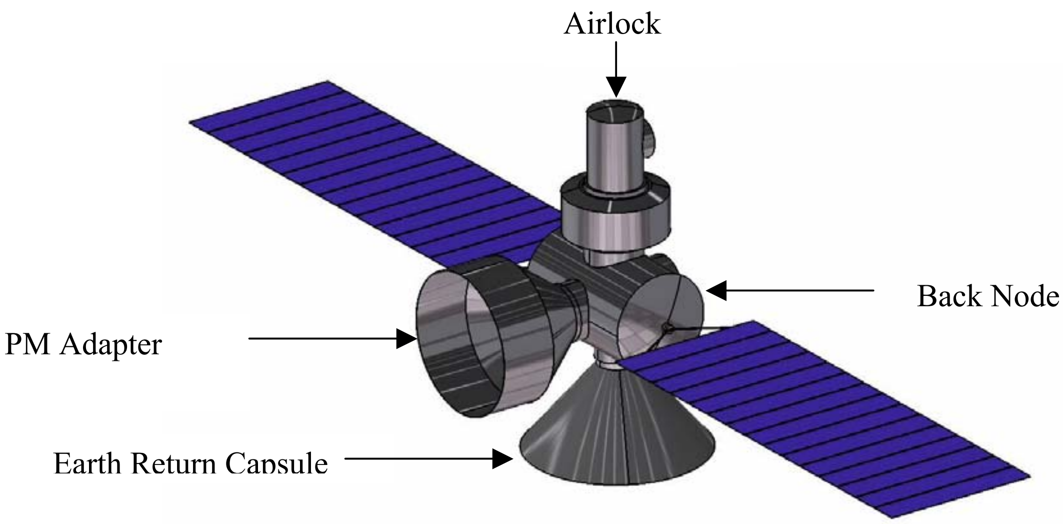


Figure 3-8: Back Node

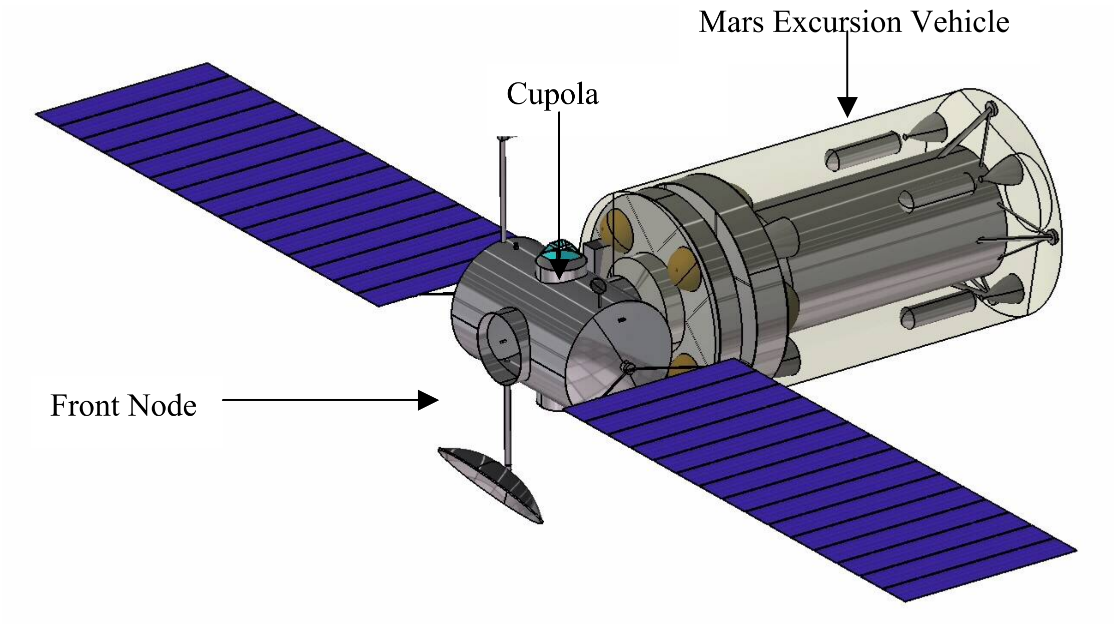


Figure 3-9: Front Node

3.2.3.1.3 EVA systems

The astronauts' ability to move around and conduct useful tasks outside the pressurised volume is a required capability. A conceptual airlock configuration was designed taking the ISS airlock as example. The part directly connected to the back node is a facility for EVA suit maintenance and consumables servicing.

3.2.3.1.4 Earth return capsule

The design of the ERC is not included in the current study. The ERC shown in Figure 3-9 is just an artistic impression of an ERC design.

3.2.3.2 Propulsion module

Figure 3-10 shows the complete Propulsion Module. A modular system has been proposed for the PM. Separate propulsion systems, which are jettisoned after its usage, are designed for the three main propulsive manoeuvres:

- Trans Mars Injection (TMI)
- Mars Orbit Insertion (MOI)
- Trans Earth Injection (TEI)

A central cylinder of 5 m diameter acts as the backbone (Figure 3-11) of the PM. All the stacks are attached to this support structure (a stack is an autonomous propulsion system with an engine and propellant tanks). The large diameter of the backbone is driven by the need to have sufficient space between four stacks in one plane.

Figure 3-10: Complete propulsion module

Figure 3-11: Backbone Structure

Figure 3-12 shows all the propulsion modules and Figure 3-13 shows the global dimensions. Within the TMI and the MOI, a staging approach is followed to increase the efficiency of the system. The TEI has only one stage with one stack and is placed inside the backbone structure of the MOI.

The TMI has three serial stages; each stage composed of four stacks. The MOI has two parallel stages, here each stage consists of two stacks.

Trans Mars Injection
(three stages, twelve stacks)

Mars Orbit Insertion
(two stages, four stacks)

Trans Earth Injection
(one stage, one stack)

Figure 3-12: All three propulsion systems

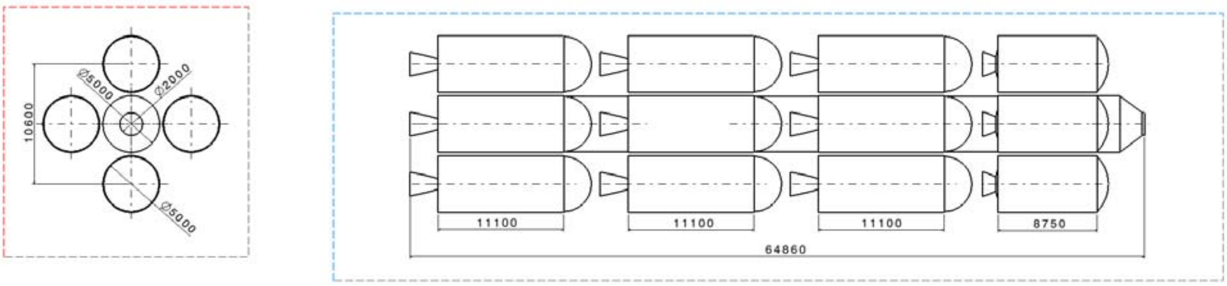


Figure 3-13: Global dimensions of the propulsion module

The front of the backbone structure is through a conical adapter structure attached to the back node of the THM. Inside this structure, five oxygen tanks and one nitrogen tank are accommodated (gasses are part of consumables), each with a diameter of 1.3 m (see Figure 3-14).

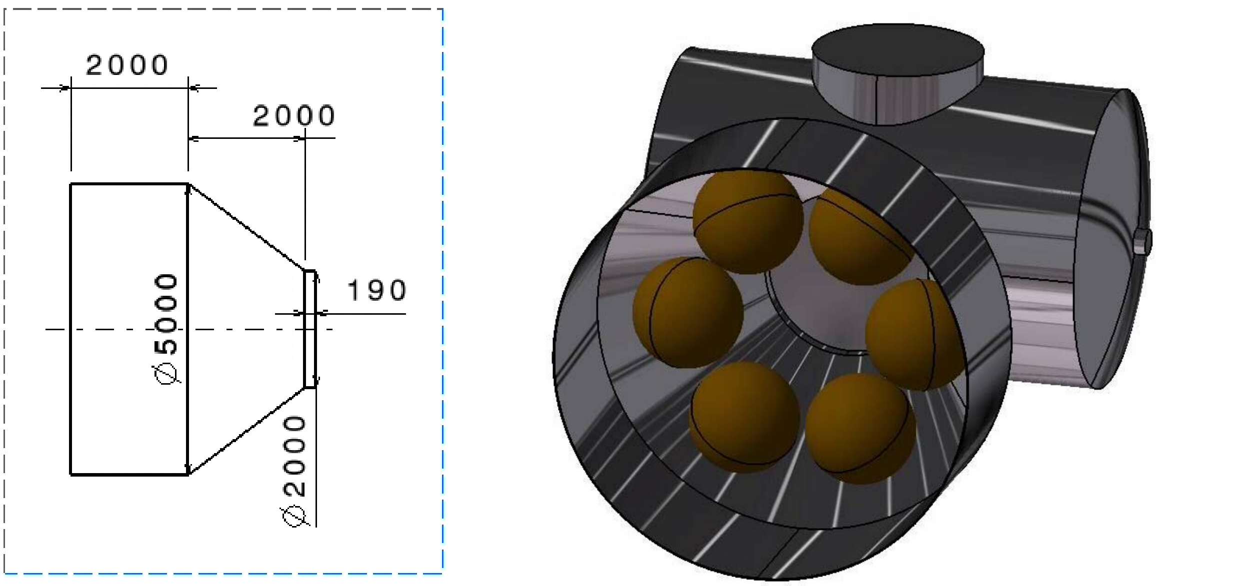


Figure 3-14: Conical adapter with tanks (consumables)

3.3 Transfer Habitation Module

3.3.1 Internal configuration

3.3.1.1 Requirements

The crew consists of six crew members, of which three are supposed to land on the surface of Mars for a 30-day surface stay. The transfer duration from Earth to Mars will be approximately 200 days and there will be a stay in the Martian orbit of about 550 days.

When designing a human mission, basic required volumes have to be integrated. Based on the Man System Integration Standards, the NASA Standards 3000 (STD) and the paper ”Habitability as a Tier One Criterion in Advanced Space Vehicle Design: Part One—Habitability” by Constance Adams (paper no.: 1999-01-2137, AIAA), functional and volume requirements for a habitat module were established:

A summary of these requirements is shown in Figure 3-15 to Figure 3-18. The internal configuration features three zones:

1. Crew quarters belong to the private zone in a spacecraft.
2. The personal zone in a spacecraft is defined by functions such as the command, the laboratory or typically the exercise facilities where the crew trains/works mostly on their own. The medical and hygiene facilities also belong to the same category.
3. The third zone is the social or communal zone.

Storage space or racks can be found in any zone though preferably not in the crew quarters so that this place stays calm and quiet and free from noisy equipment shifts.

Crew quarters

- privacy
- separation
- sleep 215x85x85 / 1.55m³
- pers. workstation 205x101x90 / 1.86m³
- personal stowage 0.63m³

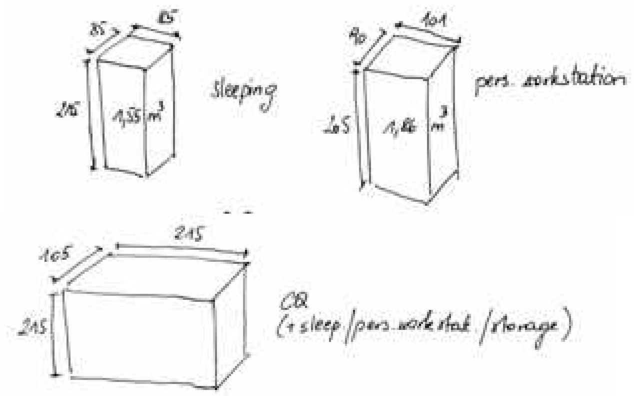


Figure 3-15: Recommendations for volumes for different areas of the private zone

Command	- 3 workstations	
Laboratory	- greenhouse	
Exercise	- centrifuge	Ø 375, 215h
	- ergometer	101-150x150x245
	- treadmill	

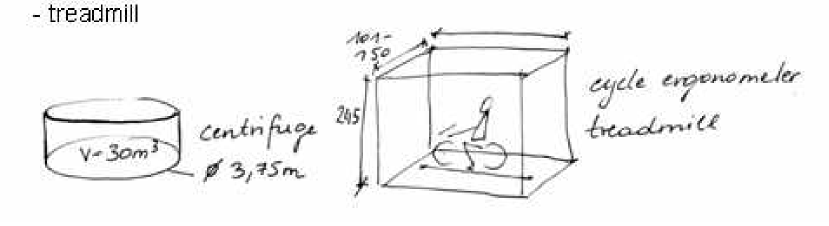
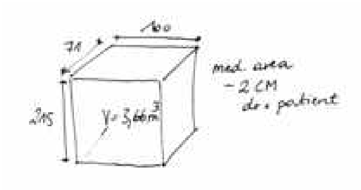
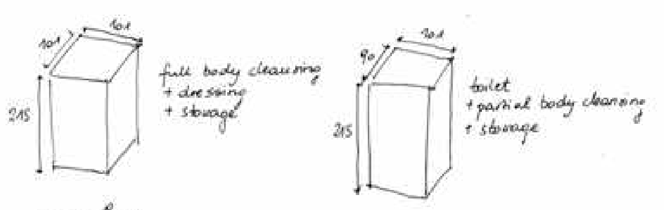


Figure 3-16: Recommendations for volumes for different areas of the personal zone

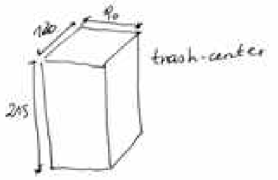
Medical	- med. Equipment	
	- private exams	71x100x215 / 3.66m³



Sanitary	- toilette	215x90x101
	- full body cleansing	215x101x101 / 2.19m³



Waste



Storage	- Food	60+85transl.+60
	- Water	
	- Propulsion	
ERV, AOCS, MEV, AL		

Figure 3-17: Recommendations for volumes for different areas of the personal zone

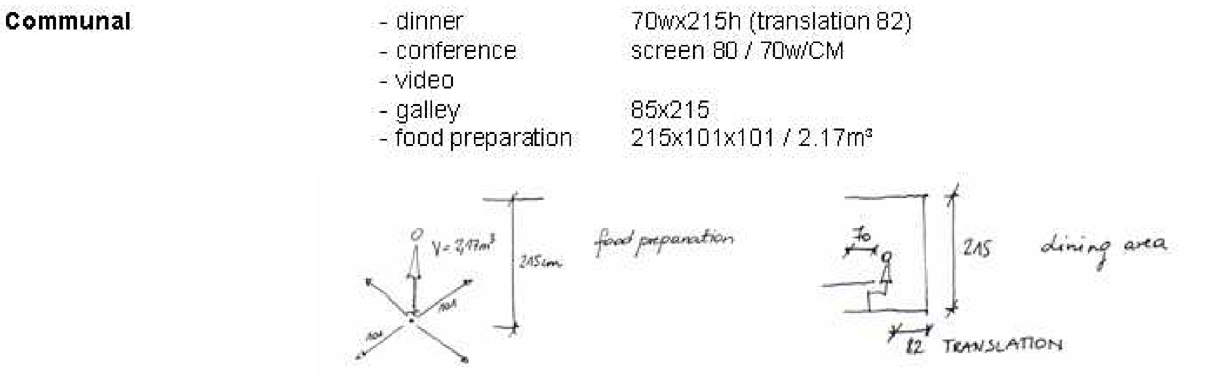


Figure 3-18: Recommendations for volumes for different areas of the social zone

To complement the requirements and numbers shown previously a set of recommendations were considered from different sources. The general architecture is shown in Table 3-7:

	Factors to be considered	**Impacts**	**Reference**
GENERAL			
Anthropometric design and layout	Neutral body posture changes the geometry of the eye`s reference point Different muscular effort in zero-g and partial-g	Different design for workstations, clothing and equipment in zero-g and partial-g	[RD92], p 147
Socialization	Support social cohesion Reduction of interpersonal tension Groups vs. Privacy of small groups	Areas designed for group interaction (dining, wardroom, entertainment area, group work sites)	[RD93]
Privacy	Individual crew activities such as sleeping, reading, personal communications	Separation of private crew quarters from: • Public view • Sounds • Vibrations	[RD93]
COMPONENTS			
Laundry	Mass reduction of textiles Psychological issue Self-cleaning	Water requirements	
Radiation shielding	Galactic Cosmic Rays (GCR)		[RD95]
	Solar Particle Event (SPE)	Storm shelter must be included in the architecture	

	Factors to be considered	Impacts	Reference
Windows	Provide the ability to observe outside the habitat	Cupola: observation of exterior of module, EVA and robotic operations Windows in private crew quarters with automatic shutters	[RD96] cosmonaut rec. [RD97]
Exercise	Fitness Micro-g Countermeasures	Dedicated areas	[RD96]
Personal Hygiene	Whole body cleansing in privacy Changing clothes	Areas for personal hygiene	[RD93]
Greenhouse	Fresh food Take care of live system	Dedicated areas	[RD98]

Table 3-7: General architecture

Table 3-9 shows factors that become especially relevant in studies dedicated to human long-term missions such as a human mission to Mars. Themes such as adjacency and separation derive from the spatial position of the functions in relation to each other inside the spacecraft. Table 3-8 shows the internal configuration:

	Factors to be considered	Impacts	Source
ADJACENCY and SEPARATION			
Simultaneous crew activities to be located far enough apart	Reduce the change of accidental interference		[RD92],p151
Easy access to potential trouble spots (leak points, motors, valves, controllers)	Easy repair and adjustments		[RD96]
Emergency routes at every stage of the habitat			[RD92],p151
Protection from electronic magnetic interference	Communications and computation equipment may be interfered	Physical separation of power cables from computer systems and data cables Layout of raceways	[RD96]
Mechanical systems (motors, pumps, LSS, waste management, toilets) far away from crew quarters	Sound, vibration and smell control		[RD96]
Separating waste management from food preparation and	Hygienic and aesthetic reasons		[RD96]

	Factors to be considered	Impacts	Source
dining			
Toilet far away from crew quarters	Noisiest item during sleep periods		[RD92],p151
Multiple volumes	Provide compartiments in case of: Fire Contamination Loss of pressure	At least two, separate, isolatable pressurised volumes within the habitat core to allow egress from one volume to the other in contingency situations.	[RD96]
FLEXIBILITY OF USE			
Crew autonomy			
Flexible design of workstations	Access to control systems from more points in any module	Elimination of dedicated workstations with their displays and controls Portable computers More autonomous working	[RD92],p151

Table 3-8: Architecture configuration

For long duration stays, communication with "home" becomes one of the major issues. Diverse studies – space mission reports or studies of humans living in extreme environments on Earth – have shown that good and efficient communication within the interior configuration is crucial for mission success in long duration missions. Table 3-9 shows recommendations for communication:

	Factors to be considered	Impacts	Reference
Training	Mission very long Psychological factor Training for 0g and xg	Provision of: Virtual-reality-training Training Software during Travel	[RD92],p143
Two-way communication	Link with home (family, children, friends)	Confidential and direct and simultaneous Video-conference, messaging Alternative communication through voice, auditive, tactile, sensory	[RD92], p165 Interview Valery Polyakov (Modern Times Spezial, ORF)
Responsibility and authority for crew	Less ground support through increasing distance		[RD92], p173

Table 3-9: Recommendations (communication)

The texture and materiality of interior spaces are considered to be a stimulating factor for a crew's health and productivity. Noise reduction must be addressed in the design for future space habitats so that noise levels become reduced to a level of comfort. Currently noise levels on ISS are as high as on a four-lane highway (about 60 dB or even more) Table 3-10 shows recommendations for lighting, colour and sounds (about 27a).

	Factors to be considered	Impacts	Reference
Visual stimulation	Psychological well-being	Through colour, lighting, sounds	[RD92], p145
	Orientation	Up and down perception	
	Biorhythm	Create day and night/winter and summer	[RD99]
	Different atmosphere	Sunny day, cloudy day, party mood	[RD99]
	Noise reduction to a comfortable level especially in crew quarters		[RD92], p155

Table 3-10: Architecture (light, colour and sound)

3.3.1.2 Design drivers

Initial interior design drivers depend on the decision if the THM is a rigid cylindrical module, or an arrangement of spheres, or an inflatable, or a hybrid construction of inflatable and hard shell modules.

The approach of the study was to organise and design a core rigid cylindrical module where all the needed hardware, parts and functional spaces needed for habitability of the required 450m^3 should be incorporated. Moreover, how should the module be organised regarding interior space and efficient use of volume mass efficiency, habitability, psychology, physiology and spatial architectural issues?

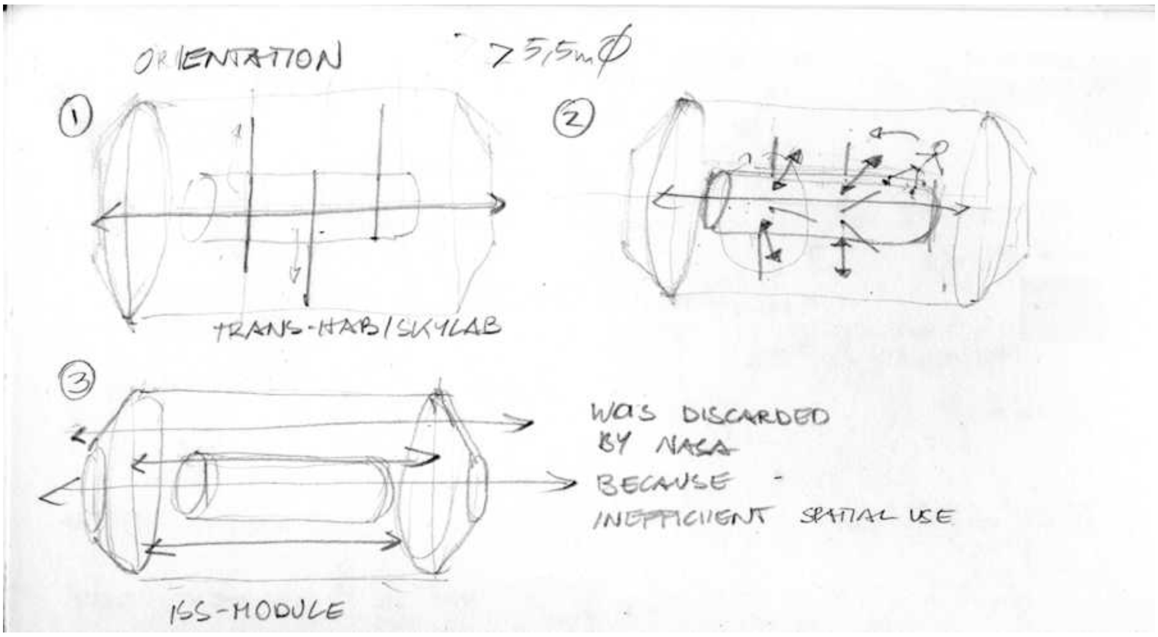


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Figure 3-19: Levels

Figure 3-19 shows habitat layout with different levels (dark lines) as from NASA's Skylab and the new Transhab. The second part shows an inner cylindrical core with a habitable surface of the outer surface of the inner cylindrical core. The third also consists of an inner core for all technical infrastructure but orientates the habitable space along the axis of the hatches and connectors towards the other modules.

Figure summarizes the possibilities for orientation. Long missions require more complex spaces are required to keep the crew healthy and productive, so the best option seems to be the mixed one (right diagram) because it implies a great variety in the space perception. This option also requires a distinct orientation so that orientation is easy and not confusing when floating through the modules.

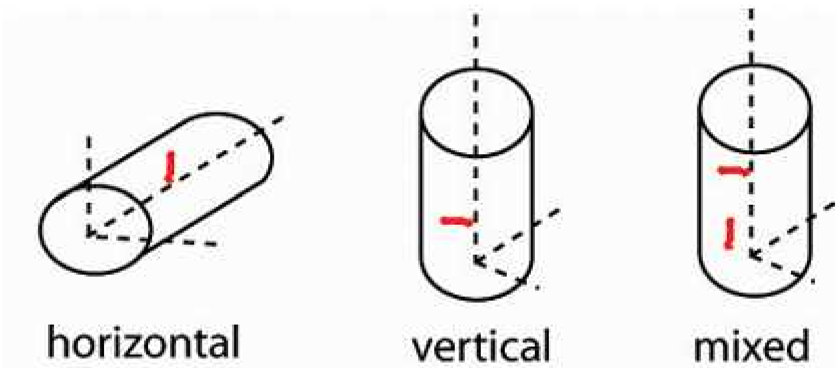

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Figure 3-20: Orientation options

Finally a one-module fail-safe with hard shell and a mixture of Skylab- and ISS-orientation was chosen.

3.3.1.3 Other design drivers

The following elements also have an impact on the design:

- 1/5 of the volume has to be dedicated to ducts and pipes
- Easy access to all ducts and pipes for maintenance is required, a well designed system and structure supports easy maintenance
- If reasons all systems (LSS, AOCS, etc.) should be modular, e.g. made out of plug-and-play parts, so that in case a part fails in one place the astronaut can put it into another place
- There shall be avoidance provisions for the following:
- Toxicity
- Fire/explosion
- Contamination
- Other biological hazards
- Enough fire detectors and isolation and recovery systems should be provided to enhance the safety of the crew

3.3.1.4 Interior configuration

The development of the baseline design for the THM included an extensive research and analysis of built and operated space stations. Before the final baseline design four main steps including nine different configuration options were developed. These were followed by a detailed design for different interior layouts for the three main zones: the private, the personal and the social zone.

Figure 3-21 shows the selected showing the selected baseline design for the THM:

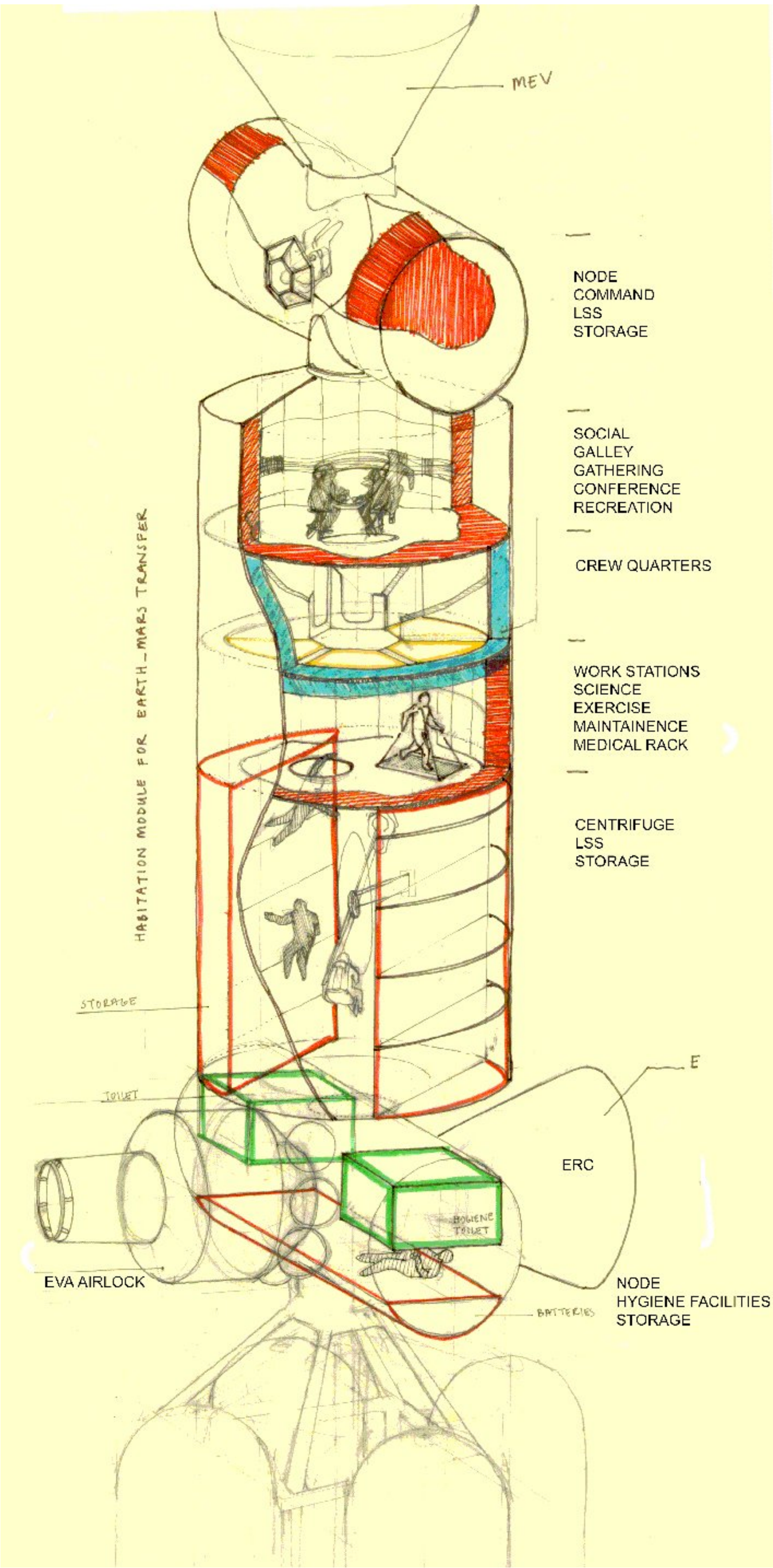


Figure 3-21: Axonometric view of the final baseline design

The configuration consists of a fail-safe core module with a length of 14.00 m and 5.90 m inner diameter (outer diameter is 6.00 m). At the extremities of the core module two nodes are connected, these nodes allow astronauts to interface with additional habitable modules, namely:

- The EVA airlock and the ERC (Earth Return Vehicle) connected onto the back node (the "lower" part of the figure)
- The MEV (Mars Excursion Vehicle) connected at the upper end.

The nodes have a length of 5.20 m, with a diameter of 3.40 m (slightly smaller than ISS modules, which are 4.00 m in diameter).

The total habitable volume has a minimum of 450 m^3; where 1/3 of the volume is used for storage, and the remaining 2/3 are the habitable volume. About 5% of the total volume has to be considered for the module structure.

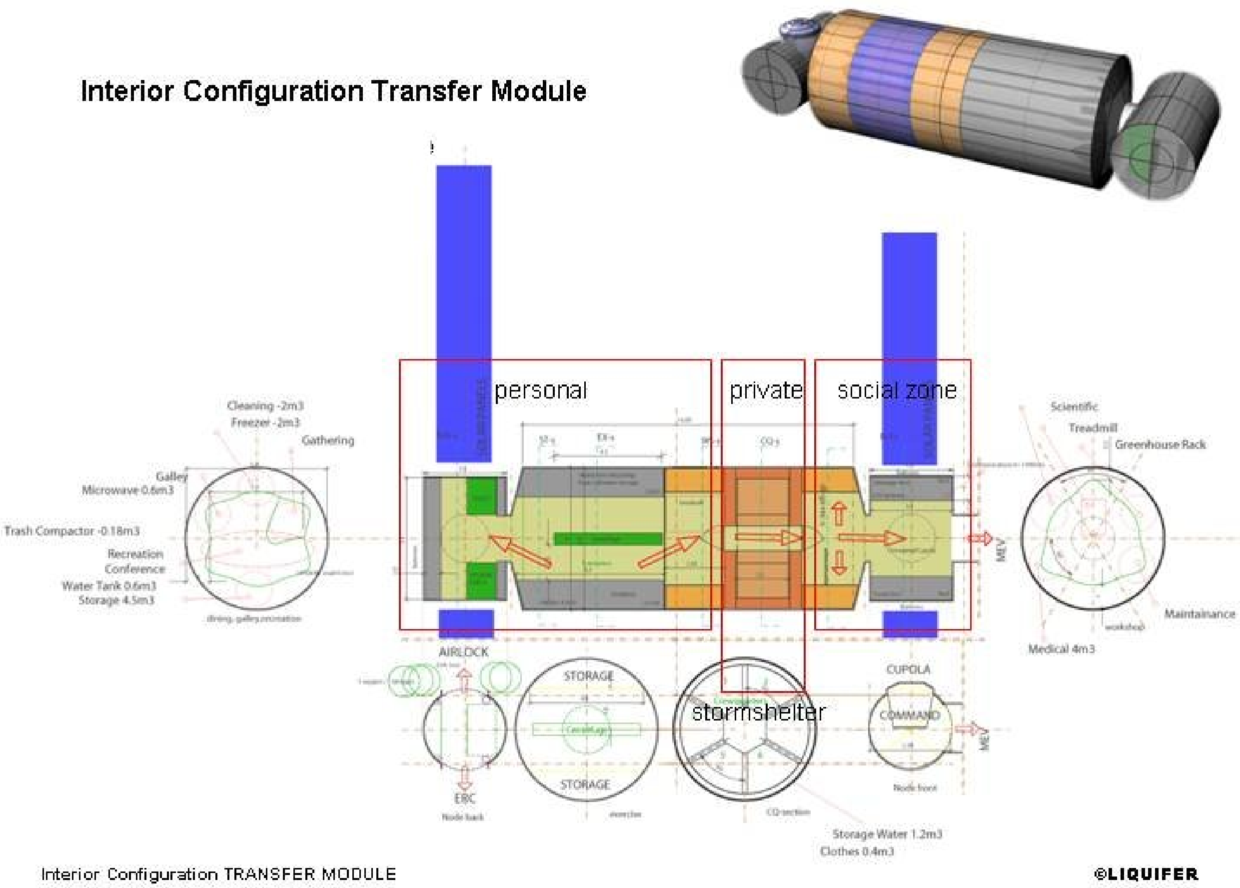


Figure 3-22: Baseline design overview - drawing

Figure 3-22 shows the division into the three basic habitation zones. Also the main translation movement is visible through the red arrows.

The overall compound of the module and the two nodes can be divided into the three basic habitation zones, as shown in of Figure 3-23. The main translation movement is visible through the red arrows.

Two main aspects have been taken into consideration with regard to safety:
The overall configuration is based on a fail-safe main module. Only the stormshelter has airtight hatches. The main cylinder is already partitioned into three zones, which can be used independently in case of emergency. Furthermore the use of the MEV and the ERC nodes provides additional safe compartments.
Special precautions have to be taken with respect to fire, toxic contamination etc. It is therefore assumed that a fire detection system, a fail-safe isolation and recovery system is implemented. Batteries should be positioned outside, adjacent to the solar panels, and the oxygen/nitrogen tanks should be also positioned outside the pressurised volume.

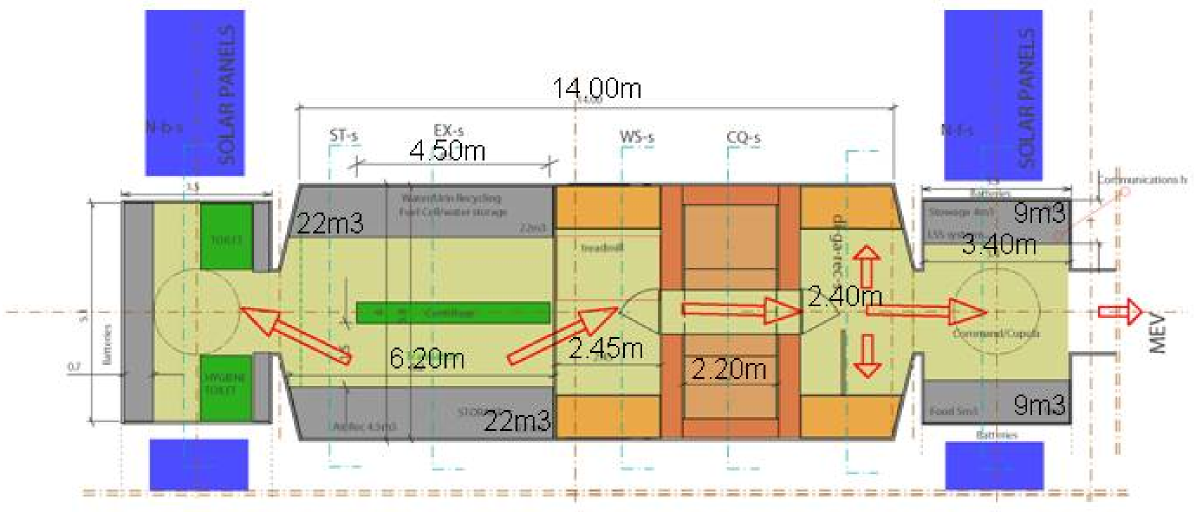


Figure 3-23: Baseline design overview – drawing of measurements

The back node is mainly a translation space to the ERC and the airlock where three space suits are stored. This habitable space is considered as more isolated than other spaces therefore the hygiene facilities are located in the personal zone. There are two hygiene facilities, one for back-up and the other one for daily use. The space towards the propulsion is used as storage space although all AOCS systems, batteries, and the propulsion tanks are positioned outside to prevent the inhabited space from being polluted by dangerous fluids or gases. The tanks are positioned around the EVA hatch for easy access in case of an emergency. The ISS-type orientation allows a maximum use of the space with these dimensions of the node. Additionally the complex hygiene facilities used on the ISS today can easily be improved without applying the latest technology on the basic volume and hardware on the existing structure. See Figure 3-24 :

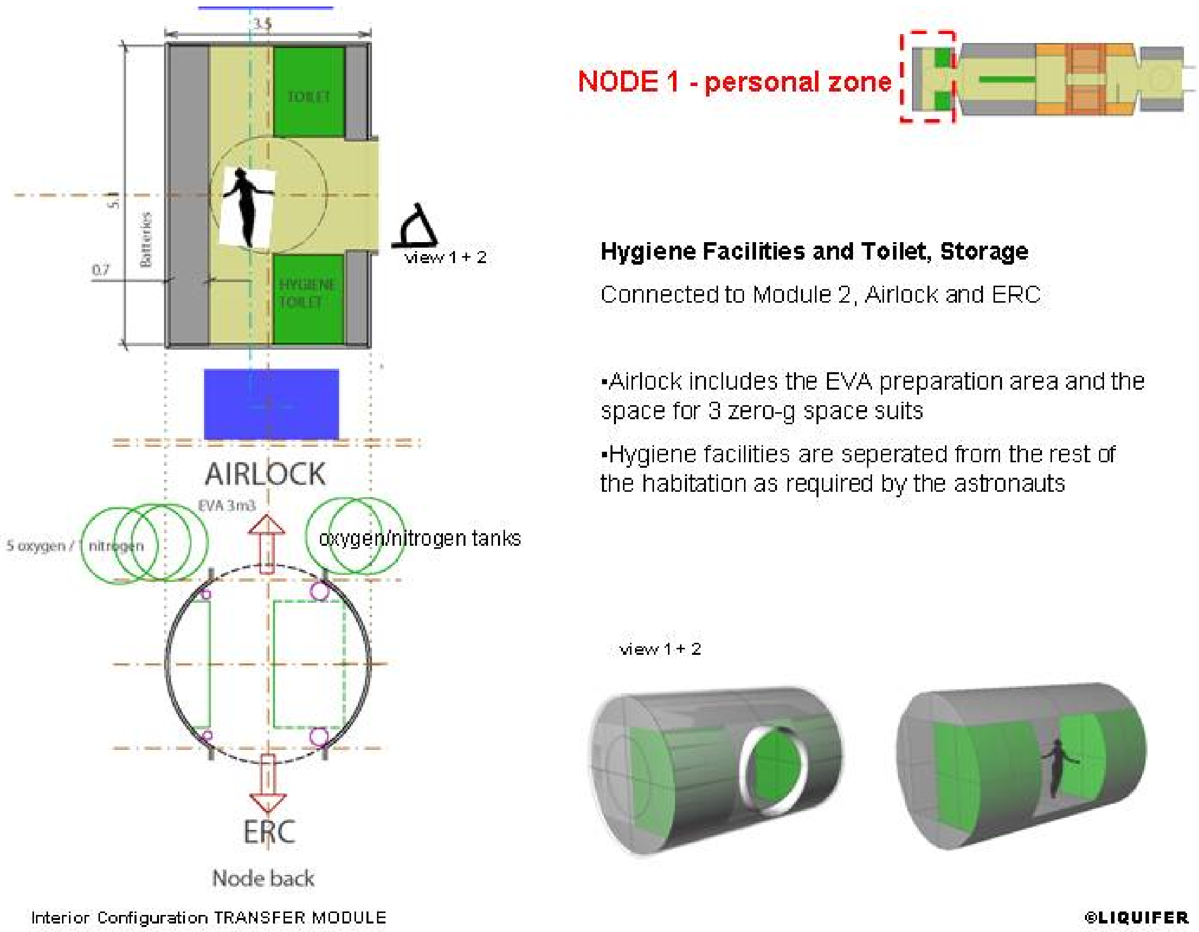


Figure 3-24: Baseline design back node – drawing

Figure 3-25 the exercise area with the centrifuge and how the astronauts can move (translate) to the other areas:

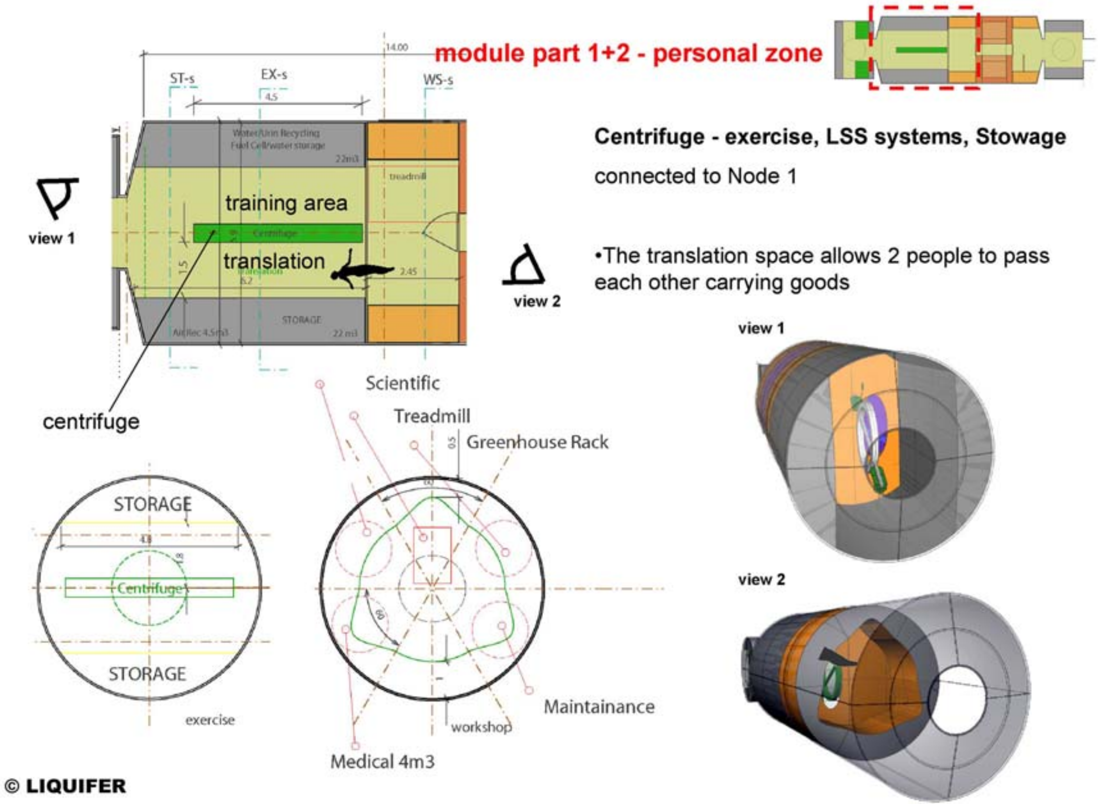


Figure 3-25: Baseline design personal area - drawing

The large exercise facility with the centrifuge (personal zone) is located in the back end of the main cylinder. It is spinning around the circumferential axis of the spacecraft. The other space is used for translation of the crew and for the life support system, which requires approximately 55 m^3. This space has an unusual orientation, which is neither Skylab-type nor ISS-type due to the spinning centrifuge. It is used by building the LSS into this part because of its easy access for maintenance. Furthermore it has a large volume, which could be used by the astronauts in their leisure time for some zero-g experiments, as on Skylab. The next adjacent compartment is the workstation and maintenance level (personal zone), which has a Skylab-type orientation due to the dimensions of the module. For efficient use of space this looks like the most adequate orientation.

In this area one treadmill, a medical, a science and a greenhouse rack and three workstations with computers are located. In addition each astronaut is equipped with a laptop computer.

Figure 3-26 shows the next compartment, which has light hatches, houses the crew quarters (private zone). It has a 50-cm thick protection wall all around the compartment, stuffed with consumables and water for radiation protection in case of a solar particle event. This storm-shelter protects the crew for up to two days from such a solar storm. Inside are six crew quarters equally distributed with a translation path wide enough for large packets to be passed through.

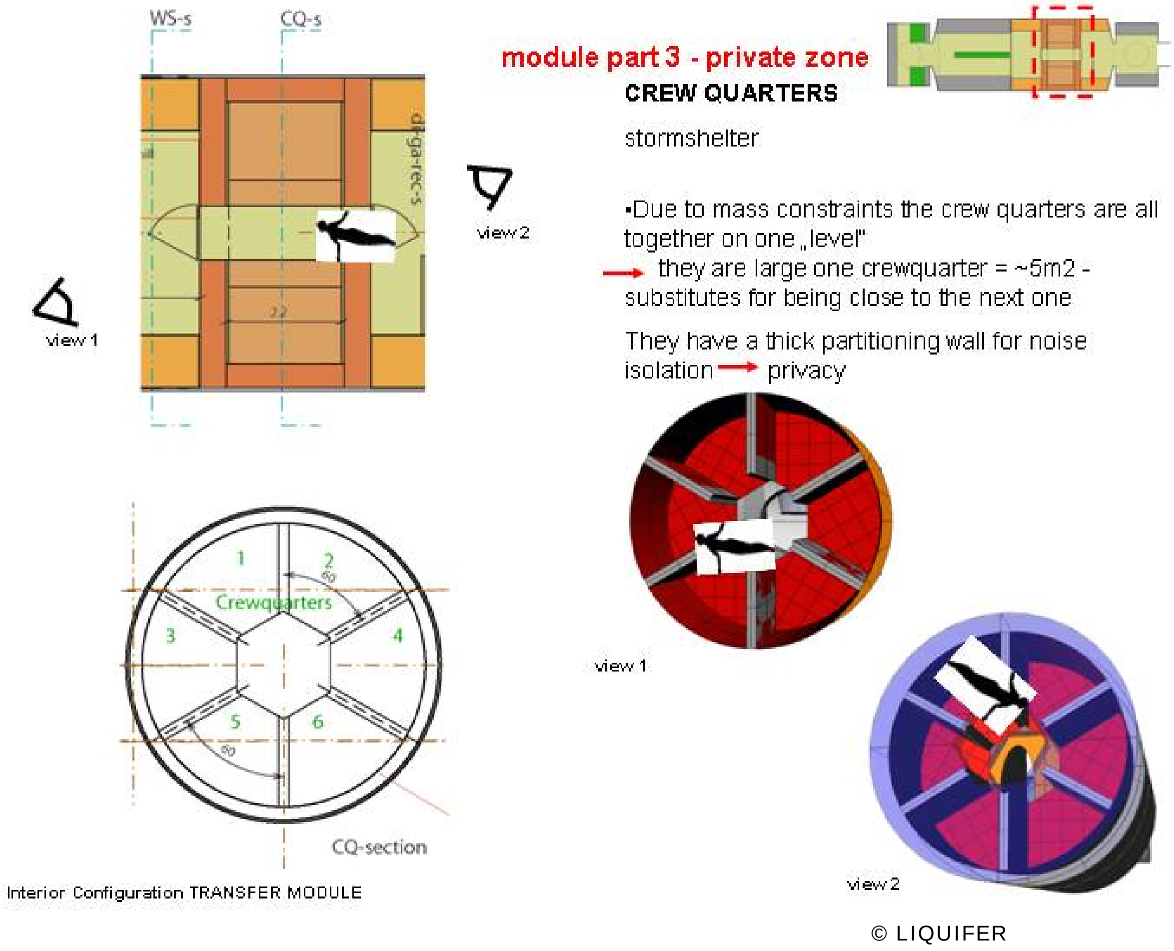


Figure 3-26: Baseline design private zone – drawing

The "top" part of the main module is occupied by the social zone which consists of the galley, the gathering area, the conference infrastructure and a recreational space. There are two windows to look outside.

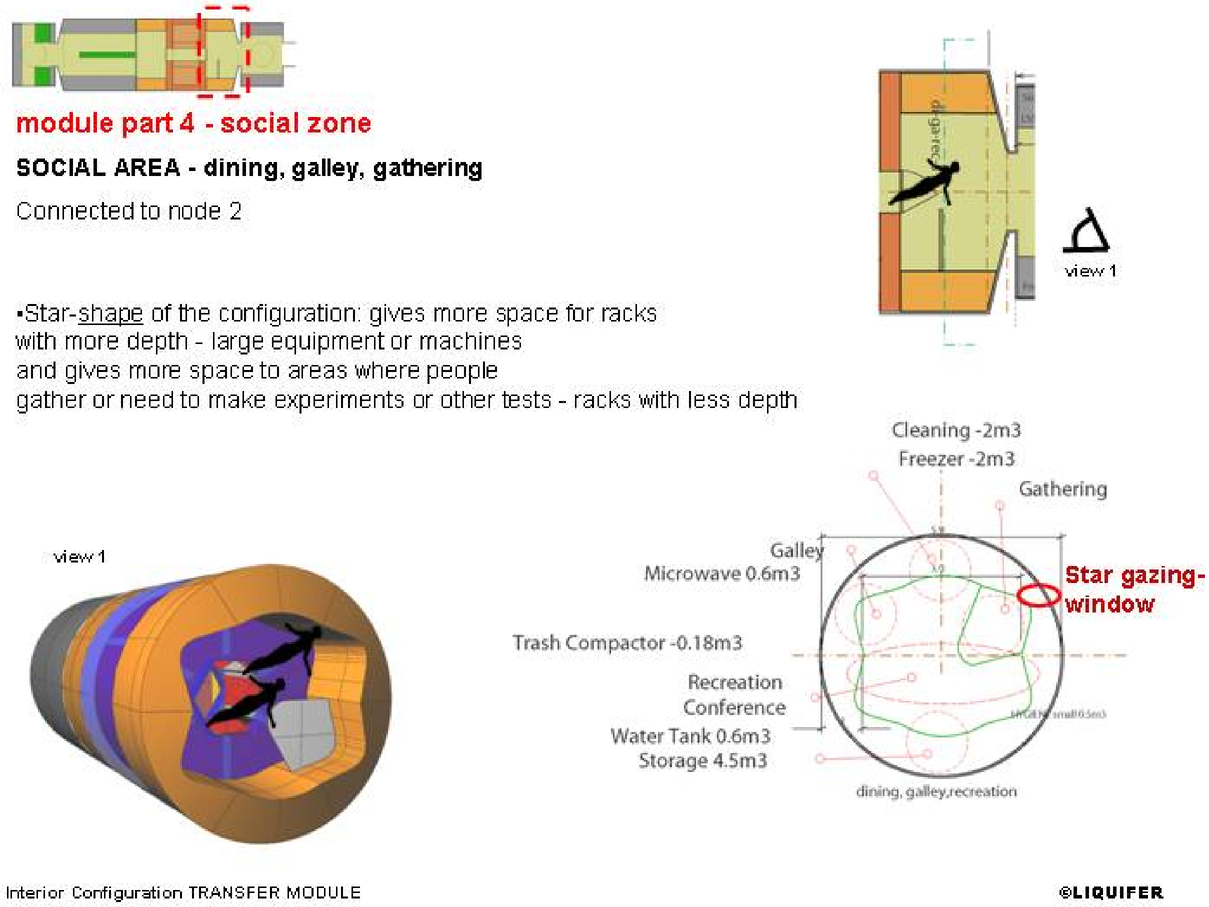


Figure 3-27: Baseline design social zone - drawing

The front node has the same dimensions as the back node and it houses the command part with the cupola for a good spacecraft overview and the translation path to the MEV (Mars Excursion Vehicle). The cupola is directed into the same direction as the airlock so during an EVA, the remaining crew can view fellow crew on the space walk. Additionally there are racks for storage.

The orientation is adjusted according to the functionals (see Figure 3-28 bottom right): there is an extra command level put in between to connect the cupola with the main command to distinguish the space "below" from the one above. In the "upper" command area six seats have to be installed for use during the spacecraft acceleration when taking the course towards Mars. Below there is the main translation path, when the astronauts go to the MEV or return from the MAV after the surface stay.

Storage is an issue for long space permanence, as already identified from previous MIR missions. The overall storage space included in this THM design is 37.6 m^3 (excluding LSS, and consumables) with the possibility of adding 8 to 10 m^3 for additional storage.

The storage space in the overall configuration is distributed as follows:

NODE 1...................= 5.1m^3

Module part 1+2.......= 23.5 m^3

Module part 3...........= 0 m^3 (all consumables or personal stowage crew)
Module part 4………= 0 m^3 (all dedicated equipment for housekeeping, cooking etc.)
NODE 2.................... = 9.0 m^3

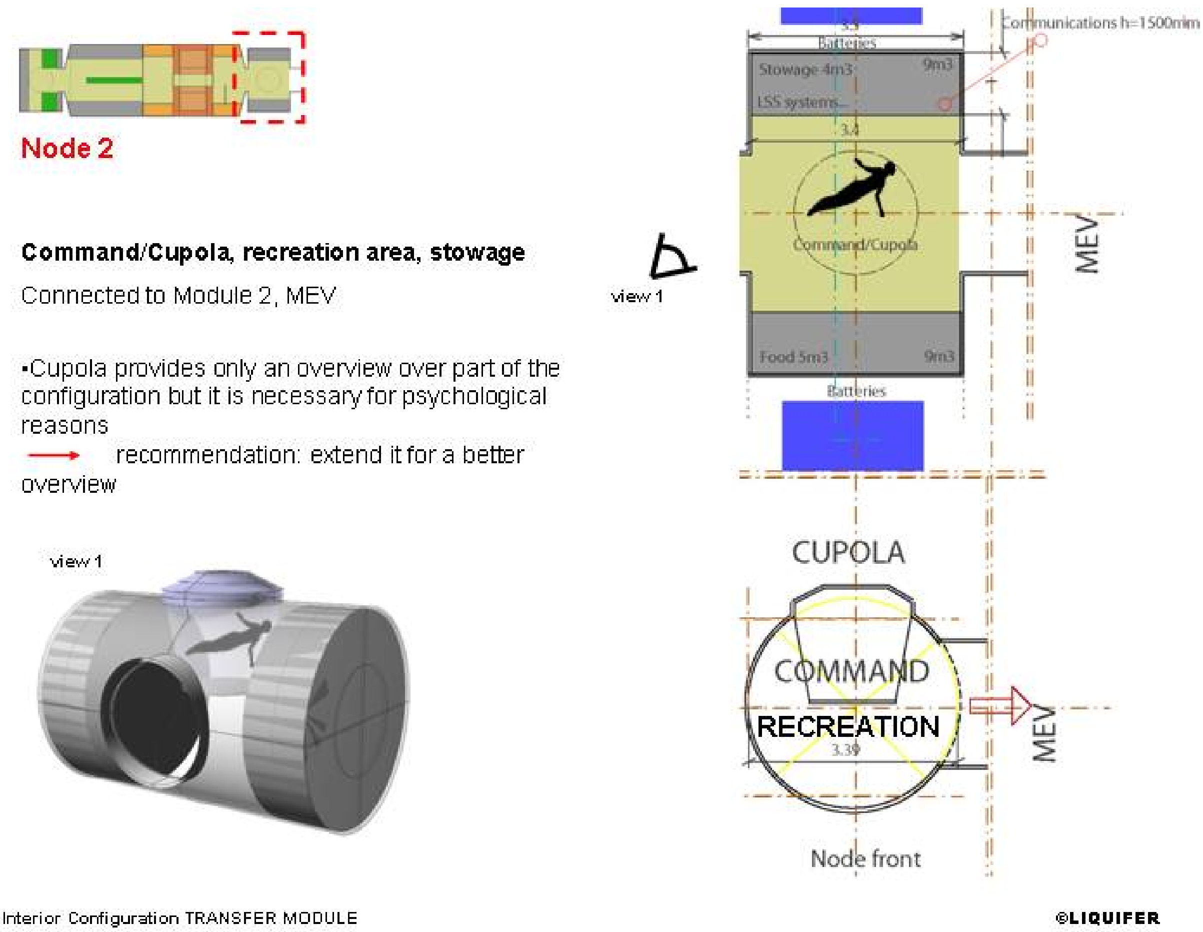


Figure 3-28: Baseline design front node - drawing

Figure 3-29 shows, a complete overview:

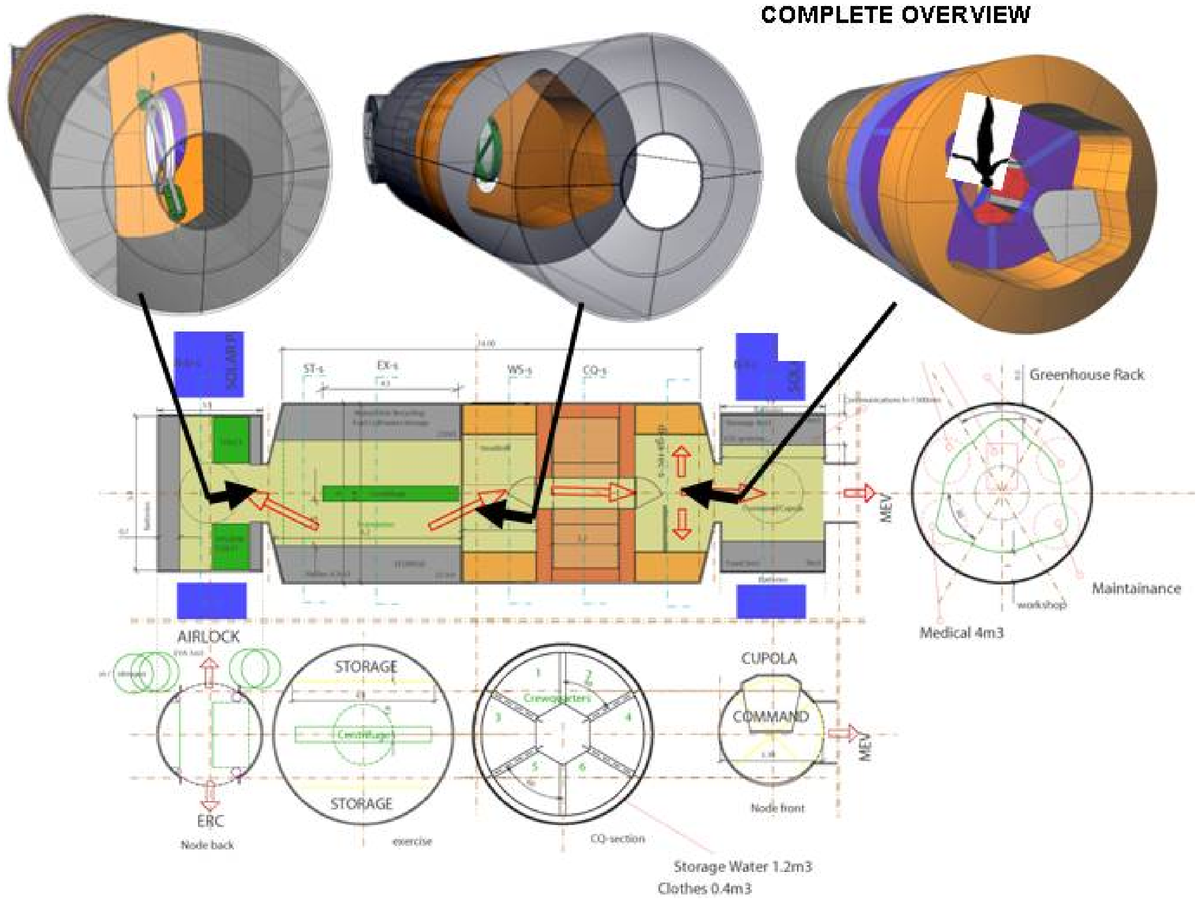


Figure 3-29: Baseline design complete overview

3.3.2 Environmental control and life support system

The life support system comprises the following subsystems:

- Atmosphere Supply and Control
- Atmosphere Revitalization
- Temperature and Humidity Control
- Water Management
- Waste Management
- Food Management
- Safety
- EVA Provisions
- Hygiene
- Crew Accommodations

3.3.2.1 Requirements and design drivers

The metabolic needs of the crew have been calculated using the correlations given in ESA standard PSS-03-406 and crosschecked with relevant sources. The entire calculations have been based on the energy expenditure of the crew. The schedule for crew activity is shown in Table 3-11. EVA has not been considered because it shall be a kept to a minimum during the interplanetary mission phase and would negatively influence the energy estimate by overestimating the needs.

Activity (hours)	Astronaut					
	1	2	3	4	5	6
Sleep	8	8	8	8	8	8
Pre- and post sleep	5	5	5	5	5	5
Leisure activities	0	0	0	0	0	0
Personal hygiene	1	1	1	1	1	1
Eating	2	2	2	2	2	2
Exercise	1.3	1.3	1.3	1.3	1.3	1.3
Station keeping	4.7	4.7	4.7	4.7	4.7	4.7
Laboratory activities	2	2	2	2	2	2
EVA mission tasks	0	0	0	0	0	0
EMU donning/doffing	0	0	0	0	0	0
Egress/ingress	0	0	0	0	0	0
Pre-EVA setup & post EVA EMU care	0	0	0	0	0	0
TOTAL TIME (24hrs)	24	24	24	24	24	24

Table 3-11: Crew Activity Schedule during Earth-Mars transits and in orbit around Mars

Based on the energy expenditure, the metabolic needs and products by the crew have been estimated. They are shown in Table 3-12:

	Per day	Per mission
Total metabolic needs		
Energy consumption (W*h)	18 053	17 385 078
Energy consumption (MJ)	65.0	62 587
Oxygen consumption (m^3)	3	3120
Oxygen consumption (kg)	5	4458
Drinking water (m^3)	0	16
Drinking water (kg)	17	16424
Dry food (kg)	4	3894
Total waste production		
Faeces (kg)	0	428
Carbon dioxide production (m^3)	3	2570
Metabolic water production (kg)	2	1842
Urine production (kg)	9	8902
Faecal liquids (kg)	0	462
Insensible water (kg)	9	8902
Additional informtion		
Hygiene water (kg)	24	23112
Nitrogen (kg)		91

Table 3-12: Metabolic needs and products of the crew

3.3.2.2 Assumptions and trade-offs

The data shown in this table refer to an open loop control. It suggests a mass of consumables of more than 47 tonnes. The life support system would have an approximate power requirement of 6.3kW and a volume of about 136m^3. Taking into account that consumables need additional hardware for storage and use, as well as the need to treat and store the metabolic products, the use of an open loop system seems prohibitive. As a reference, the main parameters for an open loop life support system are shown in Table 3-13:

CONSUMABLES TO BE LAUNCHED (kg)	
OXYGEN	4466.9
NITROGEN	91.0
POTABLE WATER	16457.9
HYGIENE WATER	23160.0
DRY FOOD	3899.4
PACKAGING	1392.3
INORGANIC MATERIAL EXCLUDING PACKAGING	3177.9
TOTAL	52645.4
WASTE PRODUCTION DURING MISSION (kg)	
WASTE GASES	6129.2
WASTE WATER	39535.8
SOLID ORGANIC WASTE	1006.2
SOLID INORGANIC WASTE EXCLUDING PACKAGING	3177.9
PACKAGING	1392.3
TOTAL	51241.4
ROUGH ESTIMATE ECLSS MASS (kg)	
TOTAL	31162.0

Table 3-13: Open Loop Life Support System Mass, Consumables and Waste production

Generally, two classes of regenerative life support systems are considered:

- Physico-chemical regeneration
- Bio-regenerative systems

These systems may be used to lower the cost for a human mission to Mars by reducing the mass of the consumables and perhaps life support system hardware. Some regenerative systems have been successfully flown on MIR and the International Space Station. The result of any trade-off would have to be measured on the criteria that have been selected. The criteria of equivalent system mass does not seem appropriate for such missions. The lack of incorporating the reliability of the system as well as the dynamic efficiency might deliver a less optimal system.

In this study the sole criteria was to reduce system mass, and providing a sufficient level of redundancy. Less critical items were not increased in their fault tolerance whereas for critical systems, which would cause a catastrophic failure, a two-fault tolerant system was implemented in the model. Furthermore, this study projects about 20 years into the future. This causes a significant uncertainty in the performance and parameters of the subsystems. Therefore, if possible, the parameters have been selected using the best guess approach or data obtained on similar systems with lower readiness level. Some systems have not been optimised for spaceflight applications and some mass savings could be expected in the future. Each item

considered in this study has been selected from a database created on purpose. This database contains a large number of units relating to life support systems with its known parameters. However, some units have been added as 'Generic' or 'ALS', which indicates that these systems are generic good guesses or advanced life support items that have not been space-qualified yet.
After arriving at the design of the life support system, the appropriate number of units has been selected and their duty cycle has been adjusted so that their performance matches the requirements on the life support system.

3.3.2.2.1 Hygiene water

During the course of this study it has been shown that the hygiene water consumption has a significant impact on the overall system mass. The study suggests assuming a daily hygiene water provision of 4 l/crew/day. This would account for:

- Flushing water (0.3 l/ day)
- Dish washing (2.4 l/day), amounting to 12.0 l /day for such purpose

- Personal hygiene water (1.6 l /day), The crew is assumed to take showers once a month, which makes the available water for this purpose around 30 litres and the daily allowance about 0.6 l/day
- Losses (0.1 l/day)

No water has been considered for washing cloths as this is assumed to be performed chemically without using water.

3.3.2.2.2 Drinking water

The water release by the crew has been calculated using standard correlations based on the energy expenditure. A literature review revealed that the water intake by the crew is to some extent equal to the water release by the crew. Therefore, the amount of water intake has been calculated using the numbers for the sensible and insensible water quantities released by the crew. The advantage of this method is that the potable water estimate is based on the energy expenditure, similar to all other crew metabolic needs.

3.3.2.2.3 Cabin atmosphere

The cabin atmosphere has been calculated as follows:

Total Cabin Pressure: 101.3 kPa
Partial Pressure Oxygen: 21.3 kPa
Partial Pressure Nitrogen: 80.0 kPa
Partial Pressure Carbon Dioxide: <0.7 kPa (<3 kPa short term exposure)

The atmosphere has been selected based on the following:

- Atmosphere composition currently used on-board manned spacecrafts

- Easy ground reference and testing possible
- Long-term exposure to other atmospheres has not been studied thoroughly

Preferably, the atmosphere would be free of any contaminants. However, as a minimum requirement, the spacecraft atmosphere shall adhere to the requirements given in ESA PSS-03-401. Based on the experiences with long-term pressurised spacecrafts there shall be more stringent limits on microbial contamination. The following limit has been proposed during this study based on the recommendation by ESA internal experts:

Total microflora count: 200 CFU/m^3 (CFU stands for colony forming units)

3.3.2.2.4 Waste production

Besides the already presented production of faecal material by the crew, the crew will produce additional organic and inorganic waste. Organic waste will consist of hair, nail clippings, skin material, kitchen waste, food leftovers. The total amount of such *organic* waste has been estimated to be 0.1 kg/crew/day. To quantify the total amount of inorganic waste produced by the crew per day was not possible due to the lack of data. However, reviewing existing data and other sizing tools, the amount of *inorganic* waste produced by the crew per day was estimated to be around 0.6 kg/crew/day. This includes:

0.05 kg/d cleaning supplies
0.1 kg/d waste collection supplies
0.1 kg/d contingency collection mitten bags
0.1 kg/d hygiene supplies
0.2 kg/d wet wipes for house cleaning

3.3.2.2.5 Packaging

A significant fraction of the inorganic waste will come from the food packaging. Packaging of food has been investigated to establish a figure that could be used during the study. The work included a paper review and a small weighing exercise to verify the numbers.
The review revealed differences in the food packaging between the food provided by the Russian and the American organisations. While the Russians use many cans for packaging their foods, most of the thermostabilized entrees offered by the U.S. are packaged in retort pouches. The U.S. discontinued the use of space food in tubes in the early 1970s. Russia has used tubes continually, but is now beginning to phase them out.
Preservation methods for Russian food is comparable to that of Shuttle food but different materials and packages are used. Preservation methods consist mostly of dehydration, thermostabilization, and intermediate moisture. Packaging includes metal tubes, cans, and plastic overwrapped in foil.
Russian cans are made of steel, require a can opener, and come in two sizes: large (101.6 mm in diameter x 38.1 mm high) and small (73.025 mm in diameter x 31.75 mm high). In addition to steel cans, the Russians use a plastic packaging material for dehydrated and intermediate moisture foods. They do not have sufficient barrier properties for extended shelf life, so the

plastic packaging material is overwrapped with a foil material to extend the shelf life (NASA Food Technology Commercial Space Centre).
Packaging on the U.S. side (considered lighter because of the use of plastics rather than cans) can be considered 220 g/(day*crew) for Shuttle missions and slightly higher for ISS due to overwrap films and the additional thermostabilized pouches.
For comparison, food with relatively low humidity content was weighed and the ratio between packaging and food was established. Test articles were dry ice cream, dried roasted seaweed, mashed potato powder, instant soup and dried fungi. The average packaging to food ratio was 0.34($kg_{packaging}/kg_{food}$) for 'meal size' ratios.
Based on the dry food demand of the crew of aboout 0.674 kg/(day*crew) and the data obtained from NASA, the current packaging to food ratio is about 0.33($kg_{packaging}/kg_{food}$) and for ISS slightly higher.
Based on this outcome, the study considered 270 $g_{packaging}$/(day*crew) based on 0.674 kg dry food per day and crew and a packing ratio of 0.4). Although this seems high, for a long duration mission some food will probably be produced in-situ and needs additional packaging as well as the more stringent demands on the food preservation.

3.3.2.2.6 EVA considerations

Based on the uncertainties regarding the conditions of crew as well as the environment during trans-planetary flight, the frequency of EVA needs to be kept to a minimum. However, provisions for emergency EVAs need to be foreseen to be able to recover to more favorable spacecraft performance. Therefore the capability of performing EVAs during transit has been implemented.

3.3.2.2.7 Contingency supply

Using a regenerative system it would be sufficient to launch the initial filling of the systems and the make-up supply depending on the recycling efficiencies. However, it is necessary to provide the crew with contingency supply if the life support system is failing. In an ideal case the contingency supply would enable the crew to safely return to Earth. Apart from that, as a result of the abort analysis it was discovered that no extra contingency is required in the ECLSS system to cope with these scenarios:
Emergency supply of oxygen: 36 days
Emergency supply of potable water: 10 days
Emergency supply of hygiene water: 5 days
Emergency supply of food: 10 days

Currently, the assumption is that the supply would be sufficient for the crew to overcome the contingency situation or to determine an alternative consumables supply strategy. It is clear that these figures may significantly change but they give a reasonable starting point for further discussions. In addition, this contingency supply shall be accessible from the radiation shelter as the supply for an eventual stay inside.

3.3.2.2.8 Food production unit (Greenhouse)

A greenhouse has not been introduced at this stage of the study. The study team is aware of the positive impact of the greenhouse on the different functions of a LSS, the nutrition of the crew and on the crew psychology. On the basis of preliminary knowledge, a greenhouse would provide::

- A high degree of closure of carbon dioxide to oxygen conversion
- A high degree of water loop closure
- A high percentage of fresh food to the crew

However, it would cause an increase in structural mass and power requirements that perhaps offset the gain of mass on the LSS. Therefore, at this stage of the study and the incomplete trade-off between LSS with greenhouse and LSS without greenhouse, the greenhouse was not considered.

3.3.2.3 Waste management strategy

The long duration of this mission inherently involves the production of substantial amounts of both organic and inorganic solid waste. This amount adds up to several tonness and has to be taken into account and dealt with in an efficient way. Several options were considered for waste management. It was decided to jettison the generated solid waste using existing airlocks to minimise the mass penalties of the mission. However, some treatment and storage is still necessary before jettisoning. This treatment depends on the nature of the waste and can be described as follows.

Waste treatment strategy

The handling of the inorganic waste (i.e. cleaning supplies, hygiene supplies, waste collection supplies, etc) is somewhat easier than for organic waste as it does not require because much treatment. Care must be taken in correctly classifying the nature of the waste. Supplies classified as inorganic upon launch become organic waste upon use by the crew.

The first step in the management of the inorganic waste would be to compact it to reduce its volume. Mass reduction is possible by reducing the waste reusable solid, gaseous and liquid compounds. After compaction, decontamination and bioresistant storage would be sufficient to have the inorganic waste safely stored before jettisoning.

The management of the organic waste (i.e. used tissues, faecal material, hair and skin material, nail clippings, food leftovers, etc) is more complicated and it requires waste stabilization. Three main technologies could be used for this purpose: chemical stabilization, sterilization or lyophilization (commonly referred to as freeze drying). Mass and power considerations, and assessing their technology readiness level, lyophilization looks most promising for the reduction of the organic waste.

Lyophilization is the process of removing water from a product by sublimation and desorption. It is performed in lyophilization equipment which consists of a drying chamber with temperature controlled shelves, a condenser to trap water removed from the product, a cooling system to supply refrigerant to the shelves and condenser, and a vacuum system to reduce the pressure in the chamber and condenser to facilitate the drying process. Lyophilizers come in a wide variety of sizes and configurations and can be equipped with options that allow system controls to range from fully manual to completely automated. Lyophilization cycles consist of three phases:

Freezing, primary drying, and secondary drying. Conditions in the dryer are varied through the cycle to ensure that the resulting product has the desired physical and chemical properties, and that the required stability is achieved. This freeze drying technique allows recovery of water, which could be fed to the water recycling system, while inactivating and compacting the remaining solid waste. This treated waste needs to be stored in airtight bags to avoid its humidification and recontamination to be then jettisoned together with the compacted and treated inorganic waste according to the strategy explained hereafter. Figure 3-30 summarizes' the waste management strategy selected for this mission:

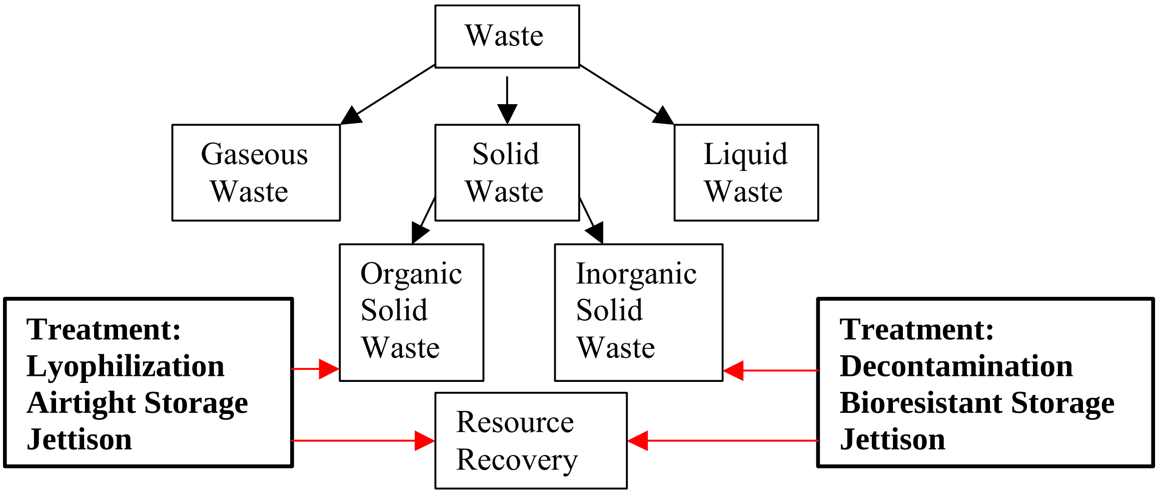


Figure 3-30: Waste strategy

The MELiSSA cycle could provide an alternative way of treating organic waste. However, its maturity is currently not advanced enough to be considered in this study.

3.3.2.3.1 Jettisoning strategy

For the proposed waste management strategy to have an optimal beneficial impact on the mission performances, the stored waste should be jettisoned at strategic mission phases; this is, before any major Δv manoeuvres to reduce the mass that needs to be accelerated or decelerated. Having this in mind, two waste discarding operations are envisioned:

- The first jettisoning would occur prior to the Mars Orbit Insertion manoeuvres and it would discard around 1210 kg of solid waste stored during the 217 days of the transfer to Mars phase.
- The second discarding operation would be to transfer the waste stored during the 533 days orbiting around Mars to the MAV after it has docked with the THM and before it is undocked.
- The waste accumulated in this phase is calculated to be about 2890 kg, which after treatment and compaction can be stored in the MAV's cabin.
- Finally, the 1170 kg of waste stored during the 210 days of transfer to Earth can remain in the THM and would be discarded with it.

Compliant with the Planetary Protection rules, it is necessary to ensure that none of the jettisoned waste units would reach the surface of Mars or Earth.

3.3.2.4 Baseline design

Given the need to provide a two-failure tolerant life support system and the dimensions of the spacecraft, the life support system has been designed in a modular approach concentrating the bulk of the life support system in particular modules similar to Node2 and Node3 of the ISS. The study suggests using two independent life support systems. The size of the vehicle, previous experience on ISS and safety considerations led to this conclusion.
In addition, an additional non-regenerative life support system has been added to the stormshelter to allow the crew full control of the life support system during their stay in the shelter. The life support system is a relatively simple open loop system based on the supply of consumables and the short-term storage of products.
The THM LSS does not take advantage of the MEV life support system features. A trade-off analysis revealed that no major mass benefit would be achieved if the THM LSS took advantage of the MEV LSS.

Figure 3-31 illustrates the design of the life support system with its major components. Only one of the redundant systems is shown in the blue box. The LSS inside the red box illustrates the stormshelter LSS.

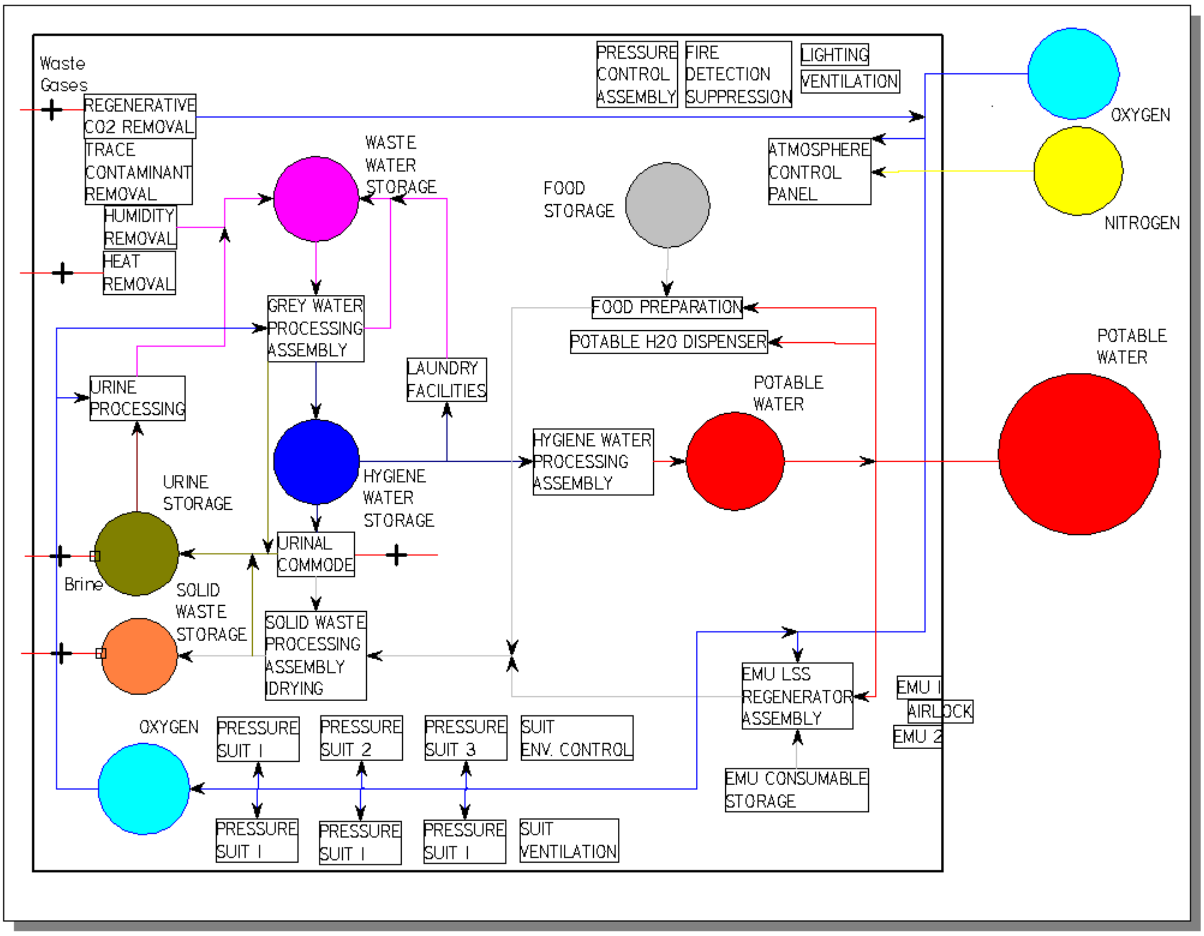


Figure 3-31: Mars THM LSS design

The anticipated LSS is a hybrid system of bio-regenerative and physico-chemical systems. Currently, a number of subsystems of both natures are under investigation by several organisations and companies, which could significantly improve the closure of the life support system. However, the level of maturity is rather low for some of those systems and do not allow implementation in the proposed THM LSS design without introducing a significant uncertainty to the system parameter. Other systems have reached maturity and a level of documentation where they could be used during this study.

This study produced two ECLS systems options. First, a LSS was designed that performs all necessary functions with *current* technologies (baseline). Second, a LSS was designed *estimating* available recycling efficiencies and implementing technology that is considered to be *available in 20 years*.

3.3.2.5 ECLSS option 1 – current technologies

The mass of the baseline life support system is 19.4 tonnes. This is significantly higher than the LSS mass presented by the NASA Mars Exploration Study Team and somewhat similar to the LSS mass shown in the ISTC 1172 study, which is also largely based on current systems. Compared to the open loop life support system, the launch mass is reduced by more than 42 tonnes.

The results of the option 1 investigation are shown in Table 3-14:

CONSUMABLES TO BE LAUNCHED (kg)	
OXYGEN	1364.0
POTABLE WATER	1387.2
HYGIENE WATER	6302.3
DRY FOOD	3906.5
PACKAGING	1358.0
INORGANIC MATERIAL EXCLUDING PACKAGING	3154.8
TOTAL	22207.7
WASTE PRODUCTION DURING MISSION (kg)	
WASTE GASES	371.9
WASTE WATER	7329.5
SOLID ORGANIC WASTE	998.9
SOLID INORGANIC WASTE EXCLUDING PACKAGING	3154.8
PACKAGING	1380.7
TOTAL	13235.7
ROUGH ESTIMATE ECLSS MASS (kg)	
TOTAL	19356.8

Table 3-14: Mass estimates for a mission using current technology

3.3.2.6 ECLSS option2 anticipated technologies

In this case, a LSS was designed estimating available recycling efficiencies and implementing technology that is considered to be available in 20 years. The anticipated recycling efficiencies are shown in Table 3-15.

Degree of Recycling (%)	
OXYGEN	95
POTABLE WATER	95
GREY WATER (condensate, used hygiene water)	95
YELLOW WATER (water in contact with urine)	95
BLACK WATER (water in contact with Faeces)	20
SOLID ORGANIC WASTE TO FOOD	20
SOLID INORGANIC WASTE	0
PACKAGING REUSE	0

Table 3-15: Anticipated degrees of recycling

The system mass has been estimated to be 13.4 tonnes. The savings on the LSS mass must be evaluated carefully due to the large uncertainty on the state of the development of certain subsystems. However, the impact on the consumables is significant as shown in Table 3-16.

CONSUMABLES TO BE LAUNCHED (kg)	
OXYGEN	394.2
NITROGEN	91.0
POTABLE WATER	1008.8
HYGIENE WATER	323.6
DRY FOOD	3830.9
PACKAGING	1392.3
INORGANIC MATERIAL ECXLUDING PACKAGING	3177.9
TOTAL	10218.6
WASTE PRODUCTION DURING MISSION (kg)	
WASTE GASES	306.5
WASTE WATER	1000.8
SOLID ORGANIC WASTE	804.9
SOLID INORGANIC WASTE EXCLUDING PACKAGING	3177.9
PACKAGING	1392.3
TOTAL	6682.4
ROUGH ESTIMATE ECLSS MASS (kg)	
TOTAL	13627.6

Table 3-16: Mass estimates for a mission using anticipated technology

Due to the anticipated high-recycling efficiencies the amount of needed consumables is reduced to 10.3 tonnes. In addition, the amount of produced waste is reduced to 6.7 tonnes making alternative options to jettison more likely.
The life support system has been estimated to have an approximate mass of 13.4 tonnes. The detailed mass budget is shown in Table 3-17.

Equipment	Number of units	Mass per unit (kg)
ALS Airlock Air Save Pump Package	2.00	70.30
ALS ARSD CDRA (ARES DEMONSTRATOR)	3.00	202.00
ALS CONVENTIONAL OVEN	2.00	50.00
ALS COOKING/EATING SUPPLY	2.00	5.00
ALS HAND/MOUNTH WASH FAUCET	2.00	8.00
ALS METOX CO2 removal (canister)	3.00	14.52
ALS METOX CO2 removal (regenerator)	1.00	47.63
ALS SHOWER	2.00	75.00

Equipment	Number of units	Mass per unit (kg)
ALS VACUUM	3.00	30.00
GENERIC ACCUMULATOR HYGIENE WATER (300L)	4.00	100.00
GENERIC ACCUMULATOR POTABLE WATER (300L)	2.00	100.00
GENERIC ACCUMULATOR URINE STORAGE TANK (57L)	1.00	30.84
GENERIC INORGANIC STORAGE(200kg)	16.00	67.00
GENERIC ORGANIC STORAGE(200kg)	20.00	80.00
ISS AAA - avionics air assembly	10.00	12.40
ISS ACCUMULATOR WASTE WATER TANK (46L)	1.00	67.59
ISS CCAA	2.00	112.00
ISS comode/urinal	3.00	50.00
ISS condensate storage	1.00	21.20
ISS EMU (Shuttle)	7.00	135.00
ALS FREEZER	1.00	400.00
ISS fuel cell water storage	4.00	21.20
ISS HEPA - BACTERIAL FILTER	40.00	5.20
ISS IMV - intermodule ventilation fan assembly	20.00	4.76
ISS IMV - intermodule ventilation valve	4.00	5.10
ISS laundry (washer/dryer)	2.00	118.00
ISS OXYGEN STORAGE TANK	5.00	109.00
ISS OXYGEN TANK PRESSURISATION SYSTEM	2.00	102.00
ISS PCWQM - process control water quality monitoring	2.00	38.00
ISS PFE - portable fire extinguisher	5.00	15.10
ISS Sample Delivery System	4.00	2.70
ISS smoke detector	10.00	1.50
ISS TCCS - trace contaminant control system	2.00	78.20
ISS trash compactor	2.00	27.00
ISS URINE PROCESSOR ASSEMBLY (VCD)	2.00	245.00
ALS BWRS1 (316l vessel)	3.00	332.80
Manual Pressure Equalization Valve (MPEV)	4.00	1.20
microwave oven	2.00	70.00
sink and spigot	2.00	15.00
GENERIC PERSONAL STOWAGE SPACE (50kg)	6.00	50.00
RESTRAINTS AND MOBILITY AIDS (100kg)	2.00	100.00
GENERIC TOOLS EQUIPMENT (20kg)	50.00	20.00
GENERIC MEDICAL/SURGICAL/DENTAL SUITE (1000kg for Mars Mission)	1.00	1000.00
SLEEP PROVISIONS	6.00	9.00
GENERIC OPERATIONAL SUPPLIES (20kg)	6.00	20.00
personal hygiene kit	18.00	1.80
Portable Breathing Aparatus	12.00	1.20
Pressure Control Assembly (PCA)	4.00	78.20
X-38 HI PRESSURE G02 REGULATOR	2.00	1.40
X-38 HI PRESSURE GN2 REGULATOR	2.00	1.40
X-38 LOW PRESSURE G02 REGULATOR	2.00	1.40
X-38 LOW PRESSURE GN2 REGULATOR	2.00	1.76
ISS NITROGEN STORAGE TANK	1.00	109.00
GENERIC CLOTHING 10kg	9.00	10.00
ALS Lyophilizer	2.00	250.00
ALS Lyophilizer	2.00	250.00

Table 3-17: Detailed mass budget for the anticipated life support system

The life support system should not be considered exhaustive. It is merely a list of major components, which give an indication of what LSS mass has to be anticipated.
Note that that the list also includes hardware based on life support and crew accommodation needs.

3.3.2.6.1 Budgets

The two system options presented have the following mass budgets:

Anticipated Technology	
Mass consumables (t)	10.3
Mass system (t)	13.7
Today's Technology	
Mass consumables (t)	22.3
Mass system (t)	19.4

Table 3-18: Mass budgets

Two power modes, a day and a night power mode, were investigated during this study. The power schemes are based on the power requirements of the equipment to achieve the optimal mass throughput through the system. Further, when possible, the power needs were optimised to provide a relatively even power requirement of the LSS independent of the time of the day to simplify power generation, power conditioning and power storage. The results are shown in Table 3-19:

Power	
Power requirement day (kW)	10.1
Power requirement night (kW)	9.3

Table 3-19: Power budgets

Only a first estimate for the volume of the life support system has been achieved in the course of the study. The internal volume requirement relates to the volume occupied by the ECLSS inside the pressurised vessel, as opposed to the external volume requirement, which relates to the volume needs outside the pressurised volume.

Volume	
Internal volume requirements (m^3)	49
External volume requirements (m^3)	7

Table 3-20: Volume requirements

3.3.3 Thermal

Objective of this part is to assess in a limited extent the feasibility of a Mars transfer vehicle thermal design and check the maturity / availability of the related technology to fulfill this mission. A possible design is discussed as an example and its budget quantified to output the constraints toward the system (mass and power budget).

3.3.3.1 Requirements and design drivers

The thermal requirements are mainly driven by the presence of humans on-board and by the complexity of a vehicle leaving Earth orbit. A particular robustness and reliability are therefore expected for the thermal control system (TCS) and required during the complete mission duration.

Mass shall be optimised, especially considering the significant contribution of the TCS to the overall budget (thermal protection in particular). Trade-offs of TCS performance against safety appear as an important driver for such study.

The main requirements are the following:

- The TCS functions are to maintain air temperature and humidity in the habitable zone within preset limits, and to thermally control the on-board systems. Therefore, TCS shall be designed:
    - to maintain the habitable zones in a certain comfort zone (temperature, humidity) but respecting also safety requirements (touch temperature, condensation avoidance). Standard figures are a medium temperature between 18 and 27 ºC and a relative humidity from 25 to 70%.
    - to maintain a uniform environment for a crew up to six members. The volume of the habitable zone is a particular constraint on the sizing of the TCS elements.
    - to maintain elements and/or dedicated zones within temperature requirements (electronics, propellants, valves, …). To optimise the thermal budget, a certain rationalization of space and grouping of elements shall be carried out. Ideally, all equipments are within a single dedicated enclosure.
    - to maintain the interfaces of all modules within temperature requirements, for all possible configuration (THM separated from MEV for example). When decoupled, heat leaks of these interfaces can be severe if not thermally accommodated (large surfaces).
- The candidate TCS architecture shall be also capable of :
    - Guaranteeing adequate flexibility and reliability of the system for a long duration mission (2.6 terrestrial years)
    - Guaranteeing the performance of the system for any spacecraft attitude and for all thermal loads derived from the mission requirements. This severe constraint for the heat rejection capability guaranties a decoupling with attitude control reliability.
    - Optimising the heat management system in term of efficiency versus penalties to the system (mass, energy consumption)
- Guaranteeing safety by adequate provision of thermal hardware for the whole mission (necessary autonomy of the crew)
- Fully verifying and testing TCS on ground.

3.3.3.2 Assumptions

- Incidental thermal fluxes are the result of the vehicle attitude against the relative location of the different heat sources (the Sun or a planet). A conservative approach is to consider envelopes through worst-case scenarios:
  - maximization of the absorbed radiative energy (normal incidence) for hot cases, minimisation of the absorbed radiative energy (maximization of the Sun angle) for the cold case
  - Environmental heat loads values are shown in Table 3-21:

	Solar flux [W/m^2]	Planet albedo	Planet IR [W/m^2]
Hot case (Earth LEO, WS, 1 AU)	1423	0.33	241
Hot case (Venus swing-by, 0.7 AU) [1]	2904	negligible	negligible
Hot case (Mars orbit, perihelion, 1.38 AU) [2]	717	0.29 (subsolar)	470 (subsolar) to 30
Cold case (Mars orbit, aphelion, 1.66 AU) [3]	493	0.29 (subsolar)	315 (subsolar) to 30

Table 3-21: Thermal cases definition

Note that the Venus swing-by is an option. That with respect to Mars arrival and departure dates, the vehicle passes aphelion and perihelion. That for the hot case with Mars, and cold cases around Mars depend in a certain extent on the orbit of the spacecraft (and thermal characteristics of the underneath regions). The worst cold case is sought with long eclipse duration: 500 km circular orbit and a coplanar Sun (beta 0) give a 13.6 mn eclipse out of a 40.2 mn orbit.

- Thermal design shall manage all internal heat loads resulting from the human activities and various dissipating equipments:
  - Total mean heat load of 12.3kW during LEO
  - Heat load turndown ratio of 1.2
- Metabolic dissipation is estimated to be 110W (steady activity) or 295W (active state) per crew.

3.3.3.3 Baseline thermal design

The design block proposed is based on the exploitation of existing heritage: space stations on one hand (ISS, Skylab), and visiting/orbiting vehicles on the other hand (STS, Apollo, Soyuz). Undergoing or foreseen technological developments, and in general non-qualified hardware, are excluded.

The thermal control philosophy adopted for such vehicle is standard and relies on the following approach:

- simplification of the heat transfer with maximal use of thermal decoupling when possible
- use of thermal regulated bus to recuperate and transfer internal heat to heat sinks
- use of switch capability to modulate this transfer and balance the heat inputs from the outputs, and thus maintain temperatures within a certain bandwidth.

This is implemented with the use of appropriate materials and technologies combining passive or active means.

3.3.3.4 Habitation module thermal control

3.3.3.4.1 Overall architecture and safety

Thermal architecture shall be designed to guaranty a sufficient performance for the complete lifetime of the vehicle. By understanding the functions needed for this performance, this report can outline the limiting factors of this subsystem and design the adequate reliability. The thermal functions required are an acquisition system, a heat transport system, a heat rejection system, an insulating system and a control and command system, as shown in Table 3-22:

Functions	Basic features	Risks and required reliability
Acquisition system	Extraction of heat from the environment (air) or from dissipating equipment	Sensible to wear out problems Internal individual units (accessible). Shall be isolable => redundancy + spare
Heat transport system	Transfer the heat via a medium (liquid in general). A pressure differential is needed between input and output. Normally associated with a primary loop.	Similar to above
Heat rejection system	The medium is cooled down thanks to a cold sink (deep-space) and its energy decreased before reentering the loop. Normally associated to a secondary loop	High sensitivity to impact, ageing External/internal: access could be difficult => oversizing or redundancy
Insulating system and thermal protection	Overall heat balance is sized to optimise thermal budget (power in general). Adiabatic walls are targeted per simplification	Low to moderate sensitivity to impact, ageing External: access and replacement difficult => oversizing possible
Control and command system, thermostatic system	A modulation of the heat transfer (depending on the heat loads) is required to optimise the system. A feedback/monitoring of temperature (medium, air) pilots this modulation.	Related to CPU/CU, telemetry problems. Redundancy + spare. All controlled units shall be operable manually

Table 3-22: Thermal systems functions

In a first approach, this report can identify different type of failures associated to the thermal control elements:

- Beginning of life or infancy-related problems occurring in the first months. The failure rate is the highest of the TCS lifetime (depends on the quality of testing). The spacecraft is still in LEO orbit (extensive commissioning probable) and replacement can be easily performed.
- Random failure such as meteorite impact on a radiator. Critical or catastrophic depending on the redundancy level. Replacement of external elements during flight is bound to feasibility of an EVA.
- Degradation and wear out problems can be solved by spare units when located internally. External thermal control elements shall also perform well in a degraded mode (EOL analysis, ageing testing)

Redundancies at different level can be foreseen to cope with diverse contingency situations, depending on the criticality of the failure and required reaction time.
The thermal control design shall be capable of operating nominally after a single failure at any point of the TCS architecture. To do so, the safest and standard approach is to have primary and secondary loops fully redundant (in cold redundancy). An alternative to a full and cold redundancy is a possible local reconfiguration (local bypass from nominal to redundant) if adequately completed by redundancy at unit level. Such an approach could eventually lead to a complementary system up to a certain level as long as each can guarantee a nominal mode. This flexibility could prevent a degraded/survival mode after a second failure. Redundancy at unit level for critical units (pumps for example) and adequate provision of spare for maintenance are foreseen depending on the redundancy level.

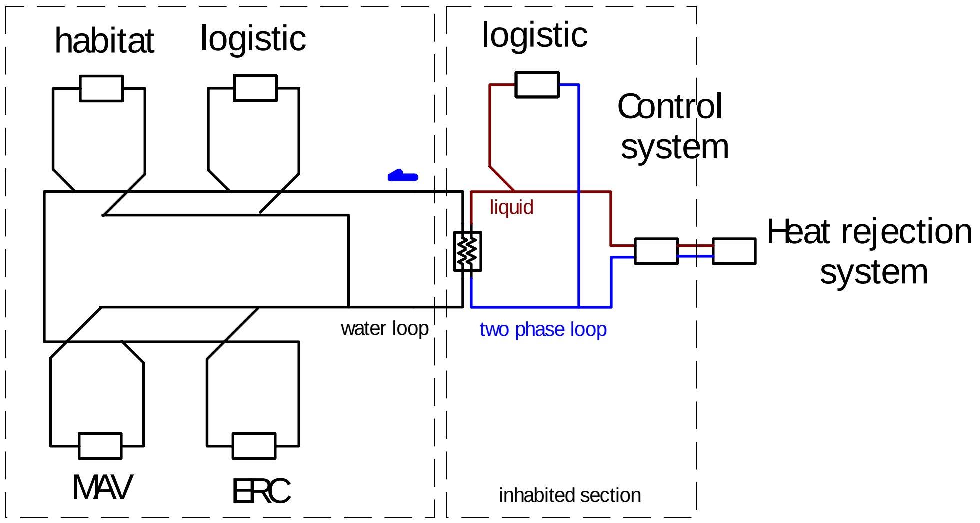


Figure 3-32: Thermal bus (primary and secondary loop)

The choice of a distributed or a centralized thermal control system depends on the system architecture and on the location/distribution of the dissipating elements. A distributed TCS offers thermal hardware simplicity (local thermal control) at the expense of a heavier configuration at system level. The requirements of modularity and flexibility do not call for a distributed architecture but for a centralized system completed by a judicious use of local thermal configuration where advantageous. A modular system is therefore proposed with parallel primary loops to pick up and convey the loads toward a central thermal bus.

3.3.3.4.2 Acquisition system

Its function is to remove locally a certain quantity of energy (heat) and the technology of the acquisition is adapted to the type of elements to control:

- high dissipative components are mounted on *cold plates*
- medium to low dissipative components are controlled via forced convection and mounted on *baseplates* thermally connected to the structure (hull or platform)
- integrated systems (within racks) are controlled via *dedicated fluid loops* (gas or liquid)
- air is sucked in by fans and canalized in a *heat exchanger* (air/liquid) and dehumidifying system.

Europe has acquired these technologies through the Spacelab and Columbus programs (including internal P/L like Biolab). These techniques can be qualified or mature and require tailoring to low/moderate development to fit specific purposes.

Figure 3-33 shows an acquisition system within primary loop:

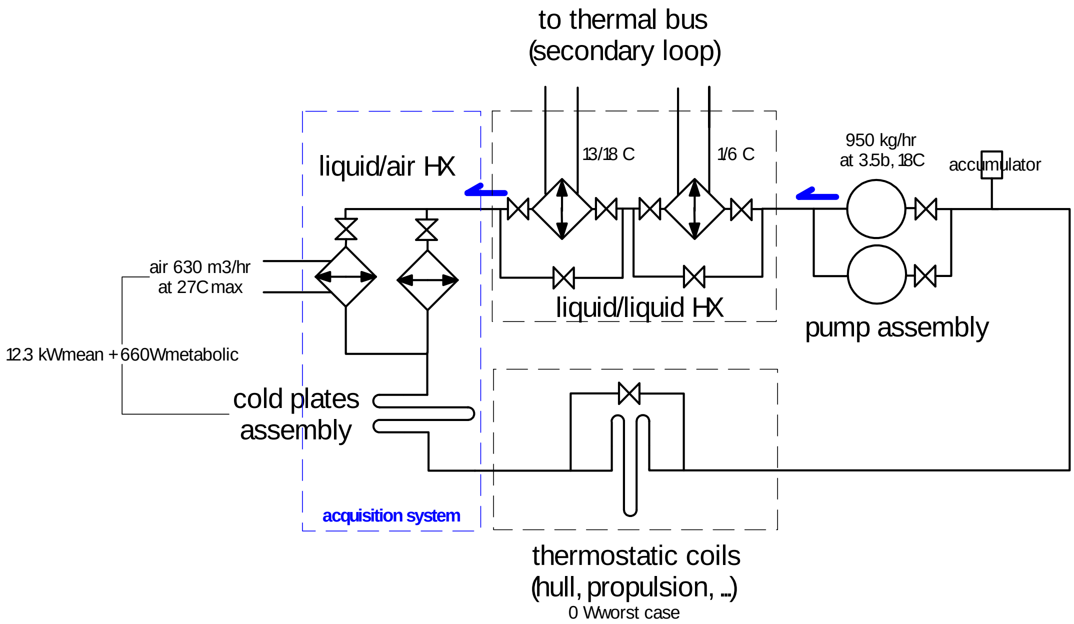


Figure 3-33: Habitat module / primary loop principles

The acquisition system is integrated within a primary loop for which two cold thermal sinks are available (second discrete temperature to cope with high peak dissipation). For certain units requiring a smaller bandwidth, the degree of heat exchange is regulated by the selection of adequate mass flow, such as in the liquid/air heat exchanger where one or two units in parallel can be activated with selectable regulation of the flow for each.
Overall recuperated heat is transferred to the loop fluid that circulates inside bypass coils to thermostatically control certain elements.
The same principles apply to the MAV and ERC main loops, connected to the thermal bus through hydraulic connectors (two-way sealing by springs) at the interfaces.

3.3.3.4.3 Heat transport system

Characterization of the heat transport system can be outlined providing basic features of the fluid loop and its working fluid:

- selection of a fluid loop depends mainly on the total power to transfer, the transport distance, and the available working fluid
- selection of the working fluid depends on its thermodynamic, hydrodynamic and safety performance (containment materials or man-related safety issues such as toxicity),

- both selections depend on the required performance and the allocated budget from the system (mass, power, environment)

Table 3-23 shows the characteristics of the fluid loop selection:

Type	Advantage	Disadvantage
Single phase fluid loop	Predictable, testable No limitation of heat removal (depends on the pump capability) Heritage: all vehicles and stations (Soyuz, Apollo, MIR, ISS)	Relative high mass and power consumption Temperature of the fluid can only increase while collecting heat through the acquisition system. Reliability of the system is mainly related to the reliability of the pump
Two phase mechanical pump loops	Lower fluid flow rate for the same quantity of heat transported for a single phase loop (due to heat of evaporation) => lower electrical consumption from the pump, lower mass for the system (lower diameter of the transport lines, lower mass of the pump) Allow a accurate temperature control (better than 1C), this relatively independently of the power transferred. No heritage known with a complete system. A loop should have been installed on ISS under Russian responsibility but its development has been interrupted.	Sizing is more complex than single-phase systems Two-phase mixture difficult to predict and test, possible instability Physics of heat transfer sensitive to acceleration Reliability of the system is mainly related to the reliability of the pump
Capillary Pump Loop (CPL)	No mechanical pump (work is provided per capillarity) => higher reliability in the long term Heritage: Granat, Mars 96, Stentor	Reliability of start up: medium to low Pumping capability of the wicks (per capillarity) is limited (<0.5 bars) which moderate the heat removal capability (<10kW)
Loop Heat Pipe	Similar to CPL but with a reservoir More tolerant to NCG (unlike heat pipes)	Limited to the amount of heat an evaporator can acquire (<1kW)
Hybrid systems (CPL + mechanical pump)	Overcome critical phases such as start-up, peak power loads	

Table 3-23: Fluid loop systems

Table 3-24 shows the characteristics of working fluid selection:

fluid	Apollo CM	Apollo LM	Soyuz	Shuttle	Spacelab	*Spacehab*	*Hermes*
Internal loop	Water	Water and Ethylene Glycol (1 TCS)	water / glycol	water	water	*Water*	*water*
External loop	Ethylene Glycol		PMS 1, 5r	Freon 21	Freon 114	-	*Freon 114*

fluid	Skylab	Mir	ISS (US)	ISS (RSA)	COF	JEM	*Hope*
Internal loop	Water and Ethylene Glycol (1 TCS)	?	water	Water-glycol	Water	Water	*Water*
External loop		?	Ammonia	ammonia	-	CFC 72	*Ammonia*

Table 3-24: Working fluids in past systems

The use of water (western program) or derivatives (water-alcohol for Russia) for all internal loops (single phase) answers mainly the safety aspects of manned programmes (flammability, toxicity). Corrosion problems related to the use of alcohol (decreases the freezing point) were reported in the Apollo programme and affected the MIR station in 1997 (leaks in one of the core loops, loss of the Kvant external loop). For external loops, fluids with a low freezing point are required: silicone-based fluid, Freons or ammonia. The progressive elimination of chlorines (CFC in 1992, HCFC by 2030) from the refrigerant industry raises a challenge in finding alternatives that can satisfy both thermal and safety requirements.

Fluid	Basic features	Data
Water	High freezing point => not suitable in external loops Low vapour pressure => not suitable in two phase loops Not compatible with aluminium	Specific heat: 4180 J/kg/K Density: 998 kg/m3 Thermal conductivity: 0.604 W/m/K latent heat: 2.2E6 J/kg (at 100C)
Ammonia	High vapour pressure No corrosion with aluminium Highly toxic	Freezing point: -78C at 1b; evap.: -33C at 1b Specific heat: 4600 J/kg/K (at 0C) Density: 642 kg/m3 (at 0C) Latent heat: 1.27E6 J/kg (at 0C)
Water - glycol mixture **(e.g. LZ-TK-5)**	Needs proper inhibitor to prevent corrosion High viscosity at low temperature Slightly toxic, flammability concern	Freezing point: -18C at 1b Specific heat: 3530 J/kg/K Density: 1057 kg/m3 Thermal conductivity: 0.43 W/m/K
Silicone mixture **(e.g. Polymethylsiloxane)**	Very low freezing point but high viscosity and low specific heat Corrosive to certain metals Toxic, flammability concern	Freezing point: -134C at 1b Specific heat: 1800 J/kg/K Density: 912 kg/m3 Thermal conductivity: 0.106 W/m/K Flash point: 56C
Freon 114	Ozone depletion fluid (ODP)	Freezing point: -94C at 1b Specific heat: 998 J/kg/K Density: 838 kg/m3 Thermal conductivity: 0.066 W/m/K

Table 3-25: Working fluids properties

Note that, if gases (CO_2, air) could also be used over a wider temperature range, their poor cooling effectiveness compared to liquid restricts their usage to very specific conditions.

Design selection

- For large systems such as a transfer vehicle, CPLs are not suitable as a primary loop because of high heat loads and long line length (>10kW, >20 m), and a mechanical pumped loop is preferred.
- Candidate fluids for a biphasic system include ammonia, propylene (excluding freons), which tends to discard their use (at least in a extensive way) in a habited module for toxicity reasons. Selection of a friendly fluid such as water in this environment is preferred.
- A single-phase primary loop currently appears to be the best solution when considering the robustness of such system, although at a certain cost for the system. So far there is no evidence that a biphasic system, although higher performance,

could challenge this robustness with the same level of safety. Significant efforts would have to be spent in this case (evaporators).

- Design of the secondary loop depends on the heat rejection requirements.

3.3.3.4.4 Heat rejection system

Radiator type	Advantages	Disadvantages
Body mounted radiator 	Highly integrated, good protection of sensible thermal connection against perforation All critical elements are potentially accessible without EVA Contribute to radiation protection Virtually independent to attitude control (and its failure) if adequately sized	low performance heat rejection capability is limited by available surface and environment system complex, shall be compatible with the radiation protection or be located in a non habitable zone
Hybrid body mounted rad. 	Increase the radiative surface when limited by configuration Heat rejection capability can be modulated with adequate opening/closing of the covers	The deployable covers radiative capability are sensitive to attitude control performance Total or partial loss of the heat rejection if covers mechanisms failed to open
Deployable radiator 	High performance (each side can be a radiative surfaces) Optimised heat rejection with possible pointing capability Pointing capability prevents coating degradation by minimising solar radiation impingement	Total or partial loss of the heat rejection if perforated (debris, meteorides), or if failed during deployment To recover from a failure requires an EVA Refolding the radiator is risky

Table 3-26: Heat rejection systems

The supporting structures of a radiative surface can be either the body of the spacecraft itself or a dedicated deployable frame. Both systems present a number of advantages and drawbacks. A deployable system has a higher performance but is more susceptible to certain failures (deployment, perforation). With an extensive commissioning in LEO, the risk of deployment can be easily overcome as long as ulterior re-folding is not considered. Regarding perforation, the highest risk comes from LEO debris (size > 1 cm) and an intervention can be assumed without excessive constraints. Meteorites remain nevertheless a permanent threat during cruise but the risk to the radiator can be limited with a bumper (protection of the fluid line by a certain aluminium thickness). The partial or complete loss of a radiator is possible, but an appropriate redundancy can be foreseen.

The feasibility of mounting a radiator on the structural spacecraft body depends on the resources allocated by the system. Less efficient than a deployable radiator because of parasitic heat loads (from environment and from the spacecraft), mass and power penalties can become significant if a certain radiator efficiency is targeted. This is especially true for a large body (14 m x 6 m cylinder) and if implemented around a habitable zone (higher insulation mass to isolate the radiator from spacecraft). The independence and flexibility in terms of heat rejection for any spacecraft attitude, gained by mounting a cylindrical radiator may not justify its mass penalty.

Using a dedicated and inhabited area for a particular type of logistic (tanks, pumps, machinery) will ease the thermal control architecture and globally decrease the power and mass budget.

- Possible decrease of this compartment temperature (10C to 20C lower than the habited module) if pressurisation is done with an inert gas,
- Will improve the performance of the thermal accommodation of stored cryogenics fluids
- Easier integration of a biphasic system possibly using different working fluids (less restrictions on the safety).

Maintenance visits, if necessary, should be possible and not raise excessive constraints (use of portable oxygen supply). An adequate redundancy of these internal units will be foreseen to minimise interventions.

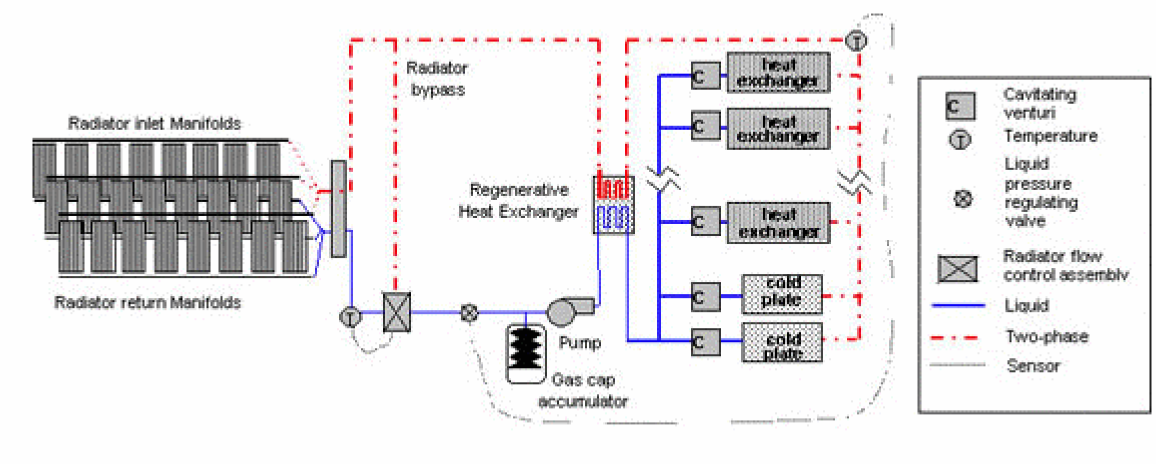


Figure 3-34: Secondary loop and radiators

There are different ways to control a fluid loop radiator:

- *bypass regulation* controlling the flow of the coolant and reducing the heat extracted from the fluid; a minimum heat must be available to prevent freezing.
- *regenerative regulation* that controls the temperature at the inlet of the radiator via a bypass valve that diverts the flow to the heat exchanger as a function of the radiator outlet temperature (advantage: the fluid differential is small => low heat load is required to prevent freezing; and almost constant mass flow can be maintained)
- *mechanical or electrically actuated systems* (louvres, pointing motorization, electro-emissive devices) that control the radiator view to space and/or limits the radiative exchange to it.

Design selection

A restricted area dedicated to thermally sensible logistic is implemented. This zone will host a biphasic system, retained as the secondary loop. This will include:

- biphasic heat exchangers, connected to the primary bus (water loop),
- biphasic cold plates for high-to-medium dissipative logistic,

- external and deployable radiators,
- regenerative regulation controlled via bypass valve
- ammonia will be used as the working fluid (higher figure of merit than alcohols)

Different sinks will be made available to the user (commutation between different heat exchangers, use of a variable bypass, or a pump with variable mass flow) to accommodate the vehicle thermal loads.

Appropriate sizing and selection of devices would require better knowledge of the heat load distribution, pressure losses and in general of the main system architecture features.

On the basis of existing programmes (Columbus, ISS, Soyuz), a preliminary sizing has been done and a budget estimated (see Section 3.3.3.4) to answer the requirements on the temperature and on the heat dissipation.

- Figure 3-35 shows the differences between monophasic and diphasic systems:

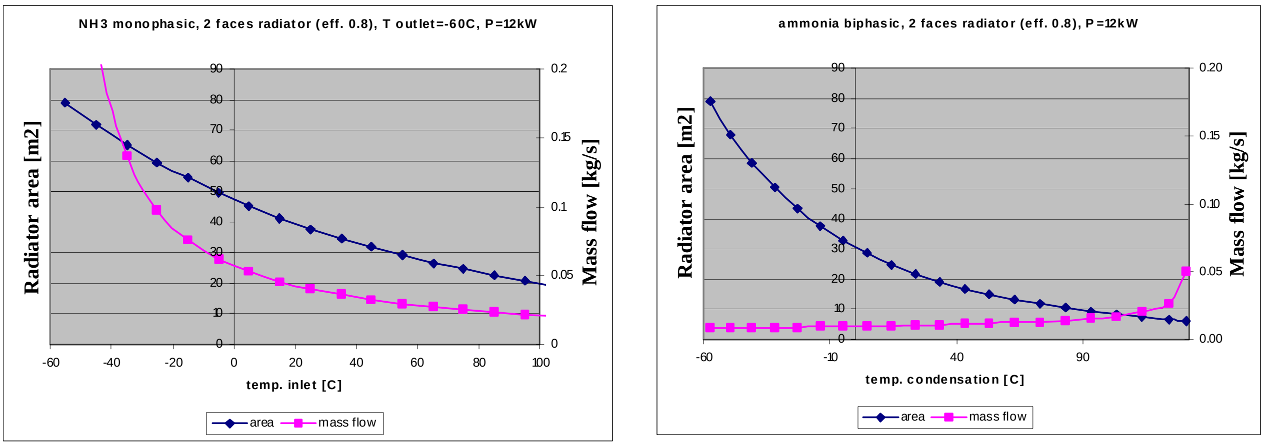


Figure 3-35: Comparative monophasic / diphasic

Figure 3-35 shows the advantages of a diphasic system: lower radiator surface and lower mass flow. Note that, for a same mass flow, equivalence of the two systems is never reached ($T_{boiling} - T_{freezing} < H/Cp$).

The parasitic heat loads on the radiators will result from a compromise between the different pointing constraints of the vehicle (solar arrays, antennas, radiators). It is premature at this stage to estimate which one would prevail, depending on the mass savings of this trade-off. Optimising the parasitic heat loads is possible if the Sun, the planet and the radiator are in the same plane, which is possible with a two-degree freedom or a one-degree freedom plus constraint on the vehicle attitude. An alternative is to constrain the radiative surface so that a certain level of absorbed energy can be tolerated, or finally to reduce the rejection to a single face.

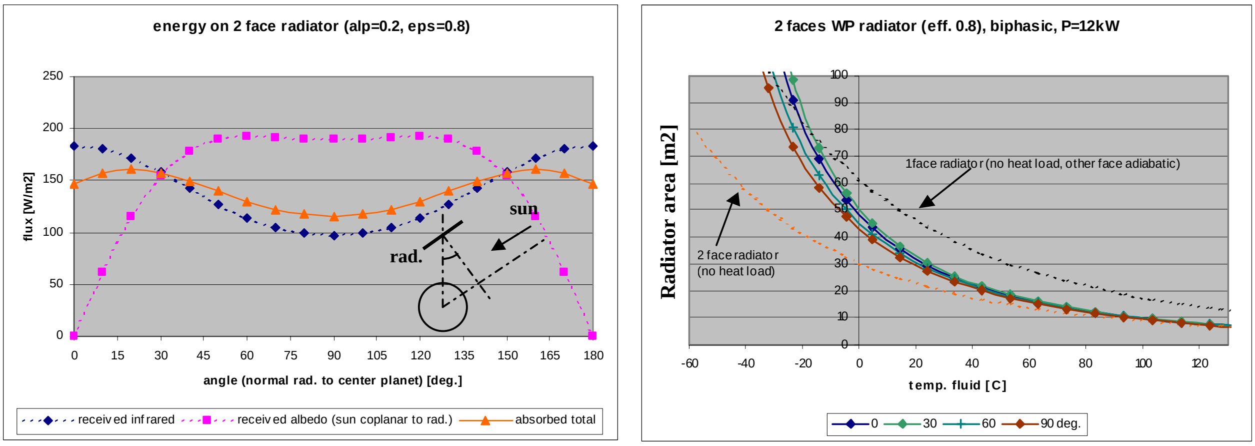


Figure 3-36: Total energy on radiator (L), Radiator size versus angle and temp. fluid (R)

Figure 3-36 (L) shows the total absorbed energy from Earth as a function of the angle it normal makes with the centre of the planet (the Sun being considered in the radiator plane) Figure 3-36 (R) shows the impact of this parasitic heat load on the radiator size. Despite planetary heat loads and the prize of the rotation angle, the size of a two-face radiator design remains inferior to a single-face (as long the radiator temperature remains superior to the planet temperature). The cost of a full tolerance of planetary heat load decreases with the fluid temperature increase (at 5C, the ratio is 1.17). The option of a fixed radiator could be therefore tolerated at this expense and with an adequate spacecraft attitude (no Sun on the radiators).

The technology to rotate a biphasic radiator is not available in Europe (nor is there a development plan), so a fixed radiator is retained with an adequate tolerance to planetary heat loads.

Out of the influence of the planet, the system performs optimally and the heat rejection capability is naturally increased (by a ratio 1.37). The flexibility of the secondary loop is provided by the bypass valve regulating the mass flow and therefore the sink temperature provided to the users.

The minimum set point (high heat load) for the primary loop is set to 4/7C (inlet 13/18C), which gives a minimum of 0/4C required from the secondary loop inlet at the heat exchanger level. Setting the bypass inlet at -5/-1C provides a possible solution, and drives a radiator area of 56.6 m^2.

The Lockheed-Martin PVR assembly mounted on the ISS truss is taken as a baseline (see last picture in Table 3-26). The following configuration is chosen: 8 panels, each 2.1 x 3.4 m^2, deployable by an electric motor ‘scissor’ mechanism. Total weight is 849 kg per assembly (x 2). White paint coating (type PSG121FD) is applied on both sides. Its degradation over time is well known and can be controlled through a careful illumination from solar UV.
Two identical systems are mounted symmetrically on the spacecraft body.

3.3.3.4.5 Insulating system and thermal protection

The materials used within the debris and insulation shielding are selected and sized so that both functions (impact and thermal) can perform optimally under their respective loads. Low conductivity materials are therefore integrated: the open cell foam (three layers of 10 cm thick, preferred to closed cells for their better thermal behaviour under vacuum), the Kevlar fabric (5 layers) and the Nextel AF10 ceramic fabric (3 layers) offering also adequate and stable (material inorganic, therefore no degradation against time to be expected) thermo-optical properties (measured values: alp=0.24, eps=0.88).

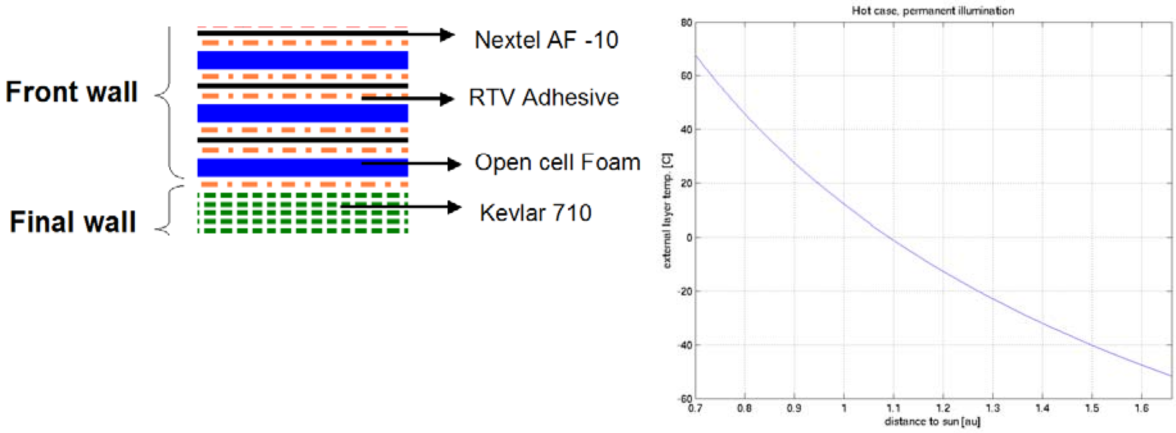


Figure 3-37: Debris shielding/thermal protection (L), Max. Temp. of external layer (Nextel) (R)

Assuming a three-axis-stabilized spacecraft, the temperature evolution of a permanently illuminated surface (Nextel AF10) is shown in Figure 3-37. Note that beyond a certain range, a correction appears necessary to come to acceptable temperatures at the hull internal structure (see following paragraph). The spinning of the spacecraft to homogenize temperatures is not considered (solar array pointing constraints).

3.3.3.4.6 Thermostatic system

Certain surfaces that cannot be protected by insulating means (docking system for the MAV) are treated (oxidation anodic, alodine) to minimise heat losses. On the internal face, coils (circulating fluid from primary loop) thermostatically control the temperature (condensation avoidance) and the heat exchanges (control of the heat losses). An adequate redistribution of the rejected heat (thermostatic coils) therefore reduces the use of heater power to the minimum.
When not directly accessible to fluid lines, externally mounted elements will require the use of strip heaters combined to an adequate insulation.

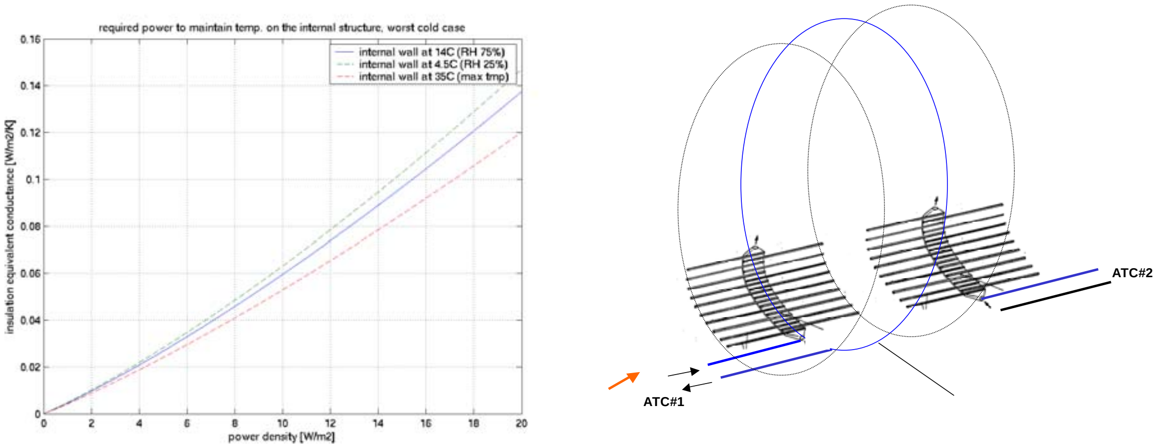


Figure 3-38: Power required to maintain temperature (L), Coil system + heat pipes schematic (R)

The insulating/debris shielding system as presented provides an equivalent thermal conductivity of 0.07 W/m^2/K. Therefore, maintaining in the worst cold case an internal wall above dew point (14C for 75% humidity) would require a power density of 11.5W/m^2.

Two systems are proposed:

- a network of heaters homogeneously distributed on the internal shell corresponding to a installed power of 5580W (assuming the vehicle as a cylinder 6 x 14 m). Two equivalent circuits (main and redundant) are foreseen, piloted each by a control unit. For safety, each circuit will be equipped with over temperature thermostats to protect against a failed-on heater switch.
- a network of coils / heat pipes mounted on the internal shell to transfer / homogenize the rejected heat from main loop.

With a mean rejected dissipation of 12 kW, there should be no need to draw power from the system for the heaters. However, to save mass, the network of coil / heat pipes will only be specifically located to sensible zones, sustained when and where necessary by the heaters system.

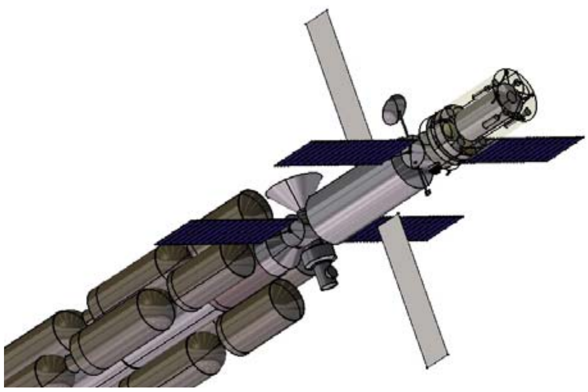


Figure 3-39: Overall view of radiators (size: 2 x 57m^2)

3.3.3.5 Cryogenic storage for oxygen tanks option

Although it was decided to store the oxygen for the life support system at high pressure, it has been also investigated the cryogenic technology. This section describes the design for this option.
Life support subsystem requires 394 kg of liquid oxygen to be stored and maintained for the duration of the mission (31.6 months). Considering the mass and related volume, it is assumed that these tanks can be accommodated in the same conditions of storage than for the fuel cells (no environmental loads from the Sun and planets but radiative environment from surrounding structures).
The required heat lift is shown in Figure 3-40. The dotted line indicates an ideal solution (heat lift exactly compensates heat losses through MLI) and the others lines where heat losses exceed heat lift, solution possible with a tolerance on the BO.
The number of tanks and their diameter are traded off so that despite possible boil-off (BO dependent on the diameter), the required capacity is reached at the end of the 31.6 months. Within that hypothesis, Figure 3-40 shows the relationship between tank diameter, number of tanks and BO. A sensitivity with two different temperatures of saturation (90 and 120K) is shown, taking into account the variation of density with temperature (higher temperature improves the performance of cryocooler).

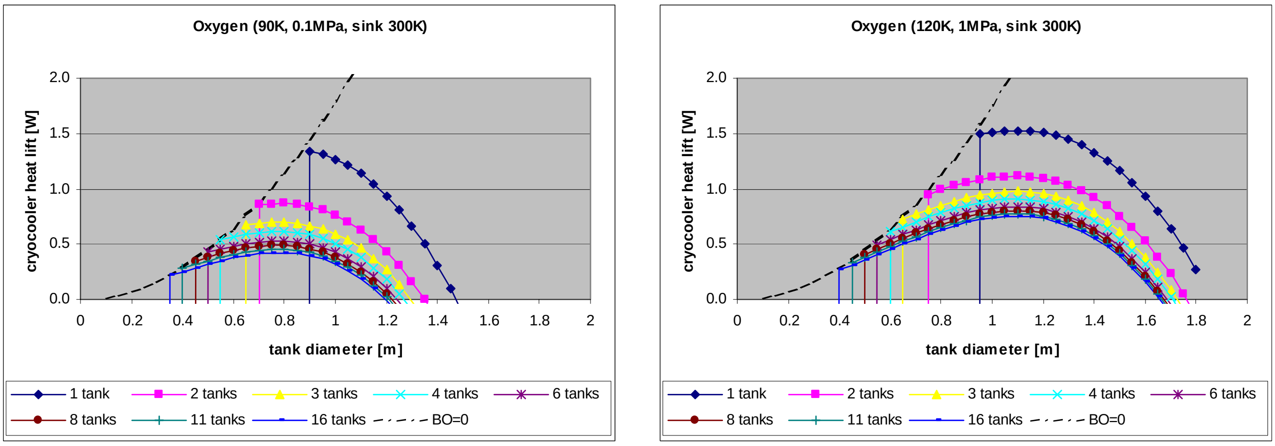


Figure 3-40: Oxygen tank dimensions

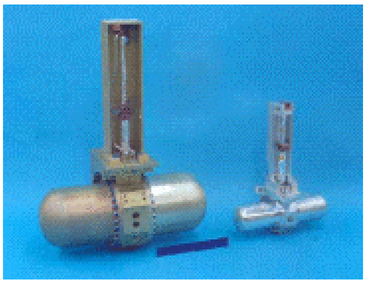
Given a sufficient tolerance on the boil-off, it is therefore not possible to cool down the tank and reach the required usable mass. The cost in terms of additional mass and volume is however acceptable, and an active cooling capability is preferred as a mass saving solution.

A hybrid thermal design is retained with maximization of insulation and use of a cooler: the number of MLI layers is set to 40 layers (DAM) and the pulse tube cooler is chosen as the most suitable cooling technology for this range of temperature (40-120K).
Significantly funded in the past years, the pulse tube offers now good performance and an increasing range of application. Very similar to Stirling coolers in its concept, the pulse tube has a simpler cold finger (no displacer mechanism) allowing a lower vibration level, an increased robustness and a wider temperature range. The compressor remains identical with the same configuration as for the Stirling (split configuration where two compressors are mounted back to back).

Note that, in Europe two ESA activities (Air liquid/Thales and Astrium/RAL TRP) are running for this technology with performance ranging from 800mW (required) to 7W at 80K. Development is oriented to detector cooling, but other applications requiring equivalent heat lift at higher temperature (up to 120K) do not present any showstoppers (performance is increased).

Dewar design

Following previous figures, two tanks of diameter 0.75 m and pressurised to 10 bars are selected. Each tank consists of a primary aluminum shell 3.5 mm thick, completed by a second shell enclosing the MLI stack (vacuum between the two shells). An external goldenized kapton foil insulates radiatively the external shell.



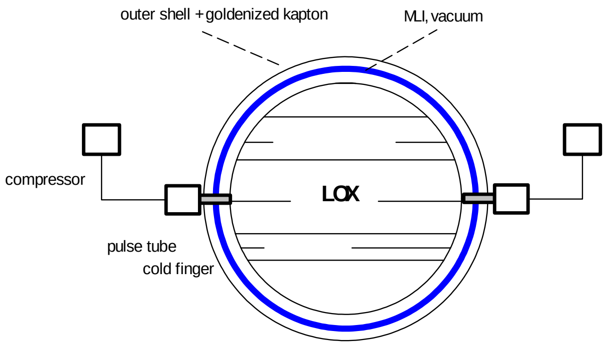


Figure 3-41: TES (L) and IMAS (R) pulse tubes (L), Cryostat and coolers schematic (R)

Two coolers are mounted on each pole of the tank, the cold finger connected to the internal shell via a thermal 'doubler' to better the thermal contact. To reduce parasitic thermal loads, Orbital Disconnect Struts (ODS) are used. Location of ODS and tubing are chosen close to the pulse tubes to reduce their thermal impacts.

3.3.3.6 Budget

Synthesis per subsystem (main features)

Fluid loops	
Primary loop	Pump assembly: 67 kg, 463W nominal (950kg/hr) (x 2) Condenser heat exchangers: 20.6 kg (x 2),cold plates: 3.4kg (x 10), valves (on/off, manual): 4kg (x 20) 120 kg of tubing (dry including insulation, brackets) + 105kg of water 4 coils mounted on the main body (38 m of heat pipes to spread energy): total 79 kg (including fluid)
Secondary loop	Pump assembly: 56.7 kg, 311W nominal (x 2) Heat exchangers: 15.9 kg (x 4), cold plates: 3.4kg (x 5), valves (on/off, manual): 4kg (x 5) 36 kg of tubing (dry including insulation, brackets) + 31kg of ammonia

Passive thermal control	
External radiators	Two radiators of 57 m2, each weighting 850 kg (15 kg/m2 from ISS PVR)
Insulation	0 kg for the main body of the transfer vehicle: the thermal properties of the MOD shield are exploited, the related budget transferred to structure. 150 kg are provisioned for specific external and internal elements insulation.
Heating system	5.58 kW installed power corresponding to 13 kg (heaters, thermostats, sensors, lines) 3 control units (1 on), each 6 kg, 29W when shell heaters are 100% duty cycle (COF)
Cryo systems	
Oxygen tank (life support)	4 coolers, each 3kg, consumption 35W each MLI: x kg for each tank (x 2)

Table 3-27: Design summary

<u>Overall budget</u> (as introduced in the system)

ement 1: Transfer Habitation Mod			MASS [kg]			
Unit	Click on button below to insert new unit	Quantit	Mass per quantity excl.	Maturity Level	Margin	Total Mass incl. margin
1	THM module / radiator	2	849.00	To be modified	10	1867.8
2	THM module / coil assembly (H	4	17.69	Fully developed	5	74.3
3	THM / insulation	1	150.00	To be modified	10	165.0
4	THM / HCU	3	6.00	To be modified	10	19.8
5	THM / Heaters, thermostat, lin	2	13.00	To be modified	10	28.6
6	THM / oxygen coolers	4	3.00	To be developed	20	14.4
7	int. loop / liquid	2	104.00	Fully developed	5	218.4
8	int. loop / dry tubing + insulatio	2	120.00	Fully developed	5	252.0
9	int. loop / pump assembly	4	67.00	To be modified	10	294.8
10	int. loop / compensator	2	3.00	To be modified	10	6.6
11	int. loop / cooler-dryer assemb	4	20.60	To be modified	10	90.6
12	int. loop / cold plates	10	3.40	Fully developed	5	35.7
13	int. loop / hydraulic connector	20	0.30	Fully developed	5	6.3
14	int. loop / choke washer	20	0.50	Fully developed	5	10.5
15	int. loop / on-off valve	20	4.00	Fully developed	5	84.0
16	int. loop / by-pass valve	2	20.30	To be modified	10	44.7
17	int. loop / manual valve	20	4.00	Fully developed	5	84.0
18	ext. loop / liquid	2	36.00	Fully developed	5	75.6
19	ext. loop / dry tubing + insulati	2	20.10	Fully developed	5	42.2
20	ext. loop / pump assembly	4	56.70	To be modified	10	249.5
21	ext. loop / cold plates	5	3.40	To be modified	10	18.7
22	ext. loop / compensator	2	3.00	To be modified	10	6.6
23	ext. loop / heat exchangers	8	15.90	To be modified	10	139.9
24	ext. loop / flow control valve	2	20.30	Fully developed	5	42.6
25	ext. loop / on-off valve	10	4.00	Fully developed	5	42.0
26	ext. loop / choke washer	10	0.50	Fully developed	5	5.3
27	ext. loop / heat exchangers	8	15.90	To be modified	10	139.9
28	ext. loop / flow control valve	2	20.30	Fully developed	5	42.6
29	ext. loop / on-off valve	10	4.00	Fully developed	5	42.0
30	ext. loop / choke washer	10	0.50	Fully developed	5	5.3
-	Click on button below to insert new unit			To be developed	20	0.0
LEMENT 1 SUBSYSTEM TOTA		30	3817.4		8.7	4149.7

Unit	Ppeak AND POWER SPECIFICATION PER MODE												
	Ppeak	OEM Pon	OEM Prtby	OEM Dc	TMIM Pon	TMIM Prtby	TMIM Dc	TMM Pon	TMM Prtby	TMM Dc	MOAM Pon	MOAM Prtby	MOAM Dc
1	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
2	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
3	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
4	29.0	29.0	13.7	100.0	29.0	13.7	100.0	29.0	13.7	100.0	29.0	13.7	100.0
5	5560.0	5560.0	0.0	2.0	5560.0	0.0	10.0	5560.0	0.0	10.0	5560.0	0.0	10.0
6	35.0	35.0	0.0	100.0	35.0	0.0	100.0	35.0	0.0	100.0	35.0	0.0	100.0
7	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
8	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
9	655.0	655.0	35.0	69.0	655.0	35.0	69.0	655.0	35.0	69.0	655.0	35.0	69.0
10	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
11	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
12	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
13	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
14	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
15	60.0	60.0	0.0	0.0	60.0	0.0	0.0	60.0	0.0	0.0	60.0	0.0	0.0
16	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
17	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
18	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
19	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
20	311.0	311.0	40.0	100.0	311.0	40.0	100.0	311.0	40.0	100.0	311.0	40.0	100.0
21	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
22	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
23	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
24	22.0	22.0	2.9	5.0	22.0	2.9	5.0	22.0	2.9	5.0	22.0	2.9	5.0
25	30.0	30.0	0.0	0.0	30.0	0.0	0.0	30.0	0.0	0.0	30.0	0.0	0.0
26	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
27	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
28	22.0	22.0	2.9	5.0	22.0	2.9	5.0	22.0	2.9	5.0	22.0	2.9	5.0
29	30.0	30.0	0.0	0.0	30.0	0.0	0.0	30.0	0.0	0.0	30.0	0.0	0.0
30	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
-													
LEMENT 1 SUBSYSTEM TOTA	6754.0	6754.0	94.5		6754.0	94.5		6754.0	94.5		6754.0	94.5	

Unit	PPEAK AND POWER SPECIFICATION PER MODE											PPEA
	OMM Pon	OMM Prtby	OMM Dc	TEIM Pon	TEIM Prtby	TEIM Dc	TEM Pon	TEM Prtby	TEM Dc	SM Pon	SM Prtby	SM Dc
1	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
2	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
3	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
4	29.0	13.7	100.0	29.0	13.7	100.0	29.0	13.7	100.0	29.0	13.7	100.0
5	5560.0	0.0	10.0	5560.0	0.0	10.0	5560.0	0.0	10.0	5560.0	0.0	100.0
6	35.0	0.0	100.0	35.0	0.0	100.0	35.0	0.0	100.0	35.0	0.0	100.0
7	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
8	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
9	655.0	35.0	69.0	655.0	35.0	69.0	655.0	35.0	69.0	655.0	35.0	69.0
10	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
11	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
12	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
13	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
14	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
15	60.0	0.0	0.0	60.0	0.0	0.0	60.0	0.0	0.0	60.0	0.0	0.0
16	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
17	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
18	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
19	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
20	311.0	40.0	100.0	311.0	40.0	100.0	311.0	40.0	100.0	311.0	40.0	100.0
21	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
22	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
23	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
24	22.0	2.9	5.0	22.0	2.9	5.0	22.0	2.9	5.0	22.0	2.9	5.0
25	30.0	0.0	0.0	30.0	0.0	0.0	30.0	0.0	0.0	30.0	0.0	0.0
26	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
27	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
28	22.0	2.9	5.0	22.0	2.9	5.0	22.0	2.9	5.0	22.0	2.9	5.0
29	30.0	0.0	0.0	30.0	0.0	0.0	30.0	0.0	0.0	30.0	0.0	0.0
30	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
-												
LEMENT 1 SUBSYSTEM TOTA	6754.0	94.5		6754.0	94.5		6754.0	94.5		6754.0	94.5	

Table 3-28: Overall budgets

3.3.4 Power

3.3.4.1 Design drivers

The main drivers of the power subsystem design are:

- The mission duration: 3 years and the required assembly time in LEO
- A safety level issued to the transportation of astronauts
- A power subsystem that fits with the assembly constraint (volume limitation, mounting,…)
- The use of technologies expected to be qualified in 2015

Future technologies for power generation, conditioning or storage are abundant. Some of them seem really interesting for space purposes but still only exist on paper or in the best case at the level of experimentation. For this design, technologies that are more mature (already qualified in space but in which improvements are expected in the coming years) and ones have reached a high state of art are the only ones taken into account.

The performances considered of each element (mechanical, electrical) are issued from the ESA roadmap technology plan.
The design proposed hereafter may therefore not be the most optimised one but is today one of the most feasible with the present technology knowledge. Therefore, the confidence in this subsystem will be higher compared to other possible designs for which technical improvements or qualifications are milestones that have less chance not to be reached on time.

3.3.4.2 Requirements

3.3.4.2.1 Mission requirements

The mission is divided into:

- An assembly phase until the Transfer Vehicle is completely built in LEO
- The cruise to Mars orbit with an arrival date expected to be November 2033
- 553 days in orbit on Mars with the separation with the MEV sometime during this phase (orbit duration: 122.95 minutes with a worst case eclipse of 41 minutes)
- The cruise back to Earth with a maximum duration of 1 year

During all the phases, power has to be supplied to the different subsystems.
Figure 3-42 shows the different modes of the mission. The time durations correspond to the reference time considered for the power design.

Mode Name	Definition	Acronym	Ref. Duration (mins)
Orbiting around Earth Mode	Crew onboard the THM Sub systems on: Lifesupport, comms, DHS, GNC, Power, Thermal	OEM	92.50
Trans-Mars Injection Mode	Crew onboard the THM Solar panels stowed Sub systems on: Lifesupport, comms, DHS, GNC, Power, Thermal	TMIM	<2880
Transfer to Mars Mode	Crew onboard the THM Sub systems on: Lifesupport, comms, DHS, GNC, Power, Thermal	TMM	1440.00
Mars Orbit Acquisition Mode	Crew onboard the THM Solar panels stowed Sub systems on: Lifesupport, comms, DHS, GNC, Power, Thermal	MOAM	-
Orbiting around Mars Mode	Crew onboard the THM MEV still attached Sub systems on: Lifesupport, comms, DHS, GNC, Power, Thermal	OMM	123.00
Trans-Earth Injection Mode	Crew onboard the THM Solar panels stowed Sub systems on: Lifesupport, comms, DHS, GNC, Power, Thermal	TEIM	-
Transfer to Earth Mode	Crew onboard the THM Sub systems on: Lifesupport, comms, DHS, GNC, Power, Thermal	TEM	1440.00

Figure 3-42: TV Mission Modes

As regards power, the OEM mode does not need closer study. Indeed, the link with a platform providing the request power is assumed. This platform has not been designed in this study.
During all the propulsion modes (TMIM, MOAM and TEIM), the mechanical load of the solar panels is too high. Therefore, the solar panels will be folded during these manoeuvres. For covering all these modes and also for safety purposes, the power system shall be able to supply the nominal required power for a duration of 6 hours without relying on the solar panels.

3.3.4.2.2 Power requirements

In this study, the power requirements have been computed module per module, unit per unit and mode per mode.
Each unit power profile is defined by 3 values:

- a peak power
- a standby power
- a duty cycle value (duration of the peak power compared to the total duration)

For every mode, the peak and standby values have been added to obtain values at system level. An equivalent duty cycle is also computed to keep the same level of energy (See Table 3-29). Table 3-29 also shows the power consumption requested for the MEV and the ERC modules.

			ERC	Thermal	MEV	Comms	Propulsion	DHS	Life Support	Mech	GNC	Harness (excl. PSS)	TOTAL CONSUMPTION
			linked	linked	linked	linked	linked	linked	linked	linked	linked		
Orbiting around Earth Mode	Solar Flux 1326 W/m² (0 if Solar Array not used)	Pon	166 W	4356 W	3466 W	370 W	0 W	80 W	31404 W	4203 W	0 W	881 W	44926 W
		Pstdby	66 W	108 W	1961 W	0 W	0 W	80 W	150 W	31 W	0 W	48 W	2444 W
		Duty Cycle	33 %	62 %	28 %	44 %	0 %	0 %	38 %	18 %	0 %		38%
Tref 93 min	Eclipse Mode NOT Included	Total Wh	153 Wh	4257 Wh	3673 Wh	248 Wh	0 Wh	123 Wh	18597 Wh	1190 Wh	0 Wh	565 Wh	28806 Wh
Trans-Mars Injection Mode	Solar Flux 0 W/m² (0 if Solar Array not used)	Pon	166 W	4356 W	3466 W	370 W	0 W	80 W	31404 W	10 W	0 W	797 W	40649 W
		Pstdby	66 W	108 W	1961 W	0 W	0 W	80 W	150 W	0 W	0 W	47 W	2413 W
		Duty Cycle	33 %	63 %	28 %	32 %	0 %	0 %	38 %	100 %	0 %		40%
Tref 2880 min	Eclipse Mode NOT Included	Total Wh	4752 Wh	133400 Wh	114355 Wh	5664 Wh	0 Wh	3840 Wh	579033 Wh	480 Wh	0 Wh	16830 Wh	858355 Wh
Transfer to Mars Mode	Solar Flux 705 W/m² (0 if Solar Array not used)	Pon	166 W	4356 W	3466 W	370 W	0 W	80 W	31404 W	470 W	0 W	806 W	41119 W
		Pstdby	66 W	108 W	1961 W	0 W	0 W	80 W	150 W	6 W	0 W	47 W	2419 W
		Duty Cycle	33 %	63 %	28 %	32 %	0 %	0 %	38 %	29 %	0 %		40%
Tref 1440 min	Eclipse Mode NOT Included	Total Wh	2376 Wh	66631 Wh	57178 Wh	2832 Wh	0 Wh	1920 Wh	289517 Wh	3350 Wh	0 Wh	8476 Wh	432279 Wh
Mars Orbit Acquisition Mode	Solar Flux 0 W/m² (0 if Solar Array not used)	Pon	166 W	4356 W	3466 W	370 W	0 W	80 W	31404 W	10 W	0 W	797 W	40649 W
		Pstdby	66 W	108 W	1961 W	0 W	0 W	80 W	150 W	0 W	0 W	47 W	2413 W
		Duty Cycle	33 %	63 %	28 %	32 %	0 %	0 %	38 %	100 %	0 %		40%
Tref 360 min	Eclipse Mode NOT Included	Total Wh	594 Wh	16675 Wh	14294 Wh	708 Wh	0 Wh	480 Wh	72379 Wh	60 Wh	0 Wh	2104 Wh	107294 Wh
Orbiting around Mars Mode	Solar Flux 493 W/m² (0 if Solar Array not used)	Pon	166 W	2280 W	3466 W	387 W	0 W	80 W	31404 W	2428 W	0 W	804 W	41014 W
		Pstdby	66 W	108 W	1961 W	17 W	0 W	80 W	150 W	6 W	0 W	48 W	2435 W
		Duty Cycle	33 %	5 %	28 %	100 %	0 %	0 %	38 %	18 %	0 %		35%
Tref 123 min	Eclipse Mode NOT Included	Total Wh	203 Wh	465 Wh	4884 Wh	792 Wh	0 Wh	164 Wh	24730 Wh	927 Wh	0 Wh	643 Wh	32807 Wh
Trans-Earth Injection Mode	Solar Flux 0 W/m² (0 if Solar Array not used)	Pon	166 W	2280 W	3466 W	370 W	0 W	80 W	31404 W	794 W	0 W	771 W	39331 W
		Pstdby	66 W	108 W	1961 W	0 W	0 W	80 W	150 W	0 W	0 W	47 W	2413 W
		Duty Cycle	33 %	6 %	28 %	32 %	0 %	0 %	38 %	1 %	0 %		35%
Tref 360 min	Eclipse Mode NOT Included	Total Wh	594 Wh	1455 Wh	14294 Wh	708 Wh	0 Wh	480 Wh	72379 Wh	61 Wh	0 Wh	1799 Wh	91771 Wh
Transfer to Earth Mode	Solar Flux 640 W/m² (0 if Solar Array not used)	Pon	166 W	2280 W	3466 W	370 W	0 W	80 W	31404 W	1254 W	0 W	780 W	39800 W
		Pstdby	66 W	108 W	1961 W	0 W	0 W	80 W	150 W	6 W	0 W	47 W	2419 W
		Duty Cycle	33 %	6 %	28 %	43 %	0 %	0 %	38 %	11 %	0 %		35%
Tref 1440 min	Eclipse Mode NOT Included	Total Wh	2376 Wh	5727 Wh	57178 Wh	3840 Wh	0 Wh	1920 Wh	289517 Wh	3354 Wh	0 Wh	7278 Wh	371190 Wh
Safe Mode	Solar Flux 0 W/m² (0 if Solar Array not used)	Pon	166 W	2280 W	3466 W	370 W	0 W	0 W	31404 W	0 W	0 W	754 W	38439 W
		Pstdby	66 W	108 W	1961 W	0 W	0 W	0 W	150 W	0 W	0 W	46 W	2331 W
Eclipse Mode		Duty Cycle	33 %	47 %	28 %	0 %	0 %	0 %	38 %	0 %	0 %		38%
Tref 60 min	Eclipse Mode NOT Included	Total Wh	99 Wh	1129 Wh	2382 Wh	0 Wh	0 Wh	0 Wh	12063 Wh	0 Wh	0 Wh	313 Wh	15987 Wh

Table 3-29: Computation of power inputs

The equivalent power profiles obtained are not realistic for all points of view. The system peak power (which consists of the sum of the individual unit peak power) is a worst case never reached. It corresponds to the case in which all the equipment is simultaneously: that is dishwasher, laundry, communications, thermal, heaters…
Better insight in the power profiles cannot be obtained at this stage of the study.

3.3.4.3 Assumptions and trade-offs

3.3.4.3.1 Power generation: solar arrays

The strategy is to first size a design without the use of nuclear technology. The high level of energy that has to be supplied implies that a power generation has to be included in the power system. Therefore, the use of photovoltaic cells is taken into account. Until now, it is the only non-nuclear power generation system used in spacecraft. In this field, important research is taking place on to increase the efficiency of the cells. Also, research is being don on the development of thin film cells that could fit on flexible or inflatable structures. Such technologies may be available in 2015. Nevertheless, their development and qualification may need more time than expected. Consequently, the design will be performed with three types of cells:

- *AsGa Multi-Junction Cells with the present state of art* (Column 1 of Table 3-30): Efficiency AM0 (28ºC): 26.8%. At end of life, the efficiency drops to 17.73%
- *AsGa Multi-Junction Cells with the performances expected for 2015* (Column 2 of Table 3-30):
  Efficiency AM0 (28ºC): 32%. At EOL, this value is estimated at 25.85%
- *Thin-Film CIS Cells also projected in 2015* (Column of Table 3-30): 15% is assumed in AM0(28º) conditions. At the end of the mission it decreases to 12.94%

	AsGa Multi-Junction Cells			AsGa Multi-Junction Cells (2015)			Thin-Film CIS Cells (2015)		
illumination									
sunlight (worst case)	**450 W/m2**			**450 W/m2**			**450 W/m2**		
Concentrator multiplier	1			1			1		
net illumination	450.00 W/m2			450.00 W/m2			450.00 W/m2		
Solar Cells			*net conversion efficiency*			*net conversion efficiency*			*net conversion efficiency*
temperature	70 C			-12 C			7 C		
AM0 cell efficiency at 28C			**26.80%**			**32.00%**			**15.00%**
temperature coefficient			0.07%			0.07%			0.09%
at operating temperature			**23.86%**			**34.80%**			**16.89%**
direct terms									
radiation	5%	95%	22.67%	5%	95%	33.06%	2%	98%	16.55%
spectrum shift	0%	100%	22.67%	0%	100%	33.06%	0%	100%	16.55%
light intensity effect on Voc	2%	98%	22.21%	2%	98%	32.40%	2%	98%	16.22%
other	0%	100%	22.21%	0%	100%	32.40%	0%	100%	16.22%
product of direct terms		93.10%	22.21%		93.10%	32.40%		96.04%	16.22%
cell efficiency on Mars			**22.21%**			**32.40%**			**16.22%**
Solar Array									
stat terms									
Mismatch	1%			1%			1%		
Calibration	5%			5%			5%		
Random failure	5%			5%			5%		
UV-micrometeorites	1%			1%			1%		
total stat terms	7.2%	92.8%	20.61%	7.2%	92.8%	30.06%	7.2%	92.8%	15.05%
electrical									
Diode loss	2.5%	97.5%	20.10%	2.5%	97.5%	29.31%	2.5%	97.5%	14.68%
Harness loss	2.0%	98.0%	19.69%	2.0%	98.0%	28.72%	2.0%	98.0%	14.38%
optical									
Orientation loss (perp =>0%)	0.0%	100.0%	19.69%	0.0%	100.0%	28.72%	0.0%	100.0%	14.38%
Packing factor loss	10.0%	90.0%	17.73%	10.0%	90.0%	25.85%	10.0%	90.0%	12.94%
Shadow	0.0%	100.0%	17.73%	0.0%	100.0%	25.85%	0.0%	100.0%	12.94%
Margin	**0%**	100.0%	17.73%	**0%**	100.0%	25.85%	**0%**	100.0%	12.94%
summary									
conversion efficiency			**17.73%**			**25.85%**			**12.94%**
including dust (last day)			**0.00%**			**0.00%**			**0.00%**
mass									
mass / m2	2.90 kg/m2			2.70 kg/m2			0.60 kg/m2		
	79.76 W/m2			116.34 W/m2			58.25 W/m2		
	27.50 W/kg			43.09 W/kg			97.08 W/kg		

Table 3-30: Comparison of multi-junction cells and thin-film cell power generation

For compliance with the high level of load during the propulsion phases, the solar panels are mounted with a deployment and folding mechanism. Therefore, the polar platform solar arrays

(see [RD19],[RD20] & [RD21]) data are considered for the design. The electrical mounting, the cells, their coverglasses and the substrate are the only part taken into account in the solar panels mass budgets for the power subsystem. The other components of the panels (deployment mechanisms, motor, yoke,..) are integrated in the mechanical subsystem.
For the multi-junction cells, coverglasses of 150 µm are assumed (see Figure 3-43).
For the thin film cells, a value of 600 g/m^2 is assumed (see [RD22])

	Option 1: 140 µm (kg/m^2)	Option 2: 100 µm (kg/m^2)
Triple junction cells	0.720	0.514
150 µm coverglass	0.397	0.397
Coverglass adhesive	0.067	0.067
Interconnects	0.013	0.013
Cell adhesive	0.213	0.213
Bus/wire/diodes	0.307	0.307
Substrate	1.144	1.144
50 µm kapton		
Total	2.878	2.878

Figure 3-43: Mass budget for multi-junction cells options

15 panels of 1 metre make up one polar wing. Changing the cells and the substrate will require a new qualification. The length of the panels will be computed for this study.
To cope with a mechanical failure during the folding/unfolding manoeuvres, the solar wings are sized taking into account one panel loss.
The solar panels are mounted with a driving mechanism. Except during eclipses and manoeuvres, the solar panels are Sun pointed.
The sizing mode for the solar panels is the “Orbiting on Mars” when the eclipse is the longest (41 minutes) and the solar irradiance at its minimum. In that case, the solar arrays are designed for being able to fully recharge the battery before the start of the next eclipse. A failure of a complete solar panel is also taken into account for this computation.

3.3.4.3.2 Power storage

The power storage has to supply power during the eclipses on LEO and in Mars orbit. But the sizing cases are the manoeuvre phases in which the solar panels may be stowed up to 6 hours. Alternatively, there are other interesting storage technologies:

- Secondary batteries. The most efficient are the Li-Ion cells with round trip efficiency around 94%. The specific mass nowadays is around 100 Wh/kg. Improvement to 150 Wh/kg for 2015 is expected (cf [RD3],[RD4] and [RD5]).
- Regenerative fuel cells. The most advance ones are the PEM providing electricity and water by combining Oxygen and Hydrogen. On the other hand, the charge is performed by electrolysed water. The efficiency is estimated to be about 50 to 60%.

The regenerative fuel cells may be interesting for high energy requirements such as in this mission, but their poor efficiency involves a huge increase of the solar generator size. Since this part is already a critical point, the use of secondary batteries is preferred.

3.3.4.3.3 Power conditioning and distributing

To limit the harness losses a higher voltage should be chosen, typically 120V. Nevertheless, this voltage cannot be too high otherwise there is a risk of electrocution of the crewmembers and also to avoid plasma interactions on the solar wings.
Due to the large amount of power units, a regulated topology shall be selected.
90% is the efficiency assumed for the Battery Charger Regulator.
85% is the efficiency assumed for the Battery Discharge Regulator.
For safety reasons, the Power Conditioning and Distributing has to be double-failure tolerant:

- First failure: the spacecraft shall remain fully operational
- Second failure: the spacecraft shall still be operational with possible limitations on the mission.

For safety, an architecture with separated and identical power systems with crosstraps to the different units connected on the bus is proposed. Each power system is composed of a dedicate solar panel, a battery module, a power conditioning and distributing unit. Compared to the power required, one power system is added in the design to cope with possible failures.

3.3.4.4 Baseline design

3.3.4.4.1 Budgets

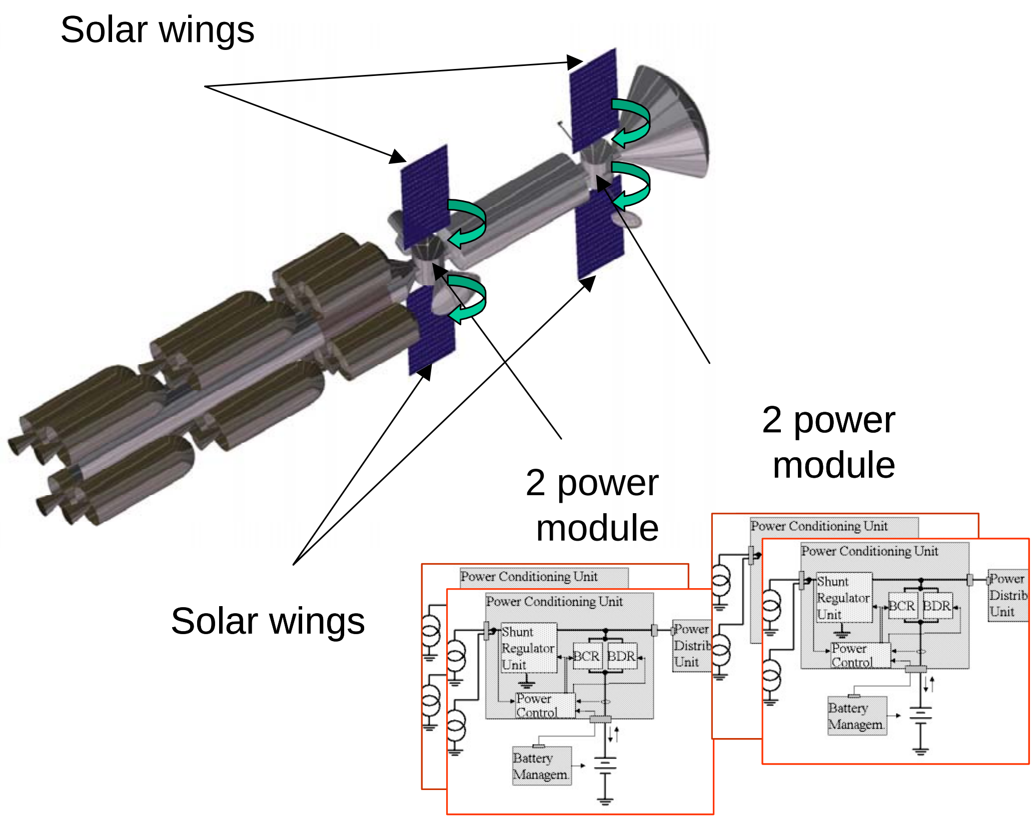


Figure 3-44: Power system overall architecture

There are two separate power modules in each of the two nodes, as shown in Figure 3-44. Each power module includes:

- One solar wing
- A Shunt Regulator Module
- Battery Charger and Discharger Regulators and the corresponding power control module
- A battery module
- A power distribution unit for delivering the regulated voltage bus to all the equipments

The solar wing is mounted with a solar-array driving mechanisms module.
The power system is designed to be able to be fully operational in case of failure of one of the four power modules.

With *advanced multi-junction cells*, the solar panels need to be 5.07 m long. The mass of one wing is estimated without margin to 205.4 kg (and without all mechanical parts that are counted in the mechanisms subsystem). A battery module is estimated to 552 kg without margin. It corresponds to the main part of the mass of the power subsystem. Such a PCDU module has a mass of about 151.8 kg.

20% of margin is taken up by all the equipments since technical improvements are taken into account for every module.

Element 1: Transfer Habitation Module			MASS [kg]			
Unit	Element 1 Unit Name Click on button below to insert new unit	Quantity	Mass per quantity excl. margin	Maturity Level	Margin	Total Mass incl. margin
1	Solar Wings	4	205.4	To be developed	20	985.7
2	PCDU	4	151.8	To be developed	20	728.4
3	Battery System	4	552.0	To be developed	20	2649.7
4				Fully developed	5	0.0
5				Fully developed	5	0.0
-	Click on button below to insert new unit		0.0	To be developed	20	0.0
ELEMENT 1 SUBSYSTEM TOTAL		3	3636.5		20.0	4363.8

Figure 3-45: Mass budget

3.3.4.5 Options

As a backup option, the solar arrays have been also computed with the present multi-junction (triple cells characteristics and also with the CIS thin film-cells.
Table 3-31 shows:

- The total area of four panels (baseline) and three panels (single case failure) of the arrays
- The mass of the solar arrays
- The length required for each panel

	Area (m^2)					
	4 panels	3 panels	Mass (kg)	Delta P (Wh)	Nb wings	Large (m)
Current AsGa TJ cells	444	333	958	0	3	7.40
Future AsGa TJ cells	304	228	610	0	3	5.07
Thin film CIS	608	456	273	0	3	10.13

Table 3-31: Solar arrays options budget

3.3.5 AOCS

The ACS system for the complete vehicle shall be modular as much as possible, trying to minimise hardware used and propellant consumption.
The functions of the ACS during the whole lifetime are listed in Table 3-32:

Phase	ACS functions
During LEO assembly	None
Between assembly completion and 1st TMI burn	Aero disturbance rejection Attitude control (no manoeuvre) No orbit maintenance
Transfer to Mars phase – 3 burns	Attitude control during non propelled phases No orbit maintenance Control during firing: TVC -> residual attitude rate 0.5 degrees/s Mid-course manoeuvre after TMI are separated
Transfer to Mars phase – orbit correction	Attitude control during non-propelled phases Control during firing: TVC -> residual attitude rate 0.5 degrees/s
Transfer to Mars phase – orbit insertion	Attitude control during non propelled phases Control during firing: TVC -> residual attitude rate 0.5 degrees/s 180 degree rotation manoeuvre (TBC)
Mars orbit	Attitude control Orbit maintenance
Return to Earth – propelled phase	Attitude control Control during firing: TVC -> residual attitude rate 0.5 degrees/s Mid-course correction
Return to Earth – cruise	Attitude control Earth retargeting

Table 3-32: ACS during mission lifetime

To limit the dimensioning of the ACS components, and considering that each individual stage has to have some manoeuvring capability as a stand alone item, an integrated control system is proposed. This means that all the stages shall participate in accomplishing the functions of each mission phase.
This results in a configuration of the ACS with actuators (i.e. thrusters) located along the vehicle in each "interstage". The thrusters' locations are shown in Figure 3-46, corresponding to the points P1 to P6.
The dimensioning of the control authority needed in each location is explained for each stage in the following chapters. The basic idea is to size the "higher" stages (from TMI to TEI and then MOI 3rd, 2nd and 1st stage), only to fulfil the functions they have to accomplish in their part of the mission. This would avoid overdesigning the ACS of the stage that has to go to Mars and back. Any ACS would however contribute to the control of the vehicle (according to its capability) throughout the mission.

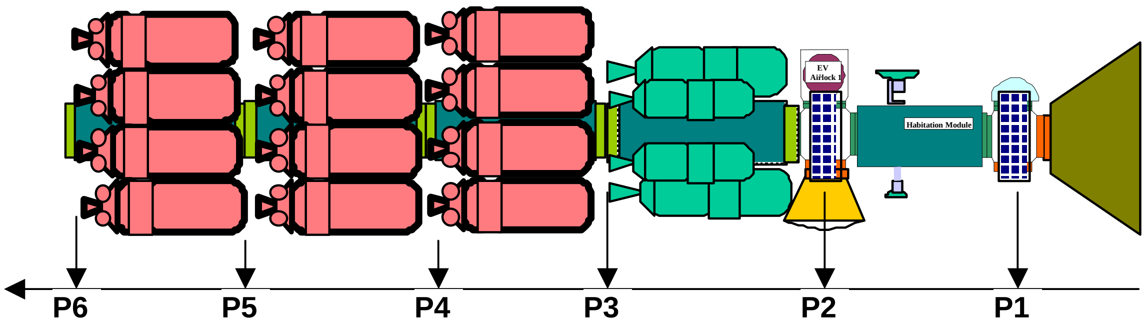


Figure 3-46: Vehicle configuration

Configuration	Length (m)	Total length (m)	Station
MEV	10	10	P1
THM	19	29	P2
ERC	0	29	P2
TEI	15	44	P3
MOI	0	44	P3
TMI 3rd	15	59	P4
TMI 2nd	15	74	P5
TMI 1st	15	89	P6

Table 3-33: Vehicle Dimensions

3.3.5.1 Habitation module AOCS

A core ACS shall be installed in the Habitation Module extremities, at locations P1 and P2.

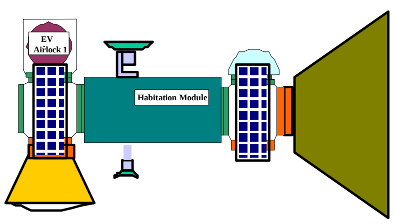


Figure 3-47: Habitation module ACS

Configuration	Length (m)	Total mass (tonnes)	Total length (m)	comp CoG (m)	Inertia total kg*m^2/1000
MEV	10	46	10	5.0	
THM	19	65	29	13.5	
ERC	0	15	29	15.3	22 874

Table 3-34: Habitation module dimensions

The functions the ACS has to accomplish are:

- Attitude control while cruising to Earth
- Earth retargeting
- Rate dumping after TEI motor separation: 0.5 degrees/s dumped in 30 sec (maximum attitude error within 15 degrees)

The most stringent requirement is the last one. To fulfil that a control authority of 6700 Nm is needed. If the thrusters are dislocated in the two stations P1 and P2, the following torque arms shall be realized (with respect to the centre of mass):
b1 = 13.7 m, b2 = 5.3 m

Thrust needed = 6700/13.7+5.3 = 360 N
This can be realized by either ATV ACS thrusters of 220 N each or Ariane-5 SCA thrusters, qualified for 350 N but easily adaptable also to 550 N.
So clusters of two thrusters could be used (or three thrusters, one redundant). This study will assume a cluster of two ATV thrusters, giving a control authority of 440 N.

To save propellant, while cruising, the attitude control shall be performed by using Control Moment Gyros (CMG) instead of thrusters. A set of CMG of the class of those used on the ISS is preliminarily baselined. The set of 4 CMG is about 230 kg. However, the suitability of these devices have to be verified against the disturbances induced by the crew, which have not been estimated yet.

3.3.5.2 AOCS for MEV+THM+ERC+TEI+MOI

Due to the integrated configuration for the TEI and MOI modules, a single ACS shall be designed, sized to comply with the worst case of the two mission phases.
In this configuration three control points shall be actives: P1, P2 and P3, as shown in Figure 3-48.

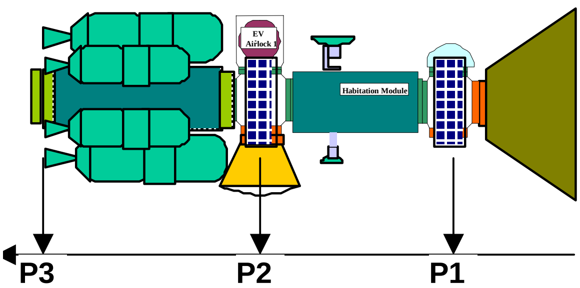


Figure 3-48: TEI and MOI configuration

Configuration	Length (m)	Total mass (tonnes)	Total length (m)	Comp CoG (m)	Inertia total kg*m^2/1000
MEV	10	46	10	5.0	
THM	19	65	29	13.5	
ERC	0	15	29	15.3	22874
TEI	15	160	44	27.2	63444
MOI	0	100	44	31.5	84414

Table 3-35: TEI and MOI dimensions

The ACS functions shall be:

- Attitude control while orbiting Mars
- Manoeuvres
- Rate dumping after MOI motor separation: 0.5 degrees/s dumped in 30 seconds (maximum attitude error 15 degrees)

Again, the last one is the most demanding

The maximum torque needed in this case is: 25 000 Nm
If ACS is dislocated in the three points, the torque arms shall be:
b1 = 21.5 m
b2 = 2.5 m
b3 = 12.5 m

Considering 440N thrust at b1 and b2, a torque contribution of 440*(21.5+2.5) = 10500 Nm is given by the ACS of the habitation module.

The additional thrusters to be located in b3 are:
Thrust needed = 25000-10500/12.5 = 1160 N
So a cluster of 2 Ariane-5 SCA thrusters (2x550=1100N) might be used here.

3.3.5.3 ACS for TMI 3rd

In this configuration four control points shall be actives: P1, P2, P3 and P4, as shown in Figure 3-49.

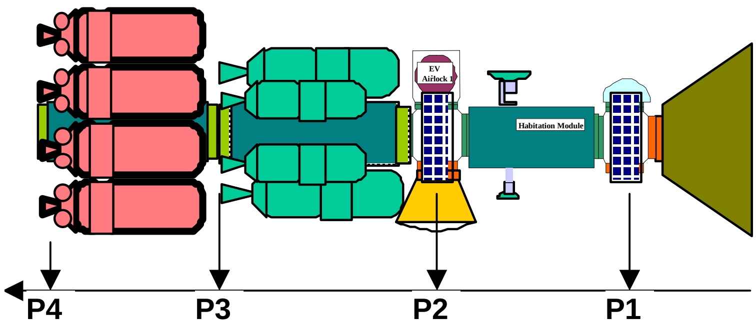


Figure 3-49: TMI 3rd configuration

Configuration	Length (m)	Total mass (tonnes)	Total length (m)	Comp CoG (m)	Inertia total kg*m^2/1000
MEV	10	46	10	5.0	
THM	19	65	29	13.5	
ERC	0	15	29	15.3	22874
TEI	15	160	44	27.2	63444
MOI	0	100	44	31.5	84414
TMI 3rd	15	249	59	39.4	172759

Table 3-36: TMI 3rd dimensions

The ACS functions shall be:

- Attitude control while cruising to Mars
- Manoeuvres
- Rate dumping after TMI 2nd stage motors separation: 0.5 degrees/s dumped in 30 seconds (maximum attitude error 15 degrees)

To cope with those functions, a maximum torque of 50 300 Nm is needed.

The available torque arms should be:
b1 = 29.4 m,
b2 = 10.4 m
b3 = 4.6 m
b4 = 19.6 m

The contribution from the other stages ACS have already been dimensioned:
Considering 440 N thrust at b1, b2, a torque contribution of 440*(29.4+10.4) = 17500 Nm is given by those ACS.
Plus 1100 N by b3 = 5000 N which is the contribution of P3,
a total control authority of 22500 N is provided by the already designed ACS

The additional thrust needed in b4 is:
Thrust needed = 50300-22500/19.6 = 1500 N

3.3.5.4 ACS for TMI 2nd

In this configuration five control points shall be actives: P1, P2, P3, P4 and P5, as shown in Figure 3-50.

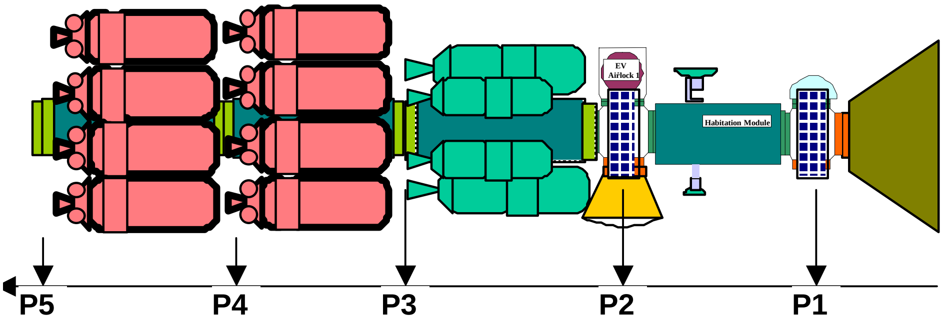


Figure 3-50: TMI 2nd configuration

Configuration	Length (m)	Total mass (tonnes)	Total length (m)	Comp CoG (m)	Inertia total kg*m^2/1000
MAV	10	46	10	5.0	
HAB	19	65	29	13.5	
ERC	0	15	29	15.3	22874
TEI	15	160	44	27.2	63444
MOI	0	100	44	31.5	84414
TMI 3rd	15	249	59	39.4	172759
TMI 2nd	15	332	74	48.7	370650

Table 3-37: TMI 2nd dimensions

ACS functions:

- Attitude control while cruising to Mars
- Manoeuvres
- Rate dumping after TMI 1st stage motors separation: 0.5 degrees/s dumped in 30 seconds (maximum attitude error 15 degrees)

Maximum torque: 108 000 Nm

Torque arms:
b1 = 38.7 m,
b2 = 19.7 m
b3 = 4.7 m
b4 = 10.3 m
b5 = 25.3 m

Considering 440 N thrust at b1 and b2, a torque contribution of 440*(38.7+19.7) = 25700 Nm is given by those ACS.
Plus 1100 N by b3 = 5100 Nm for P3
Plus 1500 N by b4 = 15400 Nm for P4

A total control authority of 46 200 Nm is provided by higher stages ACS

The additional thrusters to be located in P6 are:
Thrust needed = 108000-46200/25.3 = 2500 N

3.3.5.5 ACS for TMI 1st

In this configuration six control points shall be actives: P1, P2, P3, P4, P5 and P6, as shown in Figure 3-51.

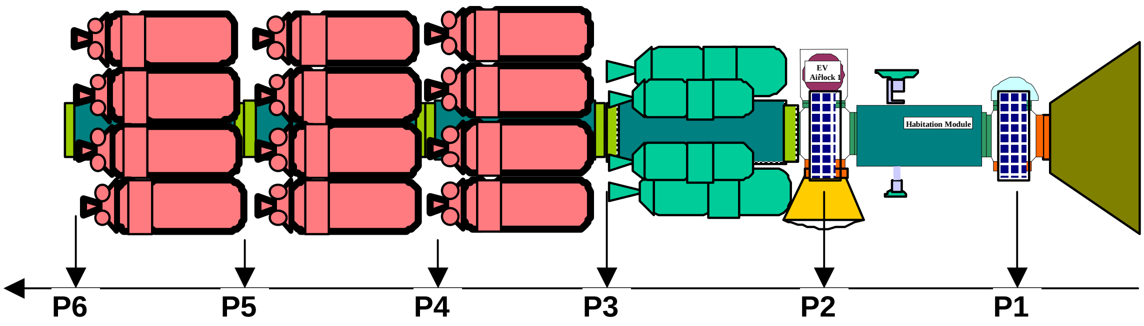


Figure 3-51: TMI 1st configuration

Configuration	Total mass (tonnes)	Total length (m)	comp CoG (m)	Inertia total kg*m^2/1000
MAV	10	46	10	
HAB	19	65	29	
ERC	0	15	29	22874
TEI	15	160	44	63444
MOI	0	100	44	84414
TMI 3rd	15	249	59	172759
TMI 2nd	15	332	74	370650
TMI 1ST	15	332	89	679909

Table 3-38: TMI 1st dimensions

The ACS functions shall be:

- Attitude control while cruising to Mars
- Atmospheric disturbance rejection
- Manoeuvres: a little re-orientation to get the optimal attitude for firing the TMI 1st stage can be assumed.

The last function is the dimensioning one. A re-orientation of 20 degrees in 4 minutes can be envisaged as dimensioning manoeuvre. This would mean a constant acceleration phase covering 10 degrees in 2 minutes and a constant deceleration phase.
The acceleration needed is: 2*Dtheta/t^2 = 2*10*pi/120/120^2= 2.4e-5 rad/sec^2

The maximum torque needed is 17 000 Nm, which can be easily realized by using the higher stages ACS.
As a result, in principle there would be no need for an ACS in the TMI 1st stage.
To give the control authority when the stage is not assembled on the main vehicle, most likely some ACS shall be envisaged as well. However the dimensioning is TBD.

3.3.5.6 Electrical architecture

Figure 3-52 shows the following electrical architecture which applies to all stages:

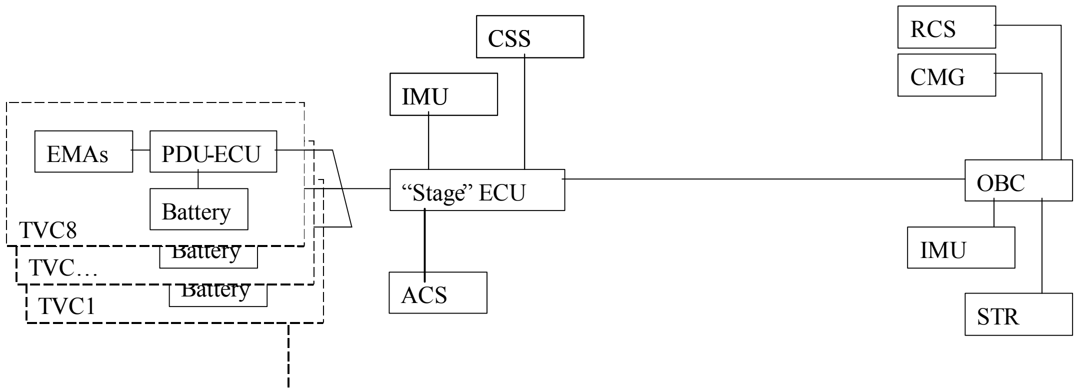


Figure 3-52: Electrical architecture

The main intelligence shall reside in the habitation module, where the principal attitude sensors (Inertial Measurement Unit, IMU and Star Tracker, STR) shall be mounted as well.
This main intelligence shall control directly the actuators mounted on the habitation module: i.e. Thrusters (RCS) and Control Moment Gyros (CMG). OBC will control the other actuators via several remote units, the "stage" ECU (Electronic Control Units). During the propelled phase the OBC shall control (via the ECU) also the Thrust Vector Control system (TVC). The inner control of the actuators of each TVC shall be realized through an additional box (PDU-ECU) mounted on the individual motor.
The stage ECUs shall be in charge of the control of the stage when not mounted on the main vehicle (before assembly or after separation, for de-orbiting), and shall work as routers when connected with the main OBC.
To accomplish the secondary function of controlling the stand alone stages, the ECU shall make use of local sensors: IMU and Coarse Sun Sensor (CSS). The IMU can be used (if needed) also during the control of the main vehicle to control the structural flexibility, which has not been considered in this study, but which might be a design issue in the future.

3.3.6 Data handling

3.3.6.1 General consideration

This section summarizes the requirements applicable to avionic systems that will fly on a long lasting human mission with target launch date 2033.
Several assumptions shall be made to generate a consequent design.

From the DHSs point of view a human mission to Mars is extremely challenging considering presently available space technologies.

Major issues that require attention are:

3.3.6.1.1 Computing power

The present and near-future European space computers are based on the SPARC RISC architecture (ERC32 then LEON2-FT). More powerful rad-hard Power-PCs are currently available, but under US ITAR control.
The roadmap for the first generation of LEON processor is now stabilized : flight models in 0.18 µm ATMEL technology should be available during the first quarter of 2005, and should fulfill the needs of the first Aurora missions.
Nevertheless, note that that the US processor available today already provide more processing power than the LEON will do in 2005: the first generation of radiation-hardened PowerPC 750 from BAE-Systems in 0.25 µm technology provides 240 to 300 MIPS, and a 370 MIPS version in 0.18 µm is foreseen before 2007.
To lower the gap with the US products, ways to improve of the internal LEON architecture should be studied. The power PC 750 architecture is far more complex than the LEON architecture: it includes several independent processing units that allow executing more than one instruction per cycle. Moreover it also supports level 2 cache.
This kind of architecture is needed to support higher-level operating systems that can guarantee soft real-time performances like Linux.
The use of COTS processors (mainly Power PC line of products) shall be assessed for non-critical payloads. The availability of commercial SOI process (see [RD28]), may boost safe use of high power chips in space, but dedicated development (especially for ASICs) is needed.

3.3.6.1.2 Maintenance, availability

The availability of space qualified electronic components has decreased in the latest years.
Moreover, the evolution of rad-hard technologies after 0.13 Pm is very difficult to predict: it is not yet known whether or not the hardening techniques at design level will compensate the increasing SEE sensitivity (see Figure 3-53). Consequently, the gap between commercially available technologies and radiation-hardened technologies may either decrease or increase.

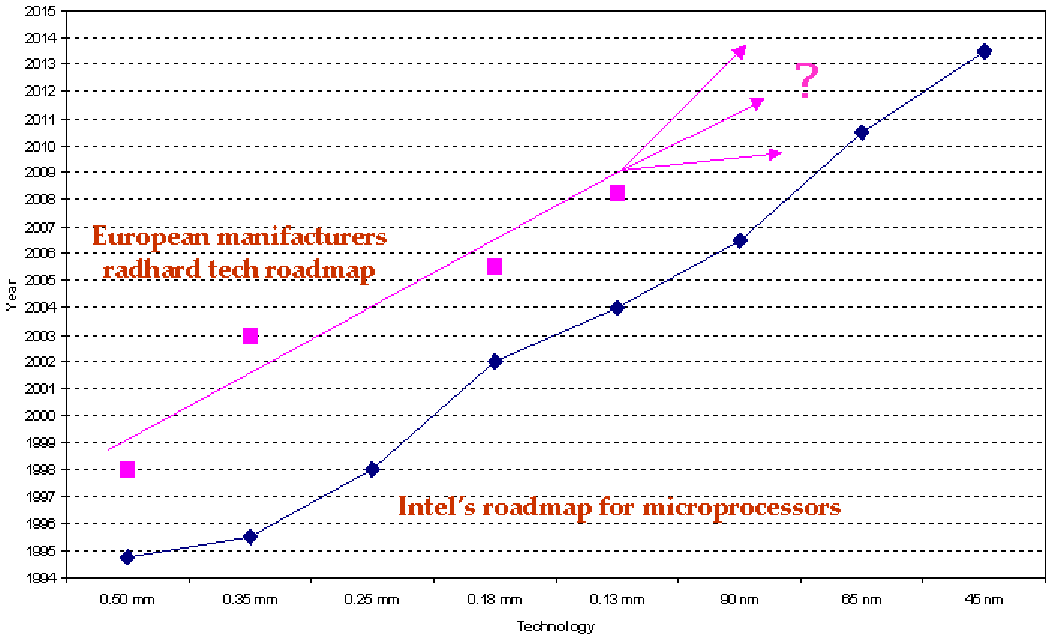


Figure 3-53: Comparison of present roadmaps for space qualified and commercial microprocessors

This introduces a problem not only in performances; already in some present designs sometimes basic EEE parts are chosen not for architecture optimisation but because of the lack of any other choice. At the moment (even including ITAR licenses) there is a dangerous lack of choice in ADCs, analogues, power components, FPGAs and memories.
To support a mission that will need 20 years to exploit (considering only FM electronics) a system of dedicated electronic supply lines shall be available. Current high-relativity electronic market in Europe is not sufficiently robust to support such an effort. Current radiation-hardened digital ICs for space applications are developed using a 0.35 µm CMOS process (ATMEL MH1RT) while the most advanced American ICs are developed in 0.16 µm (IBM). The conversion of entire modern production lines can be achieved by supporting European foundries (ATMEL and ST) up to the newest SOI processes (see Figure 3-54 below).

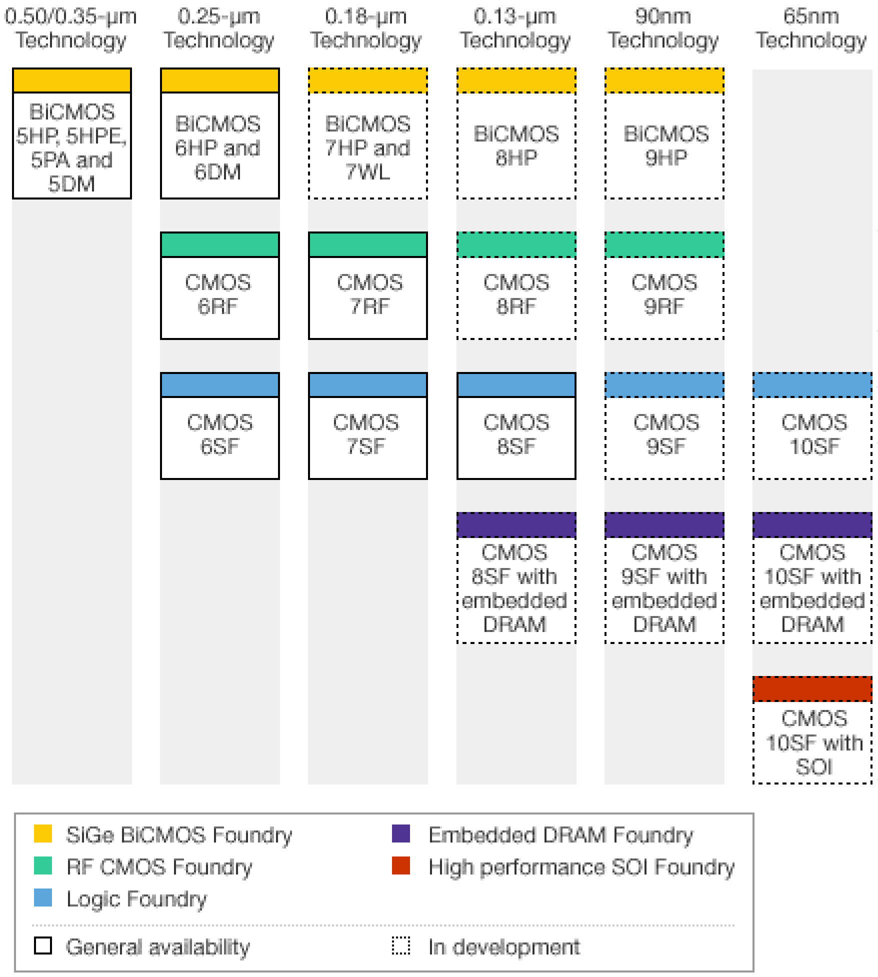


Figure 3-54: IBM foundry roadmap for high-performance microprocessors

3.3.6.1.3 Reliability

The longest manned mission ever flown in space, without the possibility of immediate escape or reentry (as in shuttle or ISS) are still the latest Apollo ones. Since then the complexity of on-board data handling systems has grown, together with its centrality in spacecraft control. On the other hand the amount of acceptable risk for the mission and the crew due to the probable failures of the on-board electronics has been reduced as a consequence mainly of the two shuttle accidents (even if they were not caused by the DHS).

Currently, it is impossible to design a truly 99.999 % reliable on-board data handling system. The biggest issue is the high degrees of customization present in any space electronics. Commercial electronic manufacturers have succeeded in having yields of some hundred of defective parts over millions about deeply embedding standardization in any electronic product.

3.3.6.1.4 Standardization in avionics

Standardization in commercial electronics started from leader computer manufacturers mainly for peripheral interconnect. Now standardization, through ISO and IEEE is involving practically any part of the design and commercial life of electronic and software products. The most recent successes in standardization were USB, that in its latest version is becoming the de-facto standard for external peripheral interconnection, POSIX, that as software standard allows the concurrent development of LINUX kernel by thousands of sparse users, IEEE 1451 standards family, including multidrop transducer bus, mixed mode transducer and wireless transducer network now used even in newest commercial airliners. The unrivalled father is the TCP/IP, so well known as the ‘standard among the standards’ taken as model for future standards development.

The current avionics are based on well-established standards as regarding ground-spacecraft communication (CCSDS, ESA PSS) and hardware level discrete interfaces (ESA TTC-01B). The future standards that will cover all the interfaces and protocols layers required to ensure the independence between the hardware level and the applications are still in development.
In the framework of the CCSDS organisation, several SOIS workgroups and “BoFs” (Bird-of-a-Feather”) are currently active:
The on-board bus and LAN workgroup defines services for the transfer of data over on-board buses and individual on-board LANs (local area networks) that constitute a single subnetwork.

The avionics architecture shall be an open architecture based on well known industrial standard. It will act at all layers, from hardware to embedded software to human interfaces, especially regarding the internal communications and interconnections for the different level of the design to facilitate the subcontracting of avionics subelements in accordance with any geographical return or other political constraint whilst minimising the risks at integration level.
Among other standardization efforts, the CCSDS 'SOIF' [RD31] providing isolation between the software applications and the communication’s logical and physical layers can be seen as the base common language towards a distributed intelligence spacecraft, capable of having functional redundancies to increase the system reliability.

3.3.6.1.5 Distributed control

From the Mars Sample Return mission on, in the framework of the Aurora project, the application of the newest standards is mandatory.
Nevertheless, the current avionics architecture and the ones foreseen for the ExoMars and Mars Sample Return missions are rather centralized (like the current HICDS demonstrator architecture). The standards required to implement centralized architecture are the ones that will be available at first. The subsequent missions (like the Human spacecrafts) will require more

processing power than available using a single processor, the architecture shall evolve later toward more distributed architectures, and possibly what is called "*integrated modular avionics*" in aeronautics (standardized multi-processor platform with transparent distribution of the functions instead of the classical allocation of processor dedicated to functions).
This last step will require the standardization of additional services to manage fault tolerance almost transparently for applications.

3.3.6.1.6 On-board interfaces

The outcome of standardization of on-board interfaces will be a set of busses covering all the levels of modular hardware integration. Each interconnection type is specialized and a stack of interconnection means is necessary to guarantee functional redundance and system optimisation. The SOIF layers will then be in charge of hiding hardware complexity at application software level. Figure 3-55 shows the complexity as a function of bandwith. Busses already qualified for space are shown in green.

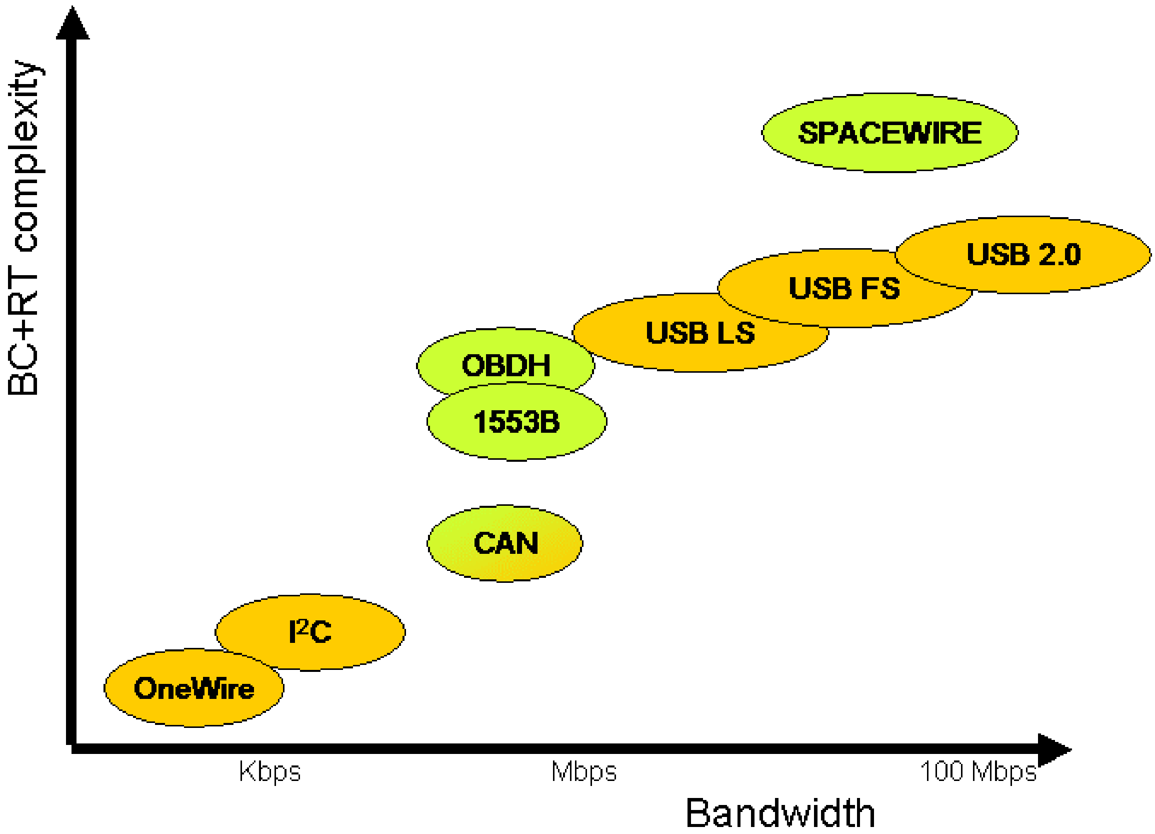


Figure 3-55: Complexity of interconnected busses as a function of bandwith

The categories to group on-board command and data interconnecting busses are:

3.3.6.1.6.1 Chip level bus (as AMBA)

Used for interconnection of IP-cores into FPGAs or ASICS. The use of reprogrammable FPGAs in future, if supported by good specification of interfaces will allow in-flight hardware reconfiguration and upgrade.

3.3.6.1.6.2 Board level and backplane level bus (as VME, PCI or PCI express [RD36], Spacewire)

Will allow the use (widespread now in commercial computers) of modular electronic boards. Once the transition between parallel busses to high-speed serial busses will be complete module-level cross-strapping will also become easier.

3.3.6.1.6.3 Very low speed bus (e.g. I2C or 1-wire)

The use of these common solutions will allow considerable mass saving on simple sensors and transducers. A very low speed bus (10 kbit/s-500 kbit/s) may then be interesting to implement a low-power sensor bus. For example, the OneWire standard [RD31] supports 16 kbit/s and 142 kbit/s communication. It relies on a single pair of wires for both power supply and communication.

3.3.6.1.6.4 Low-speed Bus

Low-speed busses are the ones commonly known as command and control busses. MIL-STD-1553B has been the most successful one, even despite the standard lack of implementation of any networking capability. CAN has recently gained the attention of space community after having became a de-facto standard for automotive industry.

3.3.6.1.6.5 High-speed Bus

Spacewire is the ideal candidate for a point-to-point data connection requiring 100 s of MBps, but the development of an Ethernet-style on-board bus is still a task to be performed. Those kinds of busses, like MIL-1773, shall have the possibility to be physical media independent, and to be put both on optical fibres or copper wirings.

3.3.6.1.6.6 Very high speed bus

Following the increase of data rates on on-board data busses in the 2020s a multi-Gigabit optical on-board connection shall be also available. Note that concurrent operation of shared computing power means heavy network load, and solutions like that will be probably necessary.

3.3.6.1.6.7 Wireless communication

Wireless communication will be the preferred mean to connect dynamically payloads or PDAs to the integrated avionics system. The standards already available (like IEEE 802.11a/b/g, Bluetooth and IEEE 802.15.4) would be enough for the needs of HMM systems.

3.3.6.1.7 Request for enhanced capabilities

A scenario imagining an integrated modular avionics for a HMM can already be quite realistic. Integrated modular avionics, with several small intelligent units concurrently running distributed software acting as a control entity of the craft, is the only way to cope with the reliability requests (together with mass power and cost savings) of such a mission and will provide a new

set of capabilities for the command & control, the data handling and the human interfaces. Given what is already available for guidance and control for military systems (like combat helicopters) or commercial airliners, it is clear that more and more controls shall be managed in complete on-board automation.
In everyday life, human beings prone to accept an involvement of automated machine intelligence in many potentially life-endangering tasks. Robotic surgery and automated driving are examples. A 3-5 year mission of a spacecraft far more complex than the actual ISS with no practical escape solution for the crew requires the acceptance of a system where the on-board computer (even if not identifiable anymore with a single box) has full control of the spacecraft health management and is the real mission control entity.
The on-board control system will also exploit capabilities like the on-board calculation of trajectories and manoeuvres, contingency management, FDIR, and all those autonomies requested by the long signal turnaround.
This approach will also save a great amount of workload for the crew (and for the ground control), during the mission and in the (presumably long) training phase.
Crewmembers shall be interfaced with the avionics by means of systems like PDAs or wrist computers and even the concept of a centralized command deck shall disappear. Human machine interfaces shall be weightless, the availability of powerful processors allows speech recognition and the use of touch screens is becoming common even in airliner cockpits.
The use of Personal Satellite Robot Assistants was already spotted (1998) as very useful on shuttle and ISS but its development by NASA-AMES is now suspended [RD35].

3.3.6.1.8 Reliable path-independent long distance multi-hop data transfer protocol

A key aspect of Martian exploration will be the ability of future missions to interoperate.

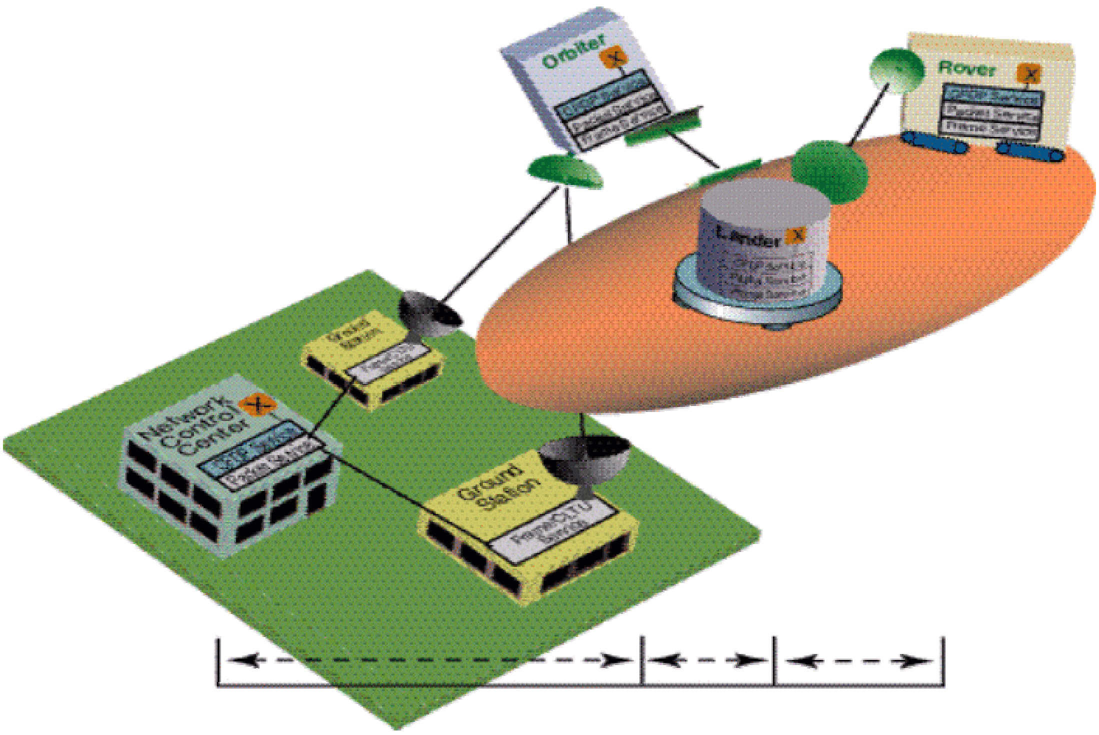


Figure 3-56: Interoperability scenario for a Mars mission.

These protocols establish a framework for interoperability by providing standard communication, navigation, and timing services. In addition, these services include strategies to recover gracefully from communication interruptions and interference while ensuring backward compatibility with previous missions from previous phases of exploration.
CFDP (CCSDS File Delivery Protocol) is a new international standard, built on the familiar Consultative Committee for Space Data Systems (CCSDS [RD40]) space data communication protocols, developed to meet a comprehensive set of deep-space file transfer requirements as stated by a number of space agencies including NASA, ESA, NASDA, CNES, and BNSC/DERA.
In addition, CFDP will serve as a prototype for the future Interplanetary Internet (IPN [RD42]) as envisioned by the IPN Study team: it encompasses a subset of the anticipated functionality of the IPN, and it implements several key IPN design concepts including store-and-forward operation with deferred transmission and concurrent transactions.

CFDP [RD28] allows an automatic, reliable file transfer between spacecraft and ground (in both directions) designed to support the operation of spacecraft by means of file transfer and remote file system management.
Its embedded transport layer provides applications the capability of transferring their data products end-to-end across the entire space link with two optional transmission modes: reliable or unreliable.
In reliable mode the data loss is automatically detected and retransmission of the lost data is performed automatically.
In unreliable mode, data are transferred in a “best effort” way over an unidirectional link.
Furthermore, CFDP provides four different selective retransmission strategies for negative acknowledgement. Its capabilities include:

- Reliable/Unreliable copying a file from the filestore of one entity (protocol engine, located in a spacecraft or ground control centre) to that of another entity
- Reliable/Unreliable transmission of arbitrary small messages, defined by the user, in the metadata accompanying a file
- Reliable/Unreliable transmission of file system management commands to be executed automatically at a remote entity – typically at a spacecraft – upon complete reception of a file
- Store-and-forward mechanism allowing an end-to-end transfers that can span multiple CFDP waypoint nodes in case source and destination entities are not in direct view.

CFDP is designed to offer these capabilities even across interplanetary distances, where data errors, data loss and out-of-sequence delivery may occur; minimising the return path overhead of the protocol for optimised performances. As such, it must function despite extremely long data propagation delays (measured in minutes or hours, rather than in milliseconds as in terrestrial networks) and frequent, lengthy interruptions in connectivity. Unlike TCP/IP, the transport layer embedded in CFDP requires no handshaking and is datagram-and transaction based to deal with space link characteristics (e.g. long RTLT and non-persistent links). Additionally, CFDP is adaptable to fit the proximity link as well. Metadata associated with each transaction describes the data transfer including data processing once the file arrives.

3.3.6.1.8.1 CFDP design concepts

It is expected that a pair of CFDP entities, which have files to exchange, may at any given moment be unable to communicate; for example, a spacecraft orbiting Mars may be on the far side of the planet, unable to transmit to Earth. Therefore, if transmission of a file from Earth to a Mars-orbiting spacecraft is interrupted when the spacecraft passes behind the planet, the CFDP entities at both ends of the transmission can react in two ways:

1. Storing their outbound protocol data units (PDUs) – possibly in non-volatile memory, to assure continued service even in the event of an unplanned system reset – until the spacecraft reemerges and transmission can resume. This approach does not affect CFDP transactions, which can continue to produce PDUs regardless of the link unavailability.
2. Freezing, at CFDP level, all the file transfer transaction related to the temporarily unavailable link and resuming it at the next communication opportunity. Both events, (freeze and resume), are triggered by an external module that holds knowledge of the link state.

A collateral benefit of this approach is that it largely insulates user applications from the state of the communication system: an instrument can record an observation in a file and "transmit" it (that is, submit it to CFDP for transmission) to Earth immediately without considering whether or not physical transmission is currently possible.
This deferred transmission mechanism, sequestering outbound data management and transmission-planning functions within CFDP, can enormously simplify flight and ground software and thereby reduce mission costs.
The large signal propagation delays that characterize interplanetary transmission limit the usefulness and efficiency of the retransmission strategies commonly used in terrestrial protocols (especially sliding-window ones). For this reason, CFDP's retransmission model is one of concurrent transmission: data PDUs for multiple files are transmitted as rapidly as possible, one after another, without waiting for acknowledgment, and requests for retransmission are handled asynchronously as they are received. As a result, portions of multiple files may be in transit concurrently.
CFDP is scalable, designed for reuse without modification in any number of communication environments as well. No specific direct interface to radio hardware, or even to any specific link-layer protocol, is mandated in the Recommendation. Instead, a minimal abstract underlying "unit data transfer" or UT layer service is assumed to be available for CFDP's use, enabling CFDP to be run on top of a wide range of services as UDP/IP on the Internet and the Packet services of CCSDS Telemetry [RD39] and Telecommand [RD40] in space.

3.3.6.1.8.2 CFDP outlook

CFDP is a stable international standard that can reduce mission operations cost and risk by enabling reliable file transfer and remote file system management over interplanetary distances. The application of emerging delay-tolerant networking technology to Interplanetary Internet operations, and specifically to the use of CFDP in complex mission configurations, will further enhance CFDP's usefulness and value to deep-space exploration missions.

3.3.6.2 Baseline design

3.3.6.2.1 Requirements and design drivers

The THM integrated avioncis has been evaluated taking into account all the function normally requested of a command and control module in a complex spacecraft.
Moreover, the data rate has been dimensioned assuming the need for almost continuous audio and video link with the ground station and the need of an internet-style connection.

3.3.6.2.2 Baseline design

The DCDMS (Distributed Control and Data Management Systems) is implemented as several distributed units called Control and Data Management Units (CDMU). Every unit exploits the following modular functions:

- processors modules (PM) each including the digital interfaces required to communicate with the platform units. Any number of processor can be powered at a given time, depending from the workload (a similar mechanism is used on multiprocessing computer farms and is totally managed by the application software)
- telemetry transfer frame generators (TFG), directly interfaced with trasponders.
- reconfiguration modules (RM), two of which are always powered, each containing a clock function and a reconfiguration function. A configuration mechanism allows choosing one of the modules as “master clock”. The other ones act as spares or as backup processors.
- distributed memory modules (DMM), each containing VRAM and NVRAM modules, the latter to be used as safeguard memory. A specific application checks the synchronization of the content of safeguard memories. The overall amount of available mass memory shall be of the order of several terabytes, even only for astronaut’s personal use.

3.3.6.2.3 Budgets

The DHS mass and power budget has been evaluated as shown in the following Table 3-39:

Property	Value
DHS+harness mass, inhabited modules	650 Kg
Percentage of DHS+harness mass, unmanned modules	5 %
Power, per module, manned	2200 W
Power, per module, unmanned	400 W

Table 3-39: DHS mass and power budgets

3.3.7 Communications

3.3.7.1 Requirements and design drivers

- The vehicle should support Tracking, Telemetry and Command (TT&C) communications during all mission phases and any attitude.
- Two-way ranging and Doppler capabilities are required during all mission phases.

- The maximum range that should be supported is 2.7 A.U. (max. distance Earth / Mars).
- The telecommand (TC) and telemetry (TM) data rates shall be selectable to improve the data rate depending on the distance.
- Data rates shall be optimised by giving realistic assumption of on-board equipment and ground segment availability.
- During all mission phases, data consists of housekeeping, high-quality audio and video channels, and any additional data (for example internet access).
- In August 2034 there is a solar superior conjunction, communications shall be provided during it.
- Minimum and average data rates requirements are shown in Table 3-40. The maximum data rate will be traded off with respect to complexity and cost, taking into account the expected technology development in the future.

	Uplink	Downlink
Maximum Data Rate (overall, kbps)	11280	9232
Average Data Rate (overall, kbps)	3484	1436
Minimum Data Rate (overall, kbps)	160	160

Table 3-40: Data rate requirements for TV

3.3.7.2 Assumptions and trade-offs

3.3.7.2.1 S-, X-, Ka-band and laser communications

The present situation of S-band (which is shared by Space Research (SR) Cat. A, Space Operation (SO) and Earth observation Services, plus high density mobile systems) is that high congestion and sharing difficulties with fixed systems have already appeared. Therefore S-band will be noisy. For this reasons, it is expected that ESA will reduce support to that band in long term.

Considering X-band versus S-band, the most favourable frequency of operation depends on the types of antenna used at both ends of the link (ground and space):

- Assuming constant apertures at both ends, the communication performance can be improved by a factor of 13.5 dB (theoretical) if the frequency of operations is increased from S- to X-band.
- Assuming constant aperture at the ground station and fixed antenna coverage on-board (e.g. communications via LGA), the communications performances of S- and X-bands are similar in clear sky conditions (atmospheric absorption and rain losses are higher in X-Band).

Considering Ka-band versus X-band, the following factors are important:

- Assuming constant apertures at both ends, the communication performance can be improved by a factor of 13 dB (theoretical) if the frequency of operations is increased from X- to Ka-band.
- The weather dependence of Ka-band is high, so the availability of the link is lower than in S- and X-bands.

- Higher pointing accuracy is required, compared with X-band and S-band. For example, 4 times more with respect to X-band and 16 times with respect to S-band.
- There is more bandwidth availability with respect to X-band, since nowadays very few spacecrafts are using Ka-band.
- During superior solar conjunction, Ka-band carrier suffers 15 % less amplitude scintillation (changes in frequencies) and 20 % less spectral broadening than X-band for the same Sun-Earth–S/C angle. Therefore, it is recommended since it will allow higher data rate than X-band. [RD44]

Considering laser communications versus Ka-band [RD50], the following factors are important:

- Higher transmission speed.
- Less technologically mature than Ka-band.
- Usable with reduced data rate, or not usable at all, for SEM (Sun-Earth-Mars) angles below 10 degrees. Therefore, it has less availability than Ka-band.
- More G/S availability since in case of clouds or rain, laser communications are not usable but Ka-band could work at a reduced data rate.
- Laser communications are difficult use for uplink from the G/S since they require a complex adaptative optics. Downlink adaptative optics are not mandatory, but if they are used, they are very different to the uplink ones. The conclusion is that the same telescope (G/S) cannot be used for both uplink and downlink.

3.3.7.2.2 Operations during first days of LEOP and contingency situations

It is assumed that, in these conditions, near omni-directional coverage is desirable and low gain antenna(s) will be used. Additionally, Earth orbit relay satellites working in X-band should be used.

The main advantage of using X-band is that LGAs based on waveguide (W/G) technology can handle power levels up to 100 W. However, typical S-Band LGAs (quadrifilar helix) can handle maximum 10 W.

Since in case of contingency, as much data rate as possible will be required, X-band would be better to obtain 100W of transmitted power. Taking into account the reference mission date (around 2025), it is considered that by that date X-band uplink and downlink capability will already be available in most stations.

3.3.7.2.3 Modulation for deep-space data transmission

To design the RF (Radio Frequency) link Mars-Earth, using the CCSDS recommendation [RD46], and because this mission is a deep-space mission (CCSDS category-B) with high data rate requirements (over 2 Mbps) two modulations have been considered; GMSK and T-OQPSK. For these missions, ESA has decided to only implement GMSK in the future; therefore it is the used one in this design. As recommended as well by CCSDS in [RD46], a coded GMSK with BTb = 0.5 has been chosen. Reasons for this election are the low Eb/N0 required and the equalization so that end-to-end losses for this modulation are in the order of 0.1 to 0.15 dB (i. e. insignificant).

3.3.7.2.4 Relay satellite

A trade-off ont he convenience of a Mars relay satellite for communications with Earth G/S is done in SHM report (see section 4.3.7). In brief, it is used to relay TV – MEV – G/S.

3.3.7.2.5 Communications during solar superior conjunction

On 19 August 2034, there will be a superior Earth-Sun-Mars conjunction that will affect the communications with Earth G/S. The minimum angular distance S-E-M will be 1.145 degrees, and in that situation, communications will be feasible but with a reduced data rate and only in Ka-band. laser link is not usable below 10 degrees of SEM angle, so for about 2 months, the mission will have to rely only on the Ka-band link, the TV-G/S or the relay satellite-G/S one. Therefore, during 2 months, the downlink data rate will be reduced (see section 3.3.7.5.4).

The effects of the solar conjunction on the received signal are amplitude scintillation, spectral broadening and phase scintillation due the fluctuating columnar electron density. Phase scintillation causes changes in the frequency of the signal, creating problems in ranging signal (orbit determination), and phase locking loop. The solution is to not plan anything critical during this time, such as orbit changes, and to increase the PLL bandwidth, respectively.

Using data obtained from Cassini mission [RD43], the study report concludes that in Ka-band a link degradation of 7 dB will be obtained. However, the real value of this degradation will depend directly from solar activity and solar transient events. Due to the solar maximum expected in 2033, close to the conjunction date, this figure should be taken cautiously.

Data rates during superior conjunction are shown in Figure 3-62.

3.3.7.3 Laser link

The maximum data rate achievable from the maximum distance Mars-Earth (2.7 AU) using only Ka-band and a reasonable antenna size, is less than 2 Mbps. For example, to obtain 9 Mbps for downlink, a Ka-band 8 metre antenna would be needed. Weight, complexity and cost make it an unrealistic option. The alternative is laser communications, which could increase the data rate to 10 Mbps at 2.7 AU, and even 250 Mbps at 0.7 AU. Despite the fact that these values are not feasible nowadays, future technology should be able to support them.

In a laser link, one of the main problems is the atmospheric distortion. To reduce the influence of the atmosphere, adaptative optics (AO) could be used in the G/S (basically optical telescopes). For the uplink as well as for the downlink, AO would significantly improve the link performances. Especially for uplink, AO are considered necessary. To limit the complexity and hence the cost of the optical G/S, and considering that nowadays much more work goes into downlink laser communications research and development, it has been decided to use laser only for downlink, while Ka-band will be used for uplink.

The technical problems to be solved to make the optical link from Mars competitive respect RF link are:

1 High power space qualified lasers of at least 5 W of transmitted power. Nowadays 1.5 W.
2 Reduce pulse length to 2 ns with high power lasers (at least 5W).
3 Adaptative optics in the G/S, to reduce atmospheric distortion.
4 System mass (around 45 Kg). It is not a serious constraint for this mission.

5 Reduce 2dB pointing precision to 2µrad.
6 Problems when angular distance between Sun- G/S – Spacecraft is low.

The development of space-qualified cavity dump lasers [RD51] is expected to solve the two first problems. As regards the third problem, adaptative optics for downlink in G/S are expensive, but are considered state of the art.

One of the main problems for laser links is the high pointing accuracy needed (around 2 µrad in this case). The approach is based on using high bandwidth inertial sensors to compensate for jitter excursions caused by spacecraft vibrations. This use of high bandwidth inertial sensors, together with a fine pointing mirror (see Figure 3-57), would enable the implementation of laser communication links by achieving submicro radian-pointing precision [RD45]. As a reference, ESA-Artemis is obtaining 3 µrad of pointing accuracy.

In this report, a minimum SEM angle of 10° for laser communications feasibility was taken, which is a conservative value. A lower minimum SEM angle (3°) is considered in [RD47].

Two laser transmitter options have been considered, the first is a 30.5 cm telescope transmitting 5W, and the second is a 50cm telescope transmitting 20W. See Table 3-41 for details.

The 30.5 cm telescope with a 5W laser is enough from the point of view of data rate (it covers the average downlink data rate shown in Table 3-40) and the pointing precision needed is lower. Nowadays, a maximum power of approximately 1.5W is achievable, 5W is a more realistic option than the 20W. Therefore, the 30.5 cm and 5W option has been selected.

Mirror diameter	Transmitted power	Expected data rate 2.7 A.U. - 0.7 A.U.	Pointing accuracy (3dB beamwidth)
30.5cm	5W	10 Mbps-250Mbps	6 µrad.
50cm	20W	100 Mbps-1Gbps	2.5 µrad

Table 3-41: Laser transmitter considered options.

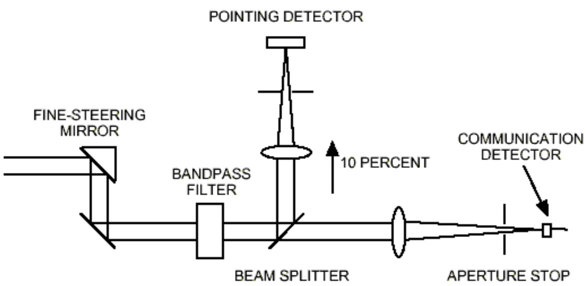


Figure 3-57: Pointing detector system. It is used together an inertial sensor.

3.3.7.4 Communications continuity

An analysis of communications during solar superior conjunction is done in section 3.3.7.2.5

The TV communications direct link availabilities, considering just the direct visibility in the worst case, with the different mission elements are:

- *G/S*: maximum blackout 41 min= 65 % availability.
- *MEV*: average visibility 15 min = 12 % total time.
- *Relay*: availability around 65 % of the time.

From relay satellite to the G/S, the maximum blackout duration is 77.5 minutes, so 95% of total time, the relay will have contact with G/S. The exact figures will depend on the season.

To increase the communications availability between MEV and TV, apart from a direct link, the solution is to use a relay satellite. With this solution, the availability will be around 65% of total time in the worst case.
In LEO (400 km, 62 degrees), each of the three ground stations will have three or a maximum of five passes per day with a maximum duration of around 8 minutes. However, in LEO data relay satellites should be used.

3.3.7.5 Baseline design

3.3.7.5.1 Band and frequency design

Laser communications has been used only for downlink while Ka-band has been used for uplink. The reason is a compromise between cost and data rate. In case of using laser communications for uplink as well, just the double of G/S would be needed in principle, so it is a too expensive option. Additionally, Ka-band data rates are higher for uplink than for downlink, mainly the higher transmitted power by the G/S.

For contingencies, X-band has been used because it has less weather dependence than Ka-band, so in case of contingency the link availability will be higher.

For the TV-relay satellite link, X-band has been choosen, since the pointing requirement is lower than Ka-band and with an small dish antenna diameter (40 cm) a high enough data rate, for the project needs, is achieved.

For the TV-MEV/MAV/SHM link, UHF is used because its low propagation losses and very low pointing requirements, make it ideal for docking-undocking manoeuvres.

The bands and frequencies used are consistent with the Space Frequency Coordination Group [RD48].

3.3.7.5.2 Ground station assumptions

Ground stations with Ka- and X-band capability and 70 m of diameter are used. See Table 3-40.

	Transmission		*Reception*	
	Frequency band	**EIRP**	**Frequency band**	**Effective G/T, 10º**
70-m antenna	7145 – 7190 MHz	89.31 dBW (1995W RF)	8400 - 8450 MHz	42.52 dB/K
70-m antenna	34200 –34700 MHz	114.69 dBW (794W RF)	31800 – 32300 MHz	56.71 dB/K

Table 3-42: Assumed ground station characteristics

3.3.7.5.3 TV EVAs

In assembly or contingency situations, EVAs could be necessary for TV. Two sets of external double (redundant) antennas have been used in TV design. The system will be able to transmit voice, biomedical data and receive voice data. The voice can come from TV or other astronaut. An ISS like antenna system has been used (see Table 3-47).

3.3.7.5.4 Links description

TV will have five different communications links with the other mission elements:

1. Ka-band link with G/S, using a 3 m-dish HGA antenna.
2. X-band link with G/S using two MGA antennas, for contingencies
3. UHF link: for communications with SHM, DM and MAV. An UHF patch antenna is used.
4. X-band link with the relay satellite, for communications with SHM and with Earth. A 0.45m dish HGA antenna is used.
5. Laser link with G/S. Only used for downlink.

Note that communications with the Earth will have a roundtrip delay of 40 minutes, in the worst case (distance TV-Mars of 2.7 AU). No real time communications are possible.

Link	**Ka-band[1]**		**X-band**		**UHF[2]**		**Laser**	**X-band**	
	Uplink	**Downlink**	**Uplink**	**Downlink**	**Uplink**	**Downlink**		**Uplink**	**Downlink**
Frequency	34.5 GHz	32 GHz	7.15 GHz	8.42 GHz	437.1 MHz	401.6 MHz	Wavelength =1064 nm	7.2 GHz	8.45 GHz
Tx power	794 W	65 W	19953 W	65 W	5 W	5 W	5 W	65 W	65 W
Modulation	NRZ/PSK/PM	GMSK. BTb=0.5	NRZ/PSK/PM	GMSK BTb=0.5	PCM-NRZ/BPSK	PCM-NRZ/BPSK	256-PPM	QPSK	QPSK
Coding	Turbo Coding ¼	Concatenated: Convolutional + RS (255, 223)	Turbo Coding ¼	Concatenated: Convolutional + RS (255, 223)	Convolutional, rate ½	Convolutional, rate ½	Reed Solomon (26143, 15685)[3]	Concatenated, Interleaving=5	Concatenated, Interleaving= 5
BER	Negligible	Negligible	BER=10^{-6}	BER=10^{-6}	10^{-6}	10^{-6}	BER=10^{-6}	FER=10^{-5}	FER=10^{-5}
Bit rate (worst case)	1.76 Mbps	1.5 Mbps	22.6 kbps	460 bps	128 kbps	128 kbps	10 Mbps	20 Mbps	20 Mbps

Table 3-43: Links description

[1] Atmospheric attenuation of 4.34dB, it corresponds with a minimum elevation of 10deg and G/S availability of 90%.
[2] Max distance 1070Km corresponding to a TV elevation of 10deg.
[3] See [[RD52]]

RECEIVER CHARACTERISTICS (X-BAND)	
Carrier acquisition threshold	-153 dBm
Carrier tracking Threshold	-156 dBm
Carrier Loop bandwidth ($2B_L$ in a C/N=10dB)	1000 Hz
Noise Figure	1.6 dB

Table 3-44: X-band transponder. Receiver characteristics

RECEIVER CHARACTERISTICS (Ka-BAND)	
Carrier acquisition threshold	-153 dBm
Carrier tracking Threshold	-156 dBm
Carrier Loop bandwidth ($2B_L$ in a C/N=10dB)	100 Hz
Noise Figure	1.6 dB
Ranging Bandwidth (double sided)	3 MHz

Table 3-45: Ka-band transponder. Receiver characteristics

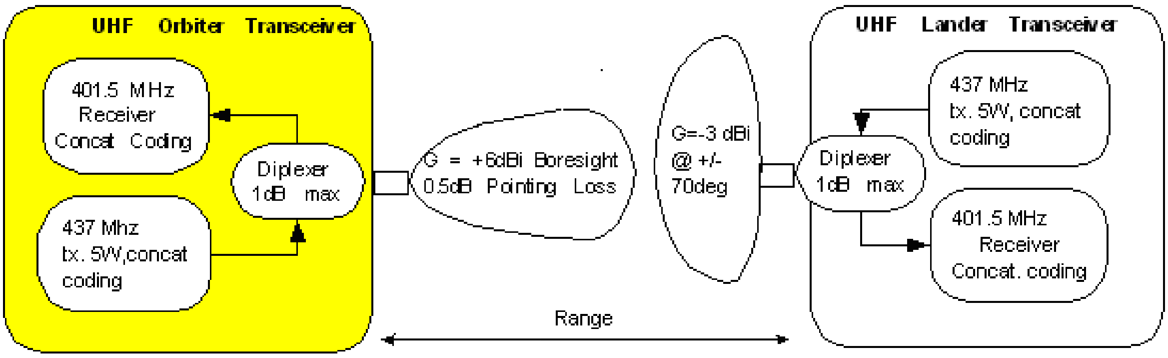


Figure 3-58: UHF TV-SHM link

The communications links are shown Figure 3-59 to Figure 3-62, under different modes (standard and contingency), distances and mission phases (cruise, surface operations, rendezvous and docking and orbiting around Mars). Three different bands are used: Ka-, X- and UHF-band. Additionally, a laser link is used.

To calculate the data rates, the distance variation depending on the mission phase has been taken into account. The maximum distance Earth-TV during cruise to Mars is 0.92 AU. The maximum and minimum distances while the MEV is on the Martian surface are 1.37 and 1.1 AU respectively. During the whole mission, the maximum distance will be 2.7 AU and will happen during the solar superior conjunction.

Data rates of laser and RF Ka-band links during the different SEM (Sun-Earth-Mars angle) of the superior solar conjunction are shown in Table 3-46.

Angular separation	Duration (days)	TV Ka-band (3 m antenna) and laser link	Relay satellite link (4 m antenna) Ka-band
SEM > 10º		•Laser communications: 10 Mbps. •RF: From G/S -> TV : 1.76 Mbps From TV -> G/S: *1.5* Mbps	•RF: From G/S -> RS : 4.1 Mbps From RS -> G/S: 3.2 Mbps
SEM < 10º	60	•Laser communications difficult. Reduced data rate. •RF: like for SEM > 10º	
2º < SEM < 3º	17	•Laser link not possible. •RF: like for SEM > 10º	
SEM < 2°	11	•No laser link. •RF: reduced rate. Worst case SEM=1.145º: From G/S -> TV : 176 Kbps From TV -> G/S: *300* Kbps	•RF: From G/S -> RS : 264 *Kbps* From RS -> G/S: 550 *Kbps*

Table 3-46: Data rates of RF Ka-band and laser links at different angular separations

In the figures below it is shown the different links for the different phases, data rates in bps.

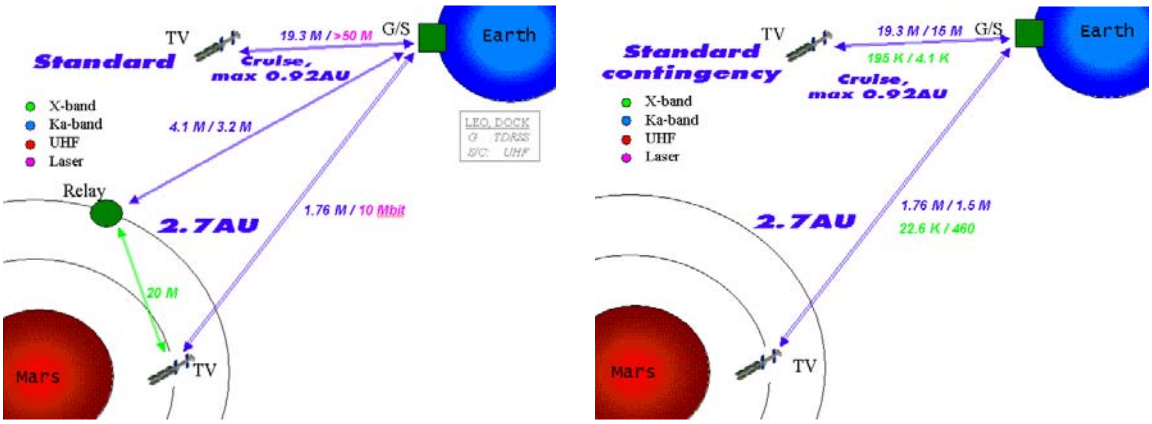


Figure 3-59: Communications links during cruise and Mars orbit phases worst cases

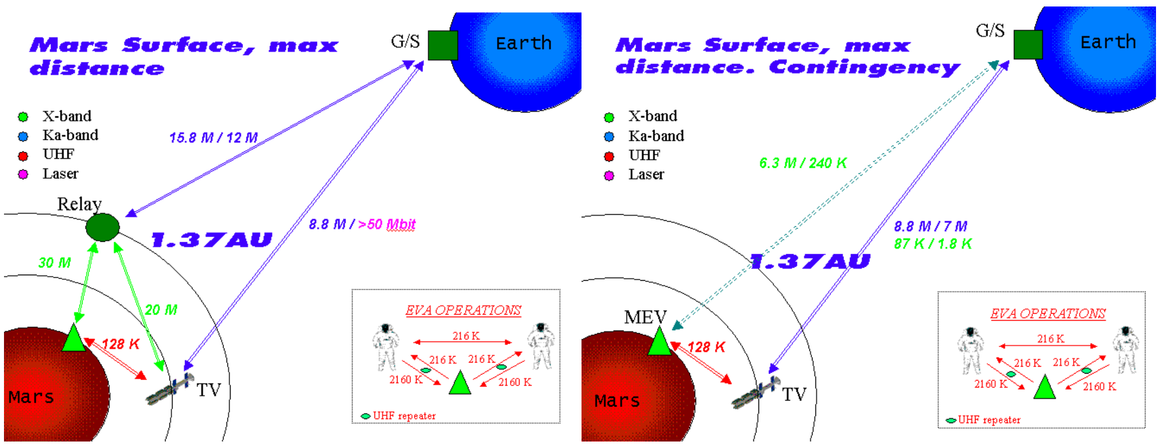


Figure 3-60: Communications links after MEV landing. Normal and contingency cases

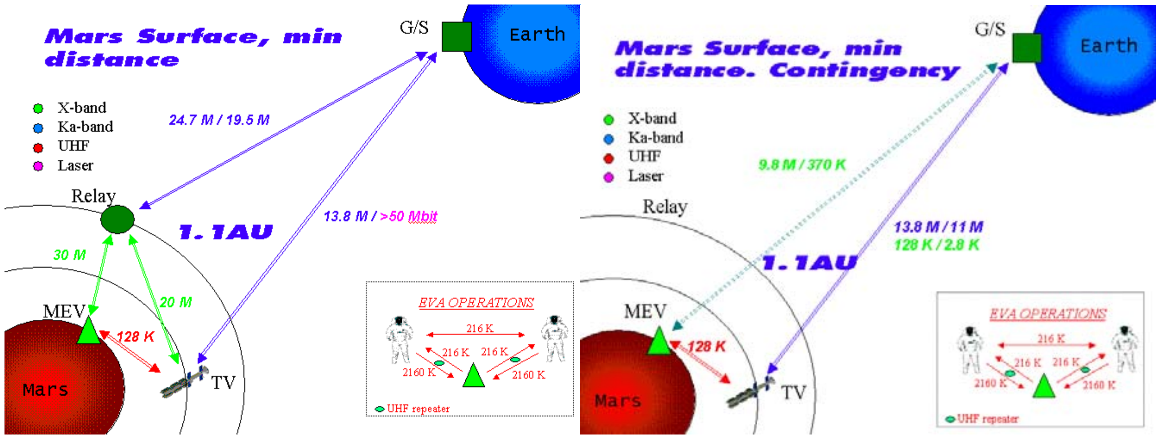


Figure 3-61: Communications links before MAV take off

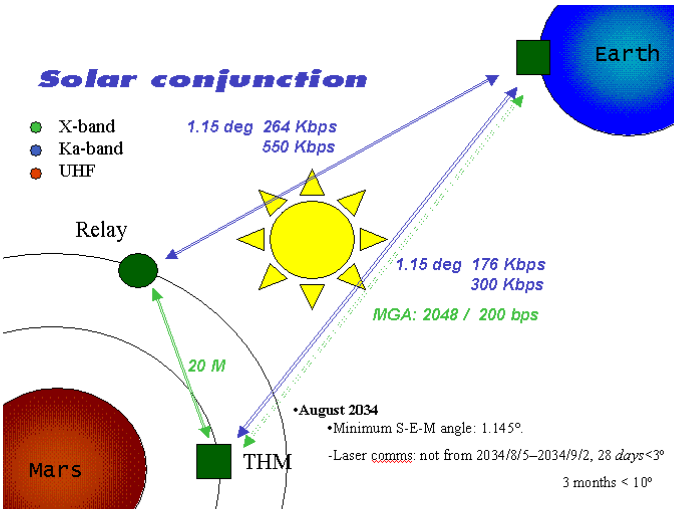


Figure 3-62: Communications during solar superior conjunction

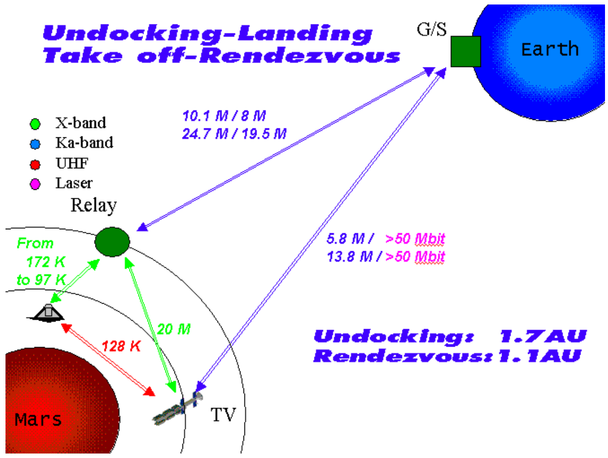


Figure 3-63: Communications MEV/MAV-TV during take off – rendezvous and undocking-landing

3.3.7.5.5 TV contingency communications

When laser downlink cannot be used, for example during the superior solar conjunction or during a contingency, Ka-band 3 m antenna will be used instead. Once in Martian orbit, the relay satellite could be used as well using the TV 40 cm X-band antenna, and would provide higher data rate than the 3m TV antenna with the G/S due to its 4 m antenna. The problem is that there is no continuous visibility, so it will be possible to use it during approximately 65% of the time.

In case of a contingency with loss of attitude, in which it is not possible to use any HGA, a communications link has been designed by using two MGAs with an intelligent steering mechanism that will be pointing one of the antennas to the Earth.

See Figure 3-59 for a summary of communications with the G/S during contingency.

3.3.7.5.6 Budgets

3.3.7.5.6.1 TV Antenna summary

In total, three booms are necessary for telescope, Ka-band antenna and X-band antenna.

Kind of antenna	Quantity	Band	Gain	Minimum required pointing precision	Size	Radiated power	Data rate Uplink	Data rate Downlink	Steering mechanism	Commentaries
Telescope	1	Optica l		2µrad	30.5 cm	5 W	No uplink	10 Mbps	180° hemispherical	LASER link, only used for downlink
Dish antenna	1	Ka-Band	59.1 dBi	0.01 deg	3 m	65 W	1.8 Mbps	1.5 Mbps	180° hemispherical	Main link.
MGA Patch	2	X-band	18 dBi	20 deg	8.2 x 8.2 x 2 cm	65 W	22 Kbps	460 bps	180° hemispherical	Intelligence to point the Earth in a contingency case, even with loss of TV attitude
Dish antenna	1	X-band	30 dBi	2.25 deg	45 cm	65 W	30 Mbps	30 Mbps	180° hemispherical	Link with relay satellite
Patch antenna	1	UHF	6 dBi	40 deg	35 x 35 cm	10 W	128 Kbps	128 kbps	180° hemispherical	UHF link with MAV/MEV
Wire antenna	4	UHF/ VHF	-3 dBi	Omnidirectio nal	20 x 10 x 10 cm	5 W	2048 Kbps	2048 kbps	None	For TV EVAs

Table 3-47: TV antennas summary

Unit	*Number of units*	*Unit mass (kg)*	*Total Mass (kg)*	*Power (W)*
Optical transmitter	2	20.0	40	150.0
Optical transmitter device (telescope)	1	25.0	25	
UHF antenna system (EVA)	4	2.0	8	
Ka-band transponder	2	6.5	13	160.0
Ka-band antenna (3m)	1	35.3	35.3	
X-band transponder	2	6.5	13	100.0
MGA (X-band), patch	2	0.6	1.2	
UHF patch antenna	1	1.0	1	
UHF transceiver	2	2.5	5	16.5
X-band dish antenna (0.45 m)	1	1.0	1	
Harness			21	
Total:			163.5	370

Figure 3-64: TV communications budget summary

3.3.7.6 Options

3.3.7.6.1 Ka+ band

One of the most straightforward ways of improving link capacity is moving to higher frequency bands. The 40 GHz up and 37 GHz down band (Ka+/Ka+) was allocated by ITU for the very purpose of human space exploration. Contrary to the 34 GHz up / 32 GHz down (Ka/Ka), the Ka+ band can be used for both deep-space (Mars) and near-Earth (Moon) missions, while Ka-band can not be used from the Moon, but from Mars. To use the same frequencies for all human missions, Ka+ band is the best option. The problems of this band are firstly that Ka+ band is new to space activities and no technology development has been performed so far. Secondly, atmospheric and rain attenuation is higher than for Ka-band in approximately 3 dB, so the improvement of gain due to the higher frequency is cancelled by the higher attenuation.

3.3.7.6.2 Laser link coding

Turbo code 4 st. for laser link could be used. A higher data rate with respect to the option was chosen in this design, Reed Solomon code, would be obtained. The problem is that the high bit rate to support a net data rate needed after coding is four times higher than before coding. Technology should be prepared for those rates, especially for short distances TV-Earth where from the link budget point of view, high net data rates could be achieved (for example 250 Mbps at 0.7 AU). See [RD54].

3.3.8 Mechanisms

3.3.8.1 Requirements and design drivers

The HMM requirements do not state any specific requirements applicable to the TV & Propulsion module Mechanisms. As a result of the TV's configuration, the following necessary mechanism and their requirements can be derived:

Transfer Habitation Module

- Power Generation System:
    - Deployable Solar Arrays to provide up to 380 m^2 area.
    - Sun-tracking ability, unlimited rotation (driven by Martian orbit)
    - Survivability of deployed array during propulsion manoeuvres
    - Potential for Re-stowage and Latching capability- this is dependent upon the loading introduced by propulsive manoeuvres
- Communication System:
    - Antenna Pointing and Tracking Mechanism Ka-band Antenna:
        - Antenna Diameter: 3 m
        - Antenna Mass: 35 kg Est
        - Coverage: 180° Hemispherical.
        - Pointing Accuracy: 0.001°
    - Antenna Pointing and Tracking Mechanism laser Communications Link:
        - Antenna Size 0.4x0.5x0.4 m (LxWxH)
        - Antenna Mass: 25 Kg Est
        - Coverage: 180° Hemispherical.
        - Pointing Accuracy: 2 μrad°.
    - Antenna Pointing Mechanism Aero-stationary satellite communications:
        - Antenna Dia: 0.5 m.
        - Antenna Mass: 5 Kg Est
        - Coverage: 180° Hemispherical.
        - Pointing Accuracy: 2.°.
    - Deployment capability for 2x 3 m booms and 1x 0.5 m boom.
    - Survivability of deployed array during propulsion manoeuvres
    - Potential for Re-stowage and Latching capability- if deployable solution selected and dependent upon the loading introduced by propulsive manoeuvres.

- Vehicle Connections:
  - Berthing & Docking Capability:
    - Earth Return Capsule:
      - Berthing & Docking in LEO
      - Un-docking during TEI
    - Martian Decent Module:
      - Berthing & Docking in LEO
      - Un-docking during Martian orbit
    - Mars Ascent Vehicle:
      - Berthing & Docking in Martian orbit
      - Un-docking during Martian orbit
  - Berthing Capability:
    - All Modules not requiring an un-docking mechanism capability will require a berthing ability for LEO assembly operations.
- Crew Egress Hatches:
  - External Hatches and Locking Mechanism at the ERC and DM Docking Ports
- Crew Conditioning and Exercise Facilities:
  - Crew Exercise Device/Facility
  - Crew Short Arm centrifuge for 1g environment simulation

3.3.8.1.1 Propulsion module

- Vehicle Connections
  - Berthing Capability
    - All support structure elements will require a berthing ability for LEO assembly operations.
    - All propulsion stacks will require a berthing ability.
  - Release Capability
    - After use of individual stages, the stage must be released at the support structure I/F.

3.3.8.2 Assumptions and trade-offs

3.3.8.2.1 Power generation system

The following assumptions have been derived as a result of the study:

- The number of arrays shall be minimised
- The technology chosen shall ensure that stowage of the array is possible

The following Solar Array Deployment systems are available:

1. Advanced rigid arrays
2. Polar platform arrays

The following highlights key features of the two concepts:

Advanced rigid arrays have the following features:

- Typically four or five panel wings with surface area of 30-35 m^2 (typical panel size 2.5 m x 2.75 m).
- Spring-driven, single-direction deployment, with latched panels for in-flight wing stiffness
- No Re-stowed latching capability

Polar platform arrays have the following features:

- Up to 16 panel capability with surface area of 80 m^2 (typical panel size 5 m x 1 m, current qualification status of 14 panels).
- Motorised, cable actuated deployment with re-stowage capability (not yet qualified).
- No Re-stowed latching capability

The Polar platform type array would better suit this application given the number of. Additionally, the requirement for re-stowage is better facilitated with a motorized deployment system.

3.3.8.2.2 Communications system

The following assumptions have been derived as a result of the study:

- All boom-mounted antennas require tracking capability.
- Tracking can be realized with two perpendicular rotational axes.

No trade-off has been performed. The choice of the chosen mechanism has been made based upon the available systems and the requirements stated earlier.

3.3.8.2.3 Vehicle connections

The following assumptions have been applied:

- Only the DM and ERC require berthing and docking ports
- Further Module assembly in LEO will be performed by Berthing Mechanisms, aided by LEO facilities i.e. robotic arm capture of module and assisted berthing
- A spare berthing and docking port is required for in-orbit contingency

Two systems have been considered:

1 Russian Docking System
2 International Docking & Berthing Mechanism (IDBM)

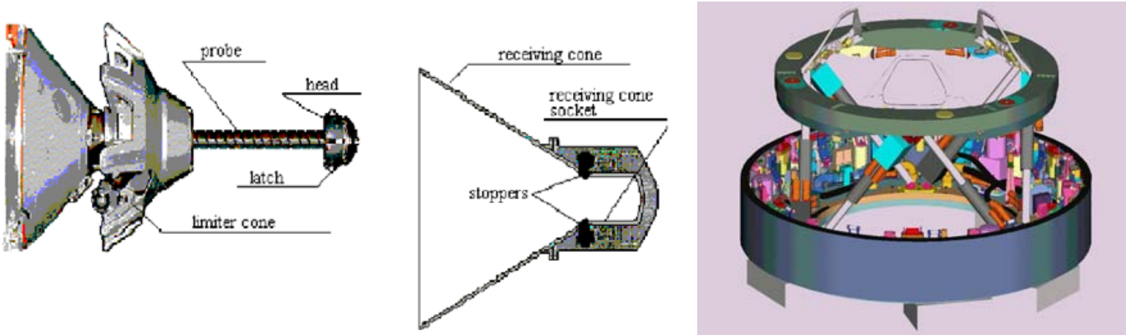


Figure 3-65: Russian Docking System & International Berthing & Docking Mechanism

Russian Docking System has the following features:

- Both half of the system are integrated in to the hatch doors
- 'Male' half situated on approaching vehicle (MAV)
- Receptacle situated on the TV
- Internal Redundancy of systems
- System level redundancy not complete-redundant receptacle can be provided, but no redundant probe can be mounted on MAV.

International Berthing & Docking Mechanism (IBDM) has the following features:

- Androgynous system- identical mechanism mounted to both vehicles
- Full redundancy of system provided
- Full Internal Mechanism redundancy
- Treble redundancy for release/emergency release
- Mechanism independent of hatch door
- Hatch door diameter limited to ingress/egress suitability (diameter 813 mm)

3.3.8.2.4 Crew conditioning and exercise facilities

The following assumptions have been applied:

- A redundant crew exercise facility shall be provided including medical (cardio-vascular) monitoring
- A Short Arm Centrifuge (SRC) shall be provided:
  - Due to the large rotational momentum, a counter rotation device shall be implemented.
  - The SRC shall have two astronaut positions
  - Urgent/Emergency stop <20 seconds

3.3.8.3 Baseline design

3.3.8.3.1 Power generation system

The baseline system shall be the polar platform type of solar array.

- Four solar array wings shall be mounted on to the TV. Two wings shall be mounted to each service node.

Improvements to the arrays are made as follows:

- Increase of panel length to 6.33 m; Individual panel size 6.33 m x 1 m
- 15 active panels per wing; wing requires 16 panels for deployment system to function correctly, panel no. 1 to be spacing panel only.
- Inclusion of the retraction/re-stowage capability; function already available, but not qualified.
- Inclusion of a re-latching and release capability in the stowed configuration.

An analysis of the loading applied to the solar array as a result of the propulsive manoeuvres shows that the deployed wing is unable to withstand the loading (see Table 3-48) and therefore a re-stowage capability is required.

Solar Array Wing Mass			
Deployment System (Kg)	Panel/Cells (Kg)	Total (Kg)	
45.7	149.6	195.3	
CoM position	8	m	
Allowable Shear	25	N	(Derived from Bending allowable, not actual shear)
Allowable Bending	185	Nm	

	Mass. Vehicle (Kg)	Thrust (N)	Acceleration ($^{m}/_{s}$)	SA Force applied at CoG/Shear Load	SA Bending Moment	Margin of Safety Bending
TMI. 1st Start	1363000	5200000	3.81511372	745.0917095	5960.733676	-0.968963552
TMI. 1st End	1084000	5200000	4.79704797	936.8634686	7494.907749	-0.975316574
TMI. 2nd Start	1045000	5200000	4.976076555	971.8277512	7774.62201	-0.976204631
TMI. 2nd End	766000	5200000	6.788511749	1325.796345	10606.37076	-0.982557653
TMI. 3rd Start	728000	3900000	5.357142857	1046.25	8370	-0.977897252
TMI. 3rd End	518000	3900000	7.528957529	1470.405405	11763.24324	-0.984273045
MOI. 1st Start	474000	1530000	3.227848101	630.3987342	5043.189873	-0.963316868
MOI 1st End	293000	1530000	5.221843003	1019.825939	8158.607509	-0.977324562
MOI. 2nd Start	277000	612000	2.209386282	431.4931408	3451.945126	-0.946407028
MOI. 2nd End	205000	612000	2.985365854	583.0419512	4664.33561	-0.960337331
TEI. Start	165000	612000	3.709090909	724.3854545	5795.083636	-0.968076388
TEI. End	82000	612000	7.463414634	1457.604878	11660.83902	-0.984134932

Table 3-48: Solar array wing loads

3.3.8.3.2 Communications system

A number of antenna pointing mechanisms exist. A schematic of a typical boom mounted deployment and pointing mechanism for the TV antennas.

For a complete 180° hemispherical coverage, the axial and azimuth axes shall rotate through 180°.

An analysis of the loading applied to the antenna boom as a result of the propulsive manoeuvres shows that the 3-metre deployed boom assemblies are highly loaded in bending (the analysis results are shown in Table 3-49). Three solutions are possible;

1. Rigidly mount the antenna and APM on a truss structure- no deployment, pre-installed during LEO assembly
2. Static or deployable boom with the deployment axis orientated along the load vector (large bending moment occurs across the hinge)
3. Deployable boom with a re-stowage and latching capability

	Ka-Band Antenna Force applied at CoG/Shear Load (N)	Ka-Band Bending Moment (Nm)	Laser Antenna Force applied at CoG/Shear Load (N)	Laser Bending Moment (Nm)	Aero-Stationary Antenna Force applied at CoG/Shear Load (N)	Aero-Stationary Antenna Bending Moment (Nm)
TMI. 1st Start	165.9574468	497.8723404	127.8063096	383.4189288	7.630227439	3.81511372
TMI. 1st End	208.6715867	626.0147601	160.701107	482.103321	9.594095941	4.79704797
TMI. 2nd Start	216.4593301	649.3779904	166.6985646	500.0956938	9.95215311	4.976076555
TMI. 2nd End	295.3002611	885.9007833	227.4151436	682.2454308	13.5770235	6.788511749
TMI. 3rd Start	233.0357143	699.1071429	179.4642857	538.3928571	10.71428571	5.357142857
TMI. 3rd End	327.5096525	982.5289575	252.2200772	756.6602317	15.05791506	7.528957529
MOI. 1st Start	140.4113924	421.2341772	108.1329114	324.3987342	6.455696203	3.227848101
MOI 1st End	227.1501706	681.4505119	174.9317406	524.7952218	10.44368601	5.221843003
MOI. 2nd Start	96.10830325	288.3249097	74.01444043	222.0433213	4.418772563	2.209386282
MOI. 2nd End	129.8634146	389.5902439	100.0097561	300.0292683	5.970731707	2.985365854
TEI. Start	161.3454545	484.0363636	124.2545455	372.7636364	7.418181818	3.709090909
TEI. End	324.6585366	973.9756098	250.0243902	750.0731707	14.92682927	7.463414634

Table 3-49: Antenna loads

The second selection is the least helpful because this will likely lead to a large and massive hinge. Additionally, the required orientation of the hinge will lead to a difficult stowed configuration.
The third solution will add the requirement for a deployment hinge. As an indication of size and mass, the Envisat DRS boom antenna deployment system had a mass of 28 kg for a 26 kg payload mass. An additional 10 kg is added for the restraint and release system. It should also be noted that this system was a spring-driven hinge.
The presence of an additional hinge in the support structure and also the rather long boom required to support the payload will lead to a reduction in the pointing accuracy/stability of the system. Therefore, to minimise these effects and to optimise the support structure required mass, The first solution was chosen as is the preferred mounting option.

Current APM systems are able to meet the pointing requirements for the laser-bench and aero-stationary antenna. However, although an APM exists that can carry the Ka-band antenna mass, the pointing accuracy will require improvement.

3.3.8.3.3 Vehicle connections

The IDBM shall be implemented for the TV/ERC and TV/DM Interface.

The IDBM has the following mechanical characteristics:

- Interface loads (at the sealing interface)- acting simultaneously while docked (Flight-limit)
  - Axial load (1200 lbf) 5338 N
  - Shear load (1000 lbf) 4448 N
  - Bending moment (80000 in*lbf) 9039 Nm
  - Torsion moment (70000 in*lbf) 7909 N*m
- Internal Pressure (16 psi) 110316.1 Pa
- Life 15 years, Functional Life 20 Berthing/un-berthing or Docking/undocking cycles

It is likely that the loading across the I/F induced by the DM and ERC will exceed the allowable loads above. Therefore, for all phases of the mission prior to I/F separation and additional a support (truss) structure will be required. This will be released prior to vehicle separation. Each individual IBDM is supported by eight electronic boxes.

For all berthing I/Fs, the common berthing mechanism shall be implemented. The following I/F are affected:

- Node-to-TV volume
- Cupola-to-Node
- Node-to-TEI backbone

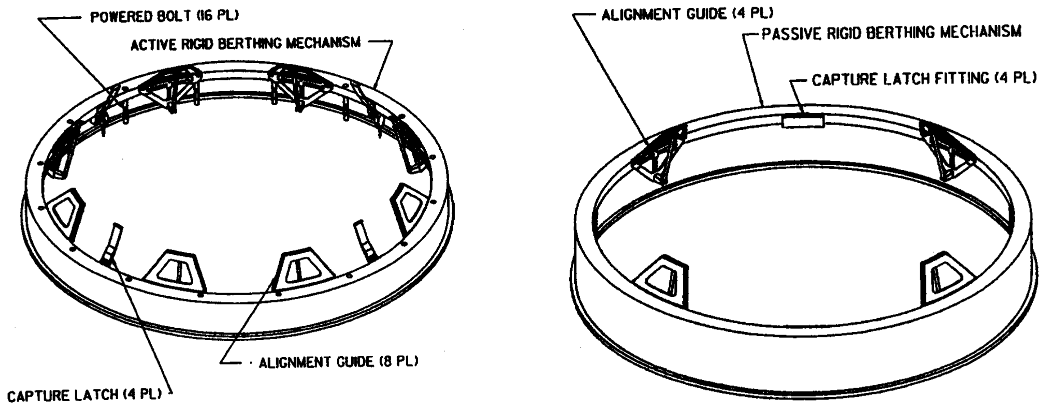


Figure 3-66: Active and passive common berthing mechanisms

For release of individual propulsion stages, a clamp-band with ejection springs shall be the baseline.

3.3.8.3.4 Crew conditioning and exercise facilities

A Flywheel Exercise Device (FWED), shall provide the general crew exercise facility. The FWED provides a crew monitoring function and also is collapsible for storage in a suitable container.

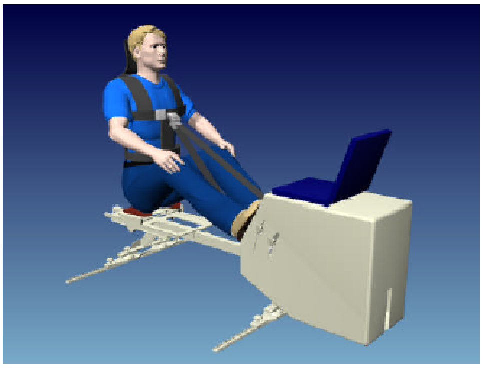


Figure 3-67: Flywheel exercise device

A short arm centrifuge of diameter 4.5 m shall be implemented. The device shall have two stations allowing the astronaut to lay on a bench with the head positioned close to the rotation point and the feet at the outer radius. In this way, the required 'g' environment is experienced at the feet with a decreasing 'g'-gradient towards the head. Table 3-50 and Figure 3-68 show the required rotation speed for a given 'g'-load at the feet and also the resulting rotational momentum

g-load	RPM	Min. Spin up time (sec)	Max. Spin-up torque [Nm]	Momentum [Nm.s]
0.38	13.03715	7.5	179.9270925	1349.45319
1	21.14905	20	109.4551074	2189.10215
2	29.90927	40	77.39644869	3095.85795
3	36.63123	60	63.19393573	3791.63614
4	42.2981	80	54.72755371	4378.2043

Table 3-50: Required rotation speed for a given g-load

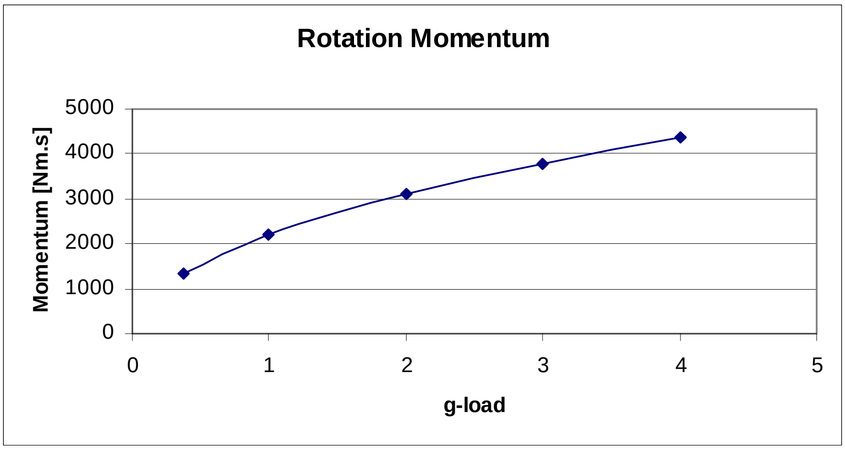


Figure 3-68: Rotation momentum

The rotational momentum can be compensated by a counter rotating mass of about 305 kg at a radius of 1.8 m. This results in an estimated additional mass of 500 kg including the static assembly and structure.

3.3.8.3.4.1 Budgets

The mechanism budgets are shown in Table 3-51 to Table 3-53, which represent the model outputs:

Element 1: Transfer Habitation Module			MASS [kg]				DIMENSIONS [m]			TEMPERATURE REQs [deg C]			
Unit	Element 1 Unit Name Click on button below to insert new unit	Quantity	Mass per quantity excl. margin	Maturity Level	Margin	Total Mass incl. margin	Dim1 Length or D	Dim2 Width or d	Dim3 Height	Operation (max)	Operation (min)	NOP (max)	NOP (min)
1	Hatch Cover-Internal	2	35.0	To be developed	20	84.0	1.5		0.007				
2	Hatch Cover Locking Mechanisms- Internal	2	200.0	To be developed	20	480.0	1.6	1.5	0.05				
3	Docking Mechanism- IBDM	3	334.4	To be modified	10	1103.5	1.371	0.813	0.254	50.0	-50.0	100.0	-100.0
4	Electronic Box- IBDM	24	8.8	To be modified	10	232.3	0.40	0.25	0.25	50.0	-20.0	70.0	-50.0
5	Berthing Mechanism- Active	2	311.0	To be modified	10	684.2	2.0	1.8	0.190				
6	Berthing Mechanism- Passive	2	177.0	To be modified	10	389.4	2.0	1.8	0.343				
7	Electronic Box- Berthing	2	8.0	To be modified	10	17.6	0.4	0.25	0.25				
8	Crew Centrifuge- SRC	1	450.0	To be developed	20	540.0	4.5		2.0				
9	Crew Centrifuge Counter Rotation Mass	1	500.0	To be developed	20	600.0	3.6		0.5				
10	Antenna Pointing Mechanism- APM Ka-band & La	2	9.4	To be developed	20	22.4							
11	Electronic Box- APM Pointing, Ka-band and Laser	2	5.0	To be modified	10	11.0							
13	Berthing Mechanism- Prop. Passive	1	143.0	To be modified	10	157.3	2.8	2.6	0.3				
15	Hatch Door-Egress External	3	18.0	To be developed	20	64.8	0.90		0.010				
16	Hatch Door Locking Mechanisms- Egress Externa	3	120.0	To be developed	20	432.0	0.95	0.80	0.050				
17	Hatch Door Radiation Protection- Egress External	3	51.0	To be developed	20	183.6	0.80		0.036				
18	Crew Exercise Machine- Flywheel Exercise Device	2	55.0	To be developed	20	132.0	2.75	0.60	1.500				
19	Solar Array Deployment Mechanism- SDM	4	40.3	To be modified	10	177.3				69.0	-32.0	70.0	-46.0
20	Solar Array SDM/Panel Hinges	4	5.6	To be modified	10	24.6						110.0	-69.0
21	Solar Array Yoke Panel	4	4.4	To be modified	10	19.4	1.0	2.50					
22	Solar array Root Hinge	4	5.5	To be modified	10	24.2	0.5	0.30	0.150				
23	Solar Array Drive/Rotation Mechanism (SEPTA 31	4	5.0	To be modified	10	21.8				70.0	-35.0	75.0	-40.0
24	Solar Array Stowed Latch(s)	4	4.0	To be developed	20	19.2							
25	Solar Array Drive Electronics	4	5.2	To be modified	10	22.7							
26	Antenna Pointing Mechanism- Aero Stationary An	1	1.0	To be modified	10	1.1	0.15	0.15	0.075				
27	Electronic Box- APM Pointing, Aero Stationary An	1	0.5	To be modified	10	0.6							
-	Click on button below to insert new unit		0.0	To be developed	20	0.0		-			-		
	ELEMENT 1 SUBSYSTEM TOTAL	25	4756.3		14.5	5445.1							

Table 3-51: TV Mechanisms Mass and Thermal Budgets

Element 1: Transfer Habitation Module			PPEAK AND POWER SPECIFICATION PER MODE							PPEAK AND POWER SPECIFICATION PER MODE					
Unit	Element 1 Unit Name Click on button below to insert new unit	Quantity	Ppeak	OEM Pon	OEM Pstby	OEM Dc	TMIM Pon	TMIM Pstby	TMIM Dc	TMM Pon	TMM Pstby	TMM Dc	MOAM Pon	MOAM Pstby	MOAM Dc
1	Hatch Cover-Internal	2													
2	Hatch Cover Locking Mechanisms- Internal	2													
3	Docking Mechanism- IBDM	3		1806.0	0.0	4.106									
4	Electronic Box- IBDM	24		152.0	0.0	24.541									
5	Berthing Mechanism- Active	2		575.0	25.0	16.216									
6	Berthing Mechanism- Passive	2													
7	Electronic Box- Berthing	2		25.0	0.0	16.216									
8	Crew Centrifuge- SRC	1		300.0	0.0	100.0				300.0	0.0	25.0			
9	Crew Centrifuge Counter Rotation Mass	1													
10	Antenna Pointing Mechanism- APM Ka-band & La	2		14.0	0.0	100.0	14.0	0.0	100.0	14.0	0.0	100.0	14.0	0.0	100.0
11	Electronic Box- APM Pointing, Ka-band and Laser	2		10.0	0.0	100.0	10.0	0.0	100.0	10.0	0.0	100.0	10.0	0.0	100.0
13	Berthing Mechanism- Prop. Passive	1													
15	Hatch Door-Egress External	3													
16	Hatch Door Locking Mechanisms- Egress Externa	3													
17	Hatch Door Radiation Protection- Egress External	3													
18	Crew Exercise Machine- Flywheel Exercise Device	2		61.0	0.0	100.0				61.0	0.0	25.0			
19	Solar Array Deployment Mechanism- SDM	4		60.0	0.0	8.649	60.0	0.0	0.5556	60.0	0.0	0.5556			
20	Solar Array SDM/Panel Hinges	4													
21	Solar Array Yoke Panel	4													
22	Solar array Root Hinge	4													
23	Solar Array Drive/Rotation Mechanism (SEPTA 31	4		20.0	4.0	100.0				20.0	4.0	100.0			
24	Solar Array Stowed Latch(s)	4													
25	Solar Array Drive Electronics	4		5.0	2.0	100.0				5.0	2.0	100.0			
26	Antenna Pointing Mechanism- Aero Stationary Ant	1		10.0		100.0	10.0		100.0	10.0		100.0	10.0		100.0
27	Electronic Box- APM Pointing, Aero Stationary An	1													
-	Click on button below to insert new unit														
	ELEMENT 1 SUBSYSTEM TOTAL	25	0.0	3038.0	31.0		94.0	0.0		480.0	6.0		34.0	0.0	

Table 3-52: TV Mechanisms Mission Modes Power Budget

Unit	Element 1 Unit Name Click on button below to insert new unit	OMM Pon	OMM Pstby	OMM Dc	TEIM Pon	TEIM Pstby	TEIM Dc	TEM Pon	TEM Pstby	TEM Dc	SM Pon	SM Pstby	SM Dc
1	Hatch Cover-Internal												
2	Hatch Cover Locking Mechanisms- Internal												
3	Docking Mechanism- IBDM	1806.0	0.0	3.088	784.0	0.0	0.0	784.0	0.0	0.0231			
4	Electronic Box- IBDM	152.0	0.0	18.455									
5	Berthing Mechanism- Active												
6	Berthing Mechanism- Passive												
7	Electronic Box- Berthing												
8	Crew Centrifuge- SRC	300.0	0.0	100.0				300.0	0.0	25.0			
9	Crew Centrifuge Counter Rotation Mass												
10	Antenna Pointing Mechanism- APM Ka-band & La	14.0	0.0	100.0	14.0	0.0	100.0	14.0	0.0	100.0			
11	Electronic Box- APM Pointing, Ka-band and Laser	10.0	0.0	100.0	10.0	0.0	100.0	10.0	0.0	100.0			
13	Berthing Mechanism- Prop. Passive												
15	Hatch Door-Egress External												
16	Hatch Door Locking Mechanisms- Egress External												
17	Hatch Door Radiation Protection- Egress External												
18	Crew Exercise Machine- Flywheel Exercise Device	61.0	0.0	25.00				61.0	0.0	25.00			
19	Solar Array Deployment Mechanism- SDM	60.0	0.0	6.504				60.0	0.0	0.5556			
20	Solar Array SDM/Panel Hinges												
21	Solar Array Yoke Panel												
22	Solar array Root Hinge												
23	Solar Array Drive/Rotation Mechanism (SEPTA 31	20.0	4.0	100.0				20.0	4.0	100.0			
24	Solar Array Stowed Latch(s)												
25	Solar Array Drive Electronics	5.0	2.0	100.0				5.0	2.0	100.0			
26	Antenna Pointing Mechanism- Aero Stationary Ant	10.0		100.0	10.0		100.0	10.0		100.0			
27	Electronic Box- APM Pointing, Aero Stationary Ant.												
	ELEMENT 1 SUBSYSTEM TOTAL	2438.0	6.0		818.0	0.0		1264.0	6.0		0.0	0.0	

Table 3-53: TV Mechanisms Mission Modes Power Budget

3.3.8.4 Options

As regards the power generation system, other systems are currently in study phases:

- Concentrator solar arrays- Use of concentrator surfaces to increase solar energy on to the cells- Potential to decrease the physical size of arrays, however mass influence is unclear.
- Flexible arrays- the power versus mass or watt/kilogram ratio of flexible arrays will increase and become greater than the W/kg of conventional fixed panel arrays, however the size capability and mass of such arrays is unclear.

3.3.9 Structures

3.3.9.1 Requirements and design drivers

For the design of the TV the following set of general requirements were taken into account:

- Habitation Module, with a diameter limitation of 6 m for compatibility with Energia fairing.
- Compatibility with the vehicle launcher Energia induced mechanical loads.
- THM shall provide a stormshelter to protect the crew in case of a solar particle event.

All module structures shall provide the mechanical support to ensure mission success.

3.3.9.2 Assumptions and trade-offs

Due to lack of information regarding the longitudinal and lateral frequencies requirements of the Energia, launcher the verification of these requirements was based on the correspondent values for the rocket Zenit. This implies the following requirements: lateral frequency > 8 Hz and longitudinal frequency > 20 Hz.
For strength calculations a safety factor of 1.5 was considered.

3.3.9.3 Baseline design

3.3.9.3.1 Habitation module skin

The Habitation Module is the main load-carrying structureIt is a 6-m diameter and 14-m long pressurised cylinder of 4 mm thick aluminium. This structure needs local framing structure around windows and other penetrations and discontinuities.
Aluminium was selected due to its low density and high strength.
The first eigen-frequency, for the THM, results in 35.6 Hz, which initially fulfils the Zenit requirements.

3.3.9.3.2 Meteoroid and debris shielding

The safety of the spacecraft in long-term space flights requires a special protection from damage by meteoroids and orbital debris.
For the spacecraft meteoroid and orbital debris protection two options were considered: a “default shielding”- monolithic shield of aluminium and a multi-shock shield, the solution proposed for the Mars Trans Habitat design study, [RD55], with Nextel and Kevlar as bumper materials.

The multi-shock shield was chosen because a Nextel/Kevlar shield provides better protection than double-aluminium bumper shields of equal weight by stopping 50% to 300% more passive projectiles, [RD56]. The objective of the bumpers is to break up debris on impact and distribute the particle energy and momentum over a large area. Due to the reduction in particle size and velocity, the resulting debris cloud does not penetrate the monolithic pressure wall behind them.
Nextel improves shield performance, compared to aluminium because it is better at shocking projectile fragments, while Kevlar improves shield performance because it is better at slowing debris expansion, having a greater strength to weight ratio than aluminium. When using a Nextel/Kevlar intermediate shield, the particle size of bumper materials within the debris cloud is smaller than for aluminium intermediate shields. Figure 3-69 shows the differences between the debris cloud resultant from the impact of a particle in an aluminium bumper and a Nextel bumper.

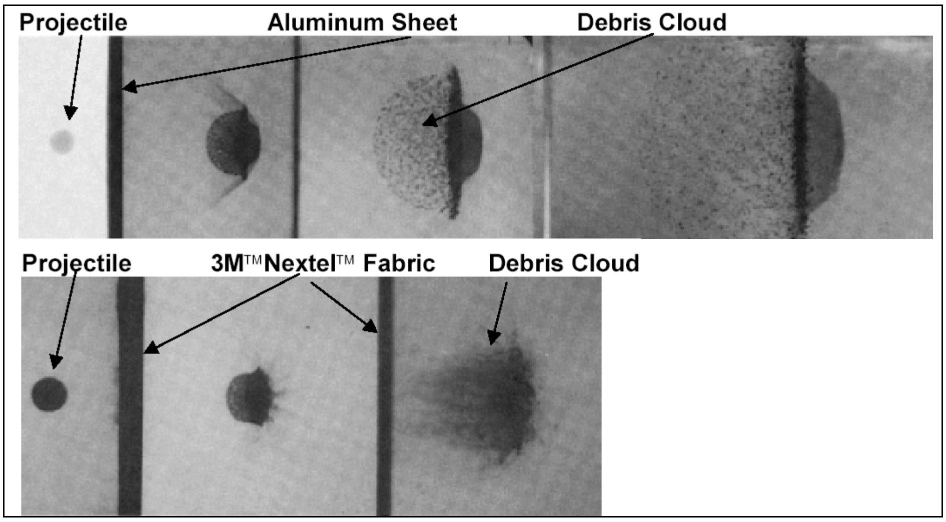


Figure 3-69: Debris cloud contact between aluminium and Nextel MOD, [RD56].

The Nextel Meteorite and Orbital Debris shield (MOD) selected consists of two walls. The front wall consists of three Nextel AF-10 bumpers, each one separated by a 10 cm standoff of low-weight open-cell foam, which serves as the support material. The rear wall consists of five layers of Kevlar fabric and provides final barrier to high penetration. This shield offers the advantage of being initially transported compressed to a thickness of 5 cm. Once in place, outside the settlement, the foam inside the shield is inflated to its full thickness of 30 cm. For this mission due to the thermal protection, it is decided to launch it at its maximum thickness. This configuration is capable of resisting impacts of incidents objects with a diameter of 6.35 mm and a velocity of 6.82 km/s.

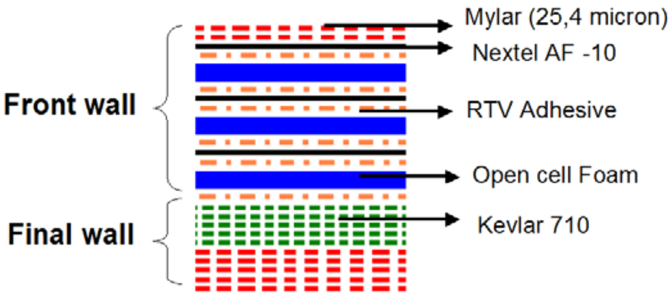


Figure 3-70 – MOD Shielding Configuration

3.3.9.3.3 Radiation shielding

According to the duration and type of mission, the radiation dose limits that an astronaut can receive during the mission are different. These requirements are translated in terms of the ratio mass per area, kg/m^2 of the protection material.

With the mass available on-board, it was analysed what was the protection provided, for the THM and for the storm shelter, and the necessary amount of mass to be added to fulfil the shielding requirements. The mass available on board that is able to provide shielding protection is shown in Table 3-54:

Material	Mass [kg]
THM Skin + Stiffening	4382.9
Internal Equipment	28754
THM Debris Shielding	964.9
MLI	525.68
Water	1200
Total mass	**43 944**

Table 3-54: Mass Budget for radiation protection

Shielding effectiveness depends largely upon the conductivity of the material. Materials with low Z are considered to be effective. For preliminary calculations the density ratio to convert from g/cm^2 to material shielding thickness was used, that is, 1 cm of H_2O is generally equivalent to a 1 cm thick slab of water, or 4 mm of aluminium. The mass required for a certain shielding requirement is therefore independent of the type of material, only the thickness will be different from one material to the other.

With the mass available on-board, two configurations for the shielding protection were analysed. The shielding effectiveness of each configuration, for nominal protection and for the storm-shelter is described in Table 3-55.

Also on-board are the consumables, weighing 10 200 kg, which were not taken into account in these calculations because it is not yet known what percentage will be able to be used as shielding protection, after their usage. The mass necessary to add in the closures corresponds to 60.6% of the consumables. This value is assumed to be viable to be used as shielding material. Shielding to this area may be provided by the consumables and by propulsion modules on one side and on the other by the MEV.

The necessary mass to be added in the skirt is smaller in case 1. Due to all the facts presented, case 1 was selected, and through this it's only necessary to add 2007kg of extra mass.

It was decided to add extra mass in the form of water. Then instead of having two separated water tanks, the extra mass is located in the closures, and in the skirt the 2000 kg will be consumables. Figure 3-71 shows the selected configuration.

Note that the shielding effectiveness of vehicle skin or equipment cases with metal walls of any reasonable thickness is limited by the apertures, joints and others discontinuities, rather than the metal itself.

<table>
<tr><th>Radiation Shielding</th><th>Habitation Module</th><th colspan="2">Storm-shelter</th></tr>
<tr><td>Shielding Requirements</td><td>9 g/cm^2</td><td colspan="2">25 g/cm^2</td></tr>
<tr><td colspan="2"></td><td>Case 1</td><td>Case 2</td></tr>
<tr><td rowspan="4">Mass available</td><td rowspan="4">Internal Equipment (85%)
MLI, Structural Skin, Debris Shielding
30341 kg</td><td colspan="2"><u>Skirt</u></td></tr>
<tr><td>15% - Internal Equipment
4500 kg
Thickness: 0.42m</td><td>H_2O – 1200 kg
Thickness: 0.03 m</td></tr>
<tr><td colspan="2"><u>Closures</u></td></tr>
<tr><td>H2O – 1200 kg
Thickness: 0.02m</td><td>15% - Internal Equipment
4500 kg
Thickness: 0.3 m</td></tr>
<tr><td rowspan="4">Protection Provided</td><td rowspan="4">9.5 g/cm^2</td><td colspan="2"><u>Skirt</u></td></tr>
<tr><td>(10.58 g/cm^2)
20.05 g/cm^2</td><td>(2.96 g/cm^2)
12.43 g/cm^2</td></tr>
<tr><td colspan="2"><u>Closures</u></td></tr>
<tr><td>(2.12 g/cm^2)
11.59 g/cm^2</td><td>(7.58 g/cm^2)
17.05 g/cm^2</td></tr>
<tr><td rowspan="5">Mass to add</td><td rowspan="4">-----------</td><td colspan="2"><u>Skirt</u></td></tr>
<tr><td>2007.2 kg</td><td>5094.4 kg</td></tr>
<tr><td colspan="2"><u>Closures</u></td></tr>
<tr><td>7582.9 kg</td><td>4495.7 kg</td></tr>
<tr><td>Total mass necessary:</td><td>9590.17 kg</td><td>9590.17 kg</td></tr>
</table>

Table 3-55: Level of protection provided by the mass on-board

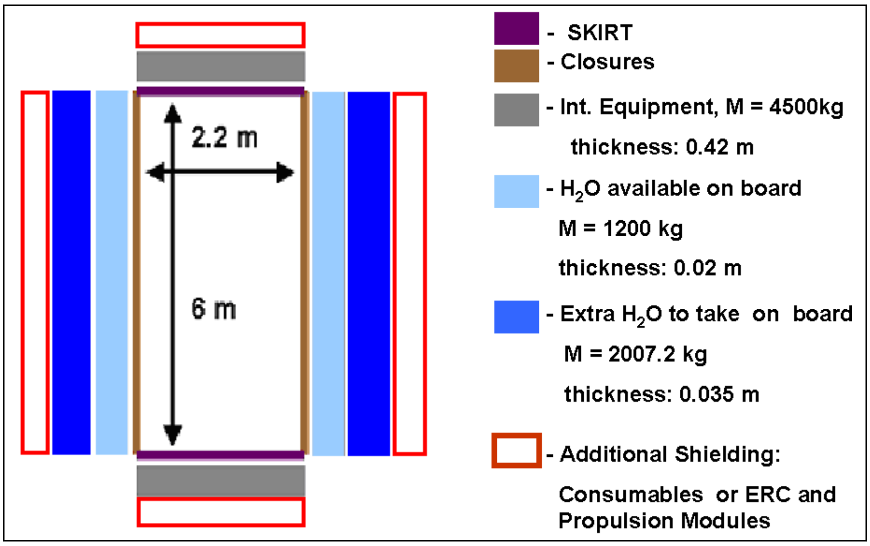


Figure 3-71: Shielding configuration for the storm-shelter.

3.3.9.4 Budgets

The structural mass composition is show in Table 3-56:

Item	Nr.	Mass [kg]	Margin [%]	Mass with Margin [kg]
THM Skin	1	2921.99	5	3068.9
Nodes_Skin	2	834.58	5	876.31
EVA	1	254.84	5	267.58
Total Stiffening	1	2005.71	5	2105.99
THM_Debris Shielding	1	964.94	5	1013.19
Nodes_Debris Shielding	2	302.73	5	317.86
Fixtures_large Tools_gloveboxes	1	1000	5	1050
Racks	1	2146	20	2575.20
TOTAL				**12468.39**

Table 3-56: THM Structural Mass budget

3.4 Propulsion module

3.4.1 Propulsion

3.4.1.1 Trans Mars Injection (TMI)

3.4.1.1.1 Requirements and design drivers

This propulsive module performs the transfer from a Low Earth Orbit to Martian orbit by three impulsive burns

The TMI is composed of several stacks that are injected in LEO by a heavy launcher (Energia) and have to be assembled in orbit. Each stack contains an autonomous propulsion system with engines, and relevant tanks fully loaded.

The TMI is composed of three stages each composed of four stacks. The four stacks in a stage will be fired together to provide the thrust to mass ratio required to avoid gravitational losses. The inertia forces generated on the solar arrays will be high, so the arrays will have to be folded.

3.4.1.1.2 Assumptions and trade-offs

The TMI module uses LH2-LOX propellant at cryogenic storage conditions. The low Isp of the storable propellants resulted in prohibitive mass penalty for the TMI.

A model to estimate the dry mass of each stack has been made including thermal protection system of the tanks, resulting in an inert dry mass of 11.5 % (9.2 tonne) for each stack.

3.4.1.1.3 Baseline design

One VULCAIN 2 gas generator (open cycle) engine has been selected as propulsion system for each cryogenic stack. The thruster is being developed for the first stage of the Ariane-5 launcher. Isp has been scaled with respect to the nominal for deep vacuum operations.
The propulsion system and its characteristics are shown in Table 3-57:

Characteristic	Value
Number of thruster	1 for each stack
Thrust	1300 kN (gas generator)
Isp	450 s
Exit diameter	2100 mm
lLength	3500 mm
Thruster mass	2035 kg
Propellant	LOX/LH2
O/F ratio	6.2
Number of tanks	1+1 Bulkhead for each stack
Tanks material	Al-Li Alloys
Max MEOP LOX	6 bar
Max MEOP LH2	2 bar
Mass of LH2 tank	1238 kg
Mass of LOX tank	902 kg

Table 3-57: TMI Summary

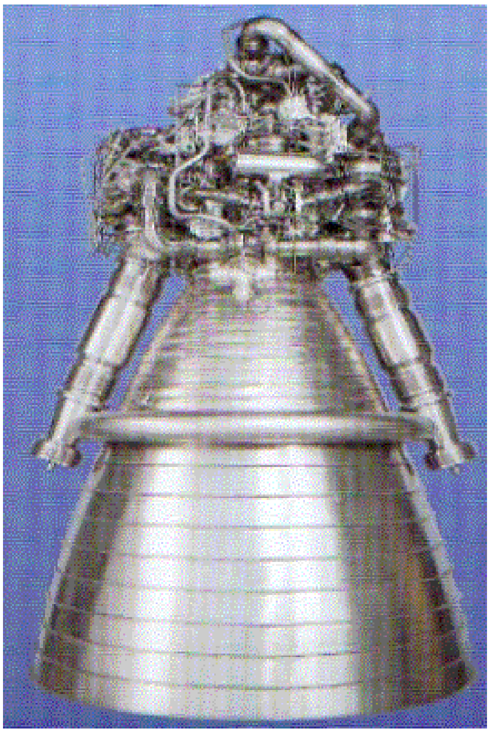


Figure 3-72: VULCAIN 2

3.4.1.1.4 Budgets

Element	Mass
Propellant mass (per stack)	70786 kg
Propulsion dry mass (including margins)	9214 kg
This mass includes the following estimation	
Structure	3561 kg
Engines, tanks, actuators feed lines, valves and regulators	4777 kg
Thermal control	843 kg
Mechanisms	33 kg

Table 3-58: TMI stack mass budget

3.4.1.2 Mars Orbit Injection (MOI) and Trans Earth Injection (TEI) modules

3.4.1.2.1 Requirements and design drivers

The Mars orbit Injection (MOI) module is the propulsion module, that performs the injection into Mars orbit.

The Trans Earth Injection (TEI) module is the propulsion module that brings the Habitation Module from Mars orbit to the Low Earth Orbit.

The MOI is composed of four stacks fuelled by storable UDMH-NTO propellant. Two 80 tonne stacks for the first stage and two 50-tonne stacks for the second stage.

The TEI is composed by the same propulsion system of the MOI first stage and uses the same engine components and storable propellant. The TEI module is composed only by one stack.

As for the TMI module the stacks are injected in LEO and assembled in orbit. Due to launchers constraints each stack weighs a max 80 tonnes and, is enveloped in a cylinder of 6 metres diameter and a length of 35 m.

The thrust level is dictated to minimise the gravitational losses. The MOI burn is divided into 2 burns and staged in two assemblies. Each stage is composed of two stacks.

3.4.1.2.2 Assumptions and trade-offs

The MOI uses only storable NTO/UDMH liquid propellants.
Semi-storable (LOX-Kerosene) engines were also considered in the analysis to increase the specific impulse and reduce propellant mass. The use of cryogenic (LOX-LH2) or partially cryo (LOX-Kerosene) propellant has been analysed and at the time of writing an assessment for LOX-Kerosene (or derivatives) is under evaluation. *Baseline design*

A Russian RD 0212 engine has been selected as propulsion system for the MOI and TEI stack. The thruster has been developed for the third stage of K/M Proton launcher.

The propulsion system and its characteristics are shown in Table 3-59:

Characteristic	Value
number of thruster	1 for each stack
Thrust	612 kN
Isp	325 s
Exit diameter	4150 mm
Length	4000 mm
Thruster mass	600 kg
Propellant	NTO/UDMH
O/F ratio	2.4
Number of tanks	1+1 Bulkhead for each stack
Tanks material	Ti
Max MEOP UDMH	5.5 bar
Max MEOP NTO	6.5 bar
Mass of UDMH tank	260 kg
Mass of NTO tank	310 kg

Table 3-59: MOI and TEI summary

Figure 3-73: RD 0212

3.4.1.2.4 Budgets

Element	Mass
Propellant mass	76 324 kg
Propulsion Dry mass (including margins)	3776 kg
This mass includes the following estimation:	
Structure	1643 kg
Engines, tanks, actuators feed lines, valves and regulators	1903 kg
Thermal	95 kg
Mechanisms	36 kg

Table 3-60: MOI/TEI stack mass budget

3.4.1.2.5 Options

For the TEI, four YUZHNOYE RD 861-G gas generator bi-propellant NTO-UDMH thrusters have been proposed instead of the RD 0212 engine. However, the final choice was dictated by thrust-to-mass ratio constraints.
Overall thrust of the module in this case is around 305 kN whereas overall mass of the propulsion system remains unchanged.

3.4.2 Structures

3.4.2.1 Requirements and design drivers

For the design of the propulsion module the following set of general requirements were taken into account:

- Maximum of a 6 m diameter for the propulsion structural elements, due to compatibility with Energia fairing.
- Compatibility with the vehicle launcher Energia induced mechanical loads.

All structures shall provide the mechanical support to the propulsion stacks.

3.4.2.2 Assumptions and trade-offs

Given information regarding the longitudinal and lateral frequencies requirements of the launcher Energia, the verification of these requirements was based in the correspondent values for the Zenit rocket. Which implies the following requirements: lateral frequency higher than 8 Hz and longitudinal frequency higher than 20 Hz.
The wall thickness of the propellant tanks will first be calculated from stresses caused by internal pressure loads, and then checked for other loads.

For strength calculations a safety factor of 1.5 was considered.

3.4.2.3 Baseline design

Each propulsion stage is composed of a number of stacks attached to an aluminium backbone, of 5 m diameter and 4 mm thick. The difference between the backbones of the TMI and MOI modules is the length, due to the size differences of the tanks and engines.

The MOD shielding, for all the propulsion modules are the same as the one for the THM.
The tanks walls form an integral part of the structure and are designed to withstand the internal pressure loads as well as the vehicle dynamic loads, without the necessity of a skin, but due to thermal requirements, an aluminium skin of 1 mm thickness is added to the tanks.

3.4.2.3.1 TMI module

The TMI propulsion module has three stages; each one consists of four stacks attached to a backbone. The liquid oxygen and liquid hydrogen tanks, connection rings, a skirt, an engine frame, skin and debris shielding constitute each stack.
The stacks are attached to the backbone through the rings and a three-point connection.
After a preliminary analysis, the first lateral Eigen-frequency for the TMI backbone results in 25.6 Hz. This initially fulfils the Zenit stiffness requirements, for the general launch environment.
The tanks are arranged in tandem configuration, Figure 3-74, with the liquid oxygen on the top and the liquid hydrogen tank below, with a common spherical bulkhead. The oxygen tank is a sphere of 4.3 mm of thickness and the hydrogen tank is a cylinder with spherical domes of 2.9 mm thickness, both with 5 m of diameter.

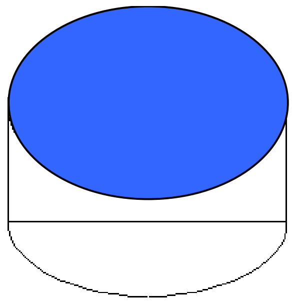


Figure 3-74: Tandem configuration

Aluminium alloys are recommended for long-term storage of cryogenic propellants, so all mass calculations were done with pure aluminium in the preliminary analysis.
The engine frame selected is the same as the one used for the main engine of Ariane-5. The engine frame consists of a cone cap, an attachment ring and cone cap stiffeners.

3.4.2.3.2 MOI module

The MOI propulsion module has two stages; each one consists of two stacks attached to a backbone. The oxide and fuel tanks, connection rings, a skirt, an engine frame, skin and debris shielding constitute each stack.
The stacks are attached to the backbone through the rings and a three-point connection.
After a preliminary analysis, the first lateral Eigen-frequency for the MOI backbone results in 71 Hz. This initially fulfils the Zenit stiffness requirements, for the general launch environment.
The tanks are arranged in tandem configuration, with the oxidizer tank on the top and the fuel tank below, with a common spherical bulkhead. The oxidizer tank is a sphere of 1.2 mm thickness and the fuel tank is a cylinder with spherical domes and 2.8 mm thickness, both with 4 m diameter.
For long-term storable propulsion, it is recommended to use titanium instead of aluminium for material skin of the tanks.

The mass of the engine frame was achieved by comparison with the one used for the TMI module.

3.4.2.3.3 TEI module

The TEI module consists of one stack equal to the MOI one. This stack is located inside the MOI backbone structure and attached to it, by means of rings and a three-point connection.
For the separation system of the last stage of the MOI module from TEI module, two options were analysed:

- Vertical guiding rail system of the backbone, to allow only the axial descent movement of the backbone
- The same separation system as the one used for Ariane-5 fairing. It consists of a pyro cord, which breaks a bolted joint around the fairing

Option 2 was selected because the mass penalty is smaller and it is a simpler system to install. For this system, the backbone has to be divided into two half cylinders, which are attached by a bolted joint.
The MOI backbone operates like a fairing to the TEI module. The interior part of the MOI backbone should be almost entirely covered with acoustic absorption panels, to counter noise levels of vibration in the structure at lift-off.

3.4.2.3.4 MOI connection to THM

The connection between the propulsion module and the habitation module is made through an aluminium cone of 4 mm thickness and 2 m length.

3.4.2.4 Budgets

The structural mass budget is shown in Table 3-61:

Item	Nr.	Mass [kg]	Margin [%]	Mass with Margin [kg]
L_{OX} Tank	1	819.82	10	901.80
L_{H2} Tank	1	1126.12	10	1238.74
Lower Skirt	1	707.57	10	778.32
Stack Ring	2	153.08	10	183.69
Backbone	1	2136.13	20	2563.36
Backbone Ring	2	162.95	10	179.24
Engine Frame	1	1500	0	1500
Skin	1	484.2	5	508.44
L_{OX} Tank Debris Shielding	1	78.48	10	86.33
L_{H2} Tank Debris Shielding	1	319.35	10	351.29
TOTAL				**8291.21**

Table 3-61: TMI stack structural mass budget

Item	Nr.	Mass [kg]	Margin [%]	Mass with Margin [kg]
Backbone	1	2847.61	20	3417.129
Backbone Ring	2	162.95	10	179.24
TOTAL				**3775.61**

Table 3-62: TMI stage backbone structural mass budget

Item	Nr.	Mass [kg]	Margin [%]	Mass with Margin [kg]
L_{OX} Tank	1	278.79	10	306.67
L_{Fuel} Tank	1	423.65	10	466.01
Lower Skirt	1	284.66	10	313.12
Stack Ring	2	167.67	10	201.20
Engine Frame	1	475	10	522.50
Skin	1	218.76	5	240.64
L_{OX} Tank Debris Shielding	1	59.52	10	65.47
L_{Fuel} Tank Debris Shielding	1	120.21	10	132.23
TOTAL				**2247.84**

Table 3-63: MOI 80 tonne Stack Structural Mass budget

Item	Nr.	Mass [kg]	Margin [%]	Mass with Margin [kg]
Backbone	1	1737.31	20	2084.77
Backbone Ring	2	203.70	10	224.07
MOI/TEI Interior Backbone ring	2	98.91	10	108.80
Separation System	1	50	20	60
TOTAL				**2477.64**

Table 3-64: MOI Backbone Structural Mass budget

Item	Nr.	Mass [kg]	Margin [%]	Mass with Margin [kg]
Connection Cone	1	243.38	10	267.72
Ring_Connect Backbone to THM	1	162.95	10	179.24
Ring Connect Cone to THM	1	68.95	10	75.86
TOTAL				**522.82**

Table 3-65: MOI connection to THM Structural Mass budget

3.4.3 Thermal

3.4.3.1 Requirements and design drivers

3.4.3.1.1 TMI

The propellants considered for Trans Mars Injection are liquid hydrogen and liquid oxygen. Main requirement is to control the evaporation of these cryogenic liquids during their operational life service (from launch till the end of the trans Mars injection firing). The requirement is

- to maintain a boil-off (BO) below 70 kg per month for the liquid hydrogen and 430 kg per month for the liquid oxygen (system requirement)

It is assumed that no orbital servicing capability will be available for refilling (improbable hypothesis) and that compliance of this requirement can only be done per design.

3.4.3.1.2 MOI and TEI

For both injection, chemical propulsion is retained with the propellant UDMH / NTO

- to maintain the propellant tanks, support structure and tubing above liquid freezing points (between 0 and 40C with margins)
- to maintain the integrity of the thermal protection if close to the thrusters nozzle
- to optimise the ratio efficiency over mass

3.4.3.2 Assumptions

3.4.3.2.1 TMI

- The envelope sizing of the tanks is based on the Russian launcher Energya capability (80T and fairing maximum diameter). The mass of the propellants needed depend on the overall vehicle mass and the number of tanks / stacks is the result of a system level trade-off provided as an input.
- Maximum life service is estimated at system level on the basis of a possible logistic for the delivery flights (build up of the assembly).

3.4.3.2.2 Heat loads

Cryogenics lifetime depends on the heat loads absorbed by the vessel, basically the result of the relative attitude between the vessel, the sun and the planet. Lifetime therefore can be drastically extended by the choice of suitable orbits (high orbit to reduce Earth radiative load) or specific orientation of the vessel (low projected surface to sun or planet). The following hypothesis are done:

- No attitude control capability of the tank. A worst case illumination is assumed (normal incidence)
- LEO is assumed (500 km) for the assembly (launcher capability)
- Influence of others other tanks and assembly (heat loads per infrared or reflection) is not considered.

- Standard illumination (1400 W/m2) and Earth thermal characteristics (albedo 0.4, infrared 273 W/m2) are assumed

3.4.3.3 Baseline thermal design

3.4.3.3.1 Hydrogen storage

Given that hydrogen storage is essential for fuel cells, life support and propulsion (Chemical, solar or nuclear), the efficiency of its storage is shown in Table 3-66:

Type	Features
Compressed hydrogen	in a gaseous state, can be stored under high pressure (up to 700 bars) within pressure vessels (aluminium, composite) increasing the hydrogen storage density
Liquid hydrogen	in a liquid state, is subject to boil-off (evaporation of liquid caused by heat leaks) depending on the vessel size, shape and thermal insulation. Density depends on saturated temperature
Metal hydrides	per absorption on transition metal, hydrogen storage density reach maximum 7% of metal weight (200-300C), 2-5% (alanates) under normal temp. and pressure. Investigations focus on more performance and lighter metal density, but so far weight is a problem for space applications
Chemical hydrides	per chemical reaction. A hydride solution (sodium borohydride for example) combined to water and catalyst produces hydrogen
Carbon nanotubes	per adsorption on activated carbon structure, hydrogen storage density could theoretically approach storage density of liquid hydrogen but mechanisms for adsorption/desorption are still under investigation (nanotubes)
Glass microspheres	per physical adsorption on micro glass sphere. Permeability is controlled per temperature

Table 3-66: Hydrogen storage options

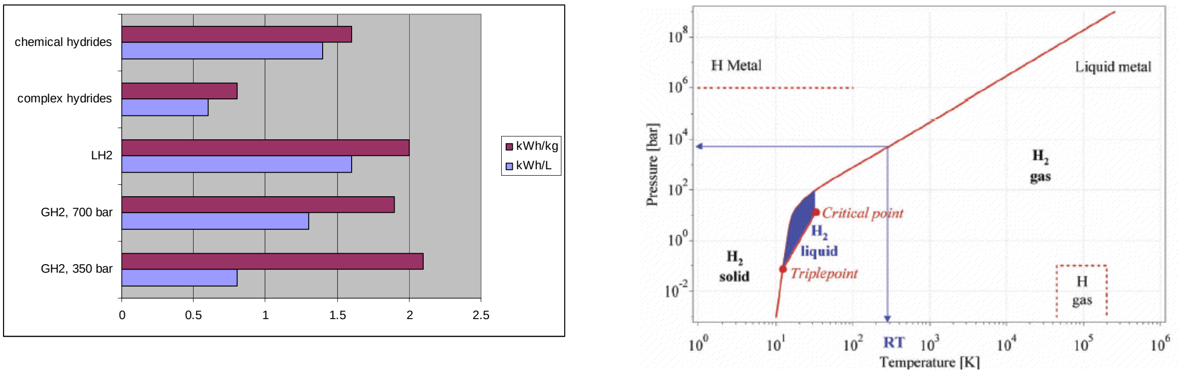


Figure 3-75:Comparative storage technology (L), Hydrogen phase diagram (R)

For propulsive applications, storage in the liquid state is currently the most efficient technique available (shall remain so at least in the mid term) and has been retained for this study. The performance of this type of storage is related to the thermal design capability to maintain cryogenic temperature (20.2K at 1 bar).
Condition of storage is a trade-off between the different constraints from the system and available thermal hardware. For example, supercritical storage eases the refrigeration requirement (higher efficiency at higher temperature) but lowers the density of the liquid (by a factor of 2.6 from triple point to critical point) and increases the pressure (by a factor of 170 from triple point to critical point).

3.4.3.4 TMI module design

The tank design includes an inner and outer vessel(s), stiffening rings, insulation system, a fluid acquisition system (piping, valves), and a thermal system.
The inner vessel will be constructed from inconel (corrosion resistant) and the outer vessels from aluminium (lower density). A cylindrical geometry (4.8 m length, 4.7 m diameter) has been retained for the hydrogen vessel and a nested spherical vessel (4.7 m diameter) for the oxygen (type Ariane-5 ECP).

The primary objective of the tank thermal design is to minimise the heat transfer to the inner vessel and optimise its related mass. The heat loads have been considered on the basis of a worst case assumption).

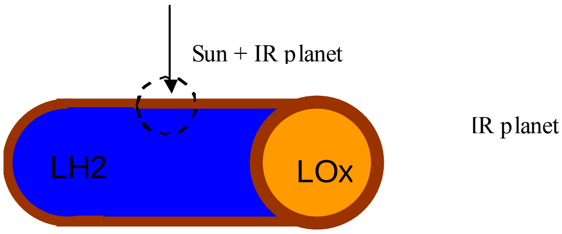


Figure 3-76: Tanks schematic

Boiling of cryogens occurs when temperature exceeds locally the saturation temperature at a certain pressure. Tanks are initially pressurised at 2 bars before launch, which gives the liquid properties shown in Table 3-67:

At 2 bars	Temperature	Latent heat of vaporization	Density (liq.)
Saturated hydrogen	22.9 K	4.29E5 J/kg	67.4 kg/m3
Saturated oxygen	97.2 K	2.06E5 J/kg	1120 kg/m3

Table 3-67: Cryogens properties

3.4.3.4.1 Shielding and insulation, passive techniques

In vacuum, Multi Layer Insulation (MLI) is the best=performing insulation compared to other types including permeable insulations (gas filled powders, evacuated powders) or solid foams with closed or open cells (Airex, Rohacell).

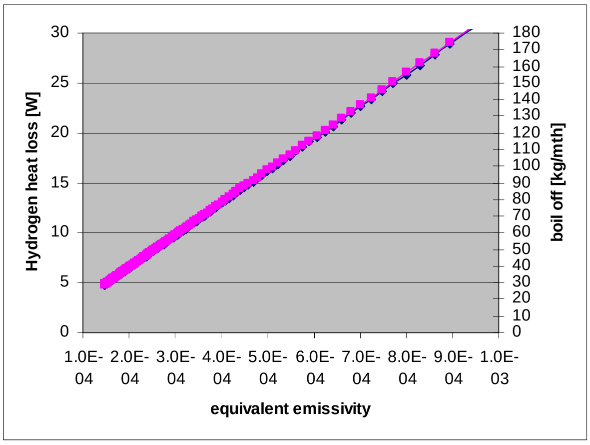
Insulation	Expanded foams	Gas-filled powder	Evacuated powders	Opacified powders	MLI
Conductivity	0.026 W/m/K	0.019 W/m/K	5.9E-4 W/m/K	3.3E-4 W/m/K	1.4E-5 W/m/K

Table 3-68: Insulation properties

MLI efficiency comes from an effective reduction of conductive and radiative coupling with the use of multiple layer radiation shields interspaced by an insulant. Without pressure loads, the equivalent efficiency depends on the number of layers, and the global heat transfer to the vessel can be simply assessed providing the following hypothesis:

- heat transfer to the fluid per conduction only (steady fluid), no contact resistance

- radiative foils are parallel. A correlation factor is considered to fit experimental data (20 layers). The variation of the efficiency at cryogenic temperatures is assumed close to zero
- parasitic heat transfers through piping, rings and other structural elements are provisioned to 5W per default (sensitivity to the design and the temperature of elements)
- antireflective external layer (requirement for visiting vehicles) is imposed with betacloth



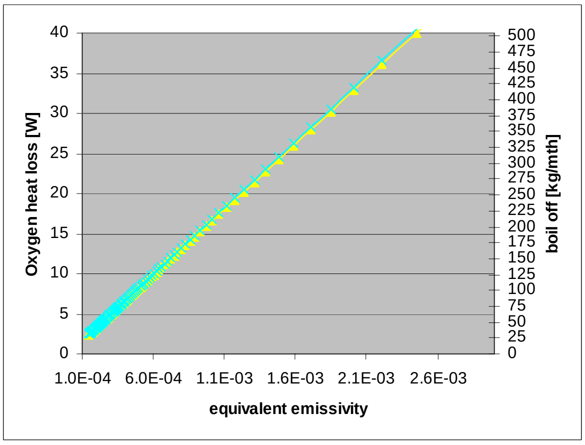
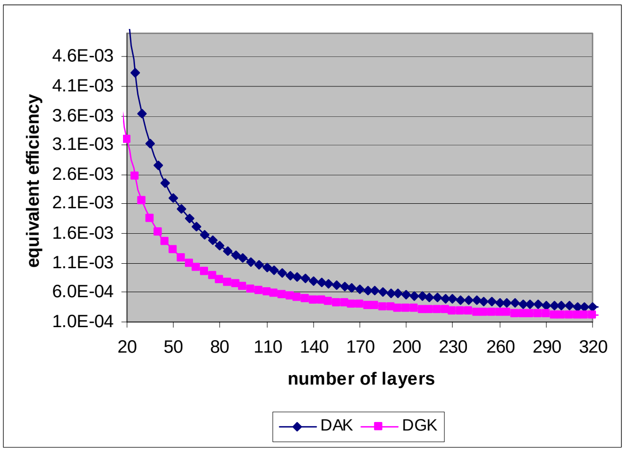


Figure 3-77: Heat loss

Therefore, considering the here above geometry,	Hydrogen	Oxygen
maintaining boil-off below	70 kg/mth	430 kg/mth
requires at least an MLI equivalent efficiency of	3.6E-4	2.2E-3
and leaves a residual heat loss of	11 W	33W

Table 3-69: Requirements

Constrained by the attachment points (compression loads), MLI performance (ratio efficiency / mass) degrades beyond a certain thickness (40-50 layers). A solution is to have different supporting structures. The equivalent efficiency for Double Aluminized Kapton (DAK) and Double Goldenized Kapton (DGK) is indicated in Figure 3-78. To meet the mentioned requirement, 190 of DGK (320 layers of DAK), interspaced with Dacron would be needed.



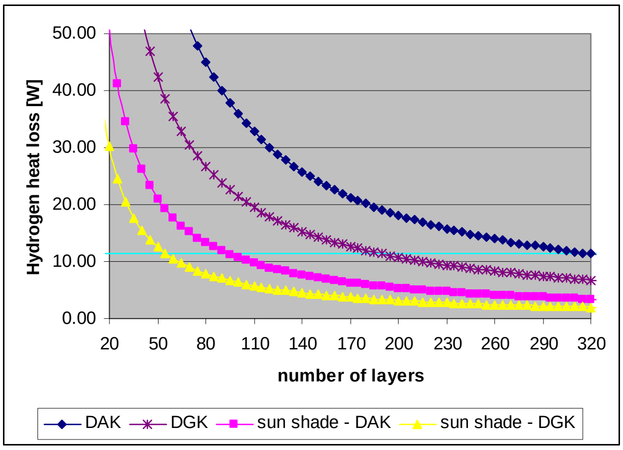


Figure 3-78; Equivalent efficiency and heat looses as a function of the number of layers

Constrained by the high solar absorptance of the betacloth, using exclusively an MLI solution would be a costly solution, either in term of mass (320 layers of DAK would weight 580kg to cover the tank surface), or financial cost (190 layers of DGK). Installing these different blankets on the shields (five blankets of 38 layers distributed on two or three external aluminium shells) might prove delicate along with probable degradation of the efficiency.
Sunshade allows a drastic reduction of heat losses along with the minimum required insulation (decrease of a factor 3.6) but its implementation raises certain constraints to be dealt at system level:

- reflective coatings (OSR, SSM) are incompatible with nearby operations requirements (docking, EVA), unless this reflective surface can be pointed away during these periods
- a non-axisymetric shield requires a pointing capability with its associated penalties (propellant, energy supply, electronics)
- a decrease of the allowable tank envelope (constraint from the launcher fairing diameter)

3.4.3.4.2 Passive vapour cooling

Using of hydrogen to cool down the shields requires an acquisition system. Extracting the vapour is a particular challenge for Newtonian fluids in zero-g (like hydrogen and oxygen) since there is no equivalence of helium superfluid properties (fountain effect) to exploit. In the absence of buoyancy, the gas stands at its evaporation point and forces have to be created (per capillarity or acceleration) to displace and separate the gas from the fluid. Related technical solutions appear globally inappropriate for the tank dimensions and complex to validate in 1-g.

Phase separation system	Advantages	Disadvantages
Artificial gravity	Distinct separation of the phases	Requires a spin of the tank or a rotation of the fluid (motorization) Friction of the fluid (possible heating, although viscosity is low) Sensitive to adverse acceleration (vehicle for example)
Capillary acquisition	Appropriate to hydrogen low surface tension Simple, no moving parts, well known system	High weight penalty with important surface Sensitive to adverse acceleration

Table 3-70: Phase separation systems

Since direct venting of the gas appears challenging, BO minimisation is to be sought. Extraction of the liquid hydrogen (or a two-phase mixture) for cooling external shields can be considered. Several schemes are possible depending if an open or closed loop is considered.
In the first case, liquid hydrogen can be throttled (Joule-Thomson effect) from the saturated state into a two-phase regime and then vapourised in a heat exchanger controlling the shell and liquid heat losses. A venting valve controlled by a tank pressure sensor regulates the system below a safety value.
The use of another cryogen (helium) to maintain the cryostat temperature could be considered provided it meets mass and volume constraints. With a rate of 5 mg/s, 271 kg (2170 l) helium would be needed for a 24-month period. Volume however is another penalty that could restrain the use of such system on the ground before launch.

3.4.3.4.3 Regenerative and thermoelectric coolers

These systems are not considered for such volume (above 10 m_3) being poorly efficient when high refrigeration loads are required in the case of regenerative system (Stirling for example), or to low temperatures with thermo-electric cooling (above 140K).

3.4.3.4.4 Recuperative coolers

Among possible recuperative systems, the Turbo-Brayton (TB) cycle presents a relatively high thermodynamic efficiency at 20K and appears to be a good candidate today for this range of cooling power (5-15W). Available for ground applications (100 watts, 4K), the challenge is to downsize the cooling power and miniaturize elements to fit space-related requirements, which was done in particular for NICMOS (5W@65K) and MELFI projects (47Wat190K). Developments are ongoing in U.S. and in Europe (ESA TRP on a compressor 50mWat6K, 300W input power).
Expected efficiency of a TB is about 10 to 15% Carnot at 35K today (an equivalent efficiency can be reasonably projected in the coming years for 20K), which would require an input power of the order of 950W.

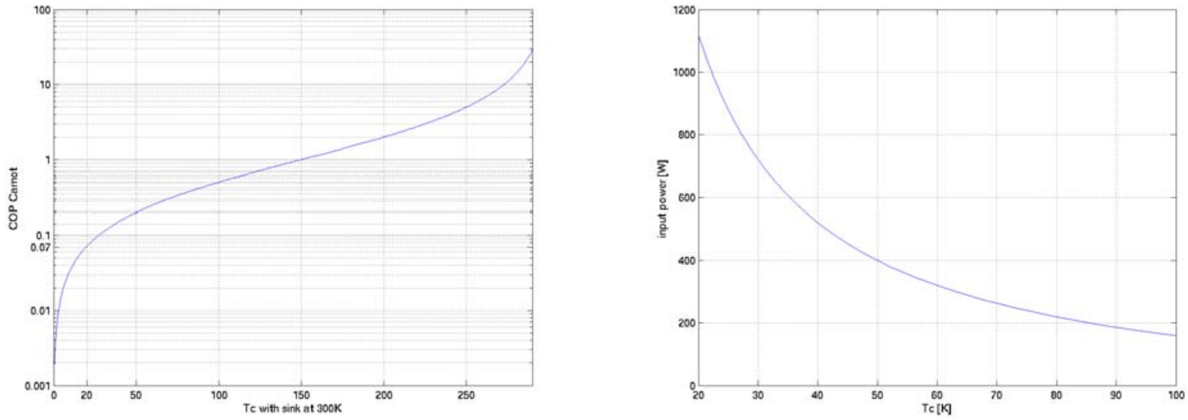


Figure 3-79: COP efficiency and input power as function of Tc

A higher boiling temperature allows significant reduction of the cooler input power: having a subcritical liquid reduces input power by 55%. The primary objective however is densification of the hydrogen to reduce the total amount at launch. Increasing the fluid temperature is therefore secondary and only possible if the launcher capability on the volume is reached.

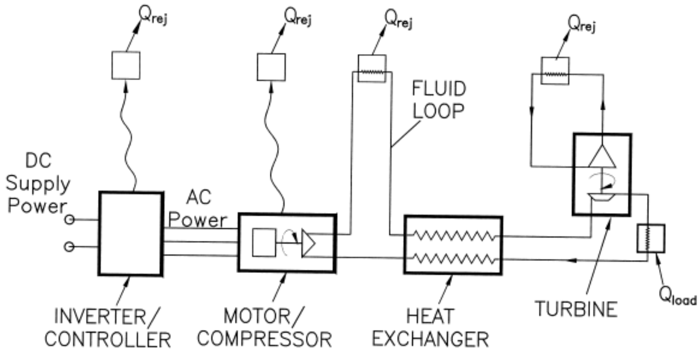


Figure 3-80: TB schematic (1 stage)

Basic components of a TB cycle are the compressor (acts on the working gas), the counterflow heat exchanger (recuperative HX) and the turbine (expansion of the gas and extraction of energy from the tank). This latest, miniaturized and operating at speeds of the order of 100 000 rpm is a critical and challenging component to optimise. Note that the high-frequency electronics raise reliability issues if used longer than 3 years.

3.4.3.4.5 Resources sharing

The tanks are mounted on a truss structure, which allows certain resources to be shared:

- a common sunshade system, deployable, mounted temporarily on the truss until the launch would be cost effective
- a common cooling system (He) does not seem very effective, increasing the number of connections (piping) and would suppose also extravehicular activities to mate the transfer lines
- a common power system would provide required energy for refrigeration

3.4.3.4.6 Synthesis

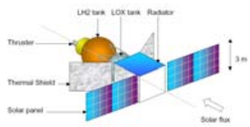
Identified system	Advantage	Disadvantage
Type 1. **Passive insulation, no sun shade**	Simplicity, no power or mechanisms required	Not effective with high mass dedicated to insulation Residual BO (and pressure rise) shall be tolerated per design Recuperation of the tank is problematic => low performance if no reflective or shade system is used, loss of propellant (BO), high structural index, high mass penalty.
Type 2. **Thermal system at tank level (sunshade, refrigeration)** **Individual tanks, orbiting until final assembly (possible fly in formation to simplify final integration)**	The assembly is delayed until all the elements are in orbit (reduction of assembly cost, EVA) 	Cost of a spacecraft per tank launched Spacecraft mass reduce propellant mass => high mass and cost penalty
Type 3. **Integrated thermal design on truss**	Sharing of resources, low mass impact per tank. Possible commonalities (power, cooling or shade system). Possible use of the transfer vehicle if already present for these functions	Requires an immediate assembly (automatic or EVA). EVA appears improbable if not in the proximity of ISS If a sunshade is deployed, has to be removed before launch => sequential assembly seems problematic
Type 4. **Thermal system at tank level: passive insulation, refrigeration** **No on-board capability** **Recuperation by a dedicated system in charge of progressive assembly**	Optimisation of the functions, little redundancy: operational capability on the dedicated system only	The dedicated system needs significant resources to be operational during the complete assembly

Table 3-71: Synthesis

Automatic assembly is seen as an advantage in particular given the uncertainties on the availability of LEO infrastructures (ISS, shuttle derivatives). The cost of an automatic capability (on-board system like docking) would mean a drastic reduction of the payload mass and proportional increase of launched elements, which could strongly affect for liquid propulsion, in particular if a heavy LV is not available.
Thermal design depends somewhat on the strategy adopted for the assembly and available means. Preferred thermal design is nevertheless a hybrid system combining passive insulation techniques, integrated thermal design on the truss, and an active refrigeration at tank level.
A breadboard model of JT closed loop is presently being tested at the Marshall Space Flight Centre and a ZBO capability should be available, possibly flight qualified in U.S. within 5 to 10 years. An equivalent capability could be available in Europe within 10 to 15 years if efforts are oriented to this achievement. A technological basis will be available (compressor in 2004, turbine possibly in 2006), but are not specifically oriented to ZBO hydrogen storage, although technically close (no major obstacle). Still, significant efforts have to be conceded (and foreseen) to reach a ZBO hydrogen tank breadboard.

3.4.3.4.7 Design

Extrapolation of the technology available in a 20-year period is illusive, depending on the efforts projected. So far, despite an increasing interest, there is no guaranty that a ZBO system or equivalent will be available in Europe at that present time. A more basic system (and less efficient) is retained for the moment for this design: integrated thermal design on the truss (deployable shade) and adequate thermal protection. The BO is accommodated per design and a tolerance is foreseen.
If required, a lower level of BO could be reached with implementation of a vapour-cooled shield alimented by centralised system (helium) on the truss. Periodical refilling of this helium tank could be also an option. Use of such system with three vapour-cooled radiation shields allows a reduction of the heat loads by a factor of 2.6.

3.4.3.5 MOI and TEI propellant tanks design

Chemical propulsion (UDMH/NTO) is retained as a baseline, with similar tanks for MOI and TEI stages. Their geometry is respectively:

- a cylinder and half sphere for the UDMH (diameter 4.08 m and length 2.08 m)
- a nested sphere for the NTO (diameter 4.08 m)

The thermal hardware (insulation and heaters) shall preserve the propellant thermal requirements. In particular, under illumination, thermo-optical properties of the external layer shall not create an unfavourable imbalance leading to a higher temperature than 40C. the figure top left in Figure 3-81 indicates a maximum ratio of 1.2 (absorptance over emittance). A betacloth layer, adequate for anti-reflective purpose (during assembly) fulfils this condition (between 0.4 and 0.5). Heaters are necessary to maintain temperature above 0C and avoid NTO freezing (-10C). Temperature is actually a trade-off between MLI performance (quantified by its equivalent emissivity) and the heater density.

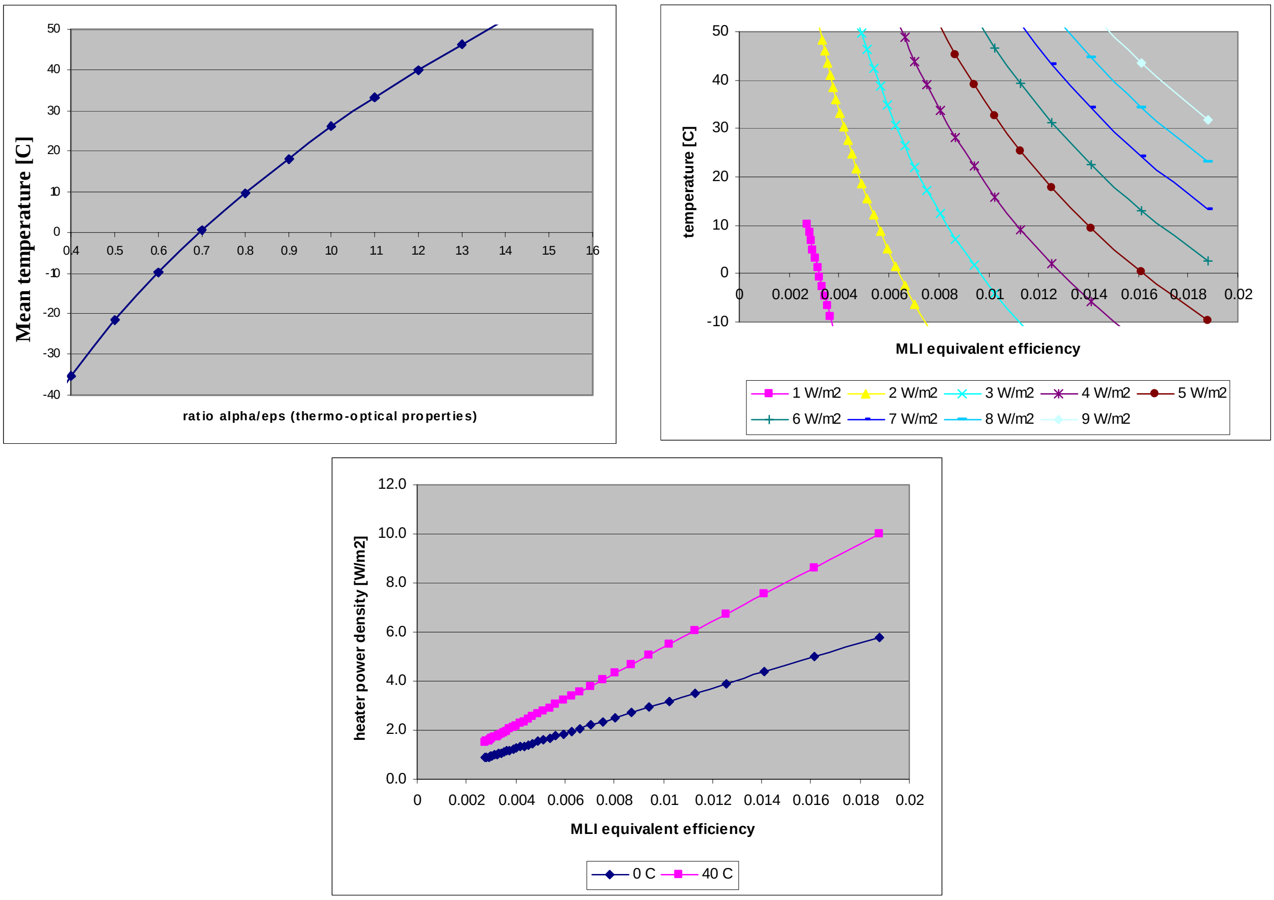


Figure 3-81: Thermo-optical properties and MLI equivalent efficiency

3.4.3.6 Budget

Overall budget (as introduced to the system):

ment 2: Trans Mars Injection Mod			MASS [kg]				
Unit	Click on button below to insert new unit	Quantit	Mass per quantity excl.	Maturity Level	Margin	Total Mass incl. margin	Ppeak
1	TMI tank assembly / MLI LH2	1	582.31	To be modified	10	640.5	0.0
2	TMI tank assembly / MLI LOx	1	184.02	To be modified	10	202.4	0.0
3	TMI tank assembly / cryo	0	0.00	To be developed	20	0.0	0.0
-	Click on button below to insert new unit			To be developed	20	0.0	
LEMENT 2 SUBSYSTEM TOTA		2	766.3		10.0	843.0	0.0

Unit	OEM Pon	OEM Pstby	OEM Dc	TMIM Pon	TMIM Pstby	TMIM Dc	TMM Pon	TMM Pstby	TMM Dc	MOAM Pon	MOAM Pstby	MOAM Dc	OMM Pon	OMM Pstby	OMM Dc	TEIM Pon	TEIM Pstby	TEIM Dc	TEM Pon	TEM Pstby	TEM Dc	SM Pon	SM Pstby	SM Dc
1	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
2	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
3	0.0	0.0	100.0	0.0	0.0	100.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
-															-									
TOTAL	0.0	0.0		0.0	0.0		0.0	0.0		0.0	0.0		0.0	0.0		0.0	0.0		0.0	0.0		0.0	0.0	

Table 3-72: TMI design

ment 3: Mars Orbit Insertion Mod			MASS [kg]				
Unit	Click on button below to insert new unit	Quantit	Mass per quantity excl.	Maturity Level	Margin	Total Mass incl. margin	Ppeak
1	MOI tank assembly / MLI	1	84.97	To be modified	10	93.5	0.0
2	MOI tank assembly / heaters	1	1.00	Fully developed	5	1.1	51.7
-	Click on button below to insert new unit			To be developed	20	0.0	
LEMENT 3 SUBSYSTEM TOTA		2	86.0		9.9	94.5	51.7

Unit	OEM Pon	OEM Pstby	OEM Dc	TMIM Pon	TMIM Pstby	TMIM Dc	TMM Pon	TMM Pstby	TMM Dc	MOAM Pon	MOAM Pstby	MOAM Dc	OMM Pon	OMM Pstby	OMM Dc	TEIM Pon	TEIM Pstby	TEIM Dc	TEM Pon	TEM Pstby	TEM Dc	SM Pon	SM Pstby	SM Dc
1	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
2	51.7	0.0	100.0	51.7	0.0	100.0	51.7	0.0	100.0	51.7	0.0	100.0	51.7	0.0	0.0	51.7	0.0	0.0	51.7	0.0	0.0	51.7	0.0	100.0
-															-									
TOTAL	51.7	0.0		51.7	0.0		51.7	0.0		51.7	0.0		51.7	0.0		51.7	0.0		51.7	0.0		51.7	0.0	

Table 3-73: MOI design

ment 4: Trans Earth Injection Mod			MASS [kg]				
Unit	Click on button below to insert new unit	Quantit	Mass per quantity excl.	Maturity Level	Margin	Total Mass incl. margin	Ppeak
1	TEI tank assembly / MLI	1	84.97	To be modified	10	93.5	0.0
2	TEI tank assembly / heaters	1	1.00	Fully developed	5	1.1	51.7
-	Click on button below to insert new unit			To be developed	20	0.0	
LEMENT 4 SUBSYSTEM TOTA		2	86.0		9.9	94.5	51.7

Unit	OEM Pon	OEM Pstby	OEM Dc	TMIM Pon	TMIM Pstby	TMIM Dc	TMM Pon	TMM Pstby	TMM Dc	MOAM Pon	MOAM Pstby	MOAM Dc	OMM Pon	OMM Pstby	OMM Dc	TEIM Pon	TEIM Pstby	TEIM Dc	TEM Pon	TEM Pstby	TEM Dc	SM Pon	SM Pstby	SM Dc
1	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
2	51.7	0.0	100.0	51.7	0.0	100.0	51.7	0.0	100.0	51.7	0.0	100.0	51.7	0.0	100.0	51.7	0.0	100.0	51.7	0.0	0.0	51.7	0.0	100.0
-															-									
TOTAL	51.7	0.0		51.7	0.0		51.7	0.0		51.7	0.0		51.7	0.0		51.7	0.0		51.7	0.0		51.7	0.0	

Table 3-74: TE design

3.4.4 Mechanisms

3.4.4.1 Requirements and design drivers

The HMM science requirements do not state any specific requirements applicable to the propulsion module mechanisms. As a result of the propulsion module's configuration, the following necessary mechanisms and their requirements can de derived:

- TMI, MOI & TEI In-orbit berthing:
    - LEO In-orbit assembly operations
- TMI Propulsion Stage Separation System:
    - Release & Separation of the Individual Propulsion stage backbone structures:
        - Horizontal separation system
- MOI Stage Separation System:
    - Release & Separation of the Propulsion stage back-bone structure:
        - Horizontal & Vertical Separation System
- TEI Stage Separation System:
    - Release & Separation of the propulsion stack:
        - Horizontal separation system

3.4.4.2 Assumptions and trade-offs

No system specific assumptions or trade-offs have been made or performed.

3.4.4.3 Baseline design

3.4.4.3.1 Backbone structure berthing

For all berthing I/Fs, the common berthing mechanism shall be implemented. The following I/F are affected:

- TMI Stage-to-stage I/F
- TMI to MOI I/F
- Node-to-TEI Stack-MOI Backbone

3.4.4.3.2 Horizontal separation system

For release of individual propulsion stages, a clamp-band with ejection springs shall be the baseline.

3.4.4.3.3 Vertical separation system

The along-axis joints of the two half cylinder parts of the MOI backbone shall be separated by a system similar to the launcher fairing jettison system. The separation mechanism consists of a pyrotechnic cord along the length of the joint which, when ignited, induces a failure along the joint.

3.4.4.4 Budgets

Element 2: Trans Mars Injection Module			MASS [kg]				DIMENSIONS [m]		
Unit	Element 2 Unit Name Click on button below to insert new unit	Quantity	Mass per quantity	Maturity Level	Margin	Total Mass incl. margin	Dim1 Length	Dim2 Width	Dim3 Height
1	Berthing Mechanism-Structure. Active	1	158.9	To be developed	20	190.7	2.8	2.6	0.2
2	Berthing Mechanism-Structure. Passive	1	143.0	To be developed	20	171.6	2.8	2.6	0.3
3	Clamp-Band	1	43.8	To be developed	20	52.5		2.8	
4	Ejection Mech- TMI Stage	1	57.56	To be modified	10	63.3			
5	Propulsion Stack Berthing Mech	1	30.0	To be modified	10	33.0			
6	Berthing Mechanism Electronics	1	8.0	To be developed	20	9.6			
7				To be developed	20	0.0			
-	Click on button below to insert new unit			To be developed	20	0.0		-	
	ELEMENT 2 SUBSYSTEM TOTAL	6	441.2		18.0	520.7			

Table 3-75: TMI mass budget

Element 3: Mars Orbit Insertion Module			MASS [kg]				DIMENSIONS [m]		
Unit	Element 3 Unit Name Click on button below to insert new unit	Quantity	Mass per quantity	Maturity Level	Margin	Total Mass incl. margin	Dim1 Length	Dim2 Width	Dim3 Height
1	Berthing Mechanism-Structure. Active	1	158.9	To be developed	20	190.7	2.8	2.6	0.2
2	Berthing Mechanism-Structure. Passive	1	143.0	To be developed	20	171.6	2.8	2.6	0.3
3	Clamp-Band	1	43.8	To be developed	20	52.5		2.8	
4	Ejection Mech- MOI Stage 1	1	15.9	To be developed	20	19.1			
5	Propulsion Stack Berthing Mech	1	30.0	To be developed	20	36.0			
6	Berthing Mechanism Electronics	1	8.0	To be developed	20	9.6			
7	Vertical Separation System	1	62.5	To be modified	10	68.8			
-	Click on button below to insert new unit			To be developed	20	0.0		-	
	ELEMENT 3 SUBSYSTEM TOTAL	7	462.0		18.6	548.2			

Table 3-76: MOI mass budget

Element 4: Trans Earth Injection Module			MASS [kg]				DIMENSIONS [m]		
Unit	Element 4 Unit Name Click on button below to insert new unit	Quantity	Mass per quantity	Maturity Level	Margin	Total Mass incl. margin	Dim1 Length	Dim2 Width	Dim3 Height
1	Clamp-Band	1	31.3	To be developed	20	37.5		2.8	
2	Ejection Mech- TEI	1	7.2	To be developed	20	8.6			
3				To be developed	20	0.0		2.6	0.3
4				To be developed	20	0.0			
5				To be developed	20	0.0			
6				To be developed	20	0.0			
-	Click on button below to insert new unit			To be developed	20	0.0		-	
	ELEMENT 4 SUBSYSTEM TOTAL	2	38.4		20.0	46.1			

Table 3-77: TEI mass budget

4 MARS EXCURSION VEHICLE

4.1 Systems

The Mars Excursion Module is the one that will allow the landing of a crew of three astronauts on the Martian surface. It has to provide life support systems for the crew for at least 30 days and the means to land and go back to orbit for a later rendezvous with the TV.
It is composed of a Descent Module (DM), mainly the heat shield for the entry, deorbit engines and parachutes, the Surface Habitation Module (SHM) in which the astronauts will live during the stay on the surface and which provides the EVA infrastructure, and the Mars Ascent Vehicle (MAV), that will bring them back to orbit.

4.1.1 System requirements

At the beginning of the study the following requirements were set:

	Achieved?	Max	Min	Ideal	Units
System Requirements					
Operational lifetime of the MEV shall be longer than 35 days			35	35.00	days
MEV shall be as passive as possible during assembly and transfer to Mars					
Exploration and Science shall be performed on the surface					
The crew ascent cabin shall support the astronauts for a period of 5 days			5	5.00	days
The surface habitat shall support the astronauts for a period of 30 days			30	30.00	days
The MEV shall provide communication with EVA, THM , and Earth					
The propulsion stages of the MAV shall insert the crew cabin in a 500 x 500 orbit					
The MAV shall RvD with the TV					
No rover is required to improve the mobility of the astronauts once on the surface					
Mission Constraints					
The MEV shall fit (mass and size wise) in the fairing of the Energya-like launcher					
MEV maximum diameter		6.00		6.00	m
Safety Requirements					
Rescue of the crew and/or abort of mission shall be possible during phases: TBD					
Single failure/fault/operator error tolerance for critical hazards.					
Two failure/fault/operator tolerance for catastrophic hazards.					
Failure detection, isolation and recovery means shall be provided (automatic and manual)					
MEV shall provide automatic detection means for at least the following hazards:					
* Fire					
* Depressurisation					
* Biohazards					
* Atmosphere degradation conditions					
* Radiation					
* Temperature					
* Food spoilage and water contamination					
The MEV shall provide a Caution and Warning System (C&W, this system must be able to receive system data, inform the crew of off-nominal events, and provide sufficient information to direct the crew to the correct response)					

Physiology Requirements					
g-loads should be lower than					
* Mars entry, descent and landing		4.00			g
* Mars ascent		4.00			g
Habitable volume per crew member in the ascent cabin shall be				1.33	m3
Habitable volume per crew member in the surface habitat shall be			16.60	16.60	m3
Surface habitat shall provide minimum conditions to sustaint best possible living and working conditions including egress					
Radiation Organ Specific Equivalent dose Limits (BFO)					
Accute event		0.15		0.15	Sv
30 days		0.25		0.25	Sv
Year		0.50		0.50	Sv
Career		1 to 4		1 to 4	Sv
Surface habitat shall provide medical equipment for the crew					
Surface habitat atmospheric pressure shall be				50.00	Kpa
Surface habitat oxygen percentage in the atmsphere shall be				50.00	%
After landing the crew will require at least 7 days to adapt to the new environment			7.00	7.00	days
Surface habitat shall provide equipment in order to assist the crew reconditioning after landing					
Operational Requirements					
The MEV shall be able to perform the entry descent and landing automatically					
The MAV shall be able to perform the take off ascent and RvD automatically					
Crew shall be able to override the automatic control at any time					
Before take off, the crew wil require 7 days for preparation			7.00	7.00	days
The crew shall perform EVAs every 2 days during the allocated period					
Total EVAs shall be			2.00	7.00	
Number of crew members per EVA shall be 2				2.00	
Time per EVA				6.00	hours
Maximum distance of EVA from MEV		1.00		1.00	km
Assembly in orbit					
The MEV shall be inserted in LEO in only one launch of Energya-like launcher					
Planetary Protection Requirements					
Surface habitat is considered as Earth for PP					
All sample material returned from Mars shall be contained, and containment shall be verified before entering the Earth-Moon system					
Interface requirements					
Interfaces between assembly elements shall be kept to a minimum in order to simplify the assembly					
MEV shall provide interfaces with the THM, allowing the crew to pass from THM to MEV and back					
Interfaces shall be standarised					
Propulsion					
Only chemical storable propellants shall be considered for the MEV (descent and ascent)					

Table 4-1: Mars Excursion Vehicle high-level requirements

One requirement has been modified since the beginning of the study. The design lifetime for the SHM has been extended by one week for contingency situations, namely from 30 days to 37 days.

4.1.2 System design drivers

The main design drivers for the MEV are:

- Habitability, the SHM has to provide the habitat for a crew of three for 30 days, which leads to a free surface of 20 m^2 and a habitable volume of 16 m^3. That is, a total pressurised volume of 79 m^3.
- Rendezvous and docking on the surface is not envisaged, therefore the habitat plus the ascent vehicle have to land together, which leads to a large mass for the entry vehicle and configuration problems.
- The MEV has to fit into the Energia fairing, diameter of less than 6 metres
- EVA and sample collection has to be performed following the planetary protection regulations and recommendations. This leads to the location of the EVA suits and sample

handling devices in the outside part of the SHM. Further analysis on the planetary protection issues needs to be done.

- Power requirements lead to big solar panels, complex to deploy. Fuel cells are envisaged
- Direct link with the ground station on Earth is only available during 50 % of the time, and only during 12 % with the TV, therefore a communications relay satellite is required

4.1.3 Mass budget

The mass budget is shown in Table 4-2.

Mars Excursion Vehicle (kg)			
Total Mass with Margin	46437		
	Descent Module (kg)	**Surface Habitation Module (kg)**	**Mars Ascent Vehicle (kg)**
Total Mass with Margin	4905	19188	22344
Total Dry Mass with Margin	3443	15846	6410
System Margin Applied	574	2641	1068
Structure	0	2769	863
Thermal Control	1722	612	524
Mechanisms	68	1351	791
DLS	620	0	0
Communications	2	25	19
Data Handling	0	37	21
GNC	204	0	158
Propulsion	254	1338	1168
Power	0	2448	91
Harness	0	1000	180
Lifesupport (Dry)	0	3515	658
Consumables			
Dry Food	0	76	11
Drinking Water	0	0	51
Hygiene Water	0	0	18
Oxygen	0	0	13
Packaging	0	27	5
Inorganic	0	11	2
Payload	0	110	584
Astronauts	0	0	285
Total Propellant Mass	1463	3227	15834

Table 4-2: Mass budget for the MEV

4.2 Configuration

The Mars Excursion Vehicle (MEV) is attached to the TV and is the mission element to land on the surface of Mars and takes off after 30 days to rendezvous and dock with the TV.

4.2.1 Requirements and design drivers

- The Mars Excursion Vehicle (MEV) is composed of a Descent Module (DM), a Surface Habitation Module (SHM) and the Mars Ascent Vehicle (MAV)
- The MEV shall provide space for three astronauts

- All main components of the MEV shall fit inside the fairing of the Energia launcher, a cylinder with a diameter of 6 m and a length of 35 m
- SHM shall provide an airlock for Extra Vehicular Activities
- Required pressurised volume of the SHM is 79 m^3
- Required habitable volume of the SHM is 50 m^3
- Required habitable volume of the MAV is 4 m^3
- There shall be a passage for the crew between SHM and MEV
- The centre of gravity has to be as low as possible, for stability reason during landing on Mars

4.2.2 Assumptions and trade-offs

For the configuration of the MEV, two options were considered:

- Horizontal configuration (Figure 4-1)
- Vertical configuration (Figure 4-2).

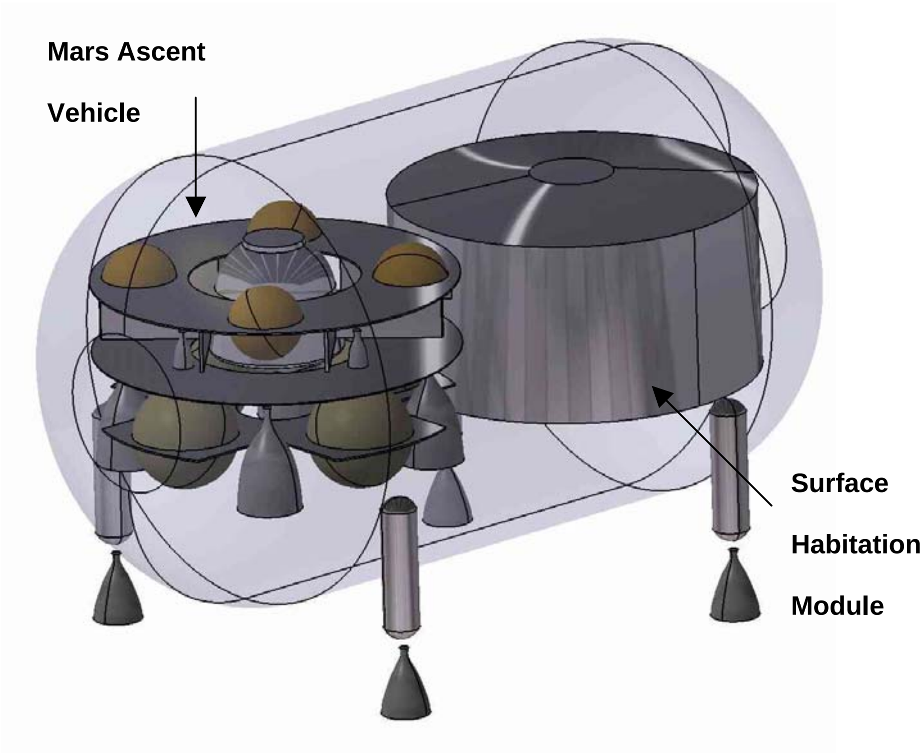


Figure 4-1: Option 1: Horizontal configuration

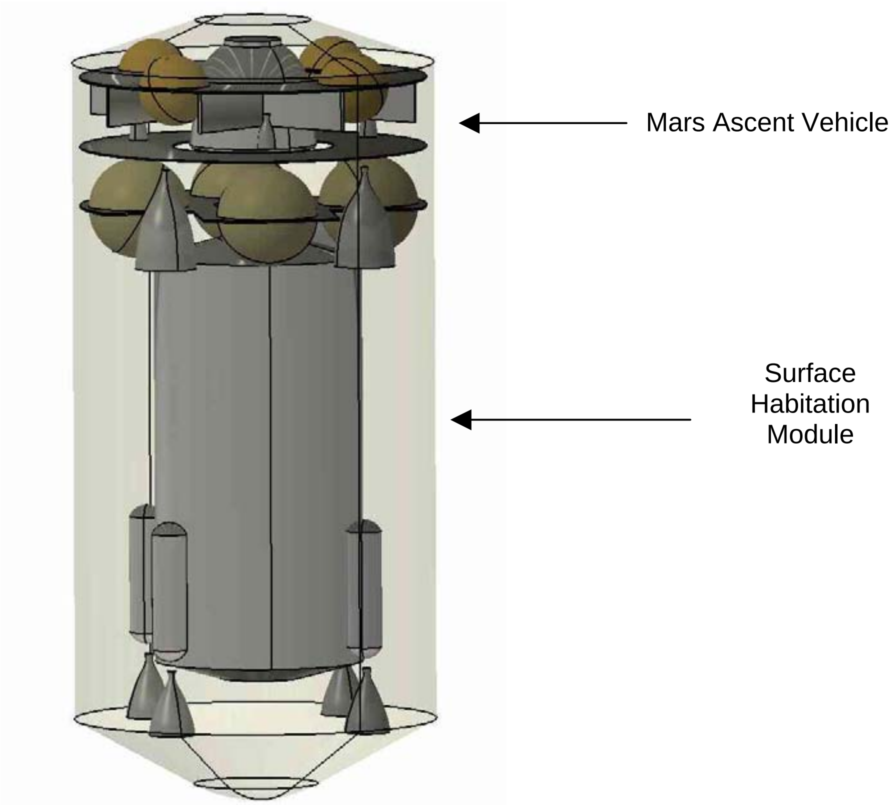


Figure 4-2: Option 2: Vertical configuration

The vertical configuration (option 2) was chosen for structural simplicity, fairing requirement and the passage MAV/SHM requirement.

4.2.3 Baseline design

In Figure 4-3 the whole vehicle is shown. The overall dimensions are presented in Figure 3-5. The SHM and MAV are contained in a heat- and a back-shield. Those elements, part of the Descent Module, are jettisoned after entry in the atmosphere of Mars. Other parts of the DM are the landing legs and the retrorockets; those parts are attached to the SHM.

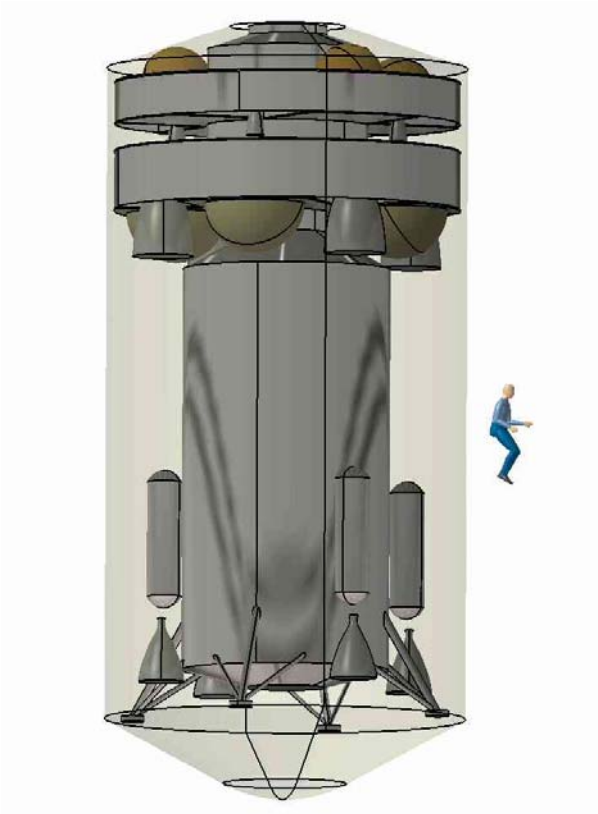


Figure 4-3: Mars excursion vehicle

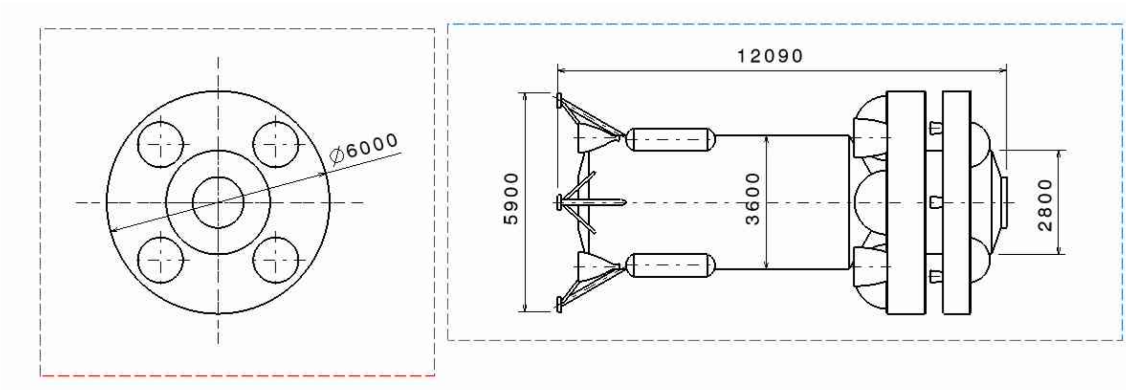


Figure 4-4: Global dimensions Mars excursion vehicle

4.2.4 Surface habitation module

The SHM is the pressurised part of the MEV and consists of a main cylinder (see Figure 4-5). The global dimensions are shown in Figure 4-6. In a dedicated paragraph the interior of this main cylinder is presented.

In Table 4-3 the pressurised volume is presented; it is less than the required 79 m^3. The diameter of the SHM is kept small because free space is needed for the exhaust gasses of the engines of the MAV during ignition. To prevent contamination of Mars, the SHM may not been damaged during the take-off of the MAV. So, to increase the volume of the SHM, than the height of the cylinder has to be increased.

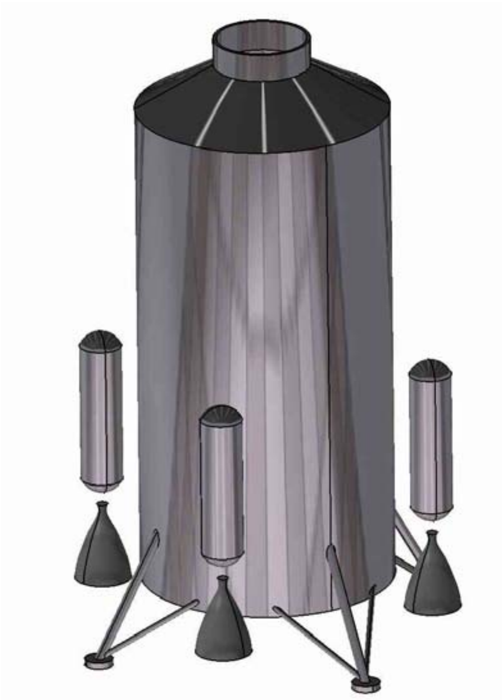


Figure 4-5: Surface habitation module

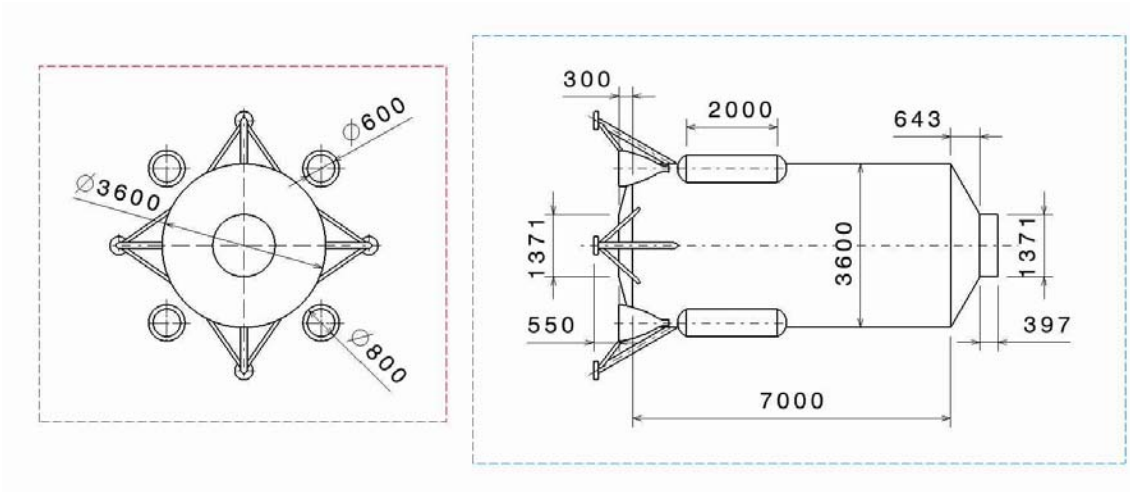


Figure 4-6: Global dimensions surface habitation module

	Length [mm]	Internal Diameter [mm]	Volume [m^3]
Cylinder	7000	3500	67.3
Lower Closure	300		1.5
Upper Closure	643		3.9
Total Pressurised Volume			**72.7**

Table 4-3: Dimensions and Volume main cylinder

4.2.5 Mars ascent vehicle

In Figure 4-7 the complete MAV is shown. It is composed of a capsule (see Figure 4-8) and two propulsion stages. In Figure 4-9 the MEV is shown after the first propulsion stage is jettisoned. In Table 4-4 the pressurised volume of the capsule is presented.

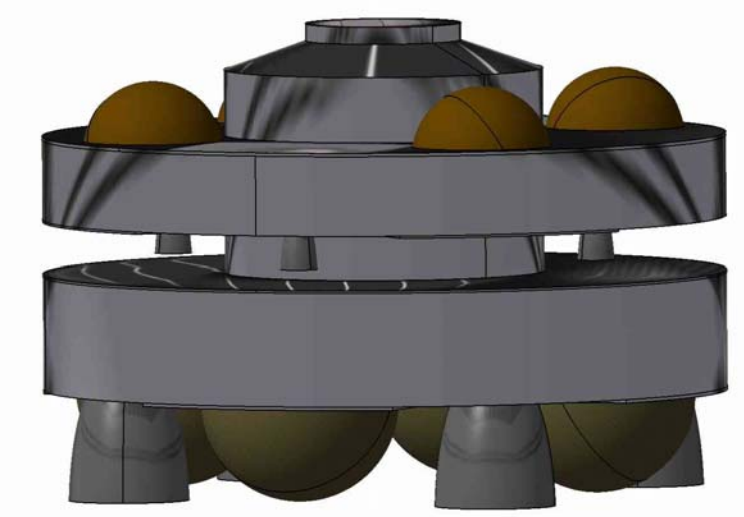


Figure 4-7: Mars ascent vehicle

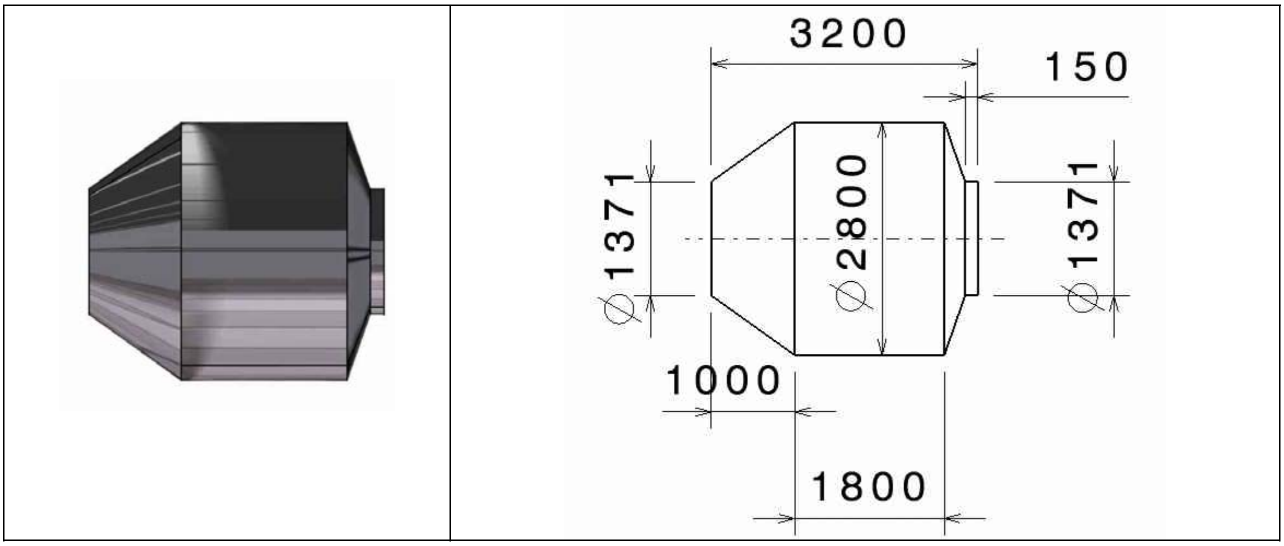


Figure 4-8: Capsule of the MAV with the conical interface to surface habitation module

	Length [mm]	Internal Diameter [mm]	Volume [m^3]
Cylinder	1800	2700	10.3
Lower Closure	1000		3.5
Upper Closure	150		0.8
Total Pressurised Volume			**14.6**

Table 4-4: Dimensions and volume capsule

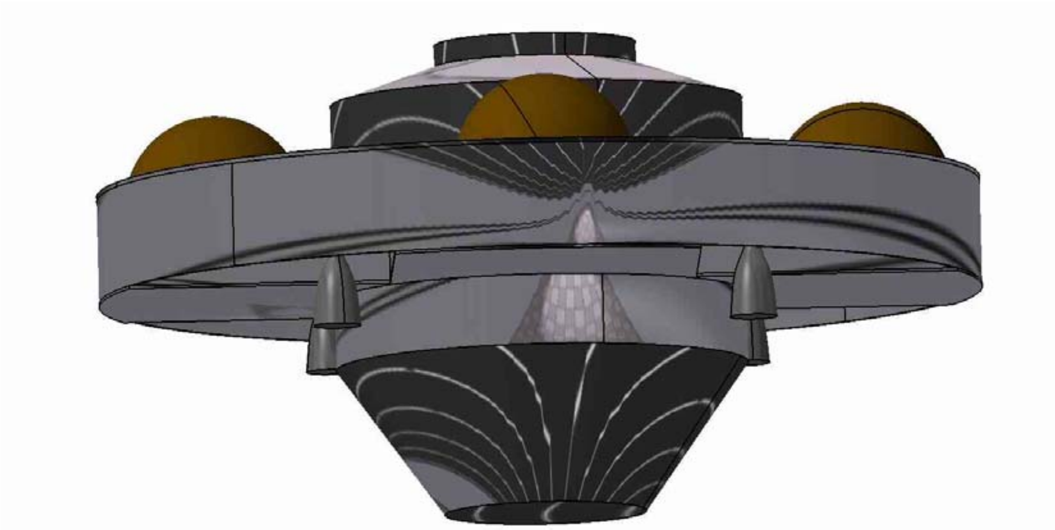


Figure 4-9: Mars excursion vehicle; Capsule with its second stage

4.3 Surface habitation module

4.3.1 Internal configuration

The development of the baseline design for the SHM included an investigation on the usage of inflatable technologies. Before the final baseline design, three main steps (including eight different designs and configurations) were developed followed by two detailed designs of different interior layouts for the three main zones: the private, the personal and the social zone. (See THM internal configuration for definitions 3.3.1)

4.3.1.1 Preparing the baseline design – investigations of SHM's interior configuration

The final diameter for the SHM is 3.60 m, derived from the limitations of the launch vehicle /diameter of 6.00 m) and the safety distance required between the plume of the MAV during take-off and the ground or the habitat.

Where and how to place the life support system optimally still remains an open issue. Three options were considered: at the top level of the habitat, in the middle, and at the bottom. The option with the LSS at the top level was discarded, because the mass of the LSS would have been too high up, considering the fact that the MAV with the propulsion tanks already has a substantially high weight within the overall MEV configuration. Prior to a final conclusion, the following two options were developed:

4.3.1.2 Option 1 – LSS in the middle of the habitat

SHM - cylinder:

Height: 7.00 m
Diameter: 3.60 m
LSS: 10 m^3 in the middle of the habitat

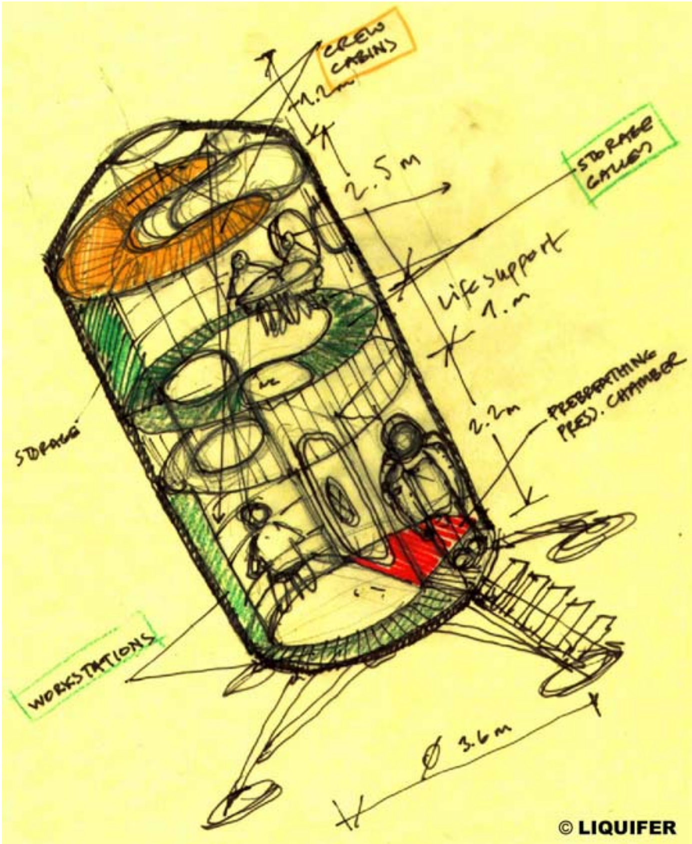


Figure 4-10: Axonometric sketch of option 1 with LSS in the middle of the SHM

This has the following advantages:

- Better access for maintenance of LSS
- Divides the habitat into compartments
- Rigidizes the structure of the shell
- Sets a distinction between private, personal and social space

Figure 4-11 shows the interior configuration for option 1. Placing the LSS in the middle of the SHM creates a natural distinction between zones. The main working zone is placed at the bottom next to the EVA area. Above the LSS are the private and personal spaces with the sleeping quarters at the top, close to the emergency exit into the MAV. This distinction creates distance while still having the possibility to overlook the whole module from the private cabins.

The best location for the LSS is at such a height so that the centre of mass is as low as possible, which is optimal for landing. As the MAV with all the propulsion is already very heavy, the other heavy part of the SHM, the LSS, should be located at the bottom.

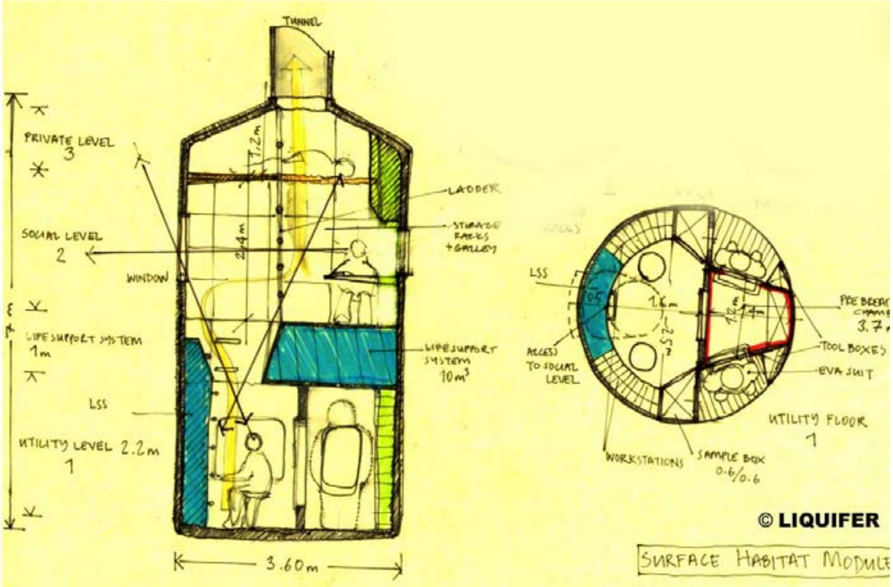


Figure 4-11: Section through the SHM and plan of the lower level – workspace and EVA access

The left half of Figure 4-11 shows the different zones of the habitat. The upper part of the habitat is the private zone with some cocoon-type crew quarters (bunk beds). Below is the social zone with the galley, the hygiene facilities, a table to accommodate all three crewmembers and the stowage area (marked in green). Arrows indicate the line of sight to enlarge the space on a perceptive level. Even from the lower level astronauts can look outside through the window placed near the table. The curved yellow arrow points towards the emergency exit – an easy path to follow and also in the line of sight.

The right half of Figure 4-11 shows the lower level with the workstation, the sample exchange box and the EVA suits docked to the SHM.

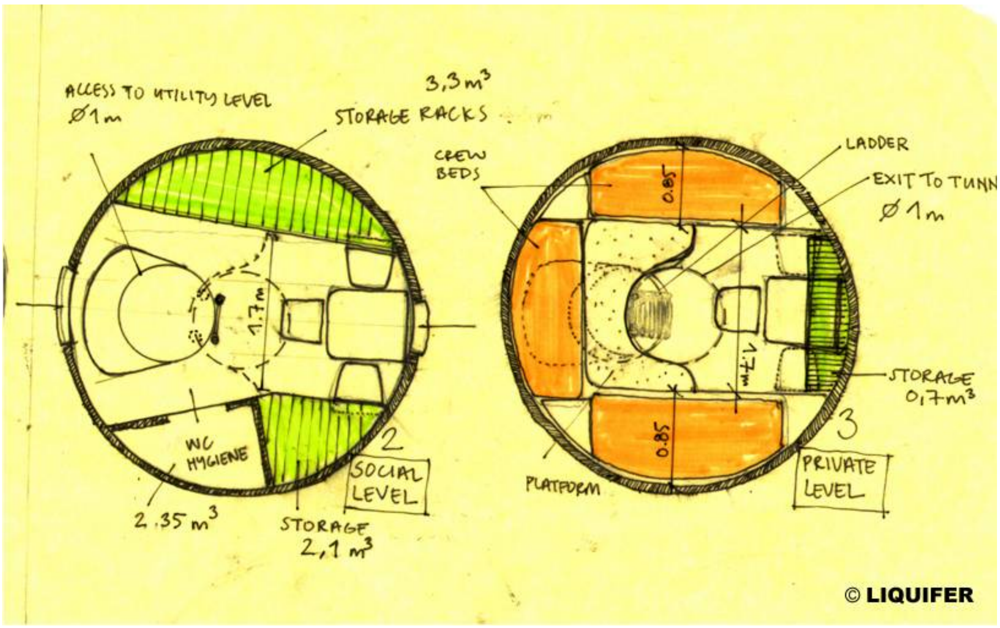


Figure 4-12: Section through the SHM (L), plan of the lower level (R) – workspace and EVA access

Figure 4-12 shows the main level with the social area and its functions, and indicates placement of the crew cabins (marked in orange) on the right.

4.3.1.3 Option 2 – LSS at the bottom of the habitat

SHM - cylinder:

Height: 7.00 m
Diameter: 3.60 m – 4.4 m (bottom level)
LSS: 10 m^3 at the bottom of the habitat

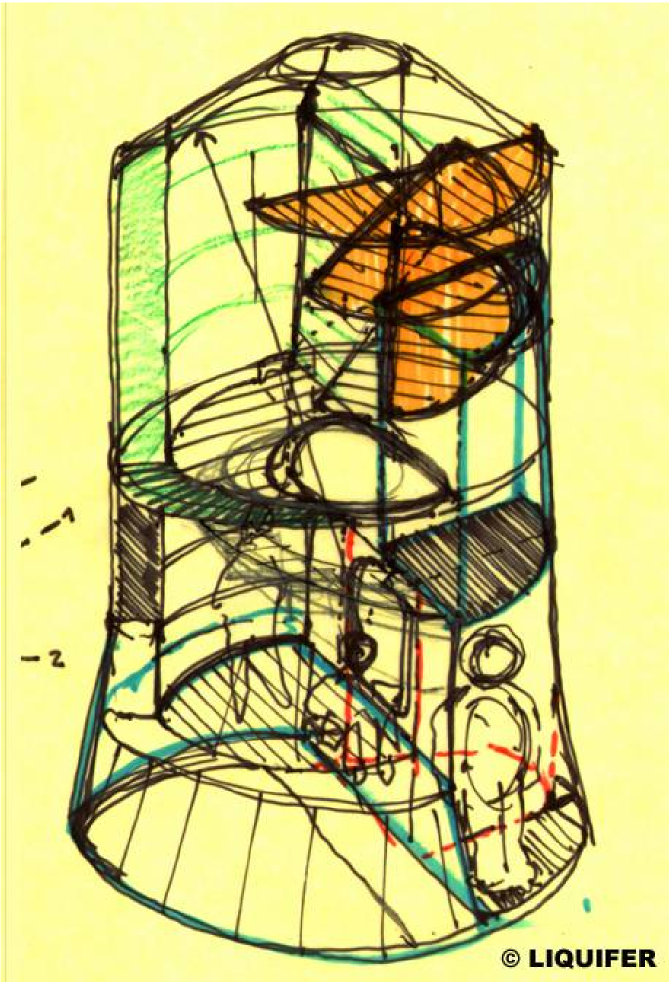


Figure 4-13: Sketch of Option 2 with LSS at the base

This has the following advantages:

- Viewing axis towards emergency exit
- Creates a strong orientation and good overview over habitat

  Axis of sight makes the space look bigger and gives astronauts an opportunity to look outside
- Centre of mass is at the bottom

Placing the LSS at the base (marked blue) of the SHM creates a strong orientation and good overview throughout the habitat. The section on the left shows that the working area is placed in a split level between the EVA area (marked red) and the main social zone. From there the astronaut can overlook the whole habitat, without interfering with the private space of fellow astronauts (marked orange).

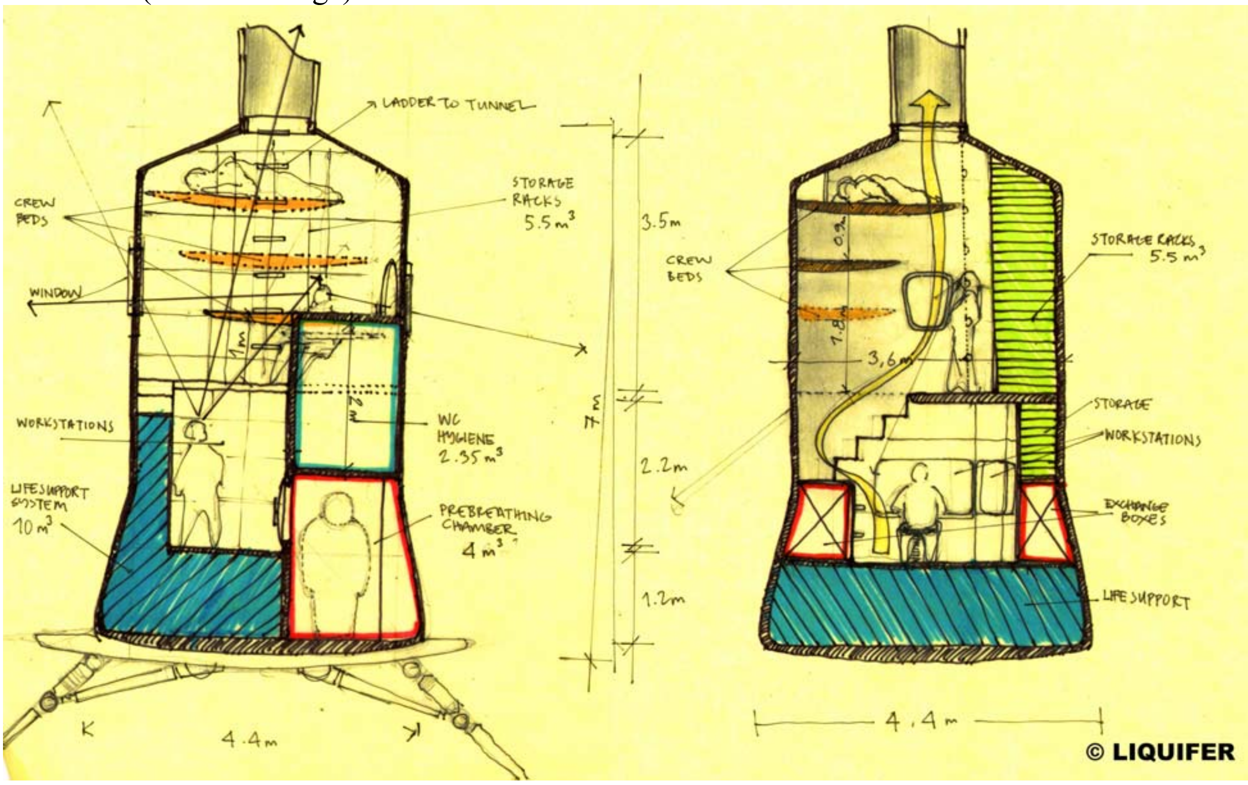


Figure 4-14: Sectional drawings of Option 2

The two sections explain in detail the different zones, functions and infrastructure of the habitat. On the left the different lines of sight are indicated by black arrows to enlarge the space on a perceptive level. The overview over the habitat is possible from each point of view but through introducing a split level zoning is made possible and therefore creates a distinct set of different spaces allocated to different functions and crew performance.

The upper part of the habitat is the private zone with some cocoon-type crew quarters (bunk beds) which can be also used as stairs for a secondary option of circulation. Below the social zone with the galley, are located the hygiene facilities, a table to accommodate all three crewmembers with a window and the stowage area (marked in green on right section).

The yellow curved arrow on are the right-hand side of Figure 4-14 points towards the emergency exit – an easy path to follow and also in the line of sight. This translation path is quite roomy and free of obstacles.

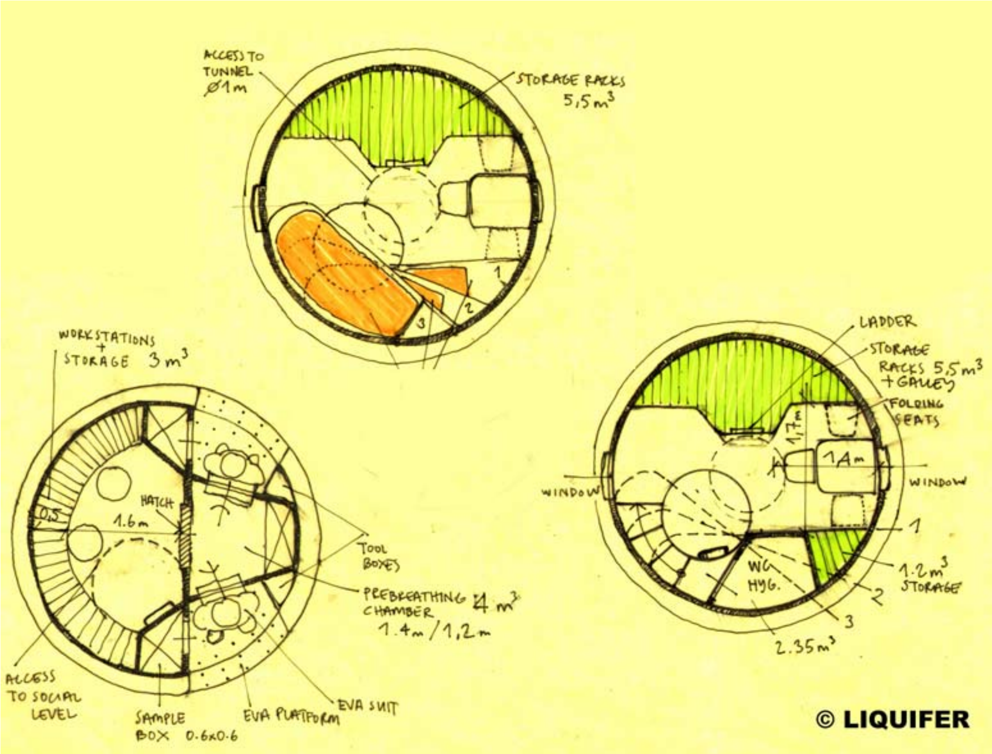


Figure 4-15: Plan of lower level, upper level and mid level, respectively, from left to right

Figure 4-15 shows the different levels with their functional layout. On the bottom left is the lowest level showing how the workstation, the sample exchange box, the toolboxes and the EVA suits docked to the SHM are integrated. In the middle right is the social level indicating the relation of the window and the table, the galley and the circulation, the hygiene facilities and the stairs coming up from the EVA deck.

This design approach was finally chosen because the heaviest part is at the bottom of the habitat and the spatial design has the most advantages and different layers of perception and habitability which makes the habitat user friendly. Also through the whole process of different options this one developed integrating all the advantages and important factors, which have been investigated earlier.

4.3.1.4 General safety issues

The following general safety issues have been taken into consideration:

- 20% of the volume has to be dedicated to ducts and pipes – this volume might be added with easy access to all ducts and pipes for maintenance
- Enough fire detectors and isolation and recovery systems should be provided to enhance the safety of the crew
- The LSS of the MAV should function independently from the SHM there must be two air-tight compartments:
    1. MAV
    2. SHM

4.3.1.5 Baseline design

With the previous proposal taken further in its conceptual detail, this baseline design was developed. Zoning of the SHM is defined by functions such as working, EVA, private areas (marked orange) and achieved by spatial planning. The LSS is placed at the bottom of the SHM, allowing free translation throughout the habitat. Figure 4-16 shows a large axonometric view of the habitat, which gives an overview of the SHM configuration.

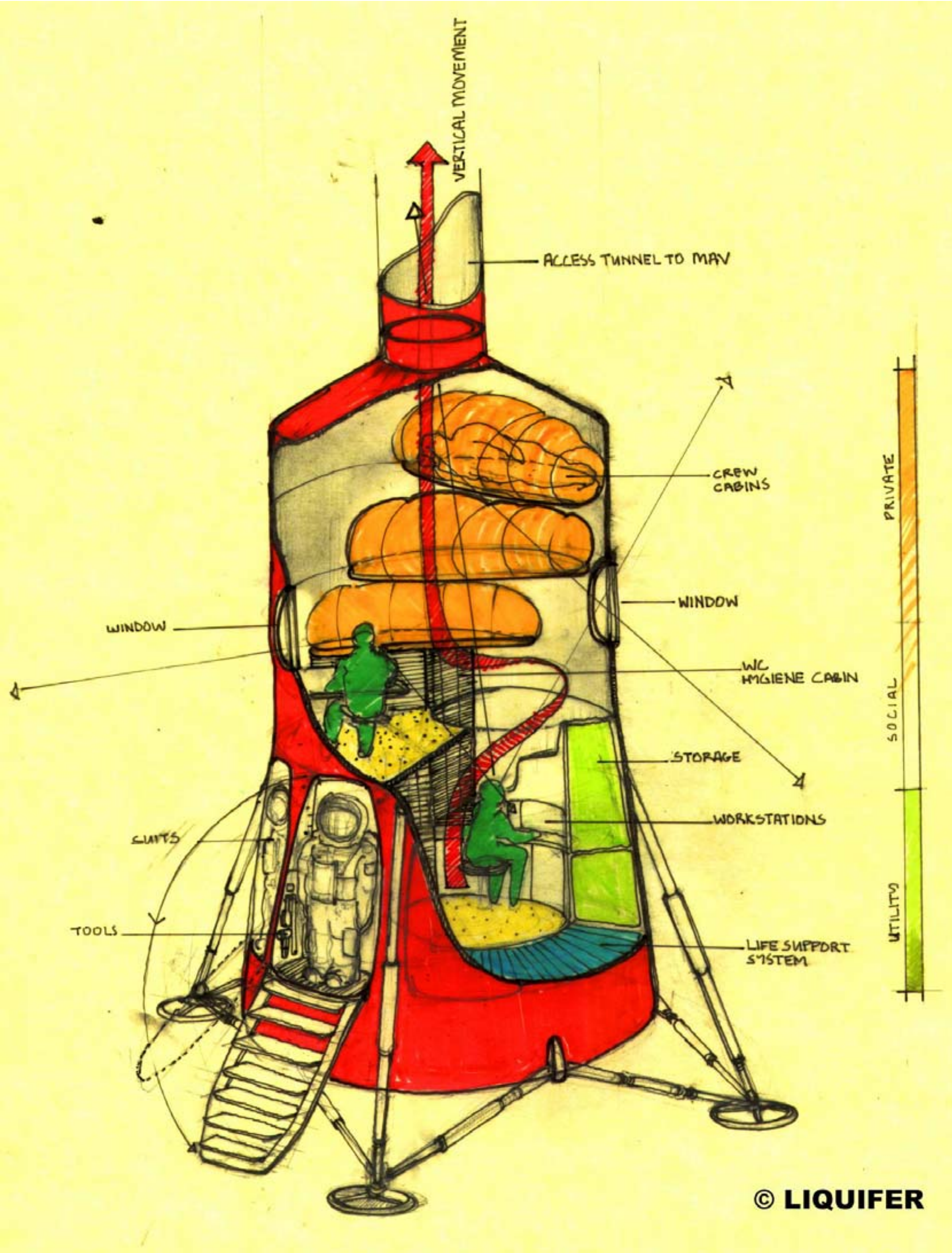


Figure 4-16: Axonometric of the final design status

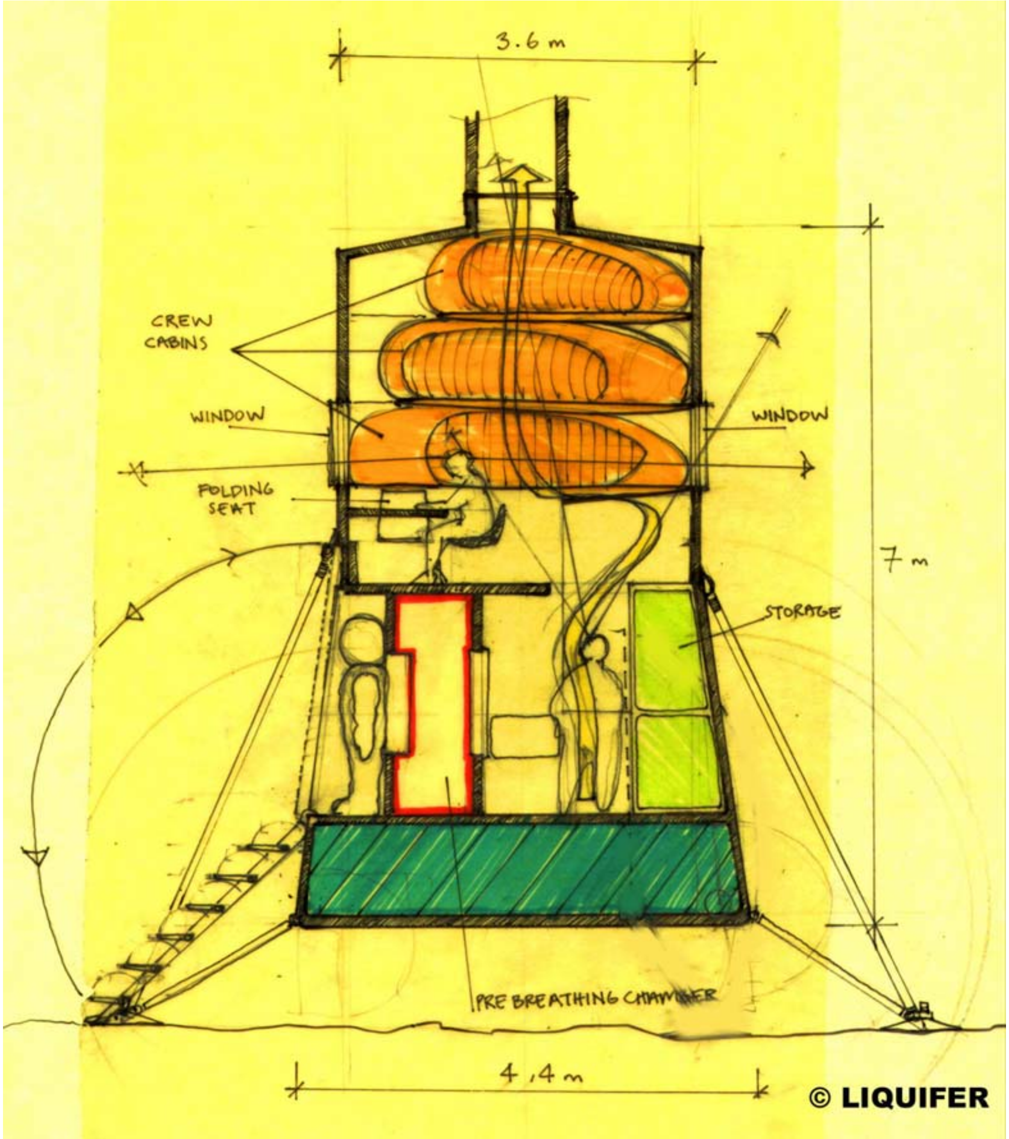


Figure 4-17: Section of the final design status

Figure 4-16 shows the different zones of the private crew cocoons (marked in orange), the social level (near the table and the window) and the working zone below. Lines of view and translation paths are marked with arrows.

This configuration is conical towards the top and allows integrating the LSS on the bottom, lowering the centre of mass.

4.3.1.5.1 Lower level

At the lower level is the EVA area and decompression chamber. The EVA system allows quick access from the inside to the outside. One spacesuit for each crewmember is provided. The study report assumes is that two astronauts go on EVA as a buddy team, all the crew members will perform EVA in rotation. On the side there are the toolboxes and the sample exchange box. The hygiene compartment is on a split-level between the main deck and the lower level. See Figure 4-18:

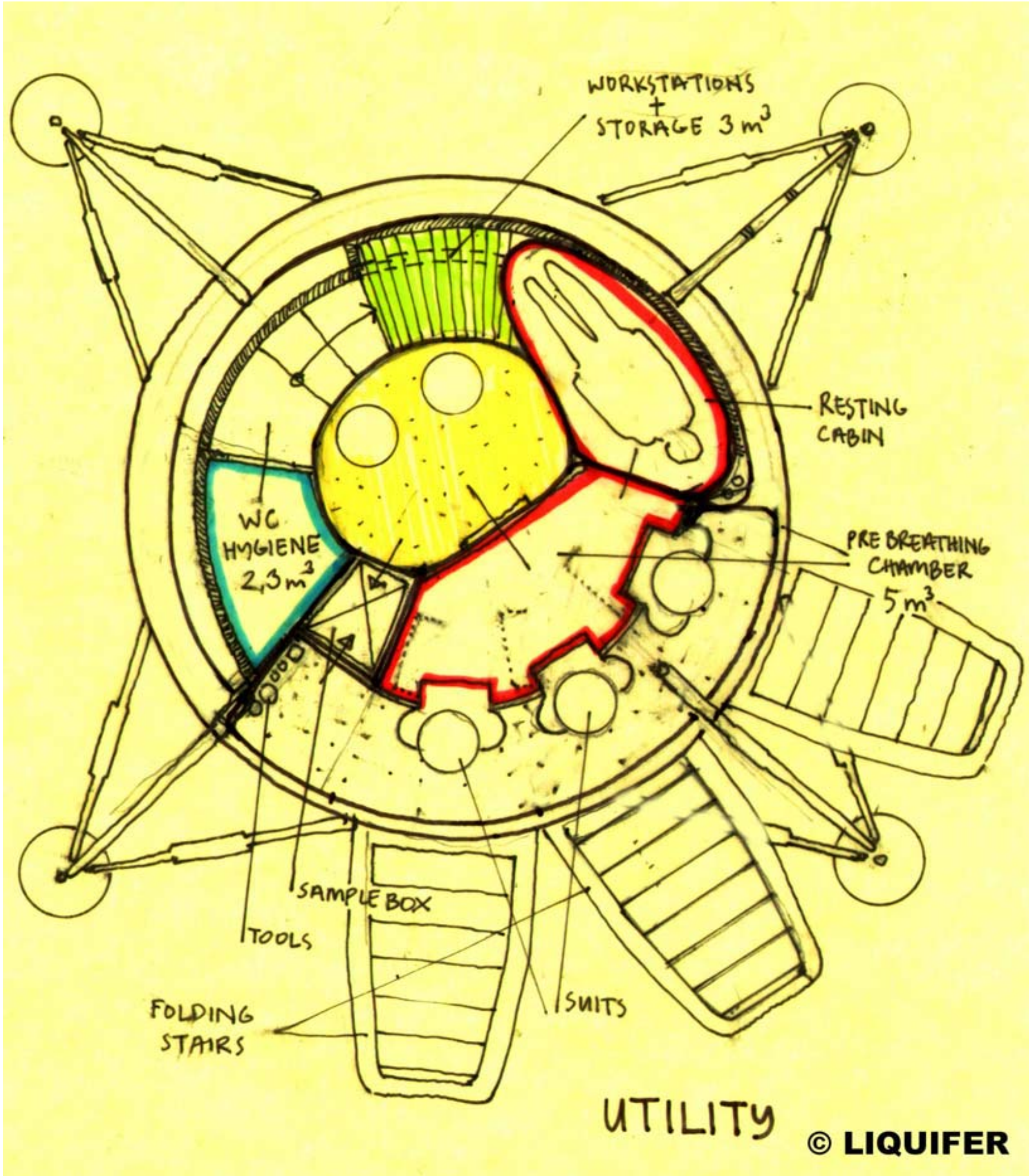


Figure 4-18: Plan of lower level

Figure 4-19 shows the sequences for EVA. The Mars walker returns to the base and connects the suit to the inner hatch. The inner hatch is opened together with the back of the Mars suit. In this way system contamination is prevented and only a small volume of gas has to be pumped away.

When going for EVA, after prebreathing, the protection cover for the spacesuits folds down and reveals a ladder for the Mars walker to climb down.

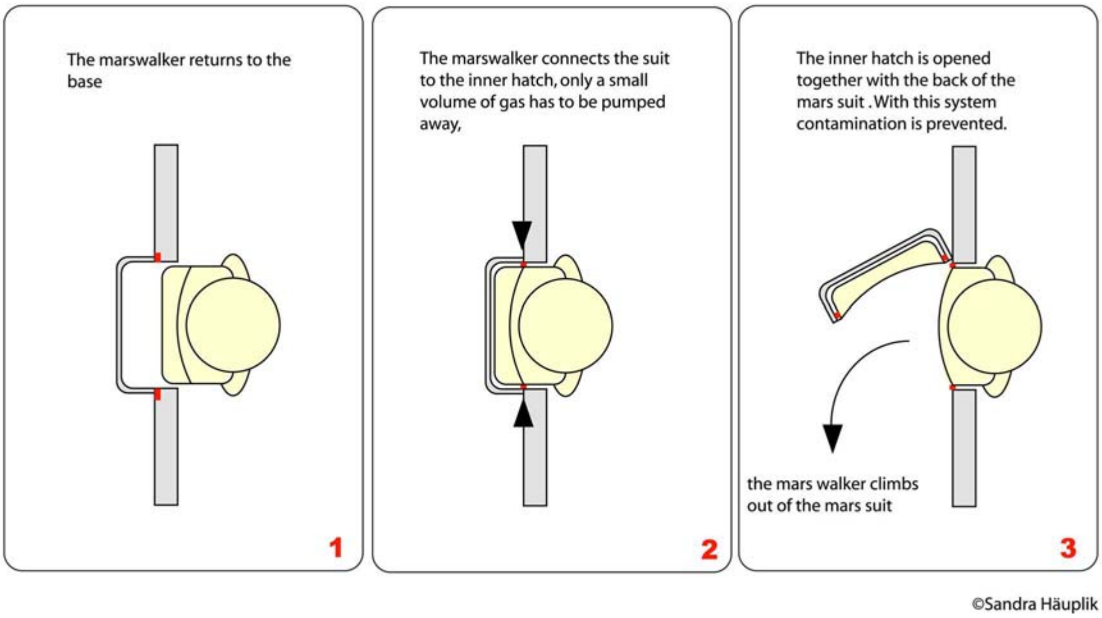


Figure 4-19: "Airlockless"-Airlock System, currently further developed at EADS, Germany

4.3.1.5.2 Middle level

This level, as shown in Figure 4-20 provides the main social area with a table to accommodate all crewmembers with folding chairs. A window at the table enables an overview of the outside area. A second window is placed on the opposite side.

While sitting at the table the crew has an excellent overview over the whole habitat, as well as over the surface of Mars outside. The table is used for gathering, conferencing, observing and recreation. A galley and necessary infrastructure – all marked in green – is located here and an easy access to the hygiene facility is ensured. The stowage system (marked green) on this level allows quick and rational access.

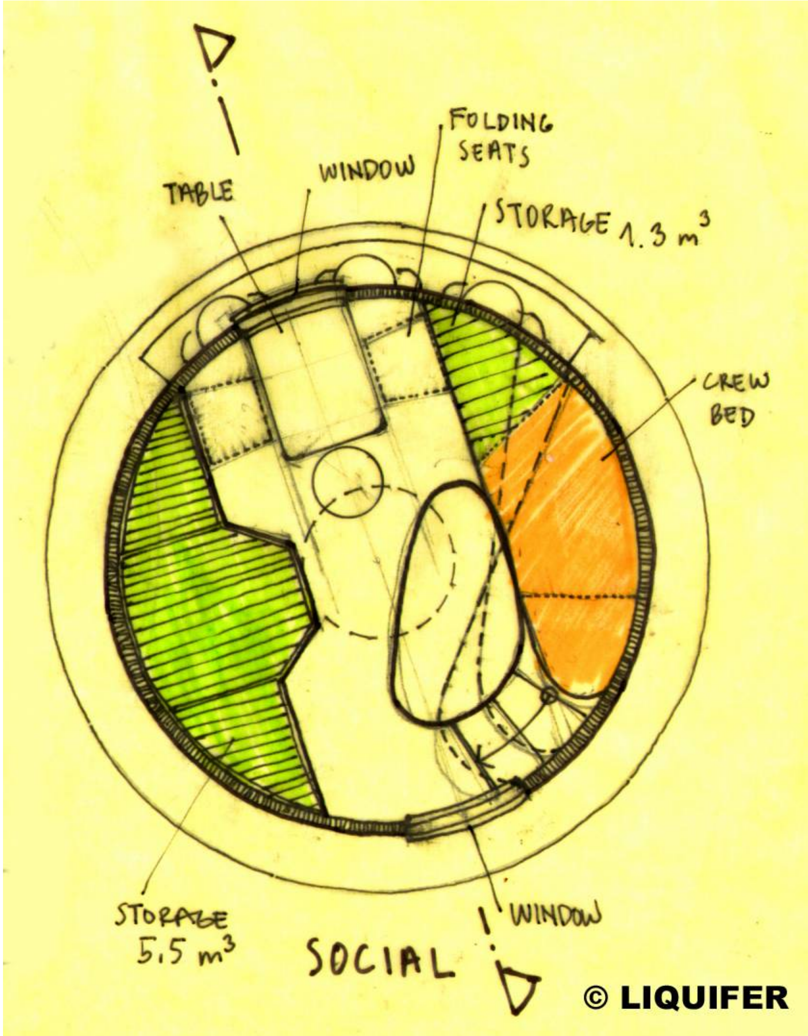


Figure 4-20: Plan of mid level

4.3.1.5.3 Top level

The crew cabins (marked in orange) are placed at the very top of the SHM – see Figure 4-21. One rationale for that is the closeness to the escape tunnel to the MEV. While resting, the crewmembers are able to observe the whole habitat, with the possibility of creating a private zone by cocooning themselves. Also the protection against radiation is better here, because the crew quarters are right underneath the MAV with its propulsion tanks.

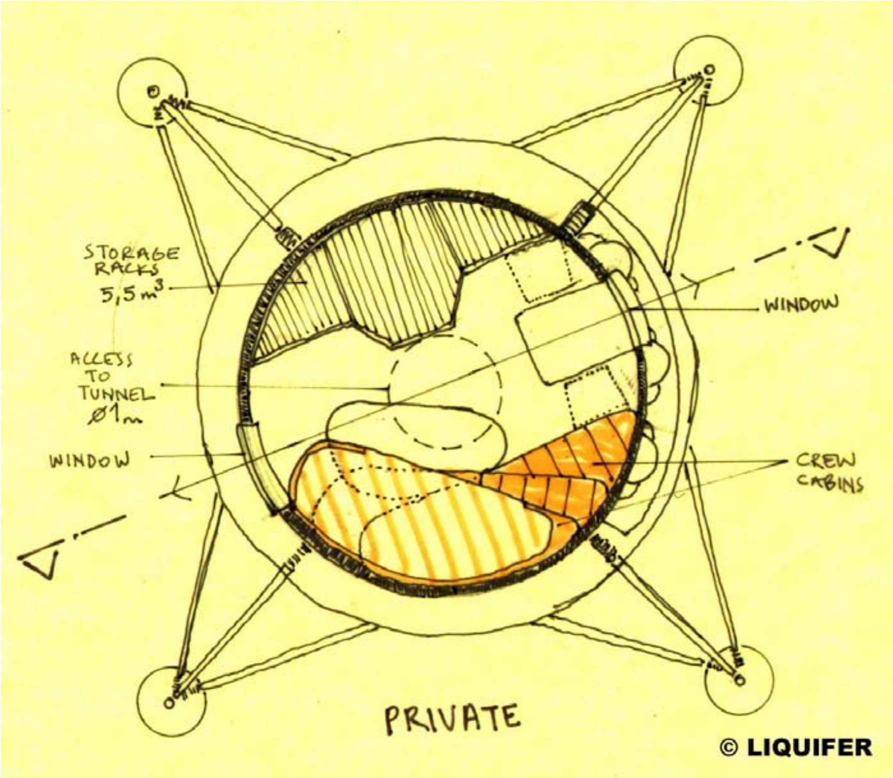


Figure 4-21: Plan of top level

4.3.2 Propulsion

4.3.2.1 Requirements and design drivers

The MEV mass prior to the landing is 42 tonne.
The gravitational acceleration is assumed 3.7 m/s^2
The engines for the descent are fired from a height of 2000 m from the Martian surface.

4.3.2.2 Assumptions and trade-offs

The starting descent velocity assumed feasible from the parachute design is 107 m/s and the maximum allowed impact velocity on the Martian surface is 2 m/s.

The choice of the thrust is obtained excluding the contribution of the Martian atmosphere, therefore the drag is not considered and the thrust level is slightly oversized.

The descent manoeuvre can be obtained with a thrust level of 280 kN with a short firing time of around 37 seconds. Due to the exceeding thrust a new design of the engine with high ability to perform full throttle (30-40 %) is required.

Only storable bi-propellant has been considered.

No attitude and steering manoeuvres have been considered.

4.3.2.3 Baseline design

Four YUZHNOYE RD 861-G gas generator bi-propellant NTO-UDMH thrusters have been chosen as propulsion system for this module.
The engine derives from a well known thruster used in Tsyklon 2nd and 3rd stage. Recently this engine was improved with an increase of the Isp performances, a higher restart capability and reduced mass. This engine was also studied as possible application for the VEGA.
The propulsion system for Surface Habitation Module presents the following characteristics:

Characteristic	Value
Number of thruster	4
Thrust	76.5 kN
Isp	325 sec
Exit diameter	808 mm
Length	1600 mm
Thruster mass	185 kg
Propellant	UDMH/NTO
O/F ratio	2.4
Number of tanks	2+2 Cylindrical
Tanks material	Ti
Max MEOP	7 bar
Mass of UDMH tank	11.1 kg (each)
Mass of NTO tank	16.5 kg (each)

Table 4-5: Landing propulsion system summary

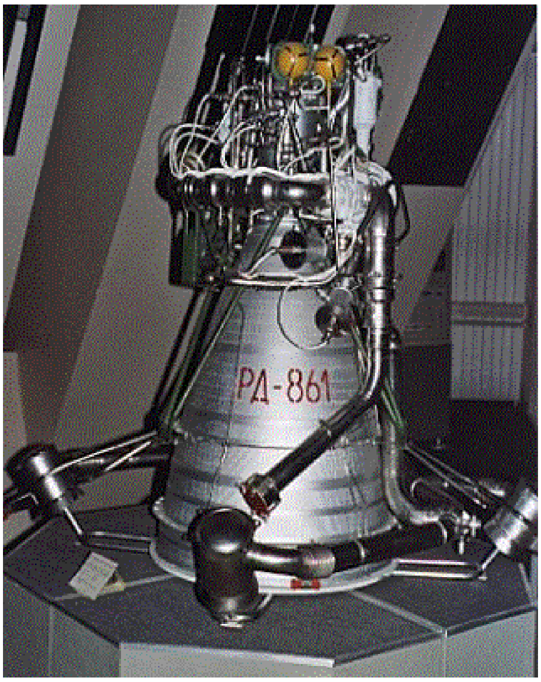


Figure 4-22: YUZHNOYE RD 861-G

4.3.2.4 Budgets

The propellant mass is 3227 kg and the propulsion dry mass (including margins) is 1338 kg.

This mass includes an estimation of thrusters mass, the tanks, and a rough estimation of feedlines, valves and regulators, propulsion thermal control, avionics, actuators. It does not consider the structure of the propulsion system, power and communication.

4.3.2.5 Options

A possible option is to use a single engine of 240 kN. This has been preliminary discarded due to poor performances in thrust level and configuration issues due to the envelope. The engine selected was the Russian RD 0235 used in the Rockot second stage.

4.3.3 Environmental control and life support system

The life support comprises the following subsystems:

- Atmosphere Supply and Control
- Atmosphere Revitalization
- Temperature and Humidity Control
- Water management
- Waste management
- Food management
- EVA provisions
- Hygiene
- Crew accommodation
- Medical equipment

Crew accommodations have been added to the classical set of life support functions as the crew accommodation engineering domain does not have a separate workbook in the CDF study, in which hardware specifications could be added.

4.3.3.1 Requirements and design drivers

The MEV complex consists of two main modules: the Surface Habitation Module (SHM) and the Mars Ascent Vehicle (MAV). The study required both modules to be equipped with life support systems, which are not interconnected. Therefore, two life support systems are presented here.
The SHM life support system is designed to provide life support to a crew of three for 37 days.

4.3.3.2 Assumptions and trade-offs

4.3.3.2.1 Metabolic requirements of the crew

The metabolic needs of the crew have been calculated using the correlations given in ESA standard PSS-03-406 and crosschecked with relevant sources. The entire calculations have been based on the energy expenditure of the crew. The schedule for crew activity is shown in Figure 4-23.

Schedule in hours for the most active day			
Activity	**Astronaut**		
	1	**2**	**3**
Sleep	8.00	8.00	8.00
Pre- and post sleep	1.50	1.50	1.50
Leisure activities	2.00	2.00	2.00
Personal hygiene	1.00	1.00	1.00
Eating	1.50	1.50	1.50
Exercise	0.00	0.00	0.00
Station keeping	1.88	1.88	1.88
Laboratory activities	7.10	7.10	7.10
Metabolic Cost of EVA			
EVA mission tasks	0.70	0.70	0.70
EMU donning/doffing	0.15	0.15	0.15
Egress/ingress	0.06	0.06	0.06
Pre-EVA setup & post EVA EMU care	0.12	0.12	0.12
TOTAL TIME (24hrs)	**24**	**24**	**24**

Figure 4-23: Crew activity schedule during Martian surface stay

Note that for the energy requirements computation that a 24-hour day has been used instead of a Martian day of 24 hours and 40 min. The error introduced is less than 3% and is covered by the contingencies.
Based on the energy expenditure, the metabolic needs and products by the crew have been estimated and are shown in Figure 4-24:

	Per day	**Per mission**
Energy consumption (W*h)	**2868.9**	**106147**
Energy consumption (J)	**10327860.0**	**382130820**
Oxygen consumption (m3)	**0.5**	**19**
Oxygen consumption (kg)	**0.7**	**27**
Drinking water (m3)	**0.0**	**0**
Drinking water (kg)	**8.5**	**315**
Dry food (kg)	**2.0**	**75**
Carbon dioxide production (m3)	**0.4**	**15**
Metabolic water production (kg)	0.3	11
Urine production (kg)	**4.3**	**158**
Faecal liquids(kg)	**0.2**	**9**
Insensible water (kg)	**4.3**	**158**
Total solid waste production		
Faeces (kg)	**0.2**	**8**
HYGIENE WATER (kg)	**3**	**111**
GREY WATER (KG)	**7**	**269**
ADDITONAL SOLID INORGANIC TRASH	**0**	**11**

Figure 4-24: Metabolic needs and products of the crew

The data shown in Figure 4-24 suggest a mass of consumables of about 671 kg. Given that consumables need additional hardware for storage and use, as well as the need to treat and store the metabolic products, the use of an open-loop system seems favorable. The data strongly suggest the use of open-loop systems except a recovery system for condensate. Furthermore, if a fuel cell option is used for power generation, the product water could be used as consumable in life support. Currently, such an option is estimated to produce roughly 800 kg of water, which is more than the demand by the crew.

4.3.3.3 Hygiene water

The hygiene water allowance for the crew has been estimated to be 1l/crew/d. It is understood that this figure is rather low. However, the supply could be supplemented by using the collected condensate water, which amounts to approximately 1.42 l/crew/d. This figure is assumed to be sufficient. For full body cleansing and hand washing, the crew would rely on wet towels similar to those provided to astronauts on Russian spacecrafts.

4.3.3.4 Drinking water

The drinking water allowance for the crew has been estimated as follows. The water release by the crew has been calculated using standard correlations based on the energy expenditure. A literature review revealed that the water intake by the crew is about the same as to the water release by the crew. Therefore, the amount of water intake has been calculated using the numbers for the sensible and insensible water quantities. The advantage of this method is that the potable water estimate is based on the energy expenditure.

4.3.3.5 Cabin atmosphere

The cabin atmosphere has been selected as follows:

Total Cabin Pressure: 50.0 kPa
Partial Pressure Oxygen: 25.0 kPa
Partial Pressure Nitrogen: 25.0 kPa
Partial Pressure Carbon Dioxide: TBD

Preferably, the atmosphere would be free of any contaminants. However, as a minimum requirement, the spacecraft atmosphere shall adhere to the requirements given in ESA PSS-03-401. Based on the experiences with long-term pressurised spacecrafts there shall be more stringent limits on microbial contamination. The following limit has been proposed during this study:

Total microflora count: 200 CFU/m^3 (CFU - colony forming units)

4.3.3.6 Waste production

Besides the already presented production of faecal material by the crew, the crew will produce additional organic and inorganic waste. Organic waste will consist of hair, nail clippings, skin material, kitchen waste, food leftovers. The total amount of such organic waste has been estimated to be 0.1 kg/crew/day. It was not possible to quantify the total amount of inorganic waste produced by the crew per day due to the lack of data. However, reviewing existing data and other sizing tools, the amount of inorganic waste produced by the crew per day was estimated to be around 0.6 kg/crew/day. This includes:

0.05 kg/d cleaning supplies
0.1 kg/d waste collection supplies
0.1 kg/d contingency collection mitten bags
0.1 kg/d hygiene supplies
0.2 kg/d wet wipes for house cleaning

However, a significant fraction of the inorganic waste will come from the food packaging. For this study the ratio of packaging weight to food content has been set to:

0.4 $kg_{packaging}/kg_{dryfood}$

4.3.3.7 EVA considerations

An efficient manned mission to Mars needs to take advantage of the unique skills of human beings. Extravehicular activities must be maximized without compromising the safety of the crew and equipment as well as keep within reasonable budgetary boundaries.
The study started by investigating current-day EVA capabilities and protocols and determine their applicability to the CDF Mars mission scenario.

4.3.3.7.1 EVA scenario

A crew of three is landing on Martian surface for duration of 30 days. The first week will be used to perform post-landing tasks inside the MEV and to get accustomed to the gravity on the surface. About two weeks of surface stay will be used for seven sorties of a crew of two. The remaining time is used for pre-departure tasks.

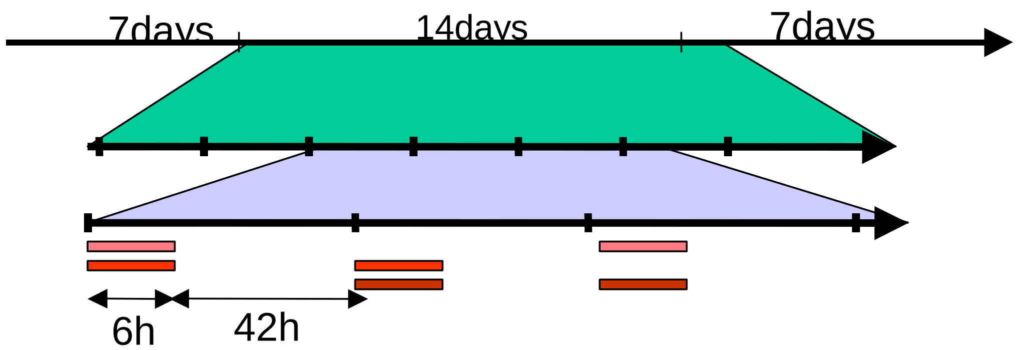


Figure 4-25: EVA scenario

Figure 4-25 illustrates the anticipated time of 6 h EVAs. Given that EVAs are scheduled for every two days, the minimal time between two scheduled EVAs is 42 h. This time is greater than the 36 h for which flight protocols exist today. Therefore, as a first cornerstone, the study assumed that such protocols are valid for mission to Mars. This implies using current technology and knowledge to assess the needs and issues concerning Martian EVAs.

4.3.3.8 Atmospheric composition

Different aspects drive the atmospheric composition selection. To schedule flexible EVAs and have the fastest contingency EVA capabilities, the SHM and MAV could use a 100% oxygen atmosphere at non-toxic pressure. However, an oxygen-rich atmosphere increases the risk of fire, poses constraints on the material selection and engineering of the MEV, and perhaps an over pressurisation during reentry and ascent will be necessary. A standard composition atmosphere of 101.3 kPa would allow flexible on-ground testing and references. Conversely, it would increase the required pre-breathing time rendering the schedule less flexible and increases the air losses.

Given a standard atmosphere (101.3 kPa, 21% O^2), the pre-breathing time would range between 30 min (Russian protocol, TR=1.8, suit pressure 39.2 kPa) and 4.5 h (American protocol, TR=1.6, suit pressure 29.6 kPa). However, both protocols assume the possibility to return to Earth quickly to treat an astronaut if symptoms of decompression sickness (DCS) occur. Such options will not exist for a mission to Mars. Therefore, more conservative pre-breathing time estimates must be considered, which would cause a significant increase in EVA preparation time. However, the scheduled activities for an astronaut on an EVA day would exceed 11 h so there would be no advantage.

Therefore, additional atmosphere compositions have been investigated that depend on the allowable tissue ratio. The results are shown in Table 4-6 and are valid for a 26 kPa suit pressure.

	TR=1.6	TR=1.4	TR=1.2
101.3kPa, 21Vol% O2	284	341	408
70.14kPa, 26.5Vol% O2	95	153	219
50kPa, 50Vol% O2	0	0	0
28kPa, 100Vol% O2	0	0	0

(suit pressure 26 kPa, 100% oxygen, half time constant 300min)

Table 4-6: Pre-breathing time as a function of suit internal pressure (min)

As shown, an atmosphere composition of 50% nitrogen and 50% oxygen at 50 kPa total pressure would be a suitable compromise, as it seems to avoid pre-breathing and still provide some marginal fire hazard reduction. NASA STD 3000 reports that the fire hazard is reduced by 50% in comparison to a 100% oxygen atmosphere yet still poses an increased fire hazard. As Figure 4-26 shows, the composition is within the zone of unimpaired performance of the astronauts. Discussions with medical personnel indicated that short and long-term health concerns for the crew due to the exposure to this atmosphere are not expected.

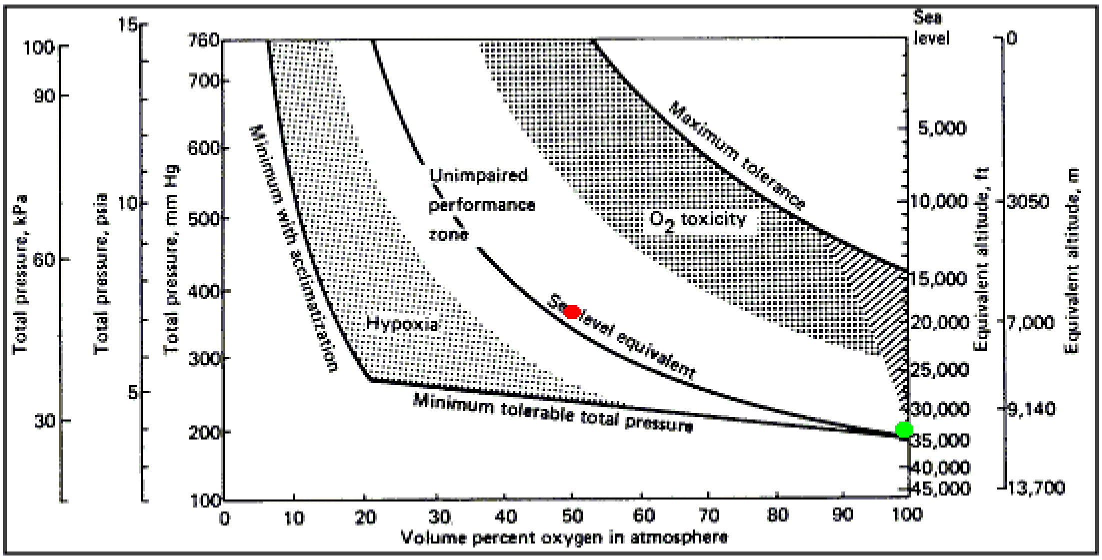


Figure 4-26: Total pressure vs. Oxygen in atmosphere

4.3.3.8.1 Impact of the selected atmosphere composition on safety considerations

To further investigate the favoured atmosphere composition, the impact on the material selection was investigated. The following equation was used to determine if the material selection would be influenced by the atmosphere composition:

$$\% = \frac{23.45}{\sqrt{p(atm)}} = \frac{23.45}{\sqrt{0.5}} = 33.16\%$$

Due to the fact that the atmosphere contains 50% oxygen and the fire-safe limit is calculated as 33%, the material selection will be affected by the atmosphere composition. However, comparing with the Apollo missions, the material selection will be less restricted and a suitable solution could be possible without severely compromising the safety of the crew and equipment.

4.3.3.8.2 Impact on testing

The selected atmosphere would have a significant impact on the testing of the equipment. Tests on the capsule as well as the systems inside it must be done at the considered atmosphere. However, the vehicle is exposed to space vacuum and therefore needs thermal testing in a suitable facility independent of the choice of internal pressure. Furthermore, an extrapolation of the experiences gained with ISS and the shuttle could help to reduce the cost of testing. In both vehicles the cabin pressure is lowered to 70 kPa prior to scheduled EVAs.

4.3.3.8.3 Impact on Martian surface habitat design

As stated earlier, no prebreathing time is necessary and the crew could enter the suits directly from the habitat. However, due to the remoteness a DCS treatment chamber is necessary. The design of a DCS treatment chamber is similar to the design of an airlock and the study suggests that the DCS treatment chamber could serve as airlock as well fulfilling several functions:

- DCS treatment chamber
- Safe area in case of contingency
- Additional level of containment

4.3.3.8.4 Investigation of alternative inert gases

In the course of this study, alternative inert gases were investigated with the objective of minimising DCS risk and prebreathing time without compromising safety. Previous studies with rats demonstrated that the use of a mix of inert gases (nitrogen/argon, argon/helium, nitrogen/helium) does not represent any advantage over the use of one of the most advantageous inert gases such as helium. It was shown that the total pressure of gases except oxygen in the tissue is important and not the partial pressure of the individual gases except oxygen that are dissolved in the tissue. Neon has been suggested as the most advantageous inert gas for space flight applications; however, there have not been any reports of tests with neon as inert gas. Most likely reason for the absence of such data is the price of neon.

4.3.3.8.5 Contingency supply

The use of an open-loop system reduces the need for an emergency supply. However, seven days have been taken in this study as the timeframe expected to prepare for ascent from the Martian surface and journey to the THM.

Currently, the assumption is that the supply would be sufficient until the crew has been able to overcome the contingency situation or to return to the THM using MAV resources.

4.3.3.8.6 In-situ resource utilization (ISRU)

ISRU has not been considered during this phase of the study.

4.3.3.9 Waste management strategy

Waste has to be stabilized and stored on the surface. No recycling or return of the waste is envisaged to minimise down and up mass for the life support.
Currently, the strategy is not compliant with the Planetary Protection rules and it is necessary to further study the waste handling strategy on the Martian surface. The only measure taken into account is to seal the SHM prior to the departure.

4.3.3.10 Baseline design

The life support system is an open-loop life support system with limited regeneration. Water is provided by the fuel cells. Oxygen is stored with the oxygen needed for power generation in the fuel cells. All subsystems of the life support system are designed as open loop with replaceable consumables. The actual technical design of the subsystems has not been yet and needs further investigation.

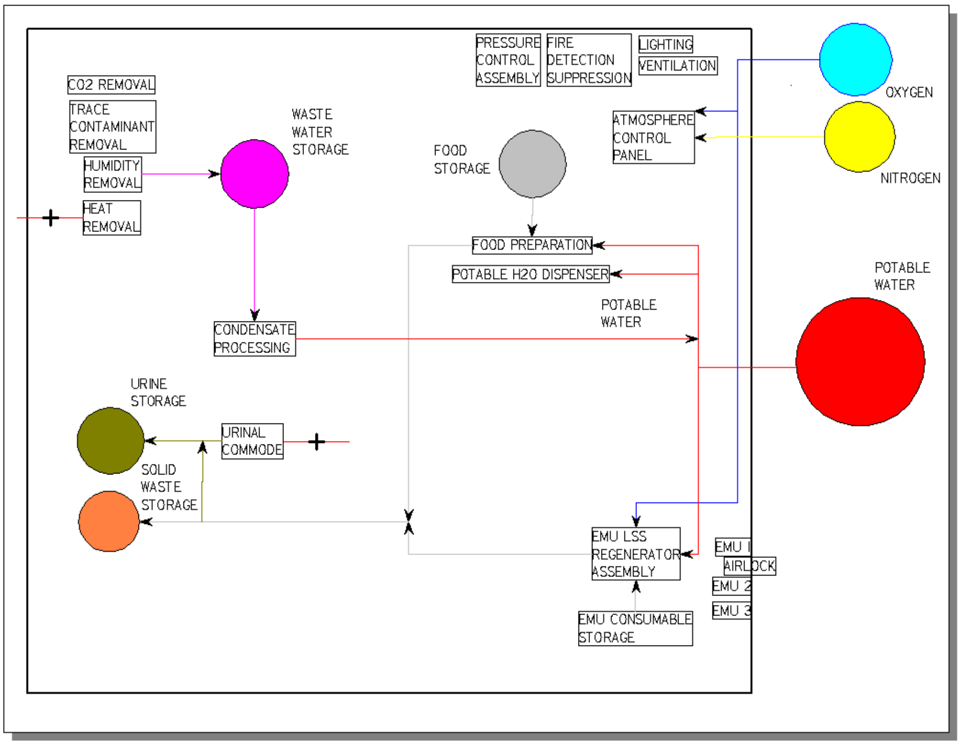


Figure 4-27: Mars SHM LSS design

ECLS system mass:

Consumables to be launched (kg)	
Oxygen	28.0
Nitrogen	91.0
Potable water	323.1
Hygiene water	114.0
Dry food	75.8
Packaging	27.4
Inorganic material excluding packaging	11.1
Total consumables to be launched	670.4
Waste production during mission (kg)	
Waste gases	37.4
Waste water	425.6
Solid organic waste	19.3
Solid inorganic waste excluding packaging	11.1
Packaging	27.4
Total waste produced	**520.9**
Rough estimate ECLSS mass (kg)	
Total ECLSS system mass	**3208.0**

Table 4-7: Mass estimates for a mission using current technology

The life support system has been estimated to have an approximate mass of 3 tonnes. The detailed unit breakdown is shown in Table 4-8:

Equipment	Number of units	Mass per unit (kg)
ALS Airlock Air Save Pump Package	2.00	70.30
ALS CONVENTIONAL OVEN	1.00	50.00
ALS COOKING/EATING SUPPLY	1.00	5.00
ALS METOX CO2 removal (canister)	6.00	14.52
ALS METOX CO2 removal (regenerator)	2.00	47.63
ALS VACUUM	2.00	30.00
GENERIC ACCUMULATOR URINE STORAGE TANK (57L)	1.00	30.84
GENERIC CLOTHING 10kg	3.00	10.00
GENERIC INORGANIC STORAGE(200kg)	1.00	67.00
GENERIC LiOH incl. Activated Carbon (1kg)	30.00	1.00
GENERIC OPERATIONAL SUPPLIES (20kg)	3.00	20.00
GENERIC ORGANIC STORAGE(200kg)	1.00	80.00
GENERIC PERSONAL STOWAGE SPACE (50kg)	1.00	50.00
GENERIC TOOLS EQUIPMENT (20kg)	1.00	20.00
ISS AAA - avionics air assembly	2.00	12.40
ISS ACCUMULATOR POTABLE STORAGE WATER TANK (57L)	15.00	30.84
ISS ACCUMULATOR WASTE WATER TANK (46L)	1.00	67.59
ISS ACCUMULATOR WASTE WATER TANK (46L)	6.00	67.59
ISS CCAA	2.00	112.00
ISS commode/urinal	2.00	50.00
ISS condensate storage	1.00	21.20
ISS EMU (Shuttle)	4.00	135.00
ISS HEPA - BACTERIAL FILTER	3.00	5.20
ISS IMV - intermodule ventilation fan assembly	2.00	4.76
ISS IMV - intermodule ventilation valve	2.00	5.10
ISS NITROGEN STORAGE TANK	1.00	109.00
ISS OXYGEN TANK PRESSURISATION SYSTEM	1.00	102.00
ISS PFE - portable fire extinguisher	2.00	15.10
ISS Sample Delivery System	1.00	2.70
ISS smoke detector	2.00	1.50
ISS TCCS - trace contaminant control system	1.00	78.20
ISS trash compactor	1.00	27.00
Manual Pressure Equalization Valve (MPEV)	1.00	1.20
microwave oven	1.00	70.00
personal hygiene kit	3.00	1.80
Portable Breathing Aparatus	3.00	1.20
Pressure Control Assembly (PCA)	1.00	78.20
RESTRAINTS AND MOBILITY AIDS (100kg)	0.00	100.00
SLEEP PROVISIONS	0.00	9.00
SOYUZ FOOD SUPPLY CONTAINER	1.00	5.00
X-38 HI PRESSURE G02 REGULATOR	1.00	1.40
X-38 HI PRESSURE GN2 REGULATOR	1.00	1.40
X-38 LOW PRESSURE G02 REGULATOR	1.00	1.40
X-38 LOW PRESSURE GN2 REGULATOR	1.00	1.76

Table 4-8 Detailed unit breakdown for the anticipated life support system

The life support system should not be considered exhaustive. It is merely a list of major components, which give an indication of what LSS mass has to be anticipated. Between the power engineering domain and the life support domain it was agreed on using the cryogenic oxygen tanks for oxygen resupply to the crew and to collect all product water of the fuel cells in the storage tanks of the life support system. Note that that the list includes hardware based on life support and a fraction of crew accommodation needs. Additional fractions are found in the Human Factors engineering domain.

4.3.3.11 Budgets

Characteristic	Value
P_{peak} (kW)	10.3
P_{night} (kW)	1.7
P_{day} (kW)	1.6
Volume (m^3)	7.9
Mass (kg)	3208

Table 4-9: Life support power budget

4.3.4 Thermal

The SHM thermal control shall be designed to perform optimally during the stay on Mars. The question whether the same performance is expected during the transfer to Mars remains open. Not necessarily a permanent habitable module (economy of a radiation shield), its functions can be hold in a dormant mode, reactivated when a crew enters the module (storable zone for example). The advantage of this is a higher tolerance on the thermal control and a lower associated budget.

4.3.4.1 Requirements and design drivers

The main requirements are the following

- The external thermal control shall be effective in vacuum and in the Martian pressurised environment.
- The TCS functions are to maintain air temperature and humidity in the HSNM zones within preset limits, and to thermally control the on-board systems. Therefore, TCS shall be designed to maintain:
    - the habitable zones in a certain comfort zone (temperature, humidity) but respecting also safety requirements (touch temperature, condensation avoidance). Standard figures are a medium temperature between 18 and 27C and a relative humidity from 25 to 70%.
    - a uniform environment for a crew up to 3 members.
    - elements and/or dedicated zones within temperature requirements (electronics, propellants, valves, …). To optimise the thermal budget, a certain rationalization of space and grouping of elements shall be carried out. Ideally, all equipments are within a single dedicated enclosure.
    - the interfaces of the others modules (ascent vehicle) within temperature requirements

- The candidate TCS architecture shall be also capable of:
    - performing effectively under Martian gravity
        - guaranteeing adequate flexibility and reliability of the system until the end of the stay on Mars
        - guaranteeing the performance of the system for any spacecraft attitude during transfer and for any orientation after landing, this for all thermal loads derived from the mission requirements
        - optimising the heat management system in term of efficiency versus penalties to the system (mass, energy consumption)
- Safety shall be guaranteed by adequate provision of thermal hardware for the whole mission (necessary autonomy of the crew)
- TCS shall be fully verifiable/testable on ground

4.3.4.2 Assumptions

4.3.4.2.1 Transfer phase thermal environment

The same environment as for the transfer vehicle applies for the Mars Excursion vehicle including the Habitation Module. A conservative approach is to consider envelopes through worst-case scenarios:

	Solar flux [W/m^2]	Planet albedo	Planet IR [W/m^2]
Hot case (Earth LEO, WS, 1 AU)	1423	0.33	241
Hot case (Mars orbit, perihelion, 1.38 AU) [2]	717	0.29 (subsolar)	470 (subsolar) to 30
Cold case (Mars orbit, aphelion, 1.66 AU) [3]	493	0.29 (subsolar)	315 (subsolar) to 30

Table 4-10: Thermal cases definition

Note that with respect to Mars arrival and departure dates, the vehicle passes aphelion and perihelion and that hot and cold cases around Mars depend to a certain extent on the orbit of the spacecraft (and thermal characteristics of the underneath regions). The worst cold case is sought with long eclipse duration: 500 km circular orbit and a coplanar Sun (beta 0) give a 13.6-minute eclipse out of a 40.2-minute orbit.

4.3.4.2.2 Martian thermal environment

Environmental thermal loads depend on the landing site and on the relative position of Mars with regards to the Sun. A mapping of Martian thermal characteristics is shown in Figure 4-28 to show the disparity of the induced environment. A higher albedo drives a lower temperature, while a low thermal inertia accelerates the variation of temperature.

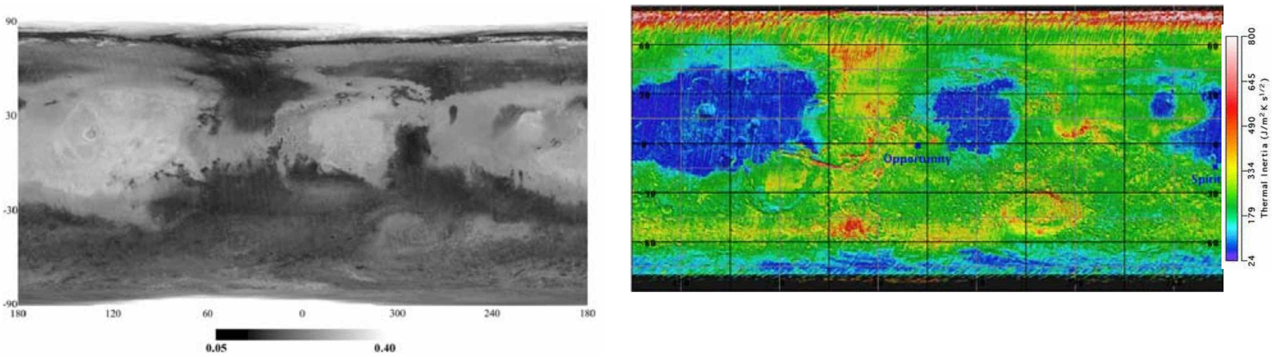


Figure 4-28: Mars albedo (L), ars thermal inertia(R)

As a preliminary approach, no particular constraints are taken regarding the landing site and the landed time, except to avoid the dust-storm period (Ls 200 to 300). A conservative approach is therefore considered based on worst-case scenarios. The following figures indicate the seasonal variation of the thermal environment and the related constraints regarding the thermal design.

Hot cases are conditioned to peak intensity (at noon time) and duration of the day. The first occurs at the perihelion (Ls = 250) and the second at the solstice (Ls = 270 for the southern latitudes, Ls = 90 for the northern latitudes).

In the contrast, the worst cold cases are found either with long duration night at the solstice (Ls = 90 for the southern latitudes, Ls = 270 for the northern latitudes) or when Mars is at its aphelion (Ls = 70).

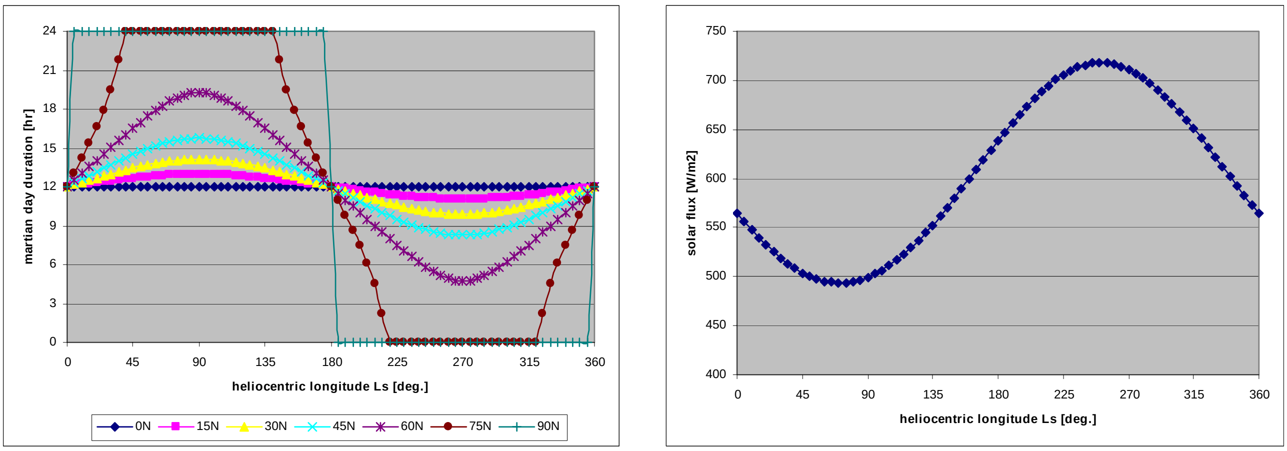


Figure 4-29: Martian day duration versus latitude and Ls (L), Solar flux versus Ls (R)

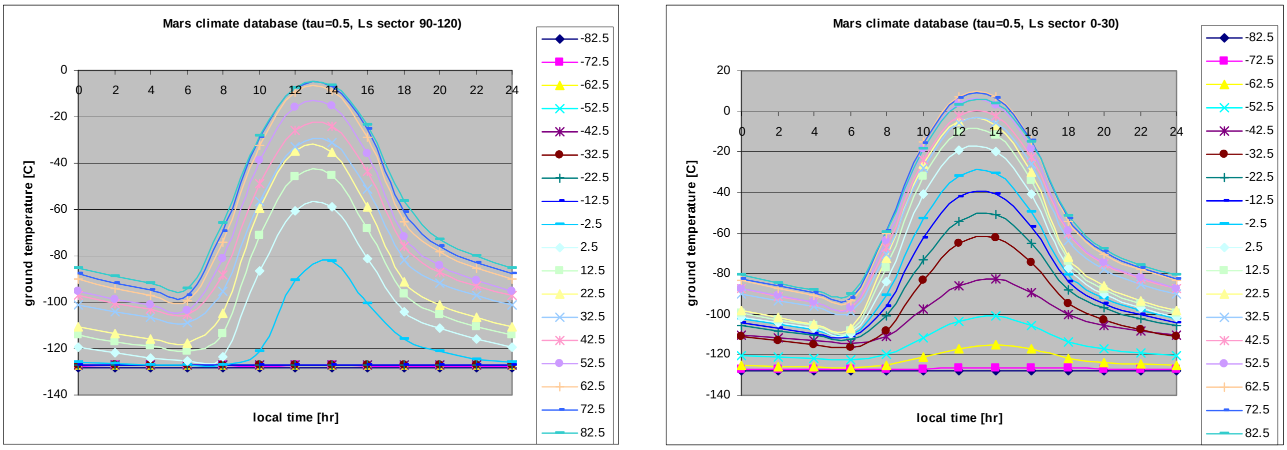


Figure 4-30: Ground temperature versus local time and latitude

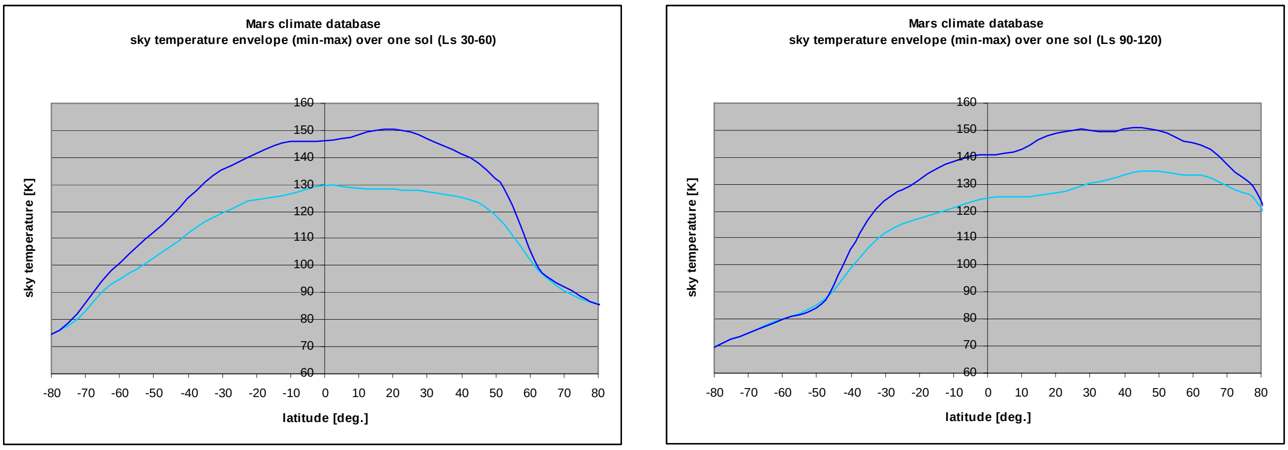


Figure 4-31: Sky temperature versus latitude (L), Sky temperature versus latitude (R)

4.3.4.2.3 Man-induced thermal loads

Thermal design shall manage all internal heat loads resulting from the human activities and various dissipating equipments:

- Total mean heat load of 2.35 kW during day, 2.41 kW during night
- Metabolic dissipation is estimated to 110W (steady activity) per crew (x 3)

4.3.4.3 Baseline thermal design

4.3.4.3.1 Surface habitation thermal control

With no direct expertise in Europe available for such vehicle, the design block proposed is based partly on the exploitation of foreign existing heritage: Apollo LM, LOK (derived Soyuz). Space station fluid-loops technologies are applicable to a certain extent.

The thermal control philosophy adopted for such vehicle is standard and relies on the following approach:

- simplification of the heat transfer with maximal use of thermal decoupling when possible
- use of thermal-regulated bus to recuperate and transfer internal heat (recuperated at the primary loop level) to heat sinks
- use of switch capability to modulate this transfer and balance the heat inputs from the outputs, and thus maintain temperatures within a certain bandwidth

This is implemented using appropriate materials and technologies combining passive or active means.

4.3.4.3.1.1 Thermal bus

- Docked phases

As long as the SHM is docked to the TV, its thermal bus is used (TV secondary loop) providing a cooling capability when necessary (crew in the cabin). Assuming a dormant mode for most of the docked phases when unmanned, a steady low cooling capability is required, mainly for thermostatic control.
Because of the low level of these loads, a direct connection with the Ascent Vehicle primary loop is foreseen, and the SHM loop capability is used to transfer its heat loads to the TV secondary loop.

- Descent phases

During de-orbit (from 500 km) and reentry phases (120 km), no external sink is possible or available because of the aerothermal loads, and a substitute has to be foreseen. The duration of these two phases being less than 2 hours (no abort possible), no specific system is required but a pre-conditioning of the vehicle before separation. Relying on the high thermal inertia of the modules, a natural cooling capability is stored in the structural mass by lowering the temperatures to a minimum setting (15C).

- Landed phase

A residual heat from the aerothermal phase will penetrate the thermal system and steadily raise the overall temperatures thanks to the thermal inertia. Until activation of the nominal thermal management mode of the landed phase (which can possibly take several hours), a system shall counter balance this temperature rise and provide a cooling capability. Sublimation of water is retained as the most efficient system for this period.

For nominal and contingency modes, an external bus is designed to provide the required sink. A biphasic system is retained for its performance with ammonia as the working fluid.

4.3.4.3.1.2 Radiator orientation and location

The choice of the radiator orientation and location is a trade-off between the available environmental sinks and the configuration constraints.

	Advantage	Disadvantage
Body-mounted radiator facing sky	Low sink temperatures (max is -120C) and good stability (20K variation max over one sol) => high performance radiator when not illuminated Possible combination with the ascent vehicle radiator necessary during RdV	Only location is on top of the ascent vehicle already occupied by the solar cells necessary during rendezvous. Radiator Sun illumination is maximal over one day Possible dust deposits
Lateral body mounted radiator	Partial viewing to the sky increases the radiator performance when not illuminated Adequate to a close location with the confined compartment (secondary loop system) if chosen on the lower part of the SHM	Sink temperature depends on the viewing with sky and ground => instability of the sink (large amplitude of the ground sink) Possible illumination during a few hours Viewing of the sky degrades when too close to the ground
Bottom body mounted radiator	Permanently shadowed by the SHM body => the amplitude of the sink remains low and the sink close to the night temperatures (high sensitivity to sun illumination) Adequate to a close location with the confined compartment (secondary loop system) The radiator view factor is closed when docked to the transfer vehicle (low activity of the SHM during transfer, the heating budget is therefore minimised)	The dust density is higher close to the ground (possible electrostatic adhesion) Conflicts with the heating from braking thrusters => requires specific protection
Deployable radiator	Possibility to implement a tracking of the Sun to minimise illumination of the radiator	Gravity and wind pressure penalize mass

Table 4-11: Radiator options

4.3.4.3.1.3 Radiator sizing

Figures here below indicate the relative penalty of the radiative surface to the daily variation of the environment, this if the cooling capability is to be maintained.
In the case of a *bottom* radiator, how much the ground can heat up with convection and the SHM rejection will depend on the nature of the ground. The rather low thermal inertia in general does not make this option attractive, in particular when considering that the ground has little means to cool down itself if the two surfaces are parallel (so the ground will tend to the mean daily radiator temperature).

In the case of a *lateral cylindrical* radiator, assuming a view factor of 0.5 to sky and ground, equinox, latitude 0, at noon the penalty is a ratio of 2.4 against the same heat rejection at –120C (end of night). The negative impact of the Sun remains acceptable when considering its angle of incidence (left figure). Increasing the view factor to the sky increases the rejection but also increases the Sun illumination (intensity and duration), however at a much lower rate.

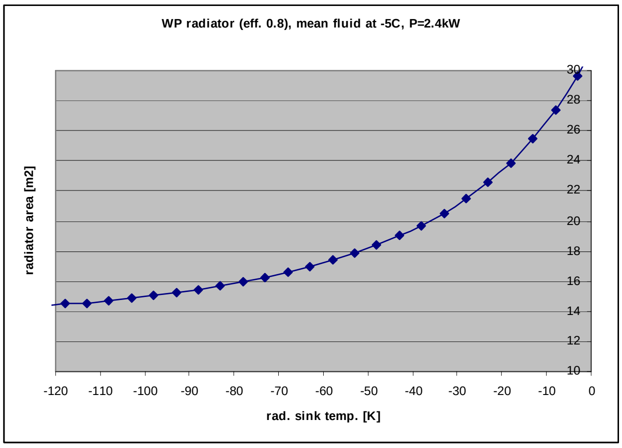


Figure 4-32: Bottom radiator facing ground sink

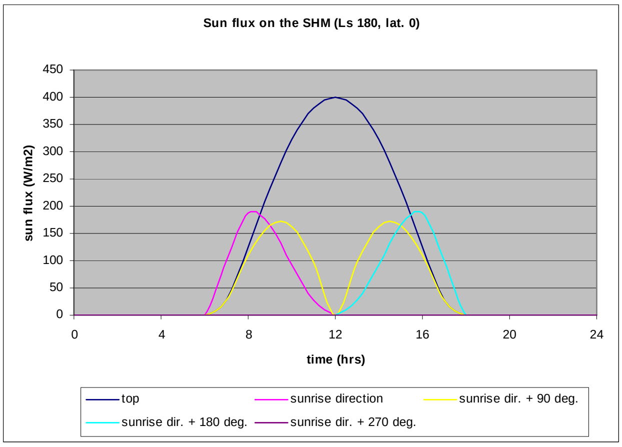


Figure 4-33: Local sun flux on SHM (equinox, lat. 0)

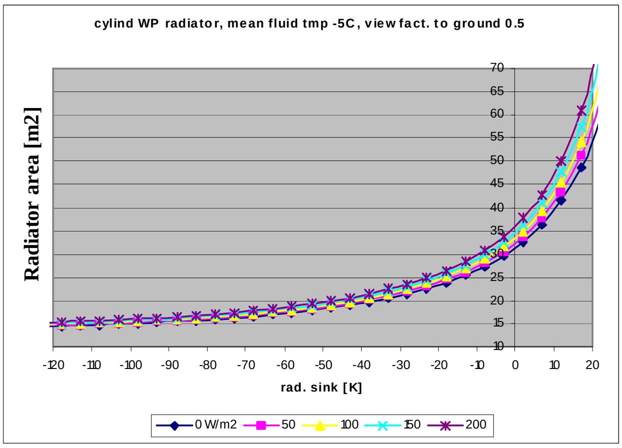


Figure 4-34: Sensitivity rad. versus sink and sun flux (equinox, lat. 0)

The proposed configuration is therefore a conical compartment upon which a body-mounted radiator is designed. Impingement from braking plumes is minimised with proper orientation and location of the thrusters. The duration of thrust is assumed short so cumulated heat and contamination of the radiator should be low. However if not sufficient, protective foil (titanium) can be installed locally. The total radiative surface allowed with this configuration (sized for a max sink of –10C) is 17.7 m^2 (total surface of the compartment zone) + 4.5 m^2 (upper cylinder 3.6 x 0.39 m length).

Flexibility exists in the fluid loop system and shall be included in the heat management to optimise the thermal design and its related budget. For example, heat rejection around noon can be delayed a few hours (permitting a certain temperature increase of the working fluid) until the sink reaches an admissible level, this to avoid an excessive radiator sizing.

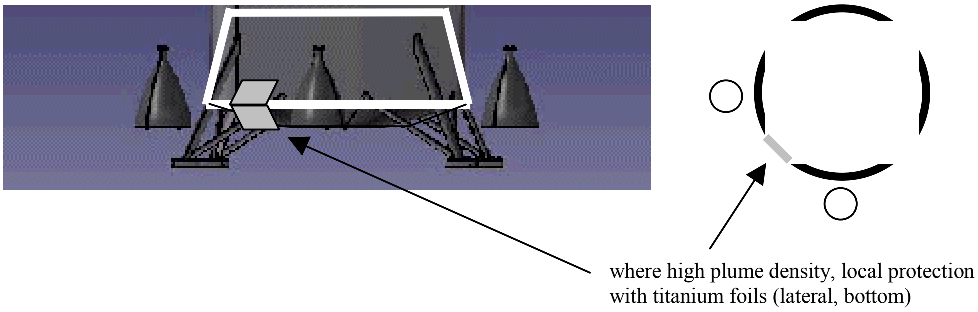


Figure 4-35: SHM radiator and location

4.3.4.3.2 Primary loop

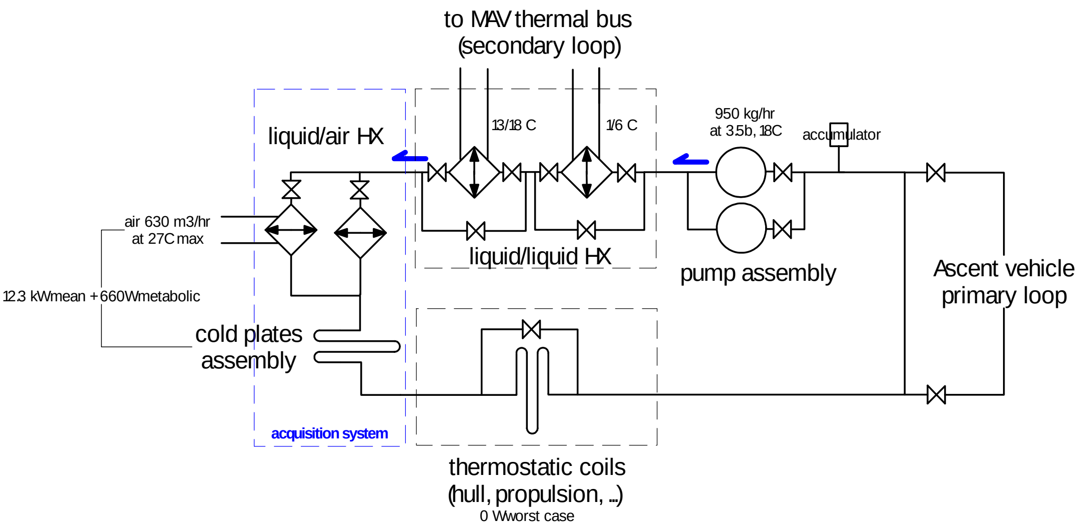


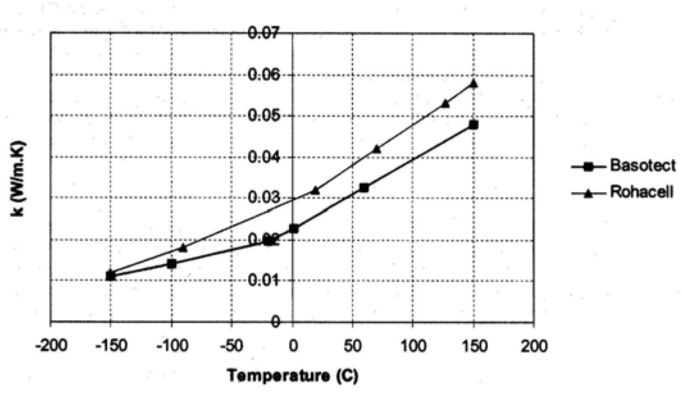
Figure 4-36: Surface Habitation Module / primary loop principles

4.3.4.3.3 The insulating system and thermal protection

The thermal design for landed vehicles differs considerably from other space applications in that Mars has an atmosphere (7 mb), which plays an important role in the thermal insulation.

The choice of insulation and structures must be traded off against each other. The vacuum compatible foams such as Basotect and Rohacell not only have different thermal properties but also are structurally different. Rohacell (a closed-cell rigid foam plastic) is stiff, impact resistant and self-supporting, whereas Basotect (an open-cell foam), a better insulator, is fibrous and lacks any structural integrity.

Used on the Pathfinder and MER rovers, Aerogel (Silica gel with carbon black) is an excellent thermal insulator, but has no structural integrity. The different thermal conductivities are shown in Figure 4-37.



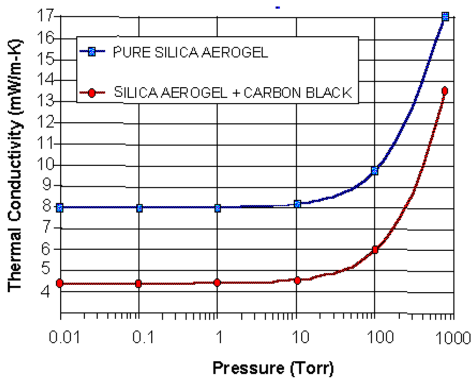


Figure 4-37: CLRC Beagle2 study (L), Aerogel thermal conductivity versus pressure (R)

The low density of Aerogel combined with its good thermal insulation makes it an ideal candidate but would need an encapsulation (honeycomb cells for example). If a monolithic structure (aluminium) is retained, insulation materials can be added in a multi-layer design combining radiative (goldenized layer) to conductive insulation (foams). The double requirement to perform in vacuum and in pressurised environment can be answered by installing different type of foams (closed cell inside, open cell outside).

Figure 4-38: Insulation layout

The choice of external layer results as a compromise between the different constraints brought by the Martian and vacuum environment. The principle of a cold radiative skin completed by heat input when necessary is preferred for its simplicity, providing the existing resources of energy (released heat from units and metabolic). Betacloth is retained as the external layer of the SHM to avoid undesirable heating from the Sun during the transfer phase. Its high emittance (high energy exchanged during nights) is somewhat counterbalanced by the high thermal inertia of the vehicle. Its strength is also seen as an advantage.

4.3.4.3.4 The thermostatic system

Certain surfaces that cannot be protected by insulating means (interface between MAV and ascent vehicle) are treated (oxidation anodic, alodine) to minimise heat transfer. On the internal face, coils (circulating fluid from primary loop) thermostatically control the temperature (condensation avoidance) and the heat exchanges (control of the heat losses). An adequate redistribution of the rejected heat (thermostatic coils) therefore reduces the use of heater power to the minimum.

When not directly accessible to fluid lines, externally mounted elements will require the use of strip heaters combined to an adequate insulation.

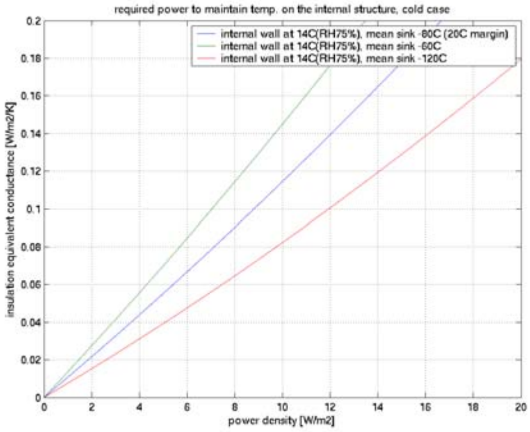


Figure 4-39: Power required to maintain temperature

The insulating system is assumed to provide an equivalent thermal conductivity of 0.13W/m^2/K (15 cm of foam). Therefore, maintaining over one sol an internal wall above dew point (14C for 75% humidity, mean sink –60C + 20C margin accounting to heat losses through inertia) would require a power density of 11.3W/m^2.

Two systems are proposed:

- a network of heaters homogeneously distributed on the internal shell corresponding to a installed power of 728W (assuming the SHM as a cylinder 3.6 x 5.7 m length). Two equivalent circuits (main and redundant) are foreseen, each piloted by a control unit. For safety, each circuit will be equipped with over temperature thermostats to protect against a failed-on heater switch.
- a network of coils / heat pipes mounted on the internal shell to transfer / homogenize the rejected heat from main loop (about 2.7 kW)

A strategy for heater power saving can be implemented with a pre-heating before the night (using the higher activity dissipation during the day through the fluid heat storage capability). Local PCM (where worst heat leaks are located) can also efficiently complete the system. An optimised heat management could request little electrical power to maintain the requirements. For safety reasons, however, a certain provision of installed power shall be designed (about 728W). No particular trade-off has been done on the landing location assuming no specific landing (polar site or winter period with high latitude).

4.3.4.3.5 Overview

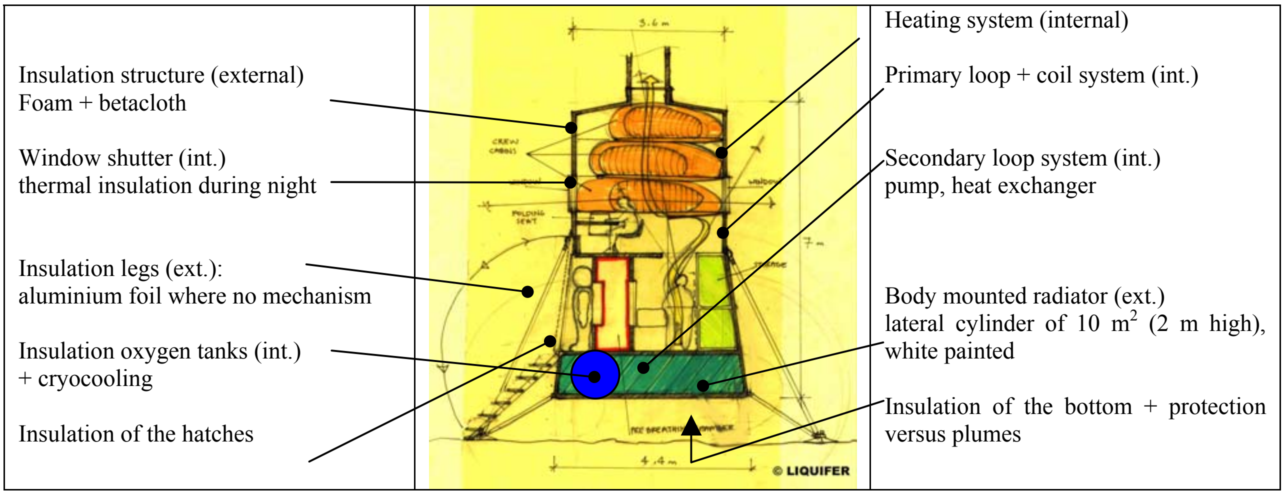


Figure 4-40: Thermal system configuration

4.3.4.3.6 Fuel cells

Because of its high energy density, hydrogen is normally retained as the fuel and oxygen as its reactant. Related technologies regarding the electrolytes have been addressed in the power section. Whichever choice, thermal management of the fuel cells is essential:

- to guarantee the operating temperature of the chemical reaction
- to preheat the cryogenics reactants before entering the stacks
- to cool down each stack, because of the highly exothermic chemical reaction (141.9 MJ/kg)

The cooling capability of the stacks is provided by a coil system connected to the coldest sink through a dedicated secondary loop heat exchanger. The fluid used for this loop is a fluorinated hydrocarbon coolant. A temperature actuated flow control valve maintains and regulates the coolant exit temperature.

As preliminary inputs for heating power, the PEM cells used in the STS (7 kW per unit) are assumed with 2.4 kW for start up and 1.1 kW in nominal mode.

Note that, the water by-product used for life support can be used also as a coolant in the thermal management system. An integrated system (thermal / power / life support) possibly based on regenerative cells seems promising for mass saving (transfer vehicle for instance).

4.3.4.3.7 Cryogenic storage for fuel cells tanks

Fuel cells are used for the MEV require the storage of liquid hydrogen and oxygen, and an appropriate thermal design to maintain the related boil-off (BO) to an acceptable level. The objective is to have after 24 months, 119 kg of hydrogen and 955 kg of oxygen (input from power subsystem).

The tanks are identical in shape and their geometry is a sphere. They are located in an unpressurised section of the MEV to minimise parasitic heat transfer from surrounding elements (gas conduction). Passive insulation is used to isolate the tanks from their radiative environment. The MLI is a stack of n layers of Double Aluminized Mylar (DAM) with an external *goldenized* layer.

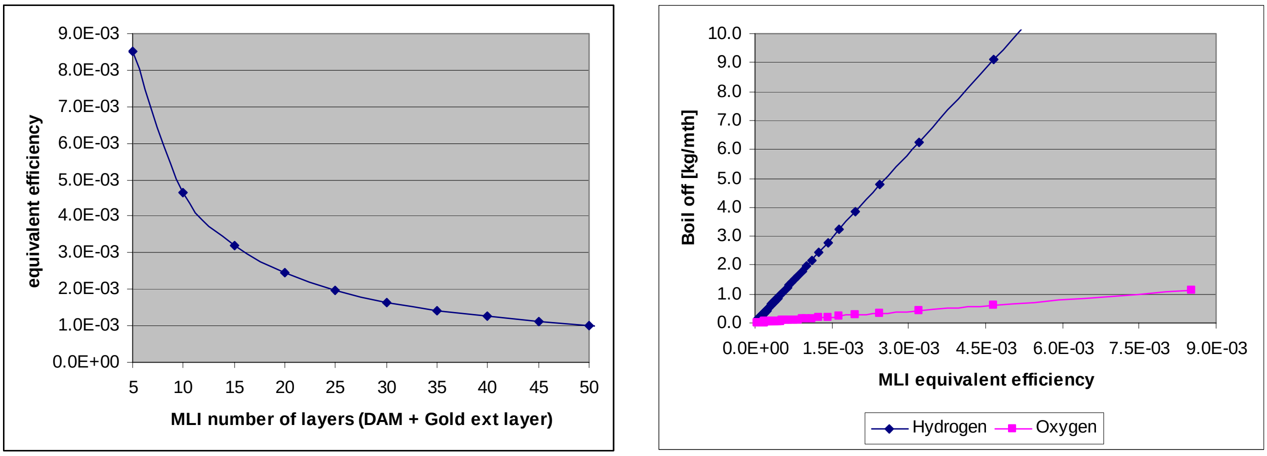


Figure 4-41: Equivalent efficiency and boil-off

The number of tanks and their diameter shall be traded off so that despite possible boil off (BO dependent of the diameter), the required capacity is reached at the end of the 24 months. Within that hypothesis, the following figures indicate the relationship between tank diameter, number of tanks and BO when no cooling capability is provided. Although acceptable for oxygen, it is not manageable for hydrogen (diameter and BO too high) and will require a cooling capability.

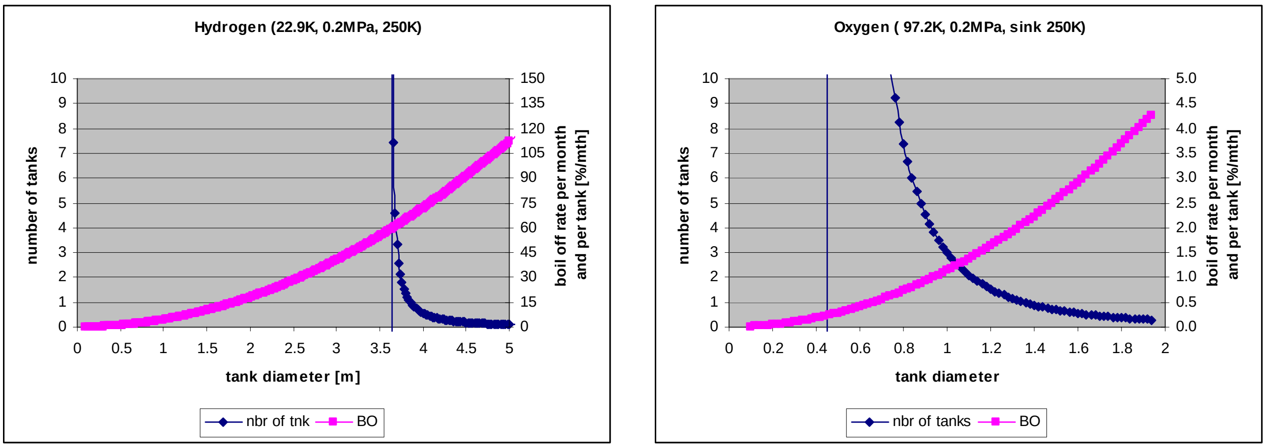


Figure 4-42: Number of tanks and boil-off as function of tank diameter for LOx and H_2

The thermal design for hydrogen tanks is a hybrid solution combining cryocooling technology and passive insulation. For the hydrogen tanks, the required heat lift from the cryocooler is indicated in Figure 4-43. The dotted line indicates an ideal solution (heat lift exactly compensates heat losses through MLI) and the others lines where heat losses exceed heat lift, solution possible with a tolerance on the BO.

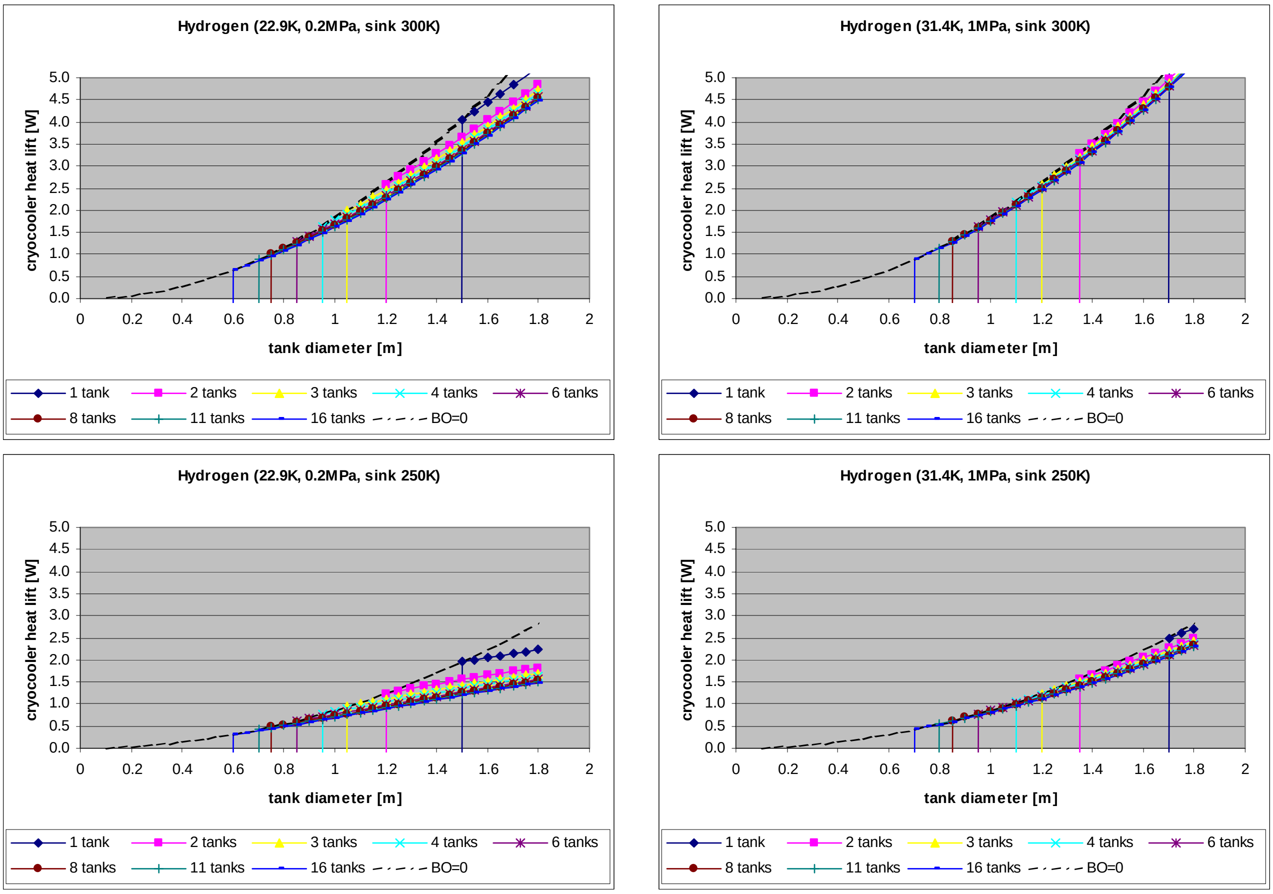


Figure 4-43: Cryocooler heat lift for the hydrogen storage

In the figure top right and bottom right, the pressure is increased to 1 MPa to reach a boiling temperature of 31.4K, while in figure left and right bottom the sink temperature is lowered to 250K. Since the density of the fluid decreases with an increase of temperature, more tanks (and more mass) are required to accommodate the same volume of liquid. Significant advantage is found however on the efficiency of the cooler that doubles (less input power) and on the BO rate (lower heat of evaporation). The number of MLI layers is traded off in the left figure in Figure 4-44, and shows that doubling this number decrease the required heat lift of a factor 2. In the same way, in the right figure in Figure 4-44, is shown that decreasing the sink temperature from 300K to 250K (still in the operative range for the electronics) allows the heat lift to drop by a factor of 2.

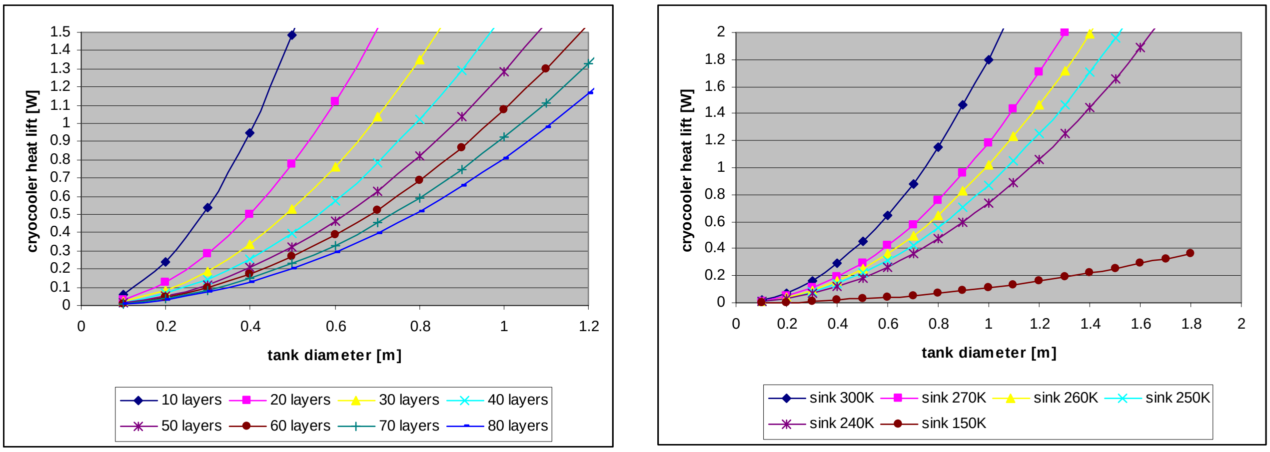


Figure 4-44: Cryocooler heat lift as function of the number of layers and tank diameter

4.3.4.3.8 Synthesis

As seen here above, optimum can be reached with:

- a performant insulation that depends on the number of layers of the MLI. This latest is set to 40 layers (DAM with goldenized external layers)
- a low heat sink that depends on the location and accommodation of the tanks. A dedicated tank compartment (not pressurised) at 250K seems a reasonable compromise with constraints from the MAV vehicle (heat soaking at the interfaces) and from the hardware located in this compartment (coolers)
- an appropriate medium for the storage: spherical tank with appropriate diameter, this is a compromise between cooling capability (that constrains the diameter downward) and the number of tanks (accommodation that constrains the number to match the allocated volume)
- a heat lift provided by a mechanical cooler

Presently available in Europe is the Astrium 20-50K two-stage Stirling cooler with a performance of 120 mW at 20K and about 300 mW at 30K, not suitable to this study (would drive a high number of tanks, more than 27 at both temperatures). The pre-cooler of this system however could be used with a performance about 800 mW at 30K.

Hydrogen		Per tank												Budget total		
case	sink [K]	nbr of tank	diameter [m]	MLI layers	liquid temp. [K]	pressure (MPa)	nbre of cryo units	heat lift [W]	mass liquid [kg]	thickness tank [mm]	structural mass [kg]	thermal mass [kg]	input power [W]	thermal mass [kg]	total dry mass [kg]	input power [W]
1	300	40	0.44	40	22.9	0.2	1	349	3.0	0.4	0.7	0.5	28.0	351.9	378.7	1120.0
2	300	15	0.62	40	22.9	0.2	2	690	8.4	0.6	1.9	1.0	56.0	263.8	291.9	840.0
3	300	8	0.76	40	22.9	0.2	3	1050	15.5	0.7	3.4	1.5	84.0	211.1	238.7	672.0
4	300	17	0.66	40	31.4	1	1	784	7.3	3.1	11.3	1.1	46.0	160.2	352.4	782.0
5	300	6	0.94	40	31.4	1	2	1588	21.1	4.4	32.7	2.3	92.0	113.3	309.2	552.0
6	300	4	1.06	40	31.4	1	3	2023	30.2	4.9	46.8	2.9	138.0	111.2	298.5	552.0
7	250	14	0.64	40	22.9	0.2	1	331	9.2	0.6	2.1	1.1	23.0	131.0	159.8	322.0
8	250	5	0.88	40	22.9	0.2	2	672	24.0	0.8	5.4	2.0	46.0	93.0	119.7	230.0
9	250	3	1.04	40	22.9	0.2	3	940	39.7	1.0	8.8	2.8	69.0	83.1	109.6	207.0
10	250	6	0.94	40	31.4	1	1	763	21.1	4.4	32.7	2.3	37.0	63.5	259.4	222.0
11	250	2	1.34	40	31.4	1	2	1555	61.1	6.2	94.6	4.6	74.0	42.4	231.7	148.0

Table 4-12: Solutions

Cases 9-11 appear interesting in terms of budgets and 11 is retained: two hydrogen tanks - diameter 1.34 m - with two single-stage Stirling coolers mounted on each pole of each tank. The

thickness (6.2 mm aluminium shell to match a internal pressure of 1 MPa) allows a good spreading of energy from the poles to the equatorial belt. Optimised mechanical support systems for the cryogen tanks should also be considered (PODS for example). The tanks are in an unpressurised section, so that –23C can be reached as a radiative environment.
In these conditions, the moderate heat lift required (0.5 to 1W between 20 to 30K) to counter BO does not require significant development but modifications of existing hardware (Stirling coolers). If the environment has to be modified (external tanks submitted to environmental loads), or because of a more integrated system (with ECLS) resulting in larger tanks, the use of higher heat lift capability may become necessary and the choice of the cooling system oriented to recuperative systems (reverse Brayton, Joule Thomson cycles).
As previously seen and within the study's hypothesis, the oxygen tanks do not necessarily require a cooling capability. A tolerance to boil-off is accepted per design with an increased initial mass of oxygen liquid.

Oxygen		Per tank												Budget total		
case	sink [K]	nbr of tank	diameter [m]	MLI layers	liquid temp. [K]	pressure (MPa)	nbre of cryo units	heat lift [W]	mass liquid [kg]	thickness tank [mm]	structural mass [kg]	thermal mass [kg]	input power [W]	thermal mass [kg]	input power [W]	total mass liquid [kg]
1	250	1	1.36	40	97.2	0.2	0	0	1454.6	1.3	19.8	4.8	0.0	4.8	0.0	1454.6
2	250	2	1.12	40	97.2	0.2	0	0	812.4	1.0	11.0	3.2	0.0	6.5	0.0	1624.9
3	250	3	1	40	97.2	0.2	0	0	578.3	0.9	7.9	2.6	0.0	7.7	0.0	1734.9
4	250	4	0.92	40	97.2	0.2	0	0	450.3	0.9	6.1	2.2	0.0	8.7	0.0	1801.2
5	250	1	1.46	40	119.6	1	0	0	1589.6	6.8	122.4	5.5	0.0	5.5	0.0	1589.6
6	250	2	1.22	40	119.6	1	0	0	927.5	5.7	71.4	3.8	0.0	7.7	0.0	1855.0

Table 4-13: Options

Case 1, a tank of a diameter 1.36 m appears as the best solution of a no coolers trade-off and could be integrated with the 2 hydrogen tanks. However, an appreciable mass saving could be obtained with a cooling capability and is the option to prefer for an optimised system.

4.3.4.4 Budget

4.3.4.4.1 Synthesis per subsystem (main features)

Fluid loops	
Primary loop	Pump assembly: 67 kg, 463W nominal (950kg/hr) (x 2) Condenser heat exchangers: 20.6 kg (x 2),cold plates: 3.4 kg (x 10), valves (on/off, manual): 4 kg (x 20) 120 kg of tubing (dry including insulation, brackets) + 20 kg of water
Secondary loop	Pump assembly: 56.7 kg, 311W nominal (x 2) Heat exchangers: 15.9 kg (x 4), cold plates: 3.4kg (x 5), valves (on/off, manual): 4 kg (x 5) 36 kg of tubing (dry including insulation, brackets) + 31 kg of ammonia
Passive thermal control	
Body-mounted radiator	Cylindrical/conical radiator of 22 m^2, 0 kg (transferred to structure budget)
Insulation	193 kg for the main body of the transfer vehicle (to check the status of PMOD). 50 kg are provisioned for specific external and internal elements insulation.
Heating system	728W installed power Two control units (1 on), each 6 kg, 29W when shell heaters are 100% duty cycle

Cryo systems	
Oxygen tank (fuel cells): (x 2)	per tank, two coolers, each 8.3 kg (7 kg compressor, 1.3 kg displacer), consumption 37W each MLI: 4.6 kg for each tank

Table 4-14: Synthesis per subsystem

4.3.4.4.2 Overall budget (as introduced to the system)

lement 2: Surface Habitation Modul			MASS [kg]				
Unit	Element 2 Unit Name Click on button below to insert new unit	Quantity	Mass per quantity excl. margin	Maturity Level	Margin	Total Mass incl. margin	Ppeak
1	SHM / radiator (body mounted)	1	0.1	Fully developed	5	0.1	0.0
2	SHM / ext. insulation	1	193.3	To be developed	20	232.0	0.0
3	SHM / coil assembly	4	17.7	Fully developed	5	74.3	0.0
4	SHM / insulation	1	50.0	Fully developed	5	52.5	0.0
5	SHM / HCU	2	6.0	To be modified	10	13.2	29.0
6	SHM / Heaters, thermostat, lines	2	13.0	Fully developed	5	27.3	728.0
7	int. loop / liquid	1	104.0	Fully developed	5	109.2	0.0
8	int. loop / dry tubing + insulation	2	120.0	Fully developed	5	252.0	0.0
9	int. loop / pump assembly	2	67.0	To be modified	10	147.4	655.0
10	int. loop / compensator	2	3.0	Fully developed	5	6.3	0.0
11	int. loop / cooler-dryer assembly	2	20.6	To be modified	10	45.3	0.0
12	int. loop / cold plates	10	3.4	Fully developed	5	35.7	0.0
13	int. loop / hydraulic connector	20	0.3	Fully developed	5	6.3	0.0
14	int. loop / choke washer	20	0.5	Fully developed	5	10.5	0.0
15	int. loop / on-off valve	20	4.0	Fully developed	5	84.0	60.0
16	int. loop / by-pass valve	2	20.3	To be modified	10	44.7	0.0
17	int. loop / manual valve	20	4.0	Fully developed	5	84.0	0.0
18	ext. loop / liquid	2	36.0	Fully developed	5	75.6	0.0
19	ext. loop / dry tubing + insulatio	2	20.1	Fully developed	5	42.2	0.0
20	ext. loop / pump assembly	2	56.7	To be modified	10	124.7	311.0
21	ext. loop / cold plates	5	3.4	Fully developed	5	17.9	0.0
22	ext. loop / compensator	2	3.0	Fully developed	5	6.3	0.0
23	ext. loop / heat exchangers	8	15.9	To be modified	10	139.9	0.0
24	ext. loop / flow control valve	2	20.3	Fully developed	5	42.6	22.0
25	ext. loop / on-off valve	10	4.0	Fully developed	5	42.0	30.0
26	fuel cell / hydrogen tank system	2	21.2	To be developed	20	50.9	148.0
27	fuel cell / oxygen tank system	1	4.8	To be developed	20	5.8	0.0
28	fuel cell / stacks thermal	1	5.0	To be developed	20	6.0	2400.0
-	Click on button below to insert new unit			To be developed	20	0.0	
LEMENT 2 SUBSYSTEM TOTA		28	1636.6		8.7	1778.7	4383.0

PPEAK AND POWER SPECIFICATION PER MODE																					
Unit	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-
	Pon	Pstby	Dc	Pon	Pstby	Dc	Pon	Pstby	Dc	Pon	Pstby	Dc	Pon	Pstby	Dc	Pon	Pstby	Dc	Pon	Pstby	Dc
1	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
2	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
3	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
4	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
5	29.0	13.7	0.0	29.0	13.7	10.0	29.0	13.7	0.0	29.0	13.7	0.0	29.0	13.7	0.0	29.0	13.7	0.0	29.0	13.7	100.0
6	728.0	0.0	0.0	728.0	0.0	10.0	728.0	0.0	0.0	728.0	0.0	0.0	728.0	0.0	0.0	728.0	0.0	0.0	728.0	0.0	100.0
7	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
8	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
9	655.0	35.0	10.0	655.0	35.0	69.0	655.0	35.0	69.0	655.0	35.0	69.0	655.0	35.0	69.0	655.0	35.0	69.0	655.0	35.0	69.0
10	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
11	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
12	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
13	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
14	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
15	60.0	0.0	0.0	60.0	0.0	0.0	60.0	0.0	0.0	60.0	0.0	0.0	60.0	0.0	0.0	60.0	0.0	0.0	60.0	0.0	0.0
16	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
17	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
18	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
19	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
20	311.0	40.0	100.0	311.0	40.0	100.0	311.0	40.0	100.0	311.0	40.0	100.0	311.0	40.0	100.0	311.0	40.0	100.0	311.0	40.0	100.0
21	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
22	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
23	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
24	22.0	2.9	5.0	22.0	2.9	5.0	22.0	2.9	5.0	22.0	2.9	5.0	22.0	2.9	5.0	22.0	2.9	5.0	22.0	2.9	5.0
25	30.0	0.0	0.0	30.0	0.0	0.0	30.0	0.0	0.0	30.0	0.0	0.0	30.0	0.0	0.0	30.0	0.0	0.0	30.0	0.0	0.0
26	148.0	0.0	100.0	148.0	0.0	100.0	148.0	0.0	100.0	148.0	0.0	0.0	148.0	0.0	0.0	148.0	0.0	0.0	148.0	0.0	100.0
27	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
28	2400.0	0.0	0.0	2400.0	0.0	46.0	2400.0	0.0	46.0	2400.0	0.0	0.0	2400.0	0.0	0.0	2400.0	0.0	0.0	2400.0	0.0	46.0
-															-						
LEMENT 2 SUBSYSTEM TOTA	4383.0	91.6		4383.0	91.6		4383.0	91.6		4383.0	91.6		4383.0	91.6		4383.0	91.6		4383.0	91.6	

Table 4-15: Overall budget

4.3.5 Power

4.3.5.1 Design drivers

The environment encountered by the MEV and the specific requirements are completely new for a space power design. Therefore, the completion of this power subsystem is one of the most challenging power designs of the coming 20 years. Until now, except nuclear technologies, power storage and generation were only performed on space applications by: solar cells, batteries (primary or secondary) or fuel cells. More extended technologies for power storage and generation have to be considered.

The Martian environment is hostile for power system for several reasons: dust deposit, daily temperature cycling, diffusion and scattering of the sunlight, high-speed winds, presence of oxidizing soil characteristics, dust storms, roughness and relief of the landing spot, low solar irradiance and long eclipse durations.

Several studies dealing with a large range of possible technologies candidates have already been performed (See [RD4],[RD5],[RD60],[RD62],[RD64],[RD65]…). The conclusions vary from stray to another.

An important rationale is the fact that the technologies shall be available and qualified only for 2015. Hence, designs that are nowadays still at experimental or even conceptual level are also considered. Moreover, the expected efficiency increase of the qualified technologies during the following 10 years can also differ.

This study therefore focuses on:

- the qualified/existing technologies: on the ESA development programmes plan
- other technologies at the current level of confidence

Consequently, although the final design presented in the report will maybe not be the one with the best performances in 2015, but is one of the most reliable with the current state of art without having a too conservative approach.

A first trade-off between the main promising candidates was performed before starting the design itself of the subsystem.

Nuclear energy has been excluded from this study.

4.3.5.2 Requirements

4.3.5.2.1 Mission requirements

During the cruise from Earth to Mars, the power required for the MEV (thermal regulation, check-ups…) is supplied by the power system of the TV.
The MEV needs to have an autonomous power system from the separation from the TV, during the descent phase, during the surface operations, and during the launch from Mars until the rendezvous with the TV.

The mission has therefore been divided into the following modes:

- Descent phase (duration estimated: 30 minutes)
- Surface operations (distinction between night and day power consumptions)
- Ascent phase (max 90 minutes)
- Parking orbit (orbit duration: 118 minutes) for several days
- Rendezvous and docking (maximum 30 minutes)
- MAV Orbital Safe Mode

The surface operations duration is 37 days long in the contingency case. The possible landing sites to take into account are in the latitude range [20ºN, 20ºS].

During all the phases, power has to be supplied to the different subsystems.
The Figure 4-45 shows the different modes of the mission. The time durations correspond to the reference time considered for the power design.

Number	Mode Name	Definition	Acronym	Ref. Duration (mins)
1	Descent	3 Crew onboard the THM and 3 crew onboard the MAV DM+MAV powered by MAV Sub systems on: Lifesupport, comms, DHS, GNC, Power, Thermal	DESM	30.00
2	Surface (day)	3 Crew onboard the Surface Habitation Module Power from surface habitation module Sub systems on: Lifesupport, comms, DHS, Power, Thermal	SDAYM	600
3	Surface (night)	3 Crew onboard the Surface Habitation Module Power from the MAV Sub systems on: Lifesupport, comms, DHS, Power, Thermal	SNGM	870.00
4	Ascent	3 Crew onboard the MAV Power from the MAV Sub systems on: Lifesupport, comms, GND, DHS, Power, Thermal	MOAM	90.00
5	Parking Orbit around Mars	3 Crew onboard the MAV Power from MAV's second stage Solar arrays deployed? Sub systems on: Lifesupport, comms, DHS, GNC, Power, Thermal	POM	118.00
6	Rendezvous and Docking Mode	3 Crew onboard THM and 3 crew onboard the MAV Batteries from MAV's second stage Sub systems on: Lifesupport, comms, DHS, GNC, Power, Thermal	RVDM	30.00
7	MAV Orbital Safe Mode	3 Crew onboard THM and 3 crew onboard the MAV Batteries from MAV's second stage Sub systems on: Lifesupport, comms, DHS, GNC, Power, Thermal	MOSM	-

Figure 4-45: MEV Modes

The MEV power subsystem has to cope with three different types of mission requirements:

- during the descent: power needs to be supplied in a short time with some peak power constraint (pyros…)
- during the surface operations: important level of power needs to be supplied for a long duration with possibility to generate power
- during the launch, the parking orbits and the rendezvous: power needs to be supplied for a few days with a possibility to store energy during the sunlight, to have enough energy available for the eclipses and non-Sun pointed phases.

Since the power system designed for the surface operations is able to cope with high power level during periods of 14 hours, it can also cover the energy requirements for the descent phase.
Therefore, it has been chosen in this study not to develop a specific power subsystem for the descent phase. Nevertheless, a dedicated DM power subsystem is not excluded given that its mass would be negligible compared to the total MEV mass.
On the MAV's side, the mass is an important mission driver. Consequently, the implementation of the huge power subsystem required for the surface operation inside the MAV has been rejected. However, such a design would be possible in some particular cases for example when using fuel cells by keeping the useless and empty reactant tanks outside of the MAV.

The MEV power system is composed of:

- one power system inside the SHM which is also used by the DM
- one power system located in the MAV for supplying the power after the launch from Mars.

These two subsystems are completely independent, except that the MAV power subsystem can be charged by the SHM power system to the launch.

4.3.5.2.2 Power requirements

In this study, the power requirements have been computed unit per unit and mode per mode (DM: Figure 4-46, SHM: Figure 4-47, MAV: Figure 4-48).
Each unit power profile is defined by three values:

- a peak power
- a standby power

- a duty cycle value (duration of the peak power compared to the total duration)

For every mode, the peak and standby values have been added to obtain the values at system level. An equivalent duty cycle has also been computed to keep the same level of energy (See Figure 4-49).

	Mode 1				Mode 2				Mode 3			
	Descent				Surface (day)				Surface (night)			
	Pon	Pstdby	Dc	Wh	Pon	Pstdby	Dc	Wh	Pon	Pstdby	Dc	Wh
Thermal												
TOTAL	0	0	0	0	0	0	0	0	0	0	0	0
Comms												
UHFslot omnidirectional anter	0	0	0	0	0	0	0	0	0	0	0	0
X-band patch antenna	0	0	0	0	0	0	0	0	0	0	0	0
TOTAL	0	0	0	0	0	0	0	0	0	0	0	0
Propulsion												
TOTAL	0	0	0	0	0	0	0	0	0	0	0	0
DHS												
0.00	80	80	100	80	80	80	100	80	80	80	100	80
TOTAL	80	80	0	80	80	80	0	80	80	80	0	80
Life Support												
TOTAL	0	0	0	0	0	0	0	0	0	0	0	0
Mechanisms												
Pyro Release system- De-orbit	-	-	-	0	-	-	-	0	-	-	-	0
Separation Spring Units- De-o	-	-	-	0	-	-	-	0	-	-	-	0
Pyro Release System- Heat-sh	-	-	-	0	-	-	-	0	-	-	-	0
TOTAL	0	0	0	0	0	0	0	0	0	0	0	0
GNC												
TOTAL	0	0	0	0	0	0	0	0	0	0	0	0

Figure 4-46: DM power inputs

	Mode 1				Mode 2				Mode 3			
	Descent				Surface (day)				Surface (night)			
	Pon	Pstdb	Dc	Wh	Pon	Pstdb	Dc	Wh	Pon	Pstdb	Dc	Wh
Thermal												
SHM / radiator	0	0	0	0	0	0	0	0	0	0	0	0
SHM / heat pipes	0	0	0	0	0	0	0	0	0	0	0	0
SHM / heat pipes fluid	-	-	-	0	-	-	-	0	-	-	-	0
SHM / MLI 1 (below HP)	-	-	-	0	-	-	-	0	-	-	-	0
SHM / MLI 2	-	-	-	0	-	-	-	0	-	-	-	0
SHM / coil	-	-	-	0	-	-	-	0	-	-	-	0
SHM / conductive insulation	-	-	-	0	-	-	-	0	-	-	-	0
SHM / HCU nom	29	14	100	29	29	14	0	14	0	0	0	0
SHM / HCU red	-	-	-	0	-	-	-	0	-	-	-	0
SHM / Heaters/Thermostat/others	-	-	-	0	-	-	-	0	-	-	-	0
int. loop / liquid	-	-	-	0	-	-	-	0	-	-	-	0
int. loop / tubing	-	-	-	0	-	-	-	0	-	-	-	0
int. loop / insulation tubing	-	-	-	0	-	-	-	0	-	-	-	0
int. loop / pump assembly	-	-	-	0	-	-	-	0	-	-	-	0
int. loop / compensator	-	-	-	0	-	-	-	0	-	-	-	0
int. loop / liq-liq heat exchanger	-	-	-	0	-	-	-	0	-	-	-	0
int. loop / cooler-dryer assembly	-	-	-	0	-	-	-	0	-	-	-	0
int. loop / liquid electric heater	-	-	-	0	-	-	-	0	-	-	-	0
int. loop / filter	-	-	-	0	-	-	-	0	-	-	-	0
int. loop / hydraulic connector	-	-	-	0	-	-	-	0	-	-	-	0
int. loop / choke washer	-	-	-	0	-	-	-	0	-	-	-	0
int. loop / on-off valve	-	-	-	0	-	-	-	0	-	-	-	0
int. loop / pyro valve	-	-	-	0	-	-	-	0	-	-	-	0
int. loop / manual valve	-	-	-	0	-	-	-	0	-	-	-	0
int. loop / temperature sensor	-	-	-	0	-	-	-	0	-	-	-	0
int. loop / pressure sensor	-	-	-	0	-	-	-	0	-	-	-	0
ext. loop / liquid	-	-	-	0	-	-	-	0	-	-	-	0
ext. loop / tubing	-	-	-	0	-	-	-	0	-	-	-	0
ext. loop / insulation tubing	-	-	-	0	-	-	-	0	-	-	-	0
ext. loop / pump	-	-	-	0	-	-	-	0	-	-	-	0
ext. loop / compensator	-	-	-	0	-	-	-	0	-	-	-	0
TOTAL	29	14	100	29	29	14	0	14	0	0	0	0
Comms												
UHF omni antenna	0	0	0	0	0	0	0	0	0	0	0	0
X-band dish antenna	0	0	0	0	0	0	0	0	0	0	0	0
UHF transceiver	0	0	0	0	2	2	10	2	2	2	1	2
X-band transponder	0	0	0	0	20	20	100	20	20	20	100	20
TWT	0	0	0	0	120	10	100	120	120	10	100	120
Global RFDU unit	0	0	0	0	0	0	0	0	0	0	0	0
TOTAL	0	0	0	0	142	32	100	142	142	32	100	142
Propulsion												
Landing Engine (RD 861 G)	-	-	-	0	-	-	-	0	-	-	-	0
Oxid Tank	-	-	-	0	-	-	-	0	-	-	-	0
Fuel Tank	-	-	-	0	-	-	-	0	-	-	-	0
Others	-	-	-	0	-	-	-	0	-	-	-	0
TOTAL	0	0	0	0	0	0	0	0	0	0	0	0
DHS												
General System Computer	19	15	75	18	19	15	75	18	19	15	75	18
Communication Bus Stub	4	4	1	4	4	4	100	4	4	4	100	4
MMI+SCREEN	20	8	33	12	20	8	33	12	20	8	33	12
TOTAL	43	27	43	34	43	27	43	34	43	27	43	34
Life Support												
ALS Airlock Air Save Pump Package	2000	0	2	40	2000	0	2	40	2000	0	0	0
ALS CONVENTIONAL OVEN	800	0	4	32	800	0	4	32	0	0	0	0
ALS COOKING/EATING SUPPLY	0	0	4	0	0	0	4	0	0	0	0	0
ALS METOX CO2 removal (canister)	0	0	0	0	0	0	0	0	0	0	35	0
ALS METOX CO2 removal (regenerator)	0	0	0	0	0	0	0	0	2000	0	35	700
ALS VACUUM	200	0	4	8	200	0	4	8	200	0	0	0
GENERIC ACCUMULATOR URINE STORAGE TANK (57L)	100	0	10	10	100	0	10	10	100	0	10	10
GENERIC CLOTHING 10kg	0	0	0	0	0	0	0	0	0	0	0	0
GENERIC INORGANIC STORAGE(200kg)	20	0	3	1	20	0	3	1	20	0	3	1
GENERIC LiOH incl. Activated Carbon (1kg)	0	0	2	0	0	0	2	0	0	0	2	0
GENERIC OPERATIONAL SUPPLIES (20kg)	0	0	2	0	0	0	2	0	0	0	0	0
GENERIC ORGANIC STORAGE(200kg)	20	0	2	0	20	0	2	0	20	0	2	0
GENERIC PERSONAL STOWAGE SPACE (50kg)	0	0	2	0	0	0	2	0	0	0	2	0
GENERIC TOOLS EQUIPMENT (20kg)	0	0	2	0	0	0	2	0	0	0	0	0
ISS AAA - avionics air assembly	290	0	100	290	290	0	100	290	290	0	66	191
ISS ACCUMULATOR POTABLE STORAGE WATER TANK (57L)	0	0	4	0	0	0	4	0	0	0	4	0
ISS ACCUMULATOR WASTE WATER TANK (46L)	0	0	4	0	0	0	4	0	0	0	4	0
ISS ACCUMULATOR WASTE WATER TANK (46L)	0	0	4	0	0	0	4	0	0	0	4	0
ISS CCAA	936	0	100	936	936	0	100	936	936	0	66	618
ISS commode/urinal	910	0	4	36	910	0	4	36	910	0	4	36
ISS condensate storage	5	0	4	0	5	0	4	0	5	0	4	0
ISS EMU (Shuttle)	8	0	100	8	8	0	100	8	0	0	0	0
ISS HEPA - BACTERIAL FILTER	0	0	4	0	0	0	4	0	0	0	4	0
ISS IMV - intermodule ventilation fan assembly	110	0	100	110	110	0	100	110	110	0	60	66
ISS IMV - intermodule ventilation valve	380	0	4	15	380	0	4	15	380	0	4	15
ISS NITROGEN STORAGE TANK	0	0	5	0	0	0	5	0	0	0	0	0
ISS OXYGEN TANK PRESSURIZATION SYSTEM	1000	0	2	20	1000	0	2	20	1000	0	2	20
ISS PFE - portable fire extinguisher	0	0	100	0	0	0	100	0	0	0	100	0
ISS Sample Delivery System	0	0	4	0	0	0	4	0	0	0	4	0
ISS smoke detector	3	0	100	3	3	0	100	3	3	0	100	3
ISS TOCS - trace contaminant control system	239	0	4	10	239	0	4	10	239	0	4	10
ISS trash compactor	0	0	4	0	0	0	4	0	0	0	0	0
Manual Pressure Equalization Valve (MPEV)	0	0	2	0	0	0	2	0	0	0	0	0
microwave oven	1000	0	2	20	1000	0	2	20	0	0	0	0
personal hygiene kit	0	0	0	0	0	0	0	0	0	0	0	0
Portable Breathing Apparatus	0	0	0	0	0	0	0	0	0	0	0	0
Pressure Control Assembly (PCA)	250	0	0	0	250	0	0	0	0	0	0	0
RESTRAINTS AND MOBILITY AIDS (100kg)	0	0	0	0	0	0	0	0	0	0	0	0
SLEEP PROVISIONS	0	0	0	0	0	0	0	0	0	0	0	0
SOYUZ FOOD SUPPLY CONTAINER	0	0	100	0	0	0	100	0	0	0	0	0
X-38 HI PRESSURE GO2 REGULATOR	0	0	2	0	0	0	2	0	0	0	2	0
X-38 HI PRESSURE GN2 REGULATOR	0	0	2	0	0	0	2	0	0	0	2	0
X-38 LOW PRESSURE GO2 REGULATOR	0	0	2	0	0	0	2	0	0	0	2	0
X-38 LOW PRESSURE GN2 REGULATOR	0	0	2	0	0	0	2	0	0	0	2	0
TOTAL	8271	0	19	[illegible]	8271	0	19	[illegible]	8213	0	20	[illegible]
Mechanisms												
Antenna Pointing Mechanism	0	0	0	0	10	0	100	10	10	0	100	10
Deployment Boom	0	0	0	0	0	0	0	0	0	0	0	0
Hatch Door- Airlock	0	0	0	0	0	0	0	0	0	0	0	0
Hatch Door Locking Mechanisms- Airlock	0	0	0	0	0	0	0	0	0	0	0	0
Hatch Door- EVA Suit	0	0	0	0	0	0	0	0	0	0	0	0
Hatch Door Locking Mechanisms- EVA Suit	0	0	0	0	0	0	0	0	0	0	0	0
Sample Bio-Lock- Quarantine Chamber	0	0	0	0	0	0	0	0	0	0	0	0
Clamp-band- SHM/MAV I/F	0	0	0	0	0	0	0	0	0	0	0	0
APM Electronics	0	0	0	0	5	0	100	5	5	0	100	5
Landing Leg Locking System	-	-	-	0	-	-	-	0	-	-	-	0
TOTAL	0	0	0	0	15	0	100	15	15	0	100	15
GNC												
TOTAL	0	0	0	0	0	0	0	0	0	0	0	0

Figure 4-47: SHM power inputs

	Mode 1				Mode 2				Mode 3				Mode 4			
	Descent				Surface (day)				Surface (night)				Ascent			
	Pon	Pstdb	Dc	Wh	Pon	Pstdb	Dc	Wh	Pon	Pstdb	Dc	Wh	Pon	Pstdb	Dc	Wh
Thermal																
TOTAL	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Comms																
UHF omni antenna	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
X-band patch antenna	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
UHF transceiver	10	10	100	10	10	10	10	10	10	10	10	10	10	10	100	10
X-band transponder	20	20	100	20	28	5	0	5	28	5	0	5	28	5	100	28
Global RFDU unit	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
TWT	120	120	100	120	120	10	0	10	120	10	0	10	120	10	0	10
TOTAL	150	150	0	150	158	25	0	25	158	25	0	25	158	25	17	48
Propulsion																
First stage engine (Aestus advance)	40	40	0	40	0	0	0	0	0	0	0	0	40	40	0	40
TOTAL	40	40	0	40	0	0	0	0	0	0	0	0	40	40	0	40
DHS																
Descent/Ascent control unit	24	19	75	23	24	19	25	20	24	19	25	20	-	-	-	0
Communication Bus Stub	4	4	100	4	4	4	75	4	4	4	75	4	-	-	-	0
MMI+SCREEN	20	8	33	12	20	8	0	8	20	8	0	8	-	-	-	0
TOTAL	48	31	45	38	48	31	7	32	48	31	7	32	0	0	0	0
Life Support																
GENERIC PIPING 1/4"X1m	0	0	100	0	0	0	100	0	0	0	100	0	0	0	100	0
ISS PFE - portable fire extinguisher	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
ISS smoke detector	1	0	100	1	1	0	100	1	1	0	100	1	1	0	100	1
SOKHOL spacesuit	0	0	100	0	0	0	100	0	0	0	100	0	0	0	100	0
SOYUZ AIR PURIFICATION UNIT	20	4	5	5	20	4	5	5	20	4	5	5	20	4	5	5
SOYUZ ARGUS PRESSURE SENSOR	15	1	5	2	15	1	5	2	15	1	5	2	15	1	5	2
SOYUZ AUTOMATIC PRESSURE CONTROL UNIT	15	0	50	8	15	0	50	8	15	0	50	8	15	0	50	8
SOYUZ CHX	0	0	100	0	0	0	100	0	0	0	100	0	0	0	100	0
SOYUZ CONDENSATE STORAGE TANK	0	0	100	0	0	0	100	0	0	0	100	0	0	0	100	0
SOYUZ FAN	16	0	100	16	16	0	100	16	16	0	100	16	16	0	100	16
SOYUZ GAS ANALYZER	15	0	100	15	15	0	100	15	15	0	100	15	15	0	100	15
SOYUZ GASEOUS O2 STORAGE SYSTEM (12l)	0	0	5	0	0	0	5	0	0	0	5	0	0	0	5	0
SOYUZ GASEOUS O2 STORAGE SYSTEM (20l)	0	0	100	0	0	0	100	0	0	0	100	0	0	0	100	0
SOYUZ HAND PUMP	0	0	100	0	0	0	100	0	0	0	100	0	0	0	100	0
SOYUZ LIGHTING (20cm fluorecent)	66	0	100	66	66	0	100	66	66	0	100	66	66	0	100	66
SOYUZ LIGHTING (spot panel lighting)	0	0	100	0	0	0	100	0	0	0	100	0	0	0	100	0
SOYUZ O2 TANK PRESSURE SENSOR	2	2	100	2	2	2	100	2	2	2	100	2	2	2	100	2
SOYUZ O2 TANK WALL TEMPERATURE SENSOR	5	5	50	5	5	5	50	5	5	5	50	5	5	5	50	5
SOYUZ ONBOARD TANK	2	0	50	1	2	0	50	1	2	0	50	1	2	0	50	1
SOYUZ OXYGEN SUPPLY FITTINGS	910	0	4	36	910	0	4	36	910	0	4	36	910	0	4	36
SOYUZ personal hygiene kit	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
SOYUZ PORTABLE SURVIVAL KIT	0	0	100	0	0	0	100	0	0	0	100	0	0	0	100	0
SOYUZ PRESSURE ALARM SENSOR	1	0	100	1	1	0	100	1	1	0	100	1	1	0	100	1
SOYUZ PRESSURE EQUALIZATION VALVE	5	0	100	5	5	0	100	5	5	0	100	5	5	0	100	5
SOYUZ PURIFICATION CARTRIDGE (REGENERATOR)	0	0	100	0	0	0	100	0	0	0	100	0	0	0	100	0
SOYUZ RECEVING DEVICE	0	0	4	0	0	0	4	0	0	0	4	0	0	0	4	0
SOYUZ REFILLING VALVES	8	0	0	0	8	0	0	0	8	0	0	0	8	0	0	0
SOYUZ RELIEF VALVE	5	0	0	0	5	0	0	0	5	0	0	0	5	0	0	0
SOYUZ RESCUE AID AUTOMATIC UNIT	15	0	100	15	15	0	100	15	15	0	100	15	15	0	100	15
SOYUZ SAFETY UNIT	0	0	40	0	0	0	40	0	0	0	40	0	0	0	40	0
SOYUZ SOKHOL SUIT BLOWER ASSEMBLY	204	0	100	204	204	0	100	204	204	0	100	204	204	0	100	204
SOYUZ SOLID WASTE STORAGE CANISTER	0	0	100	0	0	0	100	0	0	0	100	0	0	0	100	0
SOYUZ SUIT FAN CIRCUIT BRAKER PANEL	10	1	100	10	10	1	100	10	10	1	100	10	10	1	100	10
SOYUZ TOILET	30	0	30	9	30	0	30	9	30	0	30	9	30	0	30	9
SOYUZ VALVE INHIBIT PANEL	10	0	5	1	10	0	5	1	10	0	5	1	10	0	5	1
TOTAL	1355	13	29	402	1355	13	29	402	1355	13	29	402	1355	13	29	402
Mechanisms																
Docking Mechanism- IBDM	784	0	1	9	-	-	-	0	-	-	-	0	-	-	-	0
Electronic Box- IBDM	-	-	-	0	-	-	-	0	-	-	-	0	-	-	-	0
Hatch Door- Egress External	-	-	-	0	-	-	-	0	-	-	-	0	-	-	-	0
Hatch Door Locking Mechanisms- Egress External	-	-	-	0	-	-	-	0	-	-	-	0	-	-	-	0
Hatch Door Cont. Protection Cover	-	-	-	0	-	-	-	0	-	-	-	0	-	-	-	0
Clamp-band- Stage 1/Stage 2 I/F	-	-	-	0	-	-	-	0	-	-	-	0	-	-	-	0
TOTAL	784	0	1	9	0	0	0	0	0	0	0	0	0	0	0	0
GNC																
TOTAL	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0

	Mode 5				Mode 6				Mode 7				Mode 8			
	Parking Orbit around Mars				Rendezvous and Docking Mode				MAV Orbital Safe Mode				0			
	Pon	Pstdb	Dc	Wh	Pon	Pstdb	Dc	Wh	Pon	Pstdb	Dc	Wh	Pon	Pstdb	Dc	Wh
Thermal																
TOTAL	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Comms																
UHF omni antenna	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
X-band patch antenna	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
UHF transceiver	10	10	100	10	10	10	100	10	10	10	100	10	0	0	0	0
X-band transponder	28	5	100	28	28	5	100	28	28	5	100	28	0	0	0	0
Global RFDU unit	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
TWT	120	10	0	10	120	10	0	10	120	10	0	10	0	0	0	0
TOTAL	158	25	17	48	158	25	17	48	158	25	17	48	0	0	0	0
Propulsion																
First stage engine (Aestus advance)	40	40	0	40	40	40	0	40	40	40	0	40	40	40	0	40
TOTAL	40	40	0	40	40	40	0	40	40	40	0	40	40	40	0	40
DHS																
Descent/Ascent control unit	-	-	-	0	-	-	-	0	-	-	-	0	-	-	-	0
Communication Bus Stub	-	-	-	0	-	-	-	0	-	-	-	0	-	-	-	0
MMI+SCREEN	-	-	-	0	-	-	-	0	-	-	-	0	-	-	-	0
TOTAL	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Life Support																
GENERIC PIPING 1/4"X1m	0	0	100	0	0	0	100	0	0	0	100	0	0	0	100	0
ISS PFE - portable fire extinguisher	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
ISS smoke detector	1	0	100	1	1	0	100	1	1	0	100	1	1	0	100	0
SOKHOL spacesuit	0	0	100	0	0	0	100	0	0	0	100	0	0	0	100	0
SOYUZ AIR PURIFICATION UNIT	20	4	5	5	20	4	5	5	20	4	5	5	20	4	5	4
SOYUZ ARGUS PRESSURE SENSOR	15	1	5	2	15	1	5	2	15	1	5	2	15	1	5	1
SOYUZ AUTOMATIC PRESSURE CONTROL UNIT	15	0	50	8	15	0	50	8	15	0	50	8	15	0	50	0
SOYUZ CHX	0	0	100	0	0	0	100	0	0	0	100	0	0	0	100	0
SOYUZ CONDENSATE STORAGE TANK	0	0	100	0	0	0	100	0	0	0	100	0	0	0	100	0
SOYUZ FAN	16	0	100	16	16	0	100	16	16	0	100	16	16	0	100	0
SOYUZ GAS ANALYZER	15	0	100	15	15	0	100	15	15	0	100	15	15	0	100	0
SOYUZ GASEOUS O2 STORAGE SYSTEM (12l)	0	0	5	0	0	0	5	0	0	0	5	0	0	0	5	0
SOYUZ GASEOUS O2 STORAGE SYSTEM (20l)	0	0	100	0	0	0	100	0	0	0	100	0	0	0	100	0
SOYUZ HAND PUMP	0	0	100	0	0	0	100	0	0	0	100	0	0	0	100	0
SOYUZ LIGHTING (20cm fluorecent)	66	0	100	66	66	0	100	66	66	0	100	66	66	0	100	0
SOYUZ LIGHTING (spot panel lighting)	0	0	100	0	0	0	100	0	0	0	100	0	0	0	100	0
SOYUZ O2 TANK PRESSURE SENSOR	2	2	100	2	2	2	100	2	2	2	100	2	2	2	100	2
SOYUZ O2 TANK WALL TEMPERATURE SENSOR	5	5	50	5	5	5	50	5	5	5	50	5	5	5	50	5
SOYUZ ONBOARD TANK	2	0	50	1	2	0	50	1	2	0	50	1	2	0	50	0
SOYUZ OXYGEN SUPPLY FITTINGS	910	0	4	36	910	0	4	36	910	0	4	36	910	0	4	0
SOYUZ personal hygiene kit	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
SOYUZ PORTABLE SURVIVAL KIT	0	0	100	0	0	0	100	0	0	0	100	0	0	0	100	0
SOYUZ PRESSURE ALARM SENSOR	1	0	100	1	1	0	100	1	1	0	100	1	1	0	100	0
SOYUZ PRESSURE EQUALIZATION VALVE	5	0	100	5	5	0	100	5	5	0	100	5	5	0	100	0
SOYUZ PURIFICATION CARTRIDGE (REGENERATOR)	0	0	100	0	0	0	100	0	0	0	100	0	0	0	100	0
SOYUZ RECEVING DEVICE	0	0	4	0	0	0	4	0	0	0	4	0	0	0	4	0
SOYUZ REFILLING VALVES	8	0	0	0	8	0	0	0	8	0	0	0	8	0	0	0
SOYUZ RELIEF VALVE	5	0	0	0	5	0	0	0	5	0	0	0	5	0	0	0
SOYUZ RESCUE AID AUTOMATIC UNIT	15	0	100	15	15	0	100	15	15	0	100	15	15	0	100	0
SOYUZ SAFETY UNIT	0	0	40	0	0	0	40	0	0	0	40	0	0	0	40	0
SOYUZ SOKHOL SUIT BLOWER ASSEMBLY	204	0	100	204	204	0	100	204	204	0	100	204	204	0	100	0
SOYUZ SOLID WASTE STORAGE CANISTER	0	0	100	0	0	0	100	0	0	0	100	0	0	0	100	0
SOYUZ SUIT FAN CIRCUIT BRAKER PANEL	10	1	100	10	10	1	100	10	10	1	100	10	10	1	100	1
SOYUZ TOILET	30	0	30	9	30	0	30	9	30	0	30	9	30	0	30	0
SOYUZ VALVE INHIBIT PANEL	10	0	5	1	10	0	5	1	10	0	5	1	10	0	5	0
TOTAL	1355	13	29	402	1355	13	29	402	1355	13	29	402	1355	13	0	13
Mechanisms																
Docking Mechanism- IBDM	-	-	-	0	1806	0	13	229	-	-	-	0	-	-	-	0
Electronic Box- IBDM	-	-	-	0	152	0	76	115	-	-	-	0	-	-	-	0
Hatch Door- Egress External	-	-	-	0	-	-	-	0	-	-	-	0	-	-	-	0
Hatch Door Locking Mechanisms- Egress External	-	-	-	0	-	-	-	0	-	-	-	0	-	-	-	0
Hatch Door Cont. Protection Cover	-	-	-	0	-	-	-	0	-	-	-	0	-	-	-	0
Clamp-band- Stage 1/Stage 2 I/F	-	-	-	0	-	-	-	0	-	-	-	0	-	-	-	0
TOTAL	0	0	0	0	1958	0	18	344	0	0	0	0	0	0	0	0
GNC																
TOTAL	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0

Figure 4-48: MAV Power Inputs

Mode	Solar Flux		Thermal	Comms	Propulsion	DHS	Life Support	Mech	GNC	Harness (excl. PCS)	TOTAL CONSUMPTION
			linked	linked	linked	linked	linked	linked	linked		
Descent	Solar Flux 0 W/m²	Pon	29 W	150 W	40 W	170 W	9626 W	784 W	0 W	216 W	11016 W
	(0 if Solar Array not used)	Pstdby	14 W	150 W	40 W	138 W	13 W	0 W	0 W	7 W	362 W
		Duty Cycle	100 %	0 %	0 %	44 %	20 %	1 %	0 %		19%
Tref 30 min	Eclipse Mode NOT Included	Total Wh	15 Wh	75 Wh	20 Wh	76 Wh	971 Wh	4 Wh	0 Wh	23 Wh	1184 Wh
Surface (day)	Solar Flux 0 W/m²	Pon	29 W	300 W	0 W	170 W	9626 W	15 W	0 W	203 W	10344 W
	(0 if Solar Array not used)	Pstdby	14 W	57 W	0 W	138 W	13 W	0 W	0 W	4 W	226 W
		Duty Cycle	0 %	45 %	0 %	25 %	20 %	100 %	0 %		21%
Tref 600 min	Eclipse Mode NOT Included	Total Wh	137 Wh	1670 Wh	0 Wh	1458 Wh	19414 Wh	150 Wh	0 Wh	457 Wh	23286 Wh
Surface (night)	Solar Flux 0 W/m²	Pon	0 W	300 W	0 W	170 W	9568 W	15 W	0 W	201 W	10255 W
	(0 if Solar Array not used)	Pstdby	0 W	57 W	0 W	138 W	13 W	0 W	0 W	4 W	212 W
		Duty Cycle	0 %	45 %	0 %	25 %	22 %	100 %	0 %		22%
Tref 870 min	Eclipse Mode NOT Included	Total Wh	0 Wh	2422 Wh	0 Wh	2114 Wh	30052 Wh	218 Wh	0 Wh	696 Wh	35501 Wh
Ascent	Solar Flux 0 W/m²	Pon	0 W	158 W	40 W	80 W	1355 W	0 W	0 W	33 W	1666 W
	(0 if Solar Array not used)	Pstdby	0 W	25 W	40 W	80 W	13 W	0 W	0 W	3 W	161 W
		Duty Cycle	0 %	17 %	0 %	0 %	29 %	0 %	0 %		28%
Tref 90 min	Eclipse Mode NOT Included	Total Wh	0 Wh	72 Wh	60 Wh	120 Wh	603 Wh	0 Wh	0 Wh	17 Wh	872 Wh
Parking Orbit around Mars	Solar Flux 493 W/m²	Pon	0 W	158 W	40 W	80 W	1355 W	0 W	0 W	33 W	1666 W
	(0 if Solar Array not used)	Pstdby	0 W	25 W	40 W	80 W	13 W	0 W	0 W	3 W	161 W
		Duty Cycle	0 %	17 %	0 %	0 %	29 %	0 %	0 %		28%
Tref 118 min	Eclipse Mode NOT Included	Total Wh	0 Wh	94 Wh	79 Wh	157 Wh	791 Wh	0 Wh	0 Wh	22 Wh	1144 Wh
Rendezvous and Docking Mode	Solar Flux 0 W/m²	Pon	0 W	158 W	40 W	80 W	1355 W	1958 W	0 W	72 W	3663 W
	(0 if Solar Array not used)	Pstdby	0 W	25 W	40 W	80 W	13 W	0 W	0 W	3 W	161 W
		Duty Cycle	0 %	17 %	0 %	0 %	29 %	18 %	0 %		22%
Tref 30 min	Eclipse Mode NOT Included	Total Wh	0 Wh	24 Wh	20 Wh	40 Wh	201 Wh	172 Wh	0 Wh	9 Wh	466 Wh
MAV Orbital Safe Mode	Solar Flux 493 W/m²	Pon	0 W	158 W	40 W	80 W	1355 W	0 W	0 W	33 W	1666 W
	(0 if Solar Array not used)	Pstdby	0 W	25 W	40 W	80 W	13 W	0 W	0 W	3 W	161 W
		Duty Cycle	0 %	17 %	0 %	0 %	29 %	0 %	0 %		28%
Tref 1440 min	Eclipse Mode NOT Included	Total Wh	0 Wh	1152 Wh	960 Wh	1920 Wh	9650 Wh	0 Wh	0 Wh	274 Wh	13955 Wh

Figure 4-49: Power inputs at MEV level

4.3.5.3 Mars environment description

On Mars, there are specific degradation factors affecting solar energy technology performances (typically solar cells):

- Presence of direct but also diffuse light from the atmosphere and suspended dust particles ('diffusion')
- Scattering of the sunlight spectrum towards the red end, due to suspended dust particles ('scattering')
- Dust deposition effects on solar array surfaces ('dust deposition')

Note that diffusion, scattering and dust deposition effects are not constant but vary depending on seasonal and geographical conditions, as well as on the occurrence of large dust storms.
In summary, solar cell efficiencies can be broadly affected by the Martian climate changes, and to an extent that can only partially be quantified for the time being.
Other important environmental factors are:

- Potentially lower operational temperatures than orbital conditions
- High-speed winds, but with much lower force than in similar circumstances on Earth, due to the lower ambient pressure (average value is 6.4 mbar compared with Earth average 1013 mbar)
- The presence of oxidizing soil characteristics (and potential corrosion of PVAs).

For more information on the issues presented above can be found in [RD83] through [RD85].
The source of information for the Sun irradiance at Mars ground level is the Martian Climate database ([RD83]), which provides accurate information relating to the Mars latitude/longitude

and the Solar longitude Ls (indicating the Mars seasons: if Ls = 0 represents the spring equinox, then Ls=90 is the summer solstice, Ls=180 the fall equinox and Ls=270 the winter solstice).
The selected database option is the Mars Global Surveyor Dust Scenario – January 2001, 'a best guess' representing the moderately dusty planet as observed by Mars Global Surveyvor (MGS) without the dust storms. This scenario is recommended for those who seek one annual scenario to represent the Martian mean climate, which is a reference for a moderate opacity of the atmosphere.

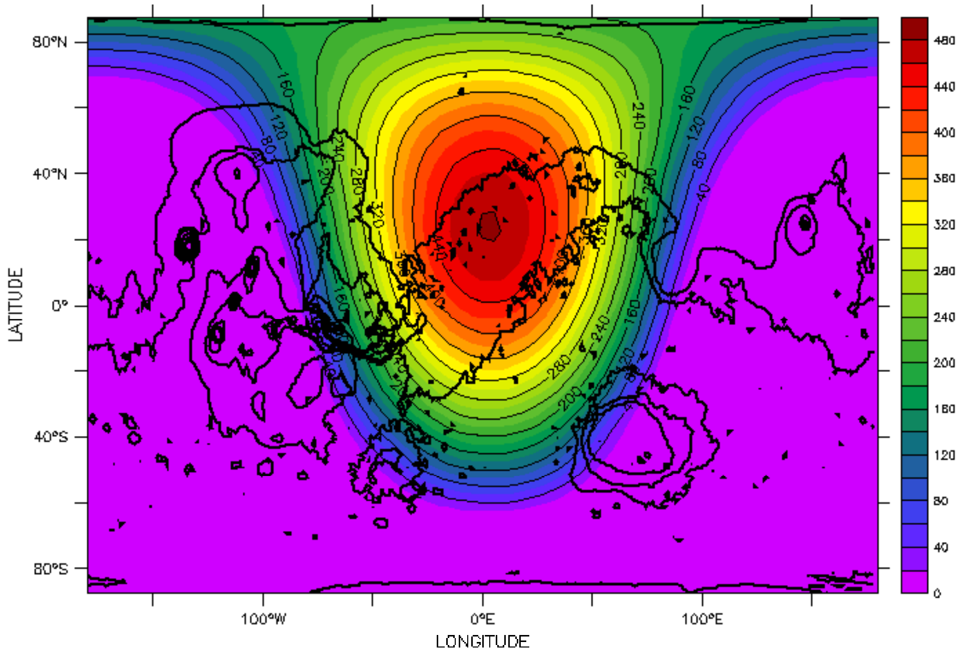


Figure 4-50: Example of Data (Solar Flux) from the Mars Climate Database

4.3.5.4 Trade-off between technologies for the SHM

Several processes to store or to generate energy on the Martian surface may be interesting. A specific chapter deals with the fuel cells since they can be used as power storage, as a power generation device or even more as part of the life support.

4.3.5.4.1 Power generation

4.3.5.4.1.1 Beamed power systems

With this concept, the energy is sent by microwaves or lasers from an orbiter to the SHM.
The main advantage is a light power collecting system.
The disadvantages are:

- requirement for a huge antenna on an orbiter
- orbiter should have an orbit providing the maximum visibility of the SHM
- technology is not proven in space
- the Martian dust may significantly affect the performances
- risk of depointment of the beam towards the crew

This power generation option was therefor rejected.
For information, a sizing has been estimated in [RD62].

4.3.5.4.1.2 Solar dynamic

This option consists of a lens or a mirror that focuses the sunlight onto a receiver. In this receiver, the collected sunlight provides heat to a thermal-conversion unit.

The NASA demonstrator has a low specific mass: 4.2W/kg with 17% of efficiency.
Moreover, this system may be more interesting by using thermal storage instead of batteries.
This system is interesting for high power needs, but the disadvantages that disqualified this concept are:

- the dust deposit of the mirror/lenses
- the need of a Sun-tracking system

4.3.5.4.1.3 Wind generator

This design could be performed on the Martian surface by mounting a wind generator on a hydrogen balloon.
It may be interesting during a dust storm when solar energy is not possible. Otherwise, this design is not reliable since it depends on the local wind strength.

4.3.5.4.1.4 Solar photovoltaic

The conversion of solar energy to electricity by solar cells is the most reliable way to generate power on the Martian surface. Moreover, it is the only power generation system already qualified in a Martian environment.
In addition to the severe environment constraint that affected the solar cells (See 4.3.5.3), the solar panels would have to be very large for being able to generate the required daily energy (more than 100m²).
To limit as much as possible this area, the most efficiency cells should be considered.
Currently, the efficiency of AsGa TJ cells is (in AM0 28°C conditions) 27%. For 2015, 30% is expected.
The inconvenience of these cells is that they are rigid and therefore mounted on a heavy rigid panel (total weight estimated: 3.33 kg/m^2).
Various developments are currently performed for optimising solar panels weight by using flexible structure or thin films. In 2015, a mass of 0.53 kg/m^2 can be expected with cells having an efficiency (AM0 28°C) around 15%. With that technology, the solar panels could consist of one (or several) blanket to unroll on the surface.

4.3.5.4.1.5 Other technologies

The following list shows the most promising other technologies with the rationales of their rejection in this study:

- Microturbines: Technology extremely immature and need to bring fuel
- Thermionic converters: Not European strength
- Thermoelectric generators: Low efficiency and still theoretical concept
- Photocatalytic decomposition of CO^2: Considerable further research required

None of these technologies is kept for this design. Nevertheless, if there are important improvements in the coming years, they should be reconsidered.

4.3.5.4.2 Power storage

4.3.5.4.2.1 Flywheels

Flywheels are mechanical batteries that convert energy to mechanical motion and when needed, convert that motion back to energy. They have a really high charge/discharge efficiency (85 to 95%) that also applied for high power demand.
The lifetime is estimated to be over 20 years.
The operating temperature range is interesting for Martian surface applications compared to chemical batteries.
Also, the expected specific energy is higher than for the secondary batteries.
In [RD77] is presented a NASA demonstrator turning at 60 000 rpm that can store up to 7.5 MJ.

The main disadvantages for Martian surface operation is the important self-discharge. The NASA demonstrator is discharged after only 12 hours. With eclipses lasting 14 hours, the flywheel cannot be used for this type of application.

4.3.5.4.2.2 Secondary batteries

The current secondary batteries that provide the best characteristics for the requirements of the surface operations are the Li-Ion cells.
The performances are:

- Specific energy: 100Wh/kg
- Efficiency Wh: 94%
- Temperature range: 0 to 40 °C
- Low self-discharge
- More than 5000 cycles

Some developments are in progress for extending the operational temperature range to –40 °C.
For this study, by looking the improvements during the last decades, a specific energy of 150Wh/kg is taken into account, which is expected to be available in 2015.

4.3.5.4.3 Fuel cells

4.3.5.4.3.1 Primary fuel cells

A fuel cell is a device that produces electricity through an electrochemical process within the fuel cell itself. This is very similar to the way a battery produces electricity. However, unlike a battery, a fuel cell only produces electricity while fuel is supplied to it. The primary fuel source for the fuel cell is hydrogen.

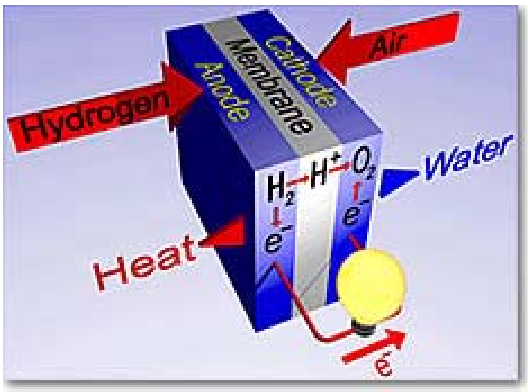


Figure 4-51: Primary FC with Air+H2 as reactants

There are five types of fuel cells:

- *Phosphoric Acid Fuel Cell (PAFC)*: phosphoric acid is used as an electrolyte. It needs to operate around 200°C and the cathode performance is inefficient.
- *Proton Exchange Membrane Fuel Cell (PEMFC)*: The mechanism is the same as PAFC. They differ in that PEMFCs operate at relatively low temperatures (about 100°C). They have high power density and can vary their output quickly to meet shifts in power demand
- *Molten Carbonate Fuel Cell (MCFC)*: An alkali metal carbonate (Li, Na, K) is used as the electrolyte. It needs to operate at about 600°C.
- *Solid Oxide Fuel Cell (SOFC)*: solid, nonporous metal oxide electrolytes are used. The cell operates at about 800-1000°C with an efficiency that can reach 60%.
- *Alkaline Fuel Cell (AFC)*: AFC uses alkaline potassium hydroxide as the electrolyte. These cells can achieve power generating efficiencies of up to 70%.

PEMFC and SOFC have the best maturity and European strength and fit the best with the Martian surface requirement.

Due to the relatively low efficiency of fuel cells, the thermal dissipation will be important and can be used for thermal regulation of fuel cells themselves but also for the rest of the spacecraft.

Using H_2/O_2 fuel cells has an important added value: water is the product of the reaction and can be used for the life support. Hence, the trade-off of the power system on the Martian surface should also include the water for the life support.

4.3.5.4.3.2 Regenerative fuel cells

Some fuel cells called "regenerative FC" can also be used in a reverse way: when power is supplied, an electrolysis process take place and the fuels are produced.

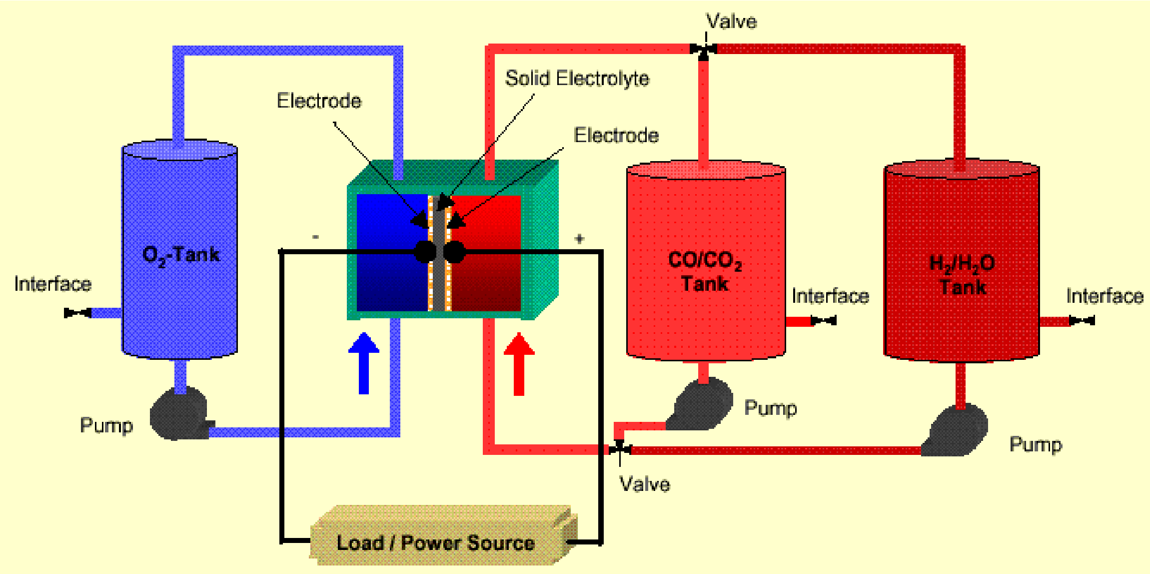


Figure 4-52: Regenerative Fuel Cell with $O_2/CO/H_2$ as reactants

Fuel cell technology becomes extremely interesting on the Martian surface, as soon as the fuels can be found there. Unfortunately, until now, except the ice located in some spots close to the poles, the only interesting element is the CO_2. Some processes are studied to reduce the CO_2 to CO and then O_2 ([RD70]), but therefore, power is first needed. Such a process should be really interesting for a permanent mars base but not for a mission of only 37 days.

[RD68] and [RD69] present the use of a regenerative-SOFC on the Martian surface using that concept of oxygen extraction from the mars atmosphere. With that concept, the fuel cells could provide:

- power for the electrical equipments
- heat for the thermal system
- fuel for the take-off from Mars
- oxygen and water for the life support

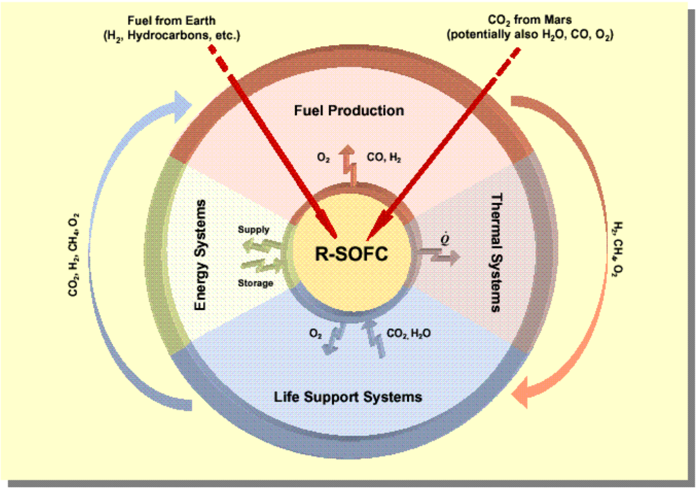


Figure 4-53: R-SOFC potential system on Mars

For this mission, the short stay on the surface is not sufficient for the fuel production. A trade-off has been performed in combination with the life support subsystem concluding that such a recycling design is not advantageous in term of mass.

Primary and regenerative fuel cells are kept as candidates for this mission.
Therefore, the fuel cell model is inspired from [RD5].

4.3.5.4.4 Conclusions

For the power subsystem of the Habitation Module, the topologies kept for the design are:

- Use of a secondary Li-Ion improved battery with solar cells (either thin film or mounted on rigid panels).
- Use of a regenerative fuel cell that can be daily recharged with solar cells (either thin film or mounted on rigid panels).
- Use of a primary H^2/O^2 fuel cells with tanks sized for providing the total energy required on the surface.

The sizing of these several subsystems are hereafter performed and compared.

4.3.5.5 Surface habitation module power design

4.3.5.5.1 Inputs and assumptions

4.3.5.5.1.1 Solar cells

The landing requirement area is all in the range between 20°N to 20°S. Figure 4-54 shows the daily solar energy on a horizontal surface of 1 m² depending on the latitude and the solar longitude of Mars. An absolute design that is fit for all landing dates would require as input an irradiance of 2000 Wh/m²/day.
Since the size of the solar panels is critical and the window of landing for the crew is about one terrestrial year, it has been chosen to add the following power constraint: “Surface Operations are not possible when the irradiance is under 3500 Wh/m²/day”.
This new operation requirement does not forbid any landing latitude. For every latitude in the range 20°N to 20°S, at least two time-opportunities are possible for a landing.
The periods in which landing on the surface will not be possible are summarized in Figure 4-55. Dust accumulation and contingency surface operations duration have been taken into account.

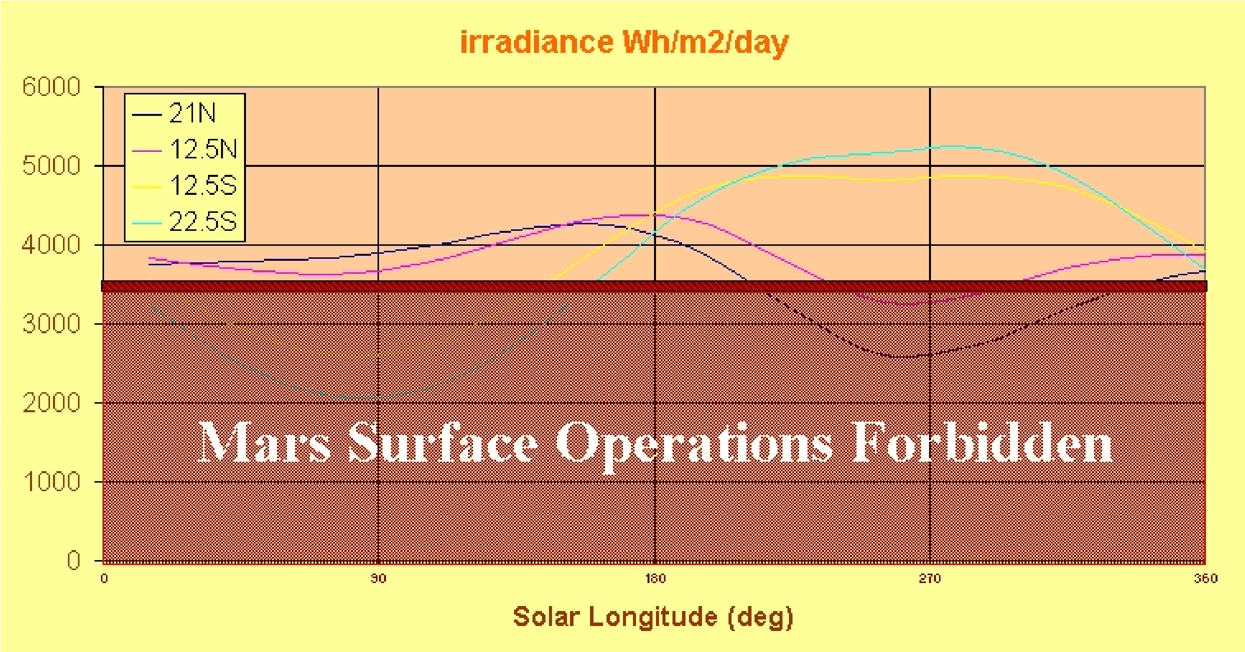


Figure 4-54: Solar irradiance on the Martian surface during one Martian year

Latitude	Start	End
12.5N	06/08/2035	04/11/2035
12.5S	16/05/2034	22/02/2035
21N	21/06/2035	19/01/2036
22.5S	23/04/2034	22/03/2035

Figure 4-55: Periods for which landing cannot be performed based on solar-cell design

For the designs without solar cells, this operations limitation simply disappears.
For example, the design based on primary fuel cells is not dependent the landing date nor the latitude of the landing site.

4.3.5.5.2 Fuel storage for FC

Hydrogen and oxygen can be stored either in gaseous or in liquid form.
For the gaseous forms, high-pressure storage is necessary to limit the volume, especially for the hydrogen that has a low density. Therefore, hydrogen is assumed to be stored at 700 bars.
The storage in a liquid state without losses due to boil-off can only be performed by cooling the tanks to 20° K for H_2 and 90° K for O_2. Therefore, the corresponding power requested for this thermal regulation has permanently to be supplied by the TV until the separation of the MEV. On the Martian surface itself, with the tanks protected by the structure, the boil-off has been estimated only around 1% per month. Consequently, the cooling of the tanks doesn't need to be continued during this phase.

4.3.5.5.3 PCDU

At this stage of definition, the power conditioning and distribution has not been analysed in details. Indeed, this module is not as critical as the power generation and storage modules for all aspects: mass, volume, technology, cost.

Nevertheless, mass, volume and efficiency of this subsystem is computed and taken into account in the design with an interpolation based on existing space PCDUs.

4.3.5.5.4 Budgets and trade-offs

The list of the selected architecture is Figure 4-56 shows the possible power options for architecture for surface operations:

Option1: Fuel Cells for 37 days with Liquid H2 + LOX
Option2: Fuel Cells for 37 days with gazeous forms
Option3: Solar Panels + Li-Ion Batteries
Option4: Solar Cells Blanket + Li-Ion Batteries
Option5: Solar Panels + Fuel Cells gazeous form
Option6: Solar Cells Blanket + Fuel Cells gazeous form
Option7: Solar Panels + Fuel Cells Liquid
Option 8: Solar Cells Blanket + Fuel Cells Liquid

Figure 4-56: List of power architecture options for surface operations

The comparison between the architectures are presented for:

- The total mass (Figure 4-57)
- The total deployed solar array area required (Figure 4-58)
- The volume of the power subsystem prior to the landing (Figure 4-59).

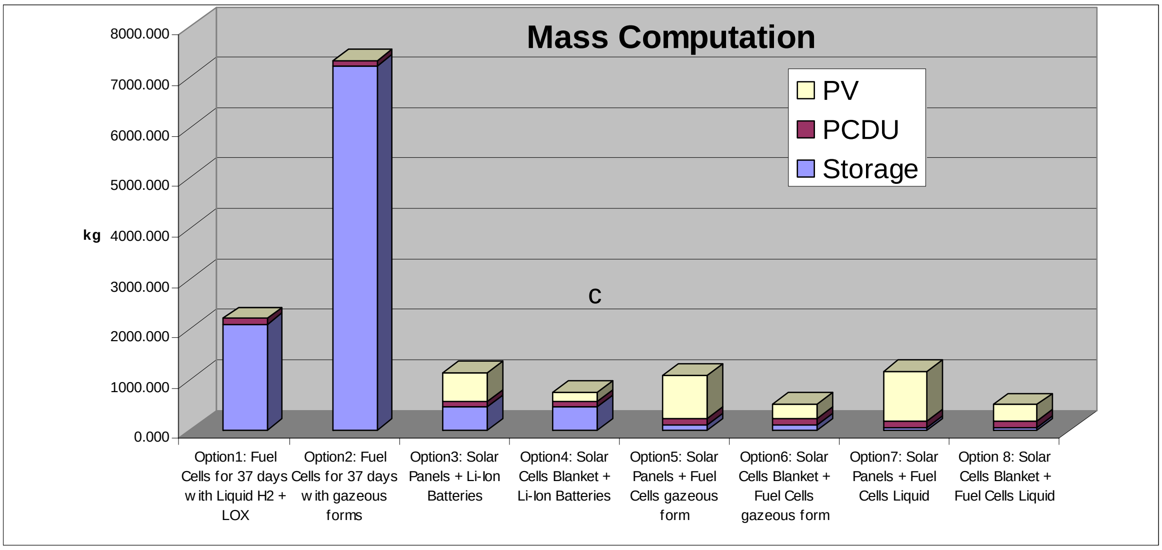


Figure 4-57: Mass comparison for the eight architecture options

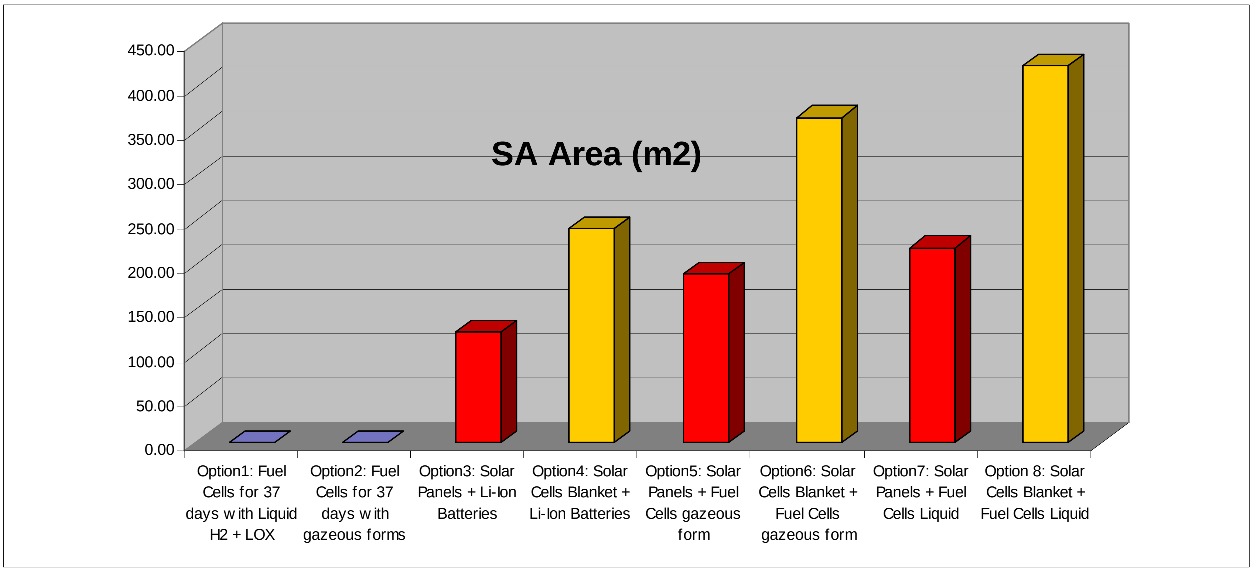


Figure 4-58: Area comparison of deployed solar arrays for the eight architecture options

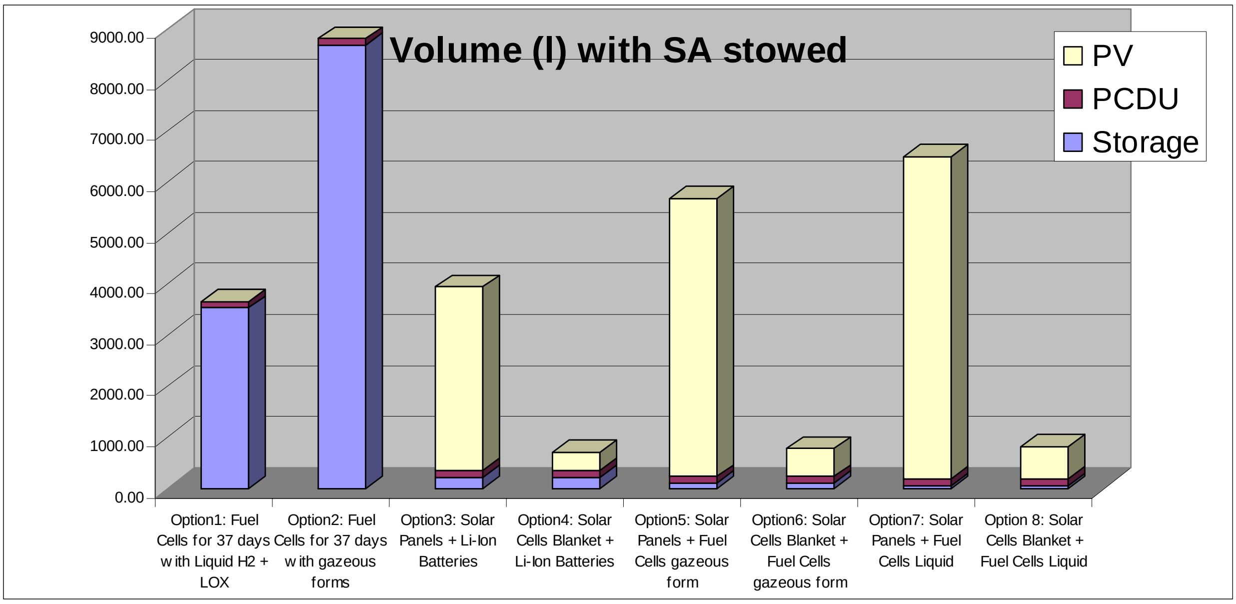


Figure 4-59: Comparison of required volume for the eight architecture options

Options 1 and 2 would produce 1075 litres of water that can be used for life support purposes (drinking water included). In total, during the surface operations, 437 litres are necessary for the astronauts. Hence, for these two options, the fact that water is not stored anymore in the life support subsystem, 437 litres and 437 kg has to be deducted from the power subsystem mass and volume figures.

Moreover, the production and consumption of the water is spread continuously for the duration of the mission. Therefore, the water storage tank does not need to be sized for the total duration of the stay on the surface.

Option 1 (liquid storage) has as main disadvantage the cooling by the TV during all the cruise to Mars. Nevertheless, the increase of the required TV solar panels seems feasible.

Option 2 (high pressure storage) has several advantages: the mass and the volume of the tanks are amazingly high, but also, the transportation of very high pressure tanks located close to the crew is a large risk in case of failure.
On the other hand, the use of regenerative system (here performed with solar cells) in the best case cannot be designed with a solar panel smaller than 100 m². Given the difficulties that the astronauts will have to face with the gravity just after the landing, it is not possible to assume a manual deployment of the panels within at least 2 days. Deployment mechanisms have to be mounted on the Habitation Module.
These mechanisms are expected to be heavy since they have to cope with the gravity on Martian surface and the huge solar array areas that need to be deployed.

In conclusion, the architecture that fits the requirements the best is Option 1: Primary hydrogen/oxygen fuel cells stored in cryogenic tanks. Another benefit of this topology is the possibility to combine the life support oxygen tanks with the ones of the power subsystem.

Figure 4-60 shows the mass budget of the selected option. The tank optimal shapes have not been studied. In this budget, they were spherical with diameters of 1.1 m. The oxygen tank includes also the part allocated for life support. The empty water tank is included in the mass budget of the life support subsystem. To avoid the boil-off during the cruise to Mars, a constant power consumption of 1800W is allocated to the thermal regulation of the tanks for the design of the TV.
The hydrogen tank mass is computed with the data given in [RD60]. For example, it is assumed that a cryogenic container can store 20 wt.% hydrogen. This value seems optimistic.
Other aspects that require closer investigation are:

- the estimation of the boil-off
- the power estimation of the tanks thermal regulation
- the gravimetric capacity of the tanks
- the shape of the tanks

Element 2: Surface Habitation Module			MASS [kg]			
Unit	Element 2 Unit Name Click on button below to **insert new unit**	Quantity	Mass per quantity excl. margin	Maturity Level	Margin	Total Mass incl. margin
1	Fuel Cells	1	35.8	To be developed	20	43.0
2	Tank O2 (Spheric)	2	733.1	To be developed	20	1759.4
3	Tank H2 (Spheric)	4	149.4	To be developed	20	717.3
4	PCDU	1	114.8	To be developed	20	137.8
5				Fully developed	5	0.0
-	Click on button below to insert new unit			To be developed	20	0.0
ELEMENT 2 SUBSYSTEM TOTAL		**4**	**2214.5**		**20.0**	**2657.4**

Figure 4-60: Mass budget of Option 1 SHM power subsystem

4.3.6 Data handling

The excursion vehicle's module integrated avionics shall be seen as a small subset of the one already described for THM.

4.3.6.1 Budgets

The mass and power for the avionics systems can be derived using the same mass/power ratio. The result lead to the following budgets, divided per functional module.

Element	Mass
Surface Habitation Module	40 kg
Mars Ascent Vehicle	30 kg

Table 4-16: Mass budget

The above figures have been derived considering a 'classical' approach to the DHS, including all the strictly necessary units. No evaluation for the harness has been made (apart from the usual 4% figure). The current assumption is that the Descent Module is 'free' of avionics.

As a comparison, the power consumption and mass of the avionics core units in Mars Express configuration are :
CDMU : 9.1 kg each, 19W (one active, one cold redundant)
RTU : 7.7 kg, 6W
AIU : 6.2 kg, 8W
SSMM : 8.7 kg, 17W (TBC)
Total : 40.8 kg, 50W

4.3.7 Communications

4.3.7.1 Requirements and design drivers

- Tracking, Telemetry and Command (TT&C) communications will be supported without any interruption longer than 1 hour.
- The maximum range that shall be supported is 1.37 AU and the minimum one 1.1 AU (maximum and minimum Earth / Mars distance respectively during surface operations)
- The telecommand (TC) and telemetry (TM) data rates shall be selectable to improve the data rate depending on the distance.
- Data rates should be optimised by giving realistic assumption of on-board equipment and ground segment availability.
- Data consists of housekeeping, high quality audio and video channels, and any additional data (for example internet access).
- Communications during EVAs shall be provided, for simultaneously two astronauts and for a maximum distance of 1 km from SHM.
- During EVAs, communications shall be possible even without direct visibility astronaut – SHM.

4.3.7.2 Assumptions and trade-offs

4.3.7.2.1 Communications availability from SHM: relay satellite

To maximize the communications from the SHM and MAV to the Earth once on the Martian surface, the visibility for communications with the TV and/or G/S has been studied. For example, for Gusev crater the black out for communications is around 6 h (so no visibility of TV or G/S). For other latitudes, the values change but are of the same order of magnitude. So, to accomplish the requirement of a maximum of 1 hour without communications with the Earth, a relay satellite must be included in the design.

Two kinds of relays have been considered: the first is satellites orbiting Mars and the second is a relay in a Sun - Earth Lagrangian point. This second option has the problem of its high distance to Mars, since basically the relay is in an Earth like orbit. So, data rate from Martian surface or TV would be low. Additionally, there is experience with relay satellites in Mars orbit, but not in Lagrangian points, so that option would be more complex. Therefore Mars relay has been selected.

To maximize the communications availability from SHM, Mars aerostationary satellites are needed, since they will be always visible from SHM. Two options have been considered: the first is a constellation of three relay satellites and the second is the minimum configuration, just one. The constellation would provide complete surface coverage and will avoid any eclipse of the Earth, as would happen with a single aerostationary satellite, but would be more complex and expensive. Since coverage is needed just in a small zone around the MEV landing point and the communications availability for relay –Earth communications is high enough (95% of time there is visibility), so the second option has been selected.

Due the situation of the SHM dish antennas and to avoid the occultation of the relay by the SHM structure, which could happen with high elevations, the relay satellite should not be placed just over SHM-MAV (in case the landing point is on the equator).

To select the link relay-G/S, Ka-band has been selected because it will give more link availability and will be simpler and cheaper than an optical link.

In addition, communications are required from SHM-MAV to the TV. Due to the low link availability for a direct transmission being about 12% of the time, to increase it TV could use the relay satellite as well.

4.3.7.2.2 Requirements for EVA

During EVA, a communication link must be provided. For EVAs, high-quality video and high quality voice is required within the astronauts' walking distance, approximately 1 km (despite a pessimistic value of 2 km will be used for the calculations). This distance is the maximum distance that an astronaut would be able to walk back to the base in case of contingency.

EVA astronaut operations are done in groups of two, so communication between both astronauts and with the SHM should be provided. Biomedical data, space suit data, high-quality voice and high-quality video are basic requirements, and the design will be done to support them. The data rate estimation for these requirements is shown in Table 4-17.

Improvement in video compression techniques is assumed by the mission date, so with 1.8 Mbps medium-high quality video can be transmitted.

The communication should be possible even if there is no direct visibility between the astronaut and the SHM. For this situation, a repeater should be used.

In the study's, attenuation because multipath effect on the Martian surface is considered, and its value is 20 dB *(*a more optimistic value of 15 dB was taken for NASA Mars Pathfinder).

*Data estimation for **return** link (EVA subject –> base):*	
Biomedical/Space suit or Vehicle	32 kbps
High Quality Voice	160 kbps
Medium quality video	1848 kbps
Other data	8 kbps
Total	***2048 Kbps***
*Data estimation for **forward** link (base -> EVA subject)*	
High Quality Voice	160 kbps
Other data	96 kbps
Total	***256 Kbps***

Table 4-17: Data requirements calculation for EVA operations.

4.3.7.3 Baseline design

4.3.7.3.1 SHM communications

SHM will have three main communication links:

- *X-band*: with the relay satellite. In case of contingency with the relay, a link to Earth G/S could be established using the same antenna
- *UHF*: for communications with the TV in case of contingency. The UHF antenna and transmitter of MAV would be used
- *UHF*: for EVAs. See section 4.3.7.3.3.

The SHM communications direct link availability, considering just the direct visibility in the worst case, with the different mission elements is:

- *G/S*: 50% of the time
- *TV*: UHF or X-band in contingencies, 12 %
- *Relay sat*: continuous communications, 100% availability

The X-band transponder characteristics are shown in Table 4-18. Two 1 m dish antennas with steering mechanism are included in the SHM, but only one will work at once depending on the aerostationary satellite or Earth position. The steering mechanism features are a coverage of 180° hemispherical, and a minimum pointing accuracy of 2º. See section 4.3.7.4 for details about the communications units.

See 3.3.7 TV communications report for a description of SHM/MAV with TV UHF link, and for X-band communications with G/S in contingency situations.

RECEIVER CHARACTERISTICS (X-BAND)	
Noise Figure	1.6 dB
Ranging Bandwidth (double sided)	3 MHz
TC Modulation Scheme, Link with relay satellite: Direct link with G/S in case of contingency:	GMSK NRZ/PSK/PM
Coding (NRZ/PSK/PM)	Turbo Code ¼ (*)
Coding (GMSK)	Concatenated: Convolutional + RS (255, 223)
TRANSMITTER CHARACTERISTICS	
RF Transmit Power	65W (SSPA)
TM Modulation Scheme	Baseline: GMSK (*)
(*) Availability of ground segment decoders-demodulators is supposed.	

Table 4-18: X-band transponder characteristics

4.3.7.3.2 Relay satellite

The aerostationary satellite will relay SHM/MAV, TV and G/S and communications will be possible between them. The link with SHM/MAV and TV will be in X-band, and will use Ka-band for the link with G/S. The time link availability relay-Earth (considering only the visibility) will be, in the worst case, 95% because of the Earth eclipses.

As an aerostationary satellite, the position will be at height 17 030 km and over the equator. A summary of the relay satellite antennas is shown in Table 4-19:

Band	Antenna characteristics	Transponder
Ka-band	Dish 4 m.	Two units, Transmitted power=65W
X-band	Dish 0.53 m, steering mechanism.	Two units, Transmitted power=30W
X-band	Dish 0.53 m, steering mechanism.	Two units, Transmitted power=30W

Table 4-19: Brief summary of the relay satellite antennas and transponders

4.3.7.3.3 EVA

Considering the drivers for EVAs described in Section 4.3.7.2.1, a summary for the designed link is shown in Figure 4-61 and Figure 4-62 (in this case, a repeater is used).

Two omni-directional antennas are used, one in the emitter and another one in the receiver part. The transmitted power is 500 mW. Proximity link [RD43] is the used protocol, in UHF. This protocol is defined for short range, bi-directional, fixed or mobile radio links, characterized by short time delays, moderate (not weak) signals, and short/independent sessions. The key parameters in the physical link between the EVA or rover and the MEV are shown in Table 4-20.

The used frequencies (in Table 4-21) and data rate will be taken from the protocol. The maximum data rate allowed, 2048 kbps, will be used. Figure 4-61 shows the frequency plan and data rate for EVAs.

Due to the wavelength of the UHF signal (around 1.75 m), the effect of Martian dust on the signal propagation is negligible.

The communications system in the space suit is similar to the one used in ISS. The radio has two single UHF channel transmitters, three single channel receivers and a switching mechanism. The low profile antenna is located in the (Portable Life Support System) PLSS.

	Forward link (EVA->Base)	Return link (Base->EVA)
Frequency	See Table 4-21.	
Data link layer protocol	Proximity link	
Data rate	2048 kbps	256 kbps
Coding	Convolutional, rate ½, constraint length 7 Viterbi code.	
Modulation scheme	Bi-Phase-L modulated directly into the carrier	
Modulation index	60º +/- 5%	
Bit Error rate	10^{-11}, obtained $Eb/N_0 = 8.4$ dB	
Doppler and Doppler rate	Negligible	

Table 4-20: Characteristics of the UHF link

UHF channel	Forward (MHz)	Return (MHz)
1	435.6	404.4
2	437.1	401.6
3	439.2	397.5
4	444.6	393.9
5	449	390

Table 4-21: UHF proximity link protocol channels.

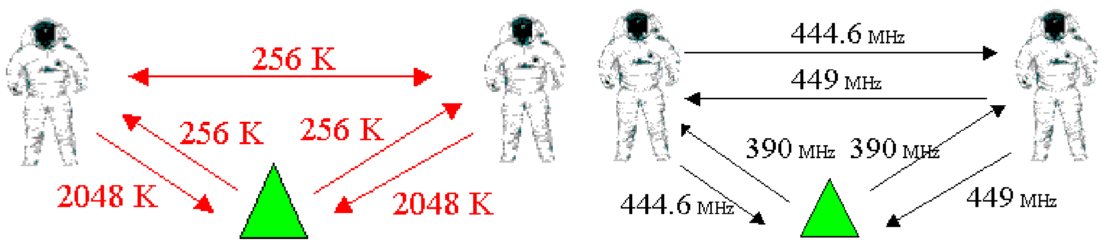


Figure 4-61: EVAs data rates and frequencies, respectively

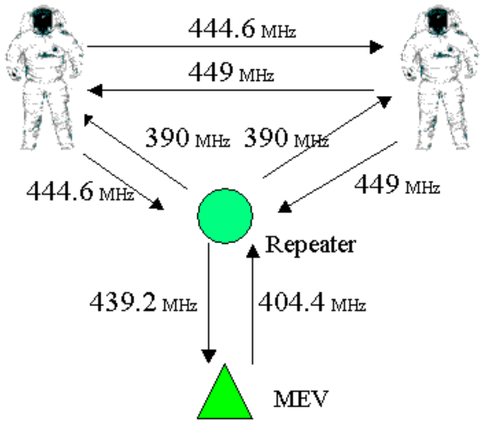


Figure 4-62: Communication frequencies using a repeater and data rates from Figure 4-61

4.3.7.4 Budgets

Unit	Number of units	Unit mass (kg)	Total mass (kg)	Power (W)
UHF omni antenna	2	1.0	2	
X-band dish antenna	2	2.5	5	
UHF transceiver	2	1.0	2	2.0
X-band transponder	2	4.6	9.2	20.0
TWT	2	0.8	1.6	120.0
Global RFDU unit	2	1.2	2.4	
Harness			3.3	
Total:			*25.5*	*142.0*

Table 4-22: SHM communications budget

Unit	Unit size (mm)	Total mass (kg)	Power (W)
UHF radio	304.8 x 109.2 x 88.9 mm		2
Omni antenna	304.9 mm length		
Total:		*3.95*	*2*

Table 4-23: Communications budget for the EVA space suit.

4.3.7.5 Options

The TV laser link could be used by the SHM to increase the data rate. The link to the TV would be done using the Mars relay satellite.

4.3.8 Mechanisms

4.3.8.1 Requirements and design drivers

The HMM science requirements do not set any specific requirements applicable to the SHM mechanisms. As a result of the SHM's configuration, the following necessary mechanisms and their requirements can be derived:

- Crew Egress Hatches:
  - External Hatches and Locking Mechanism at the SHM Separation I/F
- Vehicle Stage Separation System:
  - Release & Separation of MAV
- Crew Egress and EVA:
  - Egress Hatch (MAV/TV I/F)
  - EVA Suit Egress system
- Communication System:
  - Antenna Pointing and Tracking Mechanism Surface to Aerostationary satellite communication antenna:
    - Antenna diameter: 1 m
    - Antenna mass: 4 kg estimated
    - Coverage: 180° hemispherical.
    - Pointing Accuracy: 2°
- Sample Handling:
  - Bio-lock quarantine facility/enclosure
- Landing System:
  - Locking Latches for the Landing Leg damping system due to loads on heat shield during reentry.

- The COSPAR Planetary Protection rules apply for the in- and egress of the astronauts. The habitable volume of the SHM is assumed to be an extension of the Earth environment and therefore contamination is prohibited.

4.3.8.2 Assumptions and trade-offs

No system-specific assumptions or trade-off have been performed for the SHM mechanisms.

As regard sample handling, the quarantine enclosure forms part of the overall sample handling methodology described in the appropriate chapter 4.3.9.

The in- and egress of the astronauts presents a significant problem for the contamination of the SHM habitable volume. With adequate attention to the design of the latching a sealing mechanism associated with the restraint of the EVA suit to the SHM outer wall and the EVA suit hatch, the contamination can be minimised. However, a small surface area still remains that will see both the external Martian environment and the SHM habitable volume. Current decontamination developments are exploring chemical at the molecular level. These type of decontamination steps could be considered if the decontamination fluid is through in specific channels within the latch/seal.

4.3.8.3 Baseline design

4.3.8.3.1 Crew egress hatches

Sealable hatches are required for the following I/Fs

- SHM to MAV

The hatch diameter is sized to about 800 mm. The hatch will potentially require latch and seal mechanisms. Mass estimates shall be realised using a 'simple geometry' model.

The astronaut enters the EVA suits via a hatch in the airlock wall. Once the astronaut is ready, the airlock hatch door is closed which also closes the EVA suit by closing and sealing the life support back-pack unit. Once the suit has been verified as sealed, the suit is released from its mounting on the outside of the SHM module. Exiting the suit is the reverse process. The suit is first latched to the SHM outer wall at the location of the hatch. The hatch is then opened, which opens the rear of the EVA suit allowing the astronaut to exit the suit. The mass of such a device has been estimated. No detail design has been considered.

4.3.8.3.2 Vehicle separation

The separation of the MAV from the SHM prior to launch shall be realized with a pyrotechnic operated clamp-band of about 1.5 m.

4.3.8.3.3 Bio-lock quarantine facility

This is a circular enclosure, integral to the SHM outer wall, which houses the individual bio-lock facilities. The enclosure will provide a sealed area allowing the application of the explosive seal to the container and providing a quarantine volume to allow for seal verification.

The quarantine volumes are shown in Table 4-24:

OD (mm)	ID (mm)	Ext. Length (mm)	Int. Length (mm)	Material	Unit Mass (Kg)
360 (volume)	350	270	250	Ti	15.4
200 (lid)		50		Ti	7

Table 4-24: Quarantine volume

4.3.8.3.4 Landing system lock

This device will block the damping system of the legs due to reentry loading on the heat shield mounted to the landing system feet. No specific design has been considered. A mass estimate was used.

4.3.8.3.5 Communication antenna

Current APM systems are able to meet the pointing requirements for the antenna. A suitable unit has been chosen and an estimate of the deployable boom mass has been made.

4.3.8.4 Budgets

Element 2: Surface Habitation Module			MASS [kg]				DIMENSIONS [m]		
Unit	Element 2 Unit Name Click on button below to insert new unit	Quantity	Mass per quantity	Maturity Level	Margin	Total Mass incl. margin	Dim1 Length	Dim2 Width	Dim3 Height
1	Antenna Pointing Mechanism	2	1.0	To be modified	10	2.2	0.15	0.15	0.15
2	Deployment Boom	2	2.0	To be modified	10	4.4			
3	Hatch Door- Airlock	1	28.5	To be developed	20	34.2		0.9	
4	Hatch Door Locking Mechanisms- Airlock	1	120.0	To be developed	20	144.0	0.95	0.8	0.05
5	Hatch Door- EVA Suit	4	34.20	To be developed	20	164.2			
6	Hatch Door Locking Mechanisms- EVA Suit	4	144.0	To be developed	20	691.2			
7	Sample Bio-Lock- Quarantine Chamber	10	22.4	To be developed	20	268.8		360.0	270.0
8	Clamp-band- SHM/MAV I/F	1	15.6	To be modified	10	17.2		1.2	
9	APM Electronics	1	1.0	To be modified	10	1.1			
10	Landing Leg Locking System	4	5.0	To be developed	20	24.0			
-	Click on button below to insert new unit			To be developed	20	0.0		-	
	ELEMENT 2 SUBSYSTEM TOTAL	10	1128.0		19.8	1351.3			

Table 4-25: Mass budget

Note that the masses stated for the EVA and egress hatch and locking mechanisms are best-estimate figures.

Element 2: Surface Habitation Module			PPEAK AND POWER SPECIFICATION PER MODE									
Unit	Element 2 Unit Name Click on button below to insert new unit	Quantity	Ppeak	DESM Pon	DESM Pstby	DESM Dc	SDAYM Pon	SDAYM Pstby	SDAYM Dc	SNGM Pon	SNGM Pstby	SNGM Dc
1	Antenna Pointing Mechanism	2					10.0		100.0	10.0		100.0
2	Deployment Boom	2										
3	Hatch Door- Airlock	1										
4	Hatch Door Locking Mechanisms- Airlock	1										
5	Hatch Door- EVA Suit	4										
6	Hatch Door Locking Mechanisms- EVA Suit	4										
7	Sample Bio-Lock- Quarantine Chamber	10										
8	Clamp-band- SHM/MAV I/F	1										
9	APM Electronics	1					5.0		100.0	5.0		100.0
10	Landing Leg Locking System	4										
-	Click on button below to insert new unit						-					
	ELEMENT 2 SUBSYSTEM TOTAL	10	0.0	0.0	0.0		15.0	0.0		15.0	0.0	

Table 4-26: Power budget

4.3.8.5 Options

Further study of the lock and sealed I/F of the externally mounted EVA suits is required. Suitable decontamination methods need to be considered to guarantee no contamination can enter the habitable volume.

4.3.9 Sample handling

4.3.9.1 Requirements and design drivers

The following requirements are imposed upon the samples and the sample handling systems

1 COSPAR Planetary Protection rules apply:

- Category V applicable- In summary the following text applies:
  - **Category V** missions comprise all Earth-return missions. The concern for these missions is the protection of the terrestrial system, the Earth and the Moon. For Category V missions, in a subcategory defined as “restricted Earth return,” the highest degree of concern is expressed by the absolute prohibition of destructive impact upon return, the need for containment throughout the return phase of all returned hardware which directly contacted the target body or un-sterilzed material from the body, and the need for containment of any un-sterilised sample

collected and returned to Earth. If any sign of the existence of a non-terrestrial replicating entity is found, the returned sample must remain contained unless treated by an effective sterilizing procedure .

Sample Types- Returned to Earth

- As an indication, the total mass of sample material 100 kg, consisting of:
    - Surface samples
        - rocks, stones
        - surface soil
            - about 75%,
    - Subsurface samples
        - Core samples
            - about 20%, for example, 40 samples.
    - Atmospheric samples
            - More than 5%, 50 samples at Est. 0.1 kg
- Most samples returned to be contained with environmental control

4.3.9.2 Assumptions and trade-offs

The main hazard concerns arising from the Planetary Protection requirement above are:

- Difficult-to-control pathogen(s) capable of directly infecting human hosts (extremely unlikely).
- Life form capable of upsetting the current natural balance of Earth's ecosystem.

This has the following implication for HMM sample retrieval and handling:

- All samples to be enclosed in hermetically sealed containers with seal verification methods applied.
- The contact chain between the Martian environment and the Earth must be broken.
- The TV, MAV & SHM (and EVA suit) internal environments are an extension of the Earth's environment.

Note that all equipment that has been to the Martian surface, with the exception of the internal habitat, must be assumed to be contaminated. This will be assumed during the following discussions.

Sample handling constraint- an initial constraint applied to the samples and their containers is that they will remain outside the habitable volume.

The samples are collected during astronaut EVA activities. Table 4-27 shows an overview of the assumed method of collection and storage/transportation of the samples during EVA activities:

Sample types	EVA Retrieval Method	EVA Container
Surface samples **rocks** **Typical size shape**	Manual/by hnd Tooling- hmmer	Simple sealable bg
Stones **Typical size shape**	Manual/By Hand	Simple sealable bg
Surface soil **Type of substrate**	Manual/by hand Tooling- trowel	Cylindrical container with screw cap
Sub-surface samples **(Core samples)** **Typical depth 1-2 m** **Maintenance of core integrity** **Tooling**	Mobile handheld boring tool or station Core sample retrieved with drill head/bit (Ref MSR)	Tool head placed into cylindrical container with screw cap
Atmospheric samples	Manual/by hand	Cylindrical container with screw cap

Table 4-27: Sampling methodology

Fr sample retrieval and EVA transport, the following tooling is required:

- Trowel
- Multi-purpose hammer/pick
- Core sampling machine
- Sample transpor
- Sealable bags
- Soil containers with screw cap- typical volume 10 cm^3
- Atmospheric containers with screw cap - typical volume 10 cm^3
- Core sample drill head container with screw cap - Drill Head diameter 40 typically.

An open point for consideration is how are samples to be catalogued during EVA an the potential for different samples from different locations. Some non-complex method of cataloguing is required. Bar-coding of sample containers along with reference for time and place etc. is required.

Two Methods of 'Treatment' for the collected samples can be considered:

1. In situ (on Martian surface) detailed sample analysis:
    a. On-site during EVA by astronaut
    b. Sample returned to SHM for analysis
2. Sample returned to Earth for analysis

As regards in situ analysis, it is unlikely that the astronaut will have the equipment available to do detailed analysis during EVA. It is likely that equipment such as a microscope and an IR specrometer would be available to the astronaut to aid in the selection of sample material for further analysis.

Returning the samples to the SHM raises the following issues:

- For SHM in situ analysis, a Category V-compliant 'glovebox' is required.
    - Equivalent to Bio-safety Level 4+ facilities on Earth (these are not even possible on Earth).
    - Specialist training required for Crew?
    - All analysis equipment contained and remains within the glovebox.

Given the sample handling constraint the samples are to remain outside the habitable volume:

- The 'glovebox' must be integrated onto the external wall of the facility (possibly in the airlock for additional safety) with external direct access for depositing samples or alternatively a sealed unit external to SHM with real-time manipulator arm (less preferable).
- glovebox provides a direct barrier to Mars.
- No bio-lock, glovebox access door or additional tooling required to place samples in 'glovebox'.

This methodology will lead to a complex and fairly massive system that must allow access for the EVA astronauts to place the samples inside, but also provide a complete barrier to the habitable volume whilst allowing the astronaut inside the habitable volume to be able to see, manipulate and test the samples.

4.3.9.2.1 Modifications to sample handling constraint

Modifying the constraint will allow the samples to enter the habitable volume:

- Samples transferred to internal 'glovebox' for analysis.
- Glovebox becomes an isolated workbench in the habitable volume
- All samples must be contained in hermetically sealed containers for transfer to the 'glovebox'.
- 'Bio-lock' required at the external wall for samples.
- Additional tooling required within the glovebox to break container seal
- Glovebox has a bio-lock door for access- larger contamination risk.

Allowing the samples into the habitable volume for in situ analysis leads to a more complex system involving the requirement to seal and than access the samples once inside the 'glove-box'.

Due to complexity of sample handling (Integration of glovebox to external wall) and the Bio-safety level requirement, in situ analysis of samples will be limited to (remote/EVA) microscope and IR spectrometry analysis to aid in the selection of appropriate samples.

Returning the gathered samples to Earth requires the samples to be transferred between vehicles. If the sample handling constraint is obeyed, the following schematic for the sample transfer results, shown in Figure 4-63.

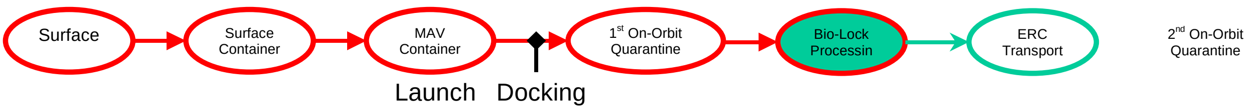


Figure 4-63: Sample transfer schematic II

The first two steps are performed by the EVA astronaut on the surface and are described briefly in the above text. The sequence will then continue as above. The steps and the consequences are briefly outlined below.

I. MAV Containers
   - External Storage bins or containers required on the MAV.
     - Containers require sealable caps (Level 1 sealing)- applied during EVA activities, no sealing verfication required.
     - Sample retention system required during MAV launch (especially for random shape rocks etc) i.e. foam inflatable filling bags.
   - Sample Environmental control required for externally mounted containers.
   - Access to containers required (difficulty for EVA ops)
     - Access Ladder along SHM and MAV required.
     - Access through potential MAV fairing required.
   - Storage Bins or Containers must be considered as contaminated (they are on the external surface of the module).

II. 1st on-orbit quarantine period to verify Level 1 sealing (sample containers are however contaminated).
   - Sample containers detached from MAV and passed to External Bio-lock facility
     - Astronaut EVA
       Due to the external surface of the MAV being contaminated, EVA suit becomes contaminated and is therefore not allowed back in the habitable volume
   - Remote manipulator required- Mass penalty
     - Complex arm required for simple task- container transfer.
     - Arm becomes contaminated due to container handling and must therefore be jettisoned prior to return.
   - MAV Container Env. Control system no longer functional- No Power

III. External ‘Bio-lock’ container sealing facility
   - Bio-Container principle has been developed mby JPL/NASA- Ref paper 001CES-131, Dolgin, Sanok, Sevilla & Bement
   - ‘Biolock’ container sealing as per MSR study- Explosive weld sealing.
     - Environmental Containment Verification Level 2
     - Container environmental control-Power.
     - Sealing Verifcation.
       - Inert gas pressurisation
       - Pressure and gas concentration monitoring

IV. Bio-Lock Sample containers placed in ERC Containers
- Externally mounted transport containers
  - Transfer performed by Astronaut EVA
  - Explosive weld sealing.
    - Environmental Containment Verification Level 3
  - Sealing Verification
    - Inert gas pressurisation
    - Pressure and gas concentration monitoring

V. Environmental Containment Verification Level 3
- Quarantine period during Mars orbit to verify Level 2 & 3 sealing (prior to EOI).
- Both powered and monitored through ERC I/F

4.3.9.2.2 Conclusions

- Maintaining the samples outside the habitable volume leads to a complex sample transfer system having the following non-preferable characteristics:
  - External Ladder access to MAV required through the potential fairing.
  - Contaminated Sample containers transported to the orbiting TV.
  - TV mounted external ‘Bio-lock’ facility required- extra mass
  - Remote manipulator required to transfer sample containers from the MAV to the ‘Bio-lock’ facility.
    1 Large disadvantage due to required complexity for such a ‘simple’ ge associated mass.
  - EVA required to transfer and install Bio-containers on the ERC.
  - (Small) risk of contamination being transferred to external surface of ERC.
  - Sample containment verification done in Mars Orbit
    1 Any non-conformance means loss of samples with no possibility of replacement.

Performing a similar analysis with a modified sample handling constraint to allow the samples to be transported internally in the MAV leads to the following.

Figure 4-64 shows a schematic for the sample transfer:

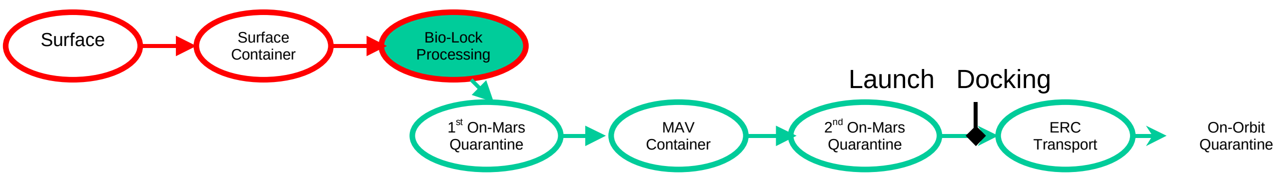


Figure 4-64: Sample transfer schematic II

The first two steps are performed by the EVA astronaut on the surface and are described briefly in the above text. The sequence will then continue as above. The steps and the consequences are briefly outlined below.

I. 'Bio-lock' container sealing facility mounted to external wall of SHM:
   Bio-Container principle has been developed by JPL/NASA- Ref paper 001CES-131, Dolgin, Sanok, Sevilla & Bement
   - 'Bio-lock' provides sample handling link through SHM wall
   - Samples placed into container by EVA Astronaut
   - 'Biolock' container sealing as per MSR study- 1st Level.
   - Double walled containment vessel
   - Explosive welding & I/F fracture
     i. Environmental Containment Verification Level 1

II. 1st Quarantine period on Mars to verify seal before transfer to MAV:
   - Container environmental control-Power connection provided by SHM
   - Minimum available quarantine time is the 7 day launch preparation
   - Sealing Verification- Level 1.
     i. Inert gas pressurisation
     ii. Pressure and gas concentration monitoring

III. Bio-Containers transferred from the quarantine enclosure to transport containers in the MAV:
   - Manually transferred by astronaut internally from SHM to MAV
   - MAV Transport Containers Sealed
   - Explosive welding
     i. Environmental Containment Verification Level 2
   - Container environmental control-Power connection provided by MAV- 1st & 2nd Levels

IV. 2nd quarantine period, on Mars to verify seal before MAV launch:
   - Container environmental control-Power connection provided by MAV
   - Time of quarantine is dependent upon when the last samples are transferred from SHM to MAV, suggest minimum of a couple of days
   - Sealing Verification Levels 1 & 2
     i. Inert gas pressurisation
     ii. Pressure and gas concentration monitoring.

V. MAV Transport Containers transferred from the MAV to the ERC:
   - Manually transferred by Astronaut internally from the MAV to the ERC.
   - ERC Transport Containers Sealed- 3rd Level
   - Clamping seal
     i. Environmental Containment Verification Level 3
     ii. (Explosive welding possible if required for risk mitigation/safety)

VI. 3rd quarantine period to verify seals before return:
   - Container environmental control-Power connection provided by ERC- 1st & 2nd Levels:

- Time of quarantine- remainder of Mars orbit period
- Sealing Verification Levels 1 & 2 (& 3)
    i. Inert gas pressurisation
    ii. Pressure and gas concentration monitoring

The conclusion that can be drawn from the above discussion is:

1 Passing the samples into the habitable volume does not create any technological difficulties (beyond those already known) or pose a significant contamination risk.
2 Further characteristics are:
    - Early sealing of samples in transport containers
    - Early verification of sample containment (before leaving Mars)
    - Ability to replace samples (collect more) if an individual Bio-lock containment cannot be verified at Level 1 (due to limitation of the number of bio-locks, other samples may have to be sacrificed)
    - Multi-level (stepwise contamination mitigation approach) sealing can be applied with individual level sealing verification
    - No EVA required beyond those for collecting the samples
    - Easy transfer of samples between vehicles
    - Minimal contamination risk to habitable volume

The baseline method for handling the samples will therefore be to transport them inside the habitable volume by using appropriate sealed containers.

4.3.9.2.3 Sample and container volumes (estimated mass)

The EVA canister volume (surface and atmospheric samples) shall be 147 cm^3- circular container of external dimension diameter 55 x 100 mm

The number of individual samples per type is assumed as follows:

Surface samples- x60, Typical individual sample volume 147 cm^3.
Atmospheric samples- x24, Typical individual sample volume 147 cm^3.
Subsurface samples- x20- Typical individual sample volume 9.72 cm^3.
Rocks- two canisters of internal volume 5890.5 cm^3

Assuming a sample mass density of 3100 kg/m^3, this surface sample volume of 20 617 cm^3 equates to a mass of 63.9 Kg (assuming 100% packing density and realising in addition that the density is only taken as an informative number on the upper bound). The total volume of atmospheric sample is 3534 cm^3. The total volume of subsurface samples is 194 cm^3, mass estimate 0.6 kg.

All mass estimates performed below will consider simple containers. This is deemed adequate to assess the mass impact at this stage.

4.3.9.3 Baseline design

4.3.9.3.1 Sample containers

Surface & atmospheric sample canisters shall be of the following size:

OD (mm)	ID (mm)	Ext. Length (mm)	Int. Length (mm)	Material	Unit Mass (kg)
55	50	100	75	Ti	0.405

Table 4-28: Surface and atmospheric sample canister characteristics

Sub- Surface (core) sample canisters shall be of the following size:

OD (mm)	ID (mm)	Ext. Length (mm)	Int. Length (mm)	Material	Unit Mass (kg)
40	35	120	75	Ti	0.440
35 (drill Head)		75		Ti. Carbide	0.357

Table 4-29: Sub-surface (core) sample canister characteristics

4.3.9.3.2 Bio-lock containers

Type 1 Container- typically for core sample containers, shall be of the following size:

OD (mm)	ID (mm)	Ext. Length (mm)	Int. Length (mm)	Material	Unit Mass (kg)
250	200	200	130	Ti	25.7

Table 4-30: Type 1 (surface and atmosphere) container characteristics

Type 2 Container - typically for surface and atmospheric sample containers, shall be of the following size:

OD (mm)	ID (mm)	Ext. Length (mm)	Int. Length (mm)	Material	Unit Mass (Kg)
310	260	200	110	Ti	41.5

Table 4-31: Type 2 (surface and atmosphere) container characteristics

Type 3 Container - typically rock samples, shall be of the following size:

OD (mm)	ID (mm)	Ext. Length (mm)	Int. Length (mm)	Material	Unit Mass (Kg)
300	250	200	120	Ti	37

Table 4-32: Type 3 (surface and atmosphere) container characteristics

4.3.9.3.3 MAV transport containers

Type 1 Container - typically for Surface and Rock Bio-containers, shall be of the following size:

OD (mm)	ID (mm)	Ext. Length (mm)	Int. Length (mm)	Material	Unit Mass (Kg)
720	710	220	210	Ti	28.8

Table 4-33: Type 1 container (surface and rock) characteristics

4.3.9.3.4 Bio-lock quarantine enclosure - SHM mounted.

The quarantine container shall be of the following size:

OD (mm)	ID (mm)	Ext. Length (mm)	Int. Length (mm)	Material	Unit Mass (Kg)
360 (enclosure)	350	270	250	Ti	15.4
200 (lid)		50		Ti	7.0

Table 4-34: Quarantine container characteristics

Figure 4-65 shows the bio-lock principle. In summary, the sample containers are placed in the internal volume. A lid is placed on the container. An explosive charge is set off which welds the I/F between the lid and Bio-lock container wall and fractures the connection to the outer support. As stated above, this principle is based upon the Bio-Container principle developed by JPL/NASA- Ref paper 001CES-131, Dolgin, Sanok, Sevilla & Bement and is adopted for this study.

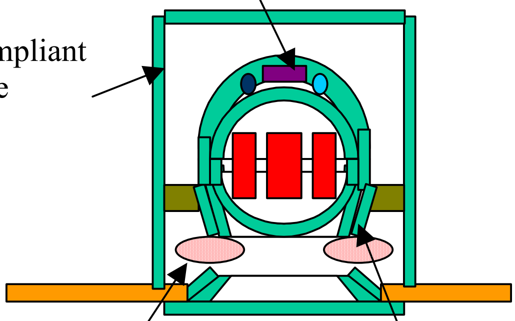


Figure 4-65: Bio-lock principle

4.3.9.4 Budgets

From the above design, the following mass budget is applicable

Individual canister masses:

Drill Head Canister	0.44	kg	
Drill Head Mass	0.36	kg	
		0.80	kg
Surface (soil) Sample Mass	0.46	kg	
Sample Canister	0.40	kg	
		0.86	kg
Atmospheric Sample Canister		0.40	kg

Bio-Lock Masses:

Core Samples		No./Bio-Container	Mass	Bio-Lock Mass	No. of Bio-Containers
Container Mass	25.68	1	25.68		
Core Sample Canister	0.80	10	7.975		
Atmospheric Sample Canister	0.40	3	1.21		
				34.87	2
					69.75

Surface Samples		No./Bio-Container		Bio-Lock Mass	No. of Bio-Containers
Container Mass	41.46	1	41.46		
Atmospheric Sample Canister Mass	0.40	3	1.21		
Surface Sample Canister Mass	0.86	10	8.61		
				51.29	6
					307.73

Rock Samples				Bio-Lock Mass	No. of Bio-Containers
Container Mass	36.94	1	36.94		
Typical Sample Mass in Bags	18.26	1	18.26		
				55.20	2
					110.41

MAV Transport Containers

Type 1		No./Transport Container	Mass	Trans. Cont. Mass	No. of Transport Containers
Container Mass	28.81	1	28.81		
Core Sample Bio-Container	34.87		0		
Surface Sample Bio-Container	51.29	2	102.58		
Rock Sample Bio-Container	55.20	1	55.20		
				186.59	2
					373.18

Type 2		No./Transport Container	Mass	Trans. Cont. Mass	No. of Transport Containers
Container Mass	28.80	1	28.80		
Core Sample Bio-Container	34.87	2	69.75		
Surface Sample Bio-Container	51.29	2	102.57		
Rock Sample Bio-Container	55.20	0	0		
				201.13	1
					201.13

Summary:

Element	Mass
Bio-lock quarantine containers- SHM	224 Kg
Container Mass Transported to Mars	509.8 Kg
Estimated Sample Mass	64.5 Kg
Sample & Container Mass returned	574.3 Kg

Table 4-35: Budget summary

4.3.10 Structures

4.3.10.1 Requirements and design drivers

For the design of the SHM on the Martian surface the following set of general requirements were taken into account:

- Compatibility with the vehicle launcher Energia-induced mechanical loads
- MEV centre of gravity has to be as low as possible, for stability during landing and on Martian surface.

All module structures shall provide the mechanical support to ensure mission success.

4.3.10.2 Assumptions and trade-off

The MAV centre of gravity is assumed to be at 1 m, with the referential at the bottom part of it. The aeroshell's centre of gravity is assumed to be in the middle of it.

4.3.10.3 Baseline design

The SHM consists of an aluminium cylinder with a cone on the top, of 4mm thickness. It has a total length of 7 m and a maximum diameter of 3.6 m.
As preliminary analysis the stiffener's mass was assumed to be half of the skin mass.
The centre of gravity of the MEV was achieved for two different situations, case 1: including the aeroshell and parachutes, SHM and MAV; case 2: excluding the aeroshell and parachutes.
To have the MEV centre of gravity as low as possible the centre of gravity of the SHM also has to be as low as possible. For the determination of it, the SHM was divided in equal parts and to each part was attributed a certain percentage of its mass. Figure 4-66 shows the mass distribution that was optimal for the centre of gravity.

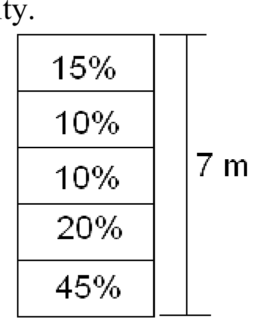


Figure 4-66: SHM Mass distribution

With the mass indicated at the bottom of the SHM, for the two cases analysed, the centre of gravity of the MEV is:

- CASE 1: 5.04 m
- CASE 2: 4.98 m

After a preliminary analysis using a cantilever beam, the first lateral eigen-frequency results in 72.8 Hz for the SHM.

The interior of the SHM is divided into two floors. The floor of the second level is assumed to be a plate, which has to support a uniform maximum load of 10 tonnes. This plate is made of aluminium, with 3.592 m diameter and 4 mm thick, [RD2]

For strength of the SHM, it was assumed to have a ring every 0.5 m. The aim of these rings is to give the necessary rigidity to the SHM. The design requirement used for the rings, for a twin-walled cylinder, was the Shanley design requirement:

$$(EI)_{Ring} = \frac{N_{cr} R^4}{1273.L}$$

Where N_{cr} is the axial load to be applied, R the radius of the cylinder and L the mutual distance between the rings. With this requirement the dimensions obtained for the aluminium rings was: 110mm for cross-section length and 5 mm for cross section thickness and web thickness, with a C cross-section area.

4.3.10.4 Budget

Item	Nr.	Mass [kg]	Margin [%]	Mass with Margin [kg]
SHM Skin	1	931.79	5	978.38
SHM Stiffeners	1	465.89	0	465.89
Ring Cylindrical part SHM	14	49.50	0	49.50
Ring Cone Part SHM	1	20.16	0	20.16
Floor-SHM	1	112.28	5	117.89
Support Cone	1	72.45	10	79.70
TOTAL				2355

Table 4-36: SHM structures mass budget

4.4 Descent Module

4.4.1 Entry Analysis

The entry analysis was performed as part of the Mission Analysis contribution to the CDF study.

4.4.1.1 Requirements and design drivers

The main requirement was to obtain a feasible entry trajectory for a low-lift inflatable aeroshell. De-orbiting shall take place from a low circular orbit. The aeroshell shall be guided via controlled variation of the bank angle, this changes the direction of the lift vector. Violent or complex control actions shall be avoided and a wide margin shall remain between the current value of the control parameter and the control boundaries, at the initial part of the entry.

4.4.1.2 Assumptions and trade-offs

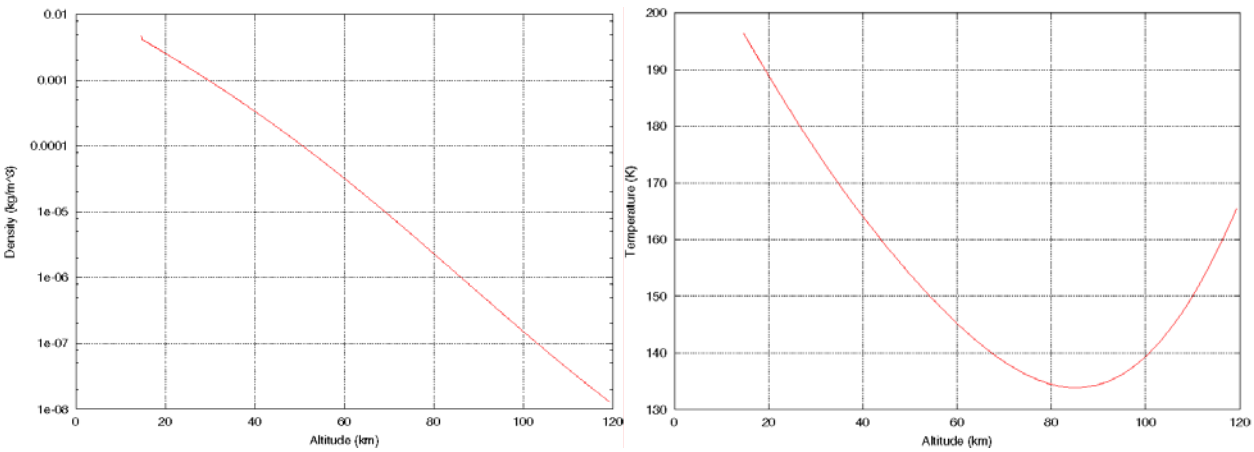


Figure 4-67: Assumed Atmospheric Density and Temperature Model

The following assumptions are made:

- Total mass of MEV at entry, including aeroshell: 46 500 kg
- Initial circular orbital altitude: 500 km
- Aerodynamic reference area 490.87 m^2
- Lift coefficient: 0.348
- Drag coefficient: 1.142
- Lift over drag: 0.305
- Aerodynamic parameters are assumed as independent of Mach number
- Angle of attack with respect to incident flow: fixed
- Control parameter: bank angle around direction of incident flow
- Atmospheric density model: Simplified approximation of a low density profile of MarsGRAM 2001, as shown above, not regarding positional, diurnal, seasonal and solar-activity-related variations
- Atmospheric temperature: Approximation of a temperature profile from MarsGRAM 2001
- Parachute deployment conditions: velocity < Mach 2 at altitude > 10 km

4.4.1.3 Baseline design

A wide corridor of possible entry angles was studied for the given configuration as described in the previous section. This corridor ranges from -2º to -15º with respect to the local horizon in a rotating Mars-fixed frame.

For all steep entry angles, starting with -3º, the bank angle is assumed to remain fixed at a value of 30º. This will leave ample control margins to cope with uncertainties in the atmospheric or aerodynamic properties.

For a very shallow entry, this control strategy is not appropriate. For the -2º case, a much larger initial bank angle is required to prevent a skip-out, for this, 110º is chosen. When the danger of skip-out is over, the body rolls to a bank angle of 0. Even shallower entry angles would require an initial bank angle of up to 180º. The upper limit to the entry corridor is at around -1.9º, where even with a full-downward lift, skip-out cannot be prevented.

Figure 4-68 shows the altitude profile for seven regarded entry angles in the specified range from 2º-15º below the local horizon. The following sets of diagrams show the comparative evolution of Mach number, dynamic pressure and g-load.

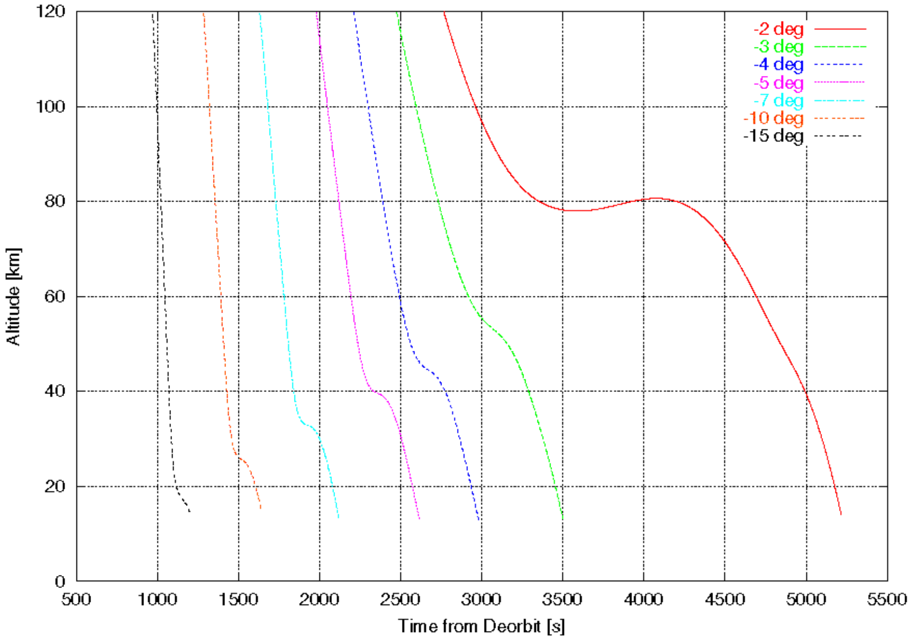


Figure 4-68: Altitude over Time for Seven Regarded Cases

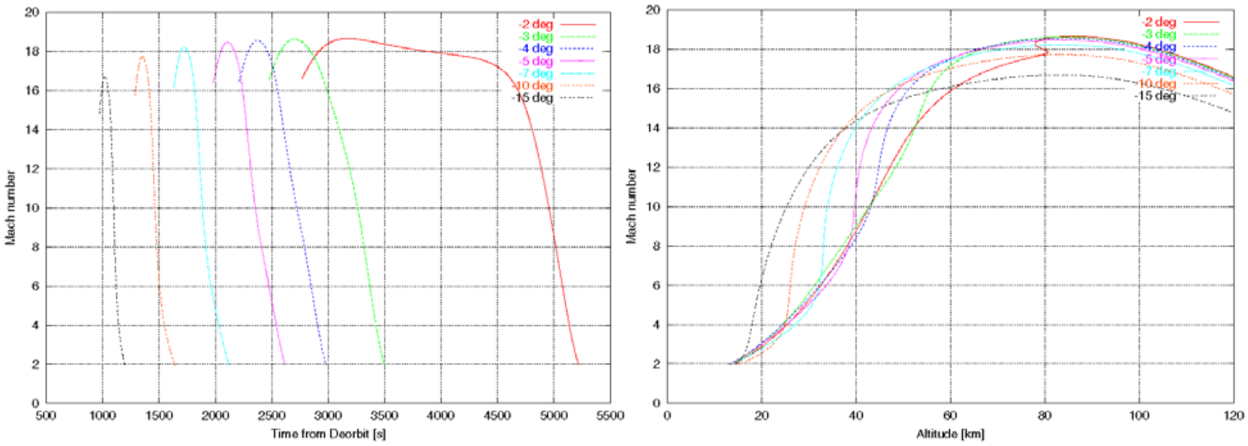


Figure 4-69: Mach Number over Time (L) and Altitude (R) for Regarded Cases

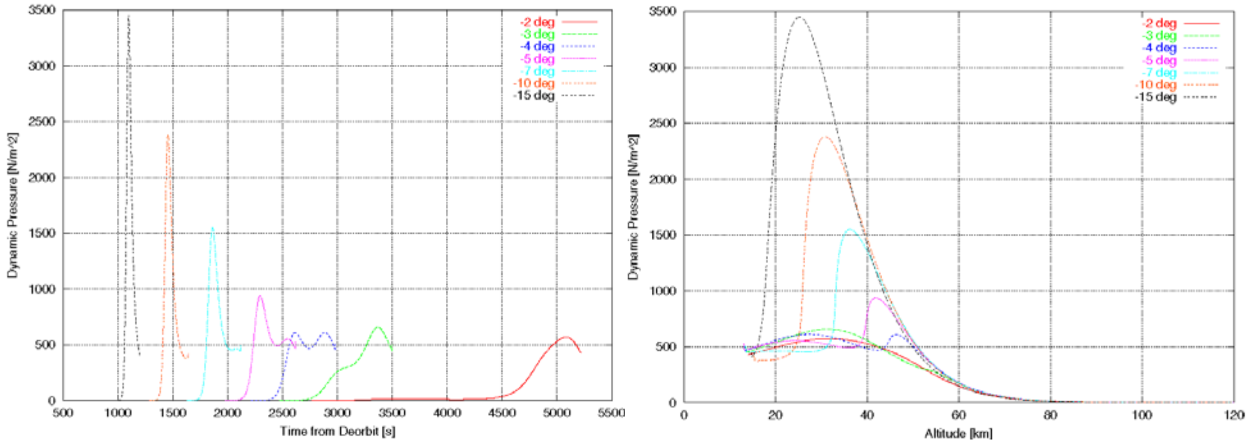


Figure 4-70: Dynamic Pressure Over Time (L) and Altitude (R) for Regarded Cases

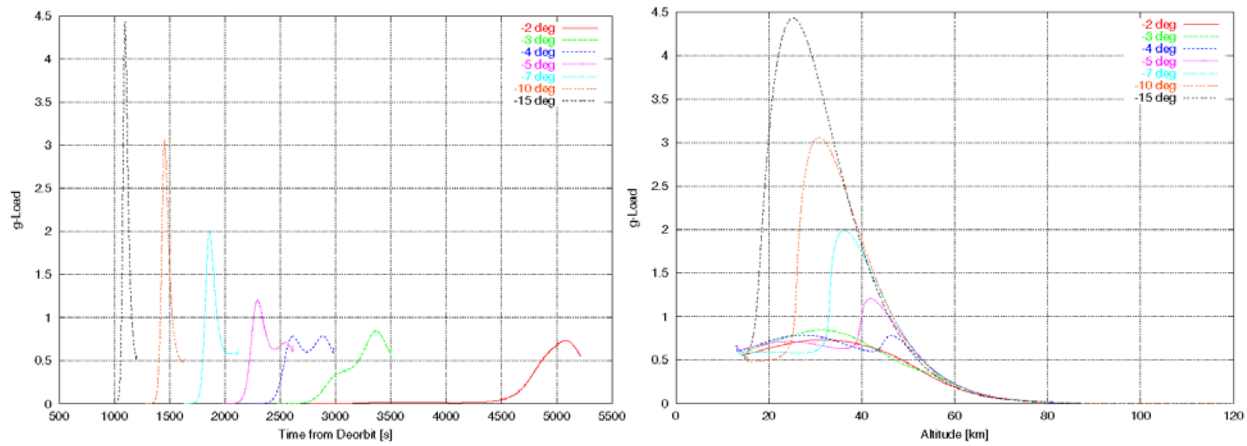


Figure 4-71: G-Load over Time (L) and Altitude (R) for Regarded Cases

As can be seen in the above figures, the duration of the entry phase varies considerably with the entry angle. Dynamic pressure and g-load increase sharply for steep entries, and also, the respective peaks shift towards lower altitudes, as shown in the right-hand plots.
It appears that an entry angle of -4° has a particularly well distributed load characteristics and therefore low structural load peaks. Of course, this feature will have to be investigated in later analysis. It might be a mere particularity of the given combination of conditions.
A further entity that needs to be investigated is the thermal load. The peak heat flux can be expected to rise sharply for a steep entry, analogously to the dynamic pressure and g-load. Conversely, the total integrated heat load can be expected to be slightly, but not dramatically, larger for shallow entry angles. Details on that analysis are given in the chapter on aerothermodynamics in this document.

4.4.1.3.1 Budgets

Table 4-37 shows the characteristics of the aerodynamic entry trajectory for the entry angle corridor with the assumed aerodynamic reference area of 490.87 m^2. Parachute deployment takes place at Mach 2, which is reached at an altitude of 13-15 km above the surface.
The entry velocity is about 10% larger at the shallowest end of the range, where also the longest phase duration is obtained. Conversely, for the steepest entry, the duration is reduced to 227 s.

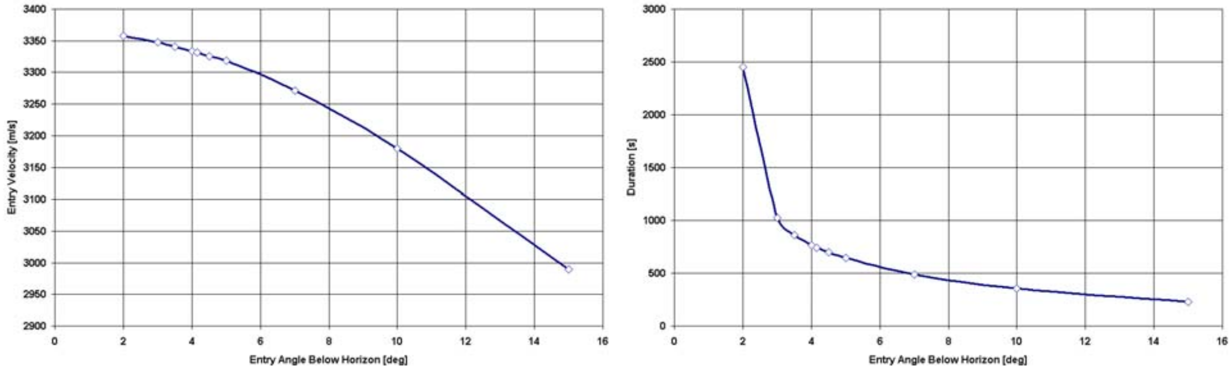


Figure 4-72: Entry Velocity (L) and Entry Phase Duration (R) as Function of Entry Angle

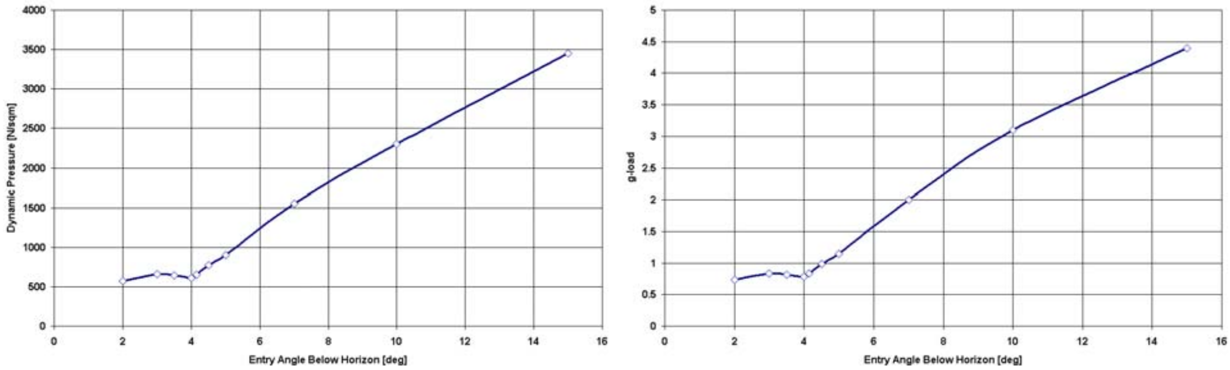


Figure 4-73: Peak Dynamic Pressure (L) and Peak G-Load (R) as Function of Entry Angle

The peak dynamic pressure and g-load show a local minimum at an entry angle of -4º and then sharply increase. This is due to the fact that the load appears to be more efficiently distributed over the entry profile for this entry angle. Note again that this feature would have to be ascertained in later analysis.

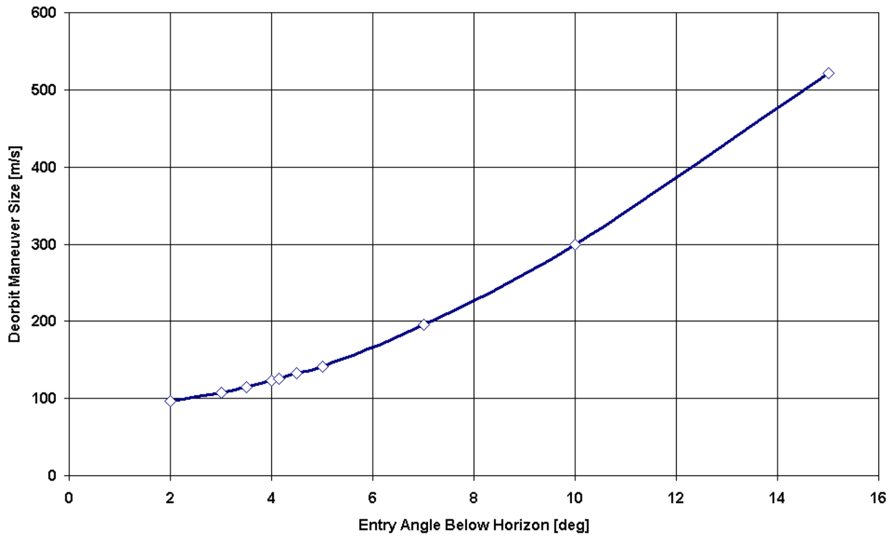


Figure 4-74: Size of Deorbit Burn as function of entry angle

The deorbit manoeuvre is at least 96 m/s and increases more than fivefold for a steep entry. This is an important factor to be taken into account in the final selection of the entry strategy. The following table lists the minimum and maximum values for all regarded trajectories occurring in the regarded entry corridor. On the basis of the provided information and taking into account the results of the aerothermodynamics analysis, this corridor must be narrowed down considerably and a baseline entry trajectory must be selected to obtain an optimal compromise between structural and thermal loads, duration, controllability, accuracy and deorbit manoeuvre propellant requirements.
Based on the current analysis, a target entry angle of -4º appears to constitute an acceptable compromise but this should be verified.

	Minimum	Maximum
De-orbit burn size [m/s]	96	521
Entry FPA [º]	-2	-15
Entry velocity [km/s]	2.990	3.358
Final altitude [km]	13	15
Flight time [s]	227	2450
Max g-load	0.73	4.4
Max dynamic Pressure [Pa]	570	3450

Table 4-37: Summary of Min/Max Characteristics for Regarded Entry Corridor

4.4.1.4 Options

Different inflatable aeroshell sizes were considered for comparison with the 490.87 m^2 baseline. A considerably smaller aeroshell reference area size of 300 m^2 is marginally acceptable. It barely meets the 10 km constraint for attaining parachute deployment velocity. This should be regarded as the lowest possible value when using inflatable aeroshell technology for the given entry mass range.
In the course of the CDF study the use of a rigid lifting body with a moderate lift/drag ratio of 0.83 was also considered instead of the inflatable aeroshell with an L/D of about 0.3. The results are not presented here but further analysis in that direction can be undertaken at a later stage.

4.4.2 Aerothermodynamics

4.4.2.1 Requirements and design drivers

The MEV design was driven by four requirements; the total mass (40 tonnes to 50m tonnes), the terminal velocity at 2000m of altitude (100m/s) due to the retro-rockets thrust and the design of the landing vehicle (vertical cylinder with the MAV at the top and retro-rockets at the bottom) and the g-load during the entry and descent phase.

Trajectories atmospheric entries are functions of parameters as entry conditions (velocity and flight path angle) and the ballistic coefficients of the vehicle (ratio of the reference surface over mass and drag coefficient).

Due to the high mass of the MEV, a vehicle with good drag coefficients to reduce the velocity and relatively good lifting coefficients is necessary to achieve the requirements and specially the terminal velocity and g-load.

4.4.2.2 Assumptions and trade-offs

Different shapes have been studied as a bent biconic shape (Figure 4-75), biconic blunt body as well as IBD model (Inflatable Braking Device) (Figure 4-76). First analysis showed that the MEV should have a reference surface greater than 300 m^2.

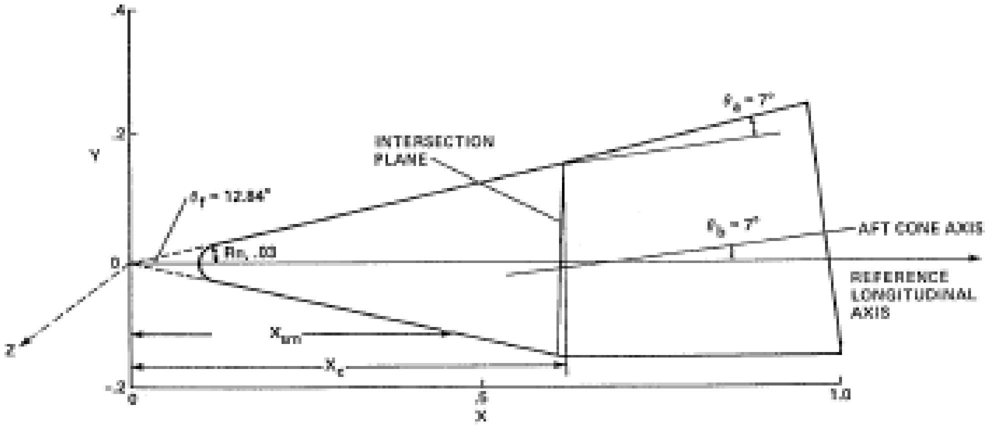


Fig. 1 Bent Biconic with $\theta_f = 12.84°$, $\theta_a = 7°$, $\theta_b = 7°$, $R_n = 0.03$, and $X_c = 0.6$.

Figure 4-75: Bent biconic shape

Figure 4-76: IBD shape

The IBD shape was preferred due to the reduced volume of the device during the launch. Three IBD shapes have been proposed corresponding to three cone angles (Figure 4-77): 45˚, 60˚ and 70˚half cone angle. Reference lengths and surface are reported in Table 4-38 shown.

Figure 4-78 shows IBD shapes that have been realised around a cylinder of 6 m diameter and 12 m long, representing the SHM and the MAV.

45˚ 60˚ 70˚

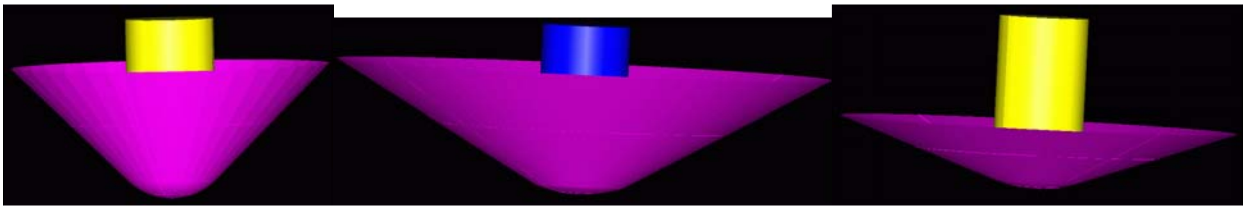


Figure 4-77: IBD shapes

		Half cone angle		
		45d	60d	70d
Lref	m	21.55	33.71	26.7
Sref	m2	364.74	892.5	560

Table 4-38: Reference IBD characteristics

The aerodynamic coefficients were computed with the Newtonian methods and are shown in the Figure 4-78:

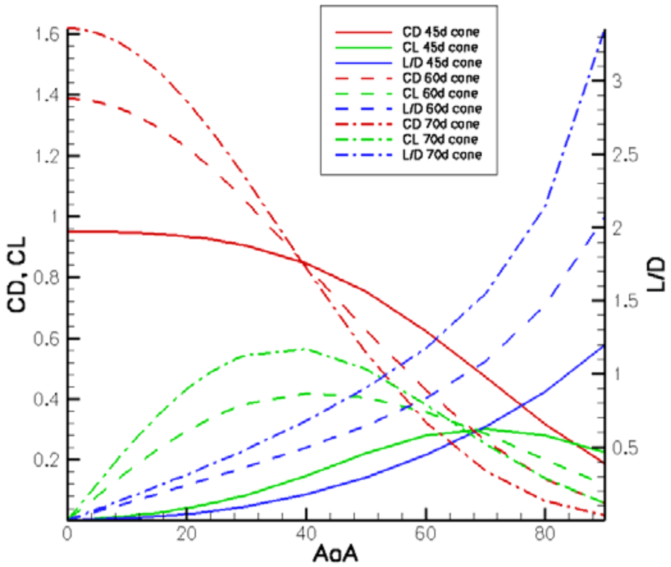


Figure 4-78: Aerodynamic coefficients comparison

The IBD shape selection is based on three coefficients: the drag and the lift coefficients and the reference surface. A drag coefficient and a reference surface should be high enough to reduce the velocity during the entry low enough to open the parachute; a lift coefficient should be enough to reduce the g-load during the entry according to the requirements for a human mission.

Another requirement to be taken into account is the stability of the vehicle. The stability can be analysed knowing the distance between the centre of pressure and the centre of gravity.

The module which has to be landed is made up from the bottom to the top by the retro-rockets needed to land, the habitation module and the MAV. The centre of gravity of this module is between half and three-quarter of the length from the bottom. Thus the centre of pressure shall be after this point, so the IBD shape should be long enough to fulfil this requirement.

4.4.2.3 Baseline design

The baseline for the design is an IBD shape with a 60˚ half cone angle, with a reference diameter of 25m (reference surface of $490m^2$). This shape was a good compromise between aerodynamic coefficients (drag, lift and L/D) and centre of pressure for the stability.

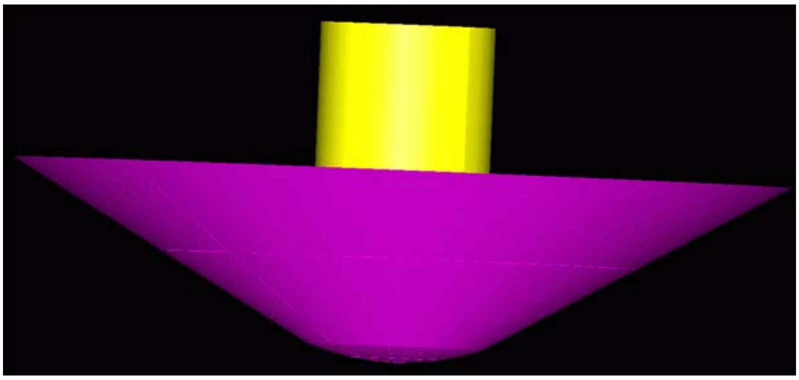


Figure 4-79: 60 degrees half cone angle, 25 m Base diameter

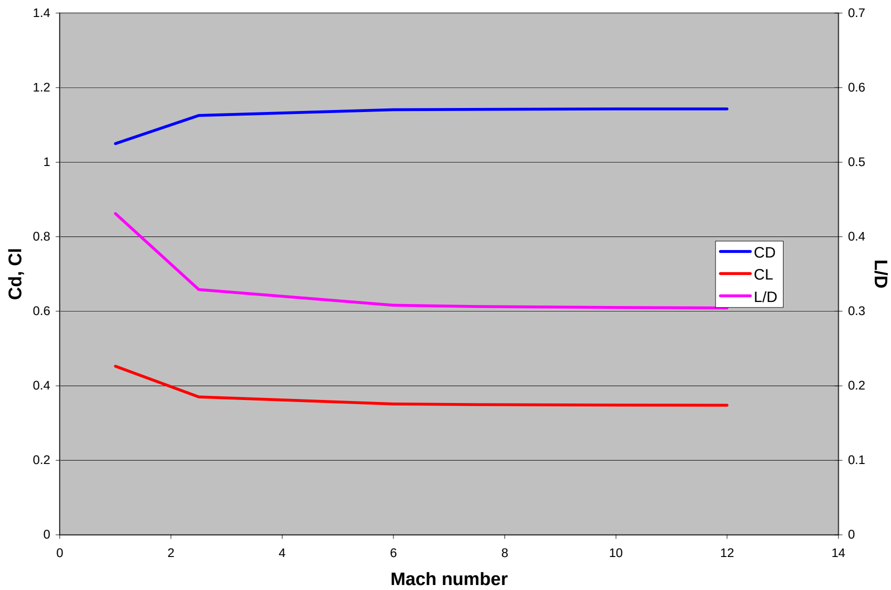


Figure 4-80: Aerodynamic coefficients vs Mach number

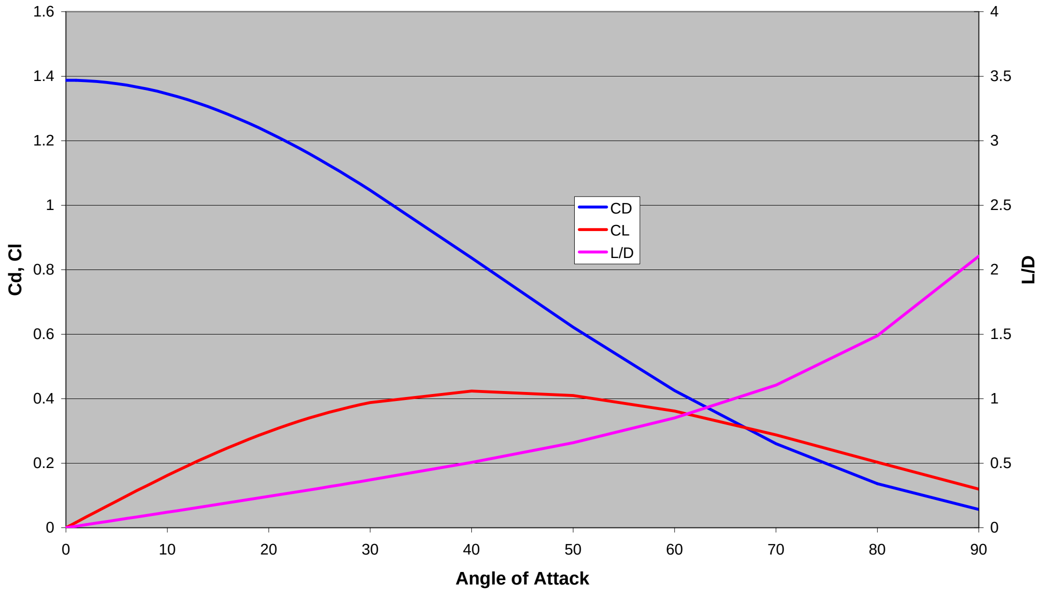


Figure 4-81: Aerodynamic coefficients vs Angle of Attack for Mach 7

4.4.2.3.1 Heat flux compilation

An aerothermodynamic computation was performed looking at two extreme cases with the following parameters:

Initial velocity: 3369 m/s
Entry angle: -4.9 d and -25 d
L/D: 0.3

The computations have been made using the MarsGramm 2001 atmospheric model.

Figure 4-82 and Figure 4-83 show the collective heat fluxes and the heat loads for the two cases:

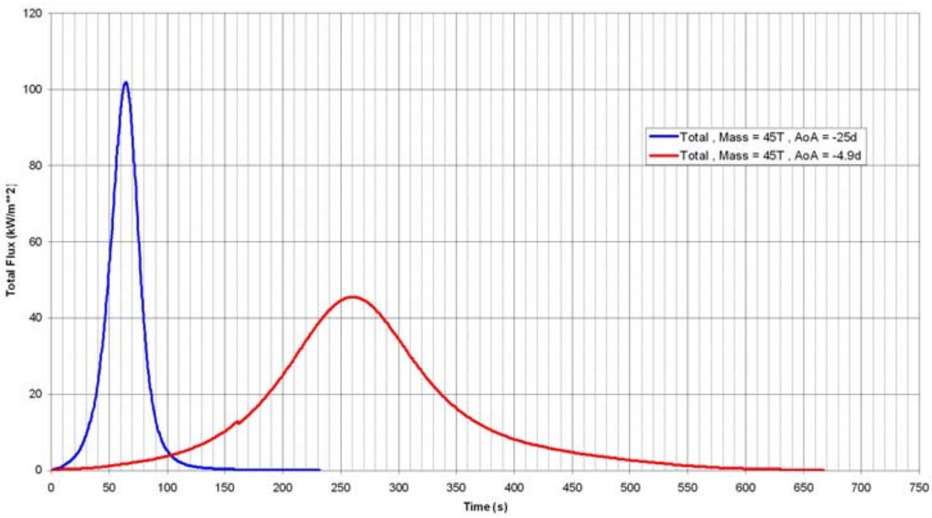


Figure 4-82: Total Heat flux versus time

The shallowest trajectory (-4.9d) has low acceleration and heat flux (around 50kW/m^2 instead of 100kW/m^2). However, this trajectory is worst for the TPS sizing and IBD design in terms of heat load (Figure 4-83). Heat load for an entry angle of -4.9d is about 7.5 MJ/m^2 whereas the maximal heat load for the steeper trajectory is about 3.5 MJ/m^2.

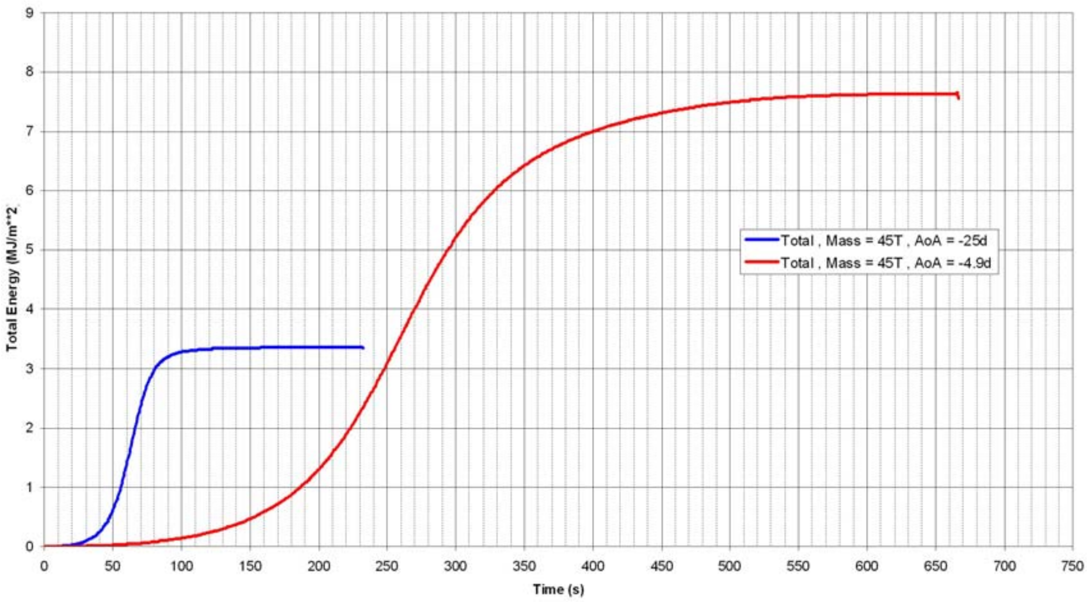


Figure 4-83: Heat load versus time

Nevertheless, for this high heat load, the shallowest entry has been selected.

Mach number requirement for the parachute is Mach 2. The corresponding altitude for the shallowest entry is about 14 km which fulfils the requirement.

4.4.2.3.2 Mass budgets

Mass evaluation of the IBD is extrapolated from the main IBD of the Exomars mass. For a 25 m diameter and 60 d half cone angle IBD shape, the mass is about 500 kg.

This mass takes into account only the IBD material, and the system for deployment.

4.4.3 Structures

4.4.3.1 Requirements and design drivers

For the design of the landing legs on Martian surface the following set of general requirements were taken into account:

- Compatibility with the vehicle launcher Energia induced mechanical loads.
- Maximum of 6 m-leg footprint, due to compatibility with Energia fairing.

All module structures shall provide the mechanical support to ensure mission success.

4.4.3.2 Assumptions and trade-offs

During descent four possible sceneries are possible. The cut-off of the engines at 2 m height with vertical velocity at this point between 0 and 2 m/s; cut-off of the engines at zero height, with velocity also between 0 and 2m/s.
In all cases, it was assumed to have a maximum horizontal velocity of 1 m/s and a deceleration of 0.5 s; consequently a maximum horizontal force of 60 000 N is present.
The vertical distance between the SHM and the Martian surface was assumed to be 1 m, due to the length of the retro rockets, and possible rocks in the landing site.
The leg footprint was assumed to be 6 m.

4.4.3.3 Baseline design

For landing stability, four legs with crushable shock-absorbing system and round footpads were selected. A three-leg design has the problem of stability in the presence of side-velocity if the spacecraft touches down moving away from one leg. A five-leg design does not improve much more since the leg structure is strongly driven by the one-leg- hits-first case. With five legs it would not be possible to make it as lighter as the number of legs increase. So the smallest number with reasonable stability is 4.
One principal leg, and two secondary legs constitute each leg. The one-leg-hits-first case was applied to the principal leg, which means that this one has to be able to withstand all loads, during touch down.

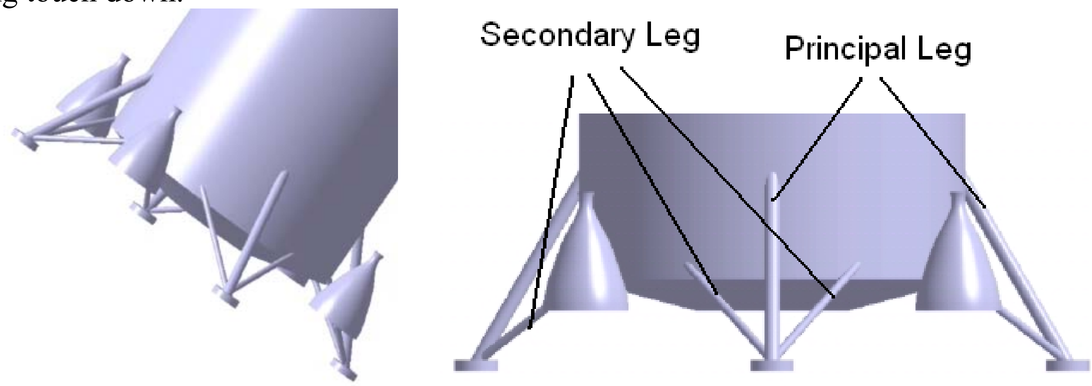


Figure 4-84: Landing-leg configuration

Aluminium was selected for the landing legs material, due to its low density and high strength.
For all the cases the horizontal and vertical forces involved were calculated, as well as the resultant force and it was concluded that the higher forces were involved when the cut-off of the engines occurred at 2 m height and with a vertical velocity of 2 m/s.

$H_{vertical}$ (m)	Min.$V_{vertical}$ (m/s)	$F_{vertical}$ due to Deceleration (N)	Total $F_{vertical}$ at motion extreme (N)	Angle (degrees)	Vector Force (N)
0	0	0	114 000	30.3	128 702.9

2	3.9	233 923.1	347 923	12.3	352 722.7

Table 4-39: Velocities and Forces with Vo=0 m/s

$H_{vertical}$ (m)	Max.$V_{vertical}$ (m/s)	$F_{vertical}$ due to Deceleration (N)	Total $F_{vertical}$ at motion extreme (N)	Angle (degrees)	Vector Force (N)
0	2	120 000	234 000	16.9	337 875.9
2	4.4	262 906.8	376 906.8	11.5	533 805.2

Table 4-40: Velocities and Forces with Vo=2 m/s

The method used for designing the landing legs consisted of assuming that each leg is a truss, case and it must be able to support all the load.
There are three possible cases for the angle of the legs with the SHM: it can be smaller, equal or higher than the angle of the resultant force. A brief analysis concluded that when these angles are equal, the force applied along the axial line of the leg is higher. Due to this the buckling and stress analysis were performed to this case. As the difference between these two angles increases, the lateral force increases, and the axial decreases. The maximum lateral force is 20% of the axial force. The case, which introduces higher stresses, is case 1; due to this the principal leg was designed to this one. A safety factor of 1.5 was applied to the resultant force, for the stress analysis, which results in a stress of 215 MPa for an axial force 1.5 higher than the expected.
Through the buckling analysis a minimum radius of 2 cm was obtained for the leg, but to fulfil the strength requirements a higher radius was necessary. A radius of 15 cm and a thickness of 4 mm was selected after the strength and buckling analysis.

4.4.3.4 Budget

Item	Nr.	Mass [kg]	Margin [%]	Mass with Margin [kg]
Principal Leg	4	61.8	10	67.96
Secondary Leg	8	16	10	17.70
TOTAL				413.44

Table 4-41: SHM Structures Mass composition

4.4.4 Communications

4.4.4.1 Requirements and design drivers

- The vehicle shall support Tracking, Telemetry and Command (TT&C) communications during all mission phases and any attitude.
- Communications availability should be maximized during all mission phases.

- The telecommand (TC) and telemetry (TM) data rates shall be selectable to improve the data rate depending on the distance to the receiving unit.
- During descent phases, data consist of housekeeping audio and any additional data.

4.4.4.2 Baseline

Communications during undocking, entry, descent and landing will be done using an UHF and a X-band link. An UHF slot antenna will be located in the docking port of MAV and will be used to communicate with the TV. Three switched X-band patches antennas will be located in the DM back shell (thermal protection) and will use the transponders and amplifiers located in the MAV (see MAV communications section 4.5.10 for further details). Therefore, the DM will not have active elements, since they will be present in SHM and MAV. After the DM shell is released, MAV antennas and transponders will be used for communications with the relay satellite.

The achieved data rates are shown in Figure 4-85. See section 4.3.8 TV report for details about the UHF link, MEV-TV link in it.

DM patch antenna **Relay antenna: 1 m with steering mechanism**		
	Uplink	**Downlink**
Frequency	7.23 GHz	8.5 GHz
Tx power	65 W	65 W
Modulation	QPSK	QPSK
Coding	Concatenated, Interleaving=5	
FER	10^{-5}	
Bit rate: *Max distance 18 600 km* *Min distance: 16 530 km*	 172 kbps 97 kbps	

Table 4-42: X-band link DM-Relay satellite

4.4.4.3 Contingency communications

Direct communications with the Earth could be possible using the X-band patches antennas, but at a very low data rate.

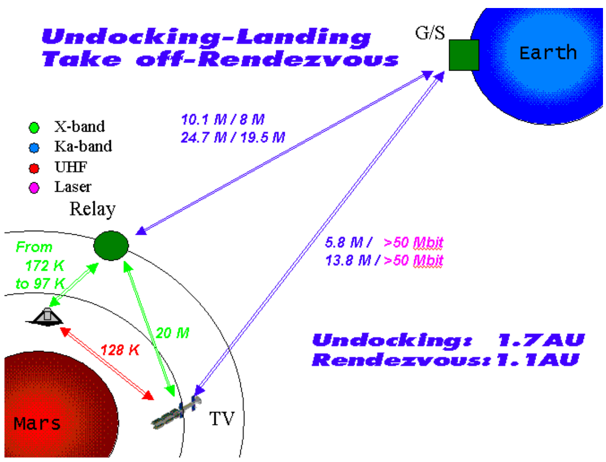


Figure 4-85: Communications MEV/MAV-TV during take off – rendezvous and undocking-landing

4.4.4.4 Budgets

Unit	*Number of units*	*Unit mass (Kg)*	*Total mass (Kg)*	*Power (W)*
UHF slot omnidirectional antenna	1	1.5	1.5	
X-band patch antenna	3	0.1	0.3	
Harness			3	
Total :			*4.8*	*0*

Table 4-43: DM communications budget

4.4.5 GNC

4.4.5.1 Requirements and design drivers

The MEV entry, descent and landing is a 6-degree of freedom (6DoF) closed loop control entry with automatic steering law that will allow the safe landing of the astronauts to the Martian surface.
The GNC requirements can be listed as follows:

- Starting from a 3DoF optimal trajectory that satisfies certain boundary and path constraints
- To find a flyable 6DoF trajectory within allowable margins that follows the optimal 3DoF trajectory previously calculated off-line
- To define a GNC equipment that controls and steers the MEV in terms of sensors and actuators

The initial and final boundary constraints as well as the path constraints are as follows:

- Safety of the astronauts with a possible manual control during entry, descent and landing
- Minimise the heat flux
- Maintain a given load factor for the health of the crew
- Minimise the descent time to be able to cope with the limited life support system available on-board the MEV
- Touch down in a specified point on the Martian surface

The output of the optimal trajectory establishes the corresponding roll, pitch and yaw profiles to be followed by the control law of the MEV.
The Figure 4-86 shows the MEV model during this study:

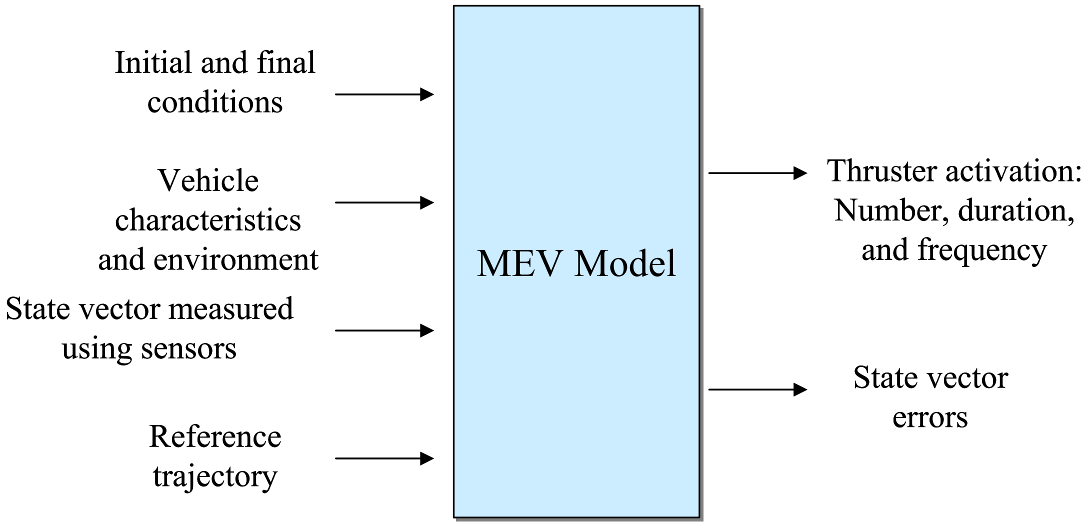


Figure 4-86: MEV CDF Model

The parameters that can be optimised are the initial flight path angle, initial azimuth, initial flight path velocity and the time duration for the entry. The purpose of the study is to select the appropriate values of those parameters. This study is also concerned to establish not only the 3DoF optimal entry trajectory but also the corresponding 6DoF full closed loop controlled trajectory.

4.4.5.2 Assumptions and trade-offs

4.4.5.2.1 MEV GNC design cyle

Figure 4-87 shows the design cycle of the GNC of the MEV:

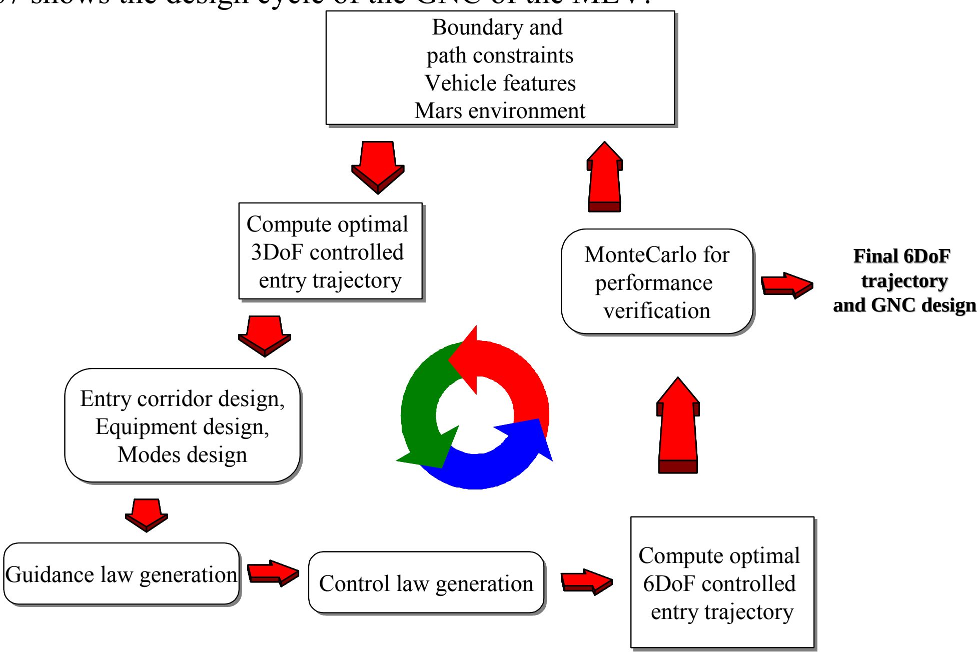


Figure 4-87: GNC design cycle

The cycle starts with the collection of all constraints. Then, an optimal 3DoF trajectory is computed. From the corridor requirements, the equipment design and the mission arc GNC modes, a guidance law is generated.
Next the control algorithms are developed to be able to follow the previously established guidance law, and finally a new 6DoF trajectory is computed in closed loop and with mathematical models of sensors and actuators.
MoteCarlo analysis are run for performance verification and validation.

Figure 4-88 shows the vehicle model coordinate and angle conventions:

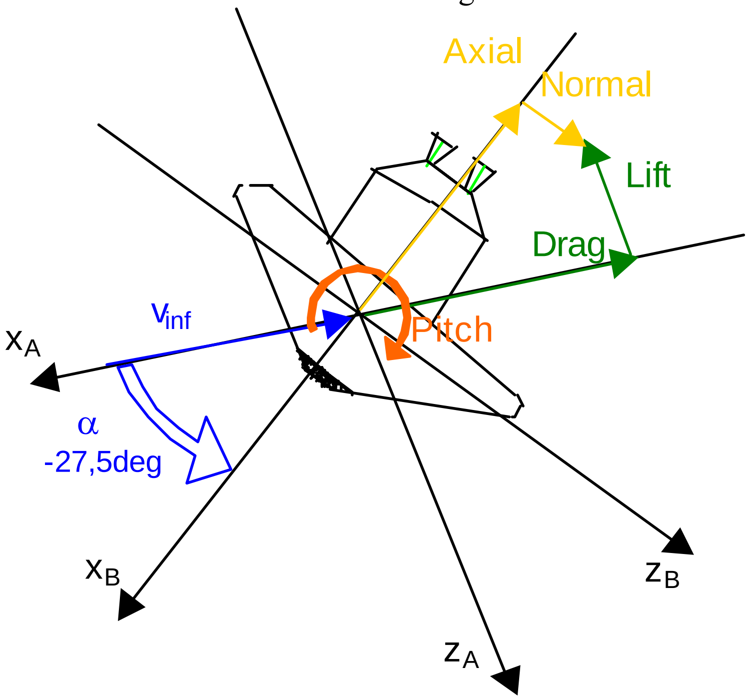


Figure 4-88: Vehicle coordinate systems

4.4.5.3 3DoF optimal trajectory

The optimal trajectory has been described in the relevant chapter 4.4.1.

4.4.5.3.1 Entry corridor design

The performance factors to take into account are controllability, stability, algorithm speed, computational loads, etc.

Predefined yellow (caution) and red tubes (warning) around the nominal path have been established to compute the controllability of the system around the pre-established optimal trajectory.

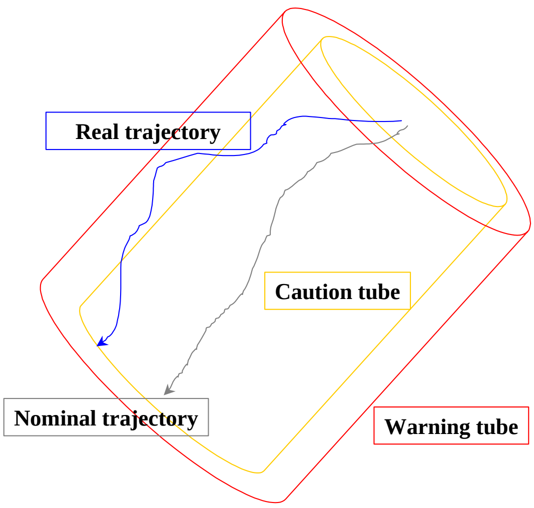


Figure 4-89: Caution and warning tubes around the nominal path

In general, the control system strategies shall be robust for the flight conditions at specific Mach numbers and dynamic pressures chosen by the control engineer along the complete flight path.

Assuming a TAEM required accuracy of 30 km, the allowable flyable corridor is a tube defined by a limit flight path angle.

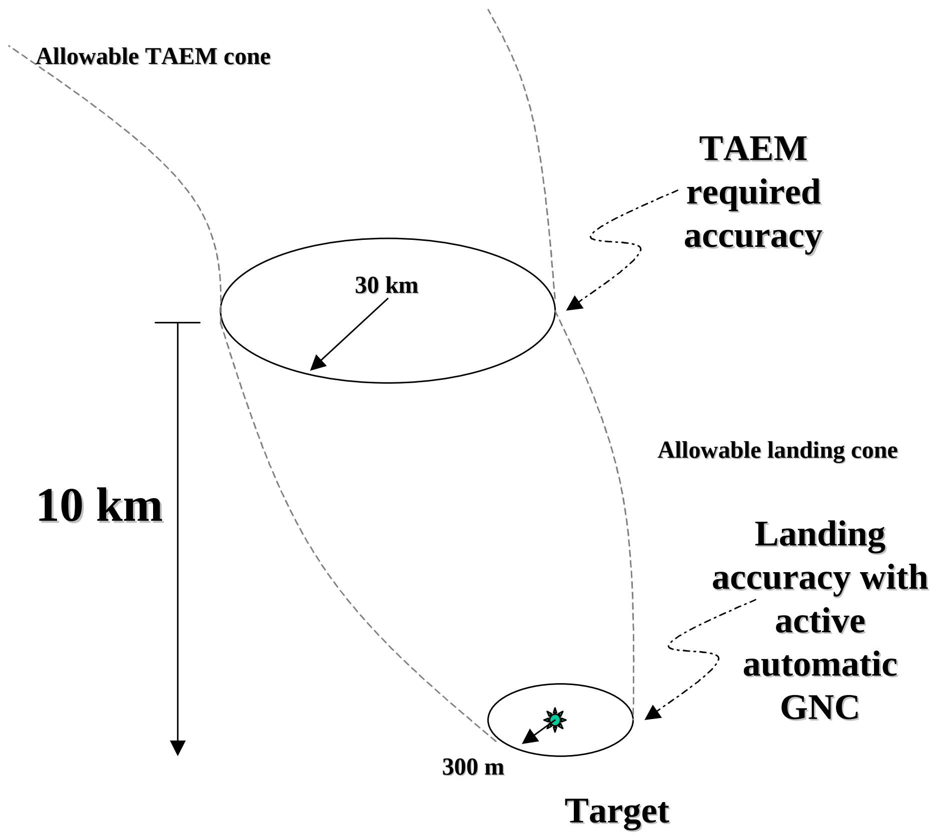


Figure 4-90: Landing cone (flyable corridor)

4.4.5.3.2 GNC equipment

The MEV will be fully three-axis stabilized control entry vehicle.
The MEV has to be able to deploy the drogue chute at a given Mach speed (Mach 2.0 is the baseline at about 10 km). This will be achieved by means of an IMU. This IMU will also be used during the powered phase. The mass and power budget can be found in the chapter 4.5.2 explaining that phase.
The time sequence for the deployment of the main chute and release of front cover will be implemented in the computer on-board. This equipment is taken into account in the Data Handling chapter 4.5.9.
Figure 4-91 shows a schematic view of the foreseen GNC equipment. There is a possibility of manual control by the crew in case of failure of the automatic system.

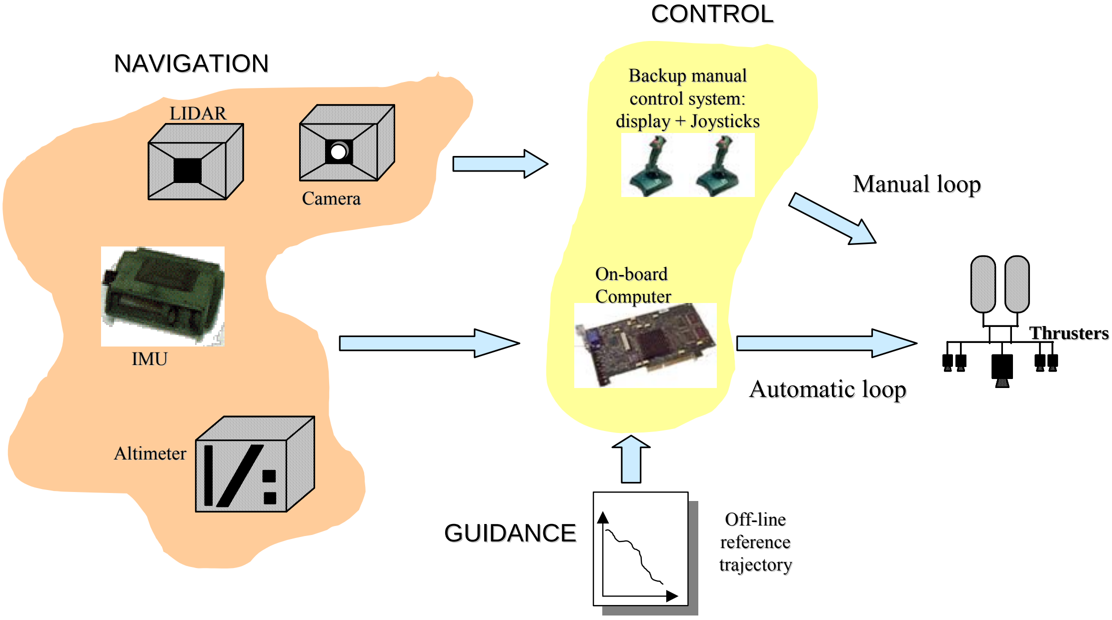


Figure 4-91: MEV GNC equipment

For stability and controllability reasons, the thrusting capability is as follows:

- The design of the reaction control system is performed based on the attitude controller requirements.
- The foreseen RCS system is composed of eight thrusters.

With 8 thrusters, it is possible to provide thrust moment of yaw, pitch and roll, according to the following scheme:

- for positive roll moment, thruster pods 1, 3, 5, and 7 are fired; no thrust force generated
- for negative roll moment, thruster pods 2, 4, 6, and 8 are fired; no thrust force generated
- for positive pitch moment, thruster pods 3, and 8 are fired; thrust force generated
- for negative pitch moment, thruster pods 4, and 7 are fired; thrust force generated
- for positive yaw moment, thruster pods 2, and 5 are fired; thrust force generated
- for negative yaw moment, thruster pods 1, and 6 are fired; thrust force generated

To generate yaw and pitch moments, forces also are generated. However, they will not have a significant impact on the point-mass motion of the spacecraft, because these forces are considerably lower than the aerodynamic forces.

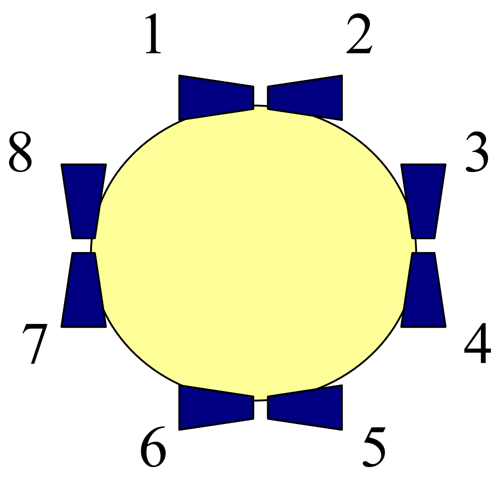


Figure 4-92: Thruster configuration

The minimum force to be produced by each thruster is 2.6 kN.

4.4.5.3.3 Entry modes design

For the EDL system, the GNC modes are cascaded as in the case of the MSR mission. Manual control is allowed during the final part of the entry, the descent and the landing phases.

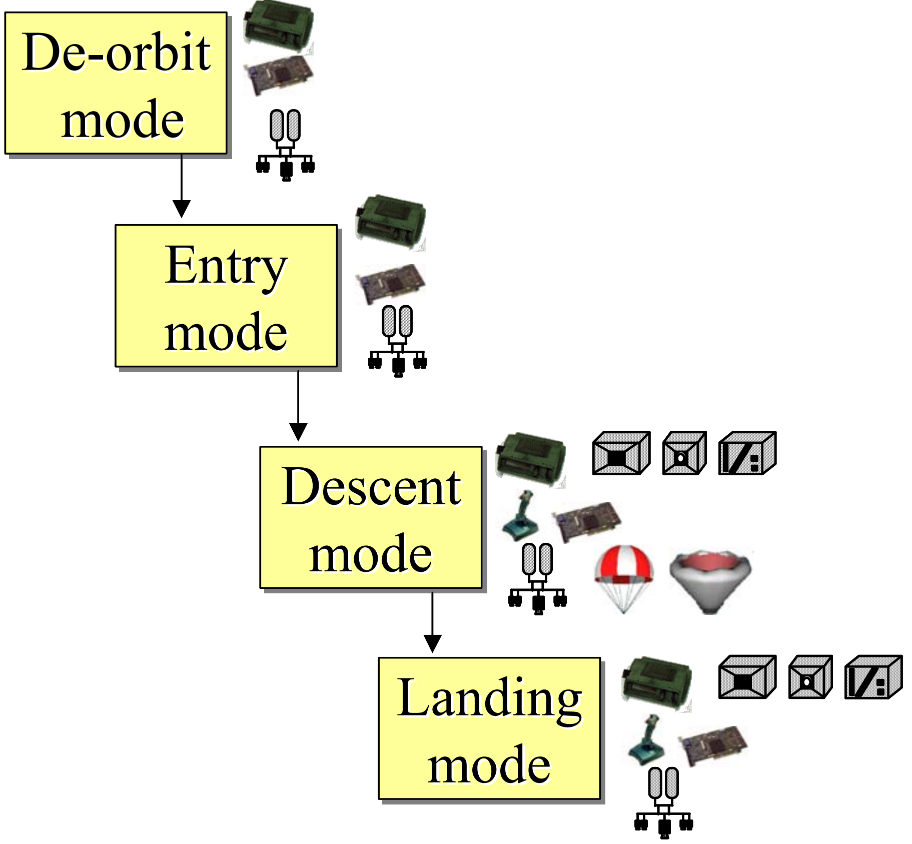


Figure 4-93: Entry modes

4.4.5.4 Control laws generation

For the control laws generation a trade-off has been made between two possible alternatives: Non-Linear Dynamic Inversion and Model-Based Predictive Control.

Non-Linear Dynamic Inversion (NLDI) control techniques uses a model of the plant and the system dynamics under control. In case of a nonlinear plant, this technique uses a two-controller level scheme design: a feedback component to linearize the dynamics and a performance enhancement component of the resulting linear system.

NLDI control technique computes a model of the dynamics of the vehicle during its flight. Then, it inverts the model to cancel all expected dynamics, and finally it inserts the desired vehicle response to the resulting plant dynamics.

Model Based Predictive Control (MBPC) involves four control elements that use a linearized model of the plant under control around a set of well pre-defined trimmed points.

The elements are as follows: a process model (a linearized system model obtained experimentally off-line), a predictor equation (a forward algorithm which will run for several steps to predict the behavior of the plant), a known future reference trajectory (previously obtained by other means and off-line), and a cost function (quadratic cost future process output error and controls).

For the NLDI solution the controller is able to handle smoothly non-linearities, coupled aerodynamics effects and other uncertainties like Earth atmospheric and gravity disturbances. By having a broad model of the plant, NLDI can cover the full flight envelope, eliminating point-per-point design gain-scheduling. In addition, NLDI can handle a variety of vehicle plants when design evolves or updates.

On the other hand, for the MBPC solution the controller is able to minimise the number of constraints when calculating the optimal trajectory and improve the failure forecasting function in the FDIR (fault detection identification and recovery) subsystem. Assuming a linearized model of the plant for a pre-defined interval of the flight, the predictor equation is based on the linearized equations of motion around this steady state flight condition.

The NLDI solution requires an acurate model of the non-linear plan (masses, moments of inertia,…), and good aerodynamic data bases for all Mach number ranges (extensive wind tunnel campaigns).

The MBPC solution requires a plant linearization on a wide rage of set points along the nominal trajectory, and on-line optimisation problem to be solved on-board inside a dedicated processor. The cost function for the quadratic optimal problem is based on a single criteria (minimum integral of the heat flux).

The final selection is done for the Non-Linear Dynamic Inversion (NLDI) shown in Figure 4-94.

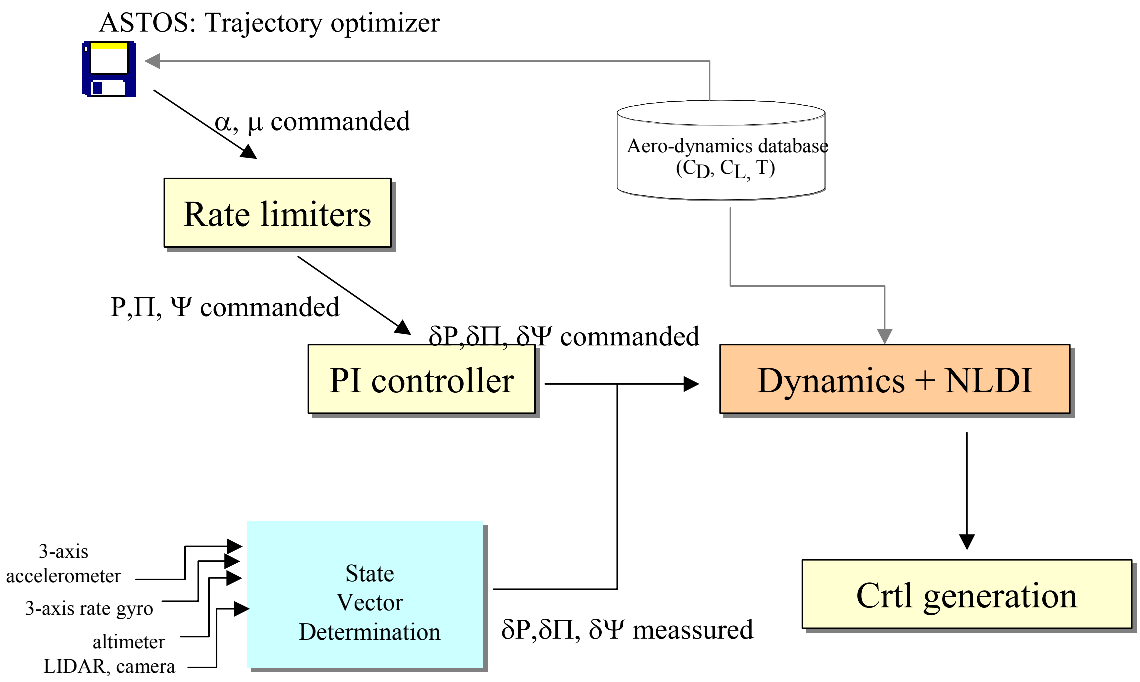


Figure 4-94: Control Law

4.4.5.5 Computation of an optimal 6DoF controlled entry trajectory

The next step in the design is to compute the 6DoF trajectory using mathematical models of sensors and actuators in the closed loop control simulation tool available at ESTEC.

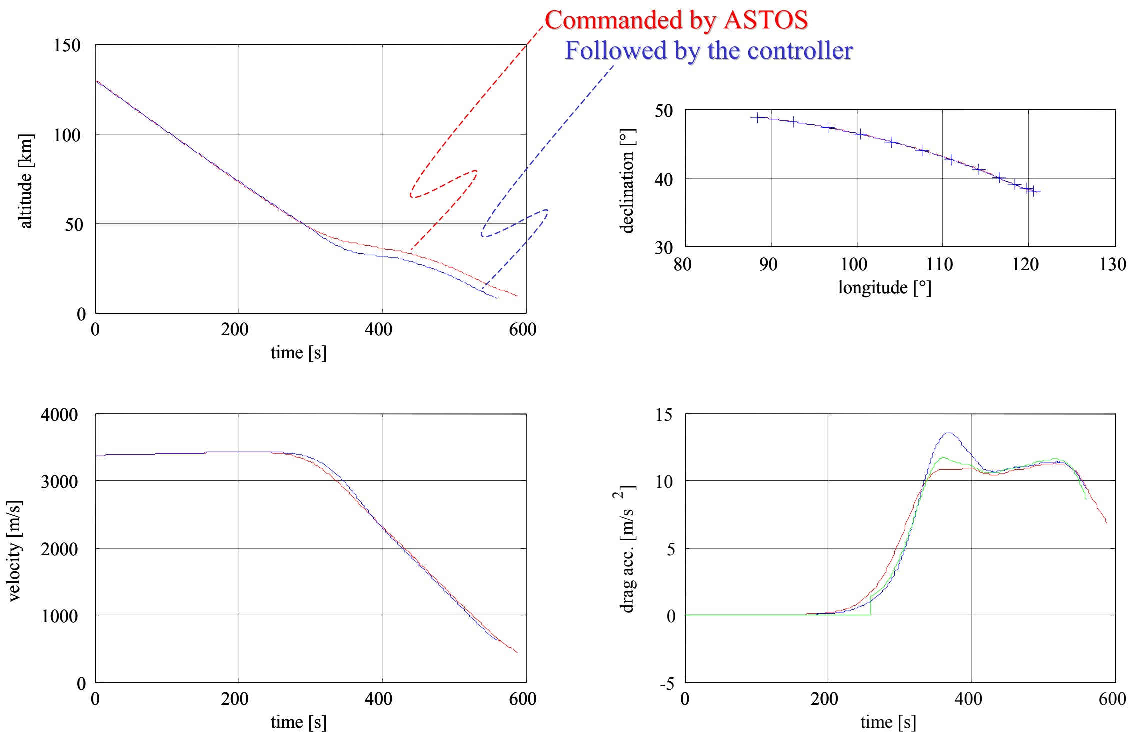
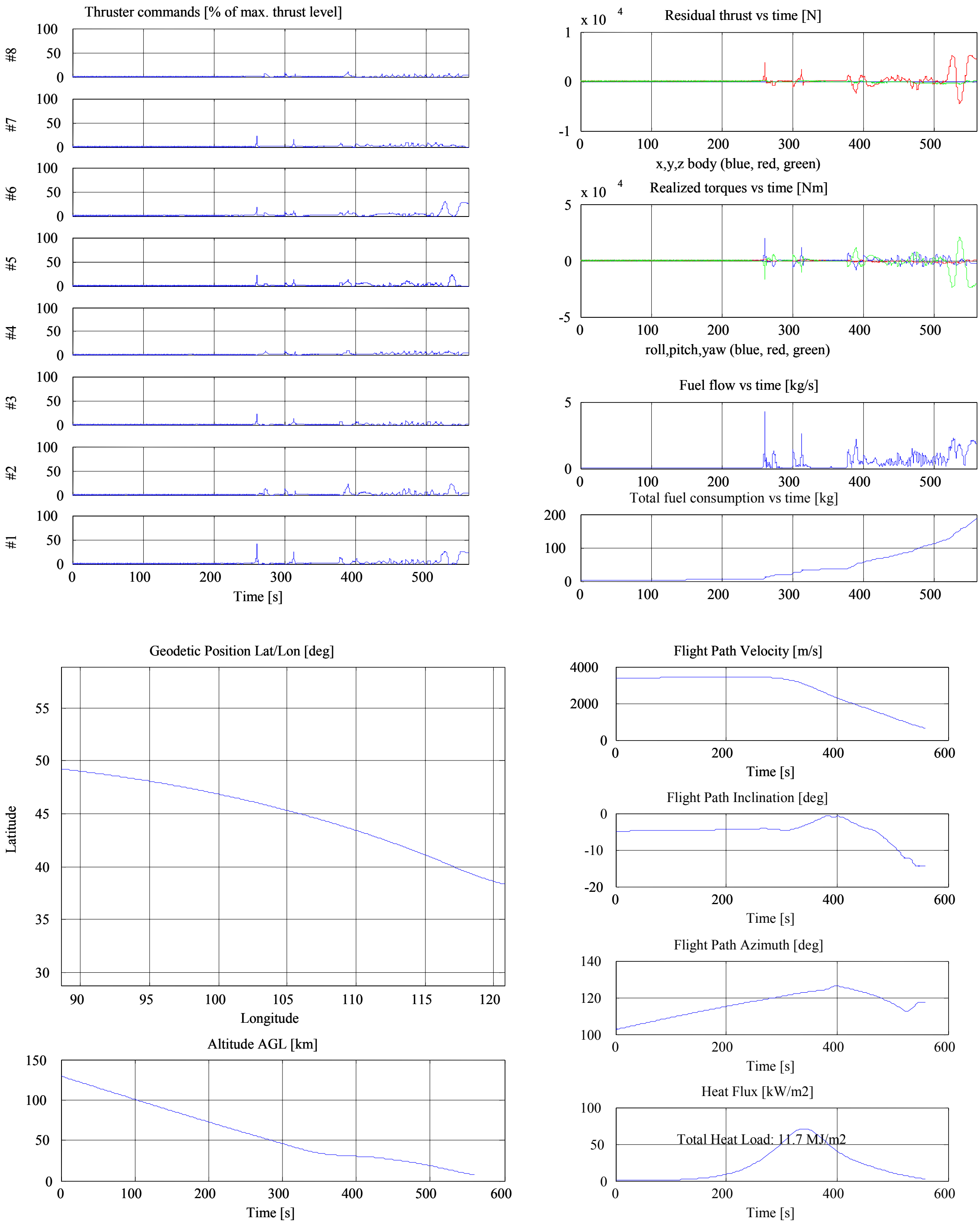


Figure 4-95: 3DoF and 6 DoF trajectories

Figure 4-95 shows two trajectories: the optimal commanded with 3DoF and the 6DoF trajectory calculated using the closed loop simulation tool.



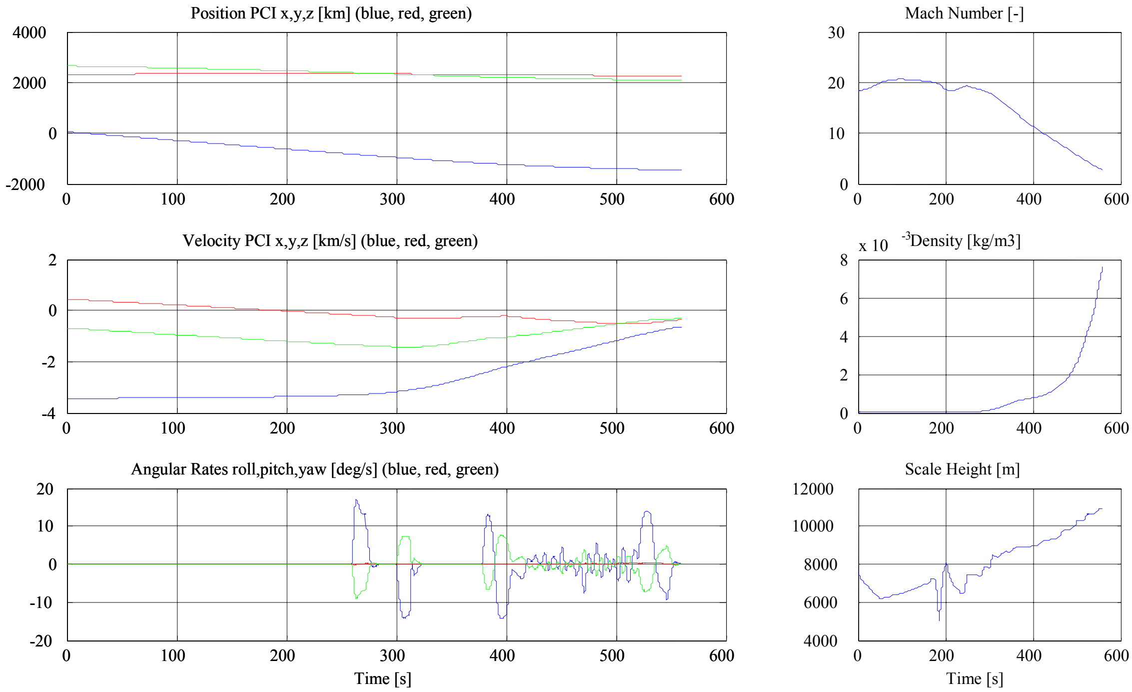


Figure 4-96: 6DoF trajectory characteristics

4.4.5.6 MonteCarlo for performance verification

The next step in the design is to run a MonteCarlo campaign for performance verification. For this step, the following parameters were scanned during the simulations:

- Uncertainty on Mass properties:
  - mass_delta = 1 % [%] Uncertainty of the nominal mass
  - CoG_x_delta = 1 % [%] Uncertainty of CoG position in X direction
  - CoG_y_delta = 1 % [%] Uncertainty of CoG position in Y direction
  - CoG_z_delta = 1 % [%] Uncertainty of CoG position in Z direction
- Uncertainty on Initial state
  - pos_alt_delta = 0.0 % [m] Uncertainty in initial altitude
  - pos_LL = 0.03 % [degrees] Uncertainty in latitude/longitude
  - vel_speed_delta = 0.39 % [m/s] Uncertainty in speed
  - FPA = 0.129 % [degrees] Uncertainty in flight path angle and azimuth
- Uncertainty on Aerodynamic model
  - total_lift = 10 %
  - total_drag = 10 %
  - aerotorque = 10 %
- Uncertainty on Atmospheric properties
  - Atmos_P_D_T = 5 %
- Uncertainty on Thrust level
  - Isp_direction = 0.5 %
- Uncertainty on Sensors
  - IMU = 0.5 %
  - RA = 10 %

Finally, the Figure 4-97 shown the MonteCarlo runs.

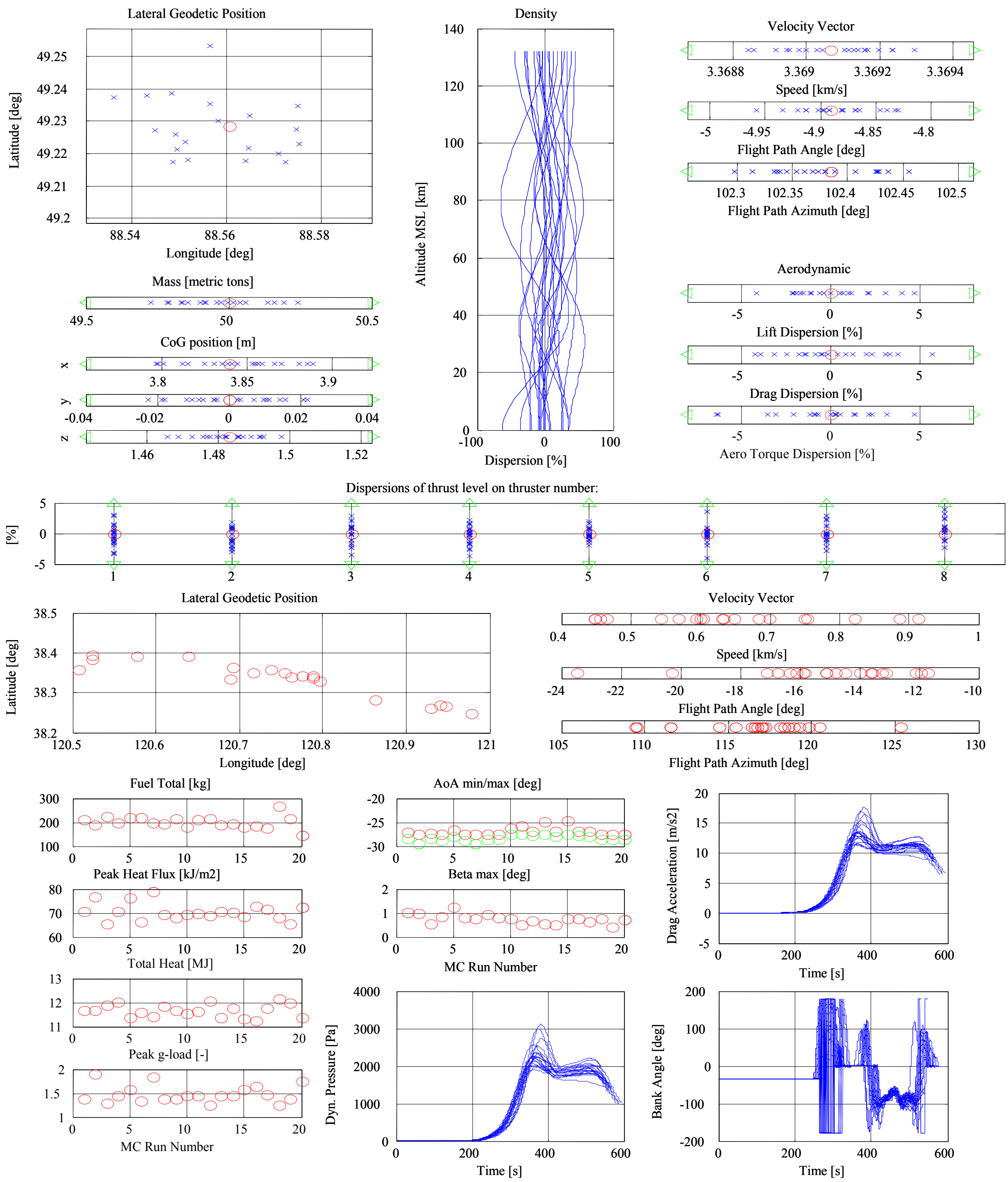


Figure 4-97: Monte Carlo runs

4.4.6 Mechanisms

4.4.6.1 Requirements and design drivers

The HMM science requirements do not state any specific requirements applicable to the DM mechanisms. As a result of the DM's configuration, the following necessary mechanisms and their requirements can be derived:

- Vehicle Stage Separation System
  - Release & Separation of De-orbit Propulsion Module.
- Separation System
  - Heat Shield Jettison System.
- Deployable Landing Leg System

4.4.6.2 Assumptions and trade-offs

The requirement for a deployable landing leg system is dependent upon the dynamics of the Landing Vehicle. The Footprint dimension is sized to prevent the vehicle from toppling when contact with the ground happens and there is a residual horizontal velocity component acting on the system.
Essentially, this can de (simplistically) analysed by looking at the situation when the horizontal and vertical force moment components are in equilibrium i.e. the point of initiating the topple moment when the horizontal moment component is greater than the vertical moment component.

Figure 4-98 shows the principle force or moment balance system:

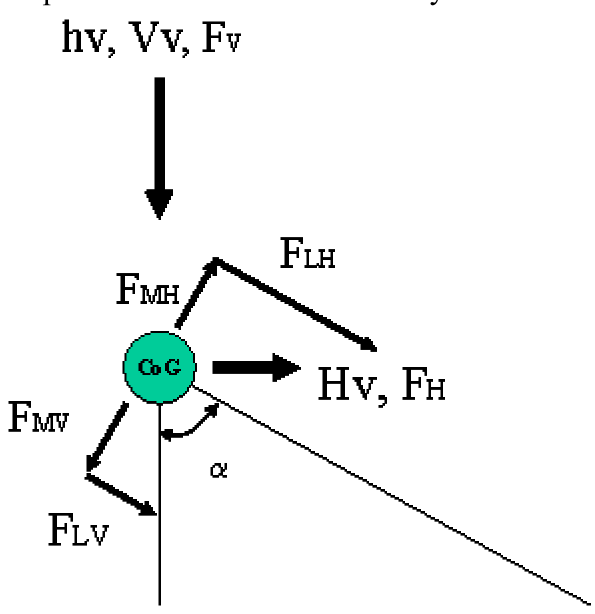


Figure 4-98: Momentum balance system

Hv = Height at which the engine thrust is cut-off.

- Max. Height for 'Thrust' cut-off = 2 m max.

Vv = The residual Vertical Velocity at the moment of engine thrust cut-off

- Final Vertical Velocity= 0→2 m/s, g=3.8 m/s2)

Hv = The maximum residual horizontal velocity at contact with the surface.

- Max. Horizontal Velocity 1 m/s

Additional Data:

- Lander Mass = 42000 Kg.
- Vehicle CoG Height about 6 m.
- Mars Gravity constant = 3.8 m/s2.

Additional assumptions:

- Assume deceleration time the same for both velocity components= 0.5 secs.
- A margin or 2.5° added to α.
- This approach excludes any Lander attitude errors or surface terrain effects.
- Damping in the leg system is not taken in to account other than applying the deceleration time.

Stability or no toppling is assumed when α is chosen such that F_{MV}=F_{MH} i.e induced moments are equal.

The following remarks can be made regarding this analysis approach:

- The problem is handled as a quasi-static problem.
- As formulated above, the analysis excludes the changing velocity vector due to the rotation over or about the foot or feet.
- Any induced rotation will be experienced as 'damping' for the toppling motion due to the Gravity vector- this can be treated as a (small) 'margin'

The analysis has been performed according to the following steps:

1. Calculate the velocity at surface impact at minimum and maximum residual velocities using v2= u2+ 2.a.s where 'u' is the residual velocity, ''is the Martian gravitational constant and 's' is the height at which the engine thrust is cut.
2. Calculate the Vertical Force component (Fv) due to the deceleration [mass * (Fv/0.5 secs)].
3. Calculate the Horizontal Force component (Fh) for 3 horizontal velocities (Hv = 0.5 m/s, 1 m/s & 1.5 m/s) [mass * (Hv/0.5)].
4. Calculate the angle α where Fmv = Fmh [α = atan(Hv/Fh) + 2.5° margin] for minimum and maximum vertical velocities.
5. Calculate the minimum footprint area required based upon the estimated CoG height.
6. Compare the required minimum footprint dimension against the available envelope dimension.

The above model has been reshown in an 'excel' spreadsheet. The following graph represents the output from the analysis.

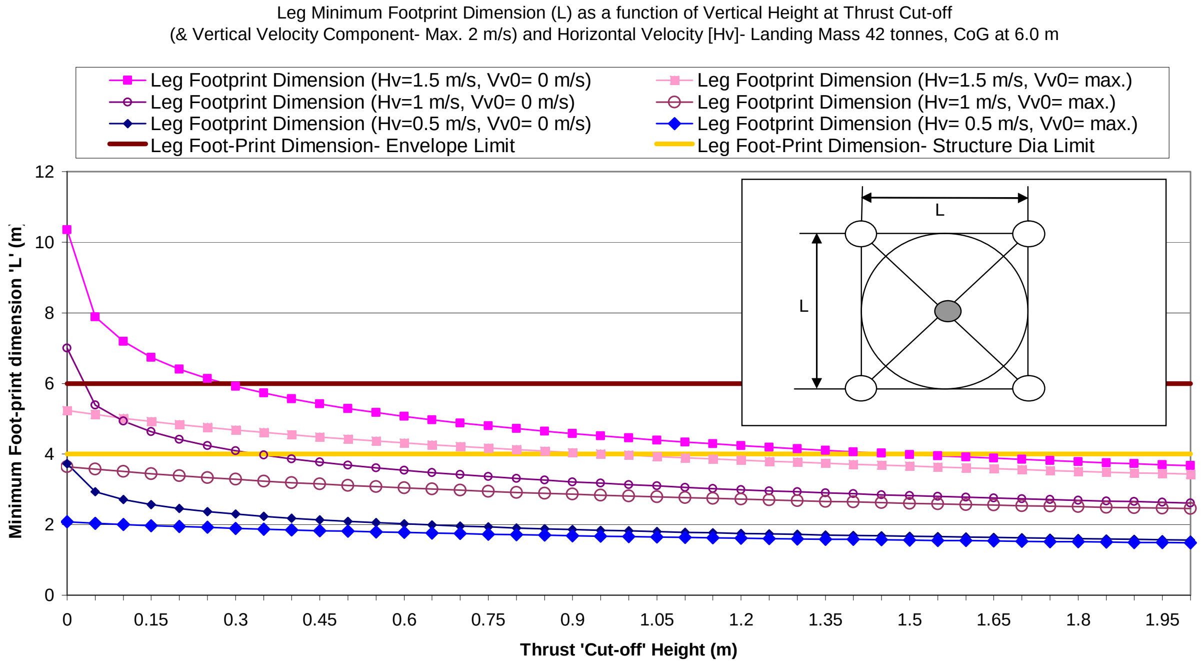


Figure 4-99: Minimum footprint vs. vertical height

The main structure of the SHM is about 4 m in diameter and the maximum allowable envelope dimension is about 6 m in diameter.
From the graph it can be seen that for a horizontal velocity component up to 1 m/s, a foot print dimension within about 6 m in diameter can be realised for all vertical velocity cases and therefore a deployable landing foot is not required. What can also be observed is that a residual vertical velocity is beneficial to the landing as this component acts against the tendency to topple and leads to a smaller footprint dimension.
For the higher horizontal velocity of 1.5 m/s, it can be seen that, to realise a footprint dimension within the enveloping limit, an additional (residual) vertical velocity component is required (either residual velocity due to the descent of the engine thrust cut-off above about 0.5 m) i.e. a near zero residual height and vertical velocity at engine thrust cut-off with a horizontal velocity of 1.5 m/s will lead to the system toppling.

In conclusion, if the a limit to the residual horizontal velocity is set to 1 m/s (requirement on the control system), a non-deployable or static landing leg system can be realised within the enveloping dimension of about 6 m in diameter that can remain stable when subject to the residual vertical velocity and height control dispertions. This shall be assumed to be the baseline.

Additional conclusions from the analysis performed are;

- A Large Mass aids stability as does a residual vertical velocity and height component..
- The landing system leg has to be designed assuming a single leg contact as worst case.
- Leg Loading of the order of 350 000 N (estimate of the load along the angular vector calculated above).
    - o This load is mainly influenced by the vertical Force components (Mass, Cut-off Height, Residual Vertical Velocity).

It should also be noted, that the landing feet minimum footprint dimension is (largely) independent of the Lander's mass as the mass term is present in both the Vertical and Horizontal resultant force components. The major parameter affecting the footprint dimension is the CoG or CoM position. The footprint dimension is a direct ration of the CoM height, which should be kept as low as possible to maintain the required footprint within the envelope dimension of ∅6 m.

4.4.6.3 Baseline design

4.4.6.3.1 Vehicle separation

Due to the potentially large diameter of the MAV to Propulsion module, the separation of the MAV from the propulsion unit prior to entry and descent launch, shall be realised with a pyrotechnically cut bolts at up to four locations around the I/F.

4.4.6.3.2 Heat shield jettison

Due to the four point mounting of the heat-shield on to the four landing system feet, the jettison shall be realised with a pyrotechnically cut bolts at the four foot locations.

4.4.6.4 Budgets

Element 1: Descent Module			MASS [kg]			
Unit	Element 1 Unit Name Click on button below to **insert new unit**	Quantity	Mass per quantity excl. margin	Maturity Level	Margin	Total Mass incl. margin
1	Pyro Release system- De-orbit	4	5.0	To be modified	10	22.0
2	Separation Spring Units- De-orbit	4	5.0	To be modified	10	22.0
3	Pyro Release System- Heat-shield	4	5.0	To be developed	20	24.0
4				To be developed	20	0.0
5				To be developed	20	0.0
-	Click on button below to insert new unit		0.0	To be developed	20	0.0
ELEMENT 1 SUBSYSTEM TOTAL		**3**	**60.0**		**13.3**	**68.0**

Table 4-44: DM Mass Budget

4.4.7 Parachute design

4.4.7.1 Requirements and design drivers

The parachutes for the MEV Descent Module provide a means of decelerating the vehicle from the high velocity reached at the end of the guided entry to a velocity that can be handed by the system of landing rockets. The design is driven by the velocity and altitude requirements at the beginning and end of the parachute descent phase. These are summarised below.

Parameter	Value
Initial Mach number	2
Initial altitude	10 km
Final velocity	100 m/s
Final altitude (above surface)	2 km

Table 4-45: Descent Requirements

The size (area) of each parachute is limited by the restrictions of manufacturing, packing and deployment. Using present technology, a parachute area of around 1000 m^2 (diameter of about 36 m) is considered to be a reasonable upper limit.

4.4.7.2 Assumptions and trade-offs

For this application disk-gap-band type parachutes are used. Of the options available, they provide the best area to mass ratio and have a sufficiently high Mach number application. Also, this type of parachute was used for the Mars Viking Lander so the Mars landing application has been successfully demonstrated (although not at the scale required here).

Initially it was intended to have a drogue chute in addition to the main parachute(s). However, it was found that with the relatively small deceleration achieved with the drogue chute, the vehicle would continue to descend at a rate that would put it at too low an altitude for main parachute deployment. Therefore, the use of a drogue was rejected.

It is assumed the MEV heat shield is jettisoned before the initiation of the parachute descent phase.

4.4.7.3 Baseline design

Given the initial and final altitude and velocity constraints and the properties of the disk-gap-band parachute, sizes and masses for the parachute system can be obtained. The total nominal parachute area required to obtain the necessary deceleration is 4384 m^2. This is divided among four parachutes to be closer to the maximum area per parachute constraint. The total design therefore consists of 4 main parachutes, each with a nominal area of 1096 m^2 and a nominal diameter of 37.4 m.

An additional, backup parachute of the same design as the main parachutes is also included in the design. This parachute will be deployed in the event that one of the main parachutes fails.

The descent of the vehicle from parachute opening to rocket firing is reshown in the following figures that show the altitude and velocity versus time. The terminal velocity of 100 m/s is not quite reached at the altitude of 2 km, but it is sufficiently close to be acceptable for the rocket system.

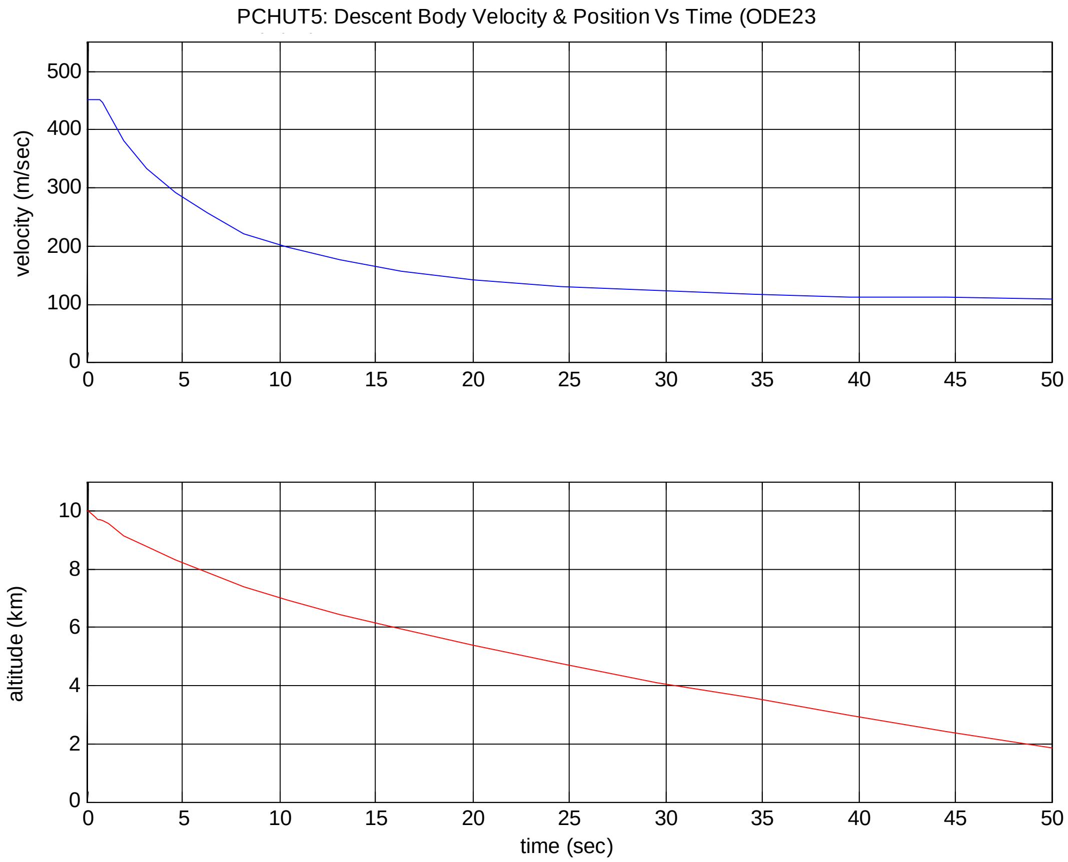


Figure 4-100: Velocity and altitude of MEV from parachute opening to landing rocket firing

4.4.7.4 Budgets

The mass of each parachute can be estimated taking into account the canopy, lines, deployment equipment, swivel, bridle, and pilot chute. The masses are based on previous built parachutes and components for applications that differ significantly from the present one, so there is a great deal of uncertainty in the mass estimate. The mass for each parachute is estimated to be about 103 kg, giving a total system mass of 515 kg. The maximum system mass margin of 20% is used to account for uncertainties.

4.4.7.5 Options

Within the framework of the current design, one option is to have fewer, larger parachutes. This would be preferable from the point of view of reducing the complexity and number of potential failure points in the system. However, further work would be required to determine if larger parachutes are feasible.

To reduce (or possibly even eliminate) the deceleration requirements for the parachutes, the heat shield of the MEV could also be used for braking from 10 km to 2 km, rather than jettisoning it as is presently done. However, having both options available as now is beneficial from the point of view of redundancy.

4.4.8 Propulsion

4.4.8.1 Requirements and design drivers

According to mission analysis the total DV required for the deorbiting manoeuvre is estimated to be 98 m/s.
Thrust required for this manoeuvre is assumed of 20 kN. This value has been selected by similarity on the basis of the thrust to mass ratio of the Soyuz module.

4.4.8.2 Assumptions and trade-offs

Module dry mass is estimated 45 tonnes

Only Storable bi-propellant are considered.

4.4.8.3 Baseline design

Four YUZHNOYE RD 869 pump-fed thruster have been chosen as propulsion system for this module. The thruster is under development for 4th stage of the European VEGA Launcher. The propulsion system presents the following characteristics.

Characteristic	Value
Number of thruster	4
Thrust	5 kN (pump- fed)
Isp	325 sec
exit diameter	325 mm
length	600 mm
thruster mass	34 kg
propellant	UDMH/NTO
O/F ratio	2.1
number of tanks	2+2
Tanks material	Ti
max MEOP	7 bar
Mass of UDMH tank	3 kg (each)
Mass of NTO tank	2.5 kg (each)

Table 4-46: De-orbit propulsion system summary

4.4.8.4 Budgets

Propellant mass 1463 kg

Propulsion Dry mass (including margins) 253 kg

This mass includes an estimation of thrusters mass, the tanks, and a roughly estimation of feedlines, valves and regulators, propulsion thermal control, avionics, actuators and does not consider the structure of the propulsion system, power and communication.

4.5 Mars Ascent Vehicle

4.5.1 Trajectories

4.5.1.1 Requirements and design drivers

The objective is to obtain an ascent trajectory which minimises the lift off mass, by means of selecting the proper propellant masses in each stage, firing time in each stage, pitch and yaw profile.

4.5.1.2 Baseline trajectory

4.5.1.2.1 Input data

Initial conditions
The optimal trajectory depends on both the altitude and latitude of the launch pad. The landing site drives the latitude of the launch site, and it was agreed to be 20 degrees North as reference. The reference altitude was assumed to be 0 Km, to be conservative. The longitude was assumed to be equal to 0, since it has no influence in the trajectory computation.

Final conditions
The final conditions are those corresponding to a circular orbit of 500 km altitude and 32 degrees of inclination.

Mass budget
Table 4-47 shows the mass budget used for the baseline trajectory. During the optimisation process the mass of the tanks was considered as a variable, being equal to 3% of the propellant mass.

1st Stage				2nd Stage				
Dry Mass (except Tanks)	Tanks	Total Dry Mass	Prop	Dry Mass (except Tanks)	Tanks	Total Dry Mass	Prop	Lift Off Mass
700	388	1088	12922	5100	81	5181	2705	21896

Table 4-47: Mass budget in Kg

Propulsion system
Table 4-48 shows the performance data of the propulsion system and the final propulsion configuration.

	Type of engine	Number of engines	Thrust (N)	Isp(s)	Nozzle Diameter (mm)
1st Stage	Aestus (advance)	4	33000	330	1070
2nd Stage	RD 869	4	5000	325	375

Table 4-48: Propulsion system

Drag coefficients and reference area

Figure 4-101 shows the drag coefficient data. The drag coefficient was defined by linear interpolation from a table of drag coefficients as function of Mach number. To make the model simple, CL was supposed to 0 for any Mach number.

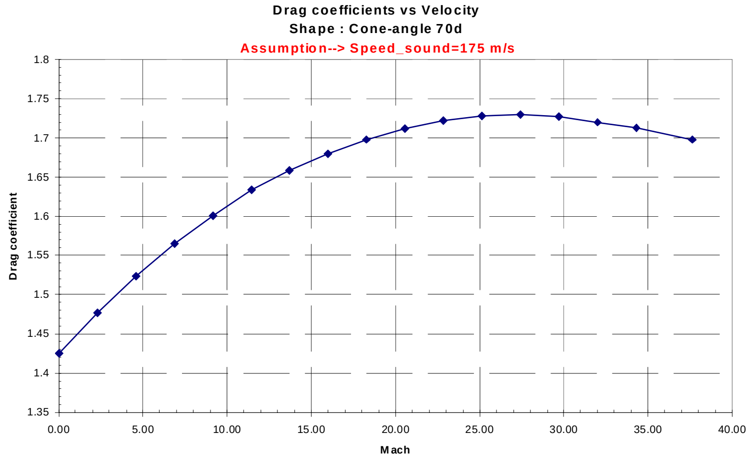


Figure 4-101: Cd Data

The reference area used for computing the aerodynamic force is 15.9 m^2, equivalent to a circle of 4.5 m of diameter.

Heat Flux Coefficients

The formula and coefficients used for the heat flux computation are the following:

$$\text{Heat flux} = C \times (\text{density})^N \times (\text{Velocity})^M \ \text{W/m}^2$$

Components	Value	Unit
C	1.5588 e-4	$W/m2 /((kg/m3)^N/(m/s)^M)$
N	0.5	ND
M	3.04	ND

Table 4-49: Reference data to calculate heat flux

Atmosphere Model

The atmosphere data used in this study is the Mars-GRAM 2001. Winds profiles were not taken into account. The density, temperature and sonic velocity profiles correspond to latitude of 15 degrees. This is conservative, since the Mars-GRAM 2001 provides higher density profiles for higher latitudes.

Planetary Model of Mars

The Table 4-50 shows the planetary model of the Mars. The gravitational acceleration was derived from the gravitational potential expressed in spherical harmonics. The main term and the $C_{2,0}$ (J2) terms were taken into account, as well as the rotation of Mars.

Term	Value	Unit
Radius of equator	3393.94	Km
Radius of polar	3376.78	Km
Gravity constant	4.28228E+13	M^3/s^2
Rotational rate	7.088218E-5	Rad/s
C2,0 (J2)	-8.75977E-4	ND

Table 4-50: Mars Reference Ellipsoid Parameter

4.5.1.2.2 Trajectory strategy and results

The most efficient way to reach the orbit is shown in Figure 4-102. The objective is to minimise the lift-off mass. The sequence is as follows:

1) 1st stage engine burn out (depletion)
2) 2nd stage engine first burn
3) Coastal arc
4) 2nd stage engine second burn, injecting the MAV into the target orbitThe coastal arc is flown along an orbit 100km x 500 km. Nevertheless, the minimum lift-off mass is achieved if that transfer orbit has a negative altitude of the perigee. This option was discarded due to safety reasons: in case of a failure in the restart of the 2nd stage, if the altitude of the perigee were negative the MAV would crash onto the surface of Mars.

10 seconds delay between the separation of the 1st stage and the ignition of the 2nd stage were assumed.

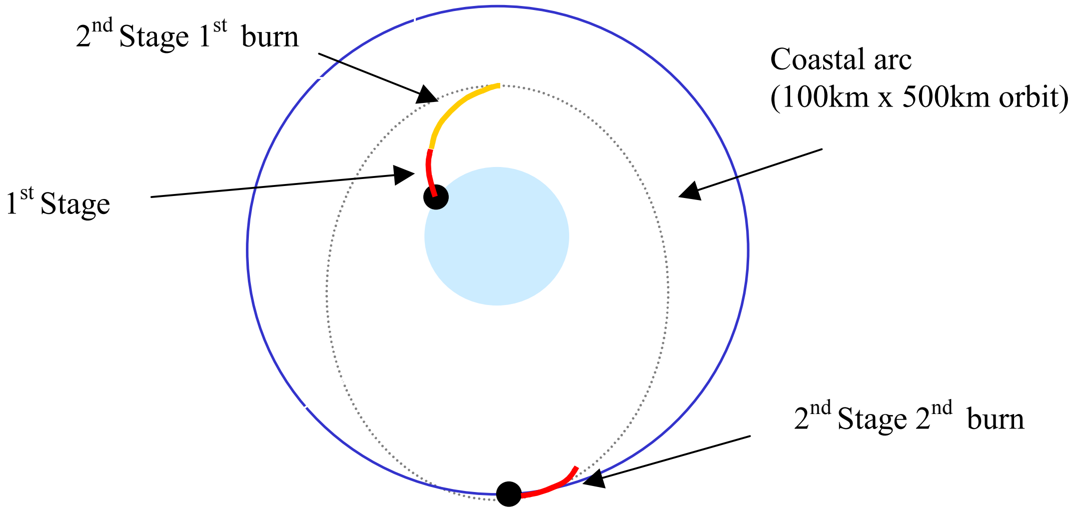


Figure 4-102: Baseline trajectory strategy

With this type of trajectory and with the input data shown above, the minimum lift-off mass is 21896 kg (Table 4-47). A detailed sequence of events can be seen in Table 4-51, and some plots with the most relevant parameters of the trajectory are shown in Figure 4-103.

Time (s)	Altitude (km)	Event
0.0	0.000	Vertical Take off
3.0	0.010	Kick turn manoeuvre
6.3	0.045	Starts Gravity Turn (Flight with aoa = 0 degrees)
316.8	74.223	1st Stage Burn Out. End of gravity turn
321.8	75.505	1st Stage Jettisoning
326.8	76.740	2nd Stage First Burn
732.4	98.960	Starts Coast Arc
4199.8	500.219	2nd Stage First Burn
4224.3	500.195	Final Orbit

Table 4-51: MAV ascent trajectory. Sequence of events

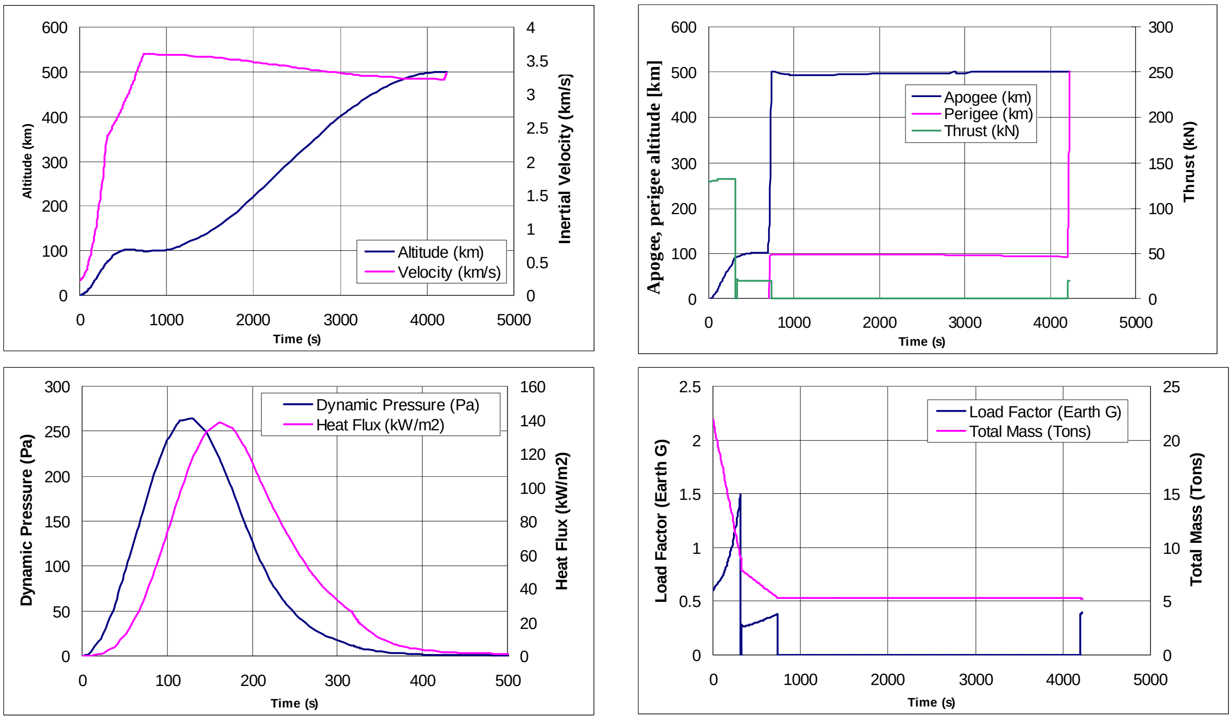


Figure 4-103: Baseline trajectory and flight environment

4.5.2 GNC and rendezvous & docking

From a GNC point of view, the ascent vehicle will cover two mission arcs:

- Ascent from the surface of Mars
- Rendezvous and docking in Mars orbit with the orbiter

Both phases have an impact in the design of the GNC subsystem. This part of the report is devoted to the rendezvous phase only.

4.5.2.1 Requirements and design drivers

The following requirements apply to the design of the GNC subsystem to fulfil the Rendezvous and docking in Mars orbit:

- To execute a safe ascent of astronauts from the Martian soil to a parking orbit
- To be able to detect, follow, and dock with the orbiter
- To be the active chaser approaching a cooperative passive target (The term "passive target" is meant from the actuation - position, attitude - point of view. Since the target – MAV – will carry a beacon and be active from a RF point of view, the term "non-cooperative" target is used instead.)
- The probability of collision between target and chaser shall be less than 0.001% over a 2 Earth days period
- To be able to establish the convenient selection of the MAV launch window
- To allow the possibility of re-trial the rendezvous in case of fail for up to 3 times
- To establish and perform the absorption of launch dispersions
- To be able to allow manual rendezvous overriding the automatic capability of the MAV.
- To be able to maintain at all times a three-axis stabilization
- To be able to accomplish all manoeuvres in less that 4 days (life support limit).
- To use as trajectory criteria the safety of the astronauts.
- To use as trajectory criteria the total rendezvous time (minimise the total time)
- To use as trajectory criteria the fuel consumption (minimum fuel consumption).
- Several sensors to be used are a radio frequency (RF) system, camera, and a LIDAR. For the very far range a radio-frequency beacon (distances between 4000 km and 5 km) would be used.
- The orbiter shall make use of data fusion between both sensors on the estimation process to improve navigation accuracy
- The sensors camera and LIDAR shall be mounted fixed to the platform of the orbiter.
- The Camera shall be axially aligned with its docking pattern in the target during normal operations.
- The accommodation of the LIDAR on the platform shall take into account the potential dissymmetrical scanning capability of the instrument to enable tracking of the target as long as possible (the origin of this remark is due to the fact that some LIDAR have dissymmetrical scanning capabilities for example: Horizontal FOV: +/- 170 degrees, Vertical FOV: +/- 40).
- During the terminal phase of the experiment, the orbiter shall be able to impart a velocity change manoeuvre of less than 1 m/s in any arbitrary direction without re-orienting its attitude

The rendezvous mission arc should rely on:

- High-thrust chemical propulsion
- A fixed orbital altitude
- A maximum total maneuvering time
- A maximum total ΔV
- High accuracy sensing technology and high-precision actuation techniques

4.5.2.2 Constraints

Figure 4-104 shows a pictorial representation of the balance between all the constraints mentioned and in particular the contradictory one of safety, ΔV consumption and time used for the rendezvous.

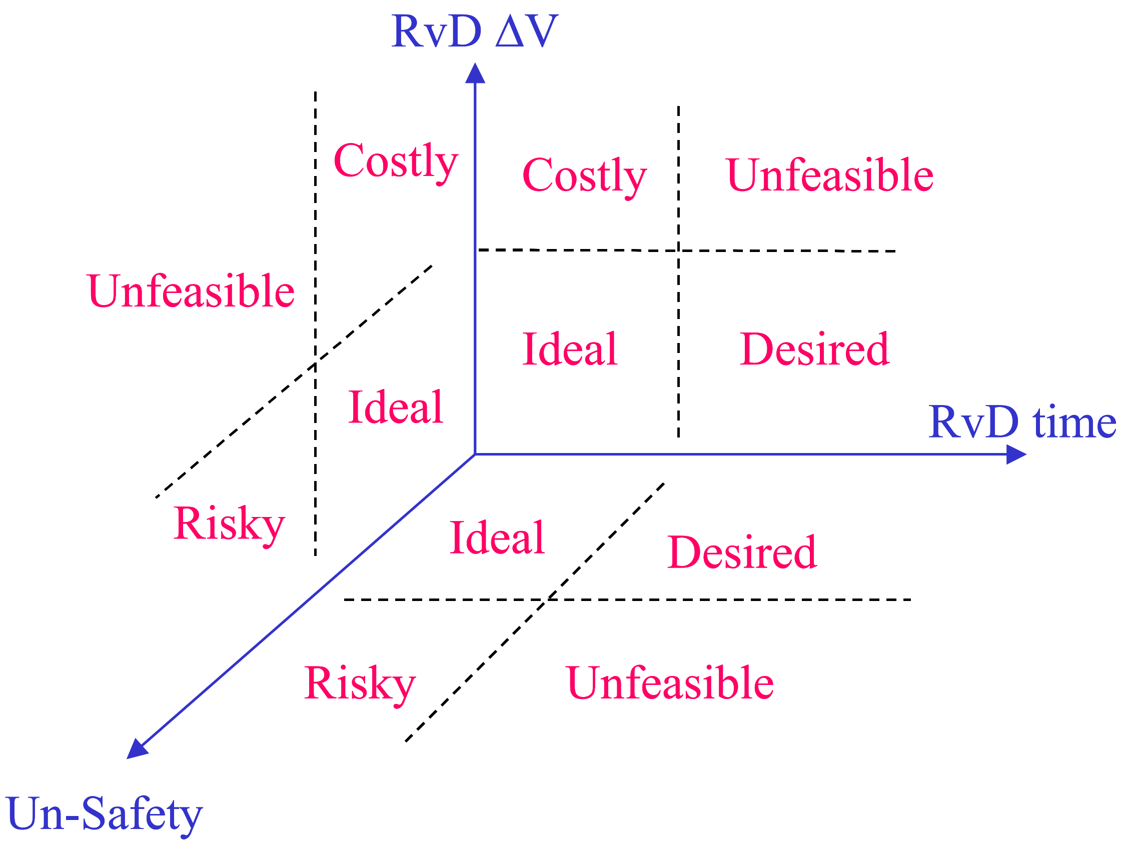


Figure 4-104: R & D constraints

To be able to elaborate a rendezvous strategy for the HMM, the past experience was reviewed. In general, there are similarities and differences of the lunar rendezvous with Mars rendezvous: they have the same purpose and target and they have the same physics and principles.

However, they take place in different planetary scenarios and they use different sensors and actuators.

The main conclusion drawn here is that the lunar rendezvous experience is not directly applicable to our problem. In particular, there is a very long distance component in all maneuvers established that leads to a time lag between Ground Segment and Astronauts that need to be properly addressed and accounted for.

4.5.2.3 Assumptions and trade-offs

The two mission arcs of ascent and rendezvous are “apparently” independent. However, for humans missions the ascent arc is much connected to the rendezvous one; In fact, the ascent and rendezvous arcs are in some strategies and trade-offs the “same” arc.

The first trade-off studied is the selection of roles and responsibilities between the ascent vehicle MAV and the orbiter.
There are two main options:

- MAV passive target, orbiter active chaser. This option leads to a slow RvD with a low MAV precision injection. It has a medium to low autonomy and the ascent crew has a strong dependency from orbiter crew
  - MAV active chaser, orbiter passive target: This option leads to a fast RvD. It requires a high MAV precision injection. It leads to a high degrees of autonomy.

For safety reasons, the second option was selected. Hence, the MAV is the active chaser, and the orbiter is the passive target. However, both vehicles will remain three-axis stabilized during all maneuvres.
The second trade-off is about the establishment of the rendezvous manoeuvring strategy. There are in essence three possible strategies: direct rendezvous, long rendezvous and short rendezvous. These are as shown in Figure 4-105.

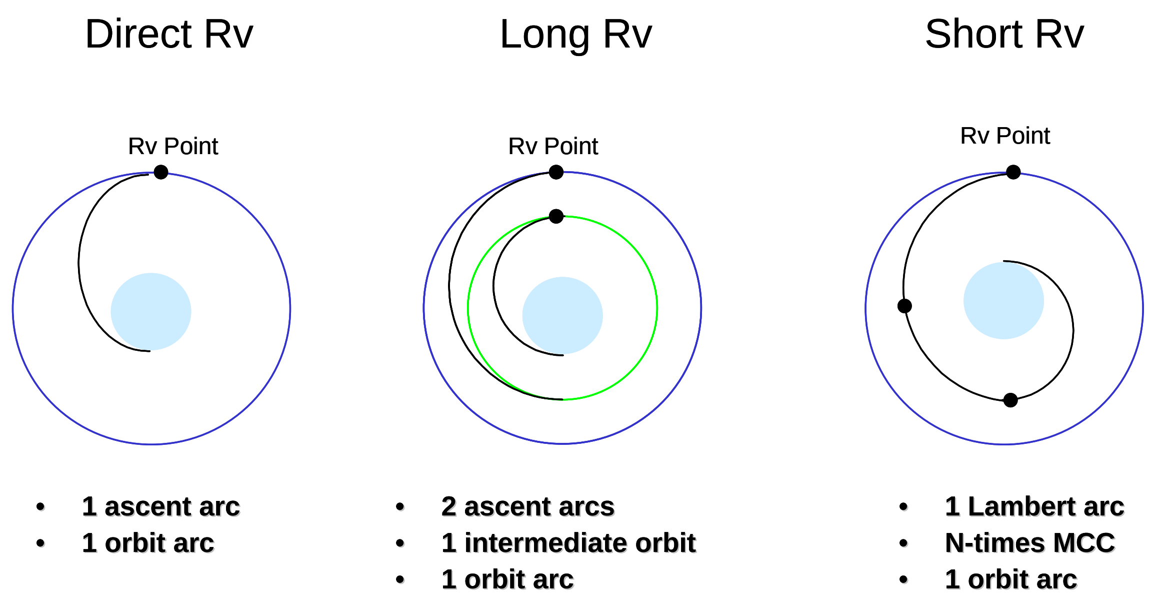


Figure 4-105: Rendezvous strategies

The Figure 4-106 and Figure 4-107 provide a list of the advantages and disadvantages of each of these strategies.

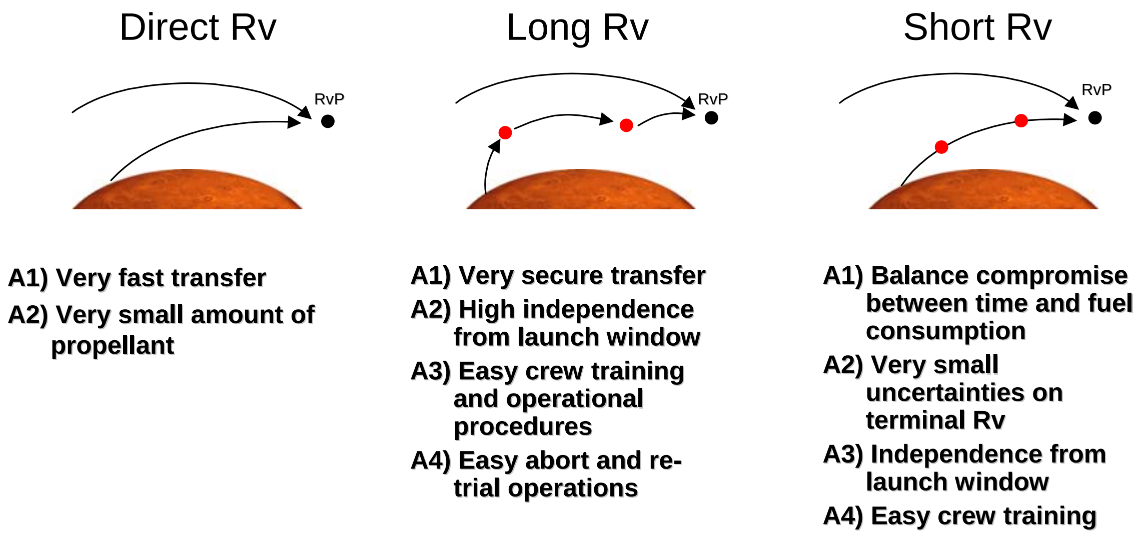


Figure 4-106: Rendezvous strategies advantages

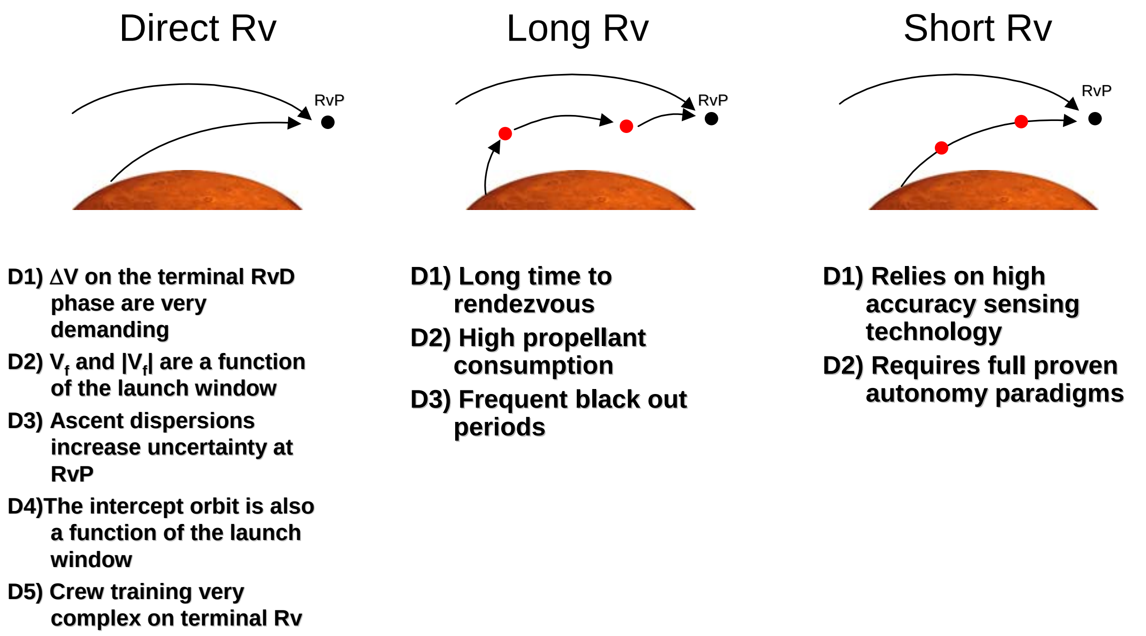


Figure 4-107: Rendezvous strategies disadvantages

As it can be seen from the figures, the disadvantages of the direct rendezvous are such that it is quickly discarded.

The trade-off remains between the long and the short rendezvous strategies.

From the Figure 4-106 it can be mentioned that the short rendezvous technique requires optimal correction manoeuvres. This strategy is direct targeting the final orbit by introducing several mid-course correction (MCC) manoeuvres. It has two possible sub-scenarios: to place the MAV in a low orbit at 500x500 or to place the MAV in a high orbit at 17000x17000 (the Mars geo-stationary orbit).

From the Figure 4-106 it can be mentioned that the long rendezvous technique does not require any optimal correction manoeuvres in between. This strategy is targeting several intermediate orbits until reaching the final one. It has also two possible sub-scenarios: to place the MAV in a low orbit at 500x500 or to place the MAV in a high orbit at 17000x17000 (the Mars geo-stationary orbit).

Supposing that the final orbit is 500x500, the long rendezvous needs to:

- adjust the nodes of the two orbits of our vehicles (the so called phasing part of the rendezvous): the Capsule Module (CM) of the MAV with three astronauts on-board at 450 km altitude with the Orbital Module (orbiter) at 500 km altitude
- establish a plane correction (PC) burn: this burn will align the orbital planes of the capsule and the orbital module. Most likely that this PC will be small, but is any case it will be needed due to the MAV ascent dispersions.
- travel the altitude difference of 50 km between the two orbits
- prepare and allow terminal rendezvous

4.5.2.4 Phases of RvD for the long rendezvous strategy

This section is devoted to explain the long rendezvous strategy as selected by the system enginner.

Let us select the LHLV coordinate system, which has the origin in the target, one axis pointing in the direction of the flight and the other one perpendicular pointing towards the planet. The selected approach for the RvD mission arc is based on the “above and ahead manoeuvre” type: the active chaser moves towards the target from a lower orbit and behind the target in the relative position.

There are three phases during this RvD in circular near co-planar orbits:

- 1st Phase. Preliminary RvD: Find the target (wherever the target is) and determine its orbit
- 2nd Phase. Intermediate RvD: Find the target (wherever the target is) and determine its orbit.
- 3rd Phase. Terminal RvD: Final approach and structural latching
- 4th Phase. Transfer of astronauts to the orbiter and de-docking of the MAV.

A general overview of the activities involved in the different phases is sown in Figure 3. The GNC units needed in each mode are listed at the end of each subsection.

4.5.2.4.1 Preliminary rendezvous

First thing to do is to achieve a precise orbit determination of the orbiter. This is done by means of Deep Space Network Doppler from Earth ground station. One week is needed to achieve an accuracy of meters and meters per second in position and velocity.

At the same time the orbit of the MAV will be computed, ranging from the orbiter to the MAV first, and via Deep Space Network after. The radio finder (RF) system will be used to locate initially the MAV.

Finally, the phase angle between the orbiter and MAV will be progressively reduced (lower orbit has shorter orbital period). This phase ends with a relative small true anomaly between chaser and target, and a small difference in orbital planes.

During this phase, the chaser will make a change of plane manoeuvre to correct the differences. Table 4-52 shows a simulation in which chaser and target are separated by a true anomaly of 12.2 degrees.

	Target	Chaser
Semi major axis (km)	3897.515	3843.515
Eccentricity	0	0
Inclination (degrees)	47	45
RAAN (degrees)	269.0307	267.0307
Argument of perigee (degrees)	25.956687	25.956687
True anomaly (degrees)	122.28026	110

Table 4-52: Simulation results

The result of the plane change manoeuvre can be seen in Figure 4-108.

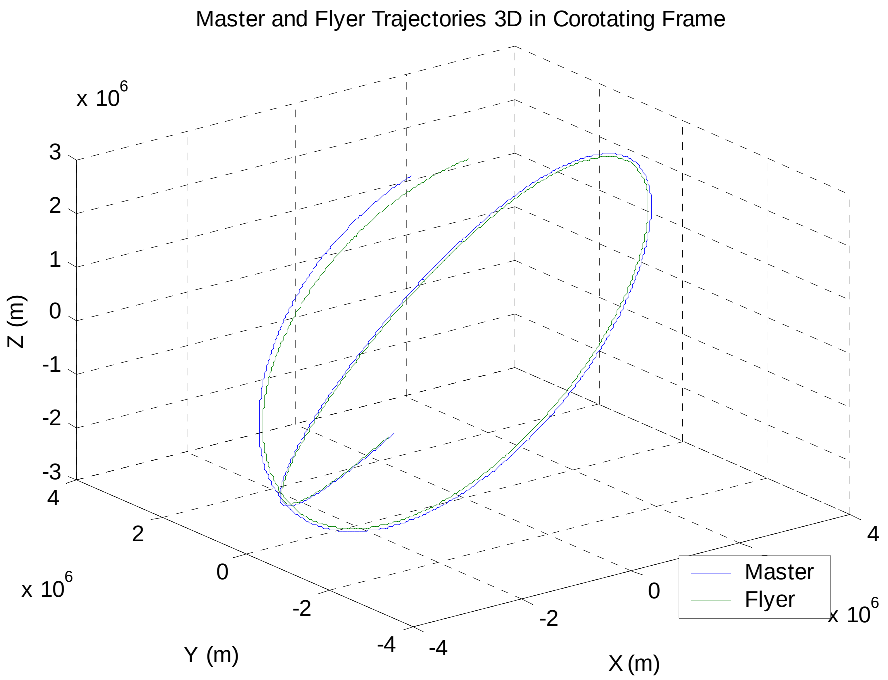


Figure 4-108: Plane change manoeuvre results

The corresponding inclination plot can be seen in Figure 4-109. This change of plane has a first Impulse at 1593 seconds. The corresponding DeltaV is negligable. Then a second Impulse is given at 1617 seconds with the following DeltaV of [27 115 -82] m/s

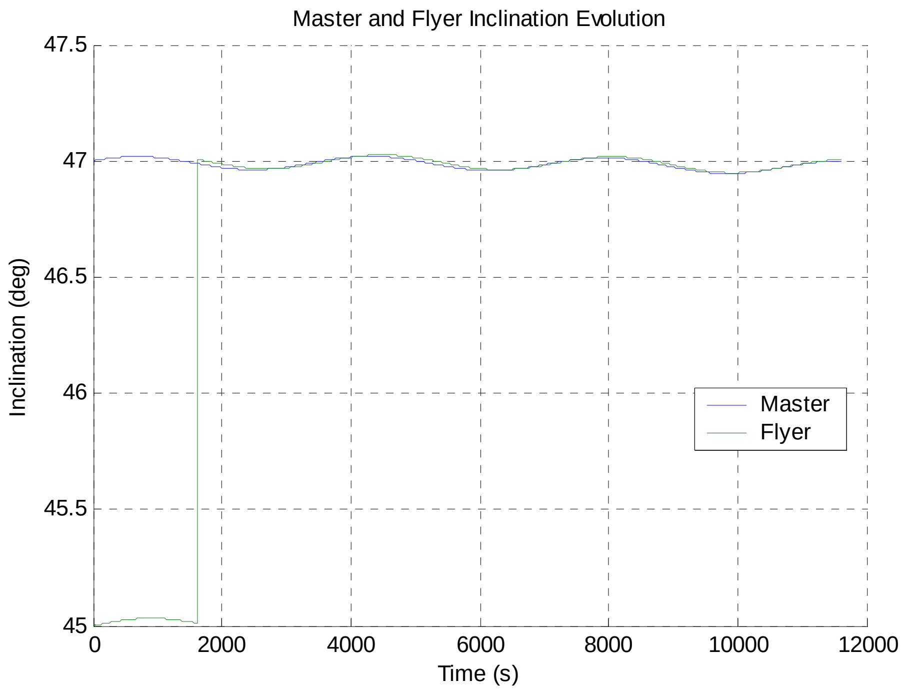


Figure 4-109: Inclination evolution

The total dV is143.5 m/s.

4.5.2.4.1.1 Intermediate rendezvous

After any plane change, the chaser is at 50 km below the target and at 140 km behind it. A Hohmann transfer is executed to move the Orbiter towards the MAV, from a point 20 km behind the target and in V-bar (see Figure 4-110).
For the intermediate rendezvous, a trade-off was mad between several substrategies:

- To place the MAV above and ahead drifts backwards towards the target
- To place the MAV below and behind drifts in front of the target
- To place the MAV below and ahead drifts the MAV in from of the orbiter

Finally, the last substrategy was selected: To place the MAV below and ahead. It is a quick transfer based on Shuttle guidance to ISS docking port that has very good manual backup features.
After the Hohmann manoeuvre, a tangential transfer in V-bar is executed to place the chaser 1 km ahead of the target maintaining it still in V-bar.
At that point a station keeping is commanded and a series of test start. These hold points will allow the astronauts to verify the authorisation to proceed for the next phase. The hold point is expensive in terms of fuel. It accounts for for the following tasks:

- Checking out subsystems.
- Occurrence for the sub-solar angle: Earth-orbiter-Sun angle more than 5 degrees.
- Communication with Earth for the ATPs (Authorisation to Proceed) to the next point.

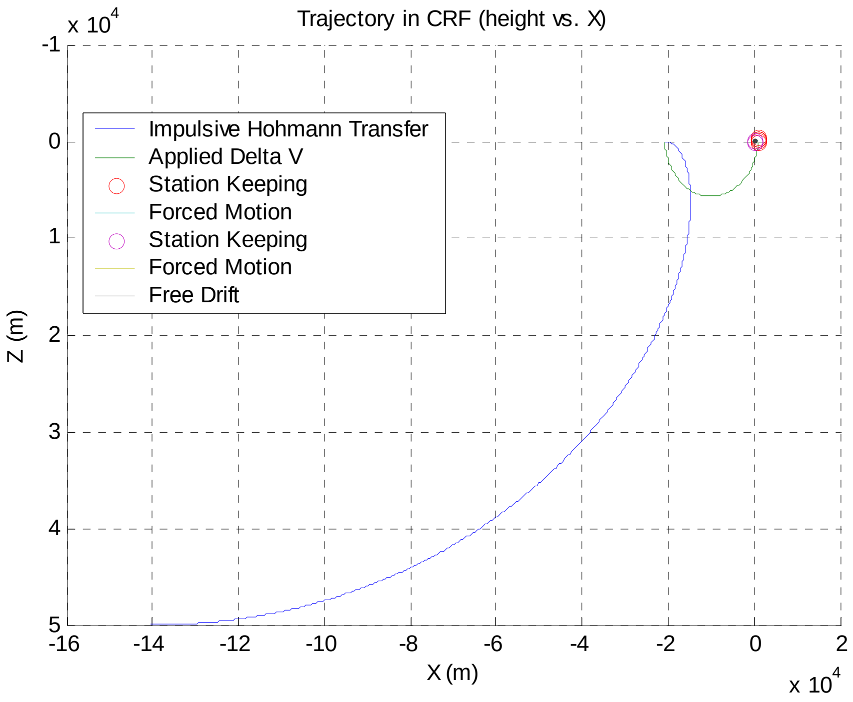


Figure 4-110: Intermediate rendezvous

4.5.2.4.2 Terminal rendezvous

Next, the crew commands a forced approach to the next hold point located at 100 meters in front of the target and also in V-bar. This new hold point will be the base of a station keeping manoeuvre that will allow the crew to check again the status of the subsystem and the availability of the docking port, etc.
Finally, a new forced approach is commanded and the MAV reaches the docking port of the Orbiter that establishes the structural latching of all required elements.

4.5.2.4.3 Transfer of crew and dedocking

Finally the structural latching occurs. The crew is transferred from the MAV to the orbiter. After that the MAV is ejected. Then there is a check out of all orbiter subsystems. Once this is done, the departure to Earth can begin.

4.5.2.5 Baseline design

The MAV final stage is a three-axis stabilised spacecraft.

The attitude determination main sensors are the star tracker (STR) and the inertial measurement unit (IMU). During Sun acquisition and safe modes, Sun acquisitions sensor and Earth infrared sensors are principally used. The two STRs that will be operated in parallel for some phases of the mission.

The attitude control actuators are a set of reaction wheels assembly (RWA), 4 clusters of 5 hydrazine thrusters plus 4 clusters of 2 hydrazine thrusters of 10 N thrust each, and two ASTRIUM engines acting with 550 N force each.

The RWA includes 4 Reaction Wheels (RWL) implemented on a skewed configuration. This configuration enables to perform most of the nominal operations of the mission with a 3 RWL configuration among 4. The Reaction wheels provide the GNC control torques during all the phases of the mission except the trajectory corrections, the attitude acquisition and back up modes. Each Reaction Wheel provides an angular momentum storage capacity of 12 Nms and a motor torque of ± 75 mNm in the speed range from + 6000 rpm to - 6000 rpm. Under nominal conditions, the RWA uses a 3-wheel configuration. The fourth wheel is used during critical mission phases to avoid any mission outage if one wheel fails and minimises the reconfiguration duration.

During nominal modes hydrazine thrusters are used to perform wheel off- loading, while during safe mode they become the attitude control actuators.

The 10 N thrusters for attitude control are all located in the second stage of the orbiter. A detailed analysis of the CoG and the controllability of the orbiter has not yet been done. However, the present configuration based on the ATV looks feasible in this first iteration.

The LIDAR expected features are the following:
- Validity range of the measurements shall be between 1 metre and 10000 metres
- Field of view (FoV) shall be of no less than 10 degrees
- The weight shall be of no more than 10 kg.
- The average power consumption shall be of no more than 20 Watts.

Even with the specifications above, the power consumption and weight values are slightly increased, according to the latest news about the LIDAR development process.

The CAMERA features to be tested shall be as follows:
- The validity range of the measurements shall be between 0 metres and 5000 metres to the target
- The field of view (FoV) shall be of no less than 20 degrees
- The weight shall be of no more than 2 kg.
- The average power consumption shall be of no more than 3 Watts.

In the same way as with the LIDAR, slightly more conservative values have been taken.
Table 4-53 summarizes some of the characteristics of the GNC equipment.

	Validity range	Accuracy	Comments
Radio Frequency Finder System (RFF)	4000 km … 0.5 km	10 m D ∈ [1km,3000m] 1 m D ∈ [100m,1000m] 0.7 m D ∈ [10m,100m] 0.07 m D ∈ [0m,10m]	4 antennae with FoV:120 deg
Star Tracker (STR)	0° … 360°	±0.3 deg	FoV: 25 deg
Inertial Measurement Unit (IMU)	0° … 360° 0°/s … 360°/s 0g … 5g	±0.3 deg 1 deg/hr drift 2% error in accel.	NA
LIDAR	5000 m .. 0m	2 m, 2mrad D ∈ [1km,5000m], 0.2 m, 2mrad D ∈ [100m,1km] 0.02 m, 20mrad D ∈ [0m,100m]	FoV: 10 deg
Camera	200m … 0m	0.1 m, 50mrad D ∈ [0m,50m]	FoV: 10 deg

Table 4-53: GNC equipment characteristics

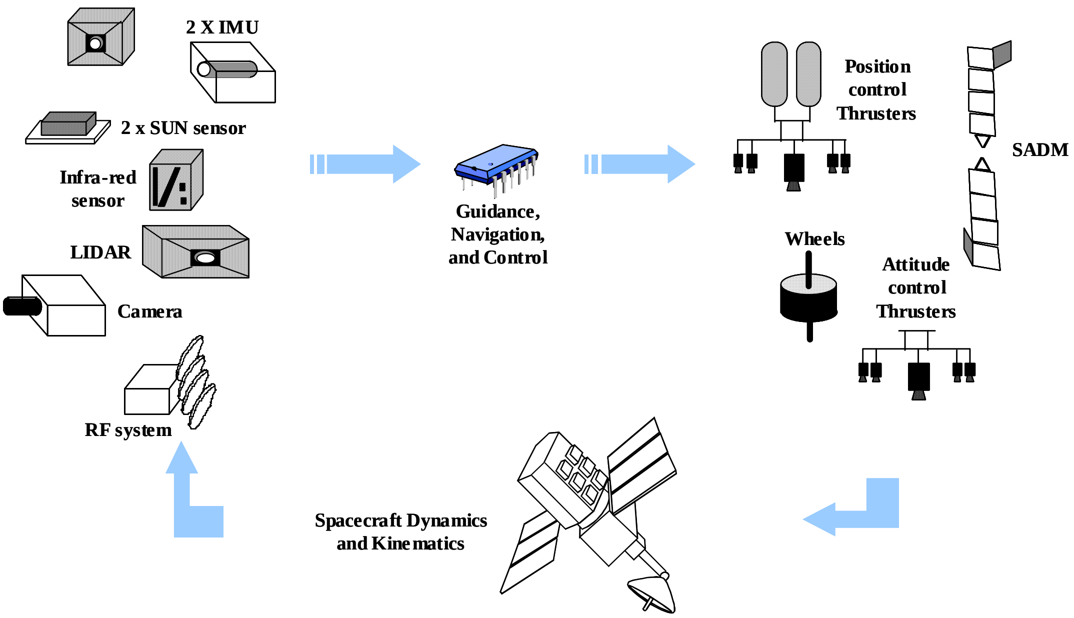


Figure 4-111: GNC schematic of the orbiter

4.5.2.6 Budgets

Table 4-54 shows the mass and power budget of the baseline design for the GNC equipments.

Element 3: Mars Ascent Vehicle			MASS [kg]			
Unit	Element 3 Unit Name Click on button below to **insert new unit**	Quantity	Mass per quantity excl. margin	Maturity Level	Margin	Total Mass incl. margin
1	IMU	2	2.500	Fully developed	5	5.3
2	Altimeter	1	1.000	Fully developed	5	1.1
3	LIDAR	1	4.000	To be developed	20	4.8
4	Camera	1	3.500	To be modified	10	3.9
5	Display	1	0.200	To be developed	20	0.2
6	Hand controls and buttons	2	0.800	To be developed	20	1.9
7	Electromecanic equipment	3	3.000	To be developed	20	10.8
8	Rendez-vous joysticks	4	0.200	To be developed	20	1.0
9	Rendez-vous TV	1	2.500	To be modified	10	2.8
10	Rendez-vous Radio Frecuency System	1	4.000	To be modified	10	4.4
11	Rendez-vous Camera	1	2.500	To be modified	10	2.8
12	Rendez-vous LIDAR	1	4.000	To be developed	20	4.8
-	Click on button below to insert new unit			To be developed	20	0.0
	ELEMENT 3 SUBSYSTEM TOTAL	**12**	**38.1**		**14.4**	**43.6**

Table 4-54: GNC equipment: mass budget

Figure 4-112 shows the RvD budget in terms of time and fuel consumption.

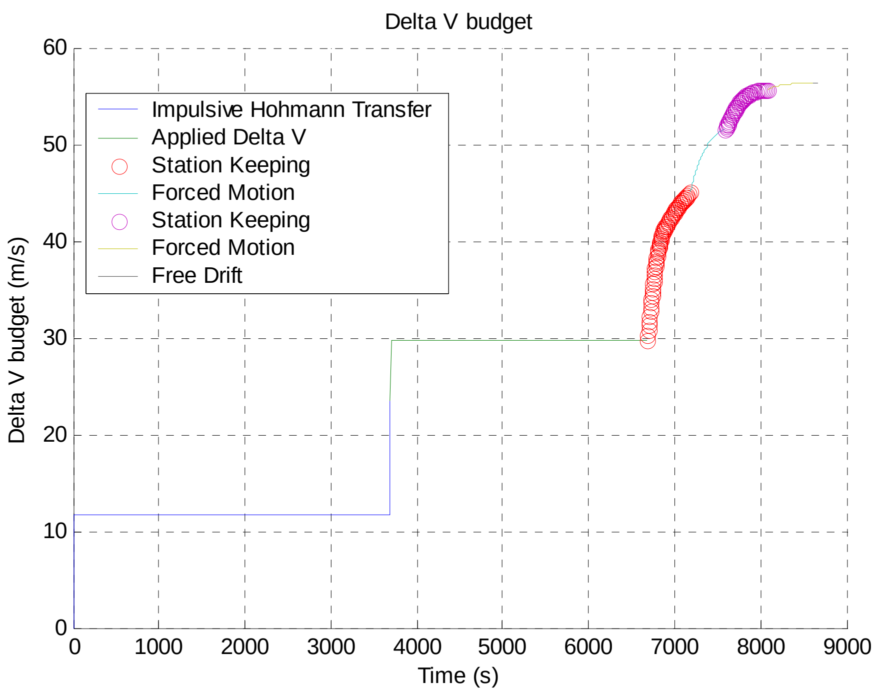


Figure 4-112: RvD budgets: time and fuel

4.5.2.6.1 Manual rendezvous backup system

This manual system is based on the one that is on-board the Soyuz. The astronaut controls the vehicle by manually translating and rotating around its centre of gravity. In order to do this the astronaut needs to perform a loop with these steps.

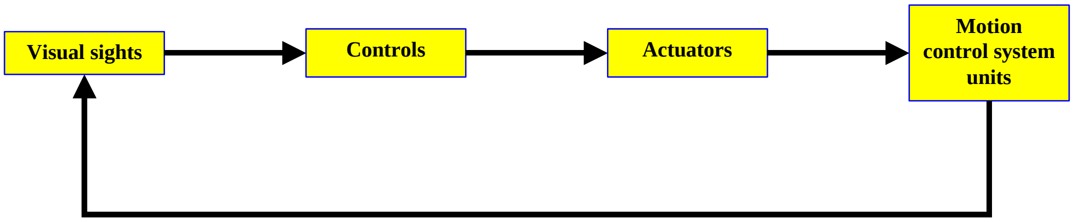


Figure 4-113: Instrumentation and equipment used for manual final approach

- For the Visual Sights: The crew of the Soyuz uses two equipments. One is a periscope (BCK-4) and the other is a television system video monitor.

  The sight(periscope) is the primary optical instrument. It has a dual screen that allows the astronaut to view the target image. The periscope has two positions: attitude sight(aimed along the transport vehicle –Y-axis) and approach sight (aimed along the transport vehicle –X-axis). A reduced-scale colour image of the target is projected onto the screen.

  The video monitor is another mean for viewing the target. It receives its image signal from the external TV camera mounted on the habitation module. The crew only uses this system for qualitative assessment of the final approach process.

- Transport Vehicle Controls.

  The astronauts have two controls. One translation hand controller and one rotation hand controller. The controls are mounted on brackets in the descent module near the center astronaut seat.

  The translational hand controller has two degrees of freedom. Tilting the stick in either direction will translate the vehicle's centre of gravity in the directions ±Y and ±Z of the spacecraft coordinate system. The controller has a toggle switch with a middle (neutral) position to generate the signals required to translate the vehicle's center of gravity in the ±X direction.

  The rotational hand controller has three degreesrees of freedom. With it you can control the vehicle's rate of rotation to control attitude.

4.5.2.6.2 Mass and power budgets (redundancy included)

	Mass (kg)	Power (W)
Stick (Two)	1.6	15
Screen	2	50
Periscope	30	TBD
Video monitor	TBD	TBD

Table 4-55: Mass and power budgets

4.5.3 Structures

4.5.3.1 Requirements and design drivers

For the design of the Mars Ascent Vehicle the following set of general requirements were taken into account:

- Compatibility with the vehicle launcher Energia induced mechanical loads.
- Compatibility with Energia fairing envelope.
- MAV centre of gravity must be low as possible.

All structures module shall provide the mechanical support to ensure mission success.

4.5.3.2 Assumptions and trade-off

The MAV centre of gravity is assumed to be at 1 m, with the referential at the bottom part of it.

4.5.3.3 Baseline design

Due to the short time frame for this module, no detailed analysis was performed. The analysis performed for the THM was used as reference.

4.5.3.3.1 MAV shell

The MAV consists of a cylinder with 2.2 m of total length and 2.8 m of diameter.
For the MAV mass determination, the first analysis began with comparison with the values of the Soyuz capsule without the heat shield. But the necessary usable volume for the MAV is higher, so as second analysis, due to the shape similarity, it was compared with the Columbus laboratory. For the thickness the same value was chosen as for the laboratory – 4 mm and for the skin material the same aluminium alloy as for the Columbus laboratory – AL 2219.
As preliminary analyses the stiffeners mass was assumed to be half of the skin mass.
For strength of the MAV it was assumed to have a ring every 0.6 m. The aim of these rings is to give the necessary rigidity to the MAV.

4.5.3.3.2 MAV propulsion module

The MAV Propulsion Module consists of two stages. Each stage has four tanks and four engines, around a central cylinder, which accommodates a passage tunnel between MAV and SHM.
For each stage there is a plate that supports the tanks and then engines. These plates consist of sandwich panels. All panels will be aluminium alloy for the face sheet, with 5 mm thickness and the core 30 mm thickness. In Figure 4-114 is described the nomenclature used for the structural parts for both stages.

Note: The MAV shape shown in Figure 4-114 shows not the last version.

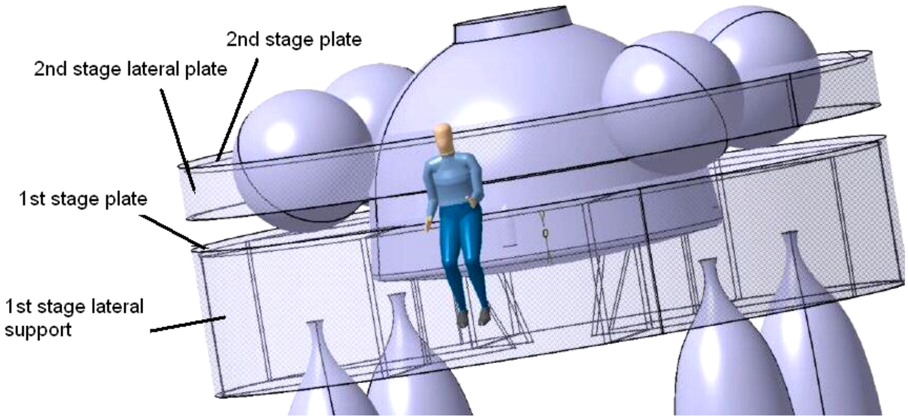


Figure 4-114: MAV Propulsion Module

4.5.3.4 Budget

Item	Nr.	Mass [kg]	Margin [%]	Mass with Margin [kg]
MAV Skin	1	219.53	20	263.43
MAV Stiffeners	1	109.76	20	131.72
1st stage plate	1	24.12	20	28.95
1st stage lateral support	1	70.68	20	84.81
MAV Rings	4	61.29	20	73.55
2nd stage plate	1	24.12	20	28.95
2nd stage lateral support	1	25.44	20	30.53
TOTAL				862.59

Table 4-56: SHM Structures mass budget

4.5.4 Power

4.5.4.1 Inputs and assumptions

4.5.4.1.1 Architecture

The mission of the MAV is to:

- Launch from the Martian surface to a parking orbit around Mars (max duration 90 minutes)
- Stay a few days on this orbit (Orbit: 118 min, max eclipse: 41 min)
- Perform the rendezvous with the TV (max duration 30 min)

Reviewing the power that needs to be supplied during all these modes, a non-regenerative power system would be too heavy, either with fuel cells or primary batteries.
Therefore, solar cells are required in the design to decrease significantly the power storage module.

As regards the parking orbit, the attitude of the satellite is not constrained at all (use of a patch antenna, no payload pointing requirement…). Consequently, solar panels can be body-mounted and always assumed sun-pointed during this phase. This solution is selected because it offers:

- the lightest system
- the most reliable system (no deployment or SADM mechanisms)

On top of the spacecraft, a flat area of 17.5 m² is available and will be used for mounting the cells (Figure 4-115).

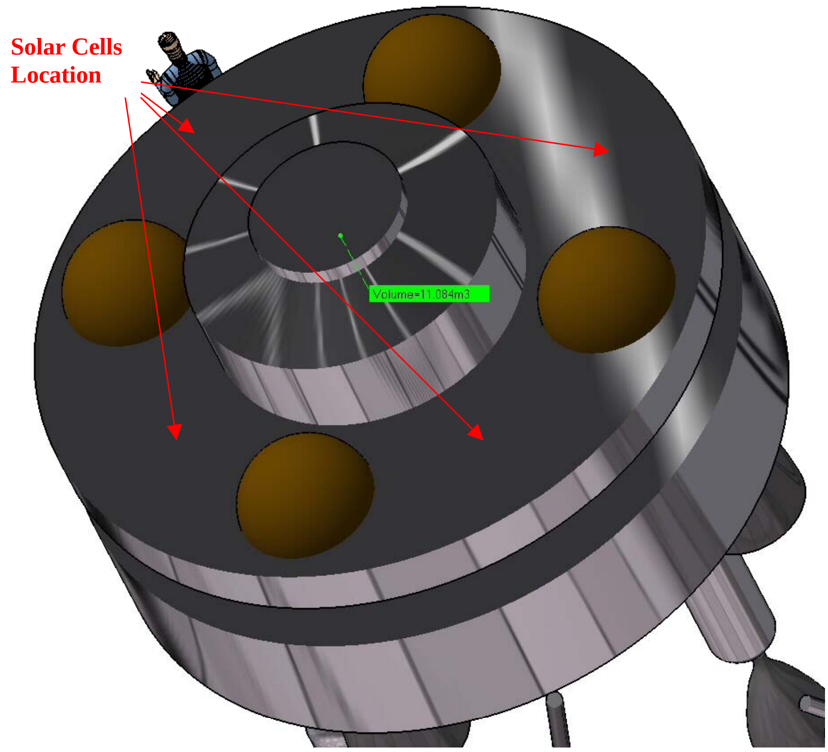


Figure 4-115: MAV solar cells location

4.5.4.1.2 Power storage

The use of a secondary battery is the best-qualified and efficient way of storing the required energy. Currently, Li-Ion cells offer the best performances (around 94% of energy round efficiency, a specific energy of 100Wh/kg). As for the Habitation Module, a specific energy of 150Wh/kg is expected to be reached in 2015 and will be considered in this study.
When these batteries will be in use (from the launch from the Martian surface until the rendezvous with the TV), the batteries will not have an important cycling effect degradation but are already between 2 to 5 years old (depending on the LEO assembly time). Hence, a maximum depth of discharge of 60% that should cover also the failure cases is selected.

4.5.4.1.3 Power generation

In this architecture, AsGa MJ cells are body-mounted on the top ring of the MEV because they offer the best conversion efficiency.

Since, the top ring is protected during the surface operations until the launch phase by a fairing, no power loss is expected from the dust deposit.
Moreover, for geometry rationales, the shape allocated for the solar cells will be difficult to optimise. Therefore, a filling factor of 80% is a low but reasonable value.
An efficiency of 32% AM0(28°C) is assumed for 2015. As a comparison, 25% efficiency cells will be body-mounted on Proba-2.
To cover the failure of a string on the solar panel, the solar panel is oversized by 5%.

4.5.4.1.4 Power conditioning and distribution

For the power conditioning, a regulated bus topology is assumed with a conservative efficiency value of 90% for the BCR (Battery Charge Regulator) and for the BDR (Battery Discharge Regulator).
This type of architecture fits the requirement of this mission. Others architectures (MPPT, S4R regulated…) may have similar or better performances. Such a trade-off is too early to be studied: The possible benefits of another architectures are relatively low in term of masses and volumes of the power modules compared to the level of detail reached on that phase of the study.

4.5.4.2 Budgets

The sizing case for the power storage and power generation system is during the parking orbit when:

- the solar flux is minimal
- the eclipse duration is maximal
- the equipments is operational during the eclipse

Solar Panels AsGa Improved	
Size (m^2)	14.52
Mass (kg)	43.77
PCU	
Mass (kg)	14.13
PDU	
Mass (kg)	9.00
Battery Li-ion Improved 150 Wh	
Capacity (Wh)	2239
Mass (kg)	16.17

Figure 4-116: MAV budget

82% of the area allocated for the solar cells is used.
During the rendezvous, the depth of discharge of the battery is estimated to 20.8%.

The total mass of the power subsystem (excluding the harness) is:

- 83.1 kg without margin on equipment level
- 91.4 kg with margin on equipment level

4.5.5 Thermal

The MAV thermal control shall be designed to perform optimally during the Mars ascent phase and the rendezvous and docking phases. Similar performance from the thermal control is expected during the landed phases, as it is assumed that the ascent compartment is an integral part of the habitable zone. During this phase, the same requirements therefore apply, with slight difference due to its upward position.
Like the MAV, the suitability of an optimal performance during the transfer to Mars is an open issue. Not necessarily a permanent habitable module (economy of a radiation shield), its functions can be hold in a dormant mode, reactivated when a crew enters the module (storable zone for example). Benefit of such scenario is a higher tolerance on the thermal control and a lower associated budget.

4.5.5.1 Requirements and design drivers

The main requirements are the following:

- The external thermal control shall be effective in vacuum (transfer and RdV phase) and in the Martian pressurised environment.
- The external thermal control shall cope with ascent aerodynamics thermal loads.
- The TCS functions are to maintain air temperature and humidity in the ascent vehicle zones within preset limits, and to thermally control the on-board systems. Therefore, TCS shall be designed to maintain:
    - the habitable zones in a certain comfort zone (temperature, humidity) but respecting also safety requirements (touch temperature, condensation avoidance). Standard figures are a medium temperature between 18 and 27C and a relative humidity from 25 to 70%.
    - a uniform environment for a crew up to three members.
    - elements and/or dedicated zones within temperature requirements (electronics, propellants, valves, …). To optimise the thermal budget, a certain rationalization of space and grouping of elements shall be carried out. Ideally, all equipments are within a single dedicated enclosure.
    - the interfaces of the others modules (Habitation Module) within temperature requirements.
- The candidate TCS architecture shall be also capable of:
    - performing effectively under Martian gravity,
    - guaranteeing adequate flexibility and reliability of the system during all phases until the end of the docking with the TV. Lower performance can be tolerated after docking
    - guaranteeing the performance of the system for any spacecraft attitude during transfer and RdV, as well as for any orientation after landing, this for all thermal loads derived from the mission requirements
    - to optimise the heat management system in term of efficiency versus penalties to the system (mass, energy consumption)

- guaranteeing by adequate provision of thermal hardware for the whole mission (necessary autonomy of the crew)
- fully verifying and testing the TCS on ground

4.5.5.2 Assumptions

4.5.5.2.1 Transfer, rendezvous and docking phases thermal environment

The same environment as for the transfer vehicle applies for the Mars Excursion vehicle including the ascent vehicle. A conservative approach is to consider envelopes through worst-case scenarios:

	Solar flux [W/m^2]	Planet albedo	Planet IR [W/m^2]
Hot case (Earth LEO, WS, 1 AU)	1423	0.33	241
Hot case (Mars orbit, perihelion, 1.38 AU) [2]	717	0.29 (subsolar)	470 (subsolar) to 30
Cold case (Mars orbit, aphelion, 1.66 AU) [3]	493	0.29 (subsolar)	315 (subsolar) to 30

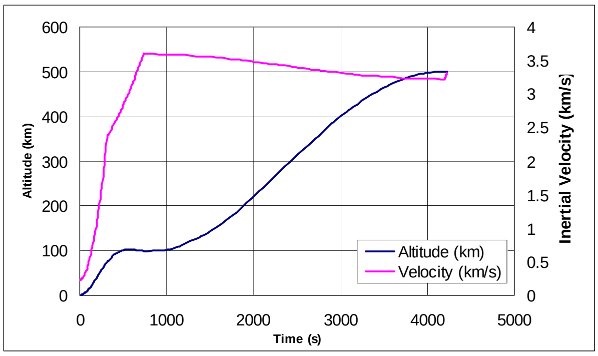
Table 4-57: Thermal cases definition

The docking has an envelope of maximal 4 days starting from the take off.

4.5.5.2.2 Martian thermal environment

The same environment as for the Habitation Module applies.

4.5.5.2.3 Martian ascent phase



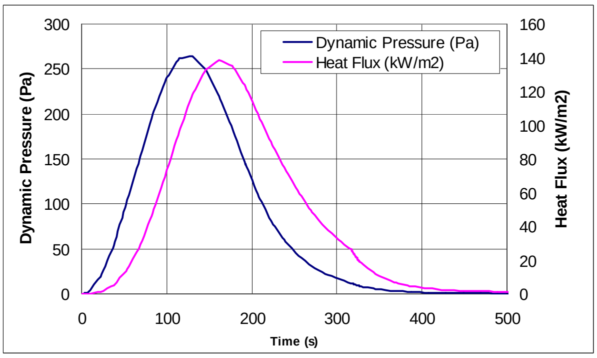


Figure 4-117: Flight Environment (from Trajectory analysis)

4.5.5.2.4 Man-induced thermal loads

The thermal design shall manage all internal heat loads resulting from the human activities and various dissipating equipments:

- Total mean heat load of 582W during ascent and parking orbit phases, 931W during rendezvous and docking phase.
- Metabolic dissipation is estimated to be 110W (steady activity) per crew (x 3)

4.5.5.3 Baseline thermal design

4.5.5.3.1 Ascent vehicle thermal control

With no direct expertise in Europe available for such vehicle, the design block proposed is based partly on the exploitation of foreign existing heritage: Apollo LM, LOK (derived Soyuz). Space station fluid loops technologies are applicable to a certain extent (shall work against gravity).

The thermal control philosophy adopted for such vehicle is standard and relies on the following approach:

- simplification of the heat transfer with maximal use of thermal decoupling when possible
- use of thermal-regulated bus to recuperate and transfer internal heat to heat sinks
- use of switch capability to modulate this transfer and balance the heat inputs from the outputs, and thus maintain temperatures within a certain bandwidth

This is implemented using appropriate materials and technologies combining passive or active means.

4.5.5.3.2 Thermal bus and radiator

- Docked and descent phases

Due to the staging with the SHM, a direct connection is designed with the SHM (quick disconnect). As long as this coupling exist, the SH module thermal bus is used providing a cooling capability when necessary.

- Landed phase

The cooling capability designed for the free flight phases is used in conjunction with the SHM heat rejection system.

- Ascent, rendezvous and docking phases

Considering the requirement of 4 days, a complete and independent thermal control system has to be designed. A Soyuz / LOK type thermal control is adopted:

- The secondary fluid loop is based on Polymethylsiloxane as working fluid, the radiator located on the lateral sides of the main cylindrical body
- The primary loop is based on water as working fluid, both lines connected via a heat exchanger

On the basis of 931W of rejected power and 330W metabolic heat, a radiator size of 8 m^2 is needed. This is implemented in a cylindrical shape type (eight surfaces of 3.6 m x 0.71 m length).

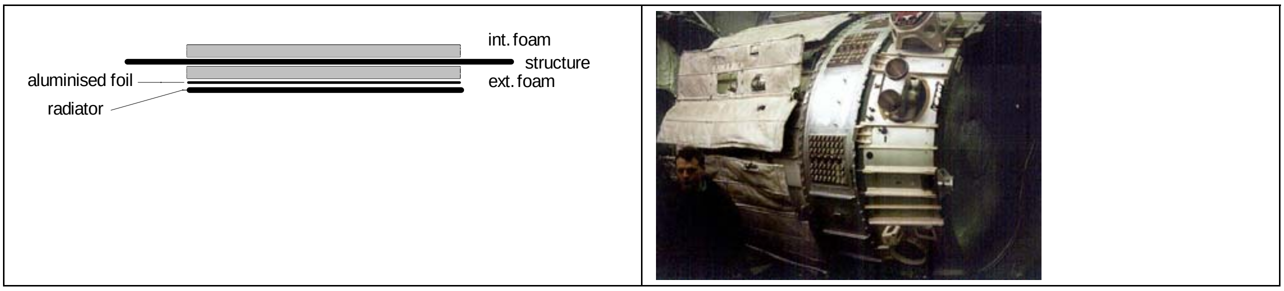


Figure 4-118: **Radiator layout (L), LOK radiators (nota: bent over the years) (R)**

4.5.5.3.3 Primary and secondary loop

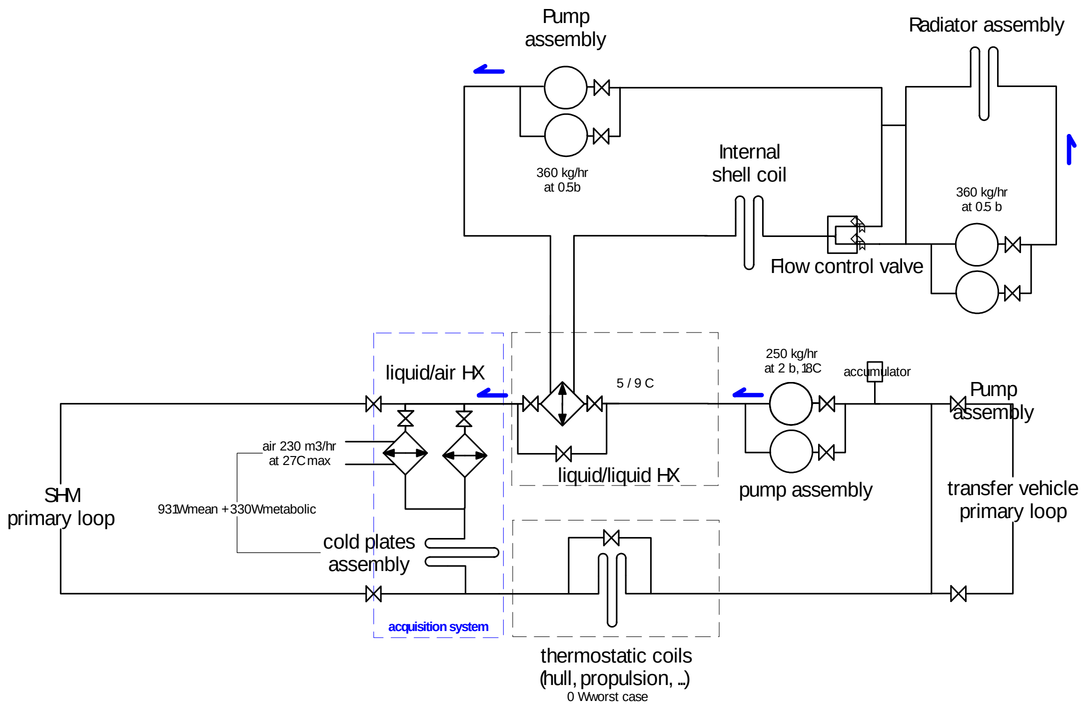


Figure 4-119: **Ascent vehicle / primary and secondary loop principles**

4.5.5.3.4 The insulating system and thermal protection

- TPS sizing to be done utterly. A provision of 50 kg has been taken into account for the mass budget
- Fuel tank insulation

The ascent vehicle fuel tanks are insulated by lightweight foam, type Basotect (open cell foam) appropriate to the Martian environment. An external goldenized kapton foil is added to reduce radiative heat exchange. The tanks are assumed to be enclosed in a protective frame whereas a thin TPS layer is added.

4.5.5.3.5 The thermostatic system

Certain surfaces that cannot be protected by insulating means (docking system for the MAV) are treated (oxidation anodic, alodine) to minimise heat losses. On the internal face, coils (circulating fluid from primary loop) thermostatically control the temperature (condensation avoidance) and the heat exchanges (control of the heat losses). An adequate redistribution of the rejected heat (thermostatic coils) therefore reduces the use of heater power to the minimum.

4.5.5.4 Budget

4.5.5.4.1 Synthesis per subsystem (main features)

Fluid loops	
Primary loop	Pump assembly: 10 kg, 50W nominal (250 kg/hr) (x 2) Condenser heat exchangers: 20.6 kg (x 1), cold plates: 3.4 kg (x 5), valves (on/off, manual): 4 kg (x 10) 26 kg of tubing (dry including insulation, brackets) + 21 kg of water
Secondary loop	Pump assembly: 6 kg, 30W nominal (x 2) Heat exchangers: 5 kg, cold plates: 3.4 kg (x 2), flow control valve: 10 kg 37.3 kg of tubing (dry including insulation, brackets) + 31kg of PMS
Passive thermal control	
External radiator	One radiator of 8 m2, weighting 43 kg (5.4 kg/m2)
Insulation	0 kg for the main body of the transfer vehicle: the thermal properties of the MOD shield are exploited, the related budget transferred to structure. 50 kg are provisioned for specific external and internal elements insulation. 50 kg are provisioned as heat shield
Heating system	300W installed power (heating of the lines) 2 control units (1 on), each 6 kg, 29W when shell heaters are 100% duty cycle
Thermal Protection System and underneath structure	
Fairing	Composite structure (core honeycomb, carbon shell) TBD mm of Norcoat (TPS), provision of 50 kg
Tanks	30 mm of basotect + goldenized kapton layer: TBD kg/m2 encapsulated structure (honeycomb TBD mm) + TBD mm Norcoat (TPS): TBD kg per tank (x 4)

Table 4-58: Main features

4.5.5.4.2 Overall budget (as introduced to the system)

Element 3: Mars Ascent Vehicle			MASS [kg]				PPEAK AND POWER SPECIFICATION PER MODE																					
Unit	Element 3 Unit Name Click on button below to insert new unit	Quantity	Mass per quantity excl. margin	Maturity Level	Margin	Total Mass incl. margin	Ppeak	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-
								Pon	Pstby	Dc	Pon	Pstby	Dc	Pon	Pstby	Dc	Pon	Pstby	Dc	Pon	Pstby	Dc	Pon	Pstby	Dc	Pon	Pstby	Dc
1	MAV / front shield	1	50.0	To be developed	20	60.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
2	MAV / tanks 1st stage	4	0.0	To be modified	10	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
3	MAV / tanks sd stage	4	0.0	To be modified	10	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
4	MAV / radiator	1	43.0	Fully developed	5	45.2	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
5	MAV / insulation	1	50.0	To be developed	20	60.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
6	MAV / HCU	2	6.0	Fully developed	5	12.6	29.0	29.0	13.7	0.0	29.0	13.7	0.0	29.0	13.7	60.0	29.0	13.7	0.0	29.0	13.7	0.0	29.0	13.7	60.0	29.0	13.7	50.0
7	MAV / Heaters/Thermostat/oth	1	1.0	Fully developed	5	1.1	300.0	300.0	0.0	0.0	300.0	0.0	0.0	300.0	0.0	60.0	300.0	0.0	0.0	300.0	0.0	0.0	300.0	0.0	60.0	300.0	0.0	50.0
8	int. loop / liquid	1	21.0	Fully developed	5	22.1	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
9	int. loop / dry tubing + insulation	1	26.1	Fully developed	5	27.4	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
10	int. loop / pump assembly	2	10.0	To be modified	10	22.0	50.0	50.0	0.0	10.0	50.0	0.0	60.0	50.0	0.0	60.0	50.0	0.0	60.0	50.0	0.0	60.0	50.0	0.0	60.0	50.0	0.0	30.0
11	int. loop / compensator	2	3.0	Fully developed	5	6.3	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
12	int. loop / liq-liq heat exchanger	1	54.5	To be modified	10	60.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
13	int. loop / cooler-dryer assembl	2	30.0	To be modified	10	66.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
14	int. loop / cold plates	5	3.4	Fully developed	5	17.9	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
15	int. loop / all valves	10	4.0	Fully developed	5	42.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
16	interm. loop / liquid	1	1.5	Fully developed	5	1.6	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
17	interm. loop / dry tubing + insula	1	1.9	Fully developed	5	2.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
18	interm. loop / pump	2	6.0	To be modified	10	13.2	50.0	50.0	0.0	10.0	50.0	0.0	10.0	50.0	0.0	10.0	50.0	0.0	10.0	50.0	0.0	60.0	50.0	0.0	60.0	50.0	0.0	30.0
19	ext. loop / compensator	2	3.0	Fully developed	5	6.3	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
20	ext. loop / liquid	1	31.0	Fully developed	5	32.6	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
21	ext. loop / dry tubing + insulatio	1	37.3	Fully developed	5	39.2	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
22	ext. loop / pump	2	6.0	To be modified	10	13.2	50.0	50.0	0.0	10.0	50.0	0.0	10.0	50.0	0.0	10.0	50.0	0.0	10.0	50.0	0.0	60.0	50.0	0.0	60.0	50.0	0.0	30.0
23	ext. loop / liq-liq heat exchanger	1	5.0	To be modified	10	5.5	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
24	ext. loop / cold plates	2	3.4	To be modified	10	7.5	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
25	ext. loop / flow control valve	1	10.0	To be modified	10	11.0	22.0	22.0	2.9	10.0	22.0	2.9	10.0	22.0	2.9	10.0	22.0	2.9	10.0	22.0	2.9	10.0	22.0	2.9	10.0	22.0	2.9	10.0
-	Click on button below to insert new unit			To be developed	20	0.0																-						
ELEMENT 3 SUBSYSTEM TOTAL		25	524.1		9.6	574.3	501.0	501.0	16.6		501.0	16.6		501.0	16.6		501.0	16.6		501.0	16.6		501.0	16.6		501.0	16.6	

Table 4-59: Overall budget

4.5.6 Mechanisms

4.5.6.1 Requirements and design drivers

The HMM science requirements do not state any specific requirements applicable to the MAV Mechanisms. As a result of the MAV's configuration, the following necessary mechanism and their requirements can be derived:

- Vehicle Connections
  - Berthing & Docking Capability
    - Mars Ascent Vehicle
      - Berthing & Docking in Martian orbit
      - Un-docking during Martian orbit
- Crew Egress Hatches
  - External Hatches and Locking Mechanism at the TV Docking Port and SHM Separation I/F.
- Vehicle Stage Separation System
  - Release & Separation of Ascent Stage1.
- Egress Hatch (MAV/TV I/F) contamination protection layer (disposable).

4.5.6.2 Assumptions and trade-offs

<u>Vehicle Connection In-orbit:</u>

The following assumptions have been derived as a result of the study:-

- The same Berthing and docking Mechanism shall be used throughout the system- IBDM
  - Androgynous system- identical mechanism mounted to both vehicles
  - Full redundancy of system provided
  - Full Internal Mechanism redundancy
  - Treble redundancy for release/emergency release.
  - Mechanism independent of hatch door.
  - Hatch door diametre limited to ingress/egress suitability (diameter 813 mm).

Crew Ingress/Egress Hatch:

Once on the Martian surface, the external face of the hatch will be exposed to the Martian atmosphere and will thus be contaminated. Once the MAV has re-docked with the TV in Mars orbit, the hatch is opened to allow egress of the astronauts. This will expose the external face of the hatch to the inner habitable volume. Therefore a disposable, cover shall be implemented, to be disposed of during MAV orbit prior to Berthing & Docking.

4.5.6.3 Baseline design

4.5.6.3.1 Vehicle connections

The International Docking and Berthing Mechanism shall be implemented for the TV/DM Interface.

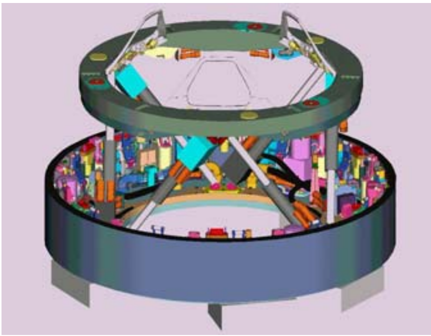


Figure 4-120: International Berthing & Docking Mechanism

The IBDM has the following mechanical characteristics:

- Interface loads (at the sealing interface)- acting simultaneously while docked (Flight-limit)
    - Axial load (1200 lbf) 5338 N
    - Shear load (1000 lbf) 4448 N
    - Bending moment (80000 in*lbf) 9039 Nm
    - Torsion moment (70000 in*lbf) 7909 N*m
- Internal Pressure (16 psi) 110316.1 Pa
- Life 15 years, Functional Life 20 Berthing/un-berthing or Docking/undocking cycles

4.5.6.3.2 Crew egress hatches

Sealable hatches are required for the following I/Fs

- MAV to TV
- MAV to SHM

The hatch diametre is sized to fit within the (current) IBDM tunnel; ≈800 mm. Both hatches will require Latch and Seal mechanisms. Mass estimates shall be realised using a ‘simple geometry’ model.

A disposable cover shall be implemented over the outer surface of the Egress hatch door. The method of separation shall be pyrotechnic bolts.

Vehicle Separation:

The separation of the MAV first and second stages shall be realised with a pyrotechnic operated Clamp-band of about ∅1.5 to ∅2.0.

4.5.6.4 Budgets

Element 3: Mars Ascent Vehicle			MASS [kg]				DIMENSIONS [m]		
Unit	Element 3 Unit Name Click on button below to insert new unit	Quantity	Mass per quantity	Maturity Level	Margin	Total Mass incl. margin	Dim1 Length	Dim2 Width	Dim3 Height
1	Docking Mechanism- IBDM	1	334.4	To be modified	10	367.8	1.371	0.813	0.254
2	Electronic Box- IBDM	6	8.8	To be modified	10	58.1	0.4	0.25	0.25
3	Hatch Door- Egress External	2	18.0	To be developed	20	43.2		0.9	0.01
4	Hatch Door Locking Mechanisms- Egress Externa	2	120.0	To be developed	20	288.0	0.95	0.80	0.05
5	Hatch Door Cont. Protection Cover	1	15.0	To be modified	10	16.5			
6	Clamp-band- Stage 1/Stage 2 I/F	1	15.6	To be modified	10	17.2		1.200	
-	Click on button below to insert new unit			To be developed	20	0.0		-	
	ELEMENT 3 SUBSYSTEM TOTAL	6	693.8		14.0	790.8			

Table 4-60: MAV Mass Budget

Unit	Element 3 Unit Name Click on button below to insert new unit	Quantity	Ppeak	DESM Pon	DESM Pstby	DESM Dc
1	Docking Mechanism- IBDM	1		784.0		1.1
2	Electronic Box- IBDM	6				
3	Hatch Door- Egress External	2				
4	Hatch Door Locking Mechanisms- Egress Externa	2				
5	Hatch Door Cont. Protection Cover	1				
6	Clamp-band- Stage 1/Stage 2 I/F	1				
-	Click on button below to insert new unit			-		
	ELEMENT 3 SUBSYSTEM TOTAL	6	0.0	784.0	0.0	

Table 4-61: Power Budget- Descent

Element 3: Mars Ascent Vehicle			AND POWER SPECIFICATION PER			
Unit	Element 3 Unit Name Click on button below to insert new unit	Quantity	Ppeak	RVDM Pon	RVDM Pstby	RVDM Dc
1	Docking Mechanism- IBDM	1		1806.0		12.7
2	Electronic Box- IBDM	6		152.0		75.7
3	Hatch Door- Egress External	2				
4	Hatch Door Locking Mechanisms- Egress Externa	2				
5	Hatch Door Cont. Protection Cover	1				
6	Clamp-band- Stage 1/Stage 2 I/F	1				
-	Click on button below to insert new unit			-		
	ELEMENT 3 SUBSYSTEM TOTAL	6	0.0	1958.0	0.0	

Table 4-62: Power Budget- Docking

Element 3: Mars Ascent Vehicle			TEMPERATURE REQs [deg C]			
Unit	Element 3 Unit Name Click on button below to insert new unit	Quantity	Operation (max)	Operation (min)	NOP (max)	NOP (min)
1	Docking Mechanism- IBDM	1	50.0	-50.0	100.0	-100.0
2	Electronic Box- IBDM	6	50.0	-20.0	70.0	-50.0
3	Hatch Door- Egress External	2				
4	Hatch Door Locking Mechanisms- Egress Externa	2				
5	Hatch Door Cont. Protection Cover	1				
6	Clamp-band- Stage 1/Stage 2 I/F	1				
-	Click on button below to insert new unit		-			
ELEMENT 3 SUBSYSTEM TOTAL		6				

Table 4-63: Thermal Constraints

4.5.7 Propulsion

4.5.7.1 Requirements and design drivers

The payload mass for ascent is estimated 4200 kg.

A thrust of 130 kN is required for the first stage to maintain the T/M ratio at acceptable value
A thrust of 20 kN is required for the second stage to maintain the T/M ratio at the same value

4.5.7.2 Assumptions and trade-offs

The ascent manoeuvre is staged in with two different modules.
Only storable bi-propellant are considered.
No attitude and steering manoeuvres are considered

4.5.7.3 Baseline design 1st stage

Four improved pump feed version of the AESTUS engine has been chosen as propulsion system for the first stage.

The engine derives from the AESTUS pressure feed engine used in Ariane-5 upper stage. Recently this engine was proposed in a pump-feed version with an increase of the Isp performances and reduced system mass derived from the relaxed pressure tank operating system typical of pump feed engines.

The engine nozzle has been resized (shortened) for the Martian atmosphere with reduced performances in Isp and thrust level in comparison to the vacuum performances

The propulsion system presents the following characteristics:

Characteristic	Value
number of thrusters	4
number of tanks	2+2
Thrust	33 kN/each (restartable)
Isp	330 sec
exit diametre	1070
lengh	1730 mm
thruster mass	120 kg
propellant	MMH/NTO
O/F ratio	2.05
tank material	Ti
max MEOP	7 bar
Mass of UDMH tank	17.1 kg (each)
Mass of NTO tank	21.1 kg (each)

Table 4-64: Propulsion system for the first stage

Figure 4-121: AESTUS Engine

4.5.7.4 Baseline design 2nd stage

Four YUZHNOYE RD 869 pump-fed thruster have been chosen as propulsion system for this module. The thruster is under development for 4th stage of the European VEGA Launcher. The propulsion system presents the following characteristics.

Characteristic	Value
Number of thruster	4
Thrust	5 kN (pump- fed)
Isp	325 sec
exit diameter	325 mm
length	600 mm
thruster mass	34 kg
propellant	UDMH/NTO
O/F ratio	2.1
number of tanks	2+2
Tanks material	Ti
max MEOP	7 bar
Mass of UDMH tank	5 kg (each)
Mass of NTO tank	5.8 kg (each)

Table 4-65: Propulsion system for the second stage

4.5.7.5 Budgets

Propellant mass 2986 kg

Propulsion Dry mass (including margins) 267 kg

This mass includes an estimation of thrusters mass, the tanks, and a roughly estimation of feedlines, valves and regulators, propulsion thermal control, avionics, actuators and does not consider the structure of the propulsion system, power and communication.

4.5.8 Environmental control and life support system

The life support comprises the following subsystems

- Atmosphere Supply and Control
- Atmosphere Revitalization
- Temperature and Humidity Control
- Water Management
- Waste Management
- Food Management
- Safety
- EVA Provisions
- Hygiene
- Crew Accommodations

Crew accommodations have been added to the classical set of life support functions as the crew accommodation engineering domain does not posses a separate workbook in the CDF study, in which hardware specifications could be added.

4.5.8.1 Requirements and design drivers

The MEV complex consists of two main modules. The Surface Habitation Module (SHM) and the Mars Ascent Vehicle (MAV). The study required both modules to be equipped with life support systems, which are not interconnected.
The MAV life support system is designed to provide life support to a crew of three for 5 days. The mission parameters and the size of the crew allow this study to use the Soyuz TM(A) design as a reference point.

4.5.8.2 Assumptions and trade-offs

Metabolic Requirements of the Crew
The metabolic needs of the crew have been calculated using the correlations given in ESA standard PSS-03-406 and crosschecked with relevant sources. The entire calculations have been based on the energy expenditure of the crew. The schedule for crew activity is shown in Figure 4-122.

SCHEDULE IN HOURS FOR THE MOST ACTIVE DAY			
ACTIVITY	ASTRONAUT		
	1	2	3
sleep	8	8	8
pre- and post sleep	5	5	5
leisure activities	0	0	0
personal hygiene	1	1	1
eating	2	2	2
exercise	0	0	0
station keeping	8	8	8
laboratory activities	0	0	0
Metabolic Cost of EVA			
EVA mission tasks	0	0	0
EMU donning/doffing	0	0	0
egress/ingress	0	0	0
pre-EVA setup & post EVA EMU care	0	0	0
TOTAL TIME (24hrs)	24	24	24

Figure 4-122: Crew Activity Schedule during Martian surface Stay

Based on the energy expenditure, the metabolic needs and products by the crew have been estimated and are shown in Figure 4-123.

	PER DAY	PER MISSION
ENERGY CONSUMPTION (W*h)	8512.8	35753.8
ENERGY CONSUMPTION (J)	30646080	128713536
OXYGEN CONSUMPTION (m3)	1.5	6.4
OXYGEN CONSUMPTION (kg)	2.1827	9.1675
DRINKING WATER (m3)	0.0051	0.0214

	PER DAY	PER MISSION
DRINKING WATER (kg)	**5.1**	**21.4**
DRY FOOD (kg)	**2.0**	**8.5**
CARBON DIOXIDE PRODUCTION (m3)	**1.3**	**5.3**
metabolic water production (kg)	0.9	3.8
URINE PRODUCTION (kg)	**4.6**	**19.3**
FAECAL LIQUIDS (kg)	**0.13**	**0.6**
INSENSIBLE WATER (kg)	**4.6**	**19.3**
TOTAL SOLID WASTE PRODUCTION		
FAECES (kg)	**0.2**	**0.9**

Figure 4-123: Metabolic needs and products of the crew

The data presented in this figure suggest a mass of consumables of about 503kg. Taking into account that consumables need additional hardware for storage and use, as well as the need to treat and store the metabolic products, the use of an open loop system seems favourable. The data strongly suggests the use of open loop systems except a recovery system for condensate.

Additional considerations

4.5.8.2.1 Hygiene water

There is no hygiene water allowance during this phase of the mission.

4.5.8.2.2 Drinking water

The drinking water allowance for the crew has been estimated to be 1.7 l/crew/d and was oriented on the drinking water allowance on-board a Soyuz TM(A) vehicle.

4.5.8.2.3 Cabin atmosphere

The cabin atmosphere has been selected as follows:

Total Cabin Pressure: 50.0 kPa
Partial Pressure Oxygen: 25.0 kPa
Partial Pressure Nitrogen: 25.0 kPa
Partial Pressure Carbon Dioxide: TBD

Preferably, the atmosphere would be free of any contaminants. However, as a minimum requirement, the spacecraft atmosphere shall adhere to the requirements given in ESA PSS-03-401. Based on the experiences with long term pressurised spacecrafts there shall be more stringent limits on microbial contamination. Following limit has been proposed during this study:

Total microflora count: 200CFU/m3 (CFU - colony forming units)

4.5.8.2.4 EVA considerations

No EVAs are to be performed during this mission phase.

4.5.8.2.5 Waste production

The produced waste is stored in waste storage canisters. The degree of stabilization has to be sufficient to guarantee

4.5.8.2.6 Waste management strategy

Waste is stored in on-board containers. Stabilization is not a major issue due to the short duration before disposing of the MAV spacecraft.

4.5.8.3 Baseline design

The design is based on the Soyuz-TM(A) life support system design.

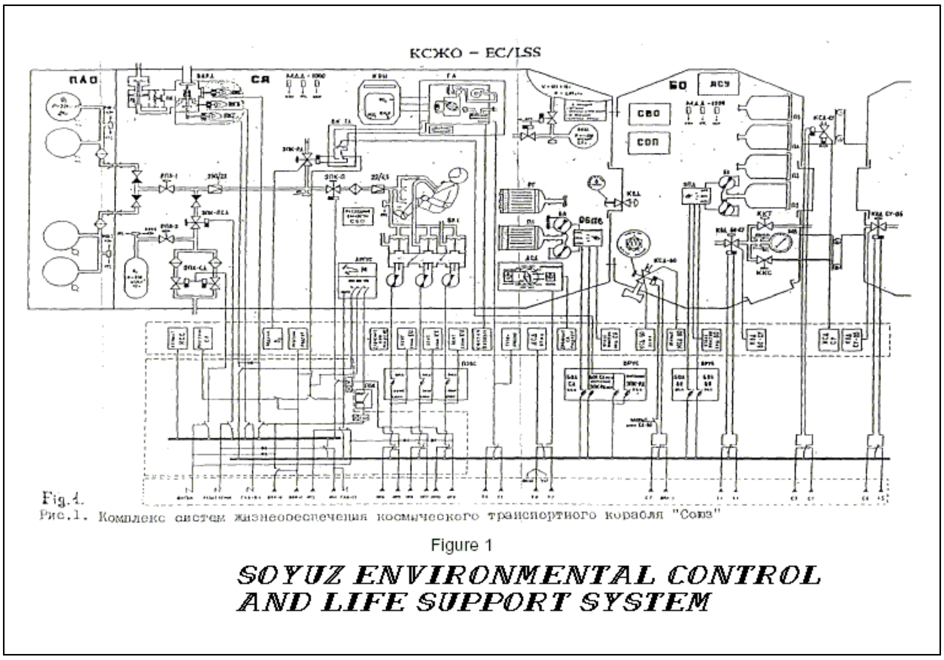


Figure 4-124: Mars MAV LSS design

ECLS system mass:

CONSUMABLES TO BE LAUNCHED (kg)	
OXYGEN	10.0
NITROGEN	0.0
POTABLE WATER	21.0
HYGIENE WATER	0.0
DRY FOOD	8.5
PACKAGING	4.0
INORGANIC MATERIAL ECXL. PACKAGING	2.0
TOTAL CONSUMABLES TO BE LAUNCHED	45.5
WASTE PRODUCTION DURING MISSION (kg)	
WASTE GASES	5.3
WASTE WATER	39.2
SOLID ORGANIC WASTE	0.9
SOLID INORGANIC WASTE excl. packaging	2.0
PACKAGING	4.0
TOTAL WASTE PRODUCED	51.4
ROUGH ESTIMATE ECLSS MASS (kg)	
TOTAL ECLSS SYSTEM MASS	549.0

Table 4-66: Mass Estimates for a Mission using Current Technology

The life support system has been estimated to have an approximate mass of 549 kg. The detailed mass budget is given in Table 4-67.

Equipment	Number of units	Mass per units (kg)
GENERIC PIPING 1/4"X1m	5.00	0.13
ISS PFE - portable fire extinguisher	1.00	15.10
ISS smoke detector	1.00	1.50
SOKHOL spacesuit	3.00	12.00
SOYUZ AIR PURIFICATION UNIT	4.00	10.00
SOYUZ ARGUS PRESSURE SENSOR	1.00	3.50
SOYUZ AUTOMATIC PRESSURE CONTROL UNIT	1.00	7.50
SOYUZ CHX	2.00	10.00
SOYUZ CONDENSATE STORAGE TANK	3.00	15.00
SOYUZ FAN	2.00	2.00
SOYUZ GAS ANALYZER	1.00	20.00
SOYUZ GASEOUS O2 STORAGE SYSTEM (12l)	1.00	10.00
SOYUZ GASEOUS O2 STORAGE SYSTEM (20l)	4.00	18.00
SOYUZ HAND PUMP	1.00	1.00
SOYUZ LIGHTING (20cm fluorecent)	2.00	1.00
SOYUZ LIGHTING (spot panel lighting)	13.00	0.20
SOYUZ O2 TANK PRESSURE SENSOR	2.00	2.00
SOYUZ O2 TANK WALL TEMPERATURE SENSOR	5.00	2.00
SOYUZ ONBOARD TANK	3.00	19.60
SOYUZ OXYGEN SUPPLY FITTINGS	17.00	4.00
SOYUZ personal hygiene kit	3.00	1.50
SOYUZ PORTABLE SURVIVAL KIT	1.00	15.00
SOYUZ PRESSURE ALARM SENSOR	1.00	0.72
SOYUZ PRESSURE EQUALIZATION VALVE	1.00	6.00
SOYUZ PURIFICATION CARTRIDGE (REGENERATOR)	1.00	10.00
SOYUZ RECEVING DEVICE	3.00	0.30
SOYUZ REFILLING VALVES	1.00	3.40
SOYUZ RELIEF VALVE	1.00	6.70
SOYUZ RESCUE AID AUTOMATIC UNIT	1.00	5.00
SOYUZ SAFETY UNIT	1.00	1.40
SOYUZ SOKHOL SUIT BLOWER ASSEMBLY	3.00	3.90
SOYUZ SOLID WASTE STORAGE CANISTER	1.00	10.00
SOYUZ SUIT FAN CIRCUIT BRAKER PANEL	1.00	1.00
SOYUZ TOILET	1.00	50.00
SOYUZ VALVE INHIBIT PANEL	1.00	1.00

Table 4-67: Detailed mass budget for the anticipated life support system

The life support system may not be considered exhaustive. It is merely a list of major components, which give an indication of what LSS mass has to be anticipated.
Note that the list includes hardware based on life support and crew accommodation needs.

4.5.8.4 Budgets

ECLSS mass requirements

Two system options were presented with following mass budgets:

Today's Technology	
Mass consumables (kg)	46
Mass system (kg)	549

Table 4-68: Mass budgets

ECLSS power requirements
The results are shown in Table 4-69.

Characteristic	Value
Power requirement day (W)	375
Power requirement night (W)	375

Table 4-69: Power budgets

ECLSS volume requirements
Only a first estimate for the volume of the life support system has been achieved in the course of the study. The internal volume requirement pertains to the volume occupied by the ECLSS inside the pressurised vessel, as opposed to the external volume requirement, which pertains to the volume needs outside the pressurised volume.

Characteristic	Value
Internal volume requirements (m3)	1.4
External volume requirements (m3)	0

Table 4-70: Volume requirements

4.5.9 Data handling

See 4.3.6.

4.5.10 Communications

4.5.10.1 Requirements and design drivers

- The vehicle shall support Tracking, Telemetry and Command (TT&C) communications during all mission phases and any attitude.
- Communications availability should be maximized.
- The telecommand (TC) and telemetry (TM) data rates shall be selectable to improve the data rate depending on the distance to the receiving unit.
- Data consists of housekeeping, audio, and any additional data.

4.5.10.2 Baseline

4.5.10.2.1 Links

During Mars ascent phase, similarly to the DM the communications will be maintained using an *UHF link* with TV and a *X-band* link with the relay satellite and therefore with Earth G/S. UHF link will be the same than for DM and the same antenna will be used. As in DM, UHF and X-band MAV transponders are used. The only difference is the location of the three X-band patch antennas, which will be in the MAV capsule chassis.

• MAV patch antenna • Relay antenna: 1m with steering mechanism.		
	Uplink	***Downlink***
Frequency	7.23 GHz	8.5 GHz
Tx power	65 W	65 W
Modulation	QPSK	QPSK
Coding	Concatenated, Interleaving=5	
FER	10^{-5}	
Bit rate: ***Max distance 18600 Km*** ***Min distance: 16530 Km***	 172 Kbps 97 Kbps	

Table 4-71: X-band link MAV-Relay satellite.

4.5.10.2.2 Contingency communications

Direct communications with the Earth could be possible using the X-band patches antennas, but with a very low data rate. See TV chapter for data rates and link schemes.

4.5.10.3 Budgets

Unit	***Number of units***	***Unit mass (Kg)***	***Total mass (Kg)***	***Power (W)***
UHF omni antenna	1	1.5	1.5	
X-band patch antenna	3	0.1	0.3	
UHF transceiver	2	1.0	2	10.0
X-band transponder	2	4.6	9.2	20.0
Global RFDU unit	2	1.2	2.4	
X-band TWT	2	0.8	1.6	120.0
Harness			2.5	
Total:			*19.5*	*150.0*

Table 4-72: MAV communications budget

5 OVERALL CONCLUSIONS

A design case for a Human Mission to Mars has been analysed. Although this does not represent a " reference" ESA mission, it contains several design elements of general applicability.
The understanding of the main technical issues and the relevant design elements will allow future definition of a reference mission and a more comprehensive exploration plan.
In particular, the issues of life support, radiation, long permanence in space, internal habitats and overall vehicle configurations, entry descent and landing, Martian surface operations, assembly in Earth orbit, etc., as far as the selected design case is concerned, have been tackled in this study and design solutions proposed.
Several simplified models have been created to deal with the issues and allow sensitivity analysis of the main mission parameters.
Whenever possible, preference in the design has been given to existing technologies or those considered within reach in relatively short time. This is to achieve results that can be trusted in this phase and to not rely on speculations on performance.
A few general conclusions can be drawn from the exercise:

- Even the simplest mission based on very limited functions and capability leads to extremely large and massive vehicles and requires assembly in Earth orbit before departure.
- The most critical technical showstopper for such a mission is the overall vehicle *assembly time* in LEO that could result in unacceptable phasing of subsequent missions and lead to unacceptable ageing before departure.
- A design point exists for an entirely " chemical" mission (e.g. all based on chemical propulsion). However, this gives a rather high mass in LEO (above 1000 tonnes) and as a consequence, high time of assembly in LEO.
- Launcher availability is critical. The study assumed that a launcher with the performance of Energia would be available for most of the launches. If this assumption is wrong, a very high penalty on the mission is expected.
- High closure of the life support system (e.g. recycling) is a must. The penalty associated with an open system would be too big for such a mission.
- The reason for the high overall mass of the mission stems from the very large dry mass of the Transfer Habitation Module and the relative inefficiency of the chemical propulsion.
- Among the possible alternatives not requiring technology leaps, aerobraking and aerocapture have been briefly investigated. It has been discovered that the implementation of these techniques will require large changes in the vehicle designs as compared to the chemical case. The detailed analysis of these options was considered outside the of this first study and will be performed in later phases.
- The verification of safety requirements has proven impossible without an overall risk model. However, mission abort cases have been investigated and the design has taken into account failure cases to a certain extent. Failures in the propulsion system cannot be recovered without unacceptable penalty on the mission; therefore systems with very high reliability need to be implemented.

As already mentioned, the design case analysed represents an oversimplified mission. Among the limitations of this approach, the following should be emphasised:

- Permanence on the surface has been limited to about 30 days to allow simplification in the design and the associated models. However, such a short duration is unlikely to be selected within the frame of a planet human exploration programme.
- Links to the overall exploration strategy and the other associated missions has not been pursued. Therefore, synergies with other missions have not been exploited at this stage.
- The Earth Return Capsule has not been designed.

5.1 Technology development

Conservative assumptions on technology availability and performance have been used throughout the whole exercise in most cases. Even so, major technology developments are needed to achieve this mission case.
Among those, some enabling ones are:

- Closure of the Life Support subsystem
- Implementation of micro-gravity countermeasures
- Techniques for reduction of boil-off in cryogenic propulsion system (for the chemical option)
- Ground and Space System infrastructures for very high data rate telecommunications and mission support
- Entry descent and landing systems for very large arrival masses
- Automatic assembly techniques in LEO
- Fuel cells
- Advanced avionic systems and architectures

Clearly the development of such technologies cannot be limited to ESA only and will require a trans-national effort.
As a result of this study, performance requirements for these technologies can be now set and programmatic assessment can be performed.

6 APPENDIX A - MARTIAN SURFACE NUCLEAR REACTOR

In the frame of the present study, the option of using a nuclear fission reactor to provide power for the surface operations was considered in an early stage of the analysis. The design of the reactor was primarily based on recent European space reactor studies.

Given the short surface stay of the proposed mission and the general conservative approach chosen, the reactor option was considered not appropriate and is given here for completeness sake and in the light of further studies involving longer surface stay times.

6.1 General parameters and initial assumptions

It is assumed that the nuclear fission reactor

- serves (only) to deliver electrical power;
- delivers 50 kW_e (nominal);
- has a lifetime of about 10 years (minimum of 6 years at operational power);
- complies with international legal standards (especially 1992 UN COPUOS principles and ICRP recommendations) in terms of operations, safety and radiation protection;
- is activated only after installation on the Martian surface (apart from zero-power testing on Earth), (no operation during cruise);
- only one reactor is delivered – no system level redundancy.

As regards the total surface power of 50 kWe, note that the final power need of the surface stay of the current mission scenario is only 3500 Wh/day, which corresponds to an average of 145W. The power level of 50 kW_e for the reactor assessment was made earlier and was maintained essentially due to two reasons: there is a certain minimum power level for fission reactor systems to become interesting in terms of specific power, furthermore, fission reactor power systems as not scale linearly with the power levels provided and even strong decreases of the power need might not change the total power system mass significantly.

6.2 Reactor type, mass and sizing aspects

6.2.1 General parameters

Compared to terrestrial and naval nuclear reactors, space reactors are orders of magnitude smaller, in size as well as in power. While the basic principles remain identical – nuclear core sustaining the controlled chain reaction, heat transport system, electricity generation from heat – some subsystems are significantly different: waste heat rejection system (e.g. absence of abundant water), natural coolant/moderator fluent circulation due to gravity, emergency systems based on gravity.

6.2.2 Main differences to terrestrial systems

Furthermore, the total mass of the systems is one of the key parameters for space reactors, while virtually non-existent for designing classical terrestrial systems (except submarine reactors). To a smaller extent this also holds true for its shape and size.

As a result of the above mentioned parameters, space reactor designs show higher reactor core temperatures and operate with much higher enriched ^{235}U fuel. This leads to lower core dimensions and mass, reduced radiator size and mass and higher efficiencies of the conversion system. In numbers, the enrichment increases from natural-20% for terrestrial up to 93% for space systems, and the core temperature increases from 400-573 K up to 900-1000 K. Technically these changes imply the abandon of the pressurised water reactor design for either liquid metal cooled or gas cooled cores. They furthermore demand the use of materials capable of withstanding for long times high temperatures as well as high radiation (neutron) fluxes.

6.2.3 Approach

Given the high complexity of designing space nuclear reactors and especially the highly interrelated subsystem dependencies, it was chosen to not develop a parametrical reactor model with completely open input parameters but to base the assessment on two recently proposed space reactor models.

For the present study, the two designs that are taken as reference were recently proposed by European nuclear industry within an ESA contract especially for Martian surface power generation purposes. They are both available to a technical design level sufficient for the present study and correspond to the power level as well as lifetime requirements.

6.3 Reactor power subsystems

Some aspects of different reactor power system components are provided. The purpose of this small section is to provide some basic elements to understand the choices and implications of the reactor designs proposed.

Reactor core

- Thermal n core: larger (if small: limited core life due to burn-up), moderator, good negative power coefficient (safety), rather for larger than10 MW_e reactors
- Fast n core: enriched fuel, smaller, more stable power distr., low burn-up, small negative power coefficient (safety), favourable in 100 kW_e range

Conversion system

Static

Thermoelectric:	mature, space proven; about 4% (adv.: cascades about7%)
Thermionic:	mature, in core vs. out core systems, life-time limiting factors, space reactor proven
AMTEC:	immature, different for ionic and electronic condition, short lifetimes, corrosion problems, 19-25%

MHD:	immature, ionic gas flow in B field, very high T, 30%
ThermoPV:	immature, high emitter T (larger than 2200K) + low PV temp, 30%

Dynamic

Brayton cycles:	He or He/Xe working gas, no corrosion problems, 1 or 2 cycles, 1500K – 550K, 20% eff.
Rankine cycles:	state change (evap-condes.), higher eff.; alkali metals (Po, N, toluene)
Stirling cycles:	kinematic or free piston, 1100K-650K, 23.5% eff., sealing integrity

Shielding

n shielding:	metal hydrides (LiH), usually contained in stainless steel (for structure and protection)
γ shielding:	Tungsten (alternatives: e.g. depleted Uranium, borated steel, Pb-W-LiH)

Shielding is for the considered type of reactor in case of close human operations certainly the most massive subsystem. Requirements dependent on mission design (transfer phase operation? partial (angular) shielding?)

Power management

Voltage issues (to decrease losses)
Cabling to overcome distance to reactor – trade-off between the mass of cabling versus mass of additional shielding in case of closer distances;

Chosen European reactor designs

The two designs retained by European industry are the liquid metal-cooled thermal core with a conservative thermoelectric conversion unit and thermal radiators as cold well and the slightly more advanced gas-cooled fast neutron particle bed reactor connected to a dynamic Brayton conversion unit using forced convection of Martian atmosphere for waste heat rejection.

1. Liquid metal cooled thermoelectric conversion reactor (LMR)
    Thermal neutron core
    ZrH_2 moderator
    Moderator/fuel ratio: 3
    NaK coolant (22/78)
    Thermoelectric conversion + radiator

2. Gas-cooled Brayton-cycle particle bed reactor (GCR)
    Fast neutron core
    Particle bed design (1mm diam. fuel particles, 93% enriched U235)
    He or He/Xe coolant (1^{st} cycle)
    Use of Martian CO_2 for cooling (2nd cycle)

The most fundamental parameters of both designs are listed and compared in Table 6-1.

	Liquid Metal cooled thermoelectric reactor	Gas-cooled particle bed Brayton reactor
Electric power [kW]	50	50
Thermal power [kW] (efficiency in %)	1250 (4%)	185 (26.9%)
Core mass (fuel mass) [kg]	186 (54)	1075 (93) (rad refl. Be: 622 kg) (axial refl. BeO: 218 kg)
Power conversion [kg]	371-712	340
Radiator mass [kg]	718 (steel: 618 kg) (mercury: 100 kg)	n/a
Packaging [kg]	111-180	250
Total Mass [kg]	1386-1796	1665
Size (diam/height) [m]	Core: 0.45/0.6 Conv.Syst: 0.5/0.4	Core: 0.8(0.17)/1.5 Conv.Syst: 0.8/1.2

Table 6-1: Comparison of basic reactor design choices

6.4 Radiation protection and shielding issues

Both designs make use of Martian regolith for additional shielding purposes. For the purpose of this preliminary assessment, an all-side shielding requirement is assumed together with an acceptable dose limit at about 100 metres from the reactor site. According to preliminary calculations such a shield would need about 10 tonnes of Martian regolith, distributed in an about 5 metre radial layer and an about 3 metre axial (assuming a cylindrical reactor core) layer. Options of using locally produced binding materials and deepening the core into an (artificial) hole need to be further explored.

An example for a buried reactor core with a subsurface heat rejection unit is shown in Figure 6-1.

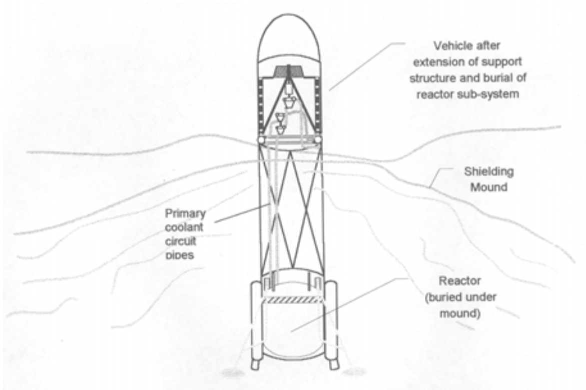


Figure 6-1: Example of buried reactor core.

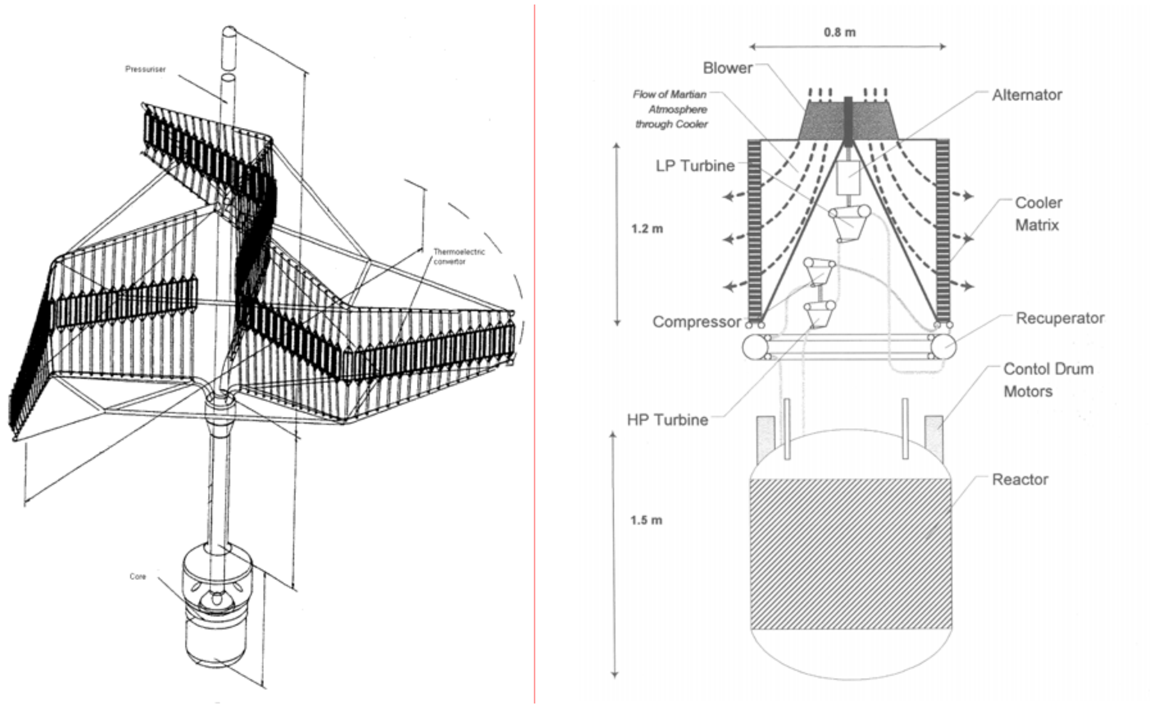


Figure 6-2: LM reactor design (L) and GC reactor design (R)

The liquid metal cooled reactor (LMR) with thermoelectric power conversion and radiative head rejection system was considered more conservative than the gas-cooled particle bed rector (GCR) with a Brayton cycle and forced convection waste head removal using the Martian atmosphere. The LMR was thus chosen as the prime choice, leaving the GCR design as the more advanced alternative.

6.5 Reactor operation

6.5.1 Pre-launch operations

During the pre-launch phase, the reactor core would undergo 0-power testing to verify the neutronics of the core. The other subsystems would be tested independently from the core operations. At launch, there would not be any significant amount of fission products present.
The reactor would not be made critical during the cruise phase.

6.5.2 Start-up

Once the reactor would be safely landed on the surface, shielding construction would need to be undertaken. Different options, including robotic ones are proposed but need further assessment. To a certain degree depending on the final design of the reactor, some sort of assembly might be required.
Reactor start-up still needs some degree of further investigations. Different options are possible. It is assumed that the start-up would be highly automatised and last of the order of several hours.

6.5.3 Nominal operations

During nominal operations, no human intervention should be necessary. The power output would probably not be adjustable, but permanent and constant.

6.5.4 Emergency operations

There certainly is a need for a possibly human intervention in reactor operations at any time. The minimum intervention options will include at least an immediate emergency shutdown, in which case the system would have enough redundancy for continued heat removal from the core.

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8 APPENDIX C - ACRONYMS

AAA	Avionics Air Assembly
ACS	Attitude Control System
AFC	Alkaline Fuel Cell
AIU	Avionics Interface Unit
AIV	Assembly, Integration and Verification
ALS	Advanced Life Support
APM	Antenna Pointing Mechanism
ARA	Adavanced Rigid Arrays
ATV	Automatic Transfer Vehicle
BCR	Battery Charge Regulator
BDR	Battery Discharge Regulator
BER	Bit Error Rate
BFO	Blood Forming Organs
CAD	Computer Aided Design
CAM	Collision Avoidance Manoeuvre
CDF	Concurrent Design Facility
CFC	Chlorofluorocarbons
CFU	Colony Forming Units
CHX	Condensing Heat Exchanger
CIS	Copper Indium Diselenide
CMG	Control Moment Gyros
COF	Columbus Orbital Facility
COP	Coefficient of Performance
CPL	Capillary Pump Loop
CPU	Central Processing Unit
CSS	Coarse Sun Sensor
CTV	Crew Transportation Vehicle
DAK	Double Aluminized Kapton
DAM	Double Aluminized Mylar
DCS	Decompression Sickness
DGK	Double Goldenized Kapton
DHS	Data Handling System
DLS	Descent and Landing System
DMM	Distributed Memory Modules
DNA	Deoxyribonucleic Acid
DOR	Delta Of Range
DRS	Dataq Relay Satellite
DSM	Deep Space Manoeuvre
EAC	European Astronaut Centre
EAP	Etage a Pondre
ECU	Electronic Control Units
EDL	Entry Descent and Landing
EEE	Electrical, Electronic, and Electromechanical
EMU	Extravehicular Mobility Unit
EOI	Earth Orbit Insertion

EOL	End Of Life
ERA	European Robotic Arm
ERC	Earth Return Capsule
ESA	European Space Agency
ETO	Earth Transfer Orbit
EVA	Extra Vehicular Activity
FER	Frame Error Rate
FOV	Field Of View
FPA	Flight Path Angle
GCR	Galactic Cosmic Rays
GEO	Geostationary Earth Orbit
GNC	Guidance, Nivigation and Control
GPS	Global Positioning System
HAB	Habitation Module
HEO	High Eliptic Orbit
HGA	High Gain Antenna
HMM	Human Mission to Mars
IBD	Inflatable Breaking Device
IEEE	Institute of Electrical and Electronic Engineers
IMU	Inertial Measurement Unit
IMV	Inter Moule Ventilation
IPN	InterPlanetary Internet
ISO	International Organisation for Standardisation
ISS	International Space Station
ITU	International Telecommunications Union
JPL	Jet Propulsion Laboratory
JSC	Johnson Space Centre
LAN	Local Area Network
LAT	Latitude
LEO	Low Earth Orbit
LGA	Low Gain Antenna
LMO	Low Mars Orbit
LMR	Liquid Metal cooled Reactor
LOX	Liquid Oxygen
LSS	Life Support System
MAV	Mars Ascent Vehicle
MCC	Mission Control Center
MCFC	Molten Carbonate Fuel Cell
MDM	Mars Descent Module
MEO	Medium Earth Orbit
MER	Mars Eploration Rover
MEV	Mars Excursion Vehicle
MEX	Mars Express
MGA	Medium Gain Antenna
MGS	Mars Global Surveyvor
MHD	Magnetohydrodynamic
MIL	Military
MIR	Soviet Unions Space Station

MLI	Multi-Layer Insulation
MMH	Mono-Methyl Hydrazine
MOD	Meteorite and Orbital Debris shield
MOI	Mars Orbit Insertion
MSR	Mars Sample Return
NCG	Non-Condensable Gas
NEP	Nuclear Electric Propulsion
NTO	Nitrogen TetrOxide
NTP	Nuclear Thermal Propulsion
OBC	On-Board Computer
OCC	Operations Control Center
ODP	Ozone Depletion Fluid
ODS	Orbital Disconnect Struts
OEM	Orbiting Around Earth Mode
OSR	Optical Solar Reflector
PAFC	Phosphoric Acid Fuel Cell
PCA	Pressure Control Assembly
PCI	Peripheral Component Interconnect
PCU	Power Control Unit
PDU	Power Distribution Unit
PEM	Proton Exchange Membrane
PEMFC	Proton Exchange Membrane Fuel Cell
PFE	Portable Fire Extinguisher
PFM	Proto-Flight Model
PLL	Phase Locked Loop
PMS	Polymathyl Siloxane
PVR	Photovoltaic Radiator
RAM	Random Access Memory
RCS	Reaction Control System
RSA	Russian Space Agency
RTU	Radio Thermal Unit
RVD	RendezVous and Docking
RWA	Reaction Wheel Assembly
RWL	Reaction Wheels
SEE	Single Event Effects
SEM	Sun Earth Mars
SEP	Solar Electric Propulsion
SEU	Single Event Upset
SHM	Surface Habitation Module
SOFC	Solid Oxide Fuel Cell
SPE	Single Particle Event
SRC	Short Arm Centrifuge
SRM	Solid Rocket Motor
SSM	SuperAmine
SSO	Sun Synchronous Orbit
STD	Standard
STM	Structural Thermal Model
STR	Star Tracker

STS	Space Shuttle
TBC	To Be Confirmed
TBD	To Be Determined
TCP	Tansport Control Protocol
TCS	Thermal Control System
TEI	TransEarth Injection
TFG	Telemetry Transfer Frame Generators
THM	Transfer Habitation Module
TMI	TransMars Injection
TPS	Thermal Protection System
TRL	Technology Readiness Level
TRP	Basic Technology Research Programme
TTC	Telemetry Tracking and Control
TVC	Thrust Vector Control
TVI	TransVenus Injection
TWT	Travelling Wave Tubes
UHF	Ultra High Frequency
USB	Universal Serial Bus
VHF	Very High Frequency
VME	VersaModule Eurocard
ZBO	Zero Boil Off