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Space Exploration Systems, Strategies and Solutions

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Il dott. Maria Antonietta Viscio ha discusso in data 21-03-2014 presso il DIMEAS del Politecnico di Torino la tesi di Dottorato avente il seguente titolo:

Space exploration systems, strategies and solutions

Le ricerche oggetto della tesi sono interessanti, di attuale interesse per la comunità scientifica e foriere di sviluppi futuri in ambito industriale.

Le metodologie appaiono adeguate allo scopo della ricerca.

I risultati sono interessanti, con applicazioni a vari sistemi di esplorazione, ed analizzati con senso critico.

Nel colloquio il candidato dimostra ottima conoscenza delle problematiche trattate.

La Commissione unanime giudica positivamente il lavoro svolto

e propone che al dott. Maria Antonietta Viscio venga conferito il titolo di **Dottore di Ricerca**.

Data 21-03-2014

Prof. Franco Bernelli Zazzera (Presidente)

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Summary

The present thesis describes the PhD research activities dealing with the topic *"Space Exploration Systems, Strategies and Solutions"*.
Traveling beyond low Earth orbit is the next step in the conquest of the solar system and so far, a human expedition to Mars is considered the most interesting goal of future human space exploration. Due to the technological and operational challenges associated with such a mission, it is necessary to define an opportune path of exploration, relying on many missions to intermediate and "easier" destinations, which would allow a gradual achievement of the capabilities required for the human Mars mission.
The main scope of this research has been the development of a rigorous and versatile methodology to define and analyze evolutionary exploration scenarios and to provide a detailed technologies' database, to support strategic decisions for human space exploration. The very innovative aspect of this work regards the development of a flexible methodology which can be followed to assess which are the next destinations for the exploration of space beyond LEO and to preliminarily define mission's architectures, identifying the most significant needed elements and advanced technologies. The obtained results should be seen as a pure technical reference, as no cost and/or political considerations have been included, and can be exploited to opportunely drive the decisions of the agencies to place investments for the development of specific technologies and get ready for future exploration missions.
The first part of the work has been devoted to the definition of a reference human space exploration scenario, which relies on both robotic and human missions towards several destinations, pursuing an increasing complexity approach and looking at a human expedition to Mars as final target. The scenario has been characterized through the assessment of the missions and

the relative phases and concepts of operations. Accordingly, the needed space elements, or building blocks, have been identified. In this frame, the concept design of two specific elements has been performed: the first is a pressurized habitation module (Deep Space Habitat) for hosting astronauts during deep space missions; the second is an electrical propulsive module (Space Tug), mainly envisioned for satellites servicing. The last part of the work has focused on the analysis of innovative and enabling technologies, with particular attention to the aspects related to their on-orbit demonstration/validation, prior to their actual implementation in real exploration missions.

The PhD has been sponsored by Thales Alenia Space - Italy and the overall work has been performed in different frameworks along the three years, as well as participating to several additional activities.

In line with the objectives of the PhD, in 2012 a collaboration between Politecnico di Torino and Massachusetts Institute of Technology has been established (MITOR Project, managed by MIT-Italy Program), with the support of Thales Alenia Space as industrial partner. The MITOR project, titled "Human Space Exploration: from Scenario to Technologies", has been aimed at identifying and investigating state of the art for Human Space Exploration, enabling elements, subsystems and technologies with reference to a selected scenario and relevant missions and architectures. Part of the nine months activities has been carried out at MIT AeroAstro department.

Besides MITOR project, the PhD activities have been carried out in synergy with some other research programs, such as ESA "Human Spaceflight & Exploration Scenario Studies" and STEPS2 project (Sistemi e Tecnologie per l'EsPlorazione Spaziale - phase 2).

Furthermore, in 2013 a specific study has been performed in collaboration with university "La Sapienza" (Rome), "Osservatorio Astrofisico di Torino" (Astrophysical Observatory of Torino) and DLR (Deutsches Zentrum fr Luft- und Raumfahrt) in Bremen; its main objective has been the analysis of an interplanetary cubesats mission, aimed at space weather evaluations and technologies demonstration.

Sommario

La presente tesi descrive le attività svolte nell'ambito del Dottorato di ricerca sulla tematica "Sistemi, Strategie e Soluzioni per l'Esplorazione Spaziale" ("Space Exploration Systems, Strategies and Solutions").

Viaggiare oltre l'orbita bassa terrestre è il prossimo passo da affrontare nella conquista del sistema solare e in particolare, una missione umana su Marte è considerato l'obiettivo più interessante per la futura esplorazione spaziale umana. A causa delle difficoltà tecnologiche e operative associate a una tale missione, è necessario definire un opportuno percorso di esplorazione, caratterizzato da diverse missioni verso destinazioni intermedie e "più semplici" per garantire un raggiungimento graduale delle capacità richieste per una la missione marziana.

L'obiettivo principale di questa ricerca è stato quello di sviluppare una metodologia rigorosa e versatile per la definizione e l'analisi di scenari per l'esplorazione spaziale, e di fornire un database dettagliato delle tecnologie abilitanti per le future missioni di esplorazione, a supporto di decisioni strategiche sia a livello di scenari e missioni, sia a livello di tecnologie. L'aspetto più innovativo di questa ricerca riguarda lo sviluppo di una metodologia flessibile da utilizzare per identificare quali sono le future destinazioni per l'esplorazione spaziale umana oltre l'orbita terrestre, e definirne le relative architetture di missione, attraverso l'analisi dei moduli spaziali necessari per lo svolgimento delle varie missioni, nonché delle tecnologie innovative ed abilitanti. I risultati ottenuti rappresentano un riferimento puramente tecnico (non sono state difatti incluse considerazioni di tipo politico/economico), che può essere sfruttato per indirizzare opportunamente investimenti, a livello strategico, per lo sviluppo tecnologico a supporto dellesplorazione futura.

La prima parte del lavoro è stata dedicata alla definizione di uno scenario

di riferimento che ha come obiettivo finale una missione umana su Marte e composto di un certo numero di missioni, sia robotiche che umane, verso destinazioni intermedie, definite in modo tale da garantire uno sviluppo tecnologico graduale attraverso missioni e sistemi di complessità crescente. Le varie missioni da includere nello scenario sono state caratterizzate tutte in dettaglio, in termini di strategia, profilo e architettura di missione. Sono stati inoltre individuati i moduli spaziali necessari e le tecnologie abilitanti associate.
In questo ambito, è stato fatto il design concettuale di due specifici moduli: il primo è un modulo pressurizzato (Deep Space Habitat) per ospitare e supportare gli astronauti durante le missioni di esplorazione; il secondo è un veicolo propulsivo (space tug) che ha come obiettivo principale quello di supportare il dispiegamento di satelliti in orbita.
La parte finale della ricerca si è focalizzata sull'analisi delle tecnologie abilitanti, con particolare attenzione agli aspetti relativi alla dimostrazione/validazione in orbita, prima della loro implementazione in missioni reali di esplorazione.
Il Dottorato di ricerca è stato finanziato da Thales Alenia Space - Italia e si è svolto in diversi contesti, partecipando a svariate attività aggiuntive.
In linea con gli obiettivi del Dottorato, nel 2012, in collaborazione con il Massachusetts Institute of Technology e con il supporto di Thales Alenia Space come partner industrial, è stata svolta un attività di ricerca (progetto MITOR) sul tema "Human Space Exploration: from Scenario to Technologies". Questa attività ha avuto come obiettivo un'analisi dello stato dell'arte dell'esplorazione spaziale umana, elementi e tecnologie abilitanti, con riferimento a uno specifico scenario e relative missioni. Una parte dell'attività è stata svolta presso il dipartimento di ingegneria aeronautica e astronautica (AeroAstro) del MIT.
Oltre al progetto MITOR, le attività di Dottorato sono state svolte in sinergia con altri programmi di ricerca, come lo studio dell'ESA "Human Spaceflight & Exploration Scenario Studies" e il progetto regionale STEPS2 (Sistemi e Tecnologie per l'EsPlorazione Spaziale - fase 2).
Infine nel 2013, è stato portato avanti uno studio in collaborazione con l'università "La Sapienza"di Roma, l'"Osservatorio Astrofisico di Torino" e

il DLR (Deutsches Zentrum fr Luft- und Raumfahrt); l'obiettivo principale è stato l'analisi di una missione interplanetaria basata su piccoli satelliti (CubeSats) finalizzata alla la dimostrazione in orbita di specifiche tecnologie (es. vele solari, sistemi di comunicazione ottica, protezione da radiazioni spaziali).

To my loving husband

“In spite of the opinions of certain narrow-minded people, who would shut up the human race upon this globe, as within some magic circle which it must never outstep, we shall one day travel to the moon, the planets, and the stars, with the same facility, rapidity, and certainty as we now make the voyage from Liverpool to New York!”

Jules Verne, From the Earth to the Moon, 1865

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Acronyms

AC Aerocapture

AV Ascent Vehicle

BB Building Block

BER Bit Error Rate

BML Big Manned Lander

CCM Contingency Consumables Module

CEV Crew Exploration Vehicle

CMOS Complementary Metal-Oxide Semiconductor

CNT Carbon NanoTube

CPS Cryogenic Propulsion System

CSM Command and Service Module

DAV Descent/Ascent Vehicle

DLR German Aerospace Center

DRA Design Reference Architecture

DSH Deep Space Habitat

DSN Deep Space Network

ECLS Environmental Control and Life Support

EDL Entry Descent and Landing

ELEO Equatorial Low Earth Orbit

EML Earth Moon Lagrangian

EML1 First Earth Moon Lagrangian

EML2 Second Earth Moon Lagrangian point

EMMU Enhanced Manned Maneuvering Unit

EP-ISS Equatorial Post-ISS

EPS Electical Power Subsystem

ERH Exploration Research Habitat

ERV Earth Reentry Vehicle

EVA Extra Vehicular Activity

FDIR Fault Detection Isolation and Recovery

FSPS Fission Surface Power System

GCR Galactic Cosmic Rays

GNC Guidance Navigation and Control

HDA Hazard Detection and Avoidance

HDPE High Density Polyethylene

HEO High Earth Orbit

HSE Human Space Exploration

HX Heat Exchanger

I/F Interface

IMLEO Initial Mass in Low Earth Orbit

IMM Inverted Metamorphic

ISRU In-Situ Resources Utilization

ISS International Space Station

IXV Intermediate eXperimental Vehicle

LCH4 Liquid Methane

LCL Lunar Cargo Lander

LEO Low Earth Orbit

LH2 Liquid Hydrogen

LLO Low Lunar Orbit

LMO Low Mars Orbit

LOX Liquid Oxygen

LPP Lunar Power Plant

LPR Lunar Pressurized Rover

LRS Lunar Relay Satellite

LSH Lunar Surface Habitat

MARSAT Mars Relay Satellite

MAV Mars Ascent Vehicle

MCT Mars Communications Terminal

MEO Medium Earth Orbit

MJ Multi Junction

MLI Multi Layer Insulation

MMOD Micro Meteoroid and Orbital Debris

MMRTG Multi Mission Radioisotope Thermoelectric Generator

MMSEV Multi Mission Space Exploration Vehicle

MMU Manned Maneuvering Unit

MPU Main Power Unit

MSR Mars Sample Return

MTO Mars Transfer Orbit

NASA National Aeronautics and Space Administration

NEA Near Earth Asteroid

NEO Near Earth Object

NEO RR NEO Robotic Reconaissance

NTR Nuclear Thermal Rocket

OPV Organic Photovoltaic

PCU Power Control Unit

PEM Proton Exchange Membrane

PGA Pressure Garment Assembly

PICA Phenolic Impregnated Carbon Ablator

PLSS Portable Life Support System

PR Pressurized Rover

PV Photovoltaic

RF Radio Frequency

RG Relative Guidance

RvD Rendezvous and Docking

RX Receiver

S/S Subsystem

SCWO Super Critical Water Oxidation

SEP Solar Electric Propulsion

SER Small Eploration Rover

SHAB Surface Habitat/Lander

SLA Super Light weight Ablator

ACRONYMS

SLS Space Launch System

SM Service Module

SML Small Manned Lander

SNR Signal to Noise Ratio

SNTR Small Nuclear Thermal Rocket

SolPS Solar Power System

SPE Solar Particle Event

TA Technological Area

TAS-I Thales Alenia Space - Italia

TCC Trace Contaminant Control

TCS Thermal Control Subsystem

TEI Trans-Earth Injection

TLC Telecommunications

TMI Trans-Mars Injection

TPS Thermal Protection System

TX Transmitter

1

Introduction

1.1 “Space Exploration”

“Space Exploration” is the discovery and study of outer space through the use of astronomy and space technology.
Physical space exploration began in Germany, where scientists developed and tested a V-2 rocket during World War II. This rocket became the first man-made object to enter space, alongside the launch of the A-4 in October of 1942. After the war ended, the United States used rockets captured from the Germans and their scientists to research and study rockets for military and civilian purposes. Although the Germans launched the first man-made object into space, the first exploration of space occurred in May of 1946, when the United States launched a V-2 for an experiment to analyze cosmic radiation. In 1947, fruit flies became the first animals in space and the first pictures of Earth were taken. Both of these experiments were conducted by using American V-2s. The Soviets also launched animal and radiation experiments in 1947, with the help of German scientists. These experiments were conducted using a variant of the V-2 known as the R-1. All of these early space exploration experiments were limited to short flights in sub-orbital space.
The Soviets conducted the first successful orbital mission in October of 1957 after launching the unmanned space vehicle Sputnik 1. This satellite weighed around 184 pounds and transmitted beeps down to radios across the earth, which were analyzed by scientists to measure the electron density in the ionosphere. The beeps also contained

encoded information about the temperature and pressure of Sputnik, which helped researchers know its safety status. Sputnik eventually burned up upon re-entering the atmosphere, but its launch and success paved the way for other missions, including the successful launch of Explorer 1 by America in 1958.
The first human flight was launched by Russia in in 1961, successfully sending cosmonaut Yuri Gagarin into space for one Earth orbit aboard Vostok 1. America launched Mercury-Redstone 3 about a month later with Alan Shepard on board, but this flight was suborbital.
The next step in space exploration was successfully landing an object on a planetary body. This was accomplished in 1959, when Russias Luna 2 landed on the moon. Americas Apollo 11 was the first manned spacecraft to reach the moon. Through the 1970s NASA ramped up its space exploration with the launches of several space orbiters, including one would much later make history. NASA launched Voyager 1 on September 5, 1977, an orbiter that was expected to last several years, exploring outer planets. It visited Jupiter in 1979 and Saturn in 1980, and its primary mission ended on November 20, 1980; however Voyager 1 is still going strong today.
Space Exploration got a huge lift when NASA launched the first Space Shuttle mission on April 12, 1981. That launch touched off a 30-year manned space program that saw 135 crewed missions into space. In 2000, the International Space Station paved a way for a continuing human presence in space. The Space Shuttle then became utilized as a vehicle to transport humans to and from the orbiting lab. After 11 years of supplying humans and cargo to the ISS, the Shuttle program was retired, leaving Russia as the only space agency capable of launching humans to and from the ISS - aboard its Soyuz spacecraft.
For more than fifty years, humans have explored space, and this has produced a continuing flow of social benefits. By its very nature, space exploration expands the envelope of human knowledge and presence throughout the solar system, and this process has been accelerated by a combination of robotic and human activities. Indeed many benefits can be obtained relying on human presence in space missions; on the other hand robotic systems can provide great support to human activities, thus reducing associated risks. Robots shall be sent as pathfinders and scouts, to decide where and when to send people later on.

Experience has demonstrated that, as long as humankind addresses the challenges of exploring space, many tangible societal benefits are produced. Space exploration has contributed to many diverse aspects of everyday life, from solar panels to implantable heart monitors, from cancer therapy to light-weight materials, and from water-purification systems to improved computing systems and to a global search-and-rescue system [1]. Today, human space exploration is limited to Low Earth Orbit (LEO), as it is mainly related to the International Space Station (ISS), which indeed continues to contribute significant benefits to humanity, supporting investigations in life and physical sciences, as well as advancing research and technology to solve problems associated with long-duration human spaceflights that have many applications on ground.
The next steps in the solar system exploration will look at beyond-LEO destinations, and at establishing sustained access to space exploration destinations such as the Moon, asteroids and Mars. The achievement of such ambitious goals will further expand the economic relevance of space. Space exploration will continue to be an essential driver for opening up new domains in science and technology, triggering other sectors to partner with the space sector for joint research and development. This will return immediate benefits back to Earth, including technological innovation, development of commercial industries and important national capabilities and contribution to our expertise in further exploration. Human exploration can contribute appropriately to the expansion of scientific knowledge and it is in the interest of both science and human spaceflight that a credible and well-rationalized strategy of coordination between them is developed.
In addition, the excitement generated by space exploration attracts young people to careers in science, technology, engineering and mathematics, helping to build global capacity for scientific and technological innovation.

1.2 Research Motivation

So far, a human expedition to Mars is considered the most interesting goal of the future Human Space Exploration (HSE). However, several limitations have to be overcome to accomplish a mission of this type, both from an economical/political and technological/operational point of view. For this reason, it is necessary to define an opportune path of exploration, relying on many missions to intermediate and “easier” destinations,

which would allow a gradual achievement of the capabilities required for the human Mars mission.
The research activities discussed in this thesis have focused on this topic, having as main objectives the establishment of a transversal survey and expertise, the identification of solutions (missions and architectures), introducing modularity and exploiting synergies, and the identification of critical and/or common technologies in support of TAS-I studies and programs relevant to various space exploration scenarios. Specifically the thesis is devoted to the evaluation and critical assessment of space exploration scenarios, together with the associated missions and systems.
Modular concepts, architectures and elements are favored, for future space exploration, in order to reduce risks and costs, thus maximizing the development effectiveness. An important point regards the identification and assessment of critical and/or common technologies. Among all the enabling technologies, those of more interest for TAS-I are further investigated, and in this regard the related roadmaps assessment is one of the most important tasks.

2

Background

2.1 State of the art

Numerous activities are being carried out by the major space agencies, industries and academia with the main scope of assessing the best path to be followed in the exploration of the solar system, with the final target of a human mission to Mars and through intermediate human missions towards multiple deep space destinations (e.g. Near Earth Asteroids).

The most significant works, which have been taken as reference for this research, are:

- Global Exploration Roadmap by the International Space Exploration Coordination Group (ISECG)
- NASA Human Spaceflight Architecture Team (HAT) activities on multi-destinations strategic analysis cycles to assess integrated development approaches for architectures, systems, mission scenarios, and concepts of operation for human space exploration,
- System architecting of exploration infrastructure works by MIT Space System Architecture (SSA) Group
- Human Spaceflight and Exploration Scenario Studies by European Space Agency (ESA)

2.2 TAS-I research programs survey

This section reports an overview of the major research programs carried out in TAS-I. It represents a first activity performed in the frame of PhD work, aimed at a preliminary study of which are the major issues related to space exploration, most of all in terms of innovative enabling technologies.
TAS-I is involved in a number of studies and research programs (Crew Commercial Transport, Exploration Studies, Lunar Lander, Human Mission to NEO) that converge to maturity in support of Space Exploration. The first task of the PhD studies is aimed at rationalizing and coordinating various parallel studies currently on-going in the field of space exploration (and others that may be acquired in future) to identify and exploit potential synergies and commonalities.
Starting from the System of Systems scenarios and the related development approach, the intent is to obtain a synergetic effect by the identification of commonalities, cross-references and interrelations among different missions that might not be obvious at first sight (e.g. the commercial development scenario for exploration missions). A synoptic map of the technology needs, identified for the various missions, is then established and maintained, with the aim of analyzing partial or full commonalities among the needs from different missions, substantiating and further justifying the research studies currently on-going, identifying critical technologies with inadequate TRL to boost or to start with and, for new technologies, performing make-team-buy selection.
The methodology adopted to collect information about the on-going research activities and to build up a correct background where to start from for further work, is based on the development of questionnaires to be filled for all the considered programs in order to have a clear and complete picture of the activities being carried out in TAS-I. One of the major outputs of the questionnaires is the description of the most critical technologies and their relative roadmaps.
Starting from that, an overall mapping of the technologies through the different programs is derived and a final summary matrix is produced, which reports synthetically the list of technologies versus the various programs, highlighting for the common technologies the dates at which they are needed.
All the considerations done for this preliminary research activity are taken into account even in the following steps of the PhD research, being the information collected in this

phase a starting point and a first state-of-the art analysis, aimed at better understanding and assessing space exploration related issues.
A questionnaire is built and used to collect information about the various programs; it consists of several parts, as hereafter described:

1. the first part includes questions about the mission to be accomplished by each mission element;
2. the second part is devoted to the description of the element and to the identification of the major technologies characterizing it;
3. the last part focuses on the critical technologies in terms of both description and roadmap assessment.

The questionnaire is proposed to the attention of the relevant *Project Focal Points* (system engineering managers or study managers), who are asked to provide an answer to the questions in order to get the required information. The considered on-going studies and projects involved in this "survey" activity are:

- AMALIA,
- STEPS - phase 1 (Lunar Lander, Capsule, SoS),
- Core Program for Exploration,
- Crew Commercial Transport,
- Cargo Commercial Transport.

Once filled, the questionnaires are used to make an analysis and to establish a transversal survey with the aim to identify common aspects with particular attention to technologies, simultaneously under study within different programs.
Hereafter, a brief overview of the listed programs is reported, with a description of the mission and related elements and technologies.
AMALIA
The aim of AMALIA project is the study of the Amalia Lunar Module, which is a module able to provide landing and mobility on lunar surface. The study is inserted in the Google Lunar X Prize competition.
Two vehicles are under study: the Amalia Lunar Lander (ALL) and the Amalia Rover (AROV), for an overall mass at launch of about 2200kg. The launch is envisaged in 2016 with a Falcon 9 launcher. The lunar lander is in charge of providing the ΔVs

necessary to accomplish the mission maneuvers, while the rover is envisioned for the surface operations.
AMALIA mission would give the chance to perform in-flight test of several advanced technologies; the most significant ones are:

- Hybrid propulsion,
- Smart skins,
- Rover locomotion S/S.

STEPS
Within STEPS program a rover to be used on the Moon or Mars surface is studied.The rover is needed to provide a means for crew mobility on lunar and Mars surface in the frame of human space exploration, in order to allow long range exploration far from the manned base/landing site. A ground demonstrator is foreseen to test on ground some advanced technologies to be later implemented on the flight pressurized rover, which would give the opportunity for in-flight tests of advanced technologies. The rover shall be sized for a crew of four astronauts in nominal conditions plus additional four in rescue situation, and for permanence up to 14 days. The rover shall be protected from external environment (meteoroids and debris) and from radiation. The overall mass of the rover is 8500kg and it provides $27m^3$ of pressurized volume. The launch is envisaged with an Ares V - like vehicle (one shot), from Cape Canaveral Air Force Station to a GTO. In addition, an Altair-like Lunar Lander (or Mars lander) is necessary, where the rover will be accommodated. The Rover to be sent on the Moon (or on Mars) would give the chance to perform in-flight test of advanced technologies. The most important ones are the following:

- Regenerative fuel cells,
- Deployable radiators,
- Phase change material (PhCM),
- Torque engine,
- Lunar dust contamination control.

CORE PROGRAM FOR EXPLORATION
Within this study several Building Blocks have been analyzed, which are:

- EUROBOT, which is a flight robotic demonstrator aimed to perform EVA operations onboard the ISS;
- Exploration Research Habitat, which is a crew-tended orbital infrastructure conceived as the first outpost beyond LEO, intended to support lunar human exploration missions, increase science return from lunar robotic surface exploration, provide a technology and research platform for exploration and support crew transportation architecture to Moon's surface and NEOs;
- Lunar Cargo Lander (LCL), which is conceived as a standardized lunar cargo lander, not demanding peculiar requirements. This type of lander is supposed to support human presence on the Moon, by delivering food, water, tools and experimental specimen and devices;
- Lunar Pressurized Rover (LPR), conceived as a platform which provides astronauts with the means necessary for the mobility on the Moon's surface, allowing the exploration of large areas of the Moon;
- Lunar Power Plant (LPP), envisaged to provide the required power to all the elements of the surface architecture (Moon base), relying on photovoltaic system, as power source, and regenerative fuel cells, as storage system;
- Space Tug, conceived as a space-based servicing vehicle, used to move and maneuver post-ISS and suitable ISS elements to different positions;
- NEO Robotic Reconnaissance (NEO RR), which is a mission aimed to place a space vehicle in the vicinity of a target Near Earth Object able to communicate with Earth and to determine the characteristics of the target NEO.

Each one of the listed building blocks, gives the chance to perform in-flight test of several advanced technologies (see table 2.1). The roadmaps derived for these technologies refer to specific exploration scenarios, which have been defined in the Core Program for Exploration study. In particular, they represent the earliest time at which a specific technology is required looking at all the developed scenarios.

CARGO COMMERCIAL TRANSPORT

This study is devoted to the Cygnus pressurized cargo module. Cygnus is an unmanned automatic vehicle, consisting of a Service Module (SM) attached to a Pressurized Cargo Module (PCM). It is conceived to provide the ISS with pressurized passive cargo as well as to transport active cargo with a dedicated configuration of the PCM internals.

At the conclusion of the mission it will remove wastes from the station performing a destructive re-entry into Earth's atmosphere.
Being a commercial program, the adopted technologies are derived from ISS heritage. The necessity to maintain a low cost profile for the PCM system pushes on utilization of existing components with proven capabilities. The only significant technology developed was the manufacturing of barrel sections of primary structure from forged cylinders. The first launch of Cygnus was initially expected in 2012, and by the way, all technologies TRL were above 7-8 in 2011 (when the study has been performed). Eventually, Cygnus demonstration mission was successfully launched in September 2013.

Summary

All the considered programs have been analyzed with the final aim of identifying the most critical technologies. Furthermore, an interesting point has been the identification of common technologies among the programs. In particular table 2.1 reports a mapping of the identified technologies versus the various programs. It is worth noticing that Cygnus PCM is not reported in table since it was ready to fly and therefore it did not require any particular critical technology. For the technologies required by more elements the date at which the technology is required is highlighted. In this way it is easy to understand that a technology potentially critical for an element, could be available when needed, having been previously developed for another program.

	AMALIA	STEPS	ERH	EUROBOT	LCL	LPR	LPP	NEO RR	Space Tug
Hybrid propulsion	X								
Smartskins	X								
Rover locomotion	X								
Regenerative fuel cells		TRL8 (2025)				TRL8 (2025)	TRL8 (2028)		
Deployable radiators		X							
Phase change materials		TRL8 (2025)		TRL8 (2025)					
Torque motors		X							
Dust contamination control		TRL8 (2025)				TRL8 (2025)			
Inflatable			X						
IBDM			TRL8 (2021)						TRL8 (2024)
Radiation shielding			X						
0-g countermeasures			X						
Advanced regenerative ECLS			X						
E-MMU			X						
In situ diagnostic / maintenance			X						
In orbit sample analysis			X						
Teleops of surface robotics			X						
Human machine I/F				X					
High temperature electronics				X					
Cryogenic propulsion					X				
OBDH					X				
Active thermal control					X				
LIDAR for landing					TRL8 (2024)			TRL8 (2020)	
Landing legs					X				
Locomotion system						X			
Autonomous GNC (surface)						X			
Deployable solar panels							X		
GNC algo for autonomous proximity ops								X	
Sampling transfer and containment								X	
Landing legs (low g)								X	
Refueling mechanism									X
cryogenic fluids mngt									X

Table 2.1: Critical technologies VS TAS-I programs

3

Methodology

3.1 System Engineering Approach

Systems engineering is the art and science of creating optimal system solutions to complex issues and problems. It is a methodical, disciplined approach for the design, realization, technical management, operations, and retirement of a system. A "system" is a construct or collection of different elements that together produce results not obtainable by the elements alone. The elements, or parts, can include people, hardware, software, facilities, policies, and documents; that is, all things required to produce system-level results. The results include system-level qualities, properties, characteristics, functions, behavior, and performance. The value added by the system as a whole, beyond that contributed independently by the parts, is primarily created by the relationship among the parts; that is, how they are interconnected. It is a way of looking at the "big picture" when making technical decisions. It is a way of achieving stakeholder functional, physical, and operational performance requirements in the intended use environment over the planned life of the systems.

3.1.1 System Engineering: Processes

Three main system engineering processes are to be mentioned [2]: system design, product realization, and technical management. The processes in each set and their interactions and flows are illustrated in figure 3.1. The processes of the system engineering

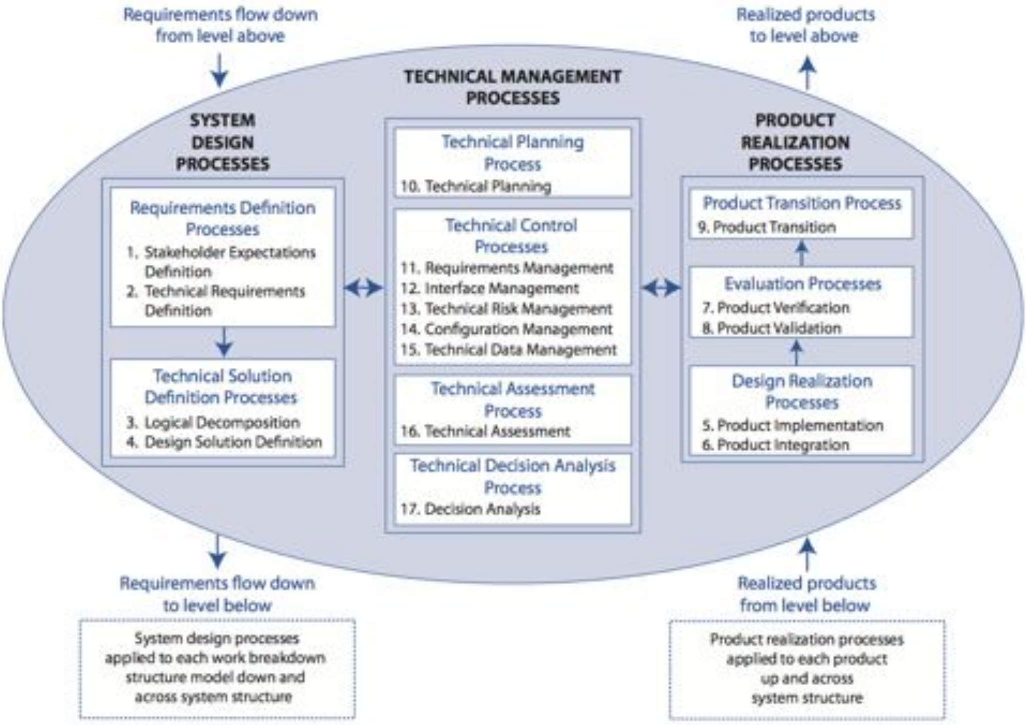


Figure 3.1: The systems engineering engine

engine are used to develop and realize the end products: steps 1 through 9 indicated in figure 3.1 represent the tasks in execution of a project, while steps 10 through 17 are crosscutting tools for carrying out the processes.

System Design Processes: The four system design processes shown in Figure 3.1 are used to define and baseline stakeholder expectations, generate and baseline technical requirements, and convert the technical requirements into a design solution that will satisfy the baselined stakeholder expectations. These processes are applied to each product of the system structure from the top of the structure to the bottom until the lowest products in any system structure branch are defined to the point where they can be built, bought, or reused. All other products in the system structure are realized by integration. Designers not only develop the design solutions to the products intended to perform the operational functions of the system, but also establish requirements for the products and services that enable each operational/mission product in the system structure.

Product Realization Processes: The product realization processes are applied to each operational/mission product in the system structure starting from the lowest level product and working up to higher level integrated products. These processes are used to create the design solution for each product (e.g., by the Product Implementation or Product Integration Process) and to verify, validate, and transition up to the next hierarchical level products that satisfy their design solutions and meet stakeholder expectations as a function of the applicable life-cycle phase.

Technical Management Processes: The technical management processes are used to establish and evolve technical plans for the project, to manage communication across interfaces, to assess progress against the plans and requirements for the system products or services, to control technical execution of the project through to completion, and to aid in the decision-making process.

The system engineering processes are used both iteratively and recursively. “Iterative” is the “application of a process to the same product or set of products to correct a discovered discrepancy or other variation from requirements,” whereas “recursive” is defined as adding value to the system “by the repeated application of processes to design next lower layer system products or to realize next upper layer end products within the system structure. This also applies to repeating application of the same processes to the system structure in the next life-cycle phase to mature the system definition and satisfy phase success criteria.”

The technical processes are applied recursively and iteratively to break down the initializing concepts of the system to a level of detail concrete enough that the technical team can implement a product from the information. Then the processes are applied recursively and iteratively to integrate the smallest product into greater and larger systems until the whole of the system has been assembled, verified, validated, and transitioned.

3.2 Methodology to support strategic decisions

3.2.1 Human Space Exploration Scenario

The developed methodology adopted for the definition of a reference scenario for future human space exploration is schematically described by the work flow reported in figure 3.2, highlighting all the main steps.

The HSE scenario is built considering as final goal a human mission to Mars by the end

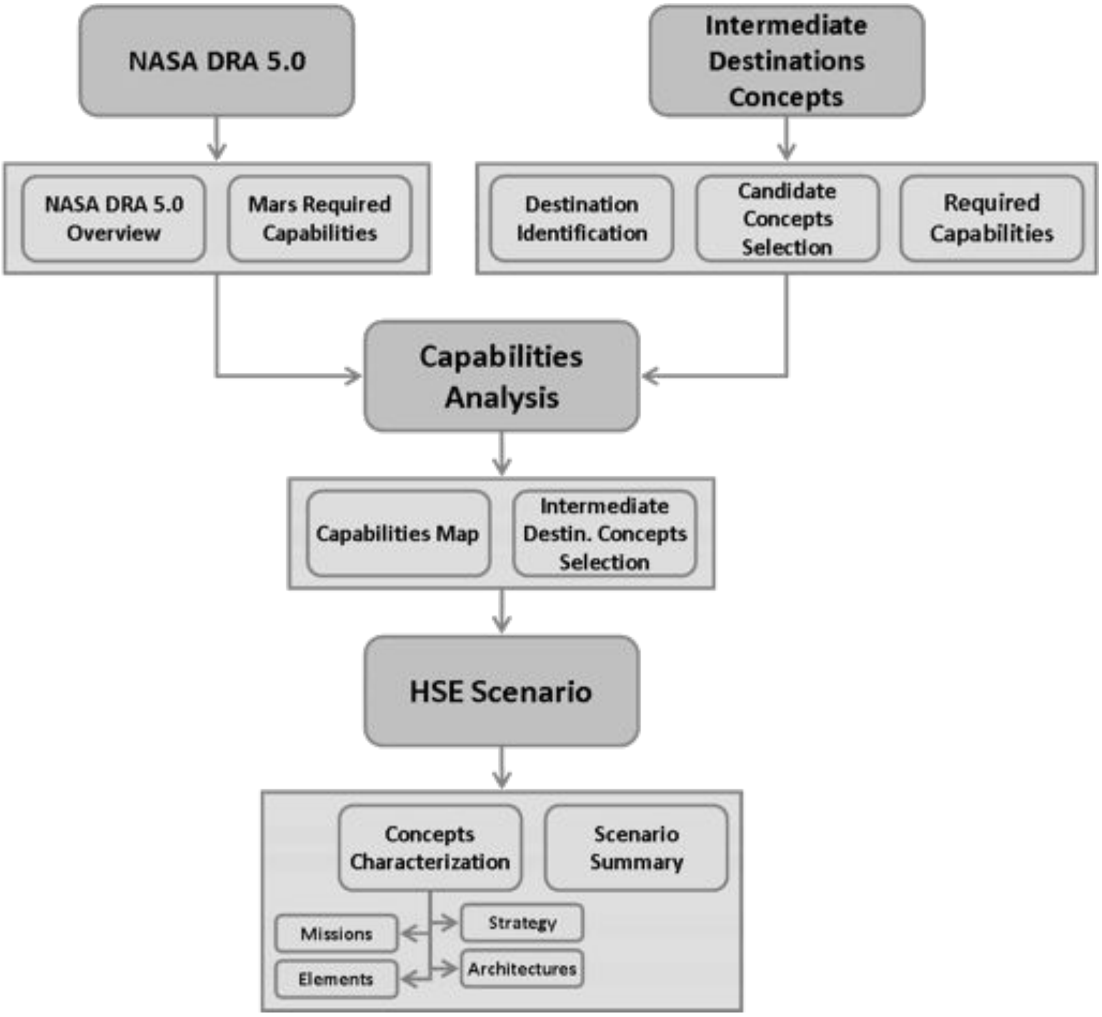


Figure 3.2: Methodology for the assessment of reference HSE scenario

of the 2030 decade. Therefore the first step of the process consists in the assessment of a significant Mars mission to take as reference for the following analyses. In particular NASA DRA 5.0 [3] is assumed as reference mission, selected among several others [4, 5, 6, 7, 8, 9], mainly due to the completeness and accuracy of the available data. Although the mission as described by NASA DRA 5.0 is quite ambitious and has several weak points in its definition, all the considerations done within this study could be easily extended to other mission opportunities, which envisage a Mars human mission as final target. Indeed, the objective of this study is to demonstrate the importance and feasibility of developing a long-term strategy for capability evolution and technology development, when considering space exploration, and specifically to provide a general methodology to be followed in the assessment of a reference scenario. According to this, even if a different “easier” architecture (e.g. with a small number of crew members) or a different time opportunity (maybe a postponed time opportunity), were considered for the final mission to Mars, the considerations done in this study, and most of all the methodology developed, would still be valid and applicable. Prior to proceed with the definition of the intermediate missions, a detailed analysis of the NASA DRA 5.0 reference mission is necessary in order to identify the needed capabilities to accomplish that mission, where the term “capability” basically refers to a function that is likely to be implemented in a subsystem of an element. As a matter of fact, the whole study is based on a pure technical/performance approach, with no risk and cost analyses, as well as no political considerations: the driving criterion for the scenario definition is given by the capabilities required for the final reference mission to Mars. In particular, the idea behind the present study is to follow a gradual path in the expansion through the solar system, which can allow a stepwise technological development and capabilities achievement that can drastically reduce the risks and costs associated to a mission like NASA DRA 5.0.

The top-right branch of the diagram of figure 3.2 refers to the analysis of the intermediate destinations to be included in the scenario. Firstly several possible destinations are identified and for them alternative “candidate concepts” are defined. For all the candidate concepts a list of capabilities is derived, starting from those required for Mars. At this point, combining the list of capabilities needed for Mars and for all the other destinations’ candidate concepts, a global capabilities map is built. Looking at this capabilities map, a down selection of a limited number of intermediate destinations concepts is performed, in order to reduce and simplify the overall scenario. Once the intermediate destinations concepts have been selected, quite a detailed characterization of all the missions to be part of the scenario is done, in terms of strategy, missions, architectures and elements. The final result is an overall scenario of exploration, which includes many missions, both human and robotic, which are conceived to allow a gradual implementation and achievement of the capabilities required to accomplish the reference human mission to Mars by the end of 2030s.

3.2.2 Enabling Technologies Assessment

The second part of the work focuses on the technologies' analysis. It is worth underlining that the final goal is the implementation of a flexible tool applicable to different final destinations (not only to the proposed scenario), in order to support strategic decisions for future space exploration specifically in terms of technologies roadmaps. This part of the work, with all the relevant analyses and assessments, tries to answer the following questions:

- What are all the technologies that can be implemented in the future HSE missions?
- In which HSE missions/elements these technologies are absolutely required?
- In which HSE missions/elements these technologies could be implemented and tested?
- What are the most required and applicable technologies?

Specifically, the methodology developed and followed for the identification of the innovative and promising not yet fully space qualified technologies and for the analysis of their applicability on the elements of the proposed reference HSE scenario is schematically described by the work flow reported in figure 3.3.
The box on the left side of figure 3.3 represents the last step of the methodology developed for the HSE reference scenario definition (see section 3.2.1 for the details), which indeed represents an input for the definition of the technologies roadmaps tool (right side of figure 3.3).
The process starts from the development of a technologies database. The most important and innovative technologies are identified, by means of an accurate review of the major space agencies recent documents on capabilities and technologies assessments and roadmaps [3, 10, 11, 12, 13]. Quite a detailed database is built, which collects a large number of innovative technologies, grouped in technological areas and sub-areas. Then a technologies mapping is carried out, including three main steps. First, an applicability map is developed to map the technologies on the elements of the reference scenario. Then, the technologies are mapped on the destinations of the scenario. Finally, a list of the "most required" technologies is derived, showing when and in which

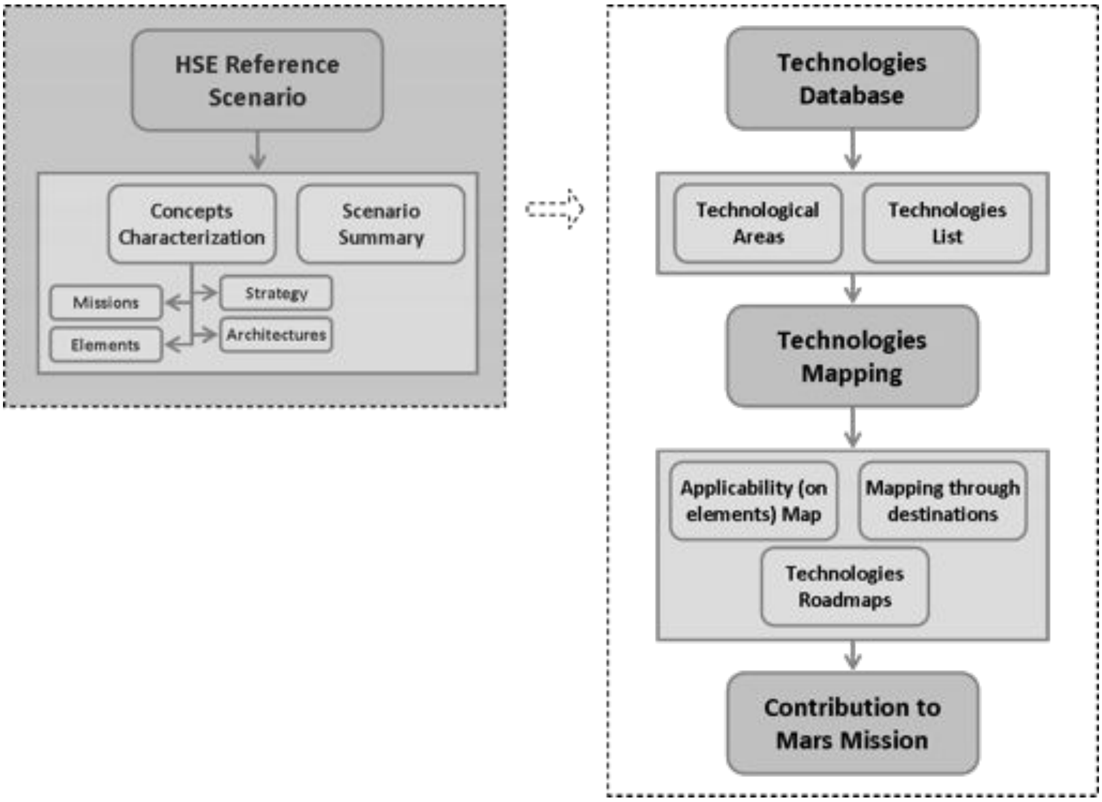


Figure 3.3: Methodology for the definition of technologies roadmaps

mission elements each technology is needed (technologies roadmaps). As last step of the process, the level of contribution of each mission concept to the demonstration of technologies needed for the reference Mars mission is evaluated.
Chapters 4 and 5 report more details about all the procedure steps, and present the most important obtained results.

3.3 Space modules conceptual design

The typical conceptual design process for a space system is depicted in figure 3.4 [14], which describes the various steps as well as the interactions among all the analyses. The process starts with the assessment of the mission statement, from which the mission objectives can be derived. A parallel activity to complete the definition of the mission objectives is the stakeholders' expectations analysis. Once the broad goals

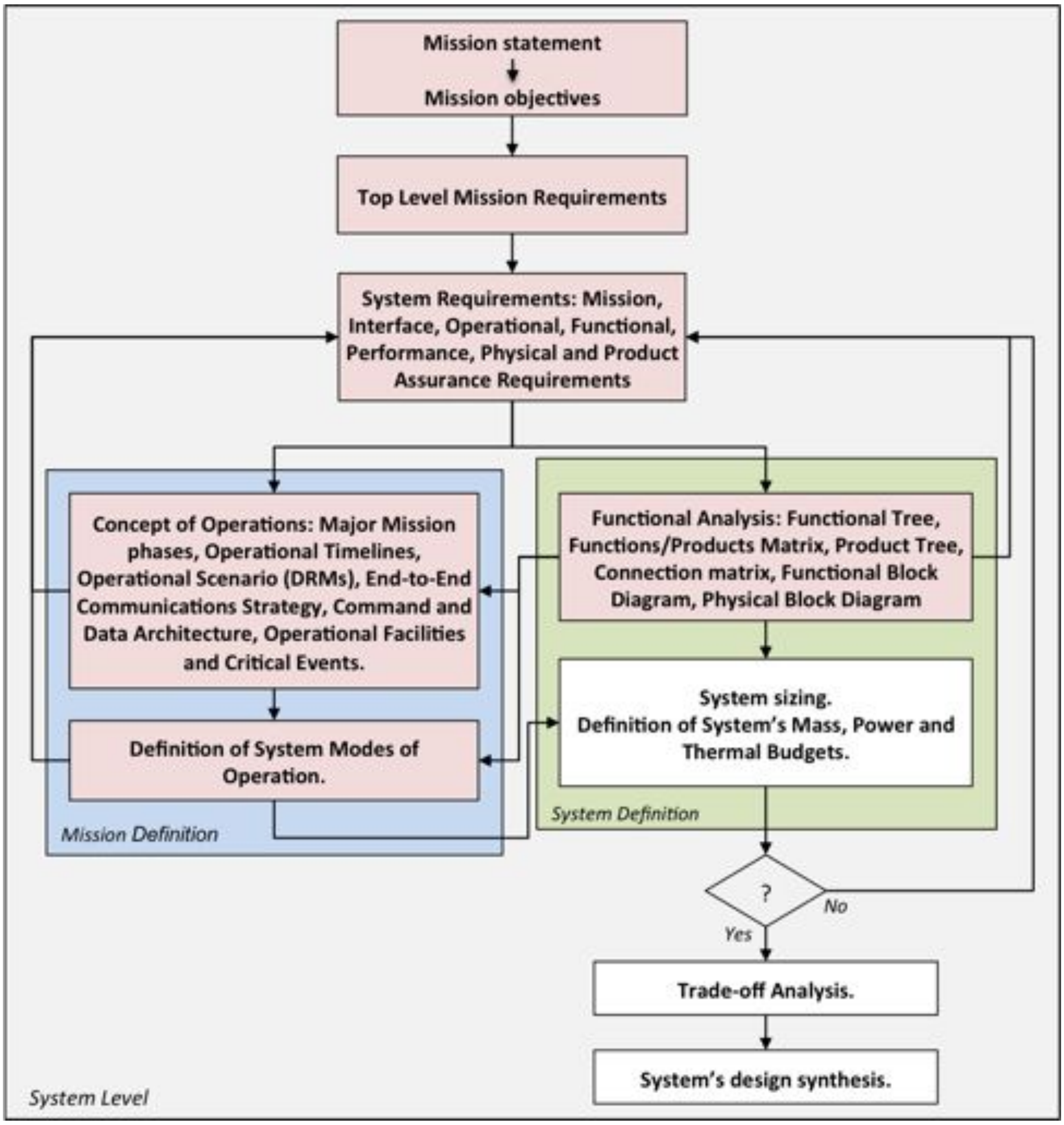


Figure 3.4: The conceptual design process

of the system, represented by the mission objectives, have been established, the system requirements can be defined. Mission statement and mission objectives drive top level requirements, which then drive system requirements, that are repeatedly iterated throughout the design process and are established on the basis of mutual interrelationships with Concept of Operations, Functional Analysis and Definition of System Modes of Operations.

On the basis of the system requirements, the conceptual design process evolves through the system and mission definition.
Starting from the mission objectives/top level system requirements or directly from the mission statement, the Functional Analysis allows identifying the physical components, the so-called building blocks, which constitute the future product, and how they are interrelated to build up the functional architecture of the future product. Physical components are identified by mapping functions to physical components. Moreover through Functional Analysis the functional requirements can be defined or anyway refined.
On the other hand the Concept of Operations (ConOps) describes how the system will be operated during all various life-cycle phases to meet stakeholder expectations. It is an important component in capturing both requirements and the architecture of a system.
Once both the mission and the system architecture have been preliminary defined, it is important to verify whether or not all system requirements have been satisfied. Being the design activity a process of successive refinements, several iterations may be necessary before achieving the system design synthesis, thus freezing the system design. Iterations may occur at every stage of the conceptual design process, thus resulting in a continuous trade or refinement of system requirements.

4

Human Space Exploration Reference Scenario

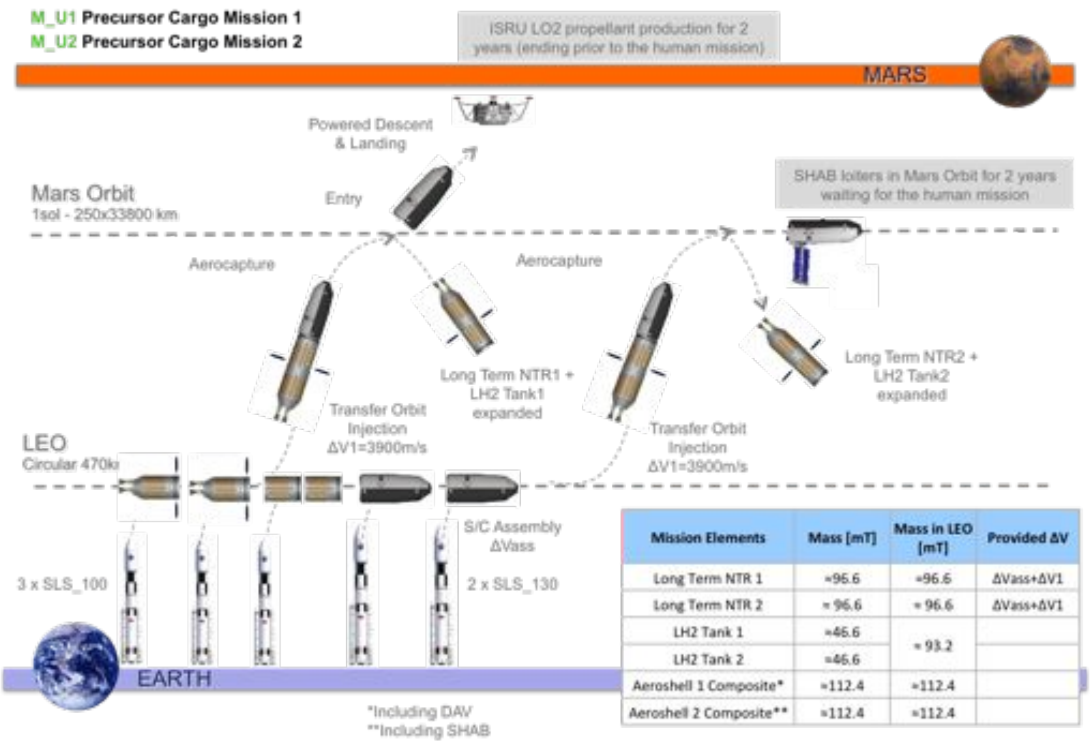
4.1 Reference human mission to Mars

According to methodology illustrated in figure 3.2, the first step of the work is the selection and characterization of the ultimate Mars reference mission. The selected reference mission has to be precisely and clearly described and understood in order to use it as a guideline for building the overall scenario. A brief overview, starting from the general strategy to the illustration of the missions, architectures and elements, is reported in the following. All the data are taken from the NASA-SP-2009-566 Report titled "Human Exploration of Mars: Design Reference Architecture 5.0" (2009) [3].
The general strategy characterizing the NASA DRA 5.0 comprises three main phases, that are:

- Cargo Missions phase, which includes two unmanned missions to Mars in 2037: the first one is envisioned to pre-deploy assets on the surface, such as power plants, mobility, utility and communications elements, ISRU plant and the Mars Ascent Vehicle (MAV); the second one is envisaged to insert into a 1-sol Mars orbit the manned lander and the surface habitat, carrying also pressurized rovers for additional surface mobility capabilities;

- Preliminary Mars Surface Operations phase, which includes two years of preliminary autonomous operations and tele-operations, aimed at the deployment and activation of various elements, power production and ISRU activities (for LOX production and storage);
- Crew Mission, which is planned to start two years later than cargo missions, given that all the LOX propellant needed for the ascent has been produced and stored in the MAV tanks; the main human mission phases are spacecraft assembly in LEO, outbound transfer, Mars orbit insertion, transfer of the crew to the manned lander, Mars entry, descent and landing, operations on the surface, ascent, rendezvous with the main orbiting S/C, inbound transfer and Earth direct re-entry.

Two different architectures characterize the three missions part of this concept; the graphical illustrations of the two architectures are reported in figures 4.1 and 4.2, which refer to the cargo missions and the crew mission, respectivelly.



Mission Elements	Mass [mT]	Mass in LEO [mT]	Provided ΔV
Long Term NTR 1	≈95.6	≈96.6	ΔVass+ΔV1
Long Term NTR 2	≈ 96.6	≈ 96.6	ΔVass+ΔV1
LH2 Tank 1	≈46.6	≈ 93.2	
LH2 Tank 2	≈46.6		
Aeroshell 1 Composite*	≈112.4	≈112.4	
Aeroshell 2 Composite**	≈112.4	≈112.4	

Figure 4.1: Mars Cargo Missions Architecture

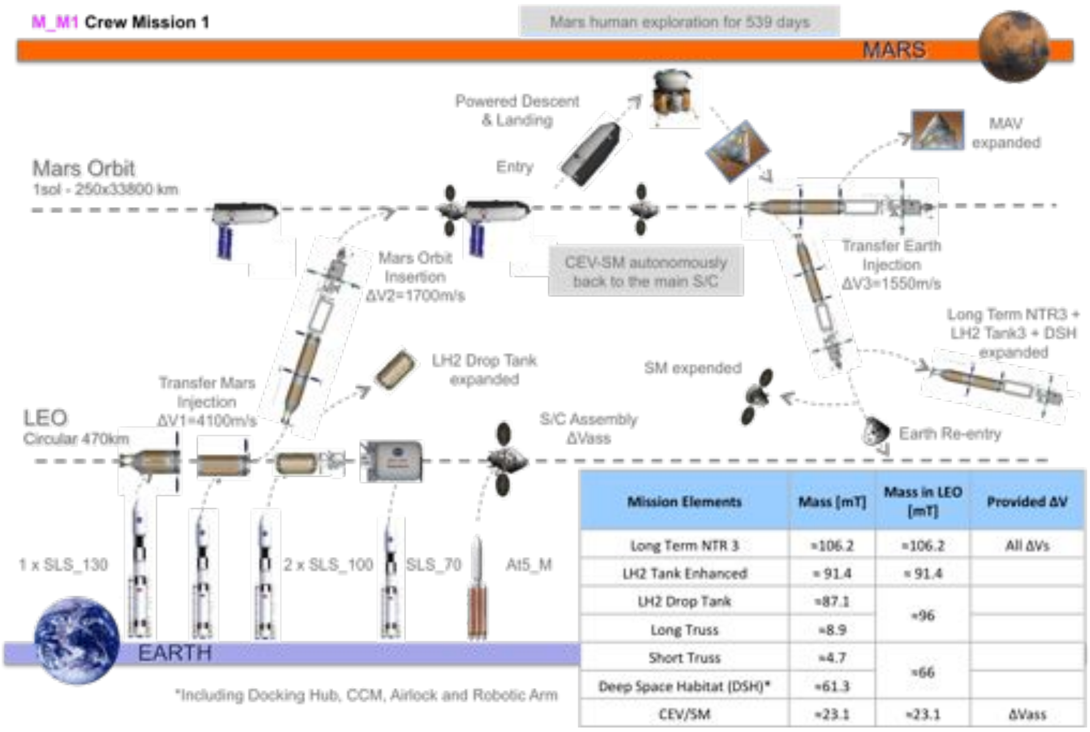


Figure 4.2: Mars Crew Mission Architecture

The pictures show the main mission phases and maneuvers, highlighting the elements involved in the mission. Moreover, the tables reported in the bottom right corner of the figures summarize the elements' masses and indicate the elements in charge of the mission's ΔV.

In order to accomplish the designed missions, NASA estimates that 28 different elements are needed, performing specific required functions. They are summarized in figure 4.3, grouped in *Transportation*, *Surface* and *In-space*. The number of recurrent units for each element is indicated as well.

4.1.1 NASA DRA 5.0 Main Elements

A brief description of the most significant elements is hereafter reported (for additional details, in particular for the elements not described in the present thesis, refer to [3]).

Nuclear Thermal Rocket (NTR)

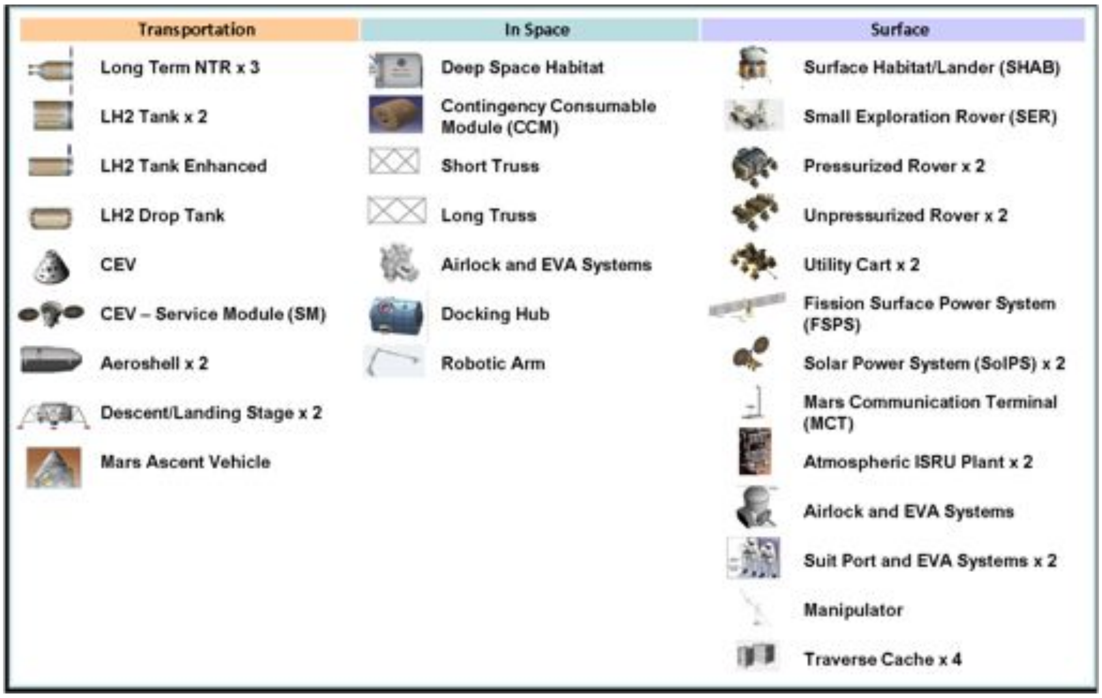


Figure 4.3: NASA DRA 5.0 Missions Elements

The nuclear thermal propulsion is the preferred transportation technology for both crew and cargo vehicles and is retained as the reference one. The NTR is a leading propulsion system option for human Mars missions because of its high thrust (tenths of kN) and high specific impulse (Isp = 900 s) capability. Three long term NTRs are needed for the NASA DRA 5.0 mission, two for the cargo missions and one for the crew mission. Each NTR shall rendezvous and dock in LEO with a liquid hydrogen tank and an aeroshell (see figure 4.1), or with a drop tank and a deep space habitat (see figure 4.2). The loiter time in LEO amounts to several months. In the cargo missions NTRs perform the trans-Mars injection and mid-cruise correction and attitude control maneuvers, while in the crew mission NTR performs also Mars orbit injection, trans-Earth injection and attitude control maneuvers in Mars orbit.
A brief summary of the NASA DRA 5.0 propulsion systems' major features is reported in table 4.1. Besides the main propulsion system, reaction control system is envisioned, relying on more conventional chemical propulsion.

Liquid Hydrogen Tanks

The LH2 tanks are additional tanks for the storage of the liquid hydrogen needed for the

	Cargo Mission	Crew Mission
Main Propulsion System - NTR		
Length [m]	28.8	28.8
Diameter [m]	10	10
LEO phase duration [days]	150-180	150
Total lifetime [days]	530	1054
Total ΔV [km/s]	3.9	7.3
Engine #/type	3/NERVA type	3/NERVA type
Engine thrust[kN]	67	67
Engine Isp[s]	900	900
Ignitions #	1	3
LH2 propellant mass [t]	59.4	59.7
Total mass [t]	96.6	106.2
LH2 propellant thermal control	active (zero boil-off cryocoolers)	active (zero boil-off cryocoolers)
Reaction Control System		
Total ΔV [km/s]	0.3	1
Propulsion type	chemical storable	chemical storable

Table 4.1: NASA DRA 5.0 Propulsion features

main orbital maneuvers. Two versions are foreseen in the NASA DRA 5.0 mission, being the second an enhanced version of the previous one, according to different requirements it has to meet. In particular, the enhanced tank is used in the crew mission and carries about doubled mass of propellant with respect to the "small" version.

Besides the two tanks just mentioned, a third type is foreseen in the NASA DRA 5.0 mission, that is a LH2 drop tank. This is envisaged to store part of the propellant needed for the TMI maneuver of the crew mission. Once the loaded propellant has been consumed the tank is detached from the truss structure and is released.

A brief summary of the major features is reported in table 4.2, where the maneuvers performed relying on the propellant stored in the tanks are highlighted as well.

Crew Exploration Vehicle

Within the framework of the Mars DRA 5.0, a future block upgrade of the Orion Crew Exploration Vehicle serves two vital functions: the transfer of as many as six crew members between Earth and LEO (where the crew transfers into a Mars Transfer Vehicle) at the beginning of the Mars mission, and the return of the six crew members

	TANKS		
	“Small” Tank	Enhanced Tank	Drop Tank
Length [m]	13.3	26.6	22
Diameter [m]	8.9	8.9	9
LEO phase duration [days]	120	120	90
Total lifetime [days]	470	1054	90
LH2 propellant mass [t]	34	70	73
Propellant Mass Fraction [%]	73	76	83
Total mass [t]	46.6	91.4	87.1
LH2 active thermal control	No	Yes (zero boil-off cryocoolers)	No
LH2 passive thermal control	Yes	Yes	Yes
TMI	Yes	Yes	Yes
MOI	No	Yes	No
TEI	No	Yes	No

Table 4.2: NASA DRA 5.0 Propellant (liquid hydrogen) tanks features

to Earth via direct entry from the Mars return trajectory.
A brief summary of the CEV major features is reported in table 4.3.

Aeroshell

The NASA DRA 5.0 foresees two similar Aeroshells with triple-use, i.e. as launch fairing, aerocapture decelerator and entry protection system. The two aeroshells are delivered into LEO by means of two separate SLS, where the RvD with the NTRs is performed (aeroshell acts as target).
Aeroshell 1 contains the descent ascent vehicle assembly and once aerocaptured in Mars Orbit the entry in the Mars atmosphere is performed.
Aeroshell 2, which carries the surface habitat, is captured in Mars Orbit where it is going to loiter for two years waiting for the crew. Eventually, a manned entry is performed and the aeroshell element is jettisoned prior to descent.
A brief summary of the aeroshells' major features is reported in table 4.4.

Descent/Landing Stage

According to NASA DRA5.0, two descent/landing stages are implemented in the human mission, respectively in the Descent Ascent Vehicle (DAV) and Surface Habitat (SHAB) system assemblies.

	CEV/CEV-SM		
Length [m]	5		
Diameter [m]	5.5		
Pressurized Volume [m^3]	40		
Habitable Volume [m^3/person]	24-4		
Crew size	6		
Active crew duration [days]	15		
Quiescent duration [days]	934		
Propulsion	1OME	8AUX	16 RCS
Engine Type	Storable	R-4D	R1-E
Engine thrust [N]	33400	490	111
Engine Isp [s]	326	308	275
Total ΔV [km/s]	1.5		
Entry speed [km/s]	<13		
CM dry mass [kg]	10830		
SM dry mass [kg]	4650		
CM propellant mass [kg]	710		
SM propellant mass [kg]	6950		
Total mass [t]	23		

Table 4.3: NASA DRA 5.0 CEV features

	AEROSHELL	
	Aeroshell 1	Aeroshell 2
Length [m]	30	30
Diameter [m]	10	10
LEO duration [days]	90	60
Deep space duration [days]	350	350
Mars duration [years]	0	2
Total duration	440 days	3.1 years
Ref. Mars orbit	1-sol period, 250x33800km	
Aerocapture velocity [km/s]	6.8	
Aerocapture max heat flux [W/cm^2]	460	
Aerocapture deceleration	4 g	
Entry velocity [km/s]	4.3	
Entry max heat flux [W/cm^2]	132	
Entry constant deceleration	2 g	
Total mass [t]	43.4	43.4

Table 4.4: NASA DRA 5.0 Aeroshell features

The DAV Descent/Landing stage shall support unmanned elements, that are Mars Ascent Vehicle (MAV), ISRU Plant, two utility carts, two Small Exploration Rovers (SER), two Unpressurized Rovers, MPU, Mars Communication Terminal (MCT), Fission Surface Power System (FSPS), Solar Power System (SolPS) and one Manipulator. It is envisaged to land on Mars Surface with the first Cargo Mission.
The SHAB Descent/Landing Stage shall carry the manned Habitat/Lander and two Pressurized Rovers. It is planned to loiter in Mars Orbit inside the Aeroshell 2 until the crew arrival. Both the elements have similar requirements and can implement similar designs.
A brief summary of the descent/landing stage major features is reported in table 4.5.

	DAV	SHAB
Length [m]	4.5	4.5
Diameter [m]	10	10
Payload mass [t]	40.4	
Propulsion type	liquid bi-propellant	
Propellant	LO2/LCH4	
Engine #/type	4/pump-fed	
Engine thrust [kN]	66	
Engine Isp [s]	369	
Ignition #	multiple	
Propellant mass [t]	10	
Descent ΔV [m/s]	600	
Landing precision [m]	10	
Contact velocity [m/s]	2.5	
Total mass [t]	26.5	26.5

Table 4.5: NASA DRA 5.0 Descent/Landing stage features

Mars Ascent Vehicle

The Mars Ascent Vehicle (MAV) is part of the DAV, which is envisioned to land on Mars surface during the first cargo mission, two years before the crew one. Its main function is the transfer of the astronauts from Mars surface up to the Deep Space Habitat (DSH) orbiting around Mars. The MAV carries LCH4 from Earth and is able to store the LOX produced by ISRU plant and needed for the propulsive ascent. Once the MAV is inserted into Martian orbit, RvD with the DSH is performed. After the

crew transfer, the MAV is jettisoned away prior to the trans-Earth injection.
A brief summary of MAV's major features is reported in table 4.6.

	MAV
Length [m]	5
Diameter [m]	5.5
Pressurized Volume [m^3]	40
Habitable Volume [m^3/person]	24-4
Crew size	6
Active crew duration [days]	5
Quiescent duration [days]	350
Prop. production/storage	2 years
Dry mass [t]	13.2
Two ascent stages	
Propulsion type	Liquid Bi-prop.
Propellant	LO2/LCH4
1^{st} stage engine #/type	4/pump-fed
2^{nd} stage engine #/type	1/pump-fed
Engine thrust [kN]	132
Engine Isp [s]	369
Ignition #	2-3
Propellant mass [t]	32.5
Ascent ΔV [km/s]	4.2
Total mass [t]	45.7

Table 4.6: NASA DRA 5.0 Mars Ascent Vehicle features

Deep Space Habitat

The Deep Space Habitat represents the module where the crew members will live during the outbound and inbound transfer phases between LEO and Mars orbit. Moreover it shall host the crew also in case of an abort of the Mars surface mission, until window availability for Earth return. The element is inserted into LEO by an SLS together with the deployable T-shaped short truss, the docking hub and the Contingency Consumables Module, 60 days before the TMI. The element, which implements the inflatable technology, shall be able to autonomously operate in a stand-by mode during the 539 days of Mars surface operations. The habitat is released just before the Earth re-entry.
A brief summary of the DSH major features is reported in table 4.7.

	DSH
Length [m]	11
Diameter - stowed [m]	4.3
Diameter - inflated [m]	8.2
Pressurized Volume [m^3]	340
Habitable Volume [m^3/person]	180-30
Crew size	6
Active crew duration [days]	395
Quiescent duration [days]	539
ECLSS air closure	100%
ECLSS water closure	100%
Power crew [kW]	50
Power quiescent [kW]	10
Total mass [t]	32.8

Table 4.7: NASA DRA 5.0 Deep Space Habitat features

Contingency Consumable Module

The Contingency Consumable Module (CCM) is an inflatable storage module attached to the docking hub connected to the DSH axial port. The CCM main function is the storage of consumables, mainly food, crew provisions and subsystem components spares. Most of the stored food and crew provisions represents the consumables needed in case of a partial or complete abort of the crew surface mission (according to the requirement of 1.5 year to be spent in Mars orbit, waiting for trans-Earth injection opportunity window). In all the scenarios, just before TEI, the module is jettisoned with all the remaining food and all the produced not-recyclable waste.
A brief summary of the CCM major features is reported in table 4.8.

Surface Habitat/Lander

The SHAB Habitat/Lander is the module where the crew has to live during the Mars surface permanence. It is contained inside the Aeroshell 2 and is envisioned to be loitering in Mars orbit for two years until the crew arrival. After the completion of rendezvous and docking maneuvers, the six astronauts will transfer to the central landing capsule part (which can also serve as surface radiation safe heaven) and the following phases of entry, descent and landing will take place. Once on Mars surface all the links with other pre-deployed elements shall be established, so that all the exploration

	CCM
Length [m]	5
Diameter - stowed [m]	1.5
Diameter - inflated [m]	4
Pressurized Volume [m^3]	60
Cargo Volume [m^3]	33
Habitable Volume [m^3]	20
Total lifetime [days]	738
Dry mass (empty) [t]	1.3
Contingency food mass [t]	7.9
Crew provisions mass [t]	1
Spare mass [t]	1
Total mass [t]	11.2

Table 4.8: NASA DRA 5.0 Contingency Consumable Module features

activities can start. After a permanence of about 540 days, the habitat will be left in quiescent mode on Mars surface.
A brief summary of the SHAB major features is reported in table 4.9.

Rovers

NASA DRA 5.0 foresees the implementation of both pressurized and unpressurized rovers.
The two pressurized rovers are stowed inside the SHAB, thus landing on Mars surface during the crew mission. Their main function is extending the crew exploration capabilities in terms of distance and duration during the Mars surface permanence. Moreover they can support easier, longer and continuative investigation and maintenance excursions to relatively far surface assets. Finally, they are also used as transfer element for the crew from SHAB to MAV at the end of the surface operations.
The two unpressurized rovers are stowed inside the DAV, thus being delivered on Mars surface during the first cargo mission. Once the exploration phase starts the Unpressurized Rovers allow extending the range of operations supporting multiple excursions (EVA), and towing particularly heavy or bulky payloads.
Their design is modular, as for the pressurized rover, and they implement a similar Mobility Chassis.
A brief summary of the rovers features is reported in table 4.10.

	SHAB
Length [m]	10-12
Diameter [m]	9.5 (not inflated)
Pressurized Volume [m^3]	200
Habitable Volume [m^3/person]	150-25
Crew size	6
Quiescent duration [years]	2
Capsule crew duration [days]	4-5
Habitat crew duration [days]	539
ECLSS air closure	100%
ECLSS water closure	100%
Power crew [kW]	12
Power quiescent [kW]	4-5
Total mass [t]	21.5

Table 4.9: NASA DRA 5.0 Surface Habitat/Lander features

	ROVER	
	Pressurized Rover	Unpressurized Rover
Length [m]	4.5	4.5
Width [m]	3.9	3.9
Height [m]	3.1	1.5
Pressurized volume [m^3]	25	-
Nominal crew	2	2
Safe/Rescue crew	4	-
Max payload [t]	-	3
Minimum excursion #	40	50
Max autonomy	14 days	8 hours
Max range [km]	100	10
Max velocity [m/s]	1.4	5.5
Sub-elements	• mobility chassis, • EVA system, • cabin, • docking ports.	• mobility chassis, • crew accommodation, • EVA support, • cargo platform.
Total mass [t]	5.8	1

Table 4.10: NASA DRA 5.0 Rovers features

Utility Cart

Two utility carts are implemented in the NASA DRA 5.0 mission. They are stowed inside the DAV and once landed on Mars they are the first elements to be activated. Their main functions are hosting, transporting, deploying and setting the Mars communications terminal, the solar and fission surface power systems, including all the interfaces between them and the main power unit in the DAV. Even if unmanned, it is probably one of the most critical elements due to the large amount and diversity of crucial tasks and the complexity of the operations it has to perform, requiring high level of autonomy and reliability. A brief summary of the utility carts' major features is reported in table 4.11.

	UTILITY CART
Length [m]	2.5
Width [m]	1.5
Height [m]	0.75
Initial operations duration [months]	5
Autonomy	24hours/7days
Total lifetime [years]	5
Max payload mass [t]	8 (FSPS)
Max operative range [km]	1
Max slope capability [deg]	30
Max velocity [m/s]	1.4
Power [kW]	3-5
Total mass [t]	1

Table 4.11: NASA DRA 5.0 Utility Cart features

Fission Surface Power System

The Fission Surface Power System (FSPS) is the reference stationary power generation system, based on a lunar design. It is stowed inside the DAV, and therefore it is delivered on Mars surface during the first cargo mission. One of the utility carts is in charge of offloading the FSPS, transporting it to the desired operative location (approximately 1 km away from the landing site and the future habitat area, mainly for radiation exposure reasons), and activating the system. The utility cart shall also ensure the correct connections with the main power unit, so that power can be correctly distributed to all the loads. The FSPS shall continuously operate for 5 years, even if with variable power requests. A brief summary of the FSPS major features is reported in table 4.12.

	FSPS
Length [m]	2.7
Width [m]	3.3
Height [m]	7
Outpost distance [km]	1
Generated power [kWe]	30-40
Radiation requirements	
<5 rem/yr at outpost	
<50 rem/yr in all other directions	
Total mass [t]	7.8

Table 4.12: NASA DRA 5.0 Fission Surface Power System features

Mars Communication Terminal

The Mars Communication Terminal (MCT) is a communications hub aimed at connecting, with high and low data rates, surface assets with Mars Relay Satellite (as primary link) or directly with Earth Deep Space Network (DSN) (as secondary link). Furthermore, a back-up proximity link between the MCT and the DSH could be envisioned. The MCT is stowed inside the DAV and is offloaded, moved to its operative site, deployed and set by means of one of the utility carts. It also provides data storage, local time and navigation for the loitering and EDL operations and routing functionalities. A brief summary of the MCT major features is reported in table 4.13.

Atmospheric ISRU plant

The In-Situ Resources Utilization (ISRU) plant is designed to convert Mars atmosphere into O_2 for use as propellants and life support. In addition to O_2, the ISRU system generates H_2O and buffer gases for use in the surface habitats and mobility systems. The plants perform a Sabatier conversion of CO_2 to CH_4 and H_2O, with subsequent H_2O electrolysis. Only half of the H_2 needed for the Sabatier is recovered from the water electrolysis process (the remaining one is brought from Earth). During the un-crewed phase, the ISRU plants will continuously operate with the nuclear power source, until the requested amount of ascent oxidizer is produced. Afterwards, the production of the ECLS consumables can start, being regulated depending on the crew needs. The two ISRU Plants are delivered by the DAV during the first cargo mission and are installed

	MCT
Length [m]	1
Width [m]	0.7
Height - stowed/deployed [m]	2/10
Landing site distance [m]	100
Requested power [kW]	421
Total lifetime [years]	5
Links in GHz	
Direct to DSN	High DR 40/37; Low DR 7.2/8.4
Mars relay satellite	High DR 32/34; Low DR 7/8
Surface assets	80Mbps - 2.4/9
SHAB	Hardwired
Local link range	10 km
Total mass [kg]	420

Table 4.13: NASA DRA 5.0 Mars Communications Terminal features

behind the MAV. A brief summary of the ISRU plant major features is reported in table 4.14.

	ISRU Plant
Volume [m^3]	0.86
Total lifetime [years]	2
LH_2 from Earth [kg]	400
Phase 1 - Ascent propellant production	
Duration [days]	300
Requested power - nuclear [kWe]	25
Produced LOX [kg]	24900
Phase 2 - ECLS consumables production	
Requested power - nuclear [kWe]	2
Produced consumables [kg]	2040
Total mass [kg]	570

Table 4.14: NASA DRA 5.0 Atmospheric ISRU plant features

4.1.2 Mars Required Capabilities

The intermediate destinations to be visited before the human mission to Mars, and their relevant concepts, are selected by means of the *Capability Analysis*, which consists in preliminary determining where and how the Mars Required Capabilities can be tested and demonstrated. In this section the capabilities required to accomplish the reference Mars mission are identified. A “capability” is basically a high level function which is likely to be implemented in a subsystem of an element. The functions/subsystems to be considered, transversal to the elements, are those defined as critical, meaning that:

- one or more not yet fully space qualified technologies are considered needed;
- possible design solutions are considered new and challenging (no significant legacy);
- operations have never been implemented and are considered challenging.

The final list of Mars required capabilities is obtained by analyzing in detail the elements of the overall mission architecture; they are divided into four groups, that are *Transportation*, *Operations*, *In Space* and *Surface Support* (in line to what defined for the classification of the elements), as summarized in table 4.15.

Transportation	**Operations**
High performance human transfer	Advanced RvD
High speed Earth manned EDL	Long range communications (high DR)
High capacity cargo transfer	Medium range communications
Orbit cargo insertion (non propulsive)	Short range communications
Destination cargo entry	Reduced gravity drilling & samples mgmt
Destination manned entry	Low-g bodies anchoring, drilling & samples mgmt
Destination cargo D&L	Robotic tele-operations
Destination manned D&L	Safe in-space elements separation
Destination manned ascent	**Support - Surface**
Destination cargo ascent	Surface multiple dockings
Support-In Space	Surface cryogenic fuel management
In-Space multiple dockings	Surface advanced power
In-Space cryogenic fuel management	Surface advanced thermal
In-Space advanced power	Surface advanced life support
In-Space advanced thermal	Surface advanced human health support
In-Space high capacity storage	Surface advanced human habitability
In-Space advanced life support	Surface radiation protection
In-Space advanced human health support	Surface advanced robotics
In-Space advanced human habitability	Atmospheric ISRU
In-Space radiation protection	Soil ISRU
In-Space advanced robotics	Surface advanced EVA
In-Space advanced EVA	Low-g bodies mobility
	Surface mobility

Table 4.15: NASA DRA 5.0 required capabilities

This list of Mars required capabilities represents the starting point for the definition of the reference scenario, which is built, as a matter of fact, on the basis of the capabilities analysis (described in section 4.2.2), aimed at identifying the intermediate destinations missions which best allow their gradual achievement.

4.2 Characterization of missions and related architectures

4.2.1 HSE Intermediate Destinations

Once fixed the final target mission, the intermediate destinations steps need to be assessed in order to proceed with the definition of the human space exploration reference scenario.
The following seven intermediate destinations are considered as possible stages in the path for exploration:

- Low Earth Orbit, considered mainly for its easy accessibility from Earth (orbital altitude approximately between 160 km and 2000 km) and for the presence of the already available International Space Station;
- Medium or High Earth Orbits, interesting because of their medium accessibility cost from Earth and for a more Deep Space-like environment than LEO;
- Cis-Lunar space (e.g. Earth-Moon Lagrangian Points), which is characterized by a deep space environment and allows an increased science return from the Moon;
- Moon, for which both surface sorties and outpost possibilities are considered, in order to perform exploration on the lunar surface as well as to prepare for extended Mars exploration;
- Near Earth Asteroids, which give the possibility to perform a significant mission (closer than Mars), with analogous Mars mission deep-space aspects;
- Mars Moons, considered as a possibility for a Mars mission rehearsal, with reduced complexity and tele-operations of Mars assets;
- Mars Orbit, as Mars mission rehearsal, with reduced complexity.

For these seven destinations several mission concepts are defined*, deriving from the combination of alternative "first-level key decisions", which are very high level concept attributes. In particular for each destination, tree diagrams are built, providing the alternative concepts for the various destinations, as described in the following.

Low Earth Orbit

Low Earth Orbit is the first destination where to start from in a future path for space exploration, as it is the closest destination to Earth and a good knowledge of its environment has already been gained, especially through the International Space Station,which indeed should be exploited as much as possible during its entire operational life. A post-ISS in LEO may also be considered as following step/destination to continue the activities necessary to support further exploration as well as scientific research. The various concepts identified for the LEO destination are graphically described by the tree diagram shown in figure 4.4. Each tree branch represents one of the possible concepts

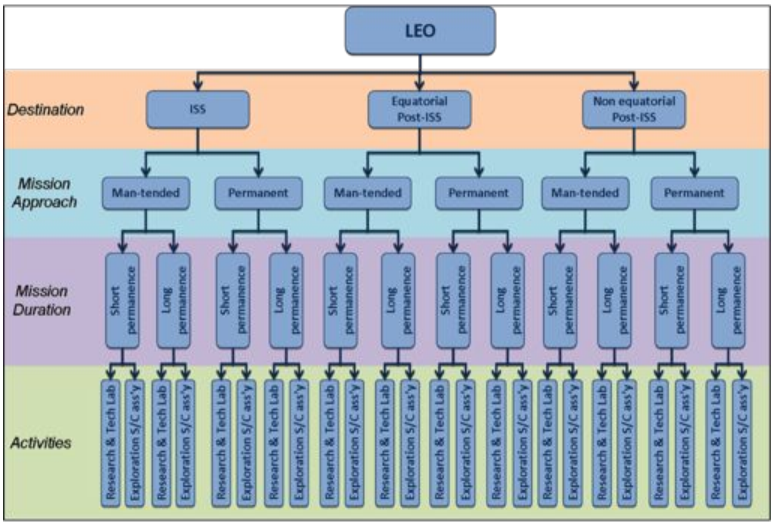


Figure 4.4: LEO Concepts Tree Diagram

derived from the combination of the following "first-level key decisions":

- destination, which can be the ISS, an Equatorial Post-ISS (EP-ISS) and a Non-Equatorial Post-ISS (Non EP-ISS);

*With "mission concept", a high level description of the exploration strategy is meant. Each mission concept shall be more deeply analyzed to identify and characterize the missions composing it.

- mission approach, that is if the LEO infrastructure is to be envisioned as a men-tended or a permanently inhabited station;
- mission duration, which refers to a short permanence (less than two weeks) versus a long permanence (more than two weeks up to several months) of the crew on the station;
- activities, which can be research and technological test, or also support for the assembly of spacecraft for further exploration.

A first down-selection of "candidate concepts" is performed by qualitatively comparing the alternative options, for each "first-level key decision". Some of the branches of the tree can be easily neglected, because of the specific characteristics of the destination. For instance for the ISS, the man-tended mission approach is neglected, given that ISS is already being used as a permanently inhabited space infrastructure. As permanently inhabited infrastructure, long permanence of astronauts is foreseen (thus another branch is canceled) and, finally, ISS is not capable to support large spacecraft assembly. Therefore, only the branch related to the permanently inhabited station, with long permanence of the crew, mainly to perform research and technological tests is considered. Analogously to the ISS case, it has no much sense to consider a mission concept envisaging a permanent crew for short stay time (too costly). Therefore, that branch is neglected for both the equatorial and non-equatorial Post-ISS. The Non-equatorial post-ISS option is discarded, since it would imply higher cost to access from Earth as well as higher costs to access transfer orbit for exploration, while not offering significant advantages neither in terms of research and technology test opportunity (ISS available) nor for supporting exploration missions. For the EP-ISS mission approach, the men-tended option is considered mainly because it implies less demanding requirements in terms of habitability, human support and logistics, and gives a greater flexibility (scientific experiments can be conducted without crew on-board while different experiments can be performed when crew on-board). However, crew long-permanence is considered (several weeks-months), in order to allow better support both for research and spacecraft assembly. Finally, the EP-ISS is intended as a station able to accomplish both the tasks of Research & Technology laboratory and Support for the assembly of future exploration spacecraft. In particular the two capabilities may be implemented in a same station, to be used for the two different scopes (e.g. first for research and then for large S/C assembly).

Medium/High Earth Orbit

The second destination considered as possible step in the exploration scenario is represented by higher Earth orbits, beyond LEO, which can allow different technologies tests as well as provide different scientific research opportunities. The tree diagram reported in figure 4.5 shows the possible concepts identified for this destination on the basis of the following "first-level key decisions":

- destination, which can be Medium Earth Orbit (MEO) or High Earth Orbit (HEO);
- mission approach, that is if the infrastructure is to be envisioned as a men-tended or a permanently inhabited station;
- mission duration, which refers to a short permanence (less than two weeks) versus a long permanence (more than two weeks up to several months) of the crew on the station;
- activities, which can be research and technological test, or also support for the assembly of spacecraft for further exploration.

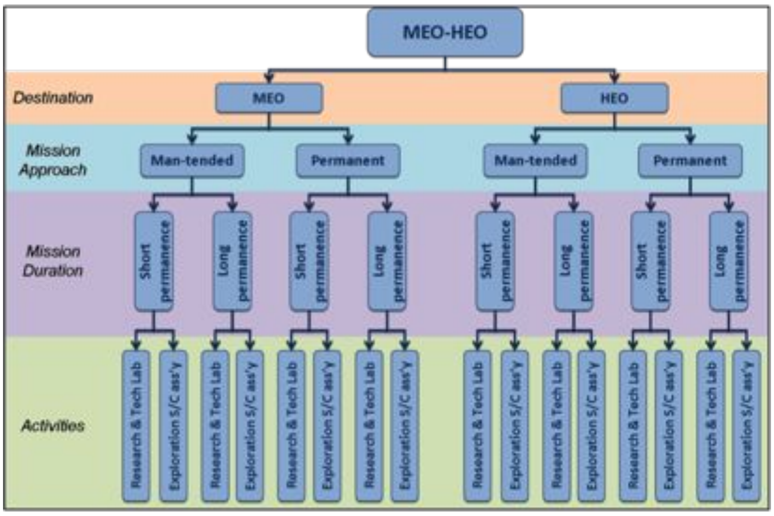


Figure 4.5: MEO/HEO Concepts Tree Diagram

Between MEO (Inner Van Allen Belts) and HEO (Outer Van Allen Belts), the latter is selected mainly because it allows extensive research beyond LEO environment. Moreover, HEOs are more interesting for interplanetary missions spacecraft assembly, because of the lower ΔV that would be required to insert the spacecraft in a transfer orbit. Therefore, all the left-hand side branches of the tree are discarded. Even in this case, a men-tended approach is preferred, since it is mainly conceived for limited research capabilities and assembly of spacecraft. Furthermore the environment of HEO is quite harsh and a permanent presence of astronauts on board is not actually needed. For what concerns the mission duration, a stepwise approach is considered, meaning that at the beginning the crew visits shall be limited to a few days/weeks. The mission duration may then be gradually increased, in order to allow large spacecraft assembly. In summary, two mission concepts are selected for this destination:

- HEO1, with a men-tended infrastructure, envisaging short crew missions mainly to perform research and technologies tests;
- HEO2, with a men-tended infrastructure, envisaging long crew missions to support exploration spacecrafts assembly.

Cis-Lunar

Cis-lunar space represents an interesting place where to deploy a space infrastructure to support human missions beyond LEO for extended stays, and provide a platform for research and technology test. Moreover, a cis-lunar station would increase science return from lunar robotic surface exploration and provide a staging post for missions to Moon's surface, as well as for deep space missions (e.g. NEA missions). The tree diagram showing the possible mission concepts is reported in figure 4.6, as obtained from the combination of the following "first-level key decisions":

- destination, which can be the first or the second Earth Moon Lagrangian (EML) point, or a Low Lunar Orbit (LLO);
- mission approach, that is if the infrastructure is to be envisioned as a men-tended or a permanently inhabited station;
- mission duration, which refers to a short permanence (less than two weeks) versus a long permanence (more than two weeks up to several months) of the crew on the station;
- activities, which can be research and technological test, or also support for the assembly of spacecraft for further exploration.

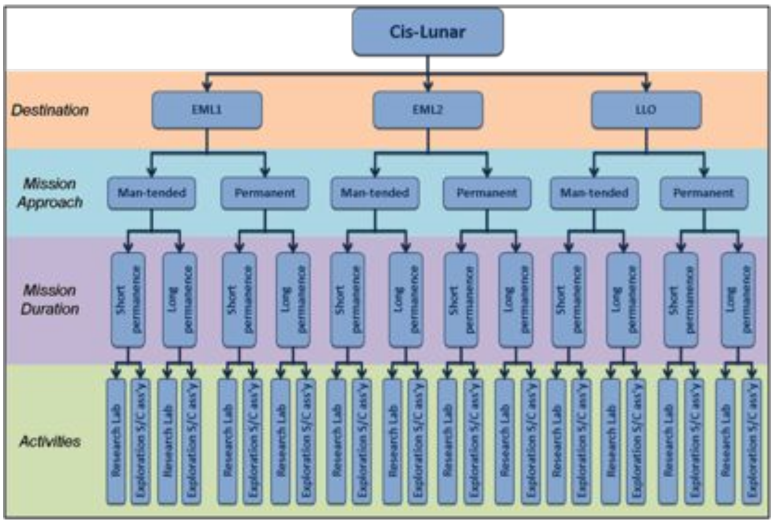


Figure 4.6: Cis-lunar Concepts Tree Diagram

As destination, only EML1 branch is considered, mainly because of the higher accessibility from Earth and lower cost, direct telecommunications visibility with ground segment, good capability to support lunar robotics (assumed to be deployed on the Near Side of the Moon). If specific robotics are to be deployed on the far side of the Moon, EML2 may be considered as option (trade-off considering the additional telecommunication needs shall be performed). The men-tended option is considered mainly because it has less demanding requirements in terms of habitability, human support and logistics, and a greater flexibility. However, the possibility to increase the duration of the permanence of the crew on board is considered, especially in view of gradually improving the capabilities for deep space exploration. According to this, a stepwise approach is considered, meaning that at the beginning the crew visits are limited to a few days/weeks, due to the high risks and limited knowledge about space environment. The mission duration may then be gradually increased, depending on the gained experience. In this regard, both the options are considered as possible, having in mind that in any case at the beginning short permanence has to be preferred (the second concept can be seen as the second part of the station lifetime). Finally, the activities to be performed are both considered applicable to the station: initially research and technologies test, and later on also support for other exploration missions.

Moon Sortie

The Moon Sortie destination/strategy is intended as composed of one or more lunar expeditions lasting up to 45 days. The tree diagram showing the possible mission concepts is shown in figure 4.7, as obtained from the combination of the following “first-level key decisions”:

- Moon approach, which refers to direct missions to Moon’s surface versus staging in cis-lunar;
- surface stay, which refers to a long permanence (14-45 days) versus a short permanence (3-7 days) of the crew on the lunar surface ;
- exploration range, which can be long range (several kilometers from the landing site - non walking back distance) versus a short range (up to 1 km from the landing site - walking back distance);
- cargo deployment, which refers to having the cargo deployed on the Moon prior to the human mission versus bringing everything during the manned mission.

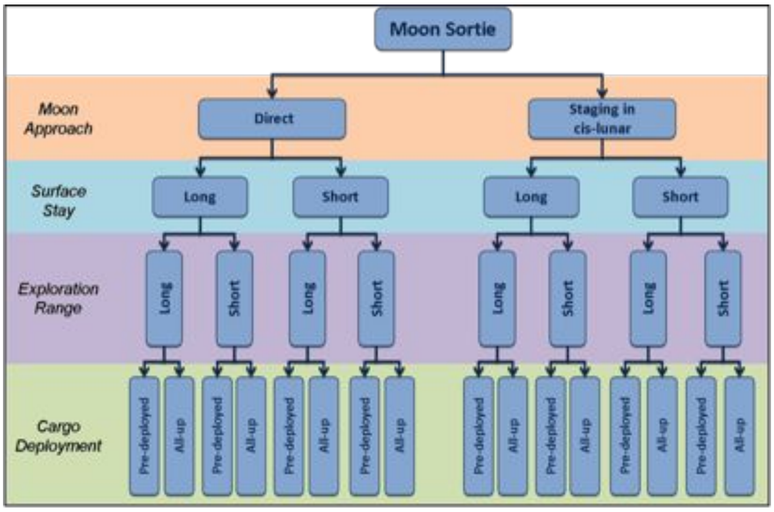


Figure 4.7: Moon Sortie Concepts Tree Diagram

For what concerns the Moon approach, both the alternatives are being considered as possible options. For the mission duration, both the options of long and short surface stay are considered, since it is strictly dependent upon the activities to be performed. Analogous considerations can be done about the exploration range, for which both the alternatives are considered. However, the long exploration range can only be combined with a long surface stay, while the short exploration range would have no sense if combined with the long surface stay option. Finally, also for the cargo deployment, both the alternatives are considered as possible options. In this regard, in the case of short surface stay combined with short exploration range, a pre-deployed cargo is not needed, while the long surface stay combined with the long exploration range is considered feasible only with the pre-deployed cargo option.

Moon Outpost

The Moon Outpost destination/strategy is intended as a concept for building up a lunar outpost on the surface of the Moon. The tree diagram illustrating the possible mission concepts is shown in figure 4.8, as obtained from the combination of the following "first-level key decisions":

- Moon approach, which refers to direct missions to the Moon's surface versus staging in cis-lunar (specifically referred to the crew);
- mission approach, that is if the outpost is envisioned as a men-tended or a permanently inhabited infrastructure;
- surface stay, which refers to a long permanence (between 250 and 600 days) versus a short permanence (up to 180 days) of the crew on the lunar surface;
- exploration range, which can be long range (up to 150km from the landing site non walking back distance) versus short range (up to 1km from the landing site walking back distance);
- cargo deployment, which refers to having the cargo deployed on the Moon prior to the human mission versus bringing everything during the manned mission.

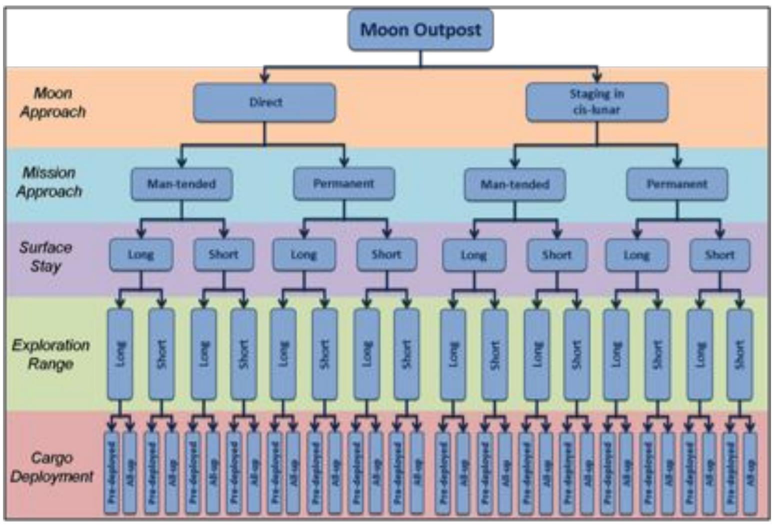


Figure 4.8: Moon Outpost Concepts Tree Diagram

For what concerns the first decision about the Moon approach, which refers to the crew missions, both the alternatives are considered as possible options. The men-tended option is considered since the permanent solution is too demanding in terms of logistics and costs requirements. For the mission duration, both the options of long and short surface stay are considered, since it is strictly dependent upon the activities to be performed on the Moon's surface. In terms of exploration range the long range option is considered: the short range option would represent a great limitation to possible activities, especially considering that the minimum surface stay duration amounts to 180 days. Finally, for what concerns the cargo delivery, the pre-deployment option is considered. The all-up alternative is not likely to be feasible considering the elements that would be necessary to support a Moon outpost with a minimum crew permanence of 180 days.

Near Earth Asteroid

A Near Earth Asteroid is included among the intermediate destinations since it represents a significant mission, with analogous Mars mission deep-space aspects but closer than Mars. The tree diagram showing the possible mission concepts is reported in figure 4.9, as obtained from the combination of the following "first-level key decisions":

- departure approach, which can be departing from LEO or departing from cis-lunar;
- cargo deployment, which refers to having cargo deployed in the NEA proximity prior to the human mission versus bringing everything during the manned mission;
- NEA proximity operations approach, that is if landing or not on the asteroid's surface;
- surface exploration approach, which can be through a dedicated lander or not, in case the "landing" option is chosen for the previous decision, and through an exploration vehicle or not, in case the "no landing" option is selected.

For the departure approach, both the options are considered possible. In particular the first strategy foresees that the overall spacecraft for the NEA mission is assembled in LEO, starting from there the journey towards the asteroid. The second strategy envisions that the overall spacecraft for the NEA mission is assembled in cis-lunar, exploiting a pre-deployed infrastructure: in this case refueling at the cis-lunar station may be envisaged. For what concerns the cargo deployment, both the alternatives are

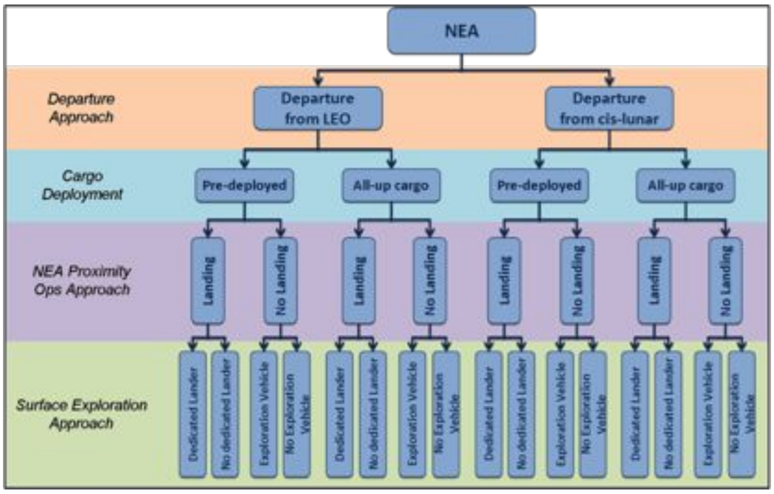


Figure 4.9: NEA Concepts Tree Diagram

considered as possible options, since it is strictly related to the specific mission and its specific requirements. The option of non-landing is selected for the NEA proximity operations, as it implies lower complexity of the overall system architecture. Furthermore, the development of a specific landing system for a NEA could not be seen as a preliminary technology development in view of the Mars mission, due to the very different environment: the increase in the complexity is not justified. Finally, for the surface exploration approach, the option envisaging the use of an exploration vehicle to perform the activities around the NEA is selected mainly because it allows safer and easier EVAs execution. Moreover, this gives the chance to test a system that can be used even in further mission to Mars (given specific minor modifications). At the end, four mission concepts are selected as "candidate concepts", depending on the departure and cargo deployment approach, but all considering a no-landing approach and relying on an exploration vehicle for the activities on the asteroid's surface.

Mars Moons

Mars Moons can represent another possible destination in the path of exploration, mainly because they can allow a Mars mission rehearsal, with reduced complexity, and tele-operations of robotic assets deployed on the martian surface. The tree diagram reporting the possible mission concepts is shown in figure 4.10, as obtained from the combination of the following "first-level key decisions":

- mission destination, that can be Phobos or Deimos;

- departure approach, which can be departing from LEO or departing from cis-lunar;
- exploration approach, that is if landing on the surface or remaining on orbit around the Moon;
- cargo deployment, which refers to having the cargo deployed prior to the human mission versus bringing everything during the manned mission;
- mission duration, that is long stay (several weeks/months) versus short stay (a few days).

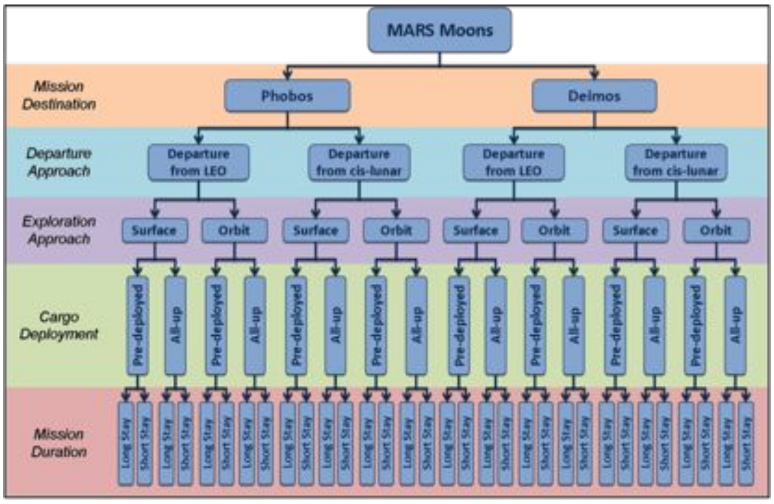


Figure 4.10: Mars Moons Concepts Tree Diagram

Between the two Moons, Deimos is chosen as destination mainly because of its lower demand in terms of ΔV, its superior coverage of sites on the martian surface and the extended durations of constant sunlight, thus allowing more extensive tele-operation activities of robotic assets deployed on Mars surface. For what concerns the departure approach, both the alternatives are considered as possible options. The surface approach is selected for the exploration of Deimos, since it allows deeper analyses and ISRU, in view of eventually providing a spaceport and refueling station, for future Mars missions. Regarding the cargo deployment, both the options are considered, since it is strictly related to the specific mission and its specific requirements. Finally, mission duration of several weeks/months is selected, since it allows much more extensive exploration activities.

Mars Orbit

The Mars Orbit is the last considered destination for the scenario and can be seen as a possibility to perform a Mars mission rehearsal, with reduced complexity, since no EDL systems would be necessary. The tree diagram showing the possible mission concepts is illustrated in figure 4.11, as obtained from the combination of the following "first-level key decisions":

- departure approach, which can be departing from LEO or departing from cis-lunar;
- deployment approach, that refers to having the station deployed to Mars prior to human mission or not;
- mission approach, which refers to having a men-tended versus a permanently inhabited infrastructure;
- mission duration, that is long stay (several months) versus short stay (a few weeks).

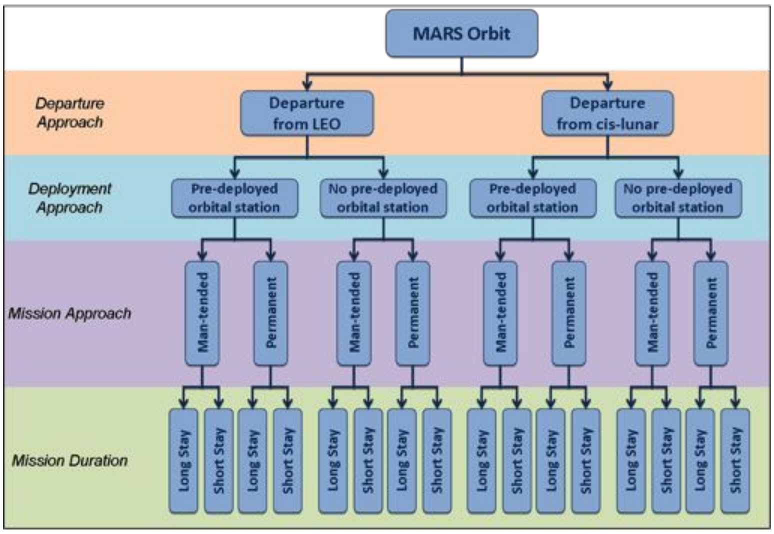


Figure 4.11: Mars Orbit Concepts Tree Diagram

For what concerns the departure approach, both the alternatives are considered as possible options. The orbital station is assumed to be deployed prior to the crewed mission, in order to have in Mars orbit an outpost which allows performing research and systems test/check even before crew arrival. The orbiting station would provide a permanent platform in Mars orbit to be used as staging post for future surface missions. The orbital station is considered as a men-tended station, since a continuous shift of the crew would be very difficult due to the long travel required to get to Mars, as well as transfer window opportunities. This would imply very complex architecture to provide astronauts with a safe environment where to live for such long periods in orbit. Finally, the permanence of the crew on the station shall last no more than a few weeks, in order to have a less complex station, with the possibility to achieve in any case the objectives of such a mission (mainly tele-operation of surface assets and robotic samples collection).

Summary

Summarizing what just discussed, 24 concepts are selected. A summary of their major features is reported in table 4.16, in which the acronyms used to refer to the specific mission concepts are also indicated. These "Candidate Concepts" are then used for the capabilities analysis, having as final objective a down-selection of the minimum number of concepts that allows achieving the capabilities required for the Mars mission (see section 4.2.2).

Destination	Candidate Concept	Main Features
LEO	ISS	• ISS • Permanent • Long Permanence • Research & technologies test lab
	EP-ISS	• Equatorial Post-ISS • Men-tended • Long Permanence • Research lab & S/C assembly
MEO/HEO	HEO1	• HEO • Men-tended • Short Permanence • Research & technologies test lab

Destination	Candidate Concept	Main Features
	HEO2	• HEO • Men-tended • Long Permanence • Exploration S/C assembly
Cis-Lunar	CL1	• EML1 • Men-tended • Short Permanence • Research laboratory
	CL2	• EML1 • Men-tended • Long Permanence • Exploration S/C assembly/support
Moon Sorties	MS1	• Direct Approach • Long Stay • Long Exploration Range • Pre-Deployed Cargo
	MS2	• Direct Approach • Short Stay • Short Exploration Range • All up Cargo
	MS3	• Staging in Cis-lunar • Long Stay • Long Exploration Range • Pre-Deployed Cargo

Destination	Candidate Concept	Main Features
	MS4	• Staging in Cis-Lunar • Short Stay • Short Exploration Range • All up Cargo
Moon Outpost	MO1	• Direct Approach • Men-tended • Long Stay • Long Exploration Range • Pre-Deployed Cargo
	MO2	• Direct Approach • Men-tended • Short Stay • Long Exploration Range • Pre-deployed Cargo
	MO3	• Staging in Cis-lunar • Men-tended • Long Stay • Long Exploration Range • Pre-Deployed Cargo
	MO4	• Staging in Cis-Lunar • Men-tended • Short Stay • Long Exploration Range • Pre-deployed Cargo
NEA	NEA1	• LEO Departure • Pre-Deployed Cargo • No-Landing • Exploration Vehicle

Destination	Candidate Concept	Main Features
	NEA2	• LEO Departure • All up Cargo • No-Landing • Exploration Vehicle
	NEA3	• Cis-Lunar Departure • Pre-Deployed Cargo • No-Landing • Exploration Vehicle
	NEA4	• Cis-Lunar Departure • All up Cargo • No-Landing • Exploration Vehicle
Mars Moons	DMS1	• Deimos • LEO Departure • Pre-Deployed Cargo
	DMS2	• Deimos • LEO Departure • All up Cargo
	DMS3	• Deimos • Cis-Lunar Departure • Pre-Deployed Cargo
	DMS4	• Deimos • Cis-Lunar Departure • All up Cargo

Destination	Candidate Concept	Main Features
Mars Orbit	MOr1	• LEO Departure • Pre-deployed station • Men-tended
	MOr2	• Cis-lunar Departure • Pre-deployed station • Men-tended

Table 4.16: Selected “Candidate Concepts” Summary

4.2.2 Capabilities Analysis

The first step of the “capabilities analysis” is the assessment of the capabilities required for the 24 “candidate concepts” selected accordingly to what described in previous section (4.2.1), analogously to what is done for Mars. Then, an analysis of the “applicability” of all the identified capabilities through the different destinations is carried out: starting from the capabilities list derived for Mars, the final scope is to identify which of them are applicable to the selected intermediate destinations concepts, guiding the selection of the most valuable ones to be included in the flexible path scenario. In particular, referring to each specific destination concept, a capability can be:

- “Required”, that means enabling or highly impacting on the overall mission/architecture,
- “Applicable”, indicating that the capability can be implemented and achieved at the specific destination, even if not strictly needed.

The obtained results are summarized in the “Capabilities Map” reported in figure 4.12, that is a matrix providing a clear mapping of the capabilities through the various destinations and according to the concepts’ features.

Apart from the first column, which refers to NASA DRA 5.0 mission, the intermediate destinations are ordered starting from the closest to the furthest locations (with respect to Earth). Besides the Mars required capabilities, the complete list reported in the map, includes some additional ones needed for other destinations, according to the

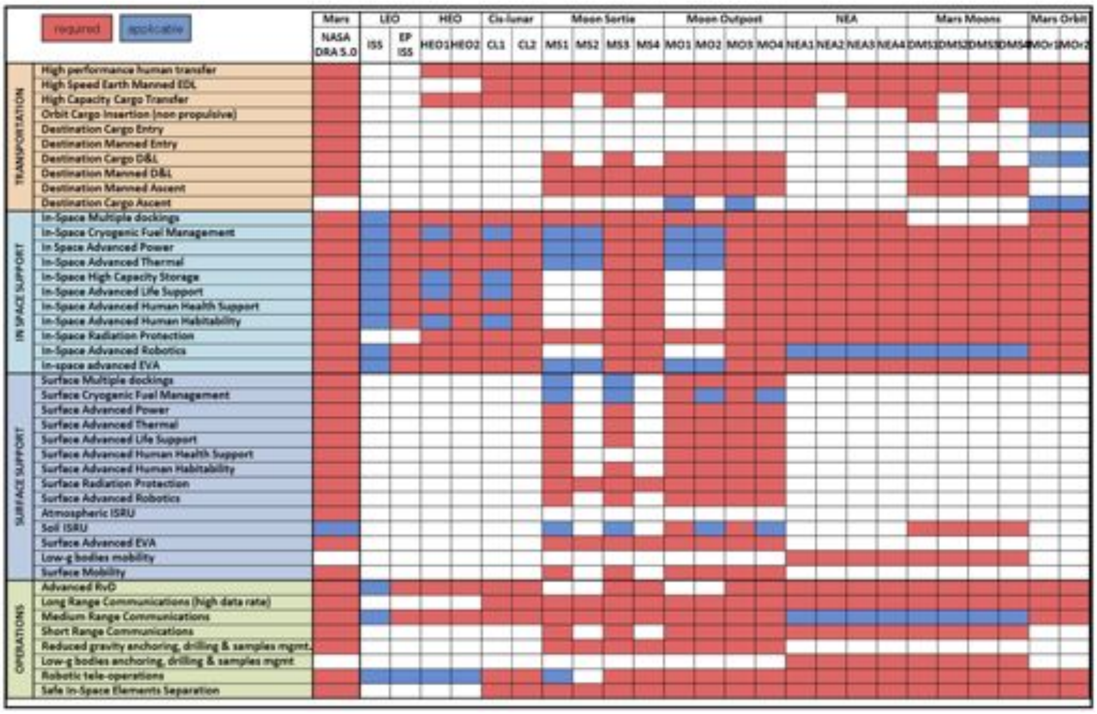


Figure 4.12: Capabilities Map

peculiarities of the missions.
The red cells indicate those capabilities are required, while the blue ones refer to the applicability of the specific capability to the different destinations. It is clear from the matrix that the ISS does not require any of the listed capabilities (as ISS is already complete and operative), but some of them can be applied there. This allows understanding that the first step shall be the exploitation as much as possible of the station. Analogous considerations can be drawn for other concepts, making the matrix a first useful tool to assess and compare the value of different destinations missions concepts.

4.2.2.1 Intermediate Destinations Concepts Down-Selection

Starting from the wide picture of concepts provided through the capabilities map, the following step in the “capabilities analysis” is the selection of the minimum number of concepts that allows the demonstration and achievement of all Mars required capabilities in intermediate locations (where they can be required or applicable). The general selection criteria adopted for this analysis are the following:

- an incremental selection process is adopted, from closer and "easier" to further and "harder" destinations: starting from locations with less demanding requirements (e.g. Cis-Lunar) and gradually moving to more challenging targets (e.g. Moon, NEA, Mars, etc.), in terms of mission durations, needed resources and propellant, psychological aspects, and possibility to "quickly" return to Earth;
- the possibility to reuse already existing space infrastructures is taken into account (e.g. ISS), in order to maximize their exploitation and reduce the overall costs (e.g. post-ISS is discarded in favor of ISS because it would imply much higher costs with almost the same demonstration opportunities);
- coupled concepts are preferred since they allow more flexibility, adaptability and reusability of elements: for example, the Moon sortie concept is envisaged to rely on the station deployed in Cis-Lunar, thus simplifying the architecture and concept of operations; indeed, the Cis-Lunar station represents a staging post, which can have also a reusable lander to support multiple Moon's surface missions or can provide the astronauts with a shelter, in case an emergency situation occurs during a Moon's surface expedition;
- no more than one concept for each destination is selected, in order to keep the overall scenario as "simple" as possible and therefore, implicitly, the cost of the overall scenario as low as possible.

According to these criteria, the various concepts are analyzed and compared and, finally, five out of the 24 concepts are selected to be part of the overall HSE scenario. Specifically, the selected mission concepts are:

- <u>ISS</u>, that relies on an already existing infrastructure, for which all the in-space support capabilities (except for the advanced radiation protection), and three operations capabilities are applicable;
- <u>CL2</u>, coupled with Moon sortie/outpost and for which all the in-space support capabilities are required (CL1 can be considered as a first operational phase of CL2);
- <u>MS3</u>, coupled with CL2 and for which three additional transportation and two additional operations capabilities are required (with respect to ISS and CL2), and

almost all the surface support capabilities and all in-space support capabilities are required or applicable;

- MO3, coupled with CL2 and for which all the in space support capabilities, advanced RvD, surface advanced human health support and soil ISRU are required (not in MS3); surface support capabilities can be demonstrated at increased level with respect to MS3;

- NEA1, which generally allows the same capabilities as CL2 except for some dedicated required capabilities (not needed for Mars) and two additional operations capabilities [15, 16].

The MEO/HEO concepts are both discarded, since they do not provide significant demonstration possibilities, also considering the ISS and CL2 concepts. Similarly, the Mars Moons and Mars orbit concepts are discarded, since they do not provide any considerable advancement in the Mars required capabilities achievement. It has to be underlined that the Mars orbit concept would foresee human mission to an infrastructure deployed in Mars orbit (human on-orbit activities), without landing on the Mars surface, and could be seen as a possibility to perform a Mars mission rehearsal, at least for what concerns the in-space phase, with reduced complexity, since no EDL systems would be necessary. According to this, a few additional capabilities could be achieved; in particular, the cargo entry, descent, landing and ascent capabilities can be considered applicable, since the possibility to carry to Mars an unmanned system (like a dedicated payload for the mission) to deploy robotic assets on the surface is not excluded (maybe to perform tele-operation activities).* However the complexity of such kind of concept (which includes manned missions to Mars orbit) would be very high and may not be justified by a so limited advancement in capabilities achievement. For this reason it is finally discarded, while a dedicated concept is introduced, which instead envisages heavy robotic missions and allows the implementation of additional capabilities not achievable in the other concepts. This sixth concept, called Mars Preparation (MP) concept (see figure 4.13), is characterized by some unmanned missions to Mars orbit

*The surface support capabilities are not considered required/ applicable, since this concept is intended as a simpler concept limited to the human on-orbit activities. Of course, specific payloads could be included, as for example ISRU demo: these aspects are specifically addressed considering an additional robotic concept.

and Mars surface, to demonstrate the missing capabilities (e.g. orbit cargo insertion, cargo entry, descent and landing and atmospheric ISRU), except for destination manned entry, descent, landing and ascent* that can be partially demonstrated through human rated missions and elements.

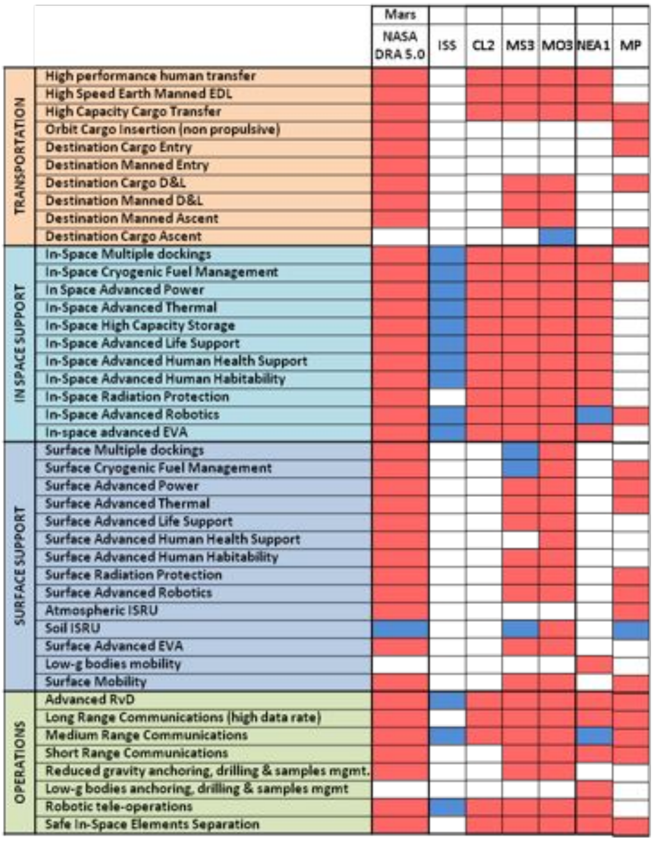


Figure 4.13: Capabilities Map - Selected Concepts including Mars Preparation

The functions listed in the capabilities map (figure 4.12) are related to phases characterized by different levels of risks. No detailed risk analyses have been performed; however, specific considerations have been done especially for the most critical functions and the associated mission elements. The followed approach takes into account that some phases are particularly risky and therefore attention is paid how to imple-

*These refer to systems with astronauts on-board and the attribute "manned" is generally used to distinguish from the cargo (just a matter of nomenclature); however, the systems used in the unmanned missions will be conceived so that they allow implementing the same technologies, considering the same constraints, in order to validate them.

ment the most critical mission elements through various destinations. In this regard, for example, considering the Mars EDL that is very critical, several aeroshell elements are included in the MP concept, which gradually improve their features till achieving the characteristics required for the human mission. Another example is represented by nuclear propulsion, which is considered very challenging and for which a demo is envisaged prior to actually implementing it. Moreover, the first missions relying on nuclear propulsion are unmanned missions, which are less critical than manned ones. Summarizing, six intermediate destinations concepts are included in the reference scenario.

4.2.3 HSE Reference Scenario Definition

To build up the HSE scenario, starting from the six mission concepts discussed in the previous section, all the missions and relative architectures are defined. All the evaluations carried out to assess the missions rely on some preliminary assumptions, hereafter reported:

- the assessment of all the destinations concepts is done always considering the NASA DRA 5.0 study as main reference at all levels, within the idea of an incremental path of Mars required capabilities demonstration;
- mission objectives different from the technological test for the Mars mission (e.g. scientific, research, space promotion) are only partially considered;
- the number of missions proposed for each destination concept is a minimum estimate; in case of failures the number of missions can increase, suggesting for repetitions (Apollo program-like approach);
- mission aborts options are not considered in the human missions of any destination concept;
- no considerations on costs and risks are performed;
- dedicated calculations are performed for the evaluation of the transportation elements or stages;

- no specific models are used for the assessment of the logistics missions, in terms of their numbers and upload capability; the reference values are first approximations based on past and current similar missions (e.g. ATV to ISS);
- the ground and launch segments are not considered in the missions' definition.

State-of-the-art and future planned launchers are considered; in particular the definition of the missions relies upon the launchers listed in table 4.17.

Name	Availability	LEO P/L mass [t]	Launch site	Notes
Ariane 5 ES (A5_ES)	Available	>20	Guiana Space Centre	Unmanned
Ariane 5 ME (A5_ME)	2016	11.2 in GTO	Guiana Space Centre	Unmanned
Falcon 9 Heavy (F9H)	2013-2014	53 (200km, 28.5°)	Cape Canaveral	Unmanned
Space Launch System (SLS_70)	2017	70	Kennedy Space Centre	Unmanned
Space Launch System (SLS_100)	?	100	Kennedy Space Centre	Unmanned
Space Launch System (SLS_130)	?	130	Kennedy Space Centre	Unmanned
Crew-rated Atlas V (At5_M)	2016-2017	28	Cape Canaveral	Manned
Space Launch System (SLS_70M)	2017	70	Kennedy Space Centre	Manned

Table 4.17: Assumed Launchers

For each mission concept the analysis goes through several steps, as schematically illustrated in figure 4.14. First of all, several options for major architecture-level attributes

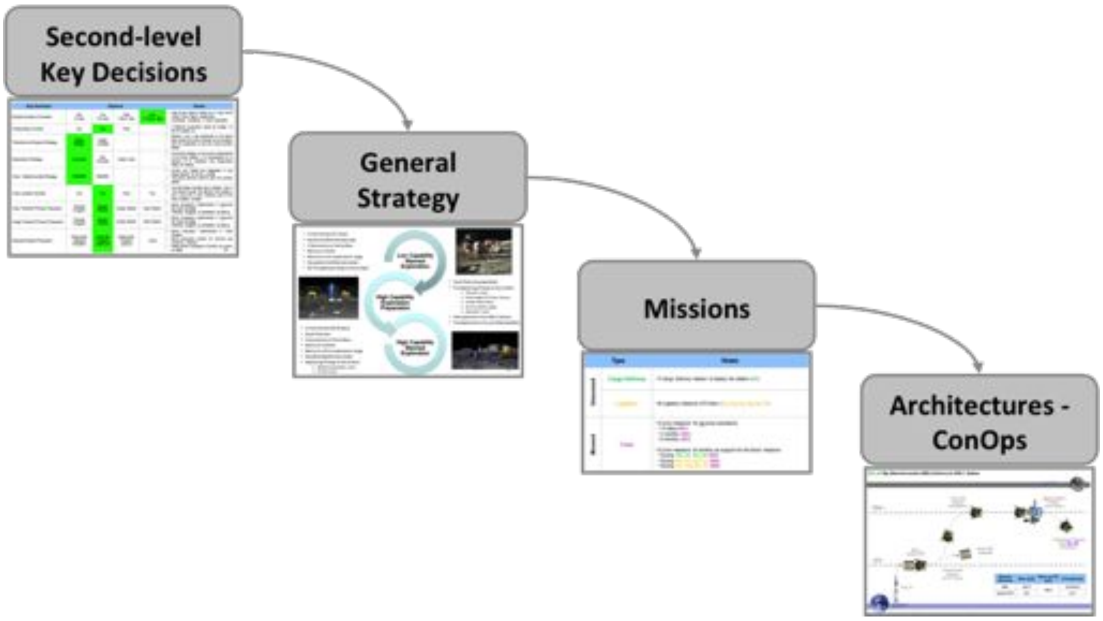


Figure 4.14: Mission Concept Analysis Work Flow

("Second-Level Key Decisions") are qualitatively evaluated in order to select the most significant ones. The second step is the definition of the "General Strategy" to be adopted: the main phases are identified and described. After having defined the general strategy, the type (e.g. manned or robotic) and the minimum total number of missions are determined. At this point, all the architectures corresponding to the identified missions are built, and an assessment of the needed launchers and space elements is performed. This process is applied to all the six mission concepts composing the overall scenario, as described in the following sections.

4.2.3.1 ISS

The ISS concept is the one selected among possible LEO mission concepts. The international space station is envisioned to be exploited as a research and technology laboratory, and to achieve some capabilities required for Mars and here applicable. In particular the ISS exploitation allows the demonstration of the capabilities listed in table 4.18.

Support-In Space	Operations
In-Space multiple dockings	Advanced RvD
In-Space cryogenic fuel management	Medium range communications
In-Space advanced power	Robotics tele-operations
In-Space advanced thermal	
In-Space high capacity storage	
In-Space advanced life support	
In-Space advanced human health support	
In-Space advanced human habitability	
In-Space advanced robotics	
In-Space advanced EVA	

Table 4.18: ISS concept capabilities

The general strategy for the ISS concept includes two different phases:

1. **Technology demo - first phase**, during which the test of several technologies implemented on an ATV-like* or a PMM-like† module is to be performed;

2. **Dedicated Technologies Demo Modules**, during which specific demos are

*Automated Transfer Vehicle (ATV) is an expendable, pressurized unmanned resupply spacecraft developed by the European Space Agency and designed to supply the ISS with propellant, water, air, payloads, and experiments. ATVs can also re-boost the station into a higher orbit.

†Leonardo Permanent Multipurpose Module (PMM) is a module of the ISS, primarily used for storage of spares, supplies and waste on the station.

brought to the station to test specific technologies (e.g. Inflatable, Nuclear Thermal Rocket,...).

A minimum of five missions is derived as needed. In particular they can be classified as *unmanned cargo delivery* mission type, referring to unmanned missions bringing to the station the technologies to be tested. Furthermore, crew missions already in the plans of the agencies will be part of the scenario. The first two missions are supposed to bring specific modules to the ISS implementing more technologies at the same time. In this case, two possible scenarios are envisaged, which are an ATV-like module or a permanent module (PMM-like). The technologies that can be implemented on these modules are listed in table 4.19.

ATV-like	Permanent Module
• Advanced Power (solar arrays, advanced batteries, fuel cells) • Advanced Thermal (advanced HX, heaters, heat-pipes, radiators) • (Advanced Life Support) • Advanced Human Health Support • Advanced Robotics • Advanced EVA (advanced suits, portable life support system, mobility units, EVA tools) • Advanced RvD • Medium Range Comms • Robotics tele-operations	• Advanced Power (solar arrays, advanced batteries, fuel cells) • Advanced Thermal (advanced HX, heaters, heat-pipes, radiators) • (Advanced Life Support) • Advanced Human Health Support • Advanced Robotics • Advanced EVA (advanced suits, portable life support system, mobility units, EVA tools) • Advanced RvD • Medium Range Comms • Robotics tele-operations

Table 4.19: ISS concept capabilities

The other three missions are meant to bring to the ISS three specific demo modules, that are Cryogenic Fuel Tank, Nuclear Thermal Rocket Demo and Inflatable Demo. Summarizing, the following five Cargo Delivery missions are envisioned (the abbreviations used to identify them are highlighted):

1. ISS_U1, bringing an ATV-like module in 2014-2016 timeframe;
2. ISS_U2, bringing a permanent module in 2014-2016 timeframe;
3. ISS_U3, bringing an Inflatable Demo in 2015;
4. ISS_U4, bringing Cryogenic Fuel Tank around 2015-2016;
5. ISS_U5, bringing the NTR demo in 2015-2016 timeframe.

Three new architectures are defined for the just listed missions. The first architecture refers to the ATV-like mission: the sequence of operations is schematically shown in figure 4.15. The mission profile shall be analogous to a typical ATV mission; the vehicle

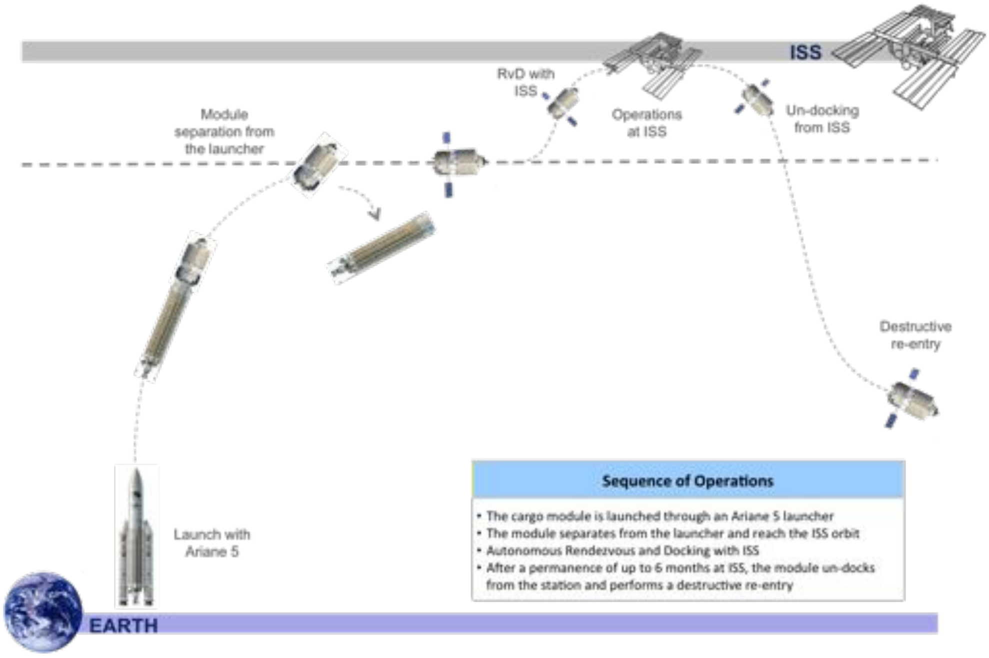


Figure 4.15: ISS concept - mission architecture 1

is launched by means of an Ariane 5, it performs rendezvous and docking with the ISS, and after several weeks spent at the station it performs a destructive re-entry.
The second mission architecture refers to the PMM-like mission: the sequence of operations is schematically shown in figure 4.16. According to this concept of operations, the PMM-like module is launched through an Ariane 5 attached to a space tug, which is in charge of performing the RvD maneuvers with ISS. The module will remain attached to the station for a long period to allow different technological tests.
The last mission architecture foresees specific demo modules to be launched and attached to the ISS: the reference sequence of operations is schematically shown in figure 4.17. According to this concept of operations, the demo module (e.g. inflatable module demo, cryogenic fuel tank demo, or NTR demo) is launched through an Ariane 5 attached to a space tug, which is in charge of performing the RvD maneuvers with ISS. The module will remain attached to the station for a six-months period, after which it will perform a destructive re-entry in the Earth's atmosphere.

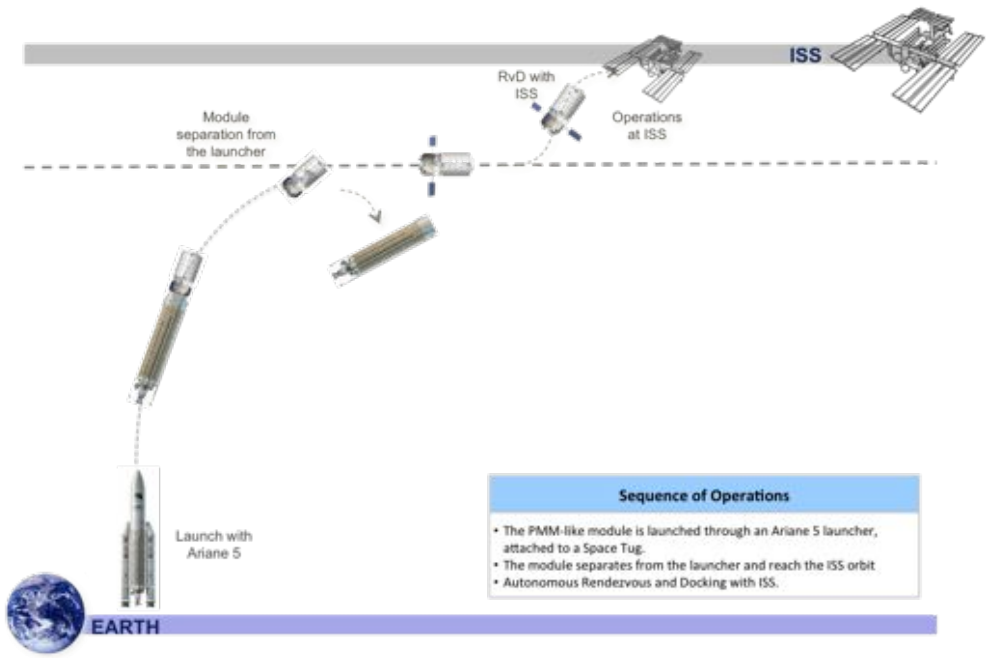


Figure 4.16: ISS concept - mission architecture 2

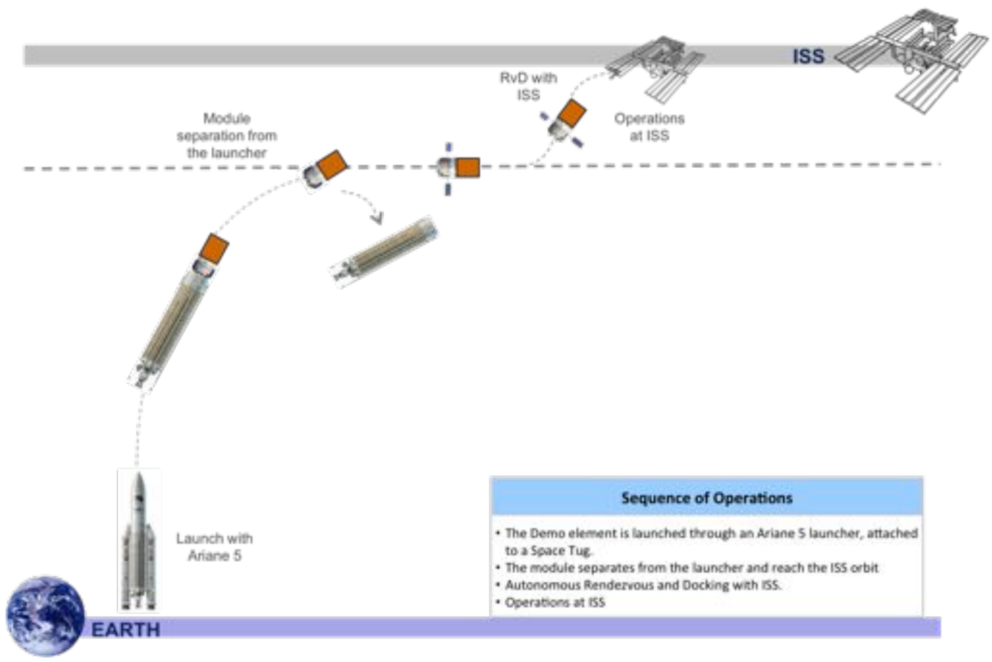


Figure 4.17: ISS concept - mission architecture 3

According to these mission architectures, five Ariane 5 ES launchers and six different elements are needed to accomplish the ISS concept missions, which are (number of needed units is reported in brackets):

- Transportation Elements
  - Cryogenic fuel tank demo (one unit)
  - NTR demo (one unit)
  - Space tug (five units)
- In-space elements
  - inflatable demo (one unit)
  - PMM-like module (one unit)
  - ATV-like module (one unit)

All these elements can be further classified as "New Project", "Upgraded Versions" and "Already Used" elements, with respect to previous steps of exploration. Obviously, since the ISS concept is the first appearing in the reference scenario, the upgraded versions refer to already in use modules. This allows easily visualizing and validating the approach adopted in the definition of the missions and of the whole scenario. The pie chart reported in figure 4.18 summarizes the number of elements needed for the ISS concept's missions, highlighting their design status: green, yellow and red colors are used to indicate already used, upgraded version and new project, respectively.

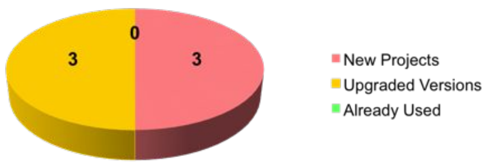


Figure 4.18: ISS concept - mission elements design status

4.2.3.2 Cis-lunar

The concept selected among the possible cis-lunar mission concepts is the CL2, which envisages a men-tended infrastructure deployed in the first Earth-Moon lagrangian point and aimed at supporting future exploration missions, for instance, by providing a staging post for expeditions to Moon's surface. This concept allows the demonstration of the capabilities listed in table 4.20. The definition of the missions and architectures

Transportation	Support-In Space	Operations
High performance human transfer	In-Space multiple dockings	Advanced RvD
High speed Earth manned EDL	In-Space cryogenic fuel management	Long range communications
High capacity cargo transfer	In-Space advanced power	Medium range communications
Orbit cargo insertion	In-Space advanced thermal	Robotics tele-operations
Orbit manned insertion	In-Space high capacity storage	Safe in-space elements separation
	In-Space advanced life support	
	In-Space advanced human health support	
	In-Space advanced human habitability	
	In-Space radiation protection	
	In-Space advanced robotics	
	In-Space advanced EVA	

Table 4.20: Cis-lunar concept capabilities

for the Cis-Lunar case starts from the identification and evaluation (qualitative) of specific "Second-Level Key Decisions" (as previously explained). For each key decision a specific option is selected, according to the philosophy behind the study and taking in mind the final objective of the human mission to Mars (NASA DRA 5.0). The key decisions for the Cis-Lunar destination are summarized in table 4.21, in which the alternative options are shown, as well as the justification of the final choice. In

Key decision	Options				Notes
Number of human missions	3	**6**	>6		Six manned missions are considered: the first three (increasing durations) for research and technologies tests, the other three (6 months) in support of the Moon missions
Crew members	2	3	**4**	>4	Crew size of four is considered, since it is representative of a Moon mission.
Cargo in-space propulsion	**CPS**	NTR	SEP		CPS is chosen because it is considered too challenging to have NTR (high capacity required) available for 2017, when the station is envisioned to be deployed.
Crew in-space propulsion	**CPS**	**NTR**			CPS is initially adopted, while NTR is implemented in the later missions (after having been tested and implemented in the logistics missions).
Logistics in-space propulsion	CPS	**NTR**			NTR is adopted for the logistics missions which represent the first possibility to implement and get that capability (low capacity NTR).

Table 4.21: Cis-lunar "Second-Level Key Decisions"

summary, six manned missions with a crew of four astronauts are considered. For what concerns the in-space propulsion, cryogenic propulsion is adopted for the station delivery to EML1 and for the first manned missions. Nuclear propulsion is instead adopted for all the logistics missions and for the last crew missions.

The following step of the analysis is the assessment of the mission strategy, which for the Cis-Lunar case foresees three main phases:

1. **Autonomous operations phase**, which starts with the deployment of the station in EML1, relying on cryogenic propulsion; during this phase of autonomous operations (before the first crew visit), the station is used for research (scientific experiments operated from ground) and technologies test;

2. **Men-Tended Cis-Lunar Operations phase**, during which three manned missions of increasing duration are envisaged and, besides scientific research and technologies tests activities, tele-operations of robotic assets on the Moon's surface is envisioned;

3. **Moon Missions Support phase**, which envisages three manned missions, in support of the Moon expeditions, in particular to perform tele-operation activities of robotic assets on the Moon's surface and provide support for the Moon base deployment and activation, as well as to support crew operating on the Moon's surface.

A minimum of 13 missions is derived as needed. In particular they can be divided into three different mission types:

- Unmanned cargo delivery mission, which refers to the unmanned mission for the delivery of the Cis-Lunar station to EML1;

- Unmanned logistics missions needed for the resupply of the station (six missions are assumed in correspondence of crew missions);

- Crew missions, which represent the crew visits at the station (six total missions).

For these three types of mission, four different architectures are identified.
The first architecture refers to the cargo delivery mission. The sequence of operations is schematically shown in figure 4.19. The transfer stage utilizes cryogenic propulsion, to insert the station (EML1-HAB) in the transfer trajectory towards EML1, while a service module attached to the habitat (HAB-SM) is in charge of Halo orbit insertion and station keeping. In the pictures illustrating the mission architectures, the masses of the various elements are indicated, as well as the ΔV provided by the propulsive stages. The propellant masses are evaluated using the classical Tsiolkovsky rocket

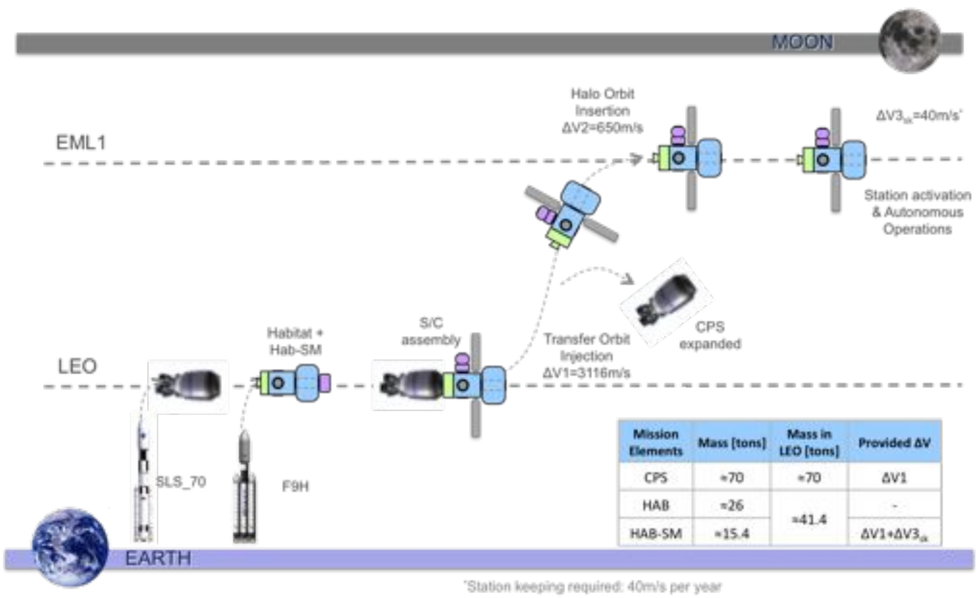


Figure 4.19: Cis-lunar concept - mission architecture 1

equation [17]*:

$$M_{prop} = M_{fin} \left[exp \left(\frac{\Delta V}{I_{sp} g_0} \right) - 1 \right] \tag{4.1}$$

where for the HAB-SM and the CPS the dry mass are assumed equal to 5 and 7 t respectively, and the I_{sp} is assumed 326 and 465 s respectively.

For what concerns the crew missions, two architectures are derived, as shown in figures 4.20 and 4.21, implementing cryogenic and nuclear propulsion, respectively.

Even in this case, the classical rocket equation is adopted for the evaluation of the initial mass in LEO. The computations are done assuming:

- CEV mass equal to 9t,
- CPS dry mass equal to 7t,
- NTR dry mass equal to 10t and $I_{sp} = 900s$.

The first two human missions are assumed to implement cryogenic propulsion, since it appears quite unlikely to have nuclear thermal rockets available for manned missions in 2018. Moreover it is assumed that before implementing nuclear propulsion in

*The same approach is followed for the analysis of all the other concepts (described in the following sections).

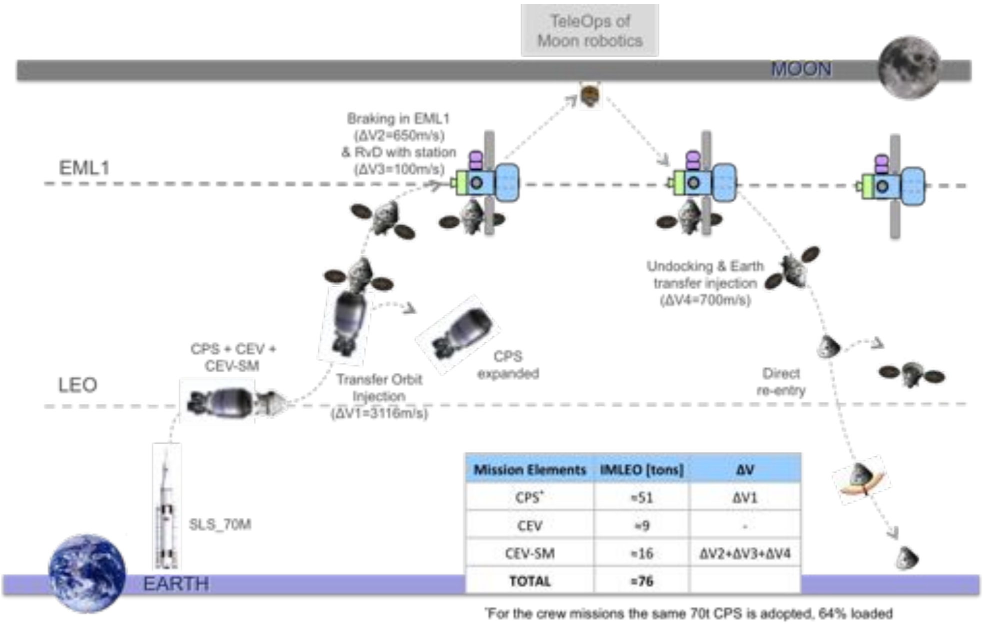


Figure 4.20: Cis-lunar concept - mission architecture 2

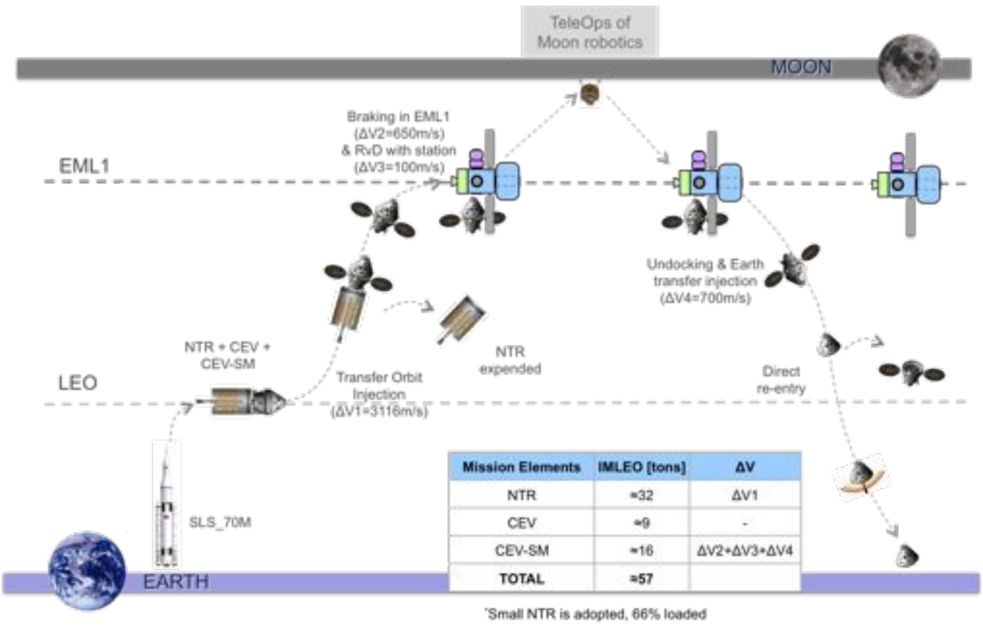


Figure 4.21: Cis-lunar concept - mission architecture 3

crewed missions, some experience shall be gained in unmanned missions (e.g. logistics missions). The following missions (starting from 2020) instead implement nuclear

propulsion, after having been tested and implemented in the unmanned logistics missions. The crew missions rely on the use of a Crew Exploration Vehicle (CEV)-like system with its service module.
The last identified architecture is shown in figure 4.22 that reports the sequence of operations for the logistics missions. The logistics delivery module is assumed to be an ATV-like system. This architecture envisages the use of a Nuclear Thermal Rocket (NTR) since the first mission, in order to validate this technology.

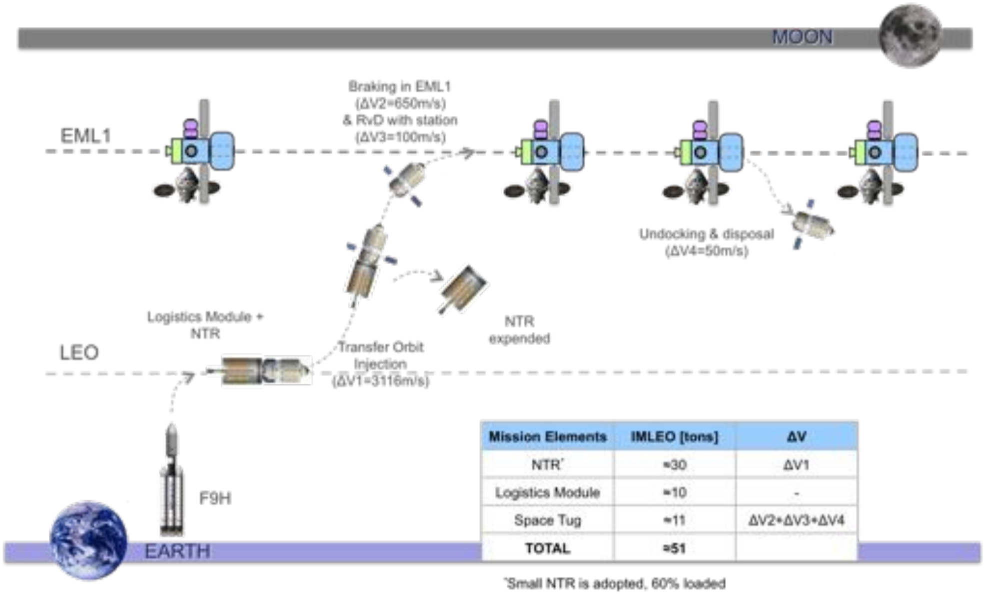


Figure 4.22: Cis-lunar concept - mission architecture 4

According to these mission architectures, ten different elements are needed to accomplish the CL2 concept missions, which are (number of needed units is reported in brackets):

- Transportation Elements
  - Habitat-service module (one unit)
  - CEV-service module (six units)
  - CEV (six units)
  - CPS (three units)
  - Small NTR (ten units)
  - Space tug (six units)

- In-space elements
  - Cis-lunar habitat (one unit)
  - Airlock (one unit)
  - Logistics module (six units)
  - Robotic arm (one unit)

These elements can be further classified as “New Project”, “Upgraded Versions” and “Already Used” elements, with respect to previous steps of exploration (in this case the ISS concept): the graph in figure 4.23 summarizes the number of elements, highlighting their design status (green, yellow and red colors are used to indicate already used, upgraded version and new project, respectively).

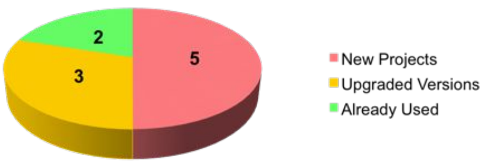


Figure 4.23: Cis-lunar concept - mission elements design status

4.2.3.3 Moon Sortie

The selected Moon sortie mission concept is the MS3, which envisages a staging in Cis-lunar, exploiting the station here deployed. Long surface staying up to 45 days and long exploration ranges up to 10 km from the landing site are envisioned. Cargo assets are assumed to be pre-deployed with dedicated unmanned missions. This concept allows the demonstration of the capabilities listed in table 4.22.

The process of analysis of the Moon sortie case for the definition of the missions and the architectures starts from the identification and evaluation (qualitative) of specific “Second-Level Key Decisions” (as previously explained). For each key decision a specific option is selected, according to the philosophy behind the study, taking in mind the final objective of the human mission to Mars (NASA DRA 5.0). The key decisions for the Moon sortie destination are summarized in table 4.23, in which the alternative options are shown, as well as the justification of the final choice.

Transportation	Support-In Space	Support - Surface Operations	Operations
High performance human transfer	In-Space multiple dockings	Surface multiple dockings	Advanced RvD
High speed Earth manned EDL	In-Space cryogenic fuel management	Surface cryogenic fuel management	Long range communications
High capacity cargo transfer	In-Space advanced power	Surface advanced power	Medium range communications
Destination cargo D&L	In-Space advanced thermal	Surface advanced thermal	Short range communications
Destination manned D&L	In-Space high capacity storage	Surface advanced life support	Reduced gravity anchoring, drilling..
Destination manned ascent	In-Space advanced life support	Surface advanced human habitability	Robotics tele-operations
	In-Space advanced human health support	Surface radiation protection	Safe in-space elements separation
	In-Space advanced human habitability	Surface advanced robotics	
	In-Space radiation protection	Soil ISRU	
	In-Space advanced robotics	Surface advanced EVA	

Table 4.22: Moon sortie concept capabilities

Key decision	Options				Notes
Sortie number & duration	One, 45 days	Two, 7-45 days	Three, 7-28-45 days	**Four, 3-7-28-45 days**	High public opinion impact of a 3-days short human Moon return (Apollo-like) and incremental confidence in Moon exploration
Visited sites numbers	1	**2**	3		Two different exploration areas for sorties 1-2 and sorties 3-4
Transition to outpost strategy	**Sortie related**	Sortie unrelated			Sortie missions 3 and 4 are performed in the area where the outpost is to be located (possibility to set and reuse surface assets)
Exploration strategy	**Commuter**	Tele-commuter	Mobile home		Commuter strategy is analogous to Mars mission (balance of fix habitation and pressurized rovers)
Crew habitat/lander strategy	**Integrated**	Separated			Lander and habitat are integrated in one single element for all the sortie missions; the same element performs surface ascent.
Crew landers number	One	**Two**	Three	Four	First lander is part of sortie missions 1 and 2, while the second one is used in sorties 3 and 4, plus two crew missions of the Moon outpost concept .
Crew transfer primary propulsion	Chemical cryogenic	**Nuclear thermal**	Nuclear electric	Solar electric	Same propulsion as cis-lunar concept (chemical cryogenic as back-up).
Cargo transfer primary propulsion	Chemical cryogenic	**Nuclear thermal**	Nuclear electric	Solar electric	Same propulsion as cis-lunar concept (chemical cryogenic as back-up).
Descent/Ascent propulsion	Pressure-fed hypergolic NTO/MMH	**Pump-fed cryogenic LOX/LCH4**	Pressure-fed cryogenic LOX/LH2	Hybrid	Same propulsion as Mars concept for both manned and unmanned missions (pressure-fed hypergolic NTO/MMH as back-up).

Table 4.23: Moon sortie “Second-Level Key Decisions”

In summary, four missions of increasing durations (3, 7, 28 and 45 days) on two different exploration sites and utilizing two crew landers are envisioned. The last two sortie missions are to be performed on the same area where the Moon outpost has to be located, in order to have the possibility of re-using same surface assets. Concerning the exploration strategy, a commuter strategy is adopted which is based on a balance of fix habitation and pressurized rovers for mobility. The lander and the habitat are

integrated in the same element, which also performs the ascent maneuver. For what concerns the propulsion, nuclear thermal propulsion is adopted for crew and cargo primary propulsion, while pump-fed cryogenic LOX/LCH4 is used for the descent/ascent propulsion.
The following step of the analysis is the assessment of the mission strategy, which for the Moon sortie case foresees three main phases:

1. **Low capability manned exploration phase**, which includes two crew sortie missions (3 - 7 days) at equatorial/mid-latitude sites, with two astronauts on the lunar surface performing up to 4 EVAs; in this phase the exploration range is limited to 3km;

2. **High capability exploration preparation phase**, which refers to the phase of cargo pre-deployment on the Moon's surface, while tele-operation activities are performed from the EML1 station;

3. **High capability manned exploration phase**, which includes two crew sortie missions (28 - 45 days) at south pole sites, with three astronauts on the lunar surface performing up to 20 EVAs; in this phase the exploration range is up to 15 km.

A minimum of 17 missions is derived as needed. In particular they can be divided into three different mission types:

- Unmanned cargo delivery mission, which refers to the unmanned missions for the delivery of the lander to the cis-Lunar station, the lunar relay satellites in LLO and surface assets to Moon south pole (seven missions);

- Unmanned logistics missions, needed for the resupply of the station to support astronauts during Moon exploration missions and for the delivery of the fuel needed for the manned landers resupply (six missions);

- Crew missions, which represent the crew surface exploration missions (four missions).

For these three mission types, eight different architectures are defined.
The first architecture refers to the first type of cargo delivery mission (delivery of Small Manned Lander to EML1 station): the sequence of operations is schematically shown

in figure 4.24.

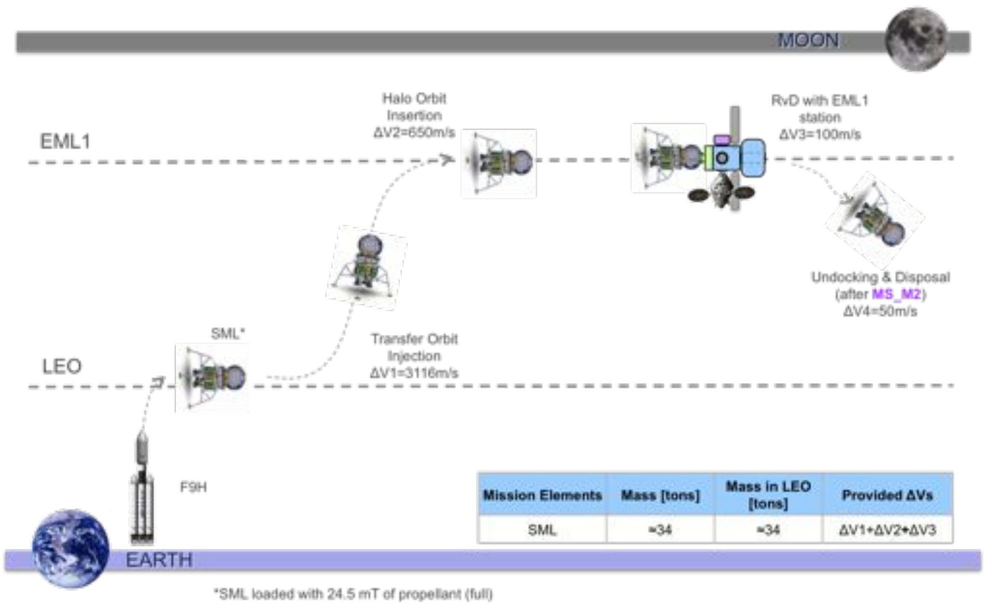


Figure 4.24: Moon sortie concept - mission architecture 1

The SML is launched in LEO by means of a Falcon 9 heavy and is in charge of the transfer and rendezvous and docking maneuvers with the EML1 station.

The second architecture refers to the second type of cargo delivery mission (Big Manned Lander delivery to EML1 station): sequence of operations is schematically shown in figure 4.25. The BML is launched in LEO by means of an SLS_70 attached to a small nuclear thermal rocket (85% filled with fuel) which is in charge of providing the first ΔV to inject the lander in the transfer trajectory. In its launch configuration the BML is only 10% loaded with fuel, being the refueling operations foreseen during its permanence at the station; it is in charge of performing the braking and RvD maneuvers with EML1 station.

The third architecture refers to the mission for the delivery of Lunar Relay Satellites: the sequence of operations is schematically shown in figure 4.26. The satellites are launched attached to a space tug (63% fuel loaded) and a small nuclear thermal rocket (69% fuel loaded). The nuclear stage is responsible for the first ΔV, to inject the spacecraft into the transfer trajectory, while the space tug performs the other maneuvers (LLO insertion). The two satellites are on the same orbit but 180deg phased. They have to support the *High Capability Exploration Phase* and the *Moon Outpost* concept

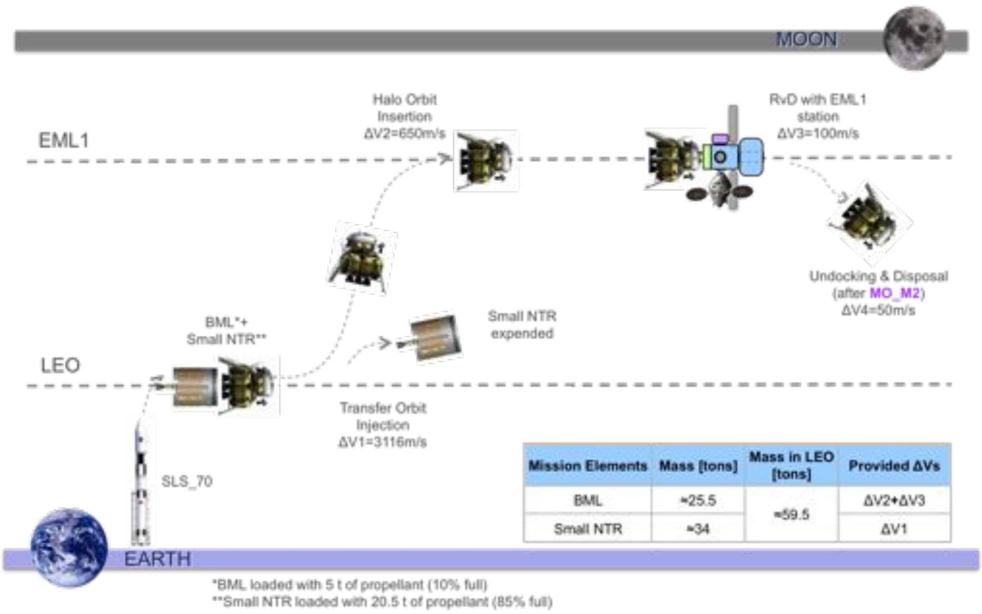


Figure 4.25: Moon sortie concept - mission architecture 2

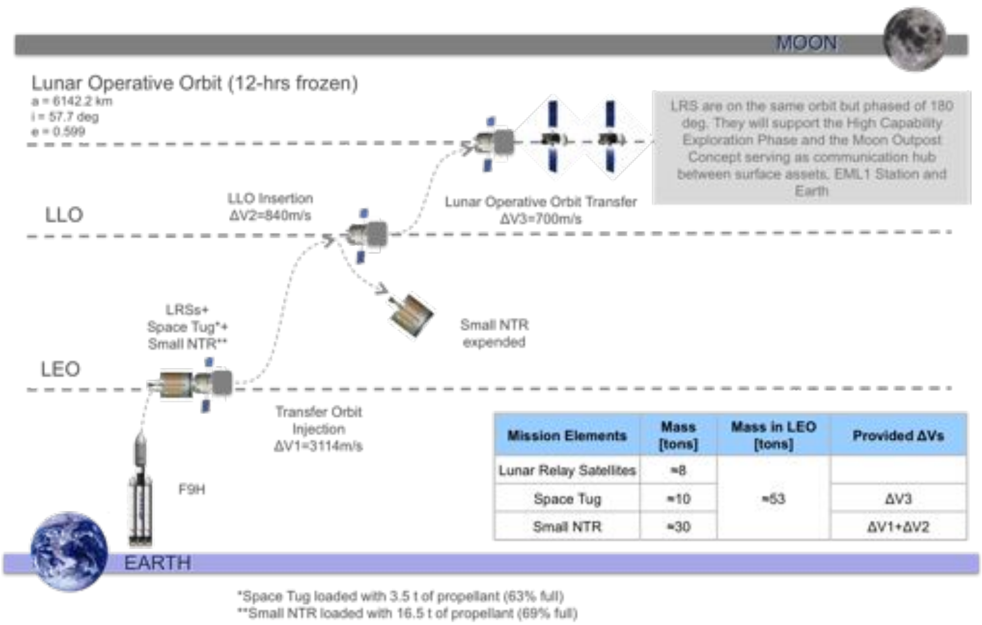


Figure 4.26: Moon sortie concept - mission architecture 3

serving as communication hub among surface assets, EML1 Station and Earth.

The fourth architecture refers to the cargo mission for the delivery of two precursor rovers on Moon's surface, that are in charge of analyzing and preparing the site for the following cargo and human landings: the sequence of operations is schematically shown in figure 4.27. The rovers, loaded on a 1-ton lander, are launched with an A5_ME,

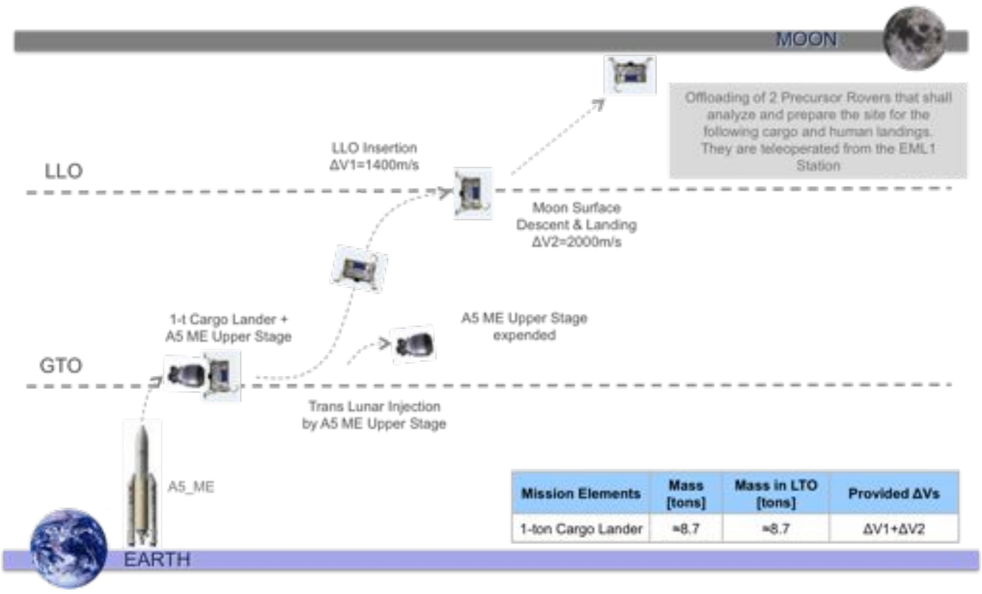


Figure 4.27: Moon sortie concept - mission architecture 4

whose upper stage provides the first ΔV to insert the spacecraft into the transfer trajectory. The lander is in charge of the other maneuvers (LLO insertion and descent and landing on the Moon's surface).

The fifth architecture refers to the other cargo delivery missions, necessary for the delivery of robotic assets to Moon south pole: the sequence of operations is schematically shown in figure 4.28. An 8-tons cargo lander is used to carry cargo assets on the Moon's surface. It is launched by means of a F9H attached to a small nuclear thermal rocket (77% fuel loaded), which provides the first burn, while the other maneuvers are performed by the lander itself. The utility cart moves, deploys and sets the power assets and the lunar communication terminal. The crew supports these activities through tele-operations from EML1 Station.

The sixth architecture refers to the logistics missions, for the delivery to EML1 station of the fuel needed for the landers: sequence of operations is schematically shown in figure 4.29. For the fuel delivery mission two SLS_70 launches are needed, to bring in LEO a fuel tank with a space tug and the small NTR with the LH2 tank. Once assembled

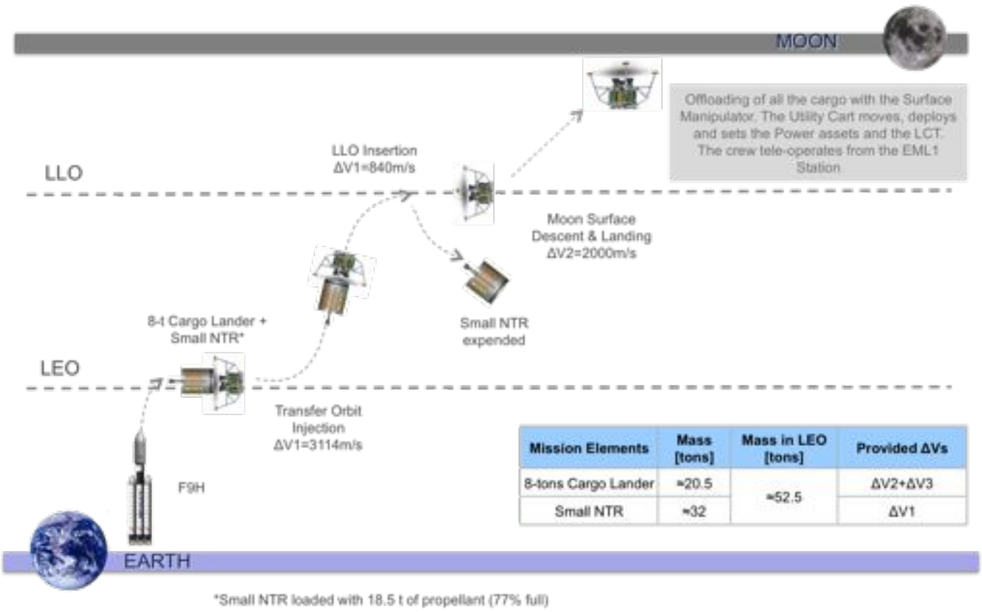


Figure 4.28: Moon sortie concept - mission architecture 5

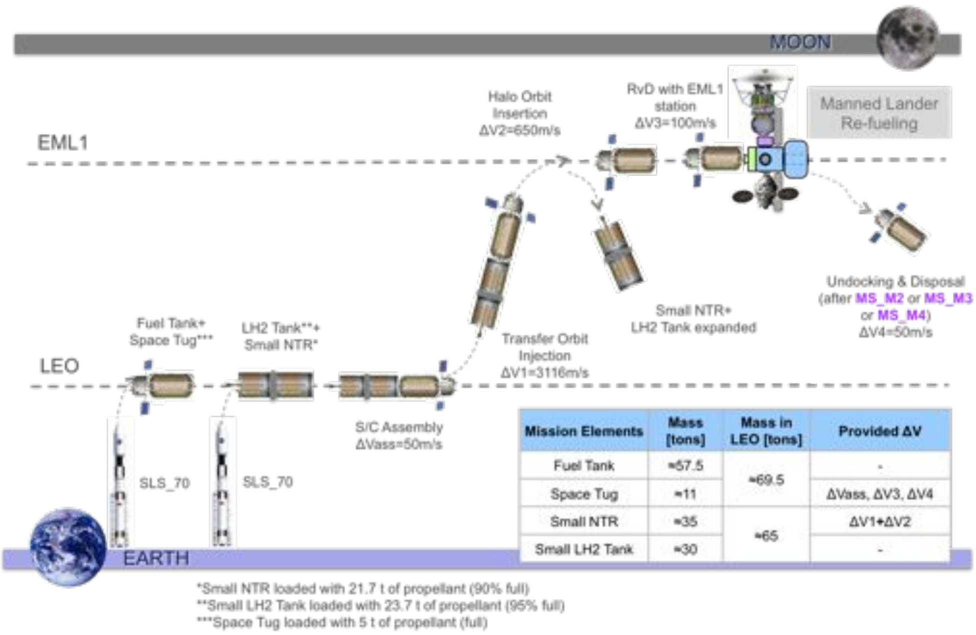


Figure 4.29: Moon sortie concept - mission architecture 6

in LEO the spacecraft transfers to EML1 station, being the maneuvers performed by the NTR using the propellant stored in its tank. The space tug is instead in charge of

the RvD maneuvers with the station. Once the manned lander refueling is completed, the space tug performs the un-docking maneuvers and the tank is expended.
The seventh architecture refers to the first two crew missions, which last 3 and 7 days, respectively, relying on the SML: sequence of operations is schematically shown in figure 4.30, which focuses on the Moon exploration mission phase. The crew arrives to EML1

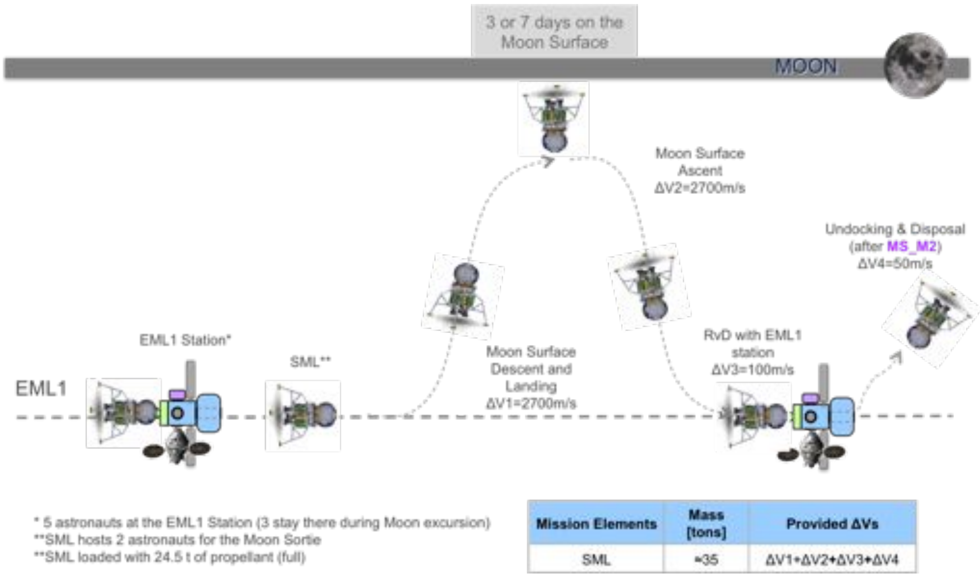


Figure 4.30: Moon sortie concept - mission architecture 7

station on board a Crew Exploration Vehicle and transfers to the SML. The lander undocks from the station and performs the descent and landing maneuvers. After a few days of surface activities, the lander comes back to the station, the crew transfers to the capsule, undocks and comes back to Earth. The SML is expended when both the missions have been completed.
The eighth architecture refers to the last two crew missions, which last 28 and 45 days, respectively, relying on the BML: sequence of operations is schematically shown in figure 4.31. The architecture is analogous to the previous one, but BML is used instead of SML.
According to these mission architectures, 25 different elements are needed to accomplish the MS3 concept missions, which are (number of needed units is reported in brackets):

- Transportation Elements
  - 1-ton lander (one unit)

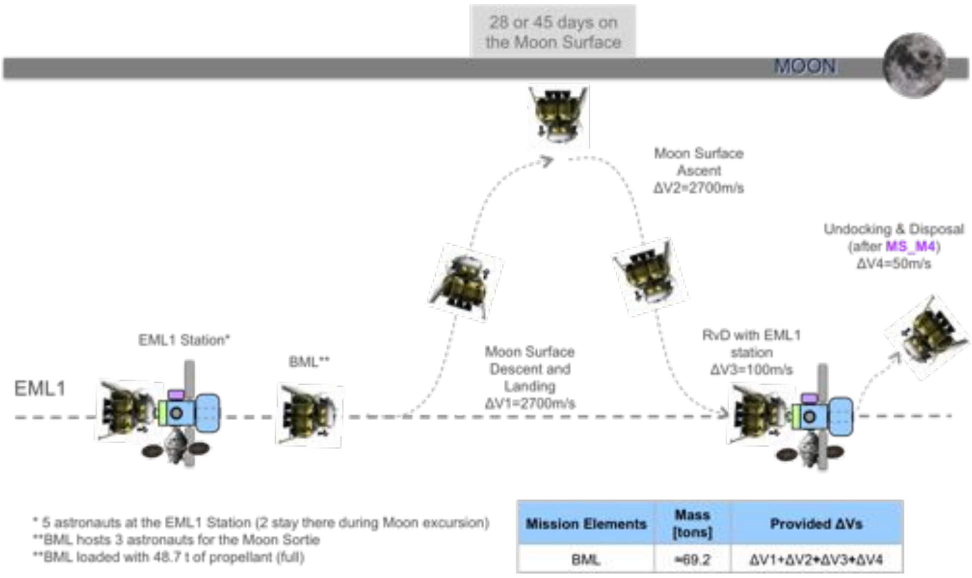


Figure 4.31: Moon sortie concept - mission architecture 8

  - Small manned lander (one unit)
  - Small LH2 tank (three units)
  - Big manned lander (one unit)
  - 8-tons lander (three units)
  - CEV-service module (six unit)
  - CEV (six units)
  - Small NTR (15 units)
  - Space tug (seven units)

- In-space elements
  - Fuel tank (three units)
  - Lunar relay satellite (two units)
  - Logistics module (three units)

- Surface elements
  - Suit port + EVA systems (two units)
  - Unpressurized rover (two units)
  - Manipulator (two units)
  - FSPS demo (one unit)
  - SolPS (two units)

– Utility cart (two units)
– ISRU demo (one unit)
– Pressurized rover demo (one unit)
– Small traverses caches (two units)
– Airlock + EVA systems (one unit)
– Precursor rover (two units)
– Small exploration rover (one unit)
– Pressurized rover (one unit)

These elements can be further classified as "New Project", "Upgraded Versions" and "Already Used" elements, with respect to previous steps of exploration (ISS and cis-lunar concepts): the pie chart reported in figure 4.32 summarizes the number of elements, highlighting their design status (green, yellow and red colors are used to indicate already used, upgraded version and new project, respectively).

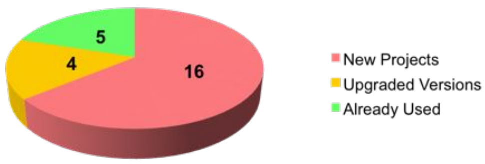


Figure 4.32: Moon sortie concept - mission elements design status

4.2.3.4 Moon Outpost

The concept selected among the possible Moon outpost mission concepts is the MO3, which is an evolution of the Moon sortie concept, with missions performed to the same site of the last two Moon sortie missions (south pole) and foreseeing the reuse of part of Moon sortie surface assets. The MO3 concept envisages a staging in Cis-lunar, exploiting the EML1 station. Long surface staying up to 600 days and long exploration ranges up to 150 km from the landing site are envisaged; cargo assets are assumed to be pre-deployed with dedicated unmanned missions. This concept allows the demonstration of the capabilities listed in table 4.24.
Analogously to what done for the previous concepts, the analysis for the definition of

Transportation	Support-In Space	Support - Surface Operations	Operations
High performance human transfer	In-Space multiple dockings	Surface multiple dockings	Advanced RvD
High speed Earth manned EDL	In-Space cryogenic fuel management	Surface cryogenic fuel management	Long range communications
High capacity cargo transfer	In-Space advanced power	Surface advanced power	Medium range communications
Destination cargo D&L	In-Space advanced thermal	Surface advanced thermal	Short range communications
Destination manned D&L	In-Space high capacity storage	Surface advanced life support	Reduced gravity anchoring, drilling..
Destination manned ascent	In-Space advanced life support	Surface advanced human health support	Robotics tele-operations
	In-Space advanced human health support	Surface advanced human habitability	Safe in-space elements separation
	In-Space advanced human habitability	Surface radiation protection	
	In-Space radiation protection	Surface advanced robotics	
	In-Space advanced robotics	Soil ISRU	
	In-Space advanced EVA	Surface advanced EVA	

Table 4.24: Moon outpost concept capabilities

the missions and the architectures of the Moon outpost case starts from the identification and evaluation (qualitative) of specific "Second-Level Key Decisions" (as previously explained), for which specific options are selected. The key decisions for the Moon outpost concept are summarized in table 4.25, in which the alternative options are shown, as well as the justification of the final choices. In summary, two missions lasting 180 and 540 days and characterized by a crew of four and six astronauts respectively, are envisioned. A commuter strategy is adopted, analogously to what done for the Moon sortie missions. The first mission relies on BML both as Lander and as Habitat, while the second mission, which is the Mars analog, uses BML as lander and a dedicated habitat as habitation module, with no logistics missions foreseen on the surface. For what concerns the propulsion, nuclear thermal propulsion is adopted for crew and cargo primary propulsion, while pump-fed cryogenic LOX/LCH4 is used for the descent/ascent phases. Soil ISRU is foreseen and nuclear fission and photovoltaic systems are implemented as primary power storage.
The mission strategy for the Moon outpost case includes four main phases:

1. **First visit preparation at EML1 station phase**, which foresees a mission to the EML1 station (in 2022), with four astronauts and lasting up to 180 days to perform preparation activities in view of the following mission on the Moon's surface (BML preparation);

2. **Mars analog preparation at Moon's surface phase**, which represents the

Key decision	Options				Notes
Crew visits number & duration	One, 540 days	**Two, 180-540 days**	Three, 180-360-540 days		First mission as demonstration of long duration Moon permanence and preparation for the Mars analog mission, which is the second mission.
Crew members on surface	3	**4**	5	**6**	First mission with four crew members , second mission (Mars analog) with six crew members
Exploration strategy	**Commuter**	Tele-commuter	Mobile home		Same commuter strategy as Moon sortie and Mars missions
Crew habitat/lander strategy	**Integrated**	**Separated**			First mission relies on BML as both lander and habitat, while second mission uses BML as lander and dedicated habitat for habitation
Surface logistics missions	Yes	**No**			
Crew transfer primary propulsion	Chemical cryogenic	**Nuclear thermal**	Nuclear electric	Solar electric	Same propulsion as cis-lunar and Moon sortie concepts (chemical cryogenic as back-up).
Cargo transfer primary propulsion	Chemical cryogenic	**Nuclear thermal**	Nuclear electric	Solar electric	Same propulsion as cis-lunar and Moon sortie concepts (chemical cryogenic as back-up).
Descent/Ascent propulsion	Pressure-fed hypergolic NTO/MMH	**Pump-fed cryogenic LOX/LCH4**	Pressure-fed cryogenic LOX/LH2	Hybrid	Same propulsion as Moon sortie and Mars concepts for both manned and unmanned missions (pressure-fed hypergolic NTO/MMH as back-up).
Soil ISRU	**Yes (oxidizer and consum-ables)**	Yes (only oxidizer)	Yes (only consum-ables)	No	Technologies similar to those foreseen for Mars atmospheric ISRU are implemented (oxidizer not used for ascent)
Primary power strategy	**Nuclear fission and PV**	Only nuclear fission	Only PV		Same strategy as Mars concept

Table 4.25: Moon outpost "Second-Level Key Decisions"

first crew visit to the Moon Outpost (in 2029), for an overall duration of 180 days, with two astronauts on the surface and two remaining onboard EML1 station; up to 25 EVAs with an exploration range up to 50km are envisioned;

3. **Mars analog preparation at EML1 station phase**, which includes a mission to the EML1 station (in 2031), with four astronauts and lasting up to 180 days to perform preparation activities in view of the following mission on the Moon's surface (BML preparation);

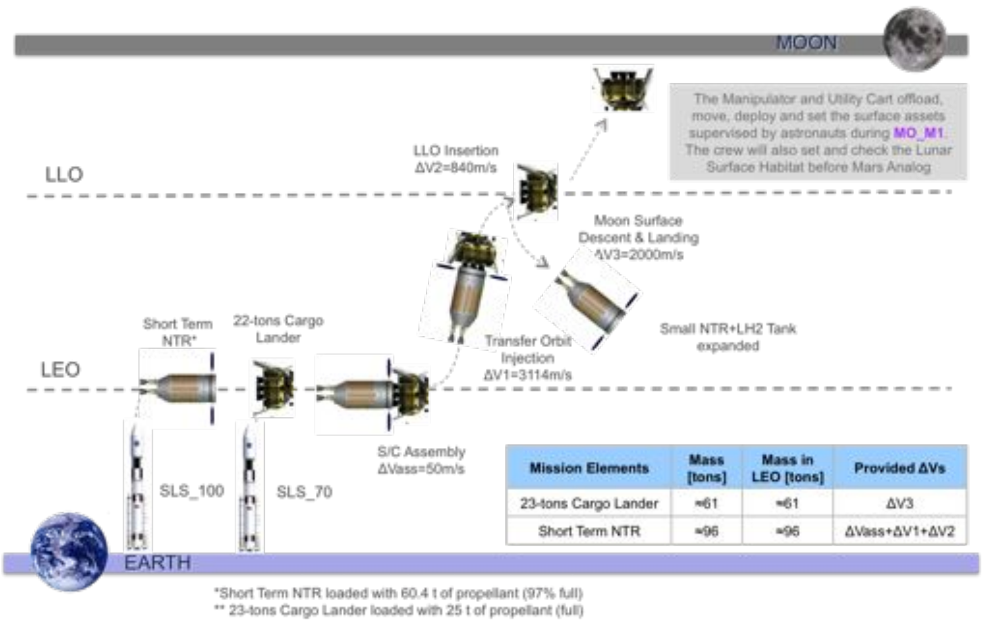
4. **Mars analog phase**, which represents the second crew visit to the Moon outpost (in 2032), for an overall duration of 540 days, with six astronauts on the surface; up to 50 EVAs with an exploration range up to 100km are envisioned.

A minimum of eight missions is derived as needed. In particular they can be divided into three different mission types:

- Unmanned cargo delivery mission, which refers to the unmanned mission for the delivery of the surface assets to Moon south pole (two missions);

- Unmanned logistics missions, needed for the delivery to EML1 station of the resources and fuel needed for the BML (four missions);
- Crew missions, which represent the crew surface exploration missions (two missions)*.

For these three mission types, one different new mission architecture is identified. It refers to the Moon outpost cargo delivery mission and is schematically illustrated in figure 4.33. The robotic assets to be deployed on the Moon's surface are loaded in a 23-tons cargo lander, which is launched by an SLS_70. Another launch (SLS_100) is needed to bring in LEO a short term NTR (97% fuel loaded), necessary for the transfer maneuvers, being the lander in charge of the descent and landing phases.



Mission Elements	Mass [tons]	Mass in LEO [tons]	Provided ΔVs
23-tons Cargo Lander	≈61	≈61	ΔV3
Short Term NTR	≈96	≈96	ΔVass+ΔV1+ΔV2

Figure 4.33: Moon outpost concept - mission architecture

According to these mission architectures, 19 different elements are needed to accomplish the MO3 concept missions, which are (number of needed units is reported in brackets):

- Transportation Elements
  - 23-tons lander (two units)

*The other two crew missions to EML1 station foreseen for the preparation activities in view of the surface missions, belong to the cis-lunar concept, where they are accounted for.

  - Short term NTR (two units)
  - Small NTR (six units)
  - Small LH2 tank (two units)
  - CEV-service module (two units)
  - CEV (two units)
  - Space tug (four units)
- In-space elements
  - Fuel tank (two units)
  - Logistics module (two units)
- Surface elements
  - Lunar communication terminal (one unit)
  - Lunar surface habitat (one unit)
  - Manipulator (one unit)
  - FSPS (one unit)
  - Traverses caches (two units)
  - Small ISRU plant (one unit)
  - Suit port + EVA systems (one unit)
  - SolPS (one unit)
  - Pressurized rover (one unit)
  - Airlock + EVA systems (one unit)

Analogously to what done for the precedent concepts, these elements can be further classified as "New Project", "Upgraded Versions" and "Already Used" elements, with respect to previous steps of exploration (ISS, cis-lunar and Moon sortie concepts): the pie chart reported in figure 4.34 summarizes the number of elements, highlighting their design status (green, yellow and red colors are used to indicate already used, upgraded version and new project, respectively).

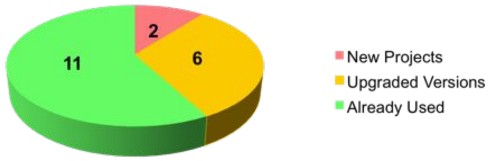


Figure 4.34: Moon outpost concept - mission elements design status

4.2.3.5 Near Earth Asteroid

The concept selected among the possible near Earth asteroid mission concepts is the NEA1. This concept envisages to depart from LEO where the spacecraft is assembled. Cargo assets are deployed to the asteroid with a precursor robotic mission. No landing on the NEA's surface is envisioned, but the spacecraft remains at a certain distance from the asteroid, while surface exploration activities are performed relying on a dedicated exploration vehicle (MMSEV-like vehicle). This concept allows the demonstration of the capabilities listed in table 4.26. Analogously to what done for the previous concepts,

Transportation	Support-In Space	Support - Surface Operations	Operations
High performance human transfer	In-Space multiple dockings	Low-g bodies mobility	Advanced RvD
High speed Earth manned EDL	In-Space cryogenic fuel management		Long range communications
High capacity cargo transfer	In-Space advanced power		Medium range communications
Orbit cargo insertion	In-Space advanced thermal		Short range communications
Orbit manned insertion	In-Space high capacity storage		Low-g bodies anchoring, drilling..
	In-Space advanced life support		Robotics tele-operations
	In-Space advanced human health support		Safe in-space elements separation
	In-Space advanced human habitability		
	In-Space radiation protection		
	In-Space advanced robotics		
	In-Space advanced EVA		

Table 4.26: NEA concept capabilities

the analysis for the definition of the missions and the architectures of the NEA case starts from the identification and evaluation (qualitative) of specific "Second-Level Key Decisions" (as previously explained), for which specific options are selected. The key decisions for the NEA concept are summarized in table 4.28, in which the alternative options are shown, as well as the justification of the final choices.

Key decision	Options				Notes
Mission duration	3-6 months	**6-12 months**	>12 months		Reference mission lasting one year to be more representative.
Crew members	2	3	**4**	>4	Four astronauts is the minimum needed crew size
Crew in-space propulsion	Chemical cryogenic	**Nuclear thermal**	Solar electric		Same propulsion as cis-lunar and Moon concepts (chemical cryogenic as back-up).
Cargo in-space propulsion	Chemical cryogenic	**Nuclear thermal**	Solar electric		Same propulsion as cis-lunar and Moon concepts (chemical cryogenic as back-up).

Table 4.27: NEA "Second-Level Key Decisions"

In summary, a reference mission not exceeding one year with a crew of four astronauts is considered. Concerning the in-space propulsion, nuclear thermal rockets are adopted for both crew and cargo missions, as this technology shall be available and already implemented in previous missions. Besides the "second-level key decisions" listed in table 4.28, additional considerations are necessary, in particular about the mission target asteroid, in order to make some preliminary calculations. In this regard, no specific analyses have been performed, and the NEA selection is mainly driven by the following specific assumptions:

- the mission duration shall not exceed 12 months (mission shorter than Mars one, but significant deep space permanence time),
- the overall required ΔV shall be at maximum 8.5 km/s.

Driven by these assumptions, the 1999 JU3 asteroid (target selected for Hayabusa 2 mission) is taken as reference target. Some of the major features of this body are reported in table 4.28. It allows a human mission in 2033, which is compatible with the overall HSE scenario in which the mission is inserted, with an overall duration not exceeding one year.

Name	Launch year	ΔV [km/s]	Mission duration [days]	Size [m]	Re-entry speed [km/s]
1999 JU3	2033	• Earth departure: 3.5 • NEA braking: 2.3 • NEA departure: 2.7 • Total: 8.5	• Outbound flight time: 217 • Stay time: 8 • Inbound flight time: 129 • Total: 354	254-1134	11.3

Table 4.28: NEA mission target

The selected asteroid requires an overall ΔV of 8.5km/s, which is the maximum reference value assumed to be conservative, even if several less demanding NEAs could have been found.

The mission strategy for the NEA case includes three major phases:

1. **NEA probe missions phase**, during which several probe missions are envisaged to explore and characterize the target asteroid: already planned missions are taken into consideration (e.g. Hayabusa-2, Osiris-Rex,), while no further dedicated assessments are provided;

2. **Precursor Robotic Mission phase**, which includes an unmanned mission for the deployment of cargo assets prior to the human mission (e.g. MMSEV, Robotic assets to support human exploration activities, transponders to support GNC, etc);

3. **Human Exploration phase**, which refers to the human mission, with four crew members and lasting about one year, including about eight days of proximity operations, during which several EVAs are performed on the NEA surface through the MMSEV.

Two missions are derived as the minimum needed, in addition to several probe missions. In particular they can be divided into three different mission types:

- Precursor robotic missions, which are mainly probe missions to take place in the timeframe 2014-2028 and needed for the characterization of the asteroid (no specific evaluations are done to precisely define how many probes missions shall be included);

- Cargo Delivery Mission, which refers to the unmanned mission for the delivery of the cargo assets in the NEA proximity (in 2030-2033);

- Crew mission, which represents the crew visit to the asteroid (in 2033-2035).

For these mission types, two different new architectures are identified. The first one refers to the cargo delivery mission and is schematically illustrated in figure 4.35. The transfer stage utilizes nuclear propulsion (Small Nuclear Thermal Rocket), to insert the spacecraft in the transfer trajectory towards the asteroid, as well as to brake in its proximity. The SNTR provides the first ignition to inject the spacecraft into the NEA transfer orbit. This maneuver is performed by using the propellant stored in the in-line tank. The SNTR is also in charge of providing the ΔV required to insert the spacecraft in the NEA parking orbit. At this point the nuclear stage is expended and the robotics assets are released at the NEA waiting for the crew to arrive. For the robotic mission propellant mass budget the same ΔV values as the manned mission are considered. Further investigation can be done, to find a better solution for the cargo transfer.
Since the propellant of the SNTR has to be stored for several months of travel, an active thermal control system must be included in the SNTR design in order to face the boil-off issue. This is clearly not necessary for the in-line tank, since the stored

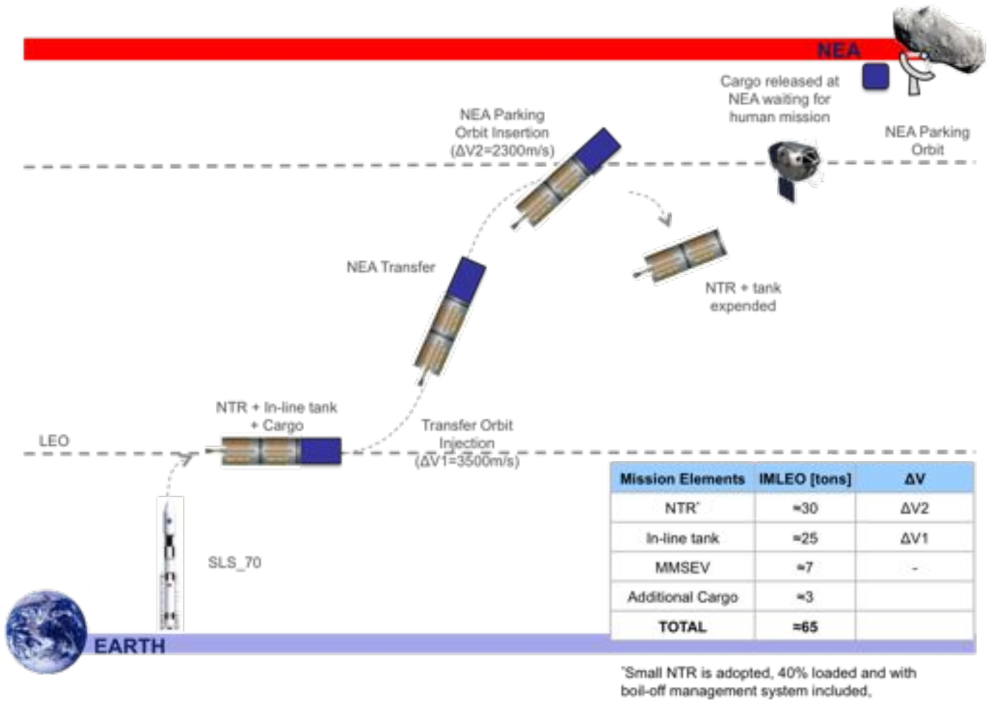


Figure 4.35: NEA concept - mission architecture 1

propellant is used at the beginning of the mission.
The second mission architecture referring to the crew mission is shown in figure 4.36. The spacecraft is assembled in LEO, where the various elements are brought by means of three launches (SLS_100, SLS_130 and At5_M launchers). The Deep Space Habitat is launched already attached to the drop tank; moreover a space tug is attached to the DSH to support the RvD maneuvers for the spacecraft assembly. When the docking between the NTR and the DSH and drop tank assembly is completed the space tug is expended. The last RvD maneuver is finally needed to dock with the CEV - CEV-SM assembly. At this point the spacecraft is completely assembled and the mission can start.
After the system checkout, the NTR provides the first ignition to insert the spacecraft in the transfer trajectory. The propellant necessary for this maneuver is stored in the drop tank, which after the burn is expended. After 217 days of travel, the NTR will provide the second ΔV to insert the spacecraft into the NEA parking orbit.
At this point, eight days will be spent in the NEA proximity and the exploration activities will be carried out by means of the MMSEV. In particular, when the spacecraft is

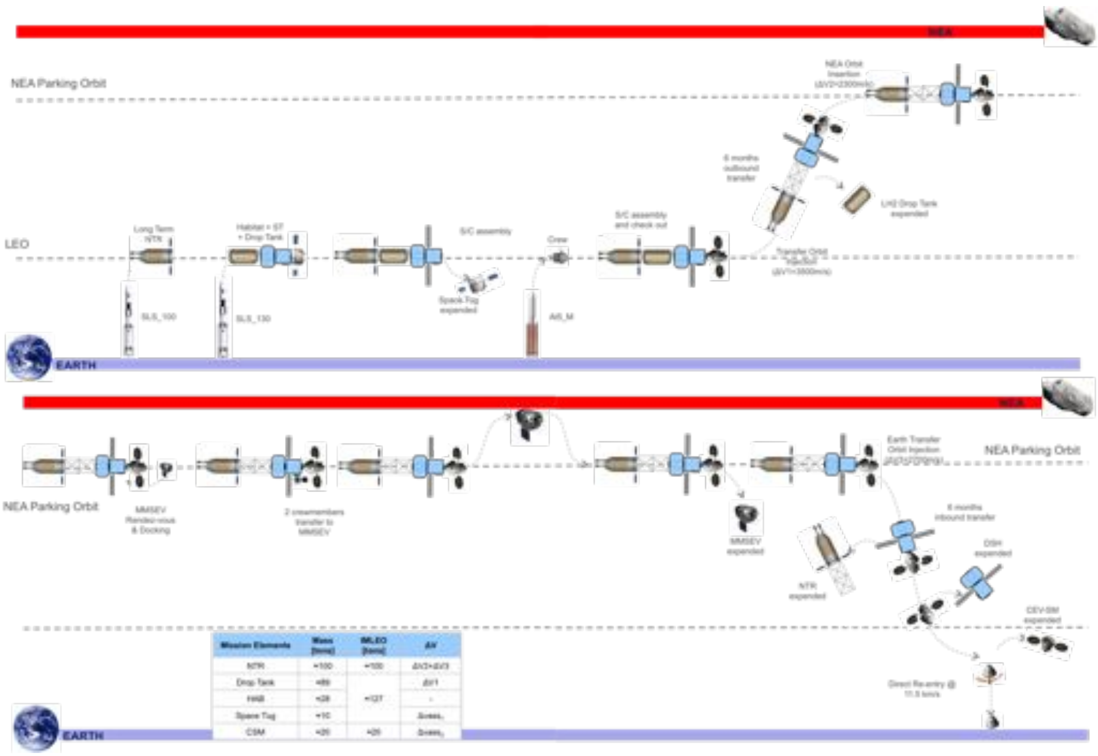


Figure 4.36: NEA concept - mission architecture 2

in the asteroid parking orbit, the MMSEV approaches and docks on the radial docking port of the DSH rigid part, allowing the transfer of two astronauts. Then the MMSEV undocks from the DSH and approaches the asteroid to observe and analyze its surface, as well as to perform EVAs. Several EVAs are envisioned to be performed and the MMSEV shall be capable to perform multiple RvD with the DSH during the NEA proximity operations phase. After eight days, the MMSEV is released and the spacecraft begins its trip back to Earth. The NTR is expended after having provided the last ΔV to insert the spacecraft into the Earth transfer orbit. The mission ends with a direct re-entry of the CEV in the Earth's atmosphere after 129 days of travel.
According to these mission architectures, ten different elements are needed to accomplish the NEA concept missions, which are (number of needed units is reported in brackets):

- Transportation Elements
  - Drop tank (one unit)

  - MMSEV (one unit)
  - Long term NTR (one unit)
  - Small NTR - enhanced (one unit)
  - Small LH2 tank (one unit)
  - CEV-service module (one unit)
  - CEV (one unit)
  - Space tug (one unit)
- In-space elements
  - Deep Space Habitat (one unit)
  - Suitport (one unit)

Analogously to what done for the precedent concepts, these elements can be further classified as "New Project", "Upgraded Versions" and "Already Used" elements, with respect to previous steps of exploration (ISS, cis-lunar and Moon concepts): the pie chart reported in figure 4.37 summarizes the number of elements, highlighting their design status (green, yellow and red colors are used to indicate already used, upgraded version and new project, respectively).

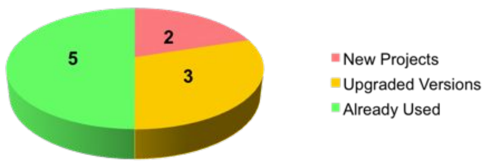


Figure 4.37: NEA concept - mission elements design status

4.2.3.6 Mars preparation

According to the *capability map* analyzed in section 4.2.2 (see figure 4.12), some of the capabilities needed for human Mars mission can be achieved only with specific missions to Mars. For this reason, as already introduced, a dedicated concept, called Mars Preparation (MP), is included in the scenario to achieve the missing capabilities. In particular, this concept allows the demonstration of the capabilities listed in table 4.29. The proposed missions have not been analyzed in details, but basic considerations on the main objectives, baseline architectures and elements are provided hereafter.

Transportation	Support-In Space	Support - Surface Operations	Operations
High speed Earth manned EDL	In-Space cryogenic fuel management	Surface cryogenic fuel management	Advanced RvD
High capacity cargo transfer	In-Space advanced robotics	Surface advanced power	Long range comms
Orbit cargo insertion		Surface advanced thermal	Medium range comms
Orbit manned insertion		Surface radiation protection	Short range comms
Destination cargo entry		Surface advanced robotics	Safe in-space elements separation
Destination cargo D&L		Atmospheric ISRU	
Destination manned D&L		Surface mobility	
Destination cargo ascent			

Table 4.29: Mars preparation concept capabilities

Analogously to what done for the previous concepts, the process of analysis of the Mars preparation case for the definition of the missions and the architectures starts from the identification and evaluation (qualitative) of specific "Second-Level Key Decisions" (as previously explained), for which specific options are selected. The key decisions for the Mars preparation concept are summarized in table 4.30, in which the alternative options are shown, as well as the justification of the final choices.

Key decision	Options				Notes
Mission strategy	**Orbiter-ERV pre-deployment**	All-in			Orbiter pre-deployed into Mars orbit with dedicated mission before the lander one.
Orbiter transfer propulsion	Solar electric	Chemical cryogenic	**Chemical storable**	Nuclear thermal	Same propellant used for Mars transfer orbit insertion, mid-term correction and attitude control maneuvers.
ERV transfer propulsion	Solar electric	Chemical cryogenic	**Chemical storable**		Same propellant used for Earth transfer orbit insertion, mid-term correction and attitude control maneuvers.
Orbiter-ERV Mars insertion strategy	**Aero-braking**	Propulsive braking			Rigid aeroshell is used, 9 months aerocapture into 500km circular orbit, 45deg inclination.
Orbiter operative life	Until ERV Earth injection	Few years after MRS end	Until Mars habitability test (7y)	**Until Mars unmanned rehearsal (10y)**	10-years operative lifetime system in Mars orbit is a valuable demo of Mars relay satellite mission.
Lander transfer propulsion	Solar electric	Chemical cryogenic	**Chemical storable**	Nuclear thermal	Same propellant used for MTO insertion, mid-term correction and attitude control maneuvers; Mars direct entry trajectory implemented.
Surface exploration strategy	Only lander	**Lander + small rover**	Lander + big rover		Samples collected in two locations by the lander and a small size rover (Spirit and Opportunity class).
Descent/Ascent propulsion	**Pressure-fed hypergolic NTO/MMH**	Pump-fed cryogenic LOX/LCH4	Pressure-fed cryogenic LOX/LH2	Hybrid	First Mars ascent demonstration supposed to implement the simplest strategy and propulsion system

Table 4.30: Mars preparation "Second-Level Key Decisions"

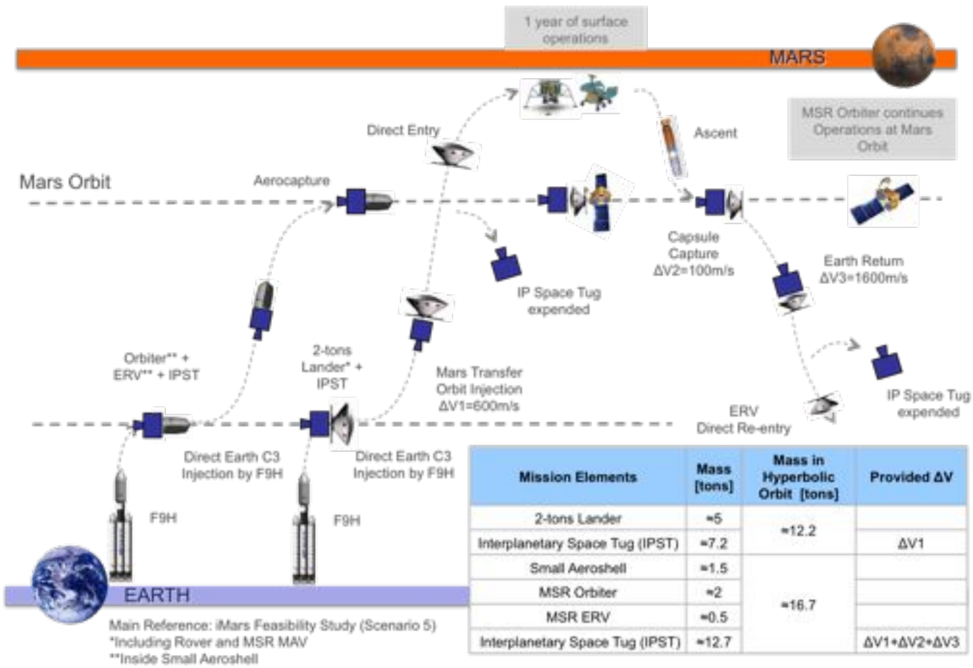
For the Mars preparation concept, three main phases can be distinguished:

1. **Mars sample return phase**, during which a sample return mission is envisioned to carry back to Earth at least 500g of samples;
2. **Mars Preparation I phase**, which includes a Mars habitability test mission to demonstrate some specific capabilities;
3. **Mars Preparation II phase**, which represents an unmanned rehearsal mission, for the demonstration of additional capabilities and to pre-deploy the Mars relay satellite.

A minimum of four missions is needed for this concept, all classified in only one type:

- Unmanned Cargo Delivery Mission, referring to unmanned missions for the demonstration of technologies in view of the human mission to Mars as well as for the pre-deployment of robotic assets.

For the mentioned missions, four different new architectures are identified. The first one refers to the Mars sample return mission and is schematically illustrated in figure 4.38.



Mission Elements	Mass [tons]	Mass in Hyperbolic Orbit [tons]	Provided ΔV
2-tons Lander	≈5	≈12.2	
Interplanetary Space Tug (IPST)	≈7.2		ΔV1
Small Aeroshell	≈1.5	≈16.7	
MSR Orbiter	≈2		
MSR ERV	≈0.5		
Interplanetary Space Tug (IPST)	≈12.7		ΔV1+ΔV2+ΔV3

Figure 4.38: Mars preparation concept - mission architecture 1

The second architecture refers to the Mars habitability test mission: the sequence of operations is shown in figure 4.39.

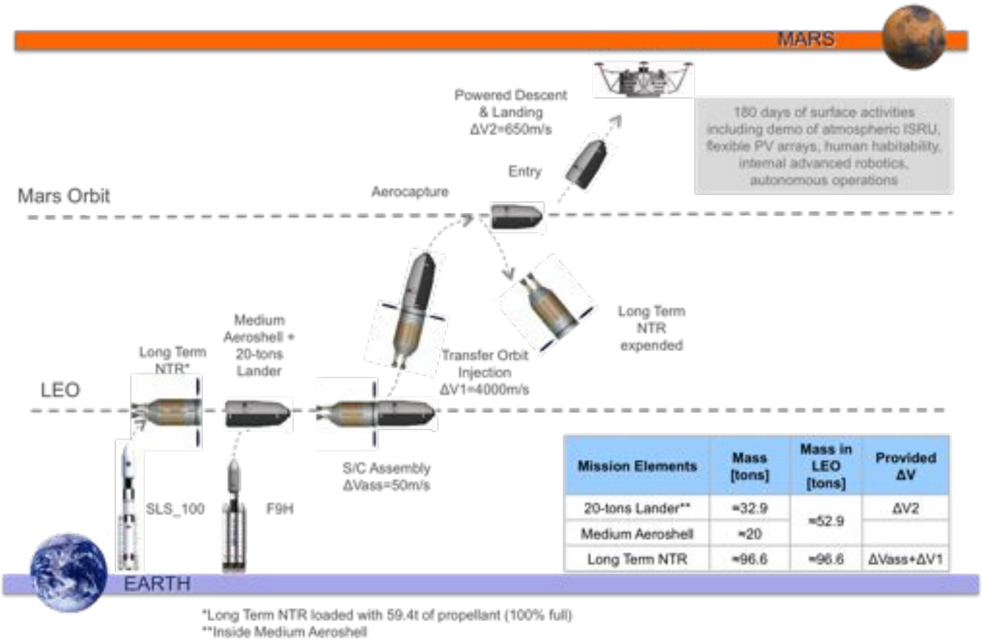


Figure 4.39: Mars preparation concept - mission architecture 2

The third architecture refers to the Mars unmanned rehearsal mission: the sequence of operations is shown in figure 4.40.

The fourth architecture refers to the Mars relay satellite deployment mission: the sequence of operations is shown in figure 4.41.

According to these mission architectures, 22 different elements are needed to accomplish the Mars preparation concept missions, which are (number of needed units is reported in brackets):

- Transportation Elements
  - Small aeroshell (two units)
  - MSR Mars ascent vehicle (one unit)
  - Interplanetary space tug (three units)
  - 2-tons lander (one unit)
  - 20-tons lander (one unit)
  - Descent/Landing stage (one unit)
  - Medium aeroshell (one unit)

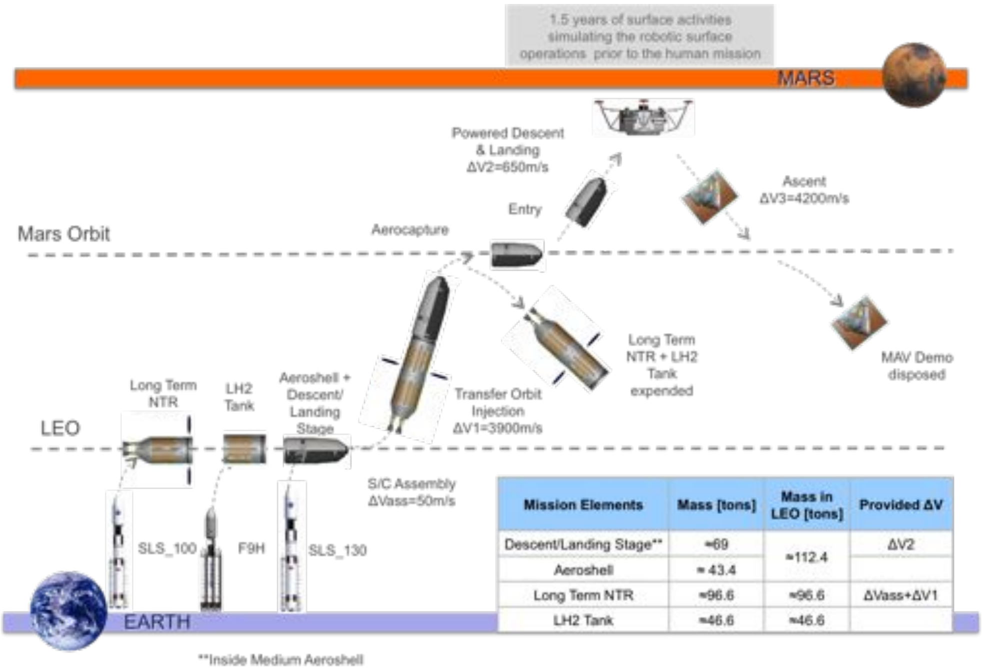


Figure 4.40: Mars preparation concept - mission architecture 3

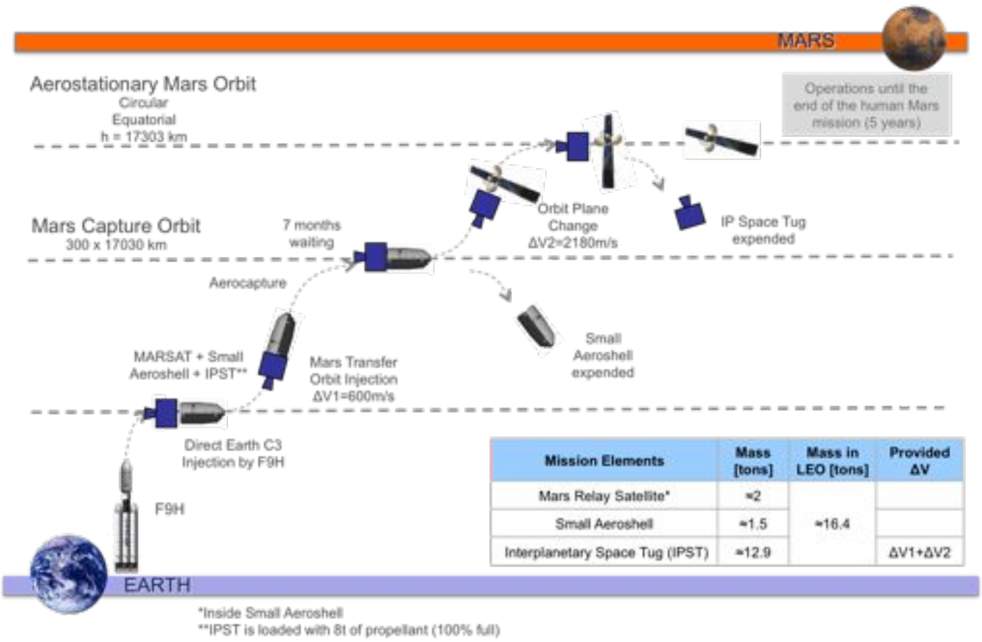


Figure 4.41: Mars preparation concept - mission architecture 4

  - Aeroshell (one unit)
  - MAV demo (one unit)
  - LH2 tank (one unit)
  - Long term NTR (two units)
- In-space elements
  - MSR ERV (one unit)
  - MSR orbiter (one unit)
  - Mars relay satellite (one unit)
- Surface elements
  - Atmospheric ISRU demo (one unit)
  - MSR rover (one unit)
  - Utility cart (one unit)
  - Manipulator (one unit)
  - SHAB demo (one unit)
  - FSPS (one unit)
  - SolPS (one unit)
  - Atmospheric ISRU plant (one unit)

Analogously to what done for the precedent concepts, these elements can be further classified as "New Project", "Upgraded Versions" and "Already Used" elements, with respect to previous steps of exploration (ISS, cis-lunar, Moon and NEA concepts): the pie chart reported in figure 4.42 summarizes the number of elements, highlighting their design status (green, yellow and red colors are used to indicate already used, upgraded version and new project, respectively).

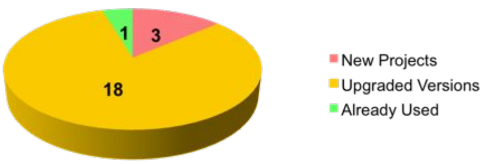


Figure 4.42: Mars preparation concept - mission elements design status

4.2.4 HSE Reference Scenario Summary

Summarizing all the results obtained for the various destinations the reference HSE scenario is built. It is shown in figure 4.43, where all the missions are indicated along the temporal reference window, going from 2014 to 2039, when the human mission to Mars is foreseen (the "star" in the top right corner of the graph refers to the NASA DRA 5.0 human expedition to Mars). The graph has to be read from the bottom

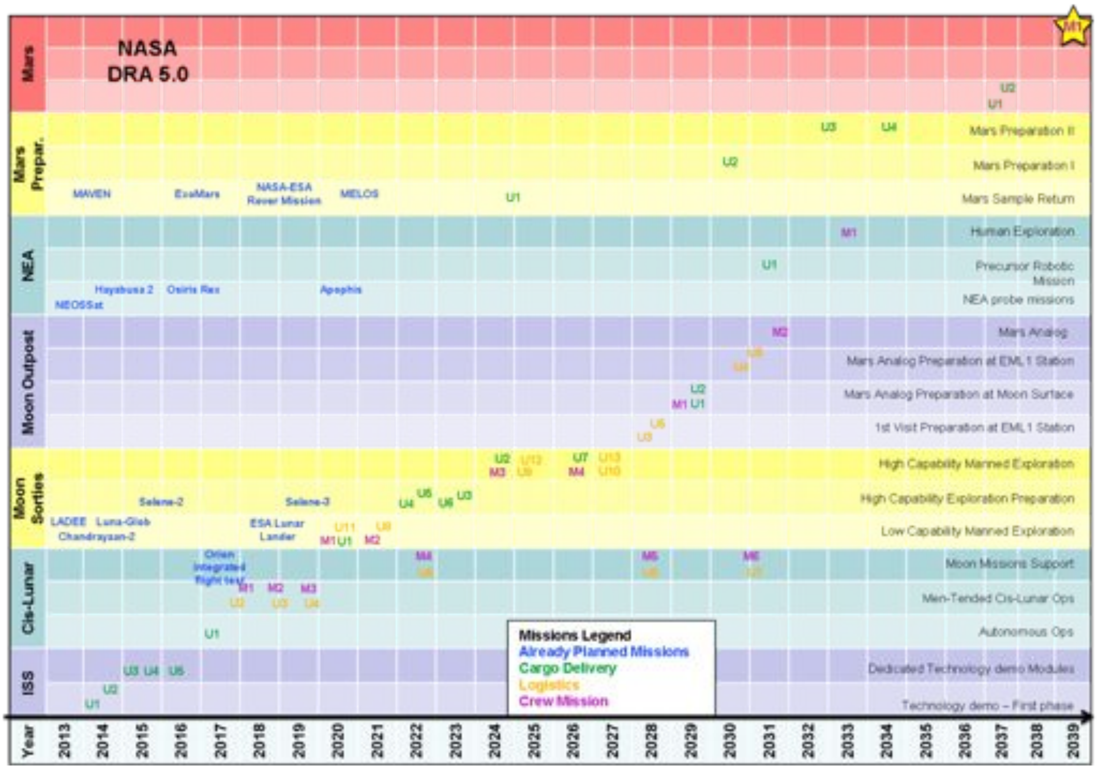


Figure 4.43: HSE reference scenario summary

to the top as the sequence of destinations is represented. For each destination the various phases of exploration are highlighted, using different color tones for the rows in which each destination area is divided. All the missions are indicated with a specific abbreviation and color, to precisely identify them. In particular, the missions labelled with a *green U* are the unmanned missions for the delivery of the cargo, those labelled with a *pink M* are the crew exploration missions and those labelled with a *yellow U* are the unmanned logistics missions. Finally, already planned robotic missions are also included in the scenario (in blue).

4.2.4.1 Launchers summary

Table 4.31 summarizes the needed launchers for the entire scenario. The total number of launchers needed for each mission derives from the mass evaluations performed according to the defined mission concepts, relying on the Tsiolkovsky rocket equation (equation 4.1). In the table the number of units needed for each destination is reported, specifying also the date when the launcher is first needed at that destination. The total units number and the planned availability date are also highlighted.

		Unmanned						Total Unmanned	Manned		Total Manned
		A5_ES	A5_ME	F9H	SLS_70	SLS_100	SLS_130		SLS_70M	At5_M	
# per concept	ISS	5									
	EML1			7	1				6		
	MS		1	8	7				4		
	MO			2	6	2			2		
	NEA				1	1	1			1	
	MP			5		2	1				
	Mars				1	5	3			1	
Total		**5**	**1**	**22**	**16**	**10**	**5**	**59**	**12**	**2**	**14**
Required Date	ISS	2014									
	EML1			2017	2017				2018		
	MS		2022	2020	2020				2020		
	MO			2028	2028	2029			2029		
	NEA				2031	2033	2033			2033	
	MP			2030		2030	2032				
	Mars				2039	2037	2037			2039	
Min		**2014**	**2022**	**2017**	**2017**	**2029**	**2032**		**2018**	**2033**	
Plan		Available	2016	2013-2014	2017	?	?		2017	?	

Table 4.31: HSE scenario launchers summary

4.3 HSE reference scenario associated Building Blocks

As described in the previous sections, an assessment of the elements necessary to accomplish the various destinations missions is done deriving from the architectures analysis. A summary of all the elements needed for the entire reference scenario is shown in figure 4.44.

The number reported next to every element's image refers to the number of units needed at the specific destination. Moreover, a different color is used to indicate if the element is a "New Project", an "Upgraded Version" or an "Already Used" element with respect to the previous steps (red, yellow or green color, respectively). The graph shall be read starting from the bottom, representing the first intermediate destination, i.e. ISS, up to the top, representing the last step, i.e. Mars Preparation. According to the philosophy

Figure 4.44: HSE reference scenario elements summary

behind the study, the considerations about the elements come from the idea to have as much as possible a gradual "improvement" through the following destinations: this can easily be seen looking at the picture. For example, if consider the nuclear thermal rocket element, the first element appearing in the scenario is represented by a demo at ISS ("New Project"). Then, there is a Small NTR ("Upgraded Version" with respect to the previous step) implemented in the Cis-Lunar concept and later on the same small NTR is used in the Moon missions ("Already Used") and so on.

The graph reported in figure 4.45 shows the minimum number of different elements needed for all the destinations concepts, highlighting their changing *Design Status* with respect to the previous concepts. In particular the green color is used to indicate elements already designed and implemented in previous destinations missions, the yellow color indicates upgraded versions of the elements and the red color is used to indicate totally new elements, not needed in previous destinations.

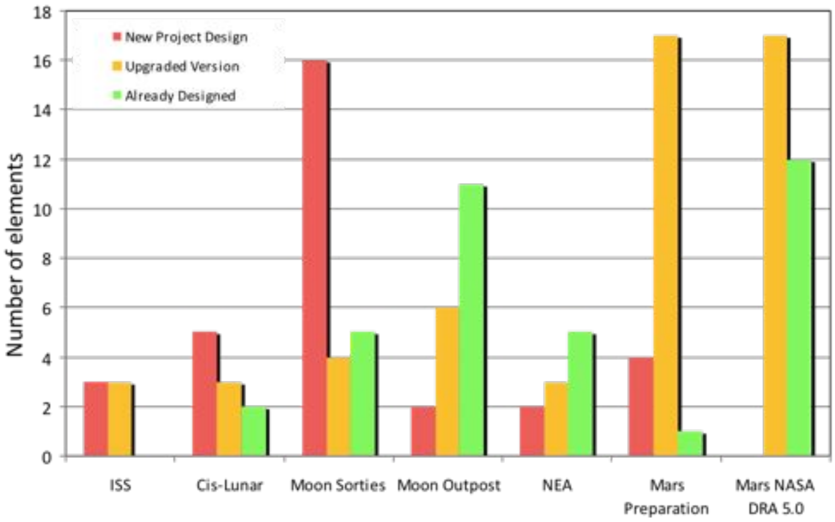


Figure 4.45: HSE reference scenario elements design status

The graph does not include the recurrent units but only the number of elements. As expectable, the graph gives evidence that in the beginning a large number of new elements is needed, but going on through the following destinations the number of already available elements increases while the number of new designs decreases. The ISS concept elements are the additional modules needed for the test of some technologies. Specifically, the new projects refer to demo modules, i.e. NTR demo, inflatable demo and cryogenic fuel tank, while the upgraded versions refer to the ATV-like or PMM-like modules envisaged to carry to the ISS several technologies to be tested. Finally, for the Mars human mission, no new project designs are needed. This is perfectly in line with the philosophy adopted for the definition of the scenario and the assessment of the intermediate destinations missions, to gradually achieve the capabilities required to accomplish the reference human mission to Mars.

4.3.1 Elements commonalities analysis

The "elements commonalities analysis" aims at identifying and verifying the commonalities among elements and at highlighting the major improvements that need to be introduced through various incremental destinations. It is performed per class of elements, in which all the elements are grouped; in particular 16 elements classes are considered:

- Nuclear Thermal Rocket
- Long Permanence Habitat
- Short Permanence Habitat
- Pressurized Modules
- Lander
- Surface Power
- Aeroshell
- Ascent Vehicle
- Earth Entry Vehicle
- Airlock and Suit ports
- Space Tug
- Tank
- Surface Mobility - Rover
- ISRU
- Robotic Arm
- Communications Assets

Each class of elements includes similar elements satisfying more and more demanding requirements, corresponding to gradually improving design and development efforts. An element can belong to more than one class depending on the analyzed requirements (e.g. CEV in Short Permanence Habitat and Earth Entry Vehicle). Within each class, preliminary commonalities analyses are carried out basing on major high-level requirements (mission, functional, operational and interface), as discussed in the following. Figures 4.46-4.61 report an overview of the requirements for the elements belonging to the various elements' classes, highlighting the major changes (yellow cells) passing from previous elements to the following ones (the tables shall be read starting from the bottom, i.e. closer destination, up to the top, i.e. furthest destination). Moreover the major improvements needed for the same element for implementation in successive missions are underlined.

Nuclear Thermal Rocket (NTR)

The nuclear thermal rocket class includes five elements:

- **NTR demo**, which is the first element to be developed and deployed to the ISS to test this technology;
- **Small NTR**, to be used for the Cis-Lunar, Moon sortie and some of the Moon outpost missions, with a maximum propellant capability of 24 t;
- **Small NTR-enhanced**, to be used in longer missions (NEA mission) and thus requiring a specific thermal control for propellant management (boil-off issue);
- **Short term NTR**, which has larger fuel loading capability and is used for mission's duration shorter than three months;
- **Long term NTR**, to be used for longer duration missions (more than three months).

The requirements evolution through successive missions, for the elements belonging to the nuclear thermal rocket class, is described in the table reported in figure 4.46.

			Requirements (The element shall..)								
			...be loaded with a propellant mass equal to N	...have a LEO permanence up to N days	...have a Deep Space permanence up to N days	...be compatible with the crew presence	...provide N number of ignitions	...provide a thrust equal to N	...be provided with active thermal control for cryogenic fuel management	...have interfaces with additional tanks	...act as chaser in RvD maneuvers
	Elements	**Concept**	[mT]	#	#	y/n	#	[kN]	y/n	y/n	y/n
Nuclear Thermal Rocket	**Long Term NTR (>3months)**	Mars Crew Mission	60	150	900	yes	3	3x111	yes	yes (LH2 tanken + drop tank)	yes
		Mars Cargo Mission	59	150-180	350	no	1	3x111	yes	yes (LH2 tank)	yes
		Mars Preparation	59	several weeks	350	no	2	3x111	yes	yes (LH2 tank)	yes
		NEA	63	several weeks	225	yes	3	3x111	yes	yes (drop tank)	no
	Short Term NTR (<3months)	Moon Outpost	60	several weeks	4	no	3	2x111	no	no	yes
	Small NTR - Enhanced	NEA	9/24	few days	225	no	2	1x111	yes	yes (small LH2 tank)	no
	Small NTR	Moon Outpost 3	16/24	few days	-	yes	1	1x111	no	no	no
		Moon Outpost 2	22/24	few days	4	no	2	1x111	no	yes (small LH2 tank)	no
		Moon Outpost 1	14/24	few days	-	no	1	1x111	no	no	no
		Moon Sortie 4	16/24	few days	-	yes	1	1x111	no	no	no
		Moon Sortie 3	22/24	few days	4	no	2	1x111	no	yes (small LH2 tank)	no
		Moon Sortie 2	19/24	few days	4	no	2	1x111	no	no	no
		Moon Sortie 1	21/24	few days	-	no	1	1x111	no	no	no
		Cis-Lunar 2	16/24	few days	-	yes	1	1x111	no	no	no
		Cis-Lunar 1	14/24	few days	-	no	1	1x111	no	no	no
	NTR Demo	ISS	6	several months	-	yes (ISS)	>1	1x67	no	no	no

Figure 4.46: NTR commonalities analysis

Long Permanence Habitat

The Long Permanence Habitat class refers to habitation elements that have to support crew for more than two months. It includes six elements divided in two groups:

- Surface
  - **Big Manned Lander (BML)**, to be used for the last Moon Sortie missions and for the Moon Outpost ones;
  - **Lunar Surface Habitat (LSH)**, which is the Habitat to host the astronauts on the Moon's surface during the Moon Outpost missions;
  - **SHAB Demo**, which is a demo to be deployed on the Mars surface during the Mars Preparation missions, to perform Mars habitability test;
  - **Surface Habitat Lander (SHAB)**, which is the surface habitat for the final human mission;
- Space
  - **Inflatable Demo**, to be tested at the ISS;
  - **Deep Space Habitat (DSH)**, which is the habitat to host the crew in cis-lunar, during the NEA mission and during the deep space phases of the Mars mission.

The requirements evolution through successive missions, for the elements belonging to the long permanence habitat class, is described in the table reported in figure 4.47.

Short Permanence Habitat

The Short Permanence Habitat class refers to habitation elements that have to support crew for less than two months. It includes five elements that are:

- **Crew Exploration Vehicle (CEV)**, which is the capsule fro crew transportation to be used in all the manned missions;
- **Small Manned Lander (SML)**, which is the lander to be used in the first two Moon sortie missions;
- **Pressurized Rover Demo**, that shall demonstrate the pressurized rover capabilities;
- **Pressurized Rover**, needed for the Moon and Mars surface missions;
- **Multi Mission Space Exploration Vehicle (MMSEV)**, which is used for the NEA proximity operations.

		Elements	Concept	Requirements (The element shall..) ...host N crew members	...support a crew permanence of N days	...implement inflatable technology	... provide protection against T external environment	...provide autonomous operation capability	...operate at a maximum N distance from Earth	...provide docking capability with N elements	...interface with T elements	...allow EVA to be performed	...have a lifetime of at least N years
				#	#	y/n	type	y/n	[AU]	#	type	y/n	#
Long Permanence HABITAT (>2 months)	Space	Deep Space Habitat (DSH)	Mars Crew Mission	6	395	yes	LEO / Deep Space / Mars Orbit	yes	2.7	2 (CEV +Docking Hub)	CEV/SM, ST, Docking Hub	no	3
			NEA	4	354	yes	LEO / Deep Space	no	0.2	1 (CEV)	CEV, ST, drop tank	no	1
			Cis-Lunar	4	180	yes	LEO / Deep Space	yes	0.002	4+airlock	HAB-SM, CEV, LM, SML, BML	yes	10
		Inflatable Demo	ISS	1-2	several months	yes	LEO	no	400 km	1 (ISS)	ISS, ST	no	several months
	Surface	Surface habitat Lander (SHAB)	Mars Crew Mission	6	540	yes	Mars orbit / surface / atmosphere	yes	2.7	2	Aeroshell, SHAB/DAV Descent/ Landing Stage, Pressurized Rover, ISRU Plant	yes	4.6
		SHAB Demo	Mars Preparation	1-2	540	yes	Mars orbit / surface / atmosphere	yes	2.7	2	Medium Aeroshell	no	2.5
		Lunar Surface Habitat (LSH)	Moon Outpost	6	540	yes	LEO/Deep Space / Moon orbit /Moon surface	yes	400.000 km	2	BML, Pressurized Rover	yes	2
		Big Manned Lander (BML)	Moon Sorties / Outpost	4	180	no	LEO/Deep Space / Moon orbit /Moon surface	yes	400.000 km	2	EML1 HAB, Pressurized Rover	yes	5

Figure 4.47: Long permanence habitat commonalities analysis

The requirements evolution through successive missions, for the elements belonging to the short permanence habitat class, is described in the table reported in figure 4.48.

Pressurized Modules

The Pressurized Module class includes five elements that are:

- **ATV-like Module**, to be used to carry to the ISS innovative technologies to be tested before being implemented in further missions;
- **PMM-like Module**, to be used at the ISS for the test of innovative technologies to be implemented in further missions;
- **Logistic Module (LM)**, for resources resupply in support of cis-lunar and Moon missions;
- **Contingency Consumables Module (CCM)**, needed for the Mars crew mission;
- **Docking Hub**, needed for Mars crew mission.

The requirements evolution through successive missions, for the elements belonging to the pressurized modules class, is described in the table reported in figure 4.49.

			Requirements (The element shall..)												
			...host N crew members	...support a crew permanence of N days	...provide exploration range capability of N	...implement inflatable technology	... provide protection against T external environment	...provide autonomous operation capability	...operate at a maximum N distance from Earth	...provide docking capability with N elements	...interface with T elements	...allow EVA to be performed	...have a lifetime of at least N years	...re-enter Earth atmosphere	...re-enter at N velocity
	Elements	Concept	#	#	[km]	y/n	type	y/n	[AU]	#	type	y/n	#	y/n	[km/s]
Short Permanence HABITAT (<2 months)	MMSEV	NEA	2	8-10	n/a	no	Deep Space/NEA	yes	0.2	1	DSH	yes (suitports)	3	no	-
	Pressurized Rover (PR)	Mars Crew mission	2 (nominal)	14	100	no	Mars surface	no	2.7	1	SHAB	yes (suitports)	2.6	no	-
		Moon Outpost	2	14	100	no	Moon Surface	yes	400.000km	1	23-ton Cargo Lander, LSH	yes (suitports)	3	no	-
		Moon Sortie	2	14	100	no	Moon Surface	yes	400.000 km	1	8-ton Cargo Lander, BML	yes (suitports)	2	no	-
	Pressurized Rover Demo	Moon Sorties	2	7	30	no	Moon Surface	yes	400.000 km	1	8-ton Cargo Lander	yes (suitports)	1	no	-
	Small Manned Lander (SML)	Moon Sorties	2	10	n/a	TBD	LEO/Deep Space / Moon orbit /Moon surface	yes	400.000 km	1	EML1 HAB	yes	2	no	-
	CEV	Mars Crew mission	6	15	n/a	no	LEO/Deep Space/Mars orbit	no	2.7	1	DSH,CEV-SM, Aeroshell	no	2.6	yes	13
		NEA	4	few hours	n/a	no	LEO/Deep Space/NEA proximity	no	0.2	1	DSH, CEV-SM	yes*	1	yes	11.5
		Moon Sorties/ Outpost	6	8	n/a	no	LEO/Deep Space	yes	0.002	1	EML1 HAB, CEV-SM	no	1.5	yes	11
		Cis-lunar	4	8	n/a	no	LEO/Deep Space	no	0.002	1	EML1 HAB, CEV-SM	no	6months	yes	11

Figure 4.48: Short permanence habitat commonalities analysis

			Requirements (The element shall..)							
			...have a total maximum mass of N	...support a crew permanence of maximum N continous time	...foresee upload and offload operations	...host T payload	... provide protection against T external environment	...provide autonomous operation capability	...interface with T elements	...have a maximum total lifetime of N
	Elements	Concept	[mT]	[hr]	y/n	type	type	y/n	type	[years]
Pressurized Module	Docking Hub	Mars Crew Mission	10	12	no	Experiments	LEO/Deep Space/Mars Orbit	no	DSH, CCM, CEV, MAV, Airlock	3.0
	Contingency Consumables Module (CCM)	Mars Crew Mission	14	3	yes	Consumables, Crew Provisions, Spares, Waste	LEO/Deep Space/Mars Orbit	no	Docking Hub	1.5
	Logistic Module (LM)	Cis-lunar Moon Sortie Moon Outpost	10	5	yes	Consumables, Crew Provisions, Spares, Waste	LEO/Cis-lunar Space	yes	Deep Space Habitat (DSH)	0.5
	PMM-like Module	ISS	10	12	no	Experiments	LEO	no	ISS Node 2	6.0
	ATV-like Module	ISS	10	5	yes	Consumables, Crew Provisions, Spares, Waste	LEO	yes	ISS Node 2	0.5

Figure 4.49: Pressurized modules commonalities analysis

Lander

The Lander class includes eight elements divided into two groups:

- Moon landers

  - **1-ton lander**, part of the second cargo mission of the Moon sortie concept;
  - **Small Manned Lander (SML)**;
  - **8-tons lander**, used in several Moon sortie cargo missions;
  - **Big Manned Lander (BML)**;
  - **23-tons lander**, used in several Moon outpost cargo missions;
- <u>Mars landers</u>
  - **2-tons lander**, present in the first Mars preparation mission (Mars Sample Return);
  - **20-tons lander**, present in the second Mars preparation mission;
  - **Descent/Landing stage**, part of the third Mars preparation mission and both the cargo and crew missions of NASA DRA 5.0.

The requirements evolution through successive missions, for the elements belonging to the lander class, is described in the table reported in figure 4.50.

				Requirements (The element shall..)								
				...land a maximum mass equal to N	...implement a type T main propulsion system	...provide a thrust value of N for high impulse maneuvers	...store the propellant up to a N time duration	...be human rated	...land with a precision of N	...deal with a previous atmospheric entry phase	...perform surface ascent	...be refuelled
		Elements	Concept	[mT]	type	[kN]	time	y/n	#	y/n	y/n	y/n
Lander	Mars	Descent/Landing Stage	Mars Cargo Mission 2	56.9	pump-fed LOX/LCH4	4 x 66 kN	3 years	yes	10 m	yes	no	no
			Mars Cargo Mission 1	56.9	pump-fed LOX/LCH4	4 x 66 kN	380 days	yes	10 m	yes	no	no
			Mars Preparation 3	56.9	pump-fed LOX/LCH4	4 x 66 kN	380 days	yes	10 m	yes	no	no
		20-tons Lander	Mars Preparation 2	27.3	pump-fed LOX/LCH4	3 x 10 kN 1 x 66 kN	380 days	no	100 m	yes	no	no
		2-tons Lander	Mars Preparation 1	4.3	pump-fed LOX/LCH4	1 x 10 kN	240 days	no	3 km	yes	no	no
	Moon	23-tons Lander	Moon Outpost 2	35.2	pump-fed LOX/LCH4	3 x 10 kN 1 x 66 kN	6 days	no	10 m	no	no	no
		Big Manned Lander	Moon Sortie 4/ Moon Outpost 1	55.2	pump-fed LOX/LCH4	3 x 10 kN 1 x 66 kN	600 days	yes	10 m	no	yes	yes
		8-tons Lander	Moon Sortie 3	11.8	pump-fed LOX/LCH4	1 x 66 kN	6 days	no	25 m	no	no	no
		Small Manned Lander	Moon Sortie 1	17.4	pump-fed LOX/LCH4	4 x 10 kN	15 days	yes	200 m	no	yes	yes
		1-ton Lander	ESA Lunar Lander / Moon Sortie 2	3.6	pump-fed LOX/LCH4	1 x 10 kN	6 days	no	500 m	no	no	no

Figure 4.50: Lander commonalities analysis

Surface Power

The Surface Power class includes five elements divided into two groups:

- Moon
  - **Fission Surface Power System (FSPS) Demo**, which is implemented in Moon sortie mission to test the nuclear power system;
  - **FSPS**, to be used during the Moon outpost missions;
  - **Solar Power System (SolPS)**, to be implemented both in the Moon sortie and outpost missions.
- Mars
  - **FSPS**, included in both Mars preparation and NASA DRA5.0 Mars cargo missions;
  - **SolPS**, included in both Mars preparation and NASA DRA5.0 Mars cargo missions.

The requirements evolution through successive missions, for the elements belonging to the surface power class, is described in the table reported in figure 4.51.

					Requirements (The element shall..)					
					...generate maximum nominal power of N	...include energy storage	...be mobile	...include radiation shielding	...interface with type T element	...have a total lifetime of N
			Elements	Concept	[kW]	y/n	y/n	y/n	-	years
Surface Power	Mars	Solar	SolPS	Mars Cargo Mission 1	Sunlight: 9 Eclipse: 5	yes	desirable	no	Utility Cart, MPU Manipulator	5
				Mars Preparation 1	Sunlight: 9 Eclipse: 5	yes	desirable	no	Utility Cart MPU Manipulator	5
		Nuclear	FSPS	Mars Cargo 2	30-40	no	no	yes	Utility Cart MPU Manipulator	5
				Mars Preparation 1	30	no	no	yes	Utility Cart MPU Manipulator	5
	Moon	Solar	SolPS	Moon Sortie 1-2/ Moon Outpost 1	Sunlight: 8, Eclipse: 5	yes	desirable	no	Utility Cart MPU Manipulator	4
		Nuclear	FSPS	Moon Outpost 1	30	no	no	yes	Utility Cart MPU Manipulator	4
			FSPS Demo	Moon Sortie 1	15	no	no	yes	Utility Cart MPU Manipulator	4

Figure 4.51: Surface power commonalities analysis

Aeroshell

The Aeroshell class includes three elements that are:

- **small aeroshell**, which is the first type of aeroshell implemented in the first Mars preparation mission;
- **medium aeroshell**, which is an evolution of the previous aeroshell still to be implemented during the Mars preparation missions;
- **aeroshell**, which is the actual aeroshell, needed for the final Mars cargo missions (NASA DRA 5.0).

The requirements evolution through successive missions, for the elements belonging to the aeroshell class, is described in the table reported in figure 4.52.

				Requirements (The element shall..)					
				...decelerate a maximum mass equal to N	...decelerate a maximum volume equal to N	...perform Aerocapture/ Aerobraking	...perform Entry	...be human rated	...have a total lifetime of N
		Elements	Concept	[mT]	[m x m]	y/n	y/n	y/n	[years]
Aeroshell	Mars	Aeroshell	Mars Cargo 2	110	10 x 30	yes	yes	yes	3
			Mars Cargo 1	110	10 x 30	yes	yes	no	1
			Mars Preparation 3	110	10 x 30	yes	yes	no	1
		Medium Aeroshell	Mars Preparation 2	52.9	7 x 15	yes	yes	no	1
		Small Aeroshell	Mars Preparation 1 (MSR)	2.5	Hypergolic NTO/ MMH	yes	no	no	1.5

Figure 4.52: Aeroshell commonalities analysis

Ascent Vehicle

The Ascent Vehicle class includes five elements divided into two groups:

- Moon
  - **Small Manned Lander**, to be used for the first Moon sortie missions;
  - **Big Manned Lander**, needed for the last Moon sortie missions and for the Moon outpost;
- Mars
  - **Mars Sample Retunr Ascent Vehicle**, which is the vehicle fro the Mars sample return mission;

- **Mars Ascent Vehicle Demo**, which is a demo to test the MAV capability during the Mars preparation missions;
- **MAV**, which is the Mars Ascent Vehicle for the Mars crew mission.

The requirements evolution through successive missions, for the elements belonging to the ascent vehicle class, is described in the table reported in figure 4.53.

				Requirements (The element shall..)								
				...lift a maximum mass equal to N	...implement a type T main propulsion system	...provide a thrust value of N for high impulse maneuvers	...store the propellant up to a N time duration	...be human rated	...RvD with a T element	...deal with an atmosphere	...perform surface descent	...be refuelled
		Elements	Concept	[mT]	type	[kN]	[days]	y/n	type	y/n	y/n	[years]
Ascent Vehicle	Mars	MAV	Mars Crew	45.7	pump-fed LOX/LCH4	4 x 66 kN	4 years	yes	DSH	yes	no	yes
		MAV Demo	Mars Preparation 2	27.1	pump-fed LOX/LCH4	3 x 66 kN	3 years	no	no	yes	no	yes
		MSR Ascent Vehicle	Mars Preparation 1 (MSR)	0.8	Hypergolic NTO/MMH	n/a	2 years	no	MSR ERV	yes	no	no
	Moon	Big Manned Lander	Moon Sortie 2/ Moon Outpost 1	26.1	pump-fed LOX/LCH4	3 x 10 kN 1 x 66 kN	600 days	yes	Cis-lunar Habitat	no	yes	yes
		Small Manned Lander	Moon Sortie 1	13.1	pump-fed LOX/LCH4	4 x 10 kN	15 days	yes	Cis-lunar Habitat	no	yes	yes

Figure 4.53: Ascent vehicle commonalities analysis

Earth Entry Vehicle

The Earth Entry Vehicle class of elements includes:

- **Crew Exploration Vehicle (CEV)**, which is the capsule to host the astronauts in all mission concepts;
- **Mars Sample Return Earth Re-entry Vehicle (MSR ERV)**, which is the capsule envisioned for the Mars Sample Return mission.

The requirements evolution through successive missions, for the elements belonging to the Earth entry vehicle class, is described in the table reported in figure 4.54.

Airlock and Suit Ports

The Airlock and Suit ports class refers to two possible options for the EVA execution:

- **Suitports** are implemented in Moon sortie and outpost missions, in the NEA proximity operations and in the Mars crew mission.
- **Airlock** is implemented also in cis-lunar concept missions.

			Requirements (The element shall..)								
			...perform Earth re-enter at N velocity	...operate in T environment	...host N astronauts	...provide autonomous operation capability	...operate at a maximum N distance from Earth	...provide docking capability with N elements	...interface with T elements	...allow EVA to be performed	...have a lifetime of at least N years
	Elements	Concept	[km/s]	type	#	y/n	[AU]	#	type	y/n	#
Earth Entry Vehicles	CEV	Mars Crew mission	13	Deep Space / Mars Orbit	6	yes	2.7	1	DSH,CEV-SM, Aeroshell	no	2.6
		NEA	11,5	Deep Space / NEA proximity	4	no	0.2	1	DSH, CEV-SM	yes	1
		Moon Outpost	11	Deep Space	6	yes	0.002	1	EML1 HAB, CEV-SM	no	1.5
		Moon Sorties	11	Deep Space	6	no	0.002	1	EML1 HAB, CEV-SM	no	2months
		Cis-lunar	11	Deep Space	4	no	0,002	1	EML1 HAB, CEV-SM	no	6months
	MSR ERV	Mars Preparation	13	Deep Space/ Mars Orbit	n/a	yes	2,7	1	IPST, small aeroshell, Mars orbiter	no	6

Figure 4.54: Earth entry vehicle commonalities analysis

			Requirements (The element shall..)							
			...allow N crew members to perform EVA	...support the execution of N EVA	...implement inflatable technology	... provide protection against T external environment	...operate at a maximum N distance from Earth	...provide docking capability with N elements	...interface with T elements	...have a lifetime of at least N years
	Elements	Concept	#	#	y/n	type	[AU]	#	type	#
Airlock & Suitports	Suitports	Mars Crew mission	2 (up to 4)		n/a	Mars surface	2.7	n/a	PR	4.6
		NEA	2	4-6	n/a	Deep Space/ NEA	0.2	n/a	MMSEV	3
		Moon Outpost	2	TBD	n/a	Moon Surface	400.000km	n/a	PR	3
		Moon Sorties	2	TBD	n/a	Moon Surface	400.000km	n/a	PR Demo, PR	2
	Airlock	Mars Crew mission (surface)	2-4	50	TBD	Mars surface	2.7	1	SHAB	4.6
		Mars Crew mission (travel)	2	2 (TBC)	TBD	LEO / Deep Space / Mars Orbit	2.7	1	Docking Hub	2.6
		Moon Outpost 2	2-4	50	TBD	Moon Surface	400.000km	1	LSH	3
		Moon Sorties 3-4 / Outpost 1	2	20+25	TBD	LEO / Deep Space / Moon Surface	400.000km	1	BML	5
		Moon sorties 1-2	2	4	TBD	LEO / Deep Space / Moon Surface	400.000km	1	SML	2
		Cis-lunar	2	6	yes	LEO / Deep Space	0.002	1	HAB	10

Figure 4.55: Airlock and suit ports commonalities analysis

The requirements evolution through successive missions, for the elements belonging to the airlock and suit ports class, is described in the table reported in figure 4.55.

Space Tug

The Space Tug class includes four elements which are:

- **Space Tug**, which is used in the ISS, cis-lunar, Moon and NEA concepts;
- **HAB-SM**, which is the propulsion module attached to cis-lunar station;
- **CEV-SM**, which is the Service Module of the Crew Exploration Vehicle;
- **Interplanetary Space Tug**, used in Mars Preparation concept missions.

The requirements evolution through successive missions, for the elements belonging to the space tug class, is described in the table reported in figure 4.56.

			Requirements (The element shall..)								
			...provide a total DV equal to N	...be compatible with T external environment	...provide autonomous operation capability	...operate at a maximum N distance from Earth	...perform max N dockings maneuvers	...interface with T elements	...have a lifetime of at least N years	...act as chaser in RvD maneuvers	...perform destructive re-entry in Earth atmosphere
	Elements	Concept	[m/s]	type	y/n	[AU]	#	type	#	y/n	y/n
Space Tug	Interplanetary Space Tug (IPST)	Mars Preparation	2300	LEO, Deep Space,Mars Orbit	yes	2.7	1	Small Aeroshell, 2-tons lander, Earth Return Vehicle	1.5	yes	yes
	CEV-SM	Mars Crew mission	1500	LEO,Deep Space,Mars Orbit	yes	2.7	3	CEV	2.6	yes	yes
		NEA	1000	LEO, Deep Space	no	0.2	1	CEV	1	yes	yes
		Moon Outpost	1450	LEO, Deep Space	yes	0.002	1	CEV, Small NTR	1.5	yes	yes
		Moon Sorties	1450	LEO, Deep Space	no	0.002	1	CEV, Small NTR	6months	yes	yes
		Cis-lunar	1450	LEO, Deep Space	no	0.002	1	CEV, CPS, Small NTR	6months	yes	yes
	HAB-SM	Cis-lunar	700	LEO, Deep Space	yes	0,002	n/a	EML1 HAB, CPS	10	no	no
	Space Tug	NEA	~100	LEO	yes	LEO	1	DSH	few weeks	yes	yes
		Moon Sorties / Moon Outpost	800	LEO, Deep Space, LLO	yes	400.000km	2	ATV-like module / Fuel Tank / Lunar Relay Sats	1	yes	no
		Cis-lunar	800	LEO, Deep Space	yes	0,002	1	Small NTR / ATV-like module	1	yes	no
		ISS	~100	LEO	yes	400km (ISS)	1	ATV-like module / PMM-like module / ISS Demos	1	yes	yes

Figure 4.56: Space tug commonalities analysis

Tank

The Tank class includes five elements which are:

- **Small LH2 Tank**, which is to be implemented for short term storage;
- **LH2 Tank**, which is implemented in the Mars preparation and Mars missions;

- **LH2 Enhanced**, which is an evolution of the previous one which shall store propellant for longer time and therefore a dedicated thermal control is needed;
- **Fuel Tank**, to carry the fuel needed for the refueling of the lunar landers;
- **Drop Tank**, needed for the NEA and Mars mission.

The requirements evolution through successive missions, for the elements belonging to the tank class, is described in the table reported in figure 4.57.

			Requirements (The element shall..)					
			...store and manage T fuel	...store fuel for a period of time N	...carry a mass of propellant N	...to provide the fuel for T maneuvers	...provide fuel storage in T external environment	...have a lifetime of at least N years
	Elements	Concept	type	days	[tons]		type	#
Tanks	Drop Tank	Mars	LH2	few weeks	87	TMI	LEO	weeks
		NEA	LH2	few weeks	76	TNI (human mission)	LEO	weeks
	Fuel Tank	Moon Outpost	LOX/LCH4	few weeks	57.5	Refuel	LEO, Deep Space	1
		Moon Sortie	LOX/LCH4	few weeks	57.5	Refuel	LEO, Deep Space	1
	LH2 Enhanced	Mars	LH2	2	70	TMI, MOI, TEI	LEO, Deep Space, Mars Orbit	2
	LH2 Tank	Mars	LH2	few weeks	34	TMI	LEO	1
		Mars Preparation	LH2	few days	34	TMI	LEO, Deep Space, Mars Orbit	
	Small LH2 Tank	NEA	LH2	few days	22/25	TNI (robotic mission)	LEO, Deep Space	months
		Moon Outpost	LH2	few days	23,7/25 (95%loaded)		LEO, Deep Space	days
		Moon Sortie	LH2	few days	23,7/25 (95%loaded)		LEO, Deep Space	days

Figure 4.57: Tanks commonalities analysis

Surface Mobility - Rover

The Surface Mobility/Rover class refers to those elements needed to move on the surface of both Moon and Mars. It includes eleven elements divided into two groups:

- <u>Moon</u>
  - **Precursor Rover**, implemented in the first cargo Moon Sortie mission;
  - **Utility Cart**, part of the second Moon Sortie cargo mission;
  - **Small Exploration Rover**, used in the third Moon Sortie cargo mission;
  - **SolPS**, present in multiple Moon Sortie missions and the first Moon Outpost mission;

  - **Unpressurized Rover**, included in the third Moon Sortie cargo mission;
  - **Pressurized Rover Demo**, used in the third Moon Sortie cargo mission;
  - **Pressurized Rover**, present in the third Moon Sortie cargo mission and the first Moon Outpost one;
- Mars
  - **Utility Cart**, present in the second Mars Preparation mission and the first cargo mission of NASA DRA 5.0;
  - **SolPS**, same as the Utility Cart;
  - **Unpressurized Rover**, part of the NASA DRA 5.0 crew mission;
  - **Pressurized Rover**, part of the NASA DRA 5.0 crew mission.

The requirements evolution through successive missions, for the elements belonging to the surface mobility/rover class, is described in the table reported in figure 4.58.

				Requirements (The element shall..)						
				...cover a N maximum range	...have a N velocity	...have a N autonomy	...be human rated	...support a payload mass of N	...support N excursions	...have a total lifetime of N
		Elements	Concept	[km]	[km/h]	[days]	y/n	[mT]	#	[years]
Surface Mobility/Rovers	Mars	Pressurized Rover	Mars Crew	100	20	14	yes	4.8	15	0.50
		Unpressurized Rover	Mars Crew	50	20	8hrs	yes	3.0	50	0.50
		SolPS	Mars Preparation 1/ Mars Cargo 1	2	0-1	8hrs	no	1.5	TBD	4
		Utility Cart	Mars Preparation 1/ Mars Cargo 1	1	0-1	continuous	no	8.0	TBD	4
	Moon	Pressurized Rover	Moon Sortie 3/ Moon Outpost 1	100	20	14	yes	4.8	15	3
		Pressurized Rover Demo	Moon Sortie 3	30	10 20	7	yes	3.0	5	5
		Unpressurized Rover	Moon Sortie 3	10	20	8hrs	yes	3.0	25	5
		SolPS	Moon Sortie 3/ Moon Outpost 1	2	0-1	8hrs	no	1.5	TBD	4
			Moon Sortie 2	2	0-1	8hrs	no	1.5	TBD	6
		Small Exploration Rover	Moon Sortie 3	20	0-5	continous	no	0.3	TBD	1
		Utility Cart	Moon Sortie 2	1	0-1	continous	no	8.0	TBD	6
		Precursor Rover	Moon Sortie 1	5	0-1	continous	no	0.2	TBD	1

Figure 4.58: Surface mobility/rover commonalities analysis

ISRU

The ISRU class of elements includes four elements divided into two groups:

- Moon
  - **ISRU Demo**, deployed during Moon Sortie concept missions to test the facility;
  - **Small ISRU Plant**, used during Moon Outpost concept missions;
- Mars
  - **Atmospheric ISRU Demo**, to test the ISRU plant;
  - **Atmospheric ISRU Plant**, deployed during the cargo mission preceding the human one to produce propellant to be exploited for the human mission.

The requirements evolution through successive missions, for the elements belonging to the ISRU class, is described in the table reported in figure 4.59.

				Requirements (The element shall..)							
				...extract T fuel resources	...produce T fuel resources	...produce a quantity N of fuel resources	...produce T ECLS resources	...produce a quantity N of ECLS resources	...interface with T elements	...actively and continuously operate for a maximum time N	...have a total lifetime of N
		Elements	Concept	type	[mT]	[mT]	type	[mT]	type	[years]	[years]
ISRU	Mars	Atmospheric ISRU Plant	Mars Cargo 1	CO2	LOX	24.9	O2,H20, Buffer Gases	2	MAV, SHAB, DAV Descent/ Landing, MPU	1.5	4.6
			Mars Preparation 2	CO2	LOX	14	no	-	MAV Demo, DAV Descent/ Landing, MPU	1.5	2.0
		Atmospheric ISRU Demo	Mars Preparation 1	CO2	LOX	3	O2,H20, Buffer Gases	0.5	20-tons Lander, SHAB Demo	6 months	1.0
	Moon	Small ISRU Plant	Moon Outpost 1	Regolith	LOX	3	O2,H2O	1	23-tons Lander, LSH, MPU	2 months	3.0
		ISRU Demo	Moon Sortie 1	Regolith	LOX	n/a	O2,H2O	n/a	8-tons Lander, BML, MPU	1 week	1.0

Figure 4.59: ISRU commonalities analysis

Robotic Arm

The Robotic Arm class includes both Robotic Arm and Manipulator. The requirements evolution through successive missions, for the elements belonging to the robotic arm class, is described in the table reported in figure 4.60.

Communications assets

The Communications Assets class includes four elements divided into two groups:

- Moon
  - **Lunar Relay Satellite (LRS)**, deployed during the Moon sortie concept missions;

			Requirements (The element shall..)							
			...be installed on T element	...work in T environment	...perform T operations	... handle and move T elements	...interface with T elements	... provide a lift capability of N	...support astronauts EVA activities	...have a lifetime of at least N years
	Elements	Concept	type	type	type	type	type	[MT]	y/n	#
Robotic Arms	Manipulator	Mars Crew mission	TBD	Mars Surface	-	-	-	-	-	-
		Mars Preparation	TBD	Mars Surface	surface assets offload,installation & 3inspection,utility cart IF & tools change	Mars surface assets	Descent/Entry Stage, utility cart, surface assets	8	no	>2
		Moon Outpost	TBD	Moon Surface	surface assets offload,installation & 3inspection,utility cart IF & tools change	Moon surface assets	utility cart, 22-t cargo lander, surface assets	8	no	1
		Moon Sorties	TBD	Moon Surface	surface assets offload,installation & 3inspection,utility cart IF & tools change	Moon surface assets	utility cart, 8-t lunar lander, surface assets	4	no	1
	Robotic Arm	Mars	TBD	Mars Orbit/ Surface	-	-	-	-	-	-
		Cis-lunar	EML1 HAB	Deep Space (EML1)	S/C reconfiguration, EVA support	airlock	HAB, airlock	8*	yes	10

Figure 4.60: Robotic arm commonalities analysis

- <u>Mars</u>
  - **MSR Orbiter**, deployed during the first Mars Preparation mission;
  - **Mars Relay Satellite (MARSAT)**, deployed during the last MP mission to support the following human expedition.

The requirements evolution through successive missions, for the elements belonging to the communications class, is described in the table reported in figure 4.61.

4.4 Discussion

The HSE scenario discussed in this chapter has been built considering as final goal a human mission to Mars by the end of the 2030 decade. In particular the NASA DRA 5.0 is taken as reference mission [3], and most evaluations and major decisions have been driven by this final objective.

Although the mission as described by NASA DRA 5.0 is quite ambitious and has several weak points in its definition, all the considerations done within this study could be easily extended to other mission opportunities, which envisage Mars human mission

				Requirements (The element shall..)				
				...have a total maximum mass of N	...communicate with a delay of N	...communicate with T	...interface with T elements	...have a total lifetime of N
		Elements	Concept	type	[s]	type	type	[years]
Communications Assets	Mars	Mars Satellite Relay (MARSAT)	Mars Preparation 2	2	5-20 minutes	Mars Assets, DSH, Earth	IP Space Tug, Small Aeroshell	6.0
		MSR Orbiter	Mars Preparation 1	2	5-20 minutes	MSR Rover, 2-tons Lander, Mars Preparation Assets, Earth	IP Space Tug, MSR ERV, Small Aeroshell	9.0
	Moon	Lunar Relay Satellite (LRS)	Moon Sortie 1	4	2.5	Moon Surface Assets, EML1 Station, Earth	Space Tug	9.0

Figure 4.61: Communications asset commonalities analysis

as final target. As also addressed in [3], the complexity and costs associated to this type of mission would be very high, thus limiting the probability to accomplish such a mission by the end of 2030s. However, unlike the NASA DRA 5.0 mission (focusing on a direct mission to Mars), the idea behind the present study is that of following a gradual path in the expansion through the solar system, which can allow a stepwise technological development and capabilities achievement that can drastically reduce the risks and costs associated to a mission like the NASA DRA 5.0, making it a more realistic opportunity. The objective of this study is to demonstrate the importance and feasibility of developing a long-term strategy for capability evolution and technology development, when considering space exploration, and specifically to provide a general methodology to be followed in the assessment of a reference scenario. According to this, even if a different "easier" architecture (e.g. with a smaller number of crew members [7, 8, 9]) or a different time opportunity (maybe a postponed time opportunity), were considered for the final mission to Mars, the considerations done in this study, and most of all the methodology developed, would still be valid and applicable. More in general, the developed methodology can be considered versatile and theoretically practicable in case the overall scenario is shifted in time, due to delays in the development of specific technologies or to available missions' opportunities.

The analysis and selection of the intermediate destinations to be included in the scenario is carried out looking at the capabilities required and/or applicable to the various concepts. The capabilities are expressed as high-level functions (which do not refer to any specific implementation or solution) and the capabilities map allows identifying in which destinations concepts the functions are implementable, even if it does not allow understanding at which level they are implementable. The selection is done on the basis of the number of implementable capabilities in each destination with the aim of guaranteeing that all Mars' required capabilities are implemented along the scenario. Pursuing this approach, six intermediate destinations concepts have been selected as the minimum number of destinations concepts necessary to gradually achieve the final reference human mission to Mars. It is worth noting that the methodology would still be applicable if the high-level functions were divided in to sub-functions, which would allow having a more detailed description of the capabilities and a more complex capabilities map.

The results obtained with the described methodology can be a good starting point to take strategic decisions about future missions, possibly considering additional objectives. For example, the NEA mission concept does not represent a very high added value in the path of exploration if only the technological point of view is considered, even if it is very interesting to be considered as a rehearsal for the Mars mission, and moreover from the scientific and planetary defense standpoints. The results discussed in this chapter rely on specific assumptions, which have actually driven some of the choices. Of course, if some assumptions change, the methodology (and all the analysis steps) will still be valid and applicable, but the final results could potentially be different. For example, all the considerations behind the reference scenario are driven by the assumption of having NASA DRA 5.0 mission to Mars as final target. According to this, nuclear propulsion is implemented through various destinations; if a different final target mission were assumed, e.g. implementing cryogenic propulsion, cryogenic propulsion would be the solution to be chosen along the scenario. Furthermore, the study is based on a pure technical approach, which does not take into account cost considerations. Accordingly, the architectures for the various missions are defined on the basis of qualitative assessment of different parameters and in such a way to guarantee a progressive achievement of technological capabilities, as also demonstrated through the commonalities analysis carried out to highlight the performance improvements foreseen for the building blocks as they are implemented in following destinations missions.

5

Technological Solutions

This chapter focuses on the technologies' analysis. As introduced in section 3.2.2, this part of the work aims at identifying the innovative and promising not yet fully space qualified technologies and determining their applicability on the elements of the proposed reference HSE scenario. The final goal is the implementation of a flexible tool applicable to different final destinations (not only to the proposed scenario), in order to support strategic decisions for future space exploration specifically in terms of technologies roadmaps. In the following sections a detailed description of the major obtained results is reported.

5.1 Technologies analysis

5.1.1 Technologies database

According to methodology illustrated in section 3.2.2, the first step in the technologies roadmaps analysis aims at building a database collecting the most significant and promising space technologies. The innovative technologies to be included in the database are identified by means of an accurate review of the major space agencies recent documents on capabilities and technologies assessment and roadmaps [3, 10, 11, 12, 18, 19]. The final scope is to have an organized list of advanced technologies, ordered according to specific technological areas (TA).

Specifically, eleven technological areas are considered, which can have a direct correspondence with subsystems:

- TA.1 Structures and Mechanisms
- TA.2 Power
- TA.3 Thermal
- TA.4 Robotics and Automation
- TA.5 Avionics
- TA.6 Communications
- TA.7 Attitude & GNC
- TA.8 Life Support
- TA.9 Propulsion
- TA.10 Environment, Humans and Safety
- TA.11 Atmospheric Descent and Landing

Each TA is further decomposed into relevant technological sub-areas, corresponding to specific functions/subsystems: a summary of all the eleven TAs and the relative sub-areas is reported in table 5.1.

TA.1 Structures and Mechanisms	**TA.5 Avionics**	**TA.9 Propulsion**
1.1 Structures	5.1 Avionics	9.1 Chemical
1.2 Mechanisms		9.2 Electric
1.3 Separations	**TA.6 Communications**	9.3 Nuclear thermal
	6.1 Communications	9.4 Electromagnetic
TA.2 Power		
2.1 Power generation	**TA.7 Attitude & GNC**	**TA.10 Environment, Humans and Safety**
2.2 Power distribution & management	7.1 Attitude	10.1 Radiation protection
2.3 Energy storage	7.2 Guidance & Navigation	10.2 Reduced gravity
	7.3 Control	10.3 Dust mitigation
TA.3 Thermal		10.4 Habitability
3.1 Thermal control	**TA.8 Life support**	10.5 EVA
3.2 Thermal protection	8.1 Air management	10.6 Crew health
3.3 Cryogenic systems	8.2 Water management	10.7 Fire detection & suppression
	8.3 Waste management	
TA.4 Robotics & Automation	8.4 Food management	**TA.11 Atmospheric descent & landing**
4.1 Sensing & perception		11.1 Atmospheric descent
4.2 Mobility, support & anchoring		11.2 Landing
4.3 Manipulation & anchoring		
4.4 Human-machine interface		
4.5 Cognition		
4.6 Autonomy		

Table 5.1: Technological areas and sub-areas

All the identified technologies are reported in tables 5.2-5.9 which represent quite a wide reference database.

Not all the database technologies are mapped on the HSE reference scenario elements

Technological area: TA.1 Structures and Mechanisms		
Technological sub-area Technical category	**Technologies**	**Variants**
1.1 Structures		
Advanced rigid structures	Advanced Al Alloy structures	Al-Li alloy Al-Ti alloy Al-Sc alloy
	Other metals structures	Titanium
	Advanced Composite structures	Al MMC Al Honeycomb Graphite epoxy resin Thermoplastic
	Open cells resin foams structures	BASF melamine - Basotec
	Advanced deployable structures	Ultra-light rigid Flexible
	Multifunctional structures	Rigid Flexible
	Smart nano-structures	
	Pressurized inflatable structures	
	Boom & modular structures	
	Advanced secondary/tertiary structures	Flexible bags
	Structures Health Monitoring and Control Techniques	Self healing structures Advanced techniques
1.2 Mechanisms		
Docking mechanisms	**In-space advanced docking mechanisms**	Unmanned docking systems IBDM/iLIDS/NDS
	Surface docking mechanisms	
Generic mechanisms	**Low-cyclic deploying mechanisms**	
	Low-cyclic extension mechanisms	
	High-cyclic long life pointing mechanisms	
	Low-speed surface deployment mechanisms	
Specific mechanisms	**Sampling mechanisms (drilling, collection)**	
1.3 Separations		
Separations	**Advanced pyrotechnique separations**	Low shock
	Non-explosive separations	
	Hot structures separations	

Table 5.2: TA.1 Structures and Mechanisms

Technological area: TA.6 Communications		
Technological sub-area Technical category	**Technologies**	**Variants**
6.1 Communications		
Link type	X-band	
	Ka-band	
	Advanced optical	Laser
	Wireless	WLAN IEEE 802.16
	Advanced fibers	
Hardware	Advanced high gain antennas - 6m	Rigid Inflatable
	Advanced transreceivers	Integrated nav/com Adjustable link
	Advanced software defined radio	Integrated nav/com Adjustable link
	IP-based radios	
	High power dual band TWTA	
Concepts - Architectures	Deep Space Network	
	Delay tolerant network	
	METERON	
	Internetworking	

Table 5.3: TA.6 Communications

Technological area: TA.2 Power		
Technological sub-area **Technical category**	**Technologies**	**Variants**
2.1 Power Generation		
Photovoltaic	Solar concentrators	
	High efficiency solar cells	Advanced MJ AsGa IMM
	Environment Resistant Solar Cells	
	Flexible solar array	CNT-enhanced Thin filmed rollout Ultra flex OPV
Electro-chemical	**Advanced non-regenerative power systems**	Batteries Fuel cells
Nuclear	**Dynamic conversion fission reactor**	Stirling - 20/30/40 kW Bryton - 50 kW Thermoelectric
	Advanced radio-isotope generators	Thermoelectric MMRTG DISP Stirling (2.5/5/10 kW)
Thermal	Heat engines (Stirling cycle)	
2.2 Power Distribution and Management		
Management	**Advanced PCU**	
	Advanced conversion/regulation systems	
Distribution	**Advanced cables/connectors**	CNT Superconductors
	Wireless power transmission	
2.3 Energy storage		
	Advanced regenerative batteries	Nano-enhanced High-specific Li-ion Supercapacitors
	Regenerative fuel cells	High-T PEM High pressure EZ - 100 bar
	Advanced flywheels	
	Electric and magnetic field storage	

Table 5.4: TA.2 Power

Technological area: TA.7 Attitude & GNC		
Technological sub-area **Technical category**	**Technologies**	**Variants**
7.1 Attitude		
	Local/terrain trackers	
7.2 Guidance & Navigation		
Guidance	**Relative guidance algorithm**	NASA ALHAT NASA crater
	Hazard detection & avoidance algorithms	SIFT
	Advanced aerocapture algorithms	
	Surface mobility algorithms	
Navigation	Fast acquisition GPS receiver	
	IMU & accelerometers suite	
	Deep Space Navigation	NASA XNAV
	Ascent navigation package	
	Low wight hybrid navigator	
7.3 Control		
	Advanced reaction wheels	

Table 5.5: TA.7 Attitude and GNC

(see section 5.1.2 for the details of the mapping analysis), but a subset is selected (represented by the technologies indicated in red in tables 5.2-5.9) according to their effective growing potential, the TAS-I interest and the actual Technology Readiness Level. In particular, technologies with TRL<2 are discarded, since having TRL<2

Technological area: TA.3 Thermal		
Technological sub-area Technical category	**Technologies**	**Variants**
3.1 Thermal Control		
Passive	Advanced coatings	Variable emissivity electro-cromic Spray on foam insulation
	Low conductivity materials	Ceramic
	Felt reusable surface insulation (FRSI)	
	Advanced MLI	Flexible Integrated MLI Load responsive MLI MMOD integrated MLI LDMLI Thinner Mars surface MLI
Active heat acquisition & generation	**Advanced heat exchanger**	Composite Dual barrier Space fission Micro channel - Water/gas separat.
	High-T heat pump/compressor	
	High performance heaters	
Active heat transfer	**2-phases heat transfer loop**	
	Advanced heat pipes	Liquid metal
	Advanced thermal fluids	
Active heat refection & accumulation	**Advanced radiators**	DPR
	Phase change materials	
	Advanced heat sinks	Ammonia boilers Ice-based fusible Evaporators
3.2 Thermal Protection		
	Advanced LI 900-2200	
	BRI-8	
	Advanced SLA	Rigid Spray
	High-density carbon phenolic	
	Mid-density carbon phenolic	
	PICA/PICA-like	
	Advanced smart reusable TPS	
	Flexible/inflatable TPS	
	Plume shielding systems	
	TPS sensors	
3.3 Cryogenic Systems		
Thermal control	**Advanced LBO-ZBO concept**	Sun shield Vacuum jacket Broad area cooling Passive thermodynamic vent sys Vapor cooled shield
	LBO-ZBO cryocoolers	Stirling Pulse tube
	Internal tank HX	
Fluid management	**Advanced cryo transfer concept**	No-vent fill Micro-g thruster settling
	Miscellaneous cryogenic components	Turbopumps CO2 freezer Propellant management device

Table 5.6: TA.3 Thermal

means that the technology development has not started yet [20] and this implies large uncertainties on its implementation; for this reason, other alternatives are preferred to those technologies with such low TRL. Obviously, all the technologies required for Mars mission are taken into account.

Technological area: TA.4 Robotics & Automation		
Technological sub-area Technical category	Technologies	Variants
4.1 Sensing & Perception		
	Stereo vision - 3D camera	
	TOF camera	
	Miniaturized optical sensor	
	Kinetics	
	LIDAR	Single Flash
	Radar altimeter	
4.2 Mobility, Support & Anchoring		
Advanced surface locomotion	Advanced scalable chassis	NASA mobility chassis
	Advanced wheels	
	Advanced suspensions	Active
	Advanced brakes	Regenerative
Support	Platform support	
Anchoring	Smart thethers	
4.3 Manipulation & Capture		
	Dexterous manipulators	
	Robotic hands	DLR hand Shadow
	Integrated tactile sensors	
	Grasping systems	
4.4 Human-machine interface		
Advanced human-machine interface	Haptic systemss	
	Exoskeletons	
	Immersed reality	
	Increased reality	
	Data glove	
	Ex-arm	
	Cyber hand	
4.5 Cognition		
	Artificial intelligence	
4.6 Autonomy		
	Autonomous VSM	
	Autonomous FDIR	
	Adjustable autonomy	
	Autonomous environment adaptation	

Table 5.7: TA.4 Robotics and Automation

Technological area: TA.5 Avionics		
Technological sub-area Technical category	Technologies	Variants
5.1 Avionics		
	Radiation-hardened multi core processor	MIC
	Next generation atomic clocks (ISS ACES)	Lightweight

Table 5.8: TA.5 Avionics

5.1.2 Technologies mapping

The mapping of the selected technologies (83 out of about 160 technologies identified in total) on the reference scenario is performed following three major steps: it starts with the technologies mapping on the scenario elements (applicability map), then it proceeds with the mapping on all destinations and eventually it ends with the assessment of the most required technologies roadmaps.

Technological area: TA.11 Atmospheric Descent & Landing		
Technological sub-area Technical category	Technologies	Variants
11.1 Atmospheric descent		
	Deployable supersonic decelerators	Supersonic parachutes Ballutes Parafoil Inflatable decelerators
11.2 Landing		
	Crashable structures	
	Active damping systems	
	Airbags	Surface Water

Table 5.9: TA.11 Atmospheric Descent & Landing

Technological area: TA.8 Life Support		
Technological sub-area Technical category	Technologies	Variants
8.1 Air Management		
Air regeneration	CO2 removal/collection systems	Micro-channel adsorption - MCATS 2-stages compressor mechanical pump
	CO2 reduction systems	Sabatier reactor Bosh reactor RWGS reactor
	O2 generation systems	PEM electrolyzer SOCE (CO2) electrolyzer
	ARES	
	Artificial photosynthesis	
	Regenerative TCC systems	Reg. activated charcoal Reg. sorption techniques Photocatalysis Reg. catalytic oxidation
8.2 Water Management		
Water regeneration	Distillation phase change	VCD VPCAR
	Filtration	MF Catalytic oxidation UV/visible photocatalyst
	Others	FDH Brine de-watering
8.3 Waste Management		
Waste compacting	Plastic waste melt compactor	
Waste processing	SCWO	
	Wet Oxidation	
	Dry/EC inceneration	
Re-using concepts	Methane recovery	
	ISS waste utilization process	
8.4 Food Management		
Preparation, conservation & packaging	Liofilization	
Low production	Food complement unit	
Close loop high production	Advanced food systems	
8.5 Hybrid Processes		
MELISSA	C1-C2-C3 waste processing	
	C4A photo-autrophic bacteria	
	C4B higher plants	

Table 5.10: TA.8 Life Support

Technological area: TA.9 Propulsion		
Technological sub-area Technical category	Technologies	Variants
9.1 Chemical		
C Micro propulsion	Cold/hot gas thrusters	
Liquid mono-prop	Hydrazine 2.5-3.5 kN	
Liquid bi-prop	**Pressure-fed storable**	MON/MMH 550 N EAM 1-1.5 kN MON/MMH 4kN MON/MMH 12kN NTO/MMH 33.5kN
	Pump-fed cryogenic	LOX/Ethanol LOX/LCH4 10 kN LOX/LCH4 66 kN LOX/LCH4 132 kN
	Cryogenic	LOX/LH2 44.5kN (RL10 deriv.) LOX/LH2 66kN (RL10 derivative) LOX/LH2 133kN (RL10 derivative) J-2X orbital LOX/LH2 1300kN
	Advanced solid	
	Advanced hybrid	
9.2 Electric		
	E Micro-propulsion	Electrospray
Solar	Hall thrusters	
	Grid-ion engine	3.5 kW >5kW
	Pulse inductive thrusters	20-30 kW
Nuclear/solar	Advanced MPDT	
	VASIMR	VF-200 200kW 12MW
9.3 Nuclear Thermal		
	NTR fission reactor	**NERVA-like 67-111kN** RD-0410 Particle bed reactor - 300kN **Bimodal NTR**
	High temperature fuels	Composite fuel - UC2-ZrC
9.4 Electromagnetic		
	Railguns	

Table 5.11: TA.9 Propulsion

5.1.2.1 Applicability map

The “applicability analysis” is performed to verify in which HSE missions/elements the identified technologies are absolutely required or can be anyway implemented, tested and validated. This analysis consists in mapping the technologies on the HSE reference scenario elements and is performed per classes of elements. As explained in section 4.3.1, in the “Elements Commonalities Analysis”, the elements are grouped in 16 classes of elements, which include similar elements satisfying more and more demanding requirements. Specifically, the objective of the “applicability analysis” is to build, for each elements class, a matrix describing if and how the technologies can be implemented in the missions elements, considering that, with respect to an element, a technology can be:

Technological area: TA.10 Environment, Humans & Safety		
Technological sub-area Technical category	**Technologies**	**Variants**
10.1 Radiation protection		
Monitoring	Passive dosimeters	
	Active dosimeters	
	Biodosimetry	
Mitigation - Passive	**Advanced shielding materials**	Carbon nanotube composites Boron carbide - B4C Boron nitride NT - BNNT Tungsten H2-filled MOF
	Local regolith shielding	
	Advanced shielding concepts	Integrated radiation suit H2O dedicated tankage H2O FlexBag
Mitigation - Active	Electromagnetic shield	
	Electrostatic plasma shield	
10.2 Reduced gravity		
	Artificial gravity concept	
	Internal centrifuge	
	Reduced gravity exercise kit	JPL 0-g exercise kit
10.3 Dust mitigation		
Inside	Dust lock oversuit	
	SPARCLED	
Outside	Dust improved HW	
	Electrodynamic dust shield	
	Lotus coatings	
10.4 Habitability		
	In-space micro-gravity habitat	Rigid concept Inflatable concept
	Surface reduced gravity habitat	Rigid concept Inflatable concept
10.5 EVA		
Preparation & ingress/egress	**Suit port**	
	Inflatable airlock	
Protection	**Advanced suit**	MIT biosuit Advanced suit and PGA
Support	**Advanced PLSS**	
	Advanced mobility jet pack	
	Advanced EVA tools	Integrated
	Advanced EVA equipment	Actuation assistance gloves
Exploration	Sample containers	
10.6 Crew Health		
	In-flight surgery	
	Biomedical sensors	
	Lab-on-a-chip	
10.7 Fire Detection & Suppression		
Detection	Advanced ionization SD	
	Advanced photo-electric SD	
Suppression	Fine mist water spray estinguisher	
	Advanced suppression systems	

Table 5.12: TA.10 Environment, Humans & Safety

- required, if enabling or significantly impacting on the overall mission/architecture;
- applicable, if possible to be implemented, even if not strictly required;
- demo, if it can be implemented as a demo while being required for a following mission;
- not applicable, if not possible to be implemented.

Figures 5.1-5.16 report the matrices obtained for the 16 elements classes. According to the color of the cell, they indicate if the listed technologies are required (red), applicable (blue), demo (yellow) or not applicable (white) on the elements belonging to the specific classes.

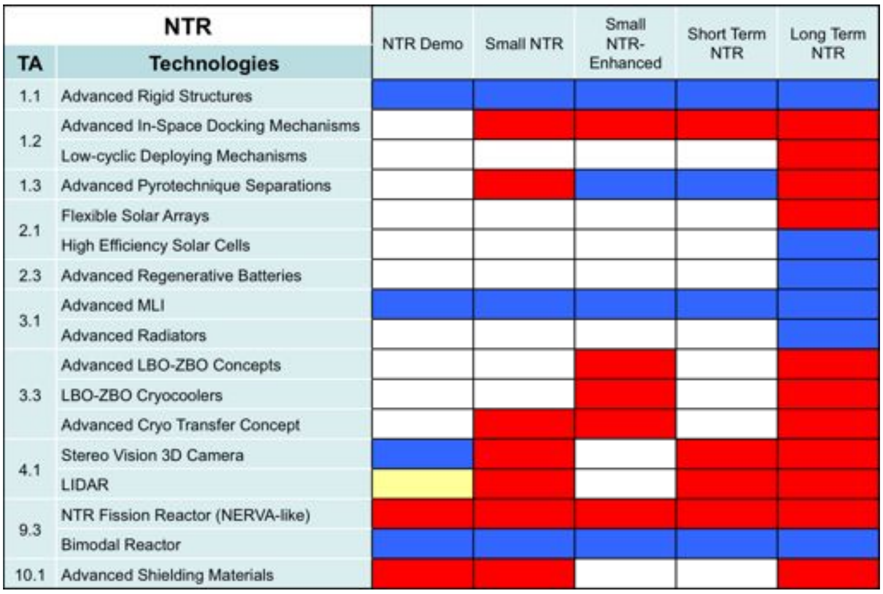


Figure 5.1: Technologies mapping on nuclear thermal rocket class

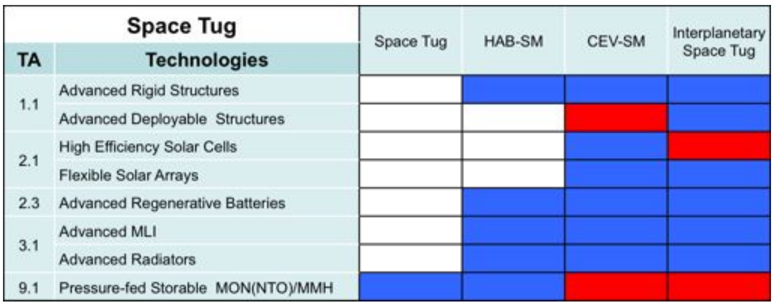


Figure 5.2: Technologies mapping on space tug class

The assessment of the "applicability" is performed by considering some reference designs [3, 21, 22, 23, 24, 25] or some assumed requirements for the elements. This is particularly true for the required technologies, while the applicable, demo and not applicable technologies mainly rely on evaluations of similar elements or on considerations about the environment and the type of module (e.g. the reference design does not fore-

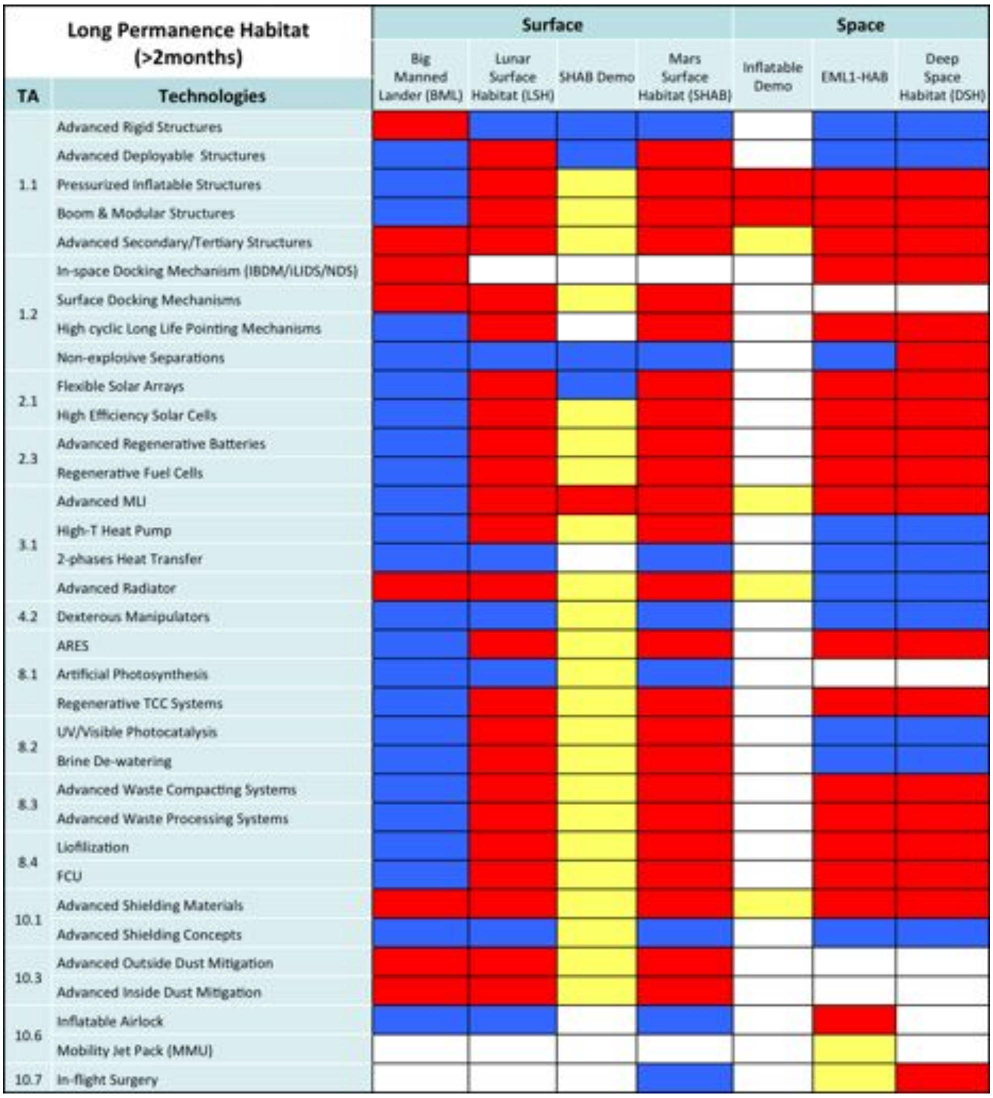


Figure 5.3: Technologies mapping on long permanence habitat class

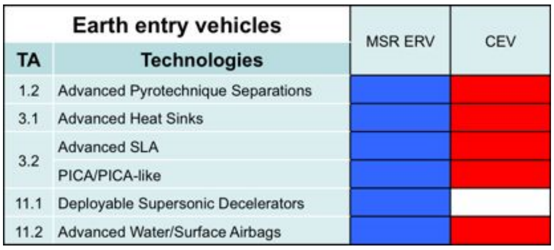


Figure 5.4: Technologies mapping on Earth entry vehicle class

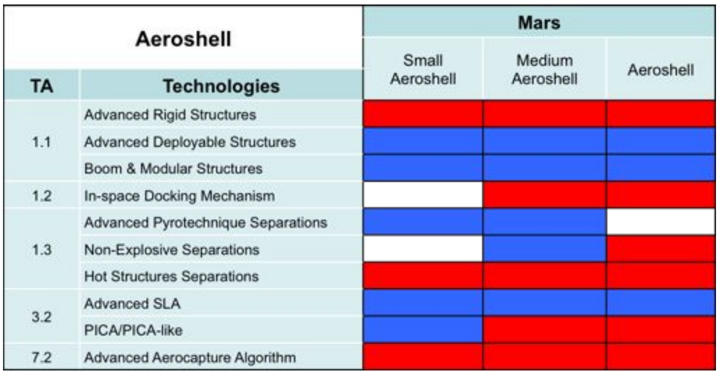


Figure 5.5: Technologies mapping on aeroshell class

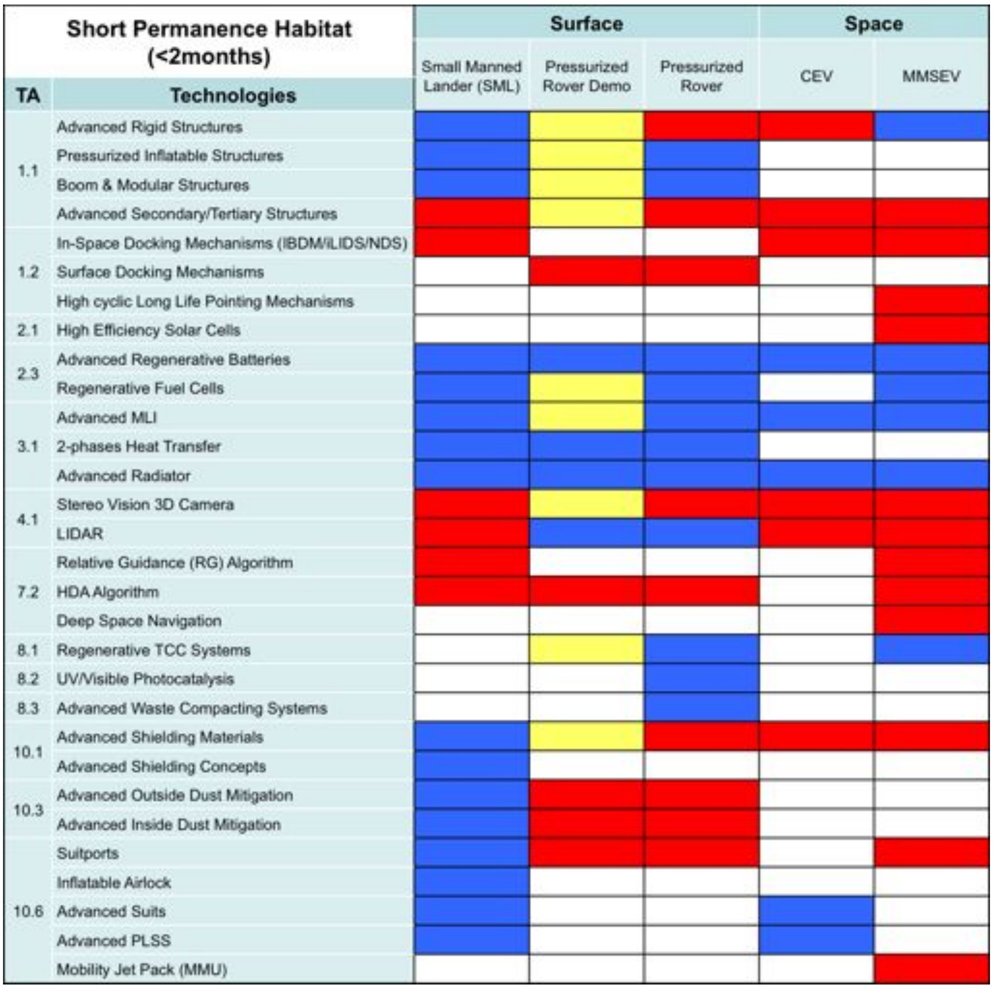


Figure 5.6: Technologies mapping on short permanence habitat class

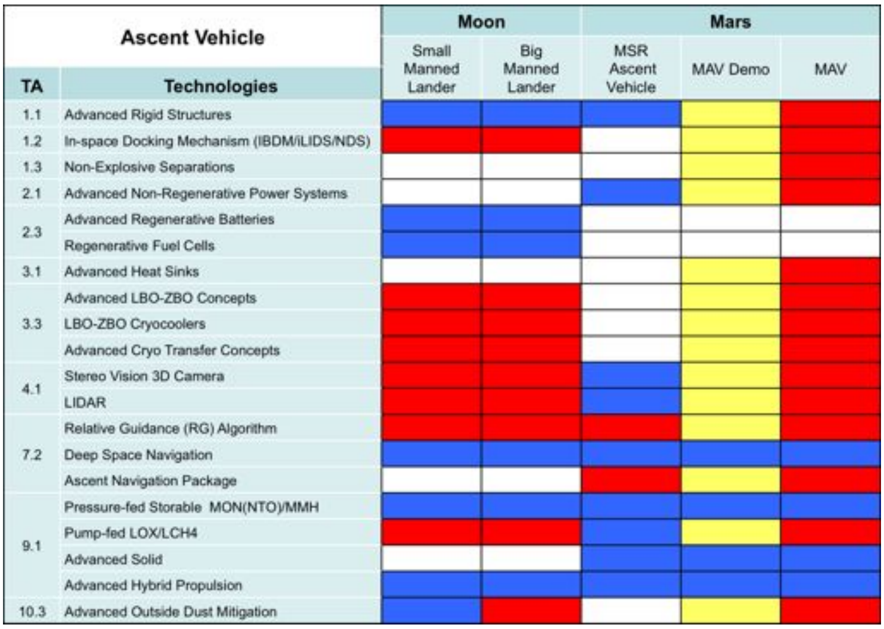


Figure 5.7: Technologies mapping on ascent vehicle class

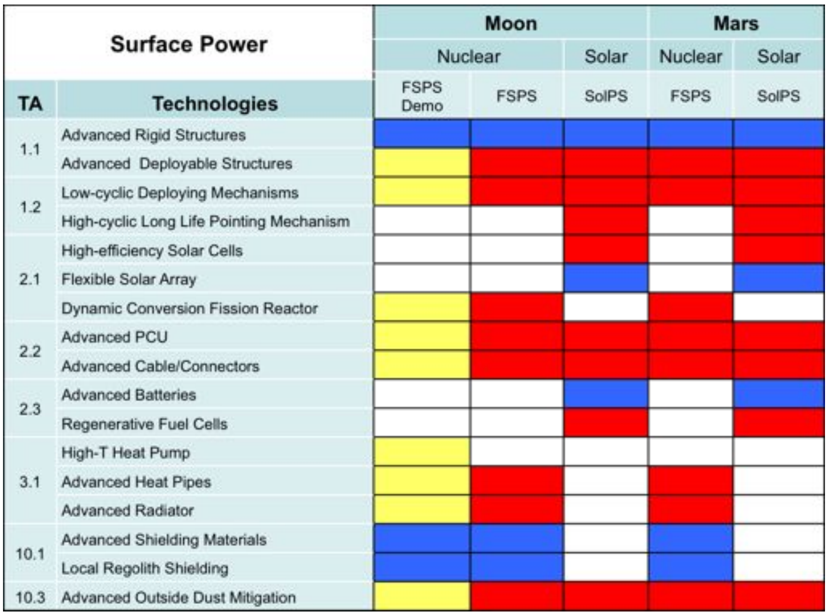


Figure 5.8: Technologies mapping on surface power class

see a specific technology, which anyway could be implemented on the module according to the mission it has to accomplish).

For example the inflatable demo element is a module envisaged to validate the in-

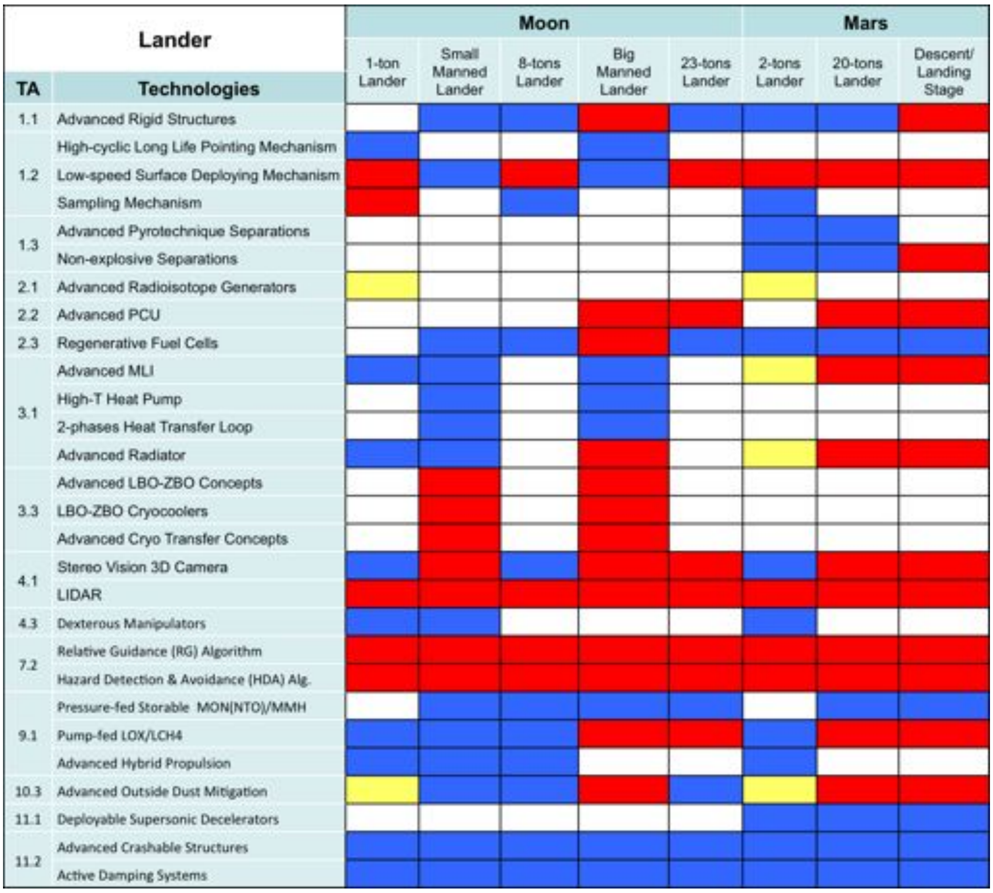


Figure 5.9: Technologies mapping on lander class

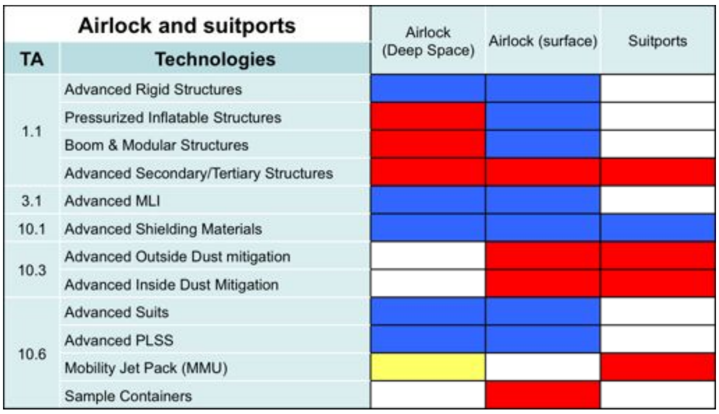


Figure 5.10: Technologies mapping on airlock and suit ports class

flatable technology, which is indeed a required technology (i.e. pressurized inflatable

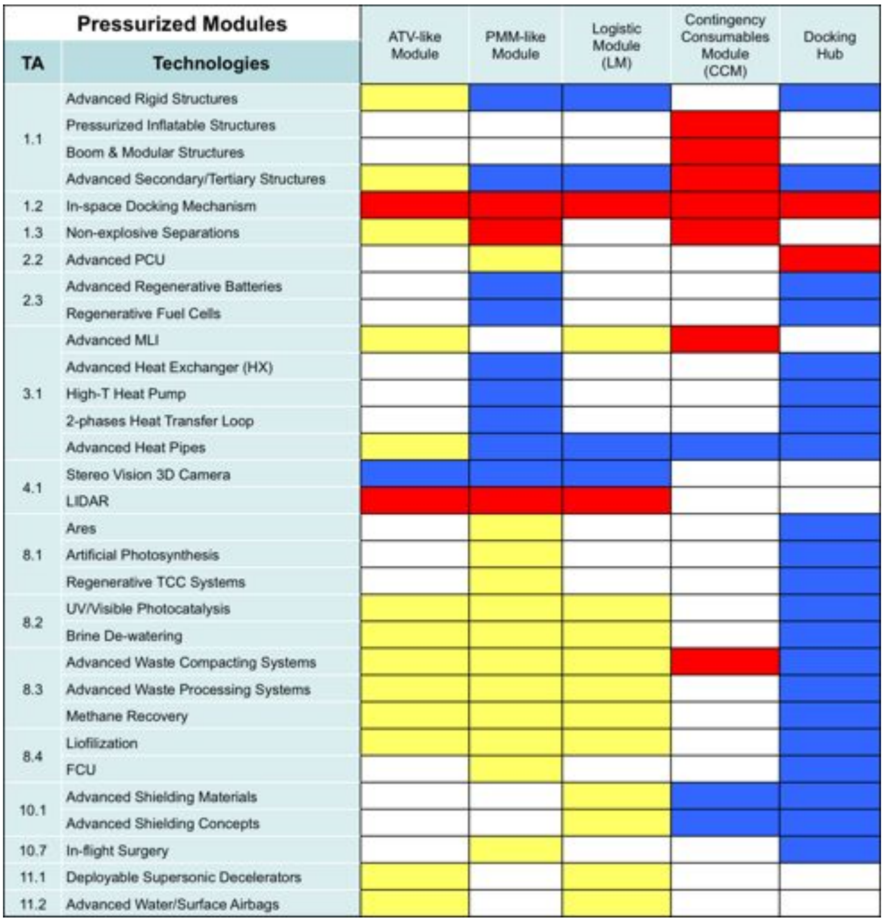


Figure 5.11: Technologies mapping on short pressurized modules class

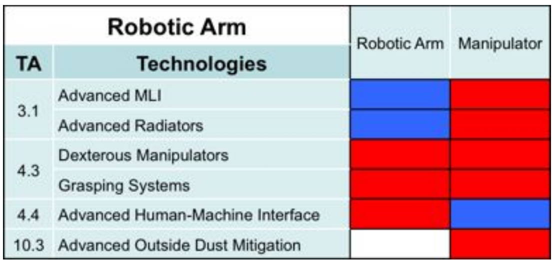


Figure 5.12: Technologies mapping on robotic arms class

structures); however some additional technologies could be included as demo on the module (e.g. advanced secondary/tertiary structure).

The obtained matrices represent the starting point to proceed with the mapping on the destinations of the HSE reference scenario.

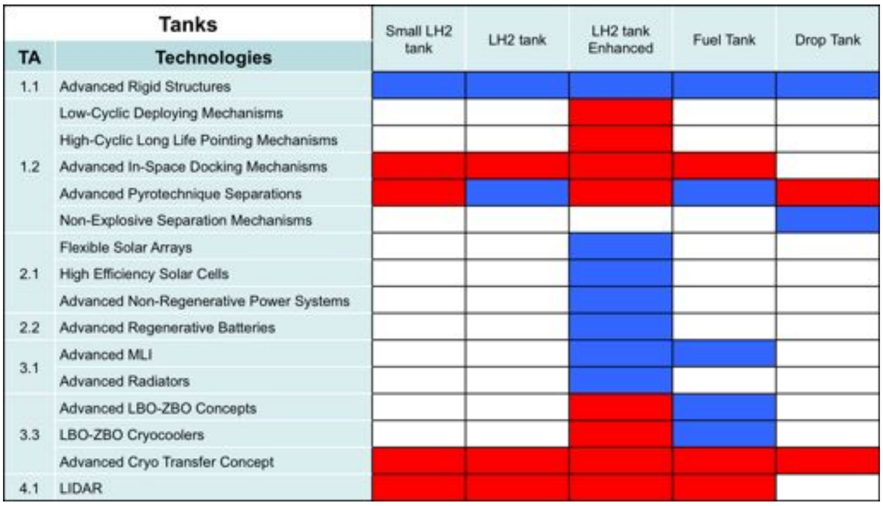


Figure 5.13: Technologies mapping on tanks class

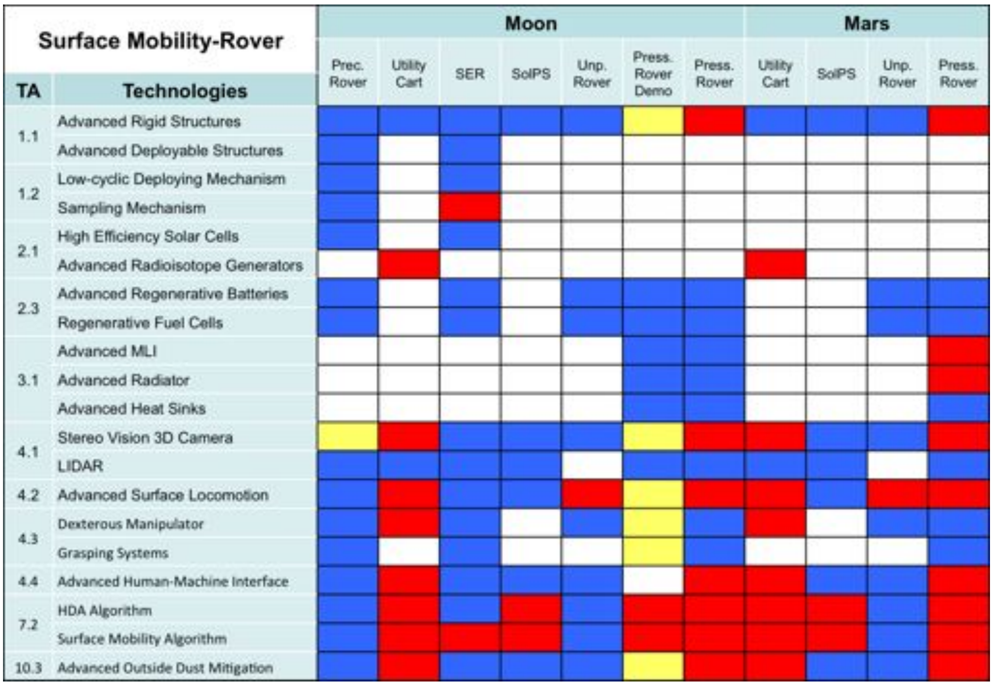


Figure 5.14: Technologies mapping on surface mobility rover class

5.1.2.2 Mapping on intermediate destinations

The technical database described in previous section can now be used to support strategic decisions, in the context of a flexible path scenario for exploration. In particular,

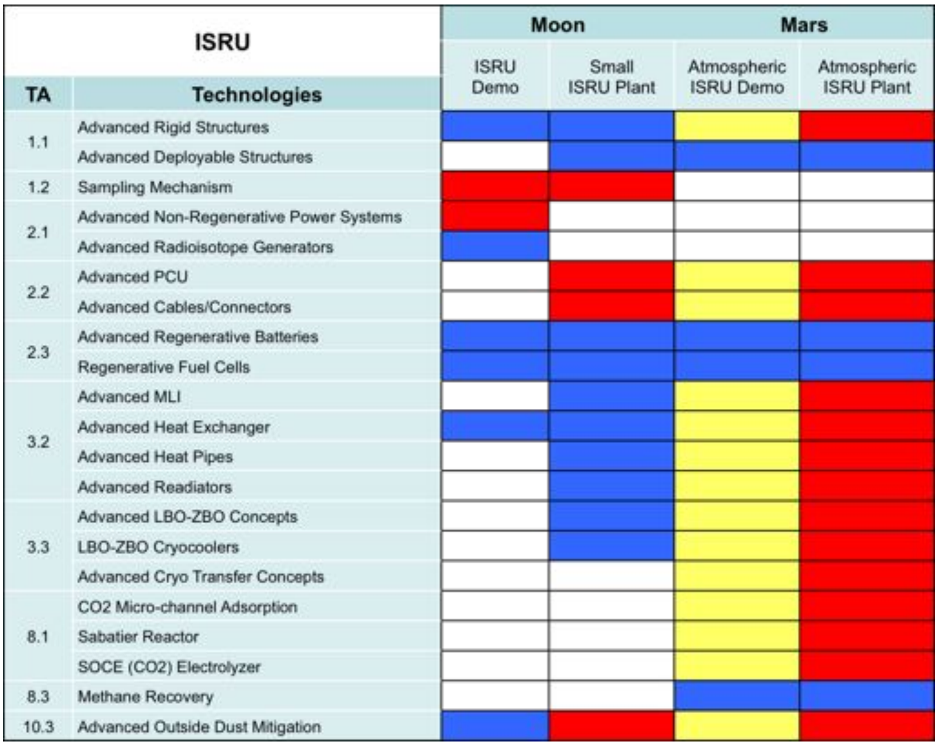


Figure 5.15: Technologies mapping on ISRU class

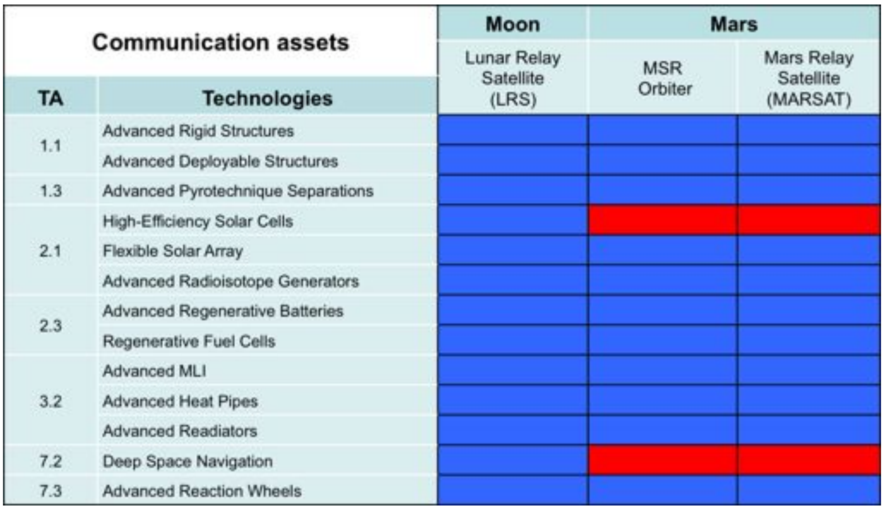


Figure 5.16: Technologies mapping on communication assets class

to support technologies development strategic decisions, the database can be exploited to understand which are the most required/applicable technologies, referring both to a

single destination and to the whole scenario. Moreover, according to the defined HSE reference scenario, it is possible to understand when the technology shall be ready to be implemented in a specific mission element.
Starting from the 16 produced matrices, the required and applicable technologies are mapped on the various destinations of the HSE scenario. Eventually, by summarizing and processing the obtained results, it is possible to rank the most required technologies, thus generating the so-called technologies roadmaps.
For each destination, the elements that need a certain technology are counted, in order to have a clearer view of which are the most required technologies. All required technologies are taken into account. Analogously, the applicable technologies are processed to have a summary of how many (and which) elements could potentially be exploited to validate these technologies before their actual implementation.
Tables 5.13 - 5.30 summarize the mapping of the technologies throughout the HSE scenario: two tables for each considered technological area* are provided, referring to required and applicable technologies, respectively.
Specifically, tables with red background refer to required technologies, which are ordered starting from those required in the largest number of elements. The numbers reported in the cells indicate the number of elements requiring the specific technology for each destination concept (recurrent units are not included in these numbers); moreover, the total number of elements with respect to the whole scenario is specified. Finally, the first time the technology is needed is highlighted, showing both the first element of the scenario in which it shall be implemented (column "1st Element") and the year when it is required for the first time (column "Year").
Similarly, tables with blue background report the elements in which the technology can be implemented as applicable, specifying the total number for each destination, as well as showing the total number of elements with respect to the whole scenario (the most significant elements are highlighted as well).
In the following, a brief discussion is reported for the different technological areas, underlining the major conclusions.

TA.1 Structures and Mechanisms

Tables 5.13 and 5.14 report the mapping on HSE scenario destinations of required

*Nine out of the eleven technological areas are considered, since only some technologies are taken into account according to what explained in section 5.1.1

and applicable technologies, respectively, referred to the "Structures and Mechanisms" technological area.

Required technologies	HSE destinations/concepts							Total		
	ISS	CL	MS	MO	NEA	MP	Mars	#	1st element	Year
In-space advanced docking mechs	2	4	5	4	5	3	7	30	ATV-like	2014
Advanced secondary/tertiary str.		3	5	5	4		7	24	EML1-HAB	2017
Advanced rigid structures		1	3	2	1	5	5	17	CEV	2018
Advanced pyrotechnic separations		2	3	3	4	1	4	17	CEV	2018
Advanced deployable structures		1	2	4	1	2	4	14	CEV-SM	2018
High-cyclic long life pointing mech		1	1	2	3	1	3	11	EML1-HAB	2017
Low-cyclic deploying mechanisms			1	2	1	3	4	11	SolPS	2022
Non-explosive separations	1				1	2	5	9	PMM-like	2014/15
Boom & modular structures	1	2		1	1		3	8	Inflat. demo	2015
Pressurized inflatable structures	1	2		1	1		3	8	Inflat. demo	2015
Low-speed surface deployment			2	1		3	1	7	1-ton lander	2022
Surface docking mechanisms			3	2			2	7	PR-demo	2023
Sampling mechanisms			3	1				4	1-ton lander/SER	2022
Hot structures separations						3	1	4	Small aeroshell	2024

Table 5.13: Required Technologies Mapping on HSE scenario destinations - TA.1 Structures and Mechanisms

A new *in-space advanced docking mechanism* is required in numerous missions and the first possibility to use it is an ATV-like cargo mission to the ISS in 2014. *Advanced secondary/tertiary structures* are needed for CL, MS, MO, NEA and human Mars mission concepts, but they can be implemented and tested in simpler missions to ISS before 2017.

Applicable/Demo technologies	HSE destinations/concepts							Total
	ISS	CL	MS	MO	NEA	MP	Mars	#
In-space advanced docking mechs						MAV demo		1
Advanced secondary/tertiary str.	ATV-like, PMM-like, infl. demo	LM	2	LM		SHAB demo	1	9
Advanced rigid structures	ATV-like, PMM-like, NTR demo	6	17	12	8	14	13	73
Advanced pyrotechnic separations			Fuel tank, LRS	2	1	8	1	14
Advanced deployable structures		EML1-HAB	5	1	1	9	3	20
High-cyclic long life pointing mech			1-ton lander, BML					2
Low-cyclic deploying mechanisms			3					3
Non-explosive separations	ATV-like	EML1-HAB	BML	LSH	Drop tank	5	2	12
Boom & modular structures			5	2		4	3	14
Pressurized inflatable structures			5	2		1	2	10
Low-speed surface deployment			SML, BML					2
Surface docking mechanisms						SHAB demo		1
Sampling mechanisms			8-ton lander prec. rover			2-tons lander		3
Hot structures separations								

Table 5.14: Applicable Technologies Mapping on HSE scenario destinations - TA.1 Structures and Mechanisms

Analogously, *advanced rigid structures* are applicable to a large number of elements.

The *pressurized inflatable structures* are required in less elements (eight in total), especially in human Mars mission concept elements, but they are applicable to Moon sortie and Moon outpost elements.
Concerning the separations, *advanced pyrotechnic separations* are required in many elements, starting with the CEV in 2018. They are also applicable to a large number of elements, especially in the Mars preparation concept.

TA.2 Power

Tables 5.15 and 5.16 report the mapping on HSE scenario destinations of required and applicable technologies, respectively, referred to the "Power" technological area. An

Required technologies	HSE destinations/concepts							Total		
	ISS	CL	MS	MO	NEA	MP	Mars	#	1st element	Year
Advanced PCU			2	4		5	5	16	SolPS	2022
High-efficiency solar cells		1	1	2	2	4	3	13	EML1-HAB	2017
Regenerative fuel cells		1	2	2	1	1	3	10	EML1-HAB	2017
Advanced cables/connectors			1	3		3	3	10	SolPS	2022
Flexible solar arrays		1		1	2	1	3	8	EML1-HAB	2017
Advanced regenerative batteries		1		1	1		2	5	EML1-HAB	2017
Advanced radioisotope generators			1			1	1	3	Utility cart	2022
Dynamic conversion fission reactor				1		1	1	3	FSPS	2029
Advanced non-regenerative power systems			1				1	2	ISRU demo	2026

Table 5.15: Required Technologies Mapping on HSE scenario destinations - TA.2 Power

Applicable/Demo technologies	HSE destinations/concepts							Total
	ISS	CL	MS	MO	NEA	MP	Mars	#
Advanced PCU	PMM-like		FSPS demo			Atm. ISRU demo		3
High-efficiency solar cells		CEV-SM	5	1	2	2	3	14
Regenerative fuel cells	PMM-like		9	3	1	8	5	27
Advanced cables/connectors			FSPS demo			Atm. ISRU demo		2
Flexible solar arrays		CEV-SM	4	2	1	5	3	16
Advanced regenerative batteries	PMM-like	HAB-SM, CEV, CEV-SM	6	3	4	3	6	26
Advanced radioisotope generators			3			3		6
Dynamic conversion fission reactor			FSPS demo					1
Advanced non-regenerative power systems						2	LH2 tank enhanced	3

Table 5.16: Applicable Technologies Mapping on HSE scenario destinations - TA.2 Power

advanced PCU shall be developed because it is considered needed for the complex surface exploration systems in the MS, MO, MP and human Mars mission concepts. It can be installed as demo in a PMM-like mission to the ISS.
High-efficiency solar cells are needed for almost all destinations. They can be firstly demonstrated on the CEV-SM.
Regenerative fuel cells are especially required for surface applications, but they can be tested at the ISS in a PMM-like module. They can also be applied to a large number

of elements in place of regenerative batteries.

Flexible solar arrays and *advanced regenerative batteries* can be exploited with significant advantages in a large set of missions.

TA.3 Thermal

Tables 5.17 and 5.18 report the mapping on HSE scenario destinations of required and applicable technologies, respectively, referred to the “Thermal” technological area.

Required technologies	HSE destinations/concepts							Total		
	ISS	CL	MS	MO	NEA	MP	Mars	#	1st element	Year
Advanced cryo transfer		1	5	3	5	3	6	23	Small NTR	2018
Advanced MLI		1	1	2	1	5	7	17	EML1-HAB	2017
Advanced radiators			2	3		5	6	16	Manipulator	2022
Advanced LBO-ZBO concept			2		2	2	4	10	SML	2020
LBO-ZBO cryocoolers			2		2	2	4	10	SML	2020
PICA/PICA-like		1	1	1	1	2	2	8	CEV	2018
Advanced heat pipes				1		2	2	5	FSPS	2029
Advanced SLA		1	1	1	1		1	5	CEV	2018
Advanced heat exchanger						1	1	2	Atm. ISRU plant	2030
High-T heat pump				1			1	2	LSH	2029
Advanced heat sinks							1	1	MAV	2037
2-phases heat transfer loop								0		

Table 5.17: Required Technologies Mapping on HSE scenario destinations - TA.3 Thermal

Advanced cryo transfer concepts are required to support cryogenic propulsion systems,

Applicable/Demo technologies	HSE destinations/concepts							Total
	ISS	CL	MS	MO	NEA	MP	Mars	#
Advanced cryo transfer						2		2
Advanced MLI	ATV-like, infl. demo, NTR demo	7	12	9	6	6	5	48
Advanced radiators	Infl. demo	5	8	4	5	7	5	35
Advanced LBO-ZBO concept			Fuel tank	2		2		5
LBO-ZBO cryocoolers			Fuel tank	2		2		5
PICA/PICA-like						Small aeroshell, MSR EVR		2
Advanced heat pipes	ATV-like, PMM-like	LM	3	2		3	2	13
Advanced SLA						4	1	5
Advanced heat exchanger	PMM like		ISRU demo	1		1	1	5
High-T heat pump	PMM-like	EML1-HAB	3		1	1	2	9
Advanced heat sinks			PR demo, Press. rov	1		MAV demo	1	5
2-phases heat transfer loop	PMM-like	EML1-HAB	4	2	1		4	13

Table 5.18: Applicable Technologies Mapping on HSE scenario destinations - TA.3 Thermal

needed for missions belonging to MS, MO, NEA, MP and the Mars human concepts. There is the possibility to firstly implement such technologies in the small NTR included in numerous missions to the cis-lunar infrastructure starting from 2018.

Advanced MLIs are also needed, especially for the MP and human Mars mission concepts, but a preliminary and deep space configuration shall be already developed for the Cis-lunar concept with a first implementation in EML1-HAB in 2017.

Advanced radiators are also required especially for surface applications in the MS, MO, MP and human Mars mission concepts.

TA.4 Robotics and Automation

Tables 5.19 and 5.20 report the mapping on HSE scenario destinations of required and applicable technologies, respectively, referred to the "Robotics and Automation" technological area.

Required technologies	HSE destinations/concepts							Total		
	ISS	CL	MS	MO	NEA	MP	Mars	#	1st element	Year
LIDAR	3	3	8	7	5	5	6	37	ATV-like	2014
Stereo vision 3D camera		2	6		2	2	6	18	Small NTR	2018
Advanced surface locomotion			3	1			3	7	Utility cart	2022
Advanced human-machine I/F		1	2	1			2	5	Robotic arm	2017
Dexterous manipulator		1	2	1			2	5	Robotic arm	2017

Table 5.19: Required Technologies Mapping on HSE scenario destinations - TA.4 Robotics and automation

Applicable/Demo technologies	HSE destinations/concepts							Total
	ISS	CL	MS	MO	NEA	MP	Mars	#
LIDAR	NTR demo		6			2	3	12
Stereo vision 3D camera	ATV-like + 2	1	7			3	2	16
Advanced surface locomotion			Prec. rover + 2				1	4
Advanced human-machine I/F			Prec. rover + 5	1		1	1	9
Dexterous manipulator		EML1-HAB	7	1	1	2	2	14

Table 5.20: Applicable Technologies Mapping on HSE scenario destinations - TA.4 Robotics and automation

LIDAR is one of the most required and applicable technologies, needed for autonomous rendezvous and docking and descent and landing operations. It is already required in 2014 for an ATV-like mission to the ISS.

Stereo vision 3D camera represents also a promising equipment to support autonomous RvD and D&L maneuvers. It is considered required for the small NTR in 2018, but it can be previously implemented and tested in an ATV-like mission to the ISS in 2014.

Advanced surface locomotion systems are needed for human Mars mission concept and considered required also for MS and MO missions. The lunar utility cart requires them in 2022 and they can be tested in the lunar precursor rovers, as well.

TA.7 Attitude and GNC

Tables 5.21 and 5.22 report the mapping on HSE scenario destinations of required and applicable technologies, respectively, referred to the "Attitude and GNC" technological area.

Required technologies	HSE destinations/concepts							Total		
	ISS	CL	MS	MO	NEA	MP	Mars	#	1st element	Year
HDA algorithms			8	4	1	5	4	22	1-ton lander	2022
Surface mobility algorithms			5	3		2	3	13	Utility cart	2022
Relative guidance algorithms			4	1		4	2	11	1-ton lander	2022
Advanced Aerocapt. algorithms						2	1	3	Small aeroshell	2024
Deep space navigation						2		2	MSR orbiter	2024
Ascent navigation package						1	1	2	MSR AV	2024

Table 5.21: Required Technologies Mapping on HSE scenario destinations - TA.7 Attitude and GNC

Applicable/Demo technologies	HSE destinations/concepts							Total
	ISS	CL	MS	MO	NEA	MP	Mars	#
HDA algorithms			Prec. rover + 2				1	3
Surface mobility algorithms			Prec. rover + 1				1	3
Relative guidance algorithms						MAV demo		1
Advanced Aerocapt. algorithms								0
Deep space navigation		HAB-SM + 2	5	3	3	3	3	20
Ascent navigation package						MAV demo		1

Table 5.22: Applicable Technologies Mapping on HSE scenario destinations - TA.7 Attitude and GNC

An advanced version of *HDA algorithms* is required for all the D&L and surface navigation operations, foreseen in several MS, MO, MP and human Mars missions elements, with a first implementation in the 1-ton lunar lander in 2022. The MMSEV in the NEA concept requires a similar technology too.

Surface mobility algorithms are also required for locomotion elements in main surface-based concepts; in particular, the first element to require them is the lunar utility cart in 2022.

An improved *relative guidance algorithm* is required for D&L operations in MS, MO, MP and human Mars mission concepts' elements. First implementation is needed in the 1-ton lunar lander in 2022.

TA.8 Life Support

Tables 5.23 and 5.24 report the mapping on HSE scenario destinations of required and applicable technologies, respectively, referred to the "Life Support" technological area. Five life support technologies are required for deep-space and surface long permanence habitats, which are ARES, regenerative TCC systems, advanced waste compacting systems, lyophilization and food complement unit. They initially are required for the EML1-HAB in 2017, but they can all be tested in a previous PMM-like module mission

Required technologies	HSE destinations/concepts							Total		
	ISS	CL	MS	MO	NEA	MP	Mars	#	1st element	Year
ARES		1		1	1		2	5	EML1-HAB	2017
Regenerative TCC system		1		1	1		2	5	EML1-HAB	2017
Advanced waste compacting sys		1		1	1		2	5	EML1-HAB	2017
Liofilization		1		1	1		2	5	EML1-HAB	2017
Food complement unit		1		1	1		2	5	EML1-HAB	2017
Advanced waste processing sys		1		1	1		1	4	EML1-HAB	2017
UV/Visible photocatalysis				1			1	2	LSH	2029
Brine De-watering				1			1	2	LSH	2029
CO2 Micro-channel adsorption							1	1	Atm. ISRU plant	2037
2-stages compressor mechanical pump							1	1	Atm. ISRU plant	2037
Sabatier reactor							1	1	Atm. ISRU plant	2037
SOCE (CO2) electrolyzer							1	1	Atm. ISRU plant	2037
Methane recovery								0	-	-
Artificial photosynthesis								0	-	-

Table 5.23: Required Technologies Mapping on HSE scenario destinations - TA.8 Life support

Applicable/Demo technologies	HSE destinations/concepts							Total
	ISS	CL	MS	MO	NEA	MP	Mars	#
ARES	PMM-like		1			1	1	4
Regenerative TCC sysm	PMM-like		3		1	1	2	8
Advanced waste compacting sys	PMM-like, ATV-like	1	1			1	2	7
Liofilization	PMM-like, ATV-like	1	1			1	1	5
Food complement unit	PMM-like		1			1	1	4
Advanced waste processing sys	PMM-like, ATV-like	1	1			1	1	6
UV/Visible photocatalysis	PMM-like, ATV-like	2	2		1	1	3	11
Brine De-watering	PMM-like, ATV-like	2	1		1	1	2	9
CO2 Micro-channel adsorption						Atm. ISRU demo		1
2-stages compressor mechanical pump						Atm. ISRU demo		1
Sabatier reactor						Atm. ISRU demo		1
SOCE (CO2) electrolyzer						Atm. ISRU demo		1
Methane recovery	PMM-like, ATV-like	1				1	1	5
Artificial photosynthesis	PMM-like		1	1		1	2	6

Table 5.24: Applicable Technologies Mapping on HSE scenario destinations - TA.8 Life support

to the ISS.

TA.9 Propulsion

Tables 5.25 and 5.26 report the mapping on HSE scenario destinations of required and applicable technologies, respectively, referred to the "Propulsion" technological area. The *NTR fission reactor (NERVA-like)* is the basis of all cargo and human high capability transfer stages belonging to the NTR element class. Several versions are envisaged in the various concepts, being the first required implementation in the NTR demo mission to ISS in 2016.

The *pump-fed LOX/LCH4* is a type of chemical cryogenic propulsion utilized for the

Required technologies	HSE destinations/concepts							Total		
	ISS	CL	MS	MO	NEA	MP	Mars	#	1st element	Year
NTR fission reactor (NERVA-like)	1	1	1	2	2	1	1	9	NTR demo	2016
Pump-fed LOX/LCH4			2	1		2	2	7	SML	2020
Pressure-fed storable MON/MMH		1	1	1	1	1	1	6	CEV-SM	2018
NTR fission reactor - bimodal								0		
Advanced hybrid propulsion								0		
Advanced solid propulsion								0		

Table 5.25: Required Technologies Mapping on HSE scenario destinations - TA.9 Propulsion

Applicable/Demo technologies	HSE destinations/concepts							Total
	ISS	CL	MS	MO	NEA	MP	Mars	#
NTR fission reactor (NERVA-like)								0
Pump-fed LOX/LCH4			SML + 2			3		6
Pressure-fed storable MON/MMH	Space tug	2	4	2		4	1	14
NTR fission reactor - bimodal	NTR demo	1	1	2	2	1	1	9
Advanced hybrid propulsion			SML + 2			1		4
Advanced solid propulsion						MSR asc. vehicle + 1	1	3

Table 5.26: Applicable Technologies Mapping on HSE scenario destinations - TA.9 Propulsion

D&L and ascent maneuvers. It is required in numerous elements of the MS, MO, MP and human Mars mission concepts with a first implementation in the SML in 2020. *Pressure-fed storable MON(NTO)/MMH* engines are required for the space tug class with a first implementation in 2018 in the CEV-SM. This technology can be previously implemented in the space tug used for ISS missions.

TA.10 Environment, Humans and Safety

Tables 5.27 and 5.28 report the mapping on HSE scenario destinations of required and applicable technologies, respectively, referred to the "Environment, Humans and Safety" technological area.

Required technologies	HSE destinations/concepts							Total		
	ISS	CL	MS	MO	NEA	MP	Mars	#	1st element	Year
Advanced outside dust mitigation			5	7		4	8	24	Utility cart	2022
Advanced shielding materials	1	3	3	4	3	1	4	17	NTR demo	2016
Advanced inside dust mitigation			3	4			3	10	PR demo	2023
Suit ports			2	1	1		1	5	PR demo	2023
In-flight surgery					1		1	2	DSH	2033
Advanced mobility jet pack					1			1	MMSEV	2031
Inflatable airlock		1						1	EML1-HAB	2017
Advanced suits			1					1	SML	2020
Local regolith shielding								0		
Advanced shielding concepts								0		
Advanced PLSS								0		

Table 5.27: Required Technologies Mapping on HSE scenario destinations - TA.10 Environment, humans and safety

Advanced outside dust mitigation technologies are highly required for surface applica-

Applicable/Demo technologies	HSE destinations/concepts							Total
	ISS	CL	MS	MO	NEA	MP	Mars	#
Advanced outside dust mitigation			SML + 7	2		5	1	16
Advanced shielding materials	Infl. demo	LM	3	2		1	4	12
Advanced inside dust mitigation			SML			1		2
Suit ports			SML					1
In-flight surgery		EML1-HAB					1	2
Advanced mobility jet pack		EML1-HAB			1		1	3
Inflatable airlock			SML, BML	1			1	4
Advanced suits		CEV	2	1	2		2	8
Local regolith shielding			FSPS demo, FSPS	1		1	1	5
Advanced shielding concepts		LM, EML1-HAB	2	1	1	1	4	11
Advanced PLSS		CEV	2	1	2		2	8

Table 5.28: Applicable Technologies Mapping on HSE scenario destinations - TA.10 Environment, humans and safety

tions in the MS, MO, MP and human Mars mission concepts. The lunar utility cart is the first element demanding such technology in a mission in 2022. Moreover the SML represents a possible technology test bench.

Advanced shielding materials are required for protection from both radiation generated from NTR and FSPS and deep space radiation. The NTR demo to ISS in 2016 is the first element needing this technology.

Advanced inside dust mitigation techniques are required for manned surface exploration activities in order to avoid inner habitats contamination. MS, MO and human Mars mission concepts' elements need these technologies. The SML can be considered a test bench element.

TA.11 Atmospheric Descent and Landing

Tables 5.29 and 5.30 report the mapping on HSE scenario destinations of required and applicable technologies, respectively, referred to the "Atmospheric Descent and Landing" technological area.

Required technologies	HSE destinations/concepts							Total		
	ISS	CL	MS	MO	NEA	MP	Mars	#	1st element	Year
Advanced water/surface airbags		1	1	1	1		1	5	CEV	2018
Deployable supersonic decelerators										
Advanced crushable structures										
Active damping system										

Table 5.29: Required Technologies Mapping on HSE scenario destinations - TA.11 Atmospheric descent and landing

Advanced water/surface airbags technology is required for CEV, that is included in-cluded in all the destinations' concepts, except for ISS and MP: the first unit is foreseen in 2018. This technology can be also applied and tested in the ATV-like and LM, for

Applicable/Demo technologies	HSE destinations/concepts							Total
	ISS	CL	MS	MO	NEA	MP	Mars	#
Advanced water/surface airbags	ATV-like	LM	LM	1		1		5
Deployable supersonic decelerators	ATV-like	LM	LM	1		4	1	9
Advanced crushable structures			4	1		3	1	9
Active damping system			4	1		3	1	9

Table 5.30: Applicable Technologies Mapping on HSE scenario destinations - TA.11 Atmospheric descent and landing

the ISS and CL concept respectively, in enhanced versions foreseeing non-destructive re-entry.

Summary

According to what just described, the obtained tables are very useful to visualize when each technology is required the first time, and to identify the possibilities to previously implement and validate it in other destinations. Moreover, they can be a support to decide where it is more urgent and/or convenient to place investments, considering the due dates and the number of missions and elements requiring the technologies.

For example, referring to “TA.1 Structures and Mechanisms”, a new In-space Advanced Docking Mechanism is required in numerous missions and the first possibility to use it is in an ATV-like cargo mission to the ISS in 2014. Advanced Secondary/Tertiary Structures are needed for CL, MS, MO, NEA and Human Mars Mission concept, but they can be implemented and tested in simpler missions to ISS before 2017. Analogous considerations apply to Advanced Rigid Structures, applicable to quite a large number of elements. Concerning separations, Advanced Pyrotechnique Separations are required in a lot of elements, starting with the CEV in 2018. They are also applicable to a large set of units, especially in the Mars Preparation concept.

Similar considerations can be drawn for all the other technological areas and finally an overall ranking of the most required technologies can be derived, with information about time and elements in which each technology is needed (see section 5.1.3).

5.1.3 Technologies roadmaps

As result of the mapping analysis discussed in the previous section, a ranking of the most interesting and critical technologies is obtained. In particular, table 5.31 summarizes the 30 most required technologies, highlighting the number of elements in which each technology is required, the year when it is needed the first time*, the first mission concept in which it is required and the related concept implementing it, according to the HSE reference scenario.

Technology	Technological area	Elements #	Needed time	1st mission concept	1st element
LIDAR	TA.4	37	2014	ISS	ATV-like
In-space advanced docking mechanisms	TA.1	30	2014	ISS	ATV-like
Advanced outside dust mitigation	TA.10	24	2022	MS	Utility cart
Advanced secondary/tertiary structures	TA.1	24	2017	CL	EML1-HAB
Advanced cryo - transfer concept	TA.3	23	2018	CL	Small NTR
HDA algorithms	TA.7	22	2022	MS	1-ton lander
Stereo vision 3D camera	TA.4	18	2018	CL	Small NTR
Advanced shielding materials	TA.10	17	2016	ISS	NTR demo
Advanced pyrotechnique separations	TA.1	17	2018	CL	CEV
Advanced rigid structures	TA.1	17	2018	CL	CEV
Advanced MLI	TA.3	17	2017	CL	EML1-HAB
Advanced PCU	TA.2	16	2022	MS	SolPS
Advanced radiators	TA.3	16	2022	MS	Manipulator
High-efficiency solar cells	TA.2	13	2017	CL	EML1-HAB
Surface mobility algorithms	TA.7	13	2022	MS	Utility cart
Relative guidance algorithms	TA.7	11	2022	MS	1-ton lander
Regenerative fuel cells	TA.2	10	2017	CL	EML1-HAB
Advanced inside dust mitigation	TA.10	10	2023	MS	PR demo
Advanced LBO-ZBO concepts	TA.3	10	2020	MS	SML
NTR fission reactor (NERVA-like)	TA.9	9	2016	ISS	NTR demo
Pressurized inflatable structures	TA.1	8	2015	ISS	Inflat. demo
Pumped-fed LOX/LCH4	TA.9	7	2020	MS	SML
Advanced surface locomotion	TA.4	7	2022	MS	Utility cart
Pressure-fed storable MON/MMH	TA.9	6	2018	CL	CEV-SM
ARES	TA.8	5	2017	CL	EML1-HAB
Regenerative TCC systems	TA.8	5	2017	CL	EML1-HAB
Advanced waste compacting systems	TA.8	5	2017	CL	EML1-HAB
Lyophilization	TA.8	5	2017	CL	EML1-HAB
Food complement unit	TA.8	5	2017	CL	EML1-HAB
Advanced water/surface airbags	TA.10	5	2018	CL	CEV

Table 5.31: Transversal ranking of required technologies

For the 30 most required technologies the roadmaps have been derived, including a survey of the actual TRL, both for Europe and US, and an assessment of the needed date for TRL 5. The results are reported in table 5.32. Please note that the TRL assessment are referred to 2011-2012; the database could be continuously be updated according to the technologies development activities. The assessment of the needed dates for TRL 5 and TRL 8 is done referring to the various missions part of the HSE reference scenario. They can seem quite “ambitious and unrealistic”, and further analysis shall be addressed to the evaluation of missions feasibility and technological development, even considering additional parameters, as for example, political and economical issues.

*The timeframes in which all the technologies are needed derive from all the considerations done for the reference scenario missions and shall be read as “desired dates”.

Technology	Europe TRL (2011)	US TRL	TRL 5	TRL 8
LIDAR	4/5	8	2012/2013	2014
In-space advanced docking mechanisms	4	4	2013	2014
Advanced outside dust mitigation	3	3/5	2018	2022
Advanced secondary/tertiary structures		3/4	2013	2017
Advanced cryo - transfer concept	2/3	4	2014	2018
HDA algorithms	3		2018	2022
Stereo vision 3D camera	5/6		2014	2018
Advanced shielding materials	4?	1/3	2013?	2016
Advanced pyrotechnique separations	3/4	3/4	2014	2018
Advanced rigid structures	2/3	4	2014	2018
Advanced MLI	3	3/4	2013	2017
Advanced PCU	3	3	2018	2022
Advanced radiators	5	2/4	-	2022
High-efficiency solar cells	4-8	3-9	(2013)	2017
Surface mobility algorithms	2		2018	2022
Relative guidance algorithms	3	3/5	2018	2022
Regenerative fuel cells	3/4	4	2013	2017
Advanced inside dust mitigation	3	3/5	2019	2023
Advanced LBO-ZBO concepts	3	3	2016	2020
NTR fission reactor (NERVA-like)	2/3	4	2013	2016
Pressurized inflatable structures	4	5	2013	2015
Pumped-fed LOX/LCH4	3/4	4/5	2016	2020
Advanced surface locomotion	4	3/7	2018	2022
Pressure-fed storable MON/MMH	7/8	7	-	2018
ARES	4	6/7	2013	2017
Regenerative TCC systems	4	3	2013	2017
Advanced waste compacting systems	4	5/6	2013	2017
Lyophilization	4	5/6	2013	2017
Food complement unit	4	5/6	2013	2017
Advanced water/surface airbags	5		-	2018

Table 5.32: Technologies TRL assessment

However this data can be a starting point to understand which are the technologies to be developed with more urgency and which are the fields where more investment shall be placed.

Finally, where the TRL 8 assessment refers to a demo mission, this means that this TRL value is achieved through the mission itself (TRL 7 is required to be launched in a demo mission).

5.1.4 Technological contribution to Mars mission

In this section, the potential contribution of each intermediate destination concept of the reference scenario to NASA DRA 5.0 is briefly discussed. Each intermediate destination can contribute to the achievement of the technological capabilities required for Mars in different percentage considering technologies required or anyway applicable at the specific destination. Table 5.33 summarizes the number and the percentage of Mars required technologies which, in each intermediate destination, are:

- required,
- applicable/demo,
- applicable/demo or required,
- not applicable.

The percentages are evaluated considering that 64 technologies in total are required for Mars, according to the NASA DRA 5.0 concept [3].

Analyzed concept	**Technologies**							
	Required		Applicable/demo		Required or Applicable/demo		Not applicable	
	#	[%]	#	[%]	#	[%]	#	[%]
ISS	7	10.9	24	37.5	28	43.8	36	56.2
Cis-lunar	30	46.9	22	34.3	37	57.8	27	42.2
Moon sortie	38	59.4	48	75	56	87.5	8	12.5
Moon outpost	48	75	24	37.5	53	82.8	11	17.2
NEA	35	54.7	18	28.1	41	64.1	23	35.9
Mars preparation	36	56.3	52	81.3	61	95.3	3	4.7

Table 5.33: Destinations concepts contribution to NASA DRA 5.0

These data are obtained starting from the mapping tables developed for all the technological areas, deriving for each destination the total number of required and applicable technologies, and expressing it as a percentage of the Mars required technologies. Table 5.33 also indicates the percentage of “required or applicable/demo”, that refers to the technologies that can actually be implemented at the specific destination (being either required or applicable/demo).

The graphs reported in figures 5.17, 5.18 and 5.19 graphically summarize the obtained results for the intermediate destinations, showing the percentages of Mars required technologies that are required or applicable in the intermediate concepts. From figure

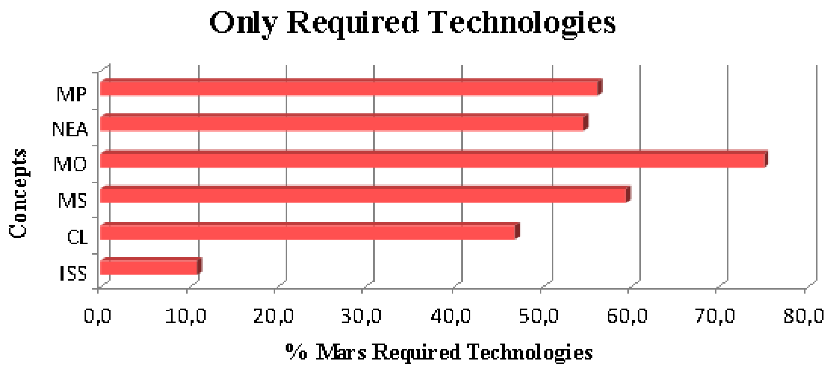


Figure 5.17: Percentage of required technologies to implement in intermediate destinations

5.17 it is evident that Moon Outpost requires 75% of the technologies required for Mars. It is followed by Moon Sortie, Mars Preparation and NEA. As foreseeable the ISS does not require many new technologies, and specifically the resulting 11% refers to the technologies needed for the new modules part of the ISS concepts (and not to the already deployed ISS modules). Considering the applicability/demo of the technologies through the intermediate destinations (graph in figure 5.18), the Mars Preparation concept represents the best test-bed with more than 80% of the Mars required Technologies. The Moon Sortie concept is also a good option to implement technologies needed for Mars (75%).

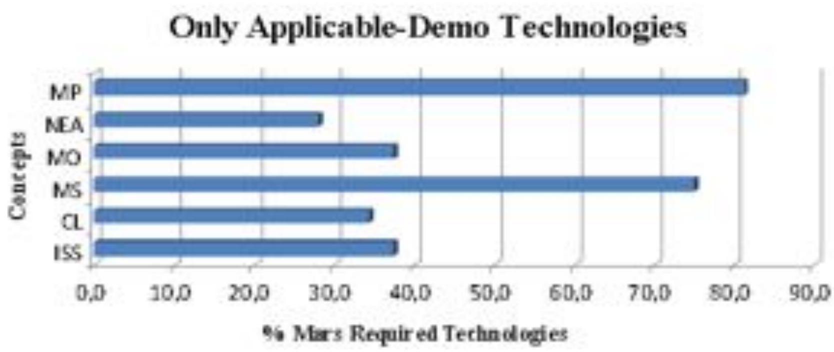


Figure 5.18: Percentage of applicable technologies to implement in intermediate destinations

Finally, the last graph (figure 5.19) provides the resulting percentage of technologies that are required or applicable at the specific destination.

Figure 5.19: Percentage of required or applicable technologies to implement in intermediate destinations

A specific technology can be required for an element while applicable to another element of the same destination concept (this explains why the "Applicable/Demo or Required" value is not given by the sum of the only "Required" and the only "Applicable/Demo" values). For example, considering the cis-lunar concept and the technology "Advanced Deployable Structure", this technology is required in one element, that is the CEV-SM (table 5.13), but is also applicable to the EML1-HAB (table 5.14). In this case, when counting the total number of technologies, it is counted as one in both the "required" and "applicable" categories, but it is counted only once in the "Applicable/Demo or Required" category (and not two as it would be by summing the "required" and "applicable" values). The same types of considerations can be done done for all other technologies.
The last graph (figure 5.19) is the one that best highlights the contribution of each destination to the achievement of the technological capabilities required for Mars. As a matter of fact, it refers to the actual number of technologies which can be validated at the destination, being them either required or applicable.

5.2 Discussion

The obtained results represent a good support for the identification of the most critical technologies to be developed, highlighting also the timeframe in which they are needed. This could be very useful, in order to well place investments in the development of specific systems necessary to allow future space exploration missions.

The complete set of obtained results is helpful to support technologies developments strategic decisions and can answer the questions about the most required/applicable technologies for the whole scenario or for a single destination. Moreover the tool gives information about when a technology shall be ready and in this respect could provide an input to define an adequate development plan.

Just as an example of how to use the tool, consider as target the cis-lunar concept and consider the technology “Advanced Secondary-Tertiary Structures”. This technology is required in three elements of the cis-lunar concept and specifically the first time it is needed is in 2017 in the EML1-HAB (see tables 5.13). However, looking at table 5.14, it appears clear that this technology can be previously implemented and tested at the ISS (in one of the elements foreseen for the ISS concept like the ATV-like module, PMM-like or inflatable demo). This type of consideration can be done for all the technologies needed for the cis-lunar concept, thus allowing the definition of an opportune roadmap for those technologies, in terms of their development and implementation in “easier” missions to validate them prior to the cis-lunar missions. Starting from these results, further analyses could be devoted to the evaluation of interdependencies between technology development activities.

The graphs discussed in section 5.1.4 can be exploited to take strategic decisions in support of future human space exploration, also in terms of target destinations selection. Indeed, looking at the technologies implementable in the various intermediate destinations, it is possible to have indications about which are the most interesting destinations for future deep space exploration, and in particular from a technological development point of view, having as final objective a human mission to Mars. For example, the lunar concepts (Moon sortie and Moon outpost) are better test-beds than NEA for what concerns the Mars required technologies. Moreover, as conceivable, the ISS concept does not require many Mars required technologies, but a large percentage of them (37.5%) is applicable there. In total, more than 43% of the technologies required for Mars are implementable (required or applicable) at the ISS where they can be tested and validated, without the need of new infrastructure or other location in space. On the basis of this result a very important conclusion can be drawn, in terms of strategic decisions: the operative life of ISS shall be extended as much as possible, in order to fully exploit its potential capabilities in the framework of future human space exploration. Furthermore, the analyses results show that the Cis-lunar concept can be a significant alternative to the NEA exploration, in terms of demonstration of Mars required technologies, even if they are not actually equivalent. However an expedition to an NEA is still a very interesting mission, since it gives the opportunity to perform a Mars-analogue mission, at least for what concerns the deep-space travel, with limited complexity. This will be very important especially for psychological issues and astronauts training.

5.3 Mission opportunities for technologies in-orbit validation

As largely discussed in the previous sections, in order to proceed in the expansion through the solar system, traveling beyond LEO, moving towards more and more challenging missions, and finally accomplish a human expedition to Mars, several technological limitations have to be overcome, through the development, test and validation of innovative technologies and advanced systems, before implementing them in real manned far missions. The philosophy behind the overall study is in line with this necessity, as demonstrated by the stepwise approach provided through increasing complexity missions, as well as by some dedicated demo missions to ISS foreseen at the beginning of the HSE reference scenario path.
Obviously, further and deeper analyses shall be performed in order to identify and define in detail opportune reference mission scenarios for the in-orbit demonstration and validation of advanced technologies. According to this, the methodology described by the work flow shown in figure 5.20 has been developed to identify a mission scenario for the verification and validation of selected key technologies (flight demonstration mission).

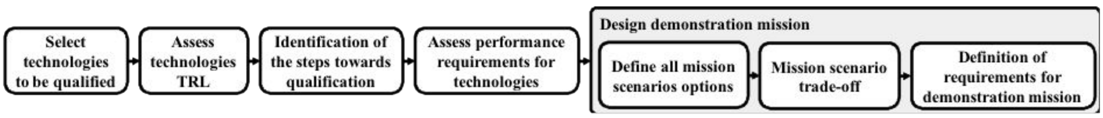


Figure 5.20: In orbit demonstration missions design methodology

In general, space technologies shall have a sufficient maturity level to be utilized during exploration missions. The European Space Agency measures the maturity level through the Technology Readiness Level. The definition of TRL is *"a set of management metrics that enable the assessment of the maturity of a particular technology and the consistent comparison of maturity between different types of technology - all in the context of a specific system, application and operational environment"* [20]. Figure 5.21 provides a high-level illustration of the TRL scale, using the well known "thermometer diagram" as a metaphor for increasing technology maturity, in the context of the progression from basic research to system operations.

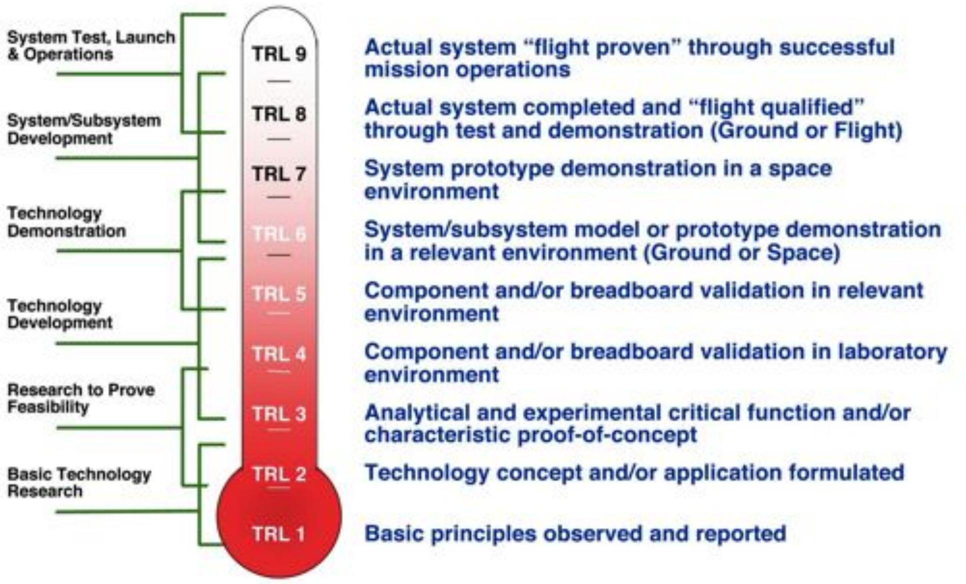


Figure 5.21: Technology Readiness Levels - Thermometer Diagram

Generally, a technology can be utilized on space systems if it is "flight qualified" (equivalent TRL 8). Tests performed both on ground and in space increase the technology TRL up to level 8. The last level of the scale (equivalent TRL is 9 for flight proven system) is obtained through successful mission operations.

The methodology adopted for the assessment of reference mission scenario for in-orbit demonstration starts with the identification of the technologies to be qualified and the assessment of the associated TRL. The obtained information is useful to plan the set of analysis, experiments and tests to be performed on breadboards and prototypes that allow reaching the desired TRL. Generally, TRL 7 and 8 require demonstration in the space environment: TRL 7 is reached through demonstration of a prototype system, TRL 8 is reached through demonstration of the actual system. The qualification

missions aim to demonstrate that the technologies meet the performance requirements. These requirements are dictated by preliminary studies and design phases performed by the developers. At this point, the demonstration mission design activity starts. Several mission scenario options are conceived and analyzed to assess the most cost-effective one. Qualitative and quantitative trade-offs are performed considering Figures of Merit (FoM) such as mass of the systems, costs, system complexity, mission risk and secondary functions (e.g. additional research capabilities). As result of the trade-off, the most cost-effective option is chosen. Finally, the detailed description of the demonstration mission scenario is provided through definition of the functional, mission, interface, environmental, physics, operations, configuration, design requirements. The developed methodology is particularly suitable for the conceptual design phases, but it can be also applied to more detailed design phases increasing the analysis detail level. Thus, the first analyses are performed to discard “bad options” with low time computational effort, whereas the successive and more detailed analyses allow increasing the results confidence level.
In the frame of STEPS2 project, some technologies are being studied and developed. For these technologies, some considerations about opportunities for on-orbit validation are done*. In particular, a set of possible missions is identified where the technologies could be implemented in order to achieve TRL 8 and get ready to be implemented in future exploration missions.

Figure 5.22: Roadmaps for STEPS2 technologies in-orbit demonstration

*The results here presented have been obtained in the framework of STEPS2 (Systems and Technologies for ExPloration of Space - Phase 2) which is a research project co-financed by Piedmont Region (Italy), firms and universities of the Piedmont Aerospace District started in 2013.

As final result, the roadmap reported in figure 5.22 is obtained which summarizes the opportunities for in flight demonstration of the technologies under study in the frame of STEPS2.

6

HSE Elements/Building Blocks

In this chapter the conceptual design of two space elements are presented: a Deep Space Habitat and a Space Tug for Earth's satellite servicing.
According to the results of the analyses described in previous chapters, these two elements are fundamental building blocks for exploration. The space tug analyzed and described in section 6.2 is designed with the main objective of supporting satellites servicing. However, an evolution of this vehicle can be exploited to support large spacecraft assembly, as required according to the reference human space exploration scenario.
In the following, the main trade-offs and analyses, which lead to the conceptual design of DSH and space tug are discussed.

6.1 Deep Space Habitat

In accordance with the defined reference HSE scenario, the Deep Space Habitat is one of the most significant elements, needed to enable future space exploration missions, which look at beyond-LEO destinations. The experience gained through the ISS could be exploited to develop a module able to support different human missions towards deep space targets. The module shall have some specific characteristics deriving from the peculiarities of the mission operations and of the environment it has to withstand that strongly influence the design of the pressurized habitat where the astronauts have to live for quite long periods. According to the necessity of a habitation module to enable travels beyond LEO, a preliminary analysis of a possible architecture for the deep space habitat has been carried out.

6.1.1 Rationale and assumptions

The deep space habitat is conceived as a cis-lunar orbital infrastructure and a space-ship for deep space exploration missions. It would represent the first outpost beyond LEO, supporting the human presence in outer space for extended stays. It will represent a platform for scientific research and technology development for space exploration as well as a support for crew transportation architecture to Moon's surface and further destinations, as for instance Near Earth Asteroids. Furthermore it will give the opportunity to increase the science return from lunar robotic surface exploration. In particular, relating to this latter point and concerning remotely controlled surface robotics, the exploration activities are assumed to be concentrated on the near side of the Moon.
The DSH is envisioned as a human-tended facility, and visits of crew of four astronauts are periodically foreseen (every six months), for a maximum permanence duration of two weeks. It is axially attached to a service module, not considered as part of the DSH system and whose features are not analyzed in details, which is in charge of providing attitude and orbital control. Additionally, the DSH is meant to demonstrate a set of critical technologies and associated operations required to perform a human exploration mission to a NEA. In particular, it is designed to enable a full asteroid mission rehearsal in a relevant environment (i.e. outside Val Allen belt). The considered NEA reference mission foresees a crew of four astronauts and has an overall duration of about 12 months, including about ten days to be spent in the proximity of the asteroid, where a certain number of EVAs are to be performed. In particular, along the entire mission seven nominal EVAs are foreseen for the NEA operations, and two contingency EVAs are considered for external maintenance. The overall reference NEA mission spacecraft (see section 4.2.3.5 for the reference architecture and concept of operations) is composed of:

- two transfer stages, utilizing nuclear thermal propulsion, in charge of providing the ΔV needed to insert the spacecraft into the NEA transfer orbit, to brake around the asteroid and for the trans-Earth injection;

- the long duration habitat to host the crew;
- the capsule for the Earth re-entry;
- the service module for the NEA proximity operations.

The spacecraft is envisioned to be assembled in low Earth orbit, where the different parts are brought by means of two heavy lift launchers and a crew vehicle for transportation of the crew.

6.1.2 Major requirements

The requirements assessment process is carried out according to the main objectives identified for the module and described in previous section. Some of the mission and system requirements, which drive the concept selection and preliminary design of the module, can be summarized in these listed hereafter.
The DSH shall provide:

- habitable volume for a crew of four astronauts for up to 12 months;
- controlled internal environment and adequate conditions for the crew activities;
- protection against external environment (a radiation shelter to protect four astronauts against SPE shall be envisaged);
- communications with ground, guaranteeing high data rate transmission;
- at least three docking ports, to allow connections with visiting vehicles;
- autonomous operation capability, being monitored and controlled from ground while un-crewed (experiments' remote control and monitoring from ground);
- tele-operation capability of robotic systems deployed on the surface of the exploration target (Moon, NEO, ...).
- interface with robotic sample return probe and sample analysis capability;
- crew EVA capability.

6.1.3 Major trade-offs

Several trade-offs are carried out in order to define the DSH architecture by comparing alternative options on the basis of a set of figures of merit. Each figure of merit is given a specific weight according to the relative importance it has with respect to the others.

The evaluation of the different options is qualitatively carried out by assigning a score of 1, 0 or -1 for each figure of merit, depending if the considered option is adequate, neutral or inadequate,. For the evaluation of the trade-offs, the lunar robotics are supposed to be deployed on the Earth side of the Moon and Lunar TLC System is considered not available (not yet deployed).
The main trades identified at system level are about:

- deployment strategy,
- deployment location,
- system architecture,
- radiation shielding approach.

Furthermore, additional trades are performed regarding:

- ECLSS closure level,
- EVA capability.

In the following sections the various trade-offs are discussed, describing the alternative options and highlighting the obtained results.

6.1.3.1 Deployment strategy

The first trade-off is performed to select the most suitable strategy of deployment to accomplish the module's mission objectives. The DSH is conceived as a testing platform for new technologies to be used in further exploration missions (e.g. to asteroids, Mars), as well as to allow long duration human mission rehearsal. A stepwise approach is foreseen to demonstrate capabilities for supporting long duration missions in deep space environment and, in this respect, the system shall be upgradeable on-orbit for supporting increasing duration missions or hosting new technologies demonstrators. Three different options are identified and traded:

- one module to be partially re-used as NEA exploration vehicle, after having been upgraded on-orbit;
- one module to be fully re-used as NEA exploration vehicle;
- two different units: the first unit envisioned as a cis-lunar station for the test of technologies, and the second unit conceived for the NEA mission; in this case, a

	Pros	Cons
Partial re-use	Lower development/manufacturing cost, Just 1+ launch	Lower optimization, More risks (referring to NEO mission) due to longer life, No station in cis-lunar
Full re-use	Much lower development/manufacturing cost, Just 1 launch	- optimization, ++ risks (referring to NEO mission) due to longer life, No station in cis-lunar
2 units	Higher optimization, Lesson learned and thus reduced risks (referring to NEO mission) Permanent cis-lunar station	Higher development/manufacturing cost, , 2 launches, thus higher risks

Table 6.1: Deployment strategy trade-off

common core shall be foreseen to make the tests representative and reduce the delta development.

The three options are compared in order to identify the major advantages and disadvantages of each one (see table 6.1) and finally the solution envisaging two units is selected as the most convenient. Indeed, the first two options imply a longer lifetime and therefore higher risks, and a less optimized design (e.g. solar arrays sized for "wrong end of life").
Furthermore, supporting lunar exploration and testing critical technologies would require different capabilities with respect to those required for deep space missions. Finally, developing two units would allow having a permanent cis-lunar station, even during and after the NEA mission.

6.1.3.2 Deployment location

Three possible locations for the cis-lunar infrastructure deployment are traded: the Earth-Moon Lagrangian points 1 and 2 and a Low Lunar Orbit (LLO). The option of a low Earth orbit is neglected since the beginning because of the infrastructure's main objectives. Indeed, this space habitat is conceived to support human mission beyond LEO for extended stays, being a technology and research platform for exploration and a support for increasing science return from lunar robotic surface exploration, and therefore a low Earth orbit would not be suitable.
The trade-off is performed considering the following figures of merit:

- accessibility to and from Earth,
- telecommunications capability with the Earth,

- lunar tele-operations capability (robotics are assumed to be on the near side of the Moon, in the South Pole zone),
- station-keeping requirement,
- accessibility to and from Moon's surface,
- deep space accessibility (for the reference NEA mission the spacecraft assembly is performed in LEO),
- sun availability,
- psychological effects (since the habitat shall allow also deep space mission rehearsal, being further away and not seeing the Earth is considered better than the opposite situation - being more challenging for the crew),
- space environment hazard,
- public outreach.

A list of the main advantages and disadvantages of having the cis-lunar infrastructure in the three different locations is reported in table 6.2. Cells are colored in green if the related location is clearly advantageous vis-à-vis the corresponding figure of merit, while red boxes are used if the location appears to be evidently disadvantageous.
Taking into account these features the comparison among the three possibilities is performed, according to what shown in table 6.3. The various figures of merits are given specific weights according to their relative importance.
EML1 results the best option, mainly thanks to its superior capability to support tele-operations of lunar surface robotics, almost constant sun availability (for power generation) and direct TLC visibility with ground segment.

6.1.3.3 System architecture

To preliminary define the system architecture for the cis-lunar habitat, the alternative configurations hereafter described are identified and traded:

- Single element, which can be
  - Rigid, or
  - Inflatable
- Assembly of more elements, that can be be given by the combination of
  - a rigid node plus a rigid habitation module, or

	EML1	EML2	LLO
Accessibility to/from Earth	once-per-day opportunity from any launch site, any time and any landing site for Earth re-entry, ΔV from LEO: 3.8 km/s, lower cost wrt LLO, higher cost wrt EML2	any time and any landing site for Earth re-entry, ΔV required LEO: 3.4 km/s, lower cost wrt LLO, lower cost wrt EML1	once-per-day opportunity from any launch site, orbit constraints on the Earth re-entry, ΔV from LEO: 4km/s, higher cost wrt EML1 and EML2
TLC with Earth	Continuous direct communications	Depending on halo orbit, generally not continuous direct comms (need for satellite)	Not continuous direct comms (need for satellite)
Lunar tele-operations capability	Round-trip delay of ~2.2s to ground Round-trip delay of ~0.2s to Moon's surface Constant visibility of the half near side	Round-trip delay of ~3s to ground Round-trip delay of ~0.2s to Moon's surface Constant visibility of the half far side	Round-trip delay of ~2.6s to ground No round-trip delay to Moon's surface Very close but short visibility windows
Station keeping	Low propellant consumption (average ΔV per year: 40 m/s)	Low propellant consumption (average ΔV per year: 40 m/s)	Average propellant consumption (ΔV per year: 1600 m/s in 100km circular polar LLO, ~0m/s in 45kmx203km polar frozen orbit)
Accessibility to/from Moon's surface	Global access of the moon (anytime and any landing site) lunar access cost ΔV~2.35km/s 1.5 days transfer to LLO	Global access of the moon (anytime and any landing site) lunar access cost ΔV~2.35km/s 1.5 days transfer to LLO	Orbit constraints on the landing site lunar access cost ΔV~2km/s few hours transfer
Deep space accessibility	Very low deep space access cost	Very low deep space access cost	Average deep space access cost
Sun availability	Long and rare shadow periods (Earth and Moon induced)	long and rare shadow periods (Earth and Moon induced)	Short and frequent shadow periods (Moon induced) Long and rare shadow periods (Earth induced)
Psychological effects	Constant view of Earth and Moon	Depending on halo orbit, generally constant view of Moon far side	Periodic passages in dark zones
Space environment hazard	High radiation Very limited space debris and meteoroids	High radiation Very limited space debris and meteoroids	High radiation Limited space debris and meteoroids
Public outreach	Far from Moon Difficult to understand	Far from Moon Difficult to understand	Close to Moon Easy to understand

Table 6.2: Deployment Locations Comparison

	Access. to/from Earth	TLC with Earth	Lunar tele-ops capability	Station keeping	Access. to/from Moon's surface	Deep space access.	Sun availability	Psycho effects	Space env. hazard	Public outreach	
Weight [%]	*15*	*10*	*15*	*5*	*15*	*5*	*15*	*5*	*10*	*5*	
EML1	1	1	1	1	-1	1	1	-1	0	-1	40
EML2	1	-1	-1	1	-1	1	1	1	0	-1	0
LLO	0	-1	0	0	1	0	-1	0	0	1	-5

Table 6.3: Deployment locations trade-off

– a rigid node plus an inflatable habitation module.

Among these alternatives, the option having a single inflatable element is discarded, since it does not match the requirements. As a matter of fact, one of the requirements states that at least three docking ports shall be available in order to allow the docking with at least three simultaneous visiting vehicles and a single inflatable module cannot

provide them.
The trade-off is performed considering the following figures of merit:

- development complexity,
- volume over mass ratio,
- flexibility/growth capability,
- operational complexity, mainly linked to the internal outfitting.

The results of the comparison among the three alternative options are shown in table 6.4, where the weight assigned to each figure of merit is shown as well.

	Development complexity	Volume/mass ratio	Flexibility / growth	Operational complexity	
Weight [%]	*20*	*30*	*30*	*20*	
Single rigid	1	-1	-1	1	-20
Rigid + rigid	0	-1	0	1	-10
Rigid + inflatable	-1	1	1	-1	20

Table 6.4: System architecture trade-off

Finally, the configuration with a rigid node attached to an inflatable habitat is selected, as it provides the best optimization of the volume over mass ratio, which is very important especially in view of future longer missions. In this regard, the selected configuration provides a meaningful opportunity for the validation of the inflatable technology, which is very attractive in order to improve the available volumes in-orbit while limiting the launch requirements, especially for long and far missions. Moreover this configuration has quite a good flexibility allowing for later docking of additional modules, such as logistics storage modules, laboratories for scientific research or a module for tourism. The module is envisaged as a modular assembly and reconfigurable in space, which is considered a preferable approach with respect to an integrated on ground configuration.

6.1.3.4 Radiation shielding

Once outside the protection of the Van Allen Belts the astronauts are constantly exposed to galactic cosmic rays (GCR), which deliver to human body a steady dose. The intensity of the GCR flux varies over the 11-year solar cycle and the maximum dose received occurs at solar minimum. In addition to the GCR, for long duration mission it must be considered the case in which a solar flare takes place. Large Solar Particle

Events (SPE) are relatively rare, usually one or two events per solar cycle, but they could be very dangerous if the spacecraft is inadequately shielded, since they deliver a very high dose in a short period of time.
The long exposure to space radiations is one of the most critical issues to be taken into account for missions beyond LEO, outside the protective shield provided by the Van Allen belts. For this reason, a specific analysis has been performed in order to identify the best approach to be adopted for protecting the crew against radiations. First of all, a high level trade between an active and a passive methodology is carried out. To perform this trade the following figures of merit are accounted for:

- shielding complexity,
- safety / reliability,
- impact on the other subsystems (e.g. interference that an active system could have with other S/Ss),
- mass.

The comparison between active and passive shielding is shown in table 6.5, where the weights assigned to the above mentioned figures of merits are reported as well.

	Architecture complexity	Safety / reliability	Impact on other S/Ss	Mass	
Weight [%]	*20*	*30*	*20*	*30*	
Active	-1	-1	-1	1	-40
Passive	1	1	1	-1	40

Table 6.5: Active vs passive radiation shielding trade-off

The passive shielding turns out to be the most convenient. Furthermore, present TRL of active technologies is very low.
For the reference mission of one year to a NEA, the protection provided by structure and racks/equipment is preliminarily evaluated sufficient as protection against GCR to remain below the maximum acceptable dose. An equivalent area density of 15g/m^2 of Aluminum is assumed, which corresponds to 20 cSv/year for GCR at solar maximum and 40cSv/y for GCR at solar minimum [26]. The inflatable part is assumed to exhibit the same shielding capability as the rigid one. This means that the total dose is within the allowable limits imposed by NASA (50cSv/y) [27]. On the contrary, a dedicated shelter is mandatory as protection against SPE, since without it the dose in case of a SPE occurrence would amount to 30-40 cSv, consequently exceeding the allowed annual

limit. For more challenging missions (e.g. towards Mars), additional shielding shall be foreseen and/or the option to switch to an active solution shall be considered.
At this point it is necessary to establish which is the most suitable material to be used for shielding the spacecraft. Materials having high hydrogen content are considered because they are the most effective for high-energy charged particle shielding per-unit-mass. Among those, the most interesting ones are:

- liquid hydrogen,
- water,
- HDPE (High Density Polyethylene).

Liquid hydrogen would be the best shielding solution but it is discarded since it is difficult to manage (very low temperature cryogenic liquid). Therefore, the trade-off is actually performed to choose between water and HDPE. Due to the closure level of ECLSS envisaged for the module (as will be addressed in the following), the amount of water on board is minimal, thus additional quantity of water should be carried exclusively for this purpose.
The trade-off between water and polyethylene is performed considering as figures of merit:

- mass,
- system complexity,
- versatility of the system.

The results are shown in table 6.6, which reports the weights assigned to the various figures of merit as well. In order to provide the same protection against SPE, a mass

	Mass	System complexity	Versatility	
Weight [%]	*40*	*30*	*30*	
Water	-1	-1	1	-40
HDPE	1	1	-1	40

Table 6.6: Water vs Polyethylene trade-off

penalty of about 300 kg is estimated for water, given the same shielded volume. Hence, polyethylene is finally chosen as shielding material.
The preliminary sizing of the shielding system is done to be compatible with a maximum allowable radiation dose. The requirements impose that the total radiation dose over

one year mission shall not exceed 50 cSv. Considering a condition of solar maximum (which is the case when SPE can occur), the equivalent dose due to GCR amounts to about 20cSv/y; this means that the shelter shall guarantee an additional dose in case of SPE occurrence not exceeding 30cSv. In order to be compatible with this limit, an additional HDPE area density of 15g/cm^2 is needed [28].* Considering a polyethylene density equal to $\rho= 0.96\text{g/cm}^3$, a shielding thickness of 16cm is needed to guarantee the desired protection.

Assuming to have a parallelepiped shelter, with length 3m and a volume of at least 5m^3, the overall mass amounts to about 3.1t.

Analogous considerations can be done for water; in this case the required area density would be 16.5g/cm^2, which corresponds to a total mass of about 3.4t (300kg more the HDPE).

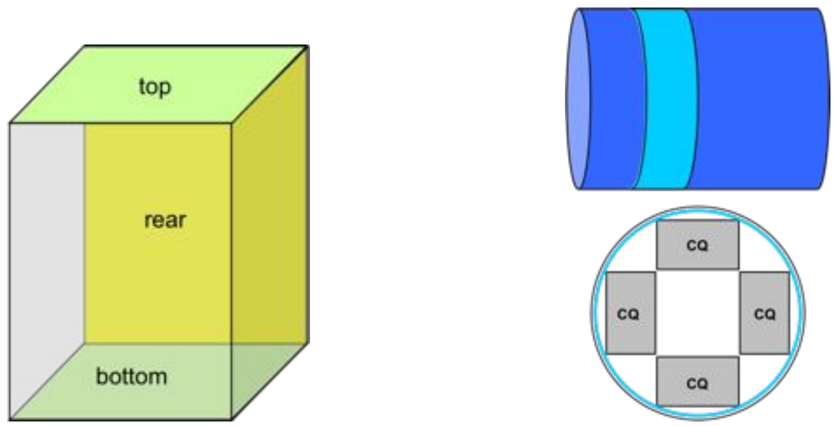
For completeness, the possibility to exploit as additional shielding contribution the water available on board (even if it is only about 170 liters) is analyzed. In particular, two alternatives are examined:

1. storing water in rear and top/bottom crew quarters walls (figure 6.1(a)),
2. storing water in the walls externally with respect to the racks (figure 6.1(b)).

For the first configuration the thickness of the water layer that could be achieved is about 0.7cm. This value is obtained considering crew quarter's dimensions of 1.5m x 2m x 1.2m. With such quantity of water the advantage in terms of radiation dose decrease is negligible ($\sim$ - 0.1cSv/year). Therefore, the increase in the system complexity would not be justified.

For the second configuration, the length of the part of the module that would be covered with water is evaluated, considering different water layer thickness and a module diameter of 4.5m. For each configuration the decrease of radiation dose is evaluated. The results are summarized in table 6.7.

*Actually, with HDPE area density of 15g/cm^2 the total amount of radiation dose in case of SPE would be lower than 30cSv, because of the additional protection provided by structure and racks. This means that a lower area density, and therefore a lighter system, could be considered However, in order to be more conservative, it has been assumed that only the shelter is in charge of protecting astronauts in case of SPE, also to account for the possibility of having part of the shelter placed not in correspondence of the racks.



(a) Crew quarters (b) Racks external

Figure 6.1: Water Storage Possible Configurations

Water layer thickness [cm]	**Length of covered module area [m]**	**∆dose [cSv/y]**
1	1.2	∼ - 0.2
2	0.6	∼ - 0.3
5	0.2	∼ - 0.9

Table 6.7: Active vs passive radiation shielding trade-off

Even with this second configuration, the gain in terms of dose is too low to justify the increase in system complexity for placing the water in the wall.

6.1.3.5 ECLSS closure level

Due to the long duration of the reference NEA mission, the possibility to have regenerative system must be considered. Different levels of closure of the ECLSS can be selected; specifically, the compared options are:

- completely open loop,
- water regeneration,
- air and water regeneration.

The different options are compared with each other, considering as figures of merit for the trade-off the following parameters:

- equivalent mass: this parameter includes mass of resources, mass of spares required for ensuring 2 failures tolerance and an equivalent mass due to the impact on power and thermal control S/Ss;
- maintenance: both operations to be performed and required hardware are considered;
- applicability to deep space missions.

Figure 6.2 reports a graph of the mass as function of the mission duration for different configurations. It appears clear that as the duration of the mission increases, the advantage of a closed loop system in terms of mass reduction becomes more and more significant.

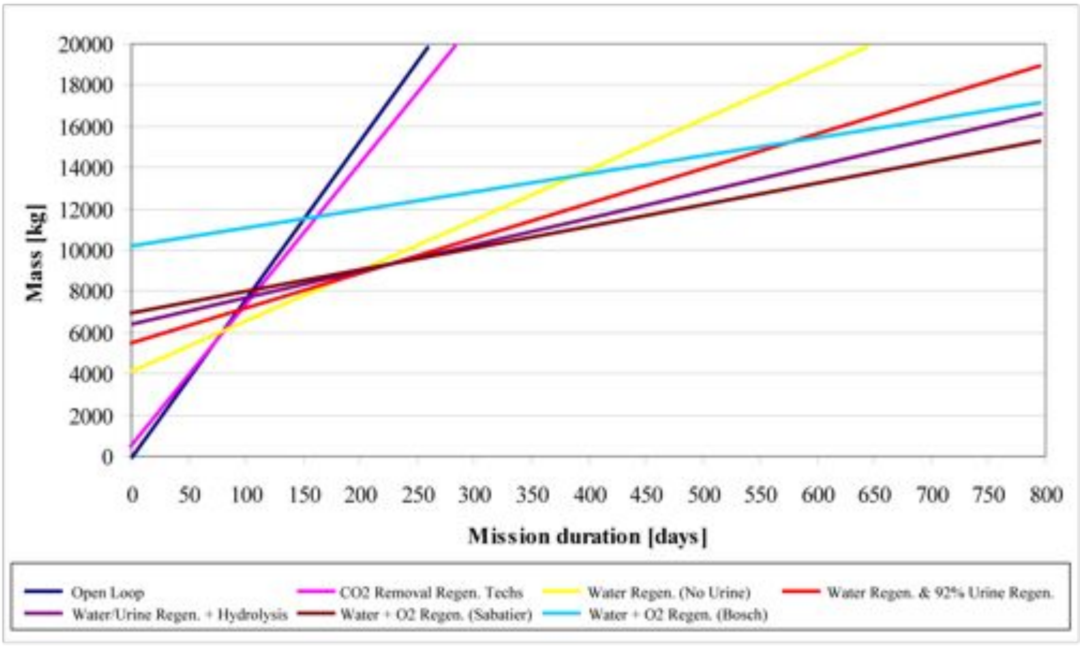


Figure 6.2: Open vs closed loop ECLSS

Table 6.8 shows the comparison among the different configurations, considering the above mentioned figures of merit, and highlights that the best solution is to adopt an air and water regeneration system, especially due to the saving in mass.

	Mass	Maintenance	Applicability to deep space	
Weight [%]	*70*	*10*	*20*	
Open loop	-1	1	-1	-80
Water reg.	1	0	-1	50
Air & water reg.	1	-1	1	80

Table 6.8: ECLSS closure level trade-off

6.1.3.6 EVA capability

For what concerns the Extra Vehicular Activities capability of the system, the trade to be performed is whether to introduce an airlock or not. Depending on the mission, different EVAs are to be performed, e.g. for maintenance, for exploration, for managing external payloads. For the reference mission to NEA lasting one year, the following EVAs are envisaged:

- seven nominal EVA, for NEA proximity operations,
- two contingency EVA, for external maintenance.

Hence, a dedicated airlock is required for this mission. Moreover, additional EVA support items are to be envisaged (e.g. Enhanced-Manned Maneuvering units, EVA tools, etc.). In case the nominal EVA are scrapped in favor of a different approach to proximity operations (e.g. dedicated proximity exploration vehicle), EVA through controlled depressurization could be a possible option (even though more risky). However, for a long term EML1 station, the presence of an airlock is the only viable approach to perform EVAs. This last point is the most important reason why introducing an airlock is finally selected as the best option.

6.1.4 Conceptual definition

In this section the DSH overall architecture is described, as obtained from all the trade-off previously discussed. The major results in terms of mass and dimensions of the module are summarized.

Two units are foreseen: the first one is deployed in EML1 while the second one is in charge of accomplishing a deep space mission to a NEA. A common core characterizes the two units, and only minor modifications are envisaged for the second unit with respect to the first one, mainly due to the peculiarities of the missions they have to

accomplish (this is perfectly in line with the philosophy of pursuing a gradual stepwise exploration approach).
A schematic overview of the resulting architecture of the first unit is shown in figure 6.3.

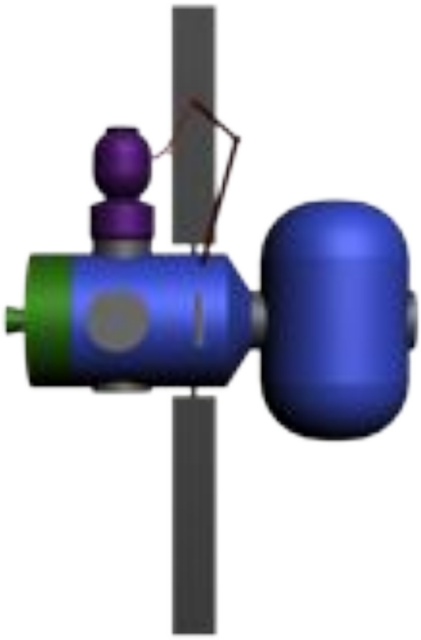


Figure 6.3: Habitat architecture: nominal configuration

It is composed of a rigid node attached to an inflatable module. The implementation of the inflatable technology is foreseen since, in view of very long missions, the comfort of the crew becomes a more and more significant design parameter, not only from a physical but also from a psychological point of view. The presence of the rigid node with its four radial ports ensures the possibility to have up to three visiting vehicles simultaneously attached and to eventually expand the module. The fourth radial ports is used to attach the airlock, which is introduced because different EVAs are to be performed (for external maintenance, for exploration, for managing external payloads). Moreover, additional EVA support items are envisaged, such as Enhanced-Manned Maneuvering Units (E-MMU), EVA tools, etc. The airlock is composed of a rigid equipment lock and an inflatable crew lock. E-MMUs and EVA tools are stored in in dedicated compartments on the external surface of the equipment lock.

The rigid node is axially attached to a propulsion module (depicted in green in figure 6.3), which is mainly in charge of providing orbit/attitude control.
For protecting the crew against radiation a passive shielding is envisaged. In particular the protection provided by structure and equipments is sufficient for protecting against GCR, while a dedicated high density polyethylene shelter is envisaged to protect the crew against SPE. These evaluations refer to a NEA reference mission lasting one year (the EML1 station shall allow a rehearsal of such a mission).
A robotic arm is introduced in the architecture to reconfigure the module from the launch configuration to the operational one and to support external maintenance activities.
According to the graph reported in figure 6.4 [29], a free volume of at least 20m^3 per crew member shall be guaranteed, for both physical and psychological reasons. The

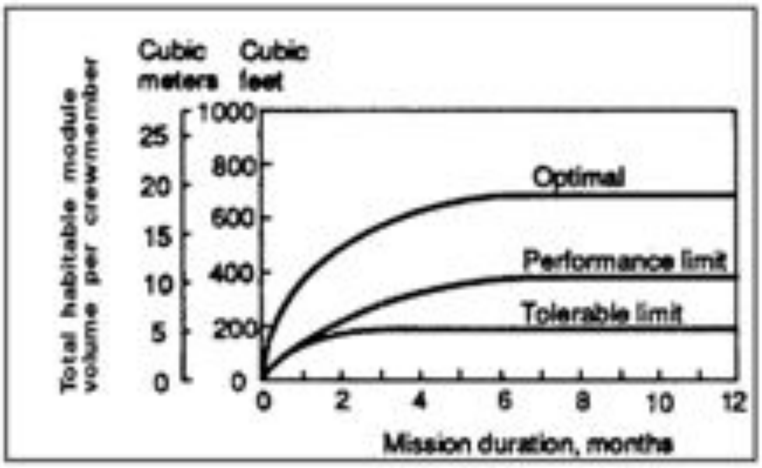


Figure 6.4: Total habitable module volume per crew member

habitat is therefore sized to ensure a total crew habitable volume of at least 80m^3 and an overall pressurized volume of ~240m^3. The main features of the pressurized elements can be synthesized as follows:

- the inflatable habitat, characterized by a rigid core and multi-layer wall, has a total pressurized volume of ~155m^3, with an external size of ~8m x 5m, when inflated;
- the rigid node is sized to guarantee a pressurized volume of ~84m^3, with an external size of ~4.5m x 5m;

- the airlock is characterized by an equipment lock of ∼2m x 1m and an inflatable crew lock of ∼2m x 2m, providing a volume of ∼10m^3.

The overall mass of the deep space module amounts to almost 26 tons, including resources and crew systems sized for 20 days of maximum stay of a crew of four astronauts. The module is deployed with this amount of resources (as per requirements) and its resupply is foreseen with the periodic crew visits.
The mass budget of the habitat deployed in EML1 is shown in table 6.9.

Subsystem	**Mass[t]**
Structure	∼4.5
Thermal control	∼1.0
Mechanisms	∼2.6
Radiation shielding	∼3.4
Communications	∼0.1
Data handling	∼0.4
Crew systems	∼3.7
Airlock	∼1.9
Power	∼0.4
Harness	∼0.2
Instruments	∼1.0
ECLSS	∼2.3
Sub-total	∼21.5
System margin	**20%**
TOTAL	**∼26**

Table 6.9: Cis-lunar habitat mass breakdown

The power subsystem is constituted of solar arrays (two flexible wings, with high efficiency triple junction cells) and Li-ions batteries. The solar arrays are sized to satisfy the requirement of 15-16 kW (total area of about 90m^{2*}).
The thermal control subsystem is sized to guarantee that all the equipment operate within the allowable temperature range along the entire mission. In particular, it comprises a passive thermal control system, characterized by Multi Layer Insulation (MLI) and heaters, and an active thermal control system, using water on the internal loop and ammonia for the external one. Deployable thermal radiators (two wings) are envisaged, capable of rejecting up to 8 kW each (to manage crew metabolic heat as well as on-board equipment waste heat).

*This value, which includes also redundancy on the arrays panels, is obtained relying on the EPS sizing process described in [30] and analogously to what done in section 6.2.4.4.

6.1.4.1 Deployment concept

The first unit is deployed in EML1 and represents the testing platform for new technologies to be used in further exploration missions (e.g. to NEO, Mars), as well as support for the exploration of the Moon. In particular the main tasks it shall accomplish are:

- remote control of surface robotics by on-board astronauts (demonstration towards future exploration, actual lunar surface robotic assets);
- sample acquisition and on-board analysis (demonstration towards future exploration, actual lunar samples);
- safe haven for crew performing lunar missions;
- science/technology research (e.g. crew operations and human psycho-physiology in deep space, long term autonomous system, etc.);
- servicing of transportation system elements (e.g. maintenance/refueling and testing of landers);
- staging post for the crew of lunar ascent/descent vehicles.

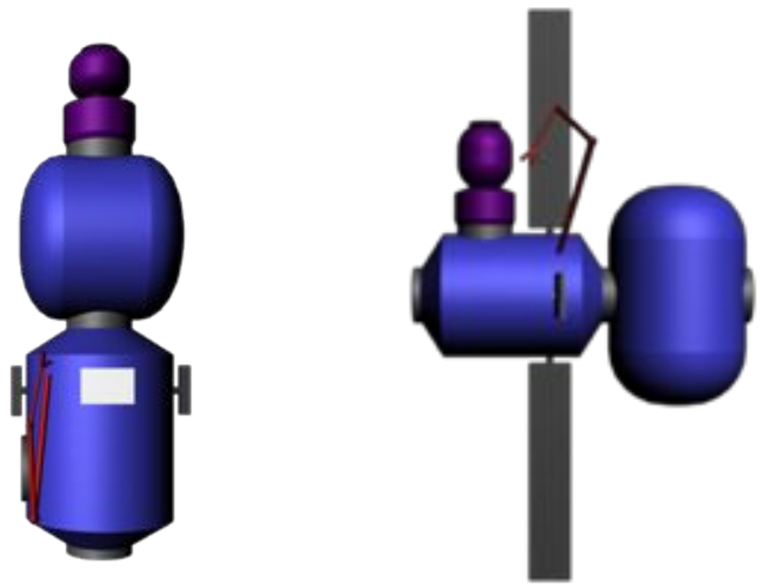
Hereafter, a brief description of the cis-lunar infrastructure deployment mission profile is reported*. The transfer stage is launched through an SLS_70 launcher, while the habitat is launched to LEO by means of a Falcon 9 heavy launcher, together with the service module. The transfer stage, which exploits chemical propulsion, is in charge of injecting the spacecraft in the transfer trajectory towards EML1. The braking maneuver to put the spacecraft in EML1 halo orbit is accomplished by the service module, which is also in charge of station-keeping ($\sim$40m/s per year).
In the launch configuration the inflatable elements are deflated and the airlock is mounted on top of the module, the solar panels and the radiators are in stowed configuration, as well as the robotic arm, as depicted i figure 6.5. The external appendices are deployed before the injection of the spacecraft into the transfer trajectory; the airlock relocation and the deployment of the inflatable habitat are performed in LEO as well, in order to allow easier recovery actions in case of issues related to these potentially critical operations.

*Refer to section 4.2.3.2 for additional details (figure 4.19).

Figure 6.5: Habitat architecture: launch configuration

6.1.4.2 Habitat for deep space missions

Introducing the deep space habitat in different mission architectures, it becomes the habitat where the astronauts have to live during the entire mission to an asteroid or to Mars. In this respect, the module described up to now can be seen as a first unit to be utilized as a precursor for the habitation module to be actually adopted for hosting the crew during the deep space mission. The second unit will exploit the experience gained through its precursor, having a common core with it and implementing those technologies previously tested on the first unit. Only minor changes shall be envisioned due to the peculiarities of the mission for which it is used, as for example the overall lifetime. In a NEA mission the DSH will be part of more complex transportation architecture and will likely have different interfaces with the propulsive module to which it is attached. In addition the three free radial docking ports will not be necessary, while an additional axial docking port would be necessary for safety and operational complexity reason. Finally, the DSH used for the NEA mission will be permanently inhabited, thus not requiring remote control and monitoring of the experiments from ground. A schematic view of the habitat to be used for deep space exploration missions is reported in figure 6.6, in both launch and operational configurations.



(a) Launch (b) Nominal

Figure 6.6: Habitat for deep space mission architecture

The overall mass of the DSH second unit amounts to about 30 tons, including resources and systems needed to accomplish a 12 months NEA mission. The average power requirements during the NEA mission is about 14kW and the solar arrays resulted to be slightly smaller than the first unit's ones, even due to the different lifetime.

Remarks

The deep space habitat presented in this section is, actually, slightly different from that envisioned in the NEA mission architecture described in section 4.2.3.5. The major difference refers to the absence of the airlock, which instead is replaced by using the MMSEV and suit ports to perform EVAs. However, the core of the module is the same: a rigid node (with one radial port to allow the MMSEV docking while in the asteroid proximity) plus an inflatable module.

The choice of considering both options is justified by the wish to keep the system as versatile as versatile. The final decision about the asteroid proximity operations approach shall be driven by an accurate characterization of the target NEA. Indeed, depending on the features of the specific asteroid one of the options may be better than the other. Therefore, at this level of the study, at which no detailed analyses have been performed to select the target asteroid, both the options are worth to be taken into account.

6.2 Space Tug

This section focuses on the conceptual design of a space tug for satellites servicing. According to the HSE reference scenario previously discussed, a propulsive module is needed for many missions, especially to support operations of RvD in orbit of large spacecraft. The space tug analyzed in this thesis is conceived with different mission objectives, but it shall be seen as a precursor of a larger system devoted to more challenging operations.

6.2.1 Introduction

Besides the aspects related to on orbit assembly of large spacecraft and orbital transfers, which indeed represent crucial points for future explorations, an issue to be faced is related to the fact that space environment is becoming extremely crowded, subject of strong competition among the potential users. In this regard, it is important to develop capabilities for on-orbit maintenance of satellites and refueling operations, as well as retrieving and/or removing space debris. In this framework, the development of a new element like a Space Tug is desirable.
Usually Earth satellites are released in a non-definitive low orbit, depending on the adopted launcher, and need to be equipped with an adequate propulsion system able to perform the transfer to their final operational locations. Exploiting a reusable space tug to support satellites deployment operations would be an attractive solution to improve the market position of the Italian VEGA launcher. Indeed, relying on the support of a pre-deployed element such as a reusable space tug in charge of performing the transfer of the satellite platform from launch orbit to the target one, allows minimizing the propulsion on the satellite and, therefore, maximizing the payload mass capability. As a consequence, the satellite platform can be optimized and standardized devoting all the non-recurring effort exclusively to the payload ("payload-oriented"). Furthermore, additional objectives can be pursued using the space tug. For example, the opportunity

of retrieving on Earth significant payload samples/parts by means of an operative reusable vehicle, such as for example an evolution of IXV (Intermediate eXperimental Vehicle), is considered. For this purpose a suitable rendezvous in LEO of the space tug with the vehicle during one of its operative mission phases is to be envisioned, in order to allow the transfer of the payloads to be re-entered on Earth.
Being conceived as a reusable system, many satellites transfers shall be accomplished and dedicated refueling operations shall be envisaged. In particular, the ISS has to be exploited as spaceport for refueling after a few services.

6.2.2 Mission scenario

The space tug is conceived to perform multiple satellites delivery missions in orbit, relying on electrical propulsion*. The choice of implementing electrical propulsion is in line with the current status of the international aerospace research and with most of the international space roadmaps. The current interest in electrical propulsion is mainly related to the fact that it uses less propellant than chemical rocketry; moreover, it may promise better reliability and simplicity than chemical systems. On the other hand it offers only low thrust propulsion, which means longer transfer times, which is actually an issue when dealing with manned spacecraft; however for applications like these discussed here, this is not a very important aspect.
The idea of increasing as much as possible the payload (either scientific or commercial) mass deployed on orbit, relying on Italian space assets, has driven the performed analyses and choices. By the way, VEGA is considered as baseline launcher and the development of the space tug as a key-element of the scenario is proposed. Specific analyses have been performed to define the most suitable scenario relying on this reusable system. Accordingly, the space tug is conceived to transfer satellites platforms from low Earth orbits, where VEGA launcher releases them, to their final operational orbits, and back, if needed. In this way it is possible to reduce the propulsion system of the

*The reference HSE scenario does not include elements implementing electrical propulsion. However, the interest in such technology is justified by the potential advantages it can have with respect to conventional propulsion. This is particularly true for Earth satellites related applications (no humans involved), and therefore, the tug here discussed could be a good opportunity to implement and validate the technology (future implementation in farther missions shall be then considered).

satellite platforms thus limiting their overall mass and volume, in favor of larger payloads. The satellite platform will be a standard platform, including specific interfaces to allow the docking/grappling by the space tug.
The tug is characterized by high level of reusability, since it is envisioned to perform many orbital transfers and servicing operations along its operational life. For this reason, periodical refueling operations are foreseen, relying on the international space station (a dedicated fuel tank is to be attached to the station, where refueling operations shall take place). The overall reference mission scenario mainly includes the following phases:

- Space Tug deployment,
- Satellite platform deployment,
- Space Tug refueling.

The number of satellites deployments before refueling is needed is evaluated considering the various constraints deriving from the VEGA launcher, as described in the following section (6.2.2.1).

6.2.2.1 Main assumptions and constraints

VEGA launcher

As already addressed, in order to improve the national space operability in terms of access to space, VEGA launcher is considered as the reference launch system. Europe's new VEGA launch vehicle flew for the first time on February 13, 2012, achieving a flawless inaugural mission; then, in May 2013 it successfully accomplished its second launch. It is designed to launch small payloads (300 to 2500 kg satellites) to polar and low Earth orbits. It is launched from Guiana Space Centre, that is the French and European spaceport near Kourou in French Guiana (approximately 500 kilometres north of the equator, at latitude of 5.2°). The performances of VEGA launcher are graphically shown in figure 6.7 [31]. Due to the still limited available data and the uncertainties related to the launch capabilities, for the present study, these curves have been scaled down by 200 kg in order to be conservative. In particular, for the reference launch orbit, which has 700 km altitude and 5.2° inclination, 1800 kg payload capability is assumed. Besides the mass compatibility, the space tug shall be compliant with

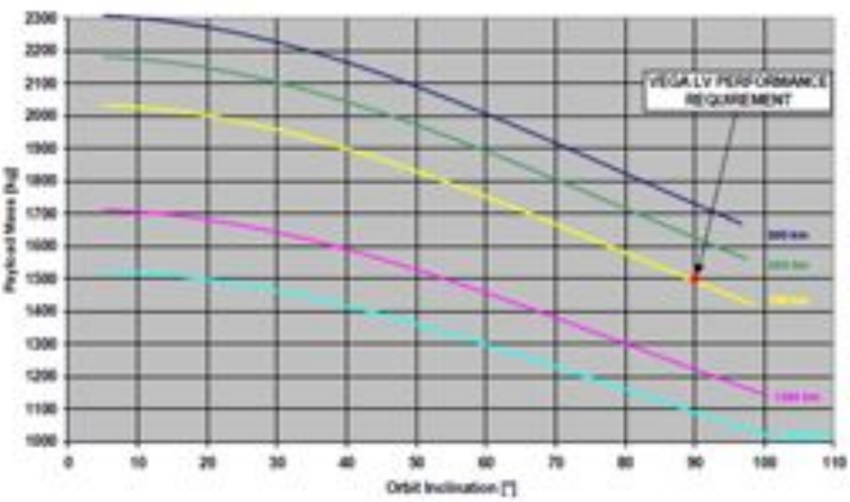


Figure 6.7: VEGA launcher performances

VEGA fairing dimensions (maximum diameter 2.6 m and maximum length 7.8 m).

ISS refueling

Refuelling at the International Space Station is considered as baseline. In order to maximize the payload launched in LEO, the reference launch orbit has a very low inclination (5.2°), where then the space tug shall dock with the satellite platform. According to this, it would be convenient to have a refueling station in an inclination orbit lower than the ISS' one, which is actually a high inclination orbit (51.6°) and therefore requires a significant inclination change. However, the ISS is an already available infrastructure and it is worth to exploit it, even considering that developing, launching and maintaining a new facility would be complex and expensive. Moreover, as addressed before, the idea is to exploit as much as possible already available infrastructures, in particular Italian space assets.

Reference satellite delivery mission

The space tug shall be capable to transfer satellites from the launch orbits to their operational ones. In order to be conservative, the most demanding case has been considered as reference one for the sizing of the space tug, that is the delivery of satellites in geostationary orbit. Firstly the space tug is launched in LEO through VEGA launcher, where it remains while waiting for the first satellite to be launched. A following VEGA launch deploys in LEO the satellite platform. At this point the space tug performs a

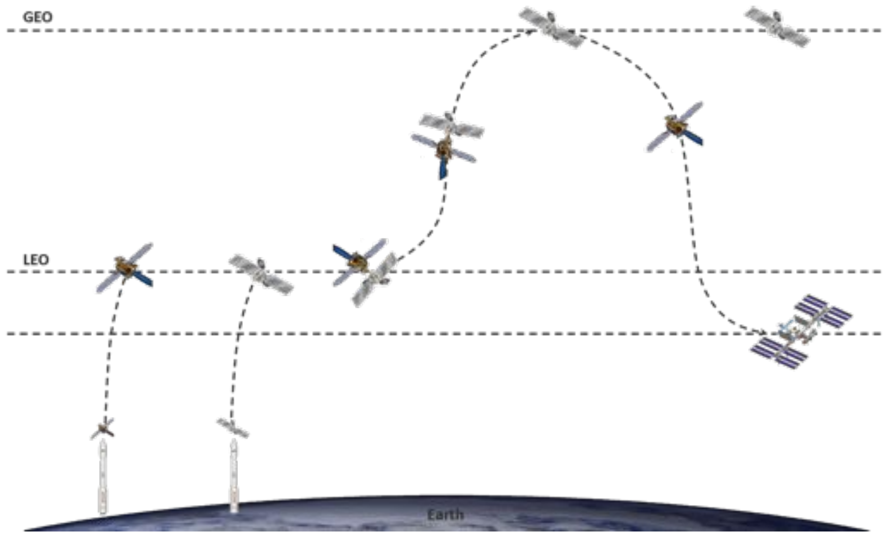


Figure 6.8: Reference satellite delivery mission in GEO

rendezvous maneuver to finally dock with the satellite platform (while this is still attached to VEGA last stage). Once the docking has been completed, the travel toward the final satellite operational orbit starts. After the release of the satellite platform in GEO, the space tug can either come back to LEO to wait another satellite to be deployed, or move to ISS for refueling operations. Figure 6.8 schematically illustrates the phases of launch, satellite delivery in GEO and space tug transfer to ISS.
An additional objective for the space tug can be to support the retrieval of payloads to be re-entered on Earth through an IXV evolution vehicle. The sequence of operations for this mission profile is shown in figure 6.9.

6.2.2.2 Mathematical model

To evaluate the mission scenario a MATLAB® script has been built and computations for different cases have been carried out. The MATLAB® script determines the characteristics of low-thrust orbital transfer between two circular orbits. The numerical method used in this script is described in chapter 14 of the book Orbital Mechanics by V. Chobotov [32]. The algorithm is valid for total inclination changes Δi given by 0 $<\Delta i<114.6°$. Firstly, the initial and final orbit velocities are computed:

$$V_0 = \sqrt{\frac{\mu}{R_0}} \tag{6.1}$$

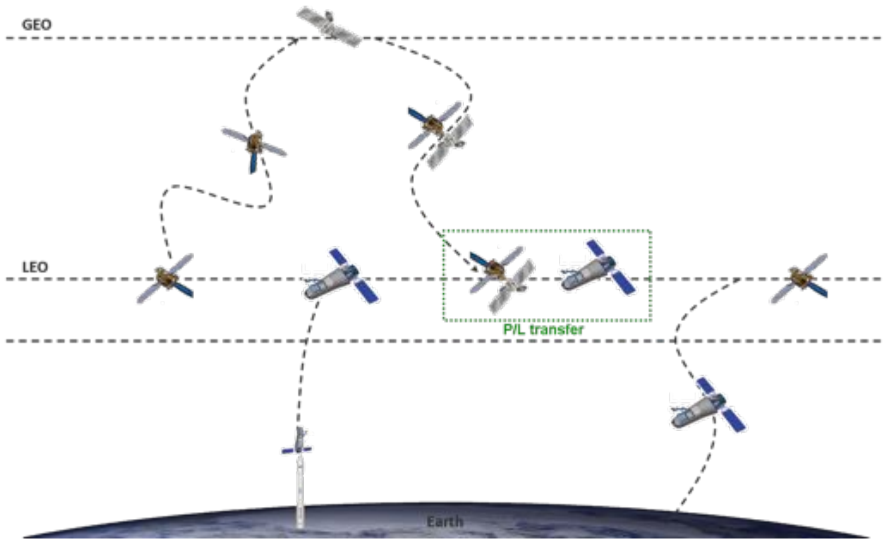


Figure 6.9: Payload retrieval scenario

$$V_f = \sqrt{\frac{\mu}{R_f}} \tag{6.2}$$

where μ is the Earth's gravity constant ($\mu = 398600.44 km^3/s^2$) and R_0 and R_f are the initial and final orbit radius, respectively (computed as $R_0 = R_E + h_0$ and $R_f = R_E + h_f$, with R_E the Earth's radius and h_0 and h_f the initial and final orbit altitudes, respectively). The initial thrust vector yaw angle β_0 is given by the following expression:

$$\tan \beta_0 = \frac{\sin\left(\frac{\pi}{2}\Delta i\right)}{\frac{V_0}{V_f} - \cos\left(\frac{\pi}{2}\Delta i\right)} \tag{6.3}$$

where Δi is the total desired inclination change.
The total velocity change required for a low-thrust orbit transfer is given by:

$$\Delta V = V_0 \cos \beta_0 - \frac{V_0 \sin \beta_0}{\tan\left(\frac{\pi}{2}\Delta i + \beta_0\right)} \tag{6.4}$$

Once obtained the total required velocity change, the initial mass in LEO is computed relying on the typical Tsiolkovsky rocket equation 4.1. Moreover the transfer time is evaluated as

$$t_f = \frac{\Delta V}{f} \tag{6.5}$$

where f indicates the low-thrust acceleration, which for this computation is assumed to be constant during the orbit transfer; the acceleration magnitude is obtained as the ratio between thrust (constant thrust is considered) and the average between final and initial mass.

6.2.3 Simulations results

6.2.3.1 Launcher compatibility

The first objective of the analysis is to evaluate the compatibility with the launcher capabilities, given that multiple delivery missions to GEO are to be performed. Different scenarios are investigated to understand how many delivery missions can be performed before refueling is required and always relying on VEGA launcher. In particular, two specific cases are considered:

1. the space tug performs two satellites delivery missions from LEO to GEO before going to ISS for refueling;
2. the space tug performs three satellites delivery missions from LEO to GEO, before going to ISS for refueling.

The number of missions to be accomplished prior to refueling has a significant impact on the sizing of the tanks, since quite different amounts of propellant shall be loaded, thus increasing the total dry mass of the tug. The possibility to accomplish four satellites delivery missions before refueling is discarded, since the overall tug launch mass would exceed the VEGA launcher capability*. Figures 6.10 and 6.11 show the results obtained for the two cases, respectively.
The graphs report the maximum tug launch mass as function of the launch orbit. In the graphs the maximum VEGA capabilities (referred to launch orbits with 5.2° inclination) are shown as well, in order to have an immediate understanding of the space tug compatibility with the launcher.
The main differences on the tug launch mass between the two cases derive from the dry mass and the amount of propellant needed for the first missions set. The tug dry mass

*This option could be considered if refueling is foreseen immediately after launch; however it is discarded because of the large amount of fuel which would be necessary for the first satellites transfers.

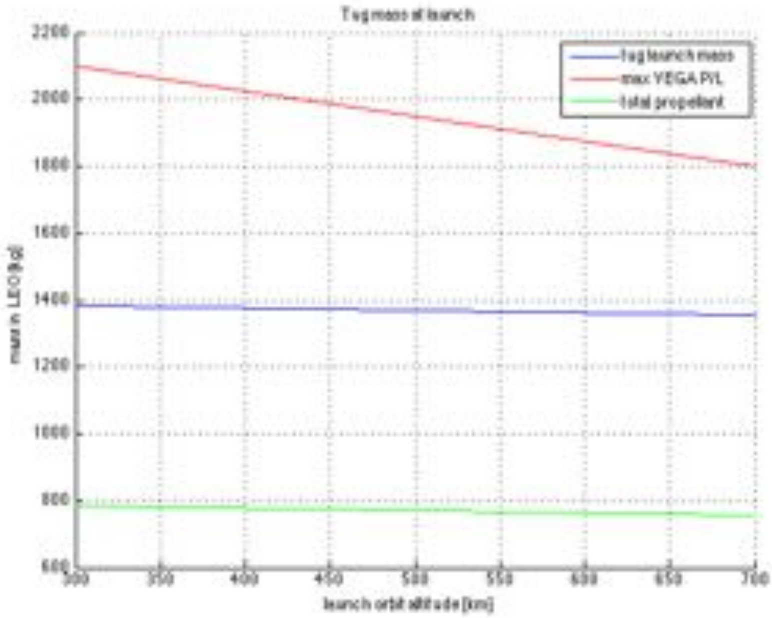


Figure 6.10: Tug launch mass: first set including 1 delivery mission and second set including 2 delivery missions

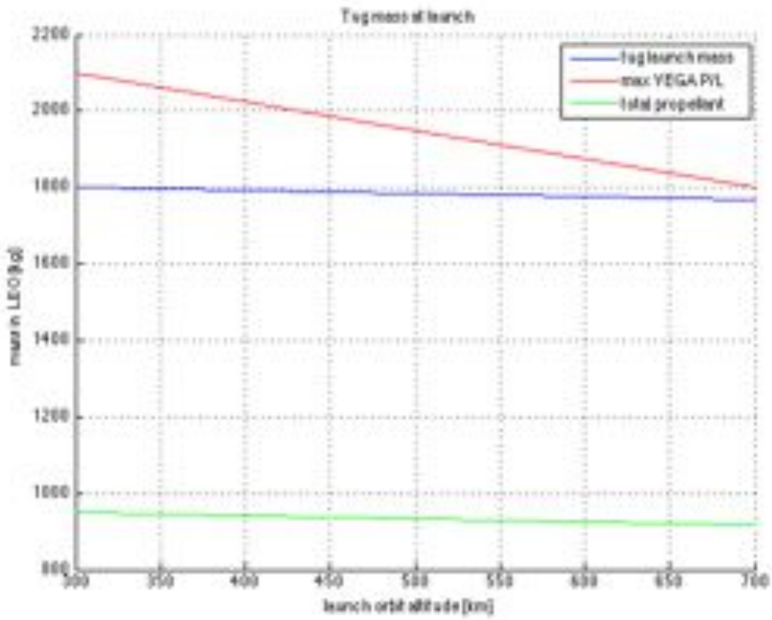


Figure 6.11: Tug launch mass: first set including 1 delivery mission and second set including 3 delivery missions

is higher in the second case, mainly because of the larger tanks needed to load larger amount of propellant for the accomplishment of three LEO-GEO trips (the payload mass is always assumed 1800kg).
However, there is the possibility to load the tug with an additional amount of fuel, in order to maximize the loading capability of the launcher. Especially in the first case (figure 6.10) the total tug launch mass is quite below the maximum VEGA capabilities for all the considered launch orbits.
It has to be underlined that the first refueling of the tug is needed after the first satellite delivery mission (the launch mass shown in figures 6.10 and 6.11 includes the propellant for only one satellite delivery mission before the first refueling at ISS). Indeed, it is not possible to launch the space tug with the amount of fuel needed for more than one delivery mission because of compatibility reasons with the launcher, as shown in figures 6.12 and 6.13.

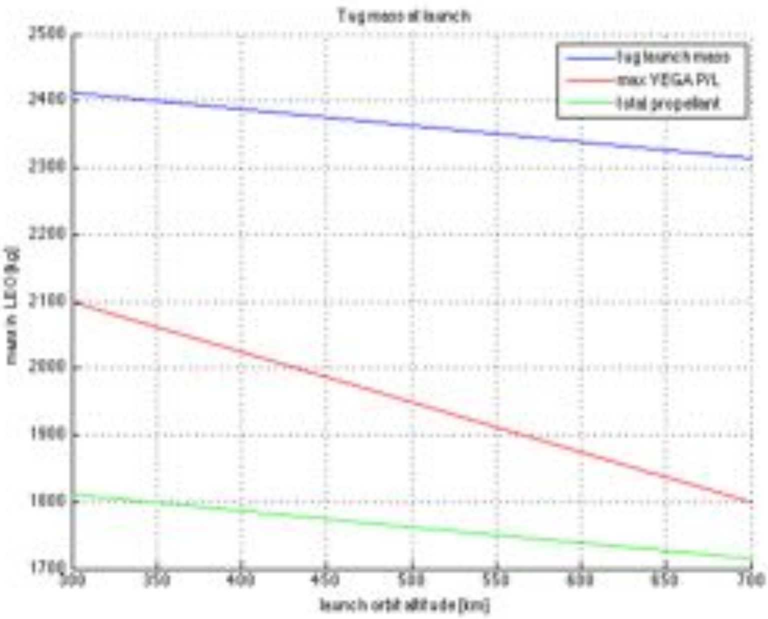


Figure 6.12: Tug launch mass: first set including 2 delivery mission and second set including 2 delivery missions

The scenario considered for the analyses is a conservative one, since all the calculations refer to the worst case of delivering 1800kg satellite to GEO. However, there could be the need of delivering a satellite to a lower orbit than GEO and in that case more

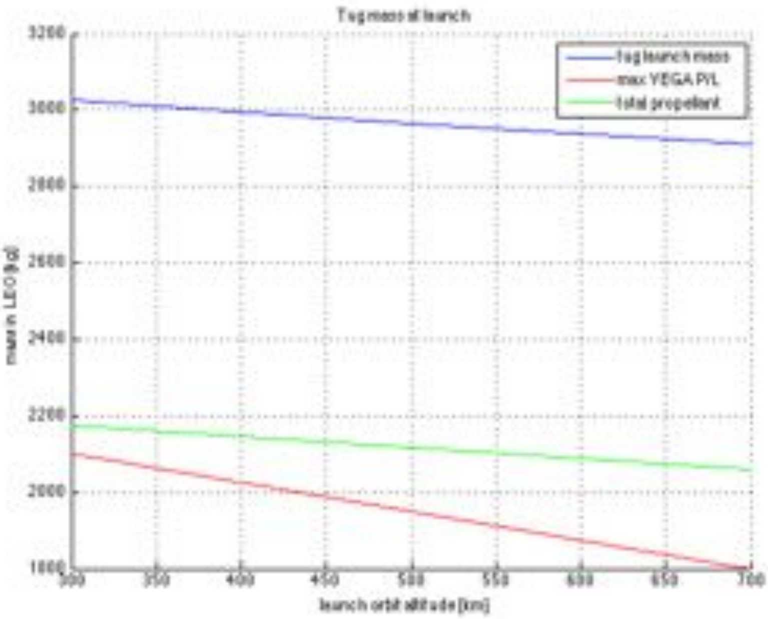


Figure 6.13: Tug launch mass: first set including 2 delivery mission and second set including 3 delivery missions

than two-three missions would be possible. The idea is to size the space tug (and in particular the tanks for the propellant) for the worst condition, but then exploit it for more satellites delivery missions.

6.2.3.2 Missions budgets

Computations for the two cases introduced in previous section have been carried out. In particular the results discussed in this section are obtained considering the following assumption:

- launch orbit (departure LEO): 700km, 5.2°,
- final orbit (arrival GEO): 36000km, 0°,
- ISS orbit (refueling orbit): 360km, 51.6°,
- constant thrust equal to 480mN and I_{sp}=2500s,
- satellite platform mass: 1800 kg (max VEGA launcher capability in the reference launch orbit).

Case 1

The sequence of operations for the first analyzed case is schematically shown in figure 6.14 (the green box represents the tug, while the orange one refers to the satellite platform). All the elements are launched in LEO by means of VEGA.

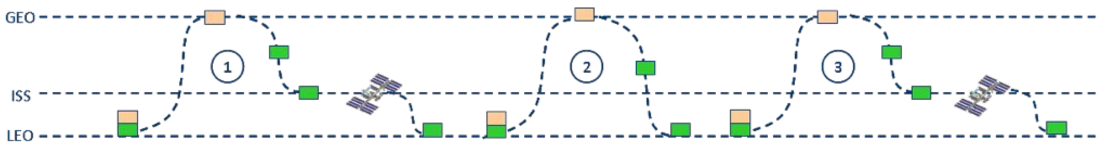


Figure 6.14: Case 1: two satellites delivery missions before refueling

The first missions set starts with the launch of the space tug, which then remains in LEO while waiting for the launch of the first satellite platform. Once the tug has docked with the satellite platform, the transfer towards the final GEO begins. After releasing the satellite in GEO, the tug moves to ISS to perform the first refueling. After refueling operations have been completed, the second missions set can start. The tug moves back in LEO, where the satellite platform has to be launched by VEGA. Once the tug has docked with the satellite platform, the transfer towards the final GEO begins. After releasing the satellite in GEO, the tug moves back to LEO to perform a second satellite delivery to GEO, prior to move again to ISS for a second refueling.
According to the scenario just described and using the developed MATLAB® tool, the various phases are analyzed and the masses of propellant needed to accomplish the various transfers have been calculated. Moreover, an estimate of the transfer time is performed, considering constant thrust acceleration. The obtained results are summarized in table 6.10. These results are obtained considering a tug dry mass equal to

Phase		Initial mass [kg]	Final mass [kg]	Propellant mass [kg]	Transfer time [days]
First mission set	LEO-GEO 1	3155	2627	528	313
	GEO-ISS 1	827	600	227	135
Second mission set	ISS-LEO 2	3345	2315	1030	616
	LEO-GEO 2	4115	3427	688	408
	GEO-LEO 2	1627	1355	272	161
	LEO-GEO 3	3155	2627	528	313
	GEO-ISS 3	827	600	227	135

Table 6.10: Case 1: Missions phases budgets

600 kg (additional details are reported in section 6.2.4.4). The total propellant mass

needed to accomplish the first missions set (one satellite delivery to GEO + transfer to ISS for refueling) is about 755 kg, while the total propellant mass needed for the second missions set (tug transfer to LEO + two delivery missions in GEO + transfer to ISS for refueling) is about 2740 kg.

The launch mass of the tug, in this case is about 1350 kg; this value is below the maximum capability of the VEGA launcher (1800kg in 700km, 5.2° LEO) and therefore additional propellant can be loaded and then exploited in the following missions. In particular the total additional amount that can be included is around 250kg still being compatible with the launcher capability. In this case, the budget for the first missions set is shown in table 6.11 (the second missions set budgets are obviously the same).

Phase		**Initial mass [kg]**	**Final mass [kg]**	**Propellant mass [kg]**	**Transfer time [days]**
First mission set	LEO-GEO 1	3568	2971	597	354
	GEO-ISS 1	1171	850	321	192

Table 6.11: Case 1 (additional propellant launched with tug): Missions phases budgets

Case 2

The sequence of operations for the second analyzed case is schematically shown in figure 6.15. It is analogous to the first scenario, described in the previous section, but in this case, the tug performs three satellites delivery missions prior to moving to ISS for refueling operations. Even for this scenario, the propellant mass needed to accomplish

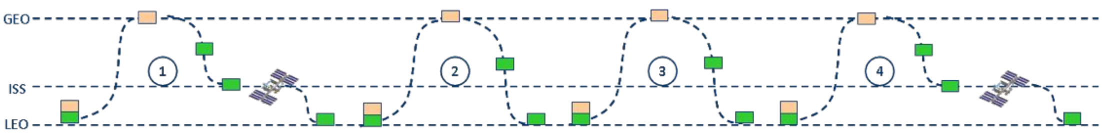


Figure 6.15: Case 2: three satellites delivery missions before refueling

the various transfers is evaluated, as well as the transfer time. The obtained results are summarized in table 6.12. These results are obtained considering a tug dry mass equal to 850 kg*.

The total propellant mass needed to accomplish the first missions set (one satellite delivery to GEO + transfer to ISS for refueling) is about 920 kg, while the total propellant

*The dry mass for this analysis case is obtained following the same process as for the previous case, analogously to what described in detail in section 6.2.4.4

Phase		Initial mass [kg]	Final mass [kg]	Propellant mass [kg]	Transfer time [days]
First mission set	LEO-GEO 1	3568	2971	597	354
	GEO-ISS 1	1171	850	321	192
Second mission set	ISS-LEO 2	6590	4561	2030	1214
	LEO-GEO 2	6361	5297	1064	631
	GEO-LEO 2	3497	2912	585	347
	LEO-GEO 3	4712	3924	788	467
	GEO-LEO 3	2124	1768	355	210
	LEO-GEO 4	3568	2971	597	354
	GEO-ISS 4	1171	850	321	192

Table 6.12: Case 2: Missions phases budgets

mass needed for the second missions set (tug transfer to LEO + three delivery missions in GEO + tug transfer to ISS for refuelling) is about 5740 kg. In this case, the launch mass of the tug amounts to almost 1770kg, and therefore no extra propellant can be loaded.

6.2.3.3 Reference Scenario Selection

In order to select the best scenario between the two analyzed ones some preliminary costs evaluations are also performed. The considered costs are the launch costs associated to the various missions including the launch of the space tug, the satellites and the propellant tank (to be attached to ISS). The total cost as function of the number of satellites delivery missions is shown in figure 6.16 for the two analyzed cases.
The total cost is evaluated considering the total number of launches necessary to launch the space tug, the satellites and the fuel for the space tug refueling. The reference launcher for the propellant tank launch is the Soyuz-FG, which has a maximum payload capability in LEO (200km, 51.6°) of 7200kg and a launch cost of 50M$. The VEGA launch cost is assumed 40M$. From the graph, it can be seen that as the total number of satellites delivery missions increases the most convenient option is that foreseeing refueling every two missions.
To have a more complete trade-off, additional issues should be taken into account. Indeed, the risks associated with the docking with the ISS and with the refueling operations are very high, and in this regard, the chosen configuration would imply larger risks than the other option. However, this analysis is focusing on a worst-case reference

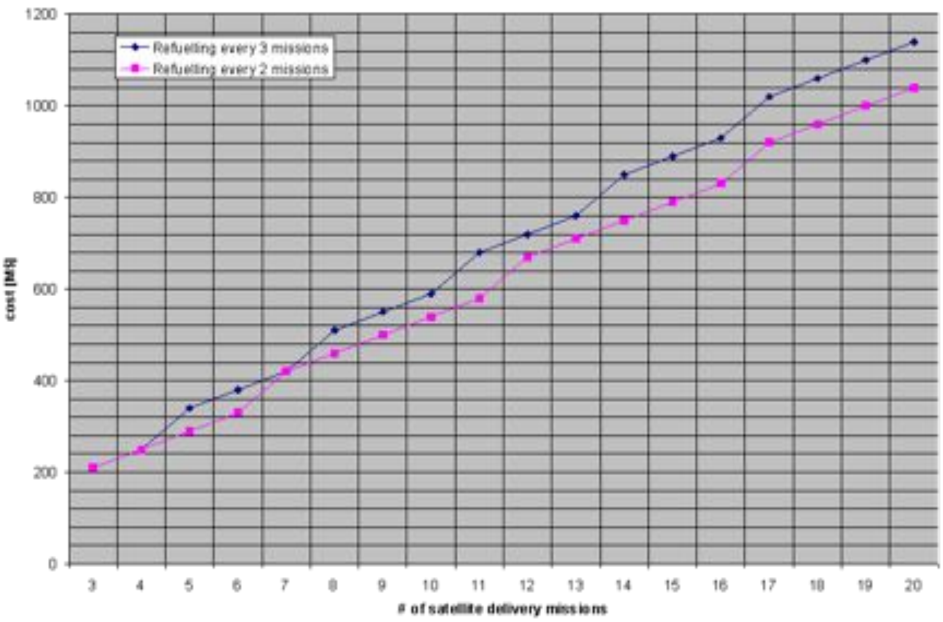


Figure 6.16: Launch costs comparison

scenario, foreseeing all satellites delivery to GEO, but, as already assessed, it is likely to have also different types of delivery missions to lower orbits, which would allow lower refueling operations frequency.

6.2.4 Conceptual design

6.2.4.1 Mission objectives

According to the typical conceptual design process in Systems Engineering (see figure 3.4 in section 3.3), the mission statement, which is reported hereafter, is firstly established:

To improve the national space operability in terms of access to space by providing a transportation system capable to transfer satellites platforms from low launch orbits to operational orbits and back, relying on Italian space assets.

Starting from the mission statement, the mission objectives are derived:

- *to perform satellites transfer from low to high Earth orbits,*

- *to retrieve satellites from high to low Earth orbits,*
- *to re-enter on Earth payloads loaded on board satellites once completed their operative cycle,*
- *to perform refuelling on orbit.*

6.2.4.2 Mission requirements

Once the broad goals of the system, represented by the mission objectives, have been identified, the system requirements are defined. On the basis of the system requirements, the system architecture is determined. In order to proceed with the sizing of the system the top-level requirements have to be assessed. Hereafter, a summary of the most significant ones is reported.
The space tug shall...

Functional Requirements

- ...release satellites in their final orbits
- ...perform multiple transfers of satellites from LEO (initial launch orbit) to high orbits (nominal orbits)
- ...perform RvD with satellite standard platform (cooperative target)
- ...perform transfers of satellites from high to low orbit to support retrieval of payloads to be re-entered on Earth
- ...perform RvD with re-entry vehicle (IXV-evolution)
- ...be provided with autonomous operation capability
- ...perform automatic RvD with refueling station (fuel tank at ISS)
- ...perform refueling operations on orbit
- ...be provided with re-start capability

Mission and Operational Requirements

- ...remain in "parking" LEO up to TBD days
- ...perform refueling operations every two GEO satellites delivery missions

Interface and Physical requirements

- ...be compatible with VEGA launcher capability payload

- ...have TBD interfaces with the satellite standard platform
- ...have TBD mechanical interface with refueling station
- ...have TBD mechanical interface with refueling station

Environmental requirements

- ...withstand LEO environment
- ...withstand GEO environment
- ...withstand the launch loads

Product Assurance and Safety requirements

- ...be maintainable on orbit
- ...comply with all safety requirements of the launch sites and launch vehicle
- ...comply with all safety requirements of the refuelling station (ISS)

6.2.4.3 Space tug configuration

As explained in section 3.3, the Functional Analysis is a fundamental tool of the design process to explore new concepts and define their architectures. This analysis is performed to refine the space tug functional requirements, to map its functions to physical components, to guarantee that all necessary components are listed and that no unnecessary components are requested and to understand the relationships among the new product's components. According to the functional analysis, the functions/components matrix is used to map functions to physical components. Specifically, figure 6.17 illustrates the functions/components matrix built for the space tug.
As result of the functional analysis the assessment of the subsystems and components needed to accomplish the mission is derived. In summary, the space tug is composed of the following subsystems:

- Propulsion Subsystem, which includes the main thruster (electric) and the reaction control system; the propellants tanks are also part of the propulsion subsystem, with all the interface and feeding system needed to provide propellant to the thrusters;

		Structure S/S						Electrical Power S/S				Thermal Control S/S					Data Management S/S			Comms S/S				Attitude & Orbit Det. & Control S/S				Propulsion S/S			Harness
FUNCTIONS	SUB-FUNCTIONS	Radiation shielding	MDPS panels	Primary structure	Secondary structure	Docking I/F	Refueling I/F	Solar arrays	Batteries	Power distribution unit	Converters	Radiators	Heaters	MLI	Piping	TCS SW	Command & Acquisit. unit	On-board computer	Mass Memory Unit	Antenna	Receiver	Transmitter	Auto-tracking equipment	Orbit sensors	Attitude sensors	Attitude Actuators	GNC SW	Main Engine	RCS thrusters	Propellant tanks	Harness
Environmental Protection	MMOD Protection		x																												
	Radiation protection	x																													
Mech. /Structural functions	Ground handling loads			x																											
	Flight accererations			x																											
	Operational loads			x	x																										
Power	Production							x																							
	Storage								x																						
	Collection							x																							
	Conversion & Distribution									x	x																				x
Thermal Control	Heat collection & transportation														x	x															
	Heat rejection											x																			
	Local heat sink												x																		
	Thermal insulation													x																	
Data Handling	Monitoring																x	x	x												x
	Control																x	x	x												x
	Command																x	x		x	x										x
GNC	Orbital Manoeuvreing																							x				x			
	RvD					x																					x		x		
	Attitude determination																								x						
	Attitude Control																									x			x		
	Station-keeping																							x				x			
Communication	Data communication																			x		x	x								x
Propellant management	Propellant storage																													x	
	Refuelling						x																								

Figure 6.17: Space tug functions/devices matrix

- Electric Power Subsystem (EPS), in charge of providing, storing and distributing power to the other subsystems; in this specific case, this is a very impacting subsystem, since electric thrusters require high power levels to function;
- Thermal Control Subsystem (TCS), to maintain all spacecraft and payload components and subsystems within their required temperature limits for each mission phase;
- Attitude and Orbit Determination and Control Subsystem (AOCS), needed to stabilize the vehicle and orient it in desired directions during the mission despite the external disturbance torques acting on it; the attitude control is particularly critical for the RvD maneuvers with the satellite platform; moreover an accurate attitude maintenance will be necessary for the refueling operations;
- Data Management Subsystem (DMS), which receives, validates, decodes, and distributes commands to other spacecraft systems and gathers, processes, and formats spacecraft housekeeping and mission data for downlink;
- Communications subsystem, which provides the interface between the spacecraft and the ground systems, transmitting mission and spacecraft housekeeping data;
- Structure subsystem, which supports all other spacecraft subsystems, and includes the attachment interfaces with the launcher and the ground support equip-

ment interfaces; moreover, it includes the RvD mechanism to dock with the satellite platform and the refueling interface with the nozzle tool (which is the tool allowing the transfer of propellant from the refueling depot attached to ISS to the space tug tank);

- Harness subsystem, which includes satellite wiring, electronics backplane, electrical interface boards.

6.2.4.4 Mass breakdown

A preliminary mass budget is performed to obtain a breakdown for the tug. The mass budget is obtained taking as reference the Dawn mission spacecraft [33], since this is a real mission implementing electric propulsion system.
The tug dry mass is computed starting from the sizing of the propulsion and the electrical power subsystems, which are the most impacting subsystems for this type of vehicle [34].
The propulsion system mass varies with the specific impulse (exhaust velocity), thrust level and total impulse. Propellant mass clearly drops off as specific impulse increases. The power source requirements, however, are proportional to I_{sp}. Thus, the mass of the power source increases with specific impulse, leading to a minimum mass of the combined system (fuel and power source) at a particular value of I_{sp}. The propulsion system can be considered as composed of two main parts:

- the thruster "subsystem", including the thruster and the power conditioning unit,
- the propellant "subsystem", including the tanks and propellant management systems.

Figures 6.18 and 6.19 show the thrust over power ratio (R) and the specific mass (SM), given as functions of the specific impulse for various types of engines, respectively [17].

The system specific mass only includes the mass of the thruster and power processor (the masses of the propellant subsystem, gimbals, and other mission specifics are not included). For the present computation, the Hall Effect Thrusters are assumed as reference and the following values are used (I_{sp} = 2500s):

- R=50 mN/kW,

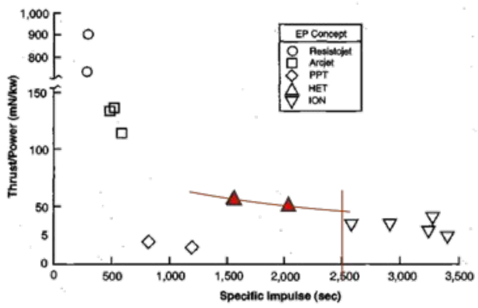


Figure 6.18: Thrust over power ratio

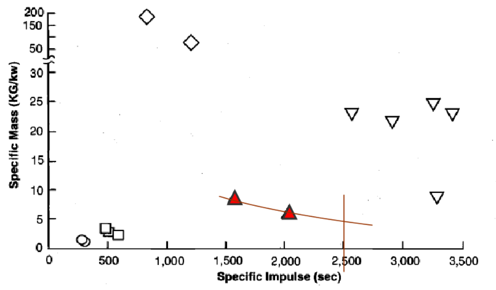


Figure 6.19: Specific mass

- SM=5kg/kW.

With these values, the power (P) needed to obtain the required thrust (T) is computed ($P = T/R$) and then the mass (M) is derived ($M = SM \cdot P$). Considering a required thrust of 480mN, the needed power amounts to about 9.6kW and the thruster mass is about 50kg. The propellant tanks mass is computed as the 4% of the total propellant mass to be loaded (about 2740kg). According to this, the overall mass of the propulsion subsystem amounts to almost 160kg. For what concerns the Electrical Power Subsystem (EPS), it includes deployable solar panels for power generation and batteries for energy storage. The EPS is sized such that propulsion is constantly guaranteed both in daylight and eclipse condition. To perform the sizing of the solar arrays, the initial LEO (h=700km) is taken as reference orbit, as it represents the worst case, having the longest eclipse time.
The needed solar arrays area is computed as:

$$A_{SA} = \frac{P_{SA}}{P_{EOL}} \tag{6.6}$$

where P_{SA} is the power that solar arrays must provide during daylight to power the spacecraft for the entire orbit, given by:

$$P_{SA} = \frac{\frac{P_e T_e}{x_e} + \frac{P_d T_d}{x_d}}{T_d} \tag{6.7}$$

where P_e and P_d are the power requirements during eclipse and daylight respectively, T_e and T_d are the length of these periods, x_e and x_d the efficiencies of the paths from the solar arrays through the batteries to the individual loads and the path directly from the arrays to the loads, respectively. The total power required to be provided by the solar arrays amounts to about 19kW, including power for batteries recharge, as well as other subsystems required power. P_{EOL} is the array power per unit area at the end of life. It can be computed by multiplying by a degradation factor the power per unit area at the beginning of life.

$$P_{EOL} = P_{BOL} L_d \tag{6.8}$$

$$L_d = (1 - degradation/yr)^{operative\ life} \tag{6.9}$$

The power per unit area at beginning of life is obtained as:

$$P_{BOL} = \varphi_{Sun} \eta I_d \cos\theta \tag{6.10}$$

where φ_{Sun} is the Sun flux, η is the conversion efficiency, I_d is the inherent degradation, which accounts for the design and assembly losses, θ is the Sun incidence angle. Considering high efficiency solar cells (triple junctions cells with 30% efficiency and 84mg/m^2 specific mass), the required area to get 19kW during daylight is about 54m^2. The corresponding solar arrays mass is about 83kg (computed assuming that the blanket mass is 55% of the total array mass [30]). Li-ion secondary batteries are foreseen to provide energy during eclipse, thus allowing continuous propulsion. For the batteries sizing the following equation is used:

$$C_r = \frac{P_e T_e}{(DOD) x_e} + self\text{-}discharge \tag{6.11}$$

The obtained required batteries total capacity is 10kWh. Considering a specific energy of 175Wh/kg, the total battery mass is about 57kg. The power control and distribution unit mass is obtained as the 20% of the overall EPS mass. Therefore the total EPS mass is about 175kg.

Starting from the mass values obtained for the propulsion and power subsystems, the total dry mass is computed referring to Dawn mission. For Dawn spacecraft the propulsion and electrical power subsystems constitute about 50% of the total dry mass. For the space tug discussed in this thesis, a bit larger percentage is considered since the power requirement and the quantity of propellant needed for the missions are higher (larger solar arrays and tanks). Specifically, a total of 60% of the total dry mass is assumed. With this percentage the obtained dry mass is about 550kg (without system margin). The mass fractions listed in table 6.13 are used for the preliminary assessment of the other subsystems masses. They are derived from the Dawn ones (readjusted to

Subsystem	Mass fraction [%]
Propulsion (w/ tanks)	28
EPS	32
TCS	7
AOCS	5
DMS	3
Communications	3
Structure	15
Harness	7

Table 6.13: Space Tug mass breakdown - S/Ss mass percentages

Subsystem	Mass [kg]
Propulsion (w/ tanks)	157
EPS	175
TCS	36
AOCS	28
DMS	16
Communications	16
Structure	83
Harness	39
TOTAL (w/o sys margin)	550
Sys margin	10%
TOTAL (w/ sys margin)	∼600

Table 6.14: Space Tug mass breakdown - S/Ss mass

take into account the larger propulsion and power subsystems mass fractions).
The obtained mass breakdown is reported in table 6.14. A system margin of 10% is included to account for the uncertainties typical of this design phase. Accordingly, the resulting total dry mass is 600kg.

7

Additional missions/systems/ideas in support to HSE and science

7.1 Interplanetary CubeSats mission

Different types of missions relying on small systems/satellites can be included in a reference exploration scenario, as they would support technologies demonstration in space environment, besides accomplishing specific scientific objectives.
According to this, an interplanetary CubeSats mission has been studied (together with university "La Sapienza" of Rome, the Astrophysical Observatory of Torino and the DLR), as described hereafter.

7.1.1 Interplanetary CubeSats mission: introduction

Interplanetary CubeSats could enable small, low-cost missions beyond low Earth orbit. CubeSats are typically characterized by 10cm x 10cm x 10cm dimensions and a mass not exceeding 1.33 kg; they can also be arranged in double and triple units systems. Although a large number of CubeSats have already been developed and launched into Earth's orbit, none have accomplished an interplanetary mission. Since big missions

are usually very costly, relying on CubeSats could be an interesting alternative to accomplish both scientific and technological tasks in deep space, as proven by the growing interest in this kind of application in the scientific community and most of all at NASA.
The CubeSats mission analyzed in this thesis envisages the deployment of a 6U CubeSats system in one of the Earth-Sun Lagrangian Points. It is aimed at supporting measurements of space weather, which is quite a critical issue especially for what concerns the human exploration of space beyond Earth's orbit where the protection of the Earth magnetic field is not available anymore. Moreover, the mission is intended as a technology validation mission, with the aim of testing advanced technologies in view of future implementation in larger missions (e.g. solar sails, far distance telecommunications).
Regarding the support to future exploration missions, another issue taken into consideration is related to the space radiation environment. Indeed, traveling outside the Van Allen belts, the CubeSats system gives the opportunity for further investigations of the deep space environment: radiation dosimeters and advanced materials are envisaged to be implemented, in order to test their response to the harsh space environment, even in view of future implementation on manned spacecrafts.

7.1.2 CubeSats mission

7.1.2.1 Mission objectives

According to the typical conceptual design process in Systems Engineering (see figure 3.4 in section 3.3), the mission statement, which is reported hereafter, is firstly established:

To perform solar observation and in-situ space weather measurements from the L_1 Earth-Sun Lagrangian point, pursuing a low-cost approach relying on interplanetary CubeSats and providing a platform for advanced technologies test.

Starting from the mission statement, the mission objectives are derived. They can be split into two different groups:

1. Scientific objectives
    - *to observe the Sun*

    - *to perform plasma measurements*
    - *to perform radiation measurements*

2. Technological objectives
    - *to develop a low-cost CubeSats platform*
    - *to implement solar sail propulsion*
    - *to communicate to Earth from very distant region (Earth-Sun L1)*
    - *to collect, store, manage and send to Earth large quantity of scientific data*

7.1.2.2 Mission requirements

Once the broad goals of the system, represented by the mission objectives, have been identified, the system requirements are defined. On the basis of the system requirements, the system architecture is then determined. In order to proceed with the sizing of the system the top-level requirements have to be assessed. Hereafter, a summary of the most significant ones is reported.

Functional Requirements

- The system shall perform an interplanetary mission to the first Earth Sun Lagrangian point.
- The system shall be provided with interfaces with the launcher.
- The system shall withstand the launch loads.
- The system shall withstand the deep space environment.
- The system shall perform plasma measurement.
- The system shall take pictures of the Sun.
- The system shall perform radiations measurements (total ionizing dose).
- The system shall allow communications with Earth.
    - command data (uplink)
    - telemetry data (downlink)
    - scientific data (downlink)

Performance requirements

- The system shall be compliant with 6U CubeSats standards
    - maximum envelope: 20cm x 30cm x 10m

    - maximum total mass: 8kg

- The total required power shall not exceed 20W.
- The max required data rate shall not exceed 500kbps.

7.1.2.3 Mission analysis

The 6U CubeSats system motion is modeled as a circular restricted three-body problem (CR3BP), in which Sun and Earth are the massive bodies moving in circular orbits around their center of mass [35]. The CubeSats system has instead negligible mass, thus it is supposed to move in the resulting force field without affecting the motion of the primaries. To bound the motion in the vicinity of an unstable point, corrective maneuvers are required. In this work the motion around L1 unstable point is considered envisaging the third body, i.e. the 6U CubeSats system, equipped with an ideal solar sail (an ideal solar sail reflects all the incoming radiation and is not interested by deformation). The sail attitude and the satellite path are obtained solving an optimal control problem with the Direct Collocation with Non Linear Programming (DCNLP) approach.
In defining the optimization process, a Halo orbit is used as initial guess for the trajectory, which is an approximated solution of the CR3BP characterized by the equality of the in-plane and out-of-plane motion frequencies.
For the L1 point of the Sun-Earth system, Halo orbits have a period T of approximately 177 days, which is roughly half a year, hence to simulate a one-year CubeSat trajectory tests for 2T are conducted.
In order to obtain a trajectory as close as possible to a periodic orbit, the optimal control problem is solved minimizing the following performance index:

$$J = \Delta \boldsymbol{r} + \Delta \boldsymbol{v}$$

which represents the difference between initial and final state, evaluated considering both the difference between positions and between velocities. In addition constraints on the control vector are imposed to limit the sail attitude rates to 5 degrees per day. It is worth pointing out that no Halo station-keeping is performed; Halo orbits are only used as initial guess for the final optimal trajectory.

7.1.3 CubeSats system configuration

7.1.3.1 Functional analysis

The functional analysis leads to the selection of the subsystems and components needed to accomplish the identified functions, by means of the functions/devices matrix. The obtained results are shown in figure 7.1, which illustrates the functions/components matrix for the complete 6U CubeSats system.

DEVICES / FUNCTIONS	Structure S/S				Electrical Power S/S					Thermal Control S/S				Command and Data Handling S/S				Communications S/S				Attitude & Orbit Det. & Control S/S				Mission Observation S/S			
	Radiation shielding	MDPS panels	Primary structure	Secondary structure	Solar arrays	Batteries	Power distribution unit	Regulators	Converters	Radiators	Heat pipes	MLI	Surface finishes	Acquisition unit	On-board computer	Memories	Watchdog timer	Antenna/optical photoreceptor	Data Bus	Receiver/Transmitter	Auto-tracking equipment	Orbit sensors	Thrusters/Solar sails	Attitude sensors	Attitude Actuators	Camera	Spectrometer	Magnetometer	Dosimeter
Radiation protection	x																												
Micro meteoroids and debris protection		x																											
Ground handling loads withstanding			x																										
Flight accelerations withstanding			x																										
Operational loads withstanding			x	x																									
Power generation					x																								
Energy storage						x																							
Power distribution							x																						
Power regulation and control								x	x										x										
Heat rejection										x																			
Local heat sink											x																		
Thermal insulation												x	x																
Commands processing														x	x	x			x										
Telemetry and mission data processing														x	x	x			x										
Attitude control functions														x	x	x			x										
Computer health monitoring provision															x		x												
Command reception																		x	x	x									
Telemetry transmission																		x	x	x									
Orbit determination (navigation)															x							x							
Orbit control (guidance)															x								x						
Attitude determination															x									x					
Attitude control															x										x				
Sun/Deep Space observation																										x			
Plasma measurements																											x	x	
Radiation measurements																													x

Figure 7.1: Functions/components matrix

As result of the functional analysis the assessment of the subsystems and components needed to accomplish the mission is derived. In summary, the following subsystems compose the 6U CubeSats system:

- structure subsystem, which supports all other spacecraft subsystems, and includes the mechanical interfaces with the launcher and the ground support equipment interfaces (to be defined);

- electrical power subsystem, which is in charge of providing, storing, distributing and controlling the spacecraft electrical power; it mainly consists of solar cells mounted on the external surfaces of the system as power source, Li-ion batteries for the energy storage and power distribution unit;
- thermal control subsystem, designed to maintain all spacecraft and payload components and subsystems within their required temperature limits for each mission phase; for this mission a passive solution is envisaged;
- command and data handling subsystem, which receives, validates, decodes, and distributes commands to other spacecraft systems and gathers, processes, and formats spacecraft housekeeping and mission data for downlink, maintains mission time and synchronization, manages operative modes and failures;
- attitude and orbit determination and control subsystem, needed to determine attitude, trajectories, angular and linear velocities, handling the measurements of inertial and not inertial sensors, to stabilize the vehicle and orient it in desired directions during the mission despite the external disturbance torques acting on it using magnetic actuators and small reaction wheels; solar sails are exploited for orbit control;
- communications subsystem, which provides the interface between the spacecraft and the ground systems, transmitting both payloads mission data and spacecraft housekeeping data; for an interplanetary CubeSats mission optical communication is likely to be implemented, in order to be compliant with mission requirements and constraints (see section 7.1.4.2 for more details);
- mission observation subsystem, which includes the scientific instruments for Sun observation and plasma measurements (see section 7.1.3.2).

Besides the allocation of the subsystems, one of the main issues related to CubeSats is how to fit big science within a small package - namely power, mass, volume, and data limitations. One of the objectives of the work is therefore to identify and size the required subsystems and equipment, needed to accomplish specific mission objectives, and to investigate the most suitable configuration, in order to be compatible with the typical CubeSats (multi units) standards.

A reference system able to fulfill the scientific objectives of the proposed mission may consist of:

- 2U occupied by the scientific payloads;
- 2U for the solar sails;
- 2U devoted to telecommunications and other bus subsystems (power subsystem, attitude control system and command and data handling).

7.1.3.2 Mission payloads

The scientific instruments to be included in the system are selected according to the main mission objectives. Specifically, the types of instruments to be considered are:

- Plasma Instruments, for plasma measurements;
- Radiation Dosimeters and Advanced Materials, to investigate the space environment and validate technologies in view of future implementation in human missions;
- Imagers/Cameras, to take pictures of the Sun.

For each instruments class, several options are considered and among them only the most significant ones are selected, also according to constraints deriving from the CubeSats standards. In particular, all the scientific payloads shall fit 2U CubeSat sizes (10cm x 10cm x 20cm, 2.66kg).
Hereafter, the main features of the instruments are briefly discussed and the justification for the selection of specific ones is reported.
Two instruments are envisaged to perform measurement of the plasma environment, a magnetometer and a plasma spectrometer.
The reference magnetometer considered for this mission is a tri-axial magnetometer utilizing Anisotropic Magneto-Resistance (AMR) [36]. It is a low cost magneto-resistive magnetometer designed for use in LEO small satellites and CubeSats, with very low mass and small size. Its main features are listed in table 7.1.

Mass	Volume	Power	Data
Sensor: 15g Electronics: 150g	Sensor: 10x10x5mm Electronics: 90x30x11mm	Power consumpt.: 400mW Power supply: +5V and +15V DC or 28V unregulated option	Measurement range: +50000nT to -50000nT Sensitivity: 10nT Update rate: 10Hz Data rate:140bps

Table 7.1: Magnetometer features

The reference spectrometer is an Ion and Neutral Mass Spectrometer (INMS), that is a miniaturized analyzer designed for sampling of low mass ionized and neutral particles in the spacecraft ram direction. The key sensor components consist of a collimator/ion filter, an ionizer and a charged particle spectrometer. Particles enter the aperture into the ion filter region where charged particles can be rejected. This is followed by a series of baffles for collimation and further charged particle suppression. Collimated neutral particles are subsequently ionized in the ionizer by a 50 eV electron beam followed by mass selection in the analyzer. The spectrometer can be operated in different modes, optimized for ions or neutral particle analysis. The INMS main features are listed in table 7.2.

Mass	**Volume**	**Power**	**Data**
Mass: 350g	Envelope: 100x100x50mm (1/2U)	Power consumption: 500mW	Data rate: ∼23bps

Table 7.2: INMS features

As introduced before, the CubeSats mission represents an opportunity to study the deep space environment, and in particular to test the response of specific materials, which can be used to shield the spacecraft.

Accordingly, radiation micro dosimeters are envisioned [37], which are compact hybrid microcircuits which directly measure the total ionizing dose absorbed by an internal silicon test mass. The test mass simulates silicon die of integrated circuits on-board a host spacecraft in critical mission payloads and subsystems. By accurately measuring the energy absorbed from electrons, protons, and gamma rays, an estimate of the dose absorbed by other electronic devices on the same vehicle can be made. The dosimeters main features are listed in table 7.3.

Mass	**Volume**	**Power**	**Data**
Mass: 20g	Envelope: 35x25x10mm	Power consumpt.: 280mW Electric I/F: 10mA at 13-40VDC	Measures up to 40krads Data rate: 1Byte/s

Table 7.3: Radiation micro dosimeter features

The dose of radiation accumulated on a system will depend on the shielding capability of the material used to shield, as discussed in section 6.1.3.4. The shielding effectiveness depends on the chemical composition of the material (for example hydrogen is

very efficient shielding and therefore materials with high hydrogen concentration shall be preferred), and according to this, very different masses of shielding could be needed, to meet the requirements on the maximum absorbed dose, while considering different materials.
In the CubeSats mission here discussed, two different materials are envisaged to be implemented and tested, through dosimeters measurements: Kevlar and High Density Polyethylene (HDPE), which indeed have good shielding performances.
As final configuration, three dosimeters are envisioned, positioned in three different spots. Two of them are coupled with Kevlar and HDPE covers, in order to measure the shielding capabilities of the two materials. In particular, it is assumed to have two equal tiles having a thickness of 20mm for both materials (each tile is 50x50x20mm, which corresponds to 72g for Kevlar and 48g for polyethylene).
A NanoCam C1U [38] is finally envisaged to take pictures of the Sun. It is a high performing camera system fitting a single unit cubesat, based on a CMOS technology. Its main features are listed in table 7.4.

Mass	Volume	Power	Data
Mass: 170g	Envelope: 96x90x58mm	Power consumpt.: Idle: 360mW Image acquisition: 634mW Image process: 600mW Supply voltage: 3.3V	Measures up to 40krads Data rate: 1Byte/s

Table 7.4: NanoCam C1U features

7.1.4 Technological challenges

The enabling technologies for this kind of mission mainly regard the solar sail control and navigation, deep space tracking and telecommunications.

7.1.4.1 Solar sails

In the last decade the possibility to execute maneuvers without requiring propellant, but exploiting an unlimited source like the solar radiation pressure, aroused more and more interest in the field of solar sails. A solar sail cancels the dependency of the mission duration on the amount of propellant stored on board and has the further advantage of

providing a continuous thrust. Unfortunately solar radiation pressure represents at the same time the advantage and the drawback of this propulsion system, since it limits the available thrust to very small ranges. The real challenge for the CubeSats mission is not just using a solar sail, but a small solar sail, since the provided thrust depends on the sail surface area and the mission restrictions on sizes and volumes considerably limit the sail dimension. For this work solar sails with characteristic acceleration a_c = 0.01 mm/s and a_c = 0.05 mm/s^2 are taken into consideration. For each value the corresponding sail mass and size are investigated and the results are briefly discussed hereafter.
The area A of the sail can be evaluated through:

$$a_c = \frac{2\eta P A}{m} \tag{7.1}$$

where m is the total CubeSats mass, η is the sail efficiency and P is the solar radiation pressure. Making use of the ideal solar sail assumption and of the CubeSats mass requirement, it results to be $\eta = 1$ and $m = 8kg$. Table 7.5 resumes the required sail areas and the corresponding side lengths when a squared sail is supposed to be used. Once the area is known, the total sail mass m_s (i.e. the mass of the sail film plus

a_c [mm/s^2]	Area [m^2]	Side [m]
0.01	8.59	2.93
0.05	42.95	6.55

Table 7.5: Sail dimensions

the mass of the sail structure) can be evaluated from the definition of the sail loading, which quantifies the structural design's performances:

$$\sigma_s = \frac{m_s}{A} \tag{7.2}$$

Considering a value of 20g/m^2 for the sail loading, the resulting masses are those reported in table 7.6. As stated above, the optimal trajectory is found for a timeframe

a_c [mm/s^2]	Mass [g]
0.01	172
0.05	859

Table 7.6: Sail mass

of 2T, where T denotes the period of the Halo orbit used as initial guess. For each value of the characteristic acceleration, tests are conducted using Halos with z-axis amplitude $A_z = 250000$ km and $A_z = 350000$ km as initial guess. An optimal trajectory obtained with $a_c = 0.01$ mm/s^2 and $A_z = 250000$ km is shown in figure 7.2. The reference frame Oxyz is used, but for easy viewing the origin O and the Sun are not included in the figure.

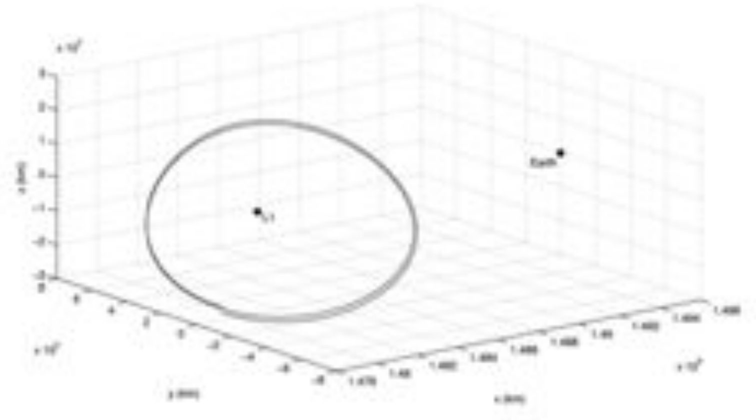


Figure 7.2: Optimal trajectory obtained with $a_c = 0.01$ mm/s^2 and with a $A_z = 250000$ km Halo

7.1.4.2 Communications

When sizing the communications subsystem for a spacecraft travelling very far from Earth, one of the issues to be faced is the choice between radio frequency (RF) and optical communications. As a matter of fact, laser communications offer many advantages over RF systems. Most of the differences arise from the very large difference in the wavelengths, which at RF are thousands of times longer than at optical frequencies. Optical crosslinks are interesting because they can support higher data rates than RF using relatively small antennas diameters, resulting in lower system masses. On the other hand, laser communications typically use narrow optical beams and therefore they are difficult to acquire and point accurately, requiring more complex pointing mechanisms.

Due to the long distance and the small CubeSats standard sizes, optical communication is to be preferred to enable very compact, low power uplink/downlink over interplane-

tary distances and to allow a good scientific data transfer capability to Earth.
In order to select the most suitable configuration for the CubeSats mission, a trade-off is performed to compare the RF solution and the optical one, and in particular, the first step of the analysis focuses on the evaluation of the link budget.

RF link budget

The link budget equation in its most general form is (expressed in dB):

$$SNR_{avail} = \left(\frac{E_b}{N_0}\right)_{avail} = P_{TX} + G_{TX} + \frac{G_{RX}}{T} - L_{fs} - L_{other} - k - R_d \tag{7.3}$$

where

- SNR_{avail} is the available signal to noise ratio on the link
- $(E_b/N_0)_{avail}$ is the available energy-per-bit to noise power density ratio (analogous of SNR_{avail})
- P_{TX} is the transmitted power
- G_{TX} is the transmitting antenna gain
- G_{RX} is the receiving antenna gain
- T is the system noise temperature of the receiver
- L_{fs} is the free space signal loss
- L_{other} is a term accounting for the other link losses (antenna pointing loss, rain, atmospheric absorption and implementation)
- k is the Boltzmann constant (-228.6 dBW/K·Hz)
- R_d is the data rate

The link margin can be computed as:

$$M = \left(\frac{E_b}{N_0}\right)_{avail} - \left(\frac{E_b}{N_0}\right)_{req'd} \tag{7.4}$$

where $(E_b/N_0)_{avail}$ is the SNR available at the receiver and $(E_b/N_0)_{req'd}$ is the SNR required to achieve a given BER. $(E_b/N_0)_{req'd}$ is a function of the modulation format and the presence of forward-error-correction coding.
The link equation 7.3 is used for sizing the transmitting antenna to be foreseen on

the satellite in order to meet the requirements in terms of data to be sent to Earth: specifically the data rate required for the present mission is 400kbps.

Hereafter, the description of how to compute the parameters included in the link equation is reported.

Transmitter power

For the present CubeSats mission, a reference value of 1.8W is assumed as output power of the transmitter. This value corresponds to an input power to the transmitter of 20W*, as obtained from the graph reported in figure 7.3. In particular, for this

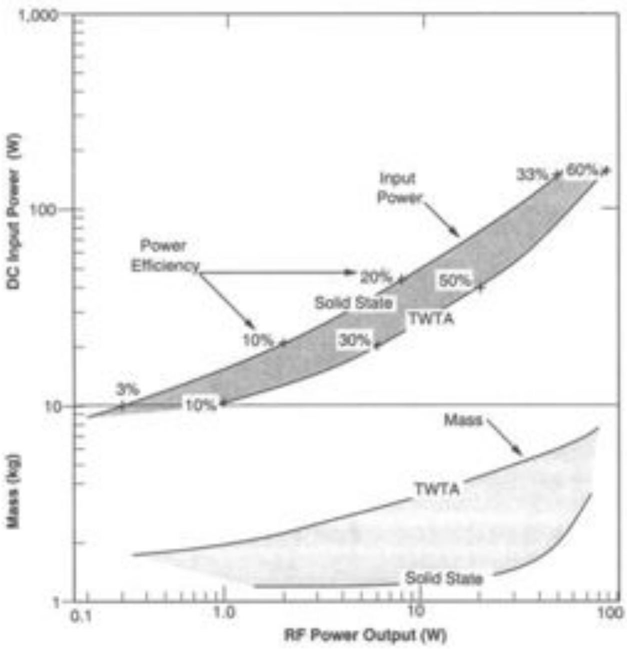


Figure 7.3: RF Transmitter power

application a solid state transmitter is assumed, which is usually considered for output power up to 5-10W; moreover, solid-state amplifiers are more reliable than the traveling wave tube amplifier, mostly because they require lower voltages.

Transmitter gain

The gain of the transmitting antenna can be expressed as:

$$G_{TX} = \eta \left(\frac{4\pi A_{TX}}{\lambda^2} \right) \tag{7.5}$$

*This reference value is assumed according to a preliminary estimation of the available power on the CubeSats system, provided by solar cells covering the spacecraft.

where

- η is the antenna efficiency (typically 0.55 for parabolic-dish antenna)
- λ is the carrier wavelength, related to the carrier frequency f as $\lambda = c/f$, with c the speed of light (3x10^8 m/s)
- A_{TX} is the antenna aperture area, which is $\pi D_{TX}^2/4$ for parabolic-dish antenna having circular aperture of diameter D_{TX}.

Expressing the transmitter gain in terms of carrier frequency and antenna diameter, the formula for the gain becomes, in dB:

$$G_{TX} = const + 10 \log \eta + 20 \log f + 20 \log D_{TX} \tag{7.6}$$

where the constant depends on the units used for the carrier frequency and the antenna diameter: it is equal to 20.4dB if expressing the frequency in GHz and the diameter in meters. This formula of the transmitter gain is used to determine the diameter of the antenna.

Receiver figure-of-merit

Usually the ratio of receiver gain to effective system temperature (G/T) is referred to as receive figure-of-merit.

Analogously to the transmitter, the gain of the receiving antenna can be expressed as:

$$G_{RX} = \eta \left(\frac{4\pi A_{RX}}{\lambda^2} \right) \tag{7.7}$$

where

- η is the antenna efficiency (typically 0.55 for parabolic-dish antenna)
- λ is the carrier wavelength, related to the carrier frequency f as $\lambda = c/f$, with c the speed of light (3x10^8 m/s)
- A_{RX} is the antenna aperture area, which is $\pi D_{RX}^2/4$ for parabolic-dish antenna having circular aperture of diameter D_{RX}.

Expressing the receiver gain 7.7 in terms of carrier frequency and antenna diameter, the formula for the gain becomes, in dB:

$$G_{RX} = const + 10 \log \eta + 20 \log f + 20 \log D_{RX} \tag{7.8}$$

where the constant depends on the units used for the carrier frequency and the antenna diameter: it is equal to 20.4dB if expressing the frequency in GHz and the diameter in meters.

For the present calculations, the reference receiving antennas on ground are those of the NASA Deep Space Network (DSN), which are parabolic antennas working in Ka-band ($f = 32GHz$) with a diameter of 34m and an efficiency of 0.494. The obtained receiver gain is 78.07 dB. The system noise temperature for the DSN antennas is T=196.112K [39].

Free-space propagation loss

The propagation loss for a signal in free space is a function of distance squared. Specifically, it can be expressed as:

$$L_{fs} = \left(\frac{4\pi S}{\lambda}\right)^2 \tag{7.9}$$

where

- S is the link range, that is the distance from source to destination;
- λ is the carrier wavelength, related to the carrier frequency f as $\lambda = c/f$, with c the speed of light ($3x10^8$ m/s)

The free-space loss equation can also be expressed as (in dB):

$$L_{fs} = const + 20\log f + 20\log S \tag{7.10}$$

where the constant depends on which units are used for the carrier frequency and the link range. Specifically it is equal to 92.45 dB while expressing the frequency in GHz and the distance in km.

For the considered case, the carrier frequency is 32GHz (Ka-band transmission) and the link distance is $1.5x10^6$ km (distance between Earth and the first Earth-Sun Libration point): the resulting free-space propagation loss is 246.07 dB.

Other losses

This term includes additional loss contributions which are:

- rain and atmospheric absorption loss,
- antenna pointing loss.

The first term for the DSN is assumed -4 dB, referring to the data given by a statistical model assuming cloudy and rainy condition (worst case) and an elevation angle of 10° [39]. The antenna pointing loss is assumed equal to -2dB [40].

Modulation

In order to obtain the required SNR, which represent the SNR needed to achieve a specific BER, it is necessary to select a specific modulation type. Indeed the $(E_b/N_0)_{reqd}$ is a function of the modulation format and the presence of forward-error-correction coding. The BER (Bit Error Ratio) represents the probability of bit error, which usually for data communications is required to be 10^{-5}-10^{-7}. In particular, assuming a $BER = 10^{-6}$ and Binary Phase Shift Keying (BPSK) modulation type, the required SNR is 10.5dB, as obtained from the graph reported in figure 7.4.

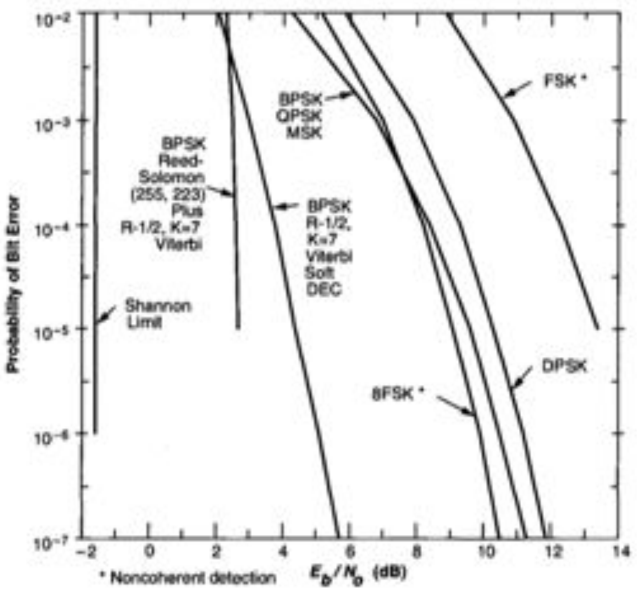


Figure 7.4: RF Modulation

Results

Using the link equation 7.3 with all the terms computed the way just discussed, the transmitting antenna gain is computed; it results 42.30dB. The antenna diameter obtained with this gain value is about **52cm**.

Optical link budget

The link equation for laser communication is very analogous to the link equation for any RF communication link. Starting with the transmit source power the designer identifies all sources of link degradation (losses) and improvements (gains) and determines the received signal level. Based on the background and receiver noise and the type of signal modulation which is to be detected, a required signal is generated. The ratio of the received signal to the required signal is the system link margin. Typical operating wavelength is 1550nm, which corresponds to a frequency f=193THz. The link equation can be simply written as [41]:

$$P_{RX} = P_{TX} \cdot G_{TX} \cdot L_{TX} \cdot L_S \cdot G_{RX} \cdot L_{RX} \tag{7.11}$$

where

- P_{RX} is the received signal power,
- P_{TX} is the transmitted signal power,
- G_{TX} is the effective transmit antenna gain,
- L_{TX} is the efficiency loss associated with the transmitter,
- L_S is the free space range loss,
- G_{RX} is the receive antenna gain
- L_{RX} is the efficiency loss associated to the receiver.

Hereafter, the description of how to compute the parameters included in the link equation is reported.

<u>Transmitter power</u>

Transmit power to be introduced in the link equation is the power out of the laser device. For the present calculation a transmit power of 500mW is assumed [42].

<u>Effective transmitter gain</u>

The effective transmit gain consists of three parts. The first contribution is the spatial distribution of energy in the far field, which is based on the aperture size (feed beam size), the near-field-energy profile and the wavelength of the laser cross-link system. The second part involves the off-axis loss factor due to pointing errors in the optical system. The third contribution is simply a geometric reduction of the far-field gain due to the wave front errors.

Transmit gain

The on-axis gain can be expressed as:

$$G_{TX} = \frac{4\pi A_{TX}}{\lambda^2}\left[\frac{2}{\alpha^2}\left(e^{-\alpha^2} - e^{-\gamma_{TX}^2\alpha^2}\right)\right] \tag{7.12}$$

where

- λ is the carrier wavelength, related to the carrier frequency f as $\lambda = c/f$, with c the speed of light (3x10^8 m/s);
- A_{TX} is the transmitter aperture area, which is obtained as $\pi D_{TX}^2/4$ with D_{TX} aperture diameter;
- $\alpha = D_{TX}/\omega$ (ω is the Gaussian feed beam waist diameter at the 1/e^2 point)
- $\gamma_{TX} = b_{TX}/D_{TX}$ (b_{TX} is the obscuration factor): for this case γ_{TX}=0.25 is assumed.

This equation defines the effective on-axis far field gain for circular apertures perturbed by obscuration and truncation effects. For $\gamma \leq$0.4, the parameter α can be computed as:

$$\alpha \approx 1.13 - 1.30\gamma^2 + 2.12\gamma^4 \tag{7.13}$$

The off-axis gain can be approximated as:

$$G_{TX}\left(\theta_{off}\right) \approx \frac{4\pi A_{TX}}{\lambda^2} e^{-8\left(\theta_{off}/\theta_{div}\right)^2} \tag{7.14}$$

where

- θ_{off} is the off-axis angle;
- θ_{div} is the 1/e^2 beam diameter.

For the present study it is assumed that no off-axis loss is present, and equation 7.12 is used for the overall transmit gain.

Pointing loss

The pointing loss can be expressed as:

$$L_{poin} = e^{-8\left(\theta_{poin}/\theta_{div}\right)^2} \tag{7.15}$$

where θ_{poin} is related to the pointing accuracy. For the present calculations a pointing loss of -6dB is assumed.

Wavefront loss

The wavefront loss associated with aberrations of the optical signal by the optical elements in the transmit path is obtained from the rms wavefront error. This error is modeled at the link level as a reduction in the on-axis intensity due to high-frequency “ripples” in the far-field intensity pattern. It can be expressed as:

$$L_{wf} = e^{-(k\sigma)^2} \tag{7.16}$$

where $k = 2\pi/\lambda$ and σ is the rms wavefront error given by λ/x, x=wavefront error. For the present calculations it is assumed null.

Free space range loss

Link range loss results from the diverging wavefront of the optical energy as it traverses the link distance. A convenient way to represent this loss is via the following equation:

$$L_S = \left(\frac{\lambda}{4\pi S}\right)^2 \tag{7.17}$$

where λ is the wavelength and S is the link distance. For the present case, at a distance of 1.5x10^6 km (distance between Earth and the first Earth-Sun libration point), the free space loss results -321.7dB.

Receive antenna gain

The effective collecting aperture of the receiver constitutes the receive antenna gain. The receive antenna gain is calculated from the collecting area of the antenna and the wavelength of the incident optical energy; it is expressed as:

$$G_{RX} = \left(\frac{\pi D_{RX}}{\lambda}\right)^2 \left(1 - \gamma_{RX}^2\right) \tag{7.18}$$

where

- D_{RX} is the telescope aperture diameter,
- λ is the carrier wavelength,
- $\gamma_{RX} = b_{RX}/D_{RX}$ (b_{RX} is the receiving telescope obscuration factor).

For the present study, the Hale telescope is assumed as reference receiver on ground. It has an aperture diameter of 5m. Considering $\gamma_{RX} = 0.36^*$, the resulting receiver gain

*This value is obtained referring to [42] where an obscuration of the ground telescope of 1.8m is considered.

is 139.5dB.

Received power

The ratio of received signal level (P_{RX}) to required signal (P_{req}) is the link margin. This is typically expressed in terms of dB and is calculated from:

$$M_{link} = 10 \log \left(\frac{P_{RX}}{P_{req}} \right) \tag{7.19}$$

For satellite cross-link, link margins greater than 3dB are typical (in the present study a 10dB margin is considered). This excess margin accounts for unexpected on-orbit link degradation that may occur due to larger than normal contamination or radiation levels or degradations in other areas of the system. The required power at the receiver is usually referred to as the sensitivity. It is the average optical power, needed at the input of the receiver in order to obtain a specific BER; it can be expressed as:

$$P_{req} = N_{RX} \cdot h \cdot f \cdot R_d \tag{7.20}$$

where

- N_{RX} is the average number of received photons per bit,
- h is the Plank's constant ($h = 6.656x10^{-34}Js$),
- f is the frequency,
- R_d is the data rate.

Assuming that the electronic amplifier and circuitry in the receiver are noiseless, the only source of noise to be considered is the "quantum noise". This noise is due to the fact that light has a certain "granularity", that means it is made up of "photons" of energy $h \cdot f$. It can be shown that due to quantum noise, the following expression holds:

$$BER = \frac{1}{2} e^{-2N_{RX}} \tag{7.21}$$

For typical optical systems, the BER values used in the sensitivity specifications range between 10^{-9} and 10^{-12}. In particular, by setting the target BER at 10^{-9}, the sensitivity in terms of photons per bit would result $N_{RX} = 10 photons/bit$. Actually, receiver electrical noise is orders of magnitude larger than quantum noise and therefore typical receivers sensitivities are larger than 10photons/bit. For the present work, a value $N_{RX} = 90 photons/bit$ is assumed. With this sensitivity, the power required amounts to -83.34dBm.

Results

Using the link equation 7.11 with all the terms computed the way just discussed, the transmitting antenna gain is computed; it results 93.86dB. The telescope aperture diameter obtained with this gain value is about **3cm**.

RF vs Optical trade-off

A summary of the comparison between the two systems' link budgets is shown in tables 7.7 and 7.8, which summarize the results obtained according to what previously described. The computations are performed considering a required data rate of 400kbps and a link range of 1.5x10^6 km (distance between Earth and the first Earth-Sun Lagrangian point). From the comparison between the two link budgets it results that the laser communications system needs a much smaller antenna, which will correspond to lower mass and easier integration requirements. Moreover, the required power is less for optical system.

RF system - Ka band	
Transmit power	1.8W 2.55 dBW
Frequency	32 GHz
Atmosphere loss	-4 dB
Antenna pointing loss	-2 dB
BER	10^{-6}
RX antenna diameter	34 m
RX antenna gain	78.07 dB
System noise	196.112 K
Link margin	10 dB
TX antenna gain	42 dB
TX antenna diameter	**52 cm**

Table 7.7: RF system link budget

Optical system	
Transmit power	500mW 27 dBm
Wavelength	1.55 μm
Frequency	193 THz
Pointing loss	-6 dB
Free space loss	-321.67 dB
RX antenna diameter	5 m
RX antenna gain	139.5 dB
RX loss	-3 dB
Sensitivity	90 photons/bit
Link margin	10 dB
TX antenna gain	93.86 dB
TX antenna diameter	**3 cm**

Table 7.8: Optical system link budget

Besides link budget considerations, to conduct a realistic trade study of RF versus laser communications, other important characteristics or factors must be identified and included in the trade [41]. In the present work the following parameters are considered for the trade-off (some of them are only qualitatively evaluated):

- mass;

- power;
- cost: the lifecycle cost includes two contributions, that are development, or non recurrent cost, and recurring costs; the development cost would be higher for laser communications, but recurring costs would be lower (overall RF are preferable);
- integration impact: it includes several factors that denote the overall effect of integrating a communications system:
  - volume needed to allocate the system (mainly related to size),
  - field of view: the requirement to provide a clear view throughout a range of angle is more stringent for RF systems due to larger antennas,
  - need to stow and deploy the antenna,
  - dynamic reaction effect (related to deployment operations),
- technical risk: it includes parts availability and level of space qualification, development and testing.

The results of the comparison are shown in table 7.9. As overall result of the trade-off,

	Mass	**Power**	**Cost**	**Integration impact**	**Technical risk**	
Weight [%]	*23*	*10*	*25*	*20*	*22*	
RF	-1	-1	1	-1	1	**-0.06**
Optical	1	1	-1	1	-1	**0.06**

Table 7.9: RF vs Optical communications trade off

the optical communications turns out to be the best solution. Moreover, optical communication could be critical for the required antenna pointing, but it is not too much more challenging than the RF case.
It is also worth underlining that one of the main objectives of the proposed CubeSats mission is to provide a platform for test and validation of advanced technologies. According to this the choice of implementing laser communications is even more significant.

7.1.5 Interplanetary CubeSats mission: conclusions

The problem of cost reduction is a significant driving factor in advancing space technologies, and it mainly involves two main points, that are the miniaturization or mass and power reduction of platform and instruments, and the implementation of new launch

strategies, mission planning and use of ground network to reduce the cost. These issues are important not only for extremely small satellites, but are significant for any bigger spacecraft, as a reduction of the mission cost is always desirable.
According to this, the interest in small satellites, and in particular CubeSats, is growing up, as they can represent valuable platforms both for scientific and technological scopes, with lower costs than big satellites. In particular a mission like that discussed in this thesis would represent a good opportunity to improve capabilities in the exploration of the solar systems, pursuing both scientific and technological objectives, foreseeing sun observation and plasma measurements, as well as advanced technologies demonstration (e.g. optical communications, solar sails), in view of their future implementation on larger spacecraft.
It is worth underlining that no specific evaluations have been performed to analyze the transfer phase, to the libration point. The study has focused on the preliminary design of the CubeSats system, starting from the assumption that an external transportation system is in charge of satellite delivery to its final destination. Further analyses shall be devoted to investigate different options for the transfer phase and select the best solution according to the constraints deriving from the objectives of such a mission.

7.2 Alternative approach for space exploration

This section provides an overview of a “Conceptual Scenario for a Global Approach to the Space Exploration Initiatives”. The resulting “Long Term Vision”, depicted here, has to be considered a kind of “A Dream for the Future”. It is constructed around an extremely theoretical and almost utopian scenario that takes into consideration non-conventional solutions, whose validity turns out to emerge only if the problem of the Space Exploration is approached on a very large scale basis and over quite long periods of time. In particular, the followed approach is aimed at bringing in evidence the benefits of highly integrated solutions applied to wide generalization of the exploration missions planning problems. Moreover, the theoretical approach hereafter described leaves out of consideration whatsoever limiting economical constraints and is not confined by the readiness of the technological developments needed to realize it. It is just, as said, a “Study Case”.

7.2.1 Strategy

The exploration of space has so far been attempted mainly through a limited sequence of missions, not strictly linked among them in terms of accumulation of achieved experience and hardware utilization. There are, nevertheless, a few examples of global exploration roadmaps that attempt to face space exploration, and particularly human space exploration, according to a rational plan, made up of an orderly sequence of different destinations [18, 43]. Among these studies we propose an innovative approach for the exploration beyond Low Earth Orbit, which foresees a sequence of interlinked missions, targeted to specific locations in space, where human outposts are put in place with the aim to progressively enlarge the boundaries of human presence in the Solar System. Each New Human Outpost, established at a specific location, is built "on the shoulders" of the previous one, through the physical transfer to the new next location of the elements of the previous outpost itself. The final architecture of the new outpost is properly re-adapted to comply with additional mission requirements deriving from the new location, through the aggregation of dedicated new elements. In some specific cases, if the outpost established at a given location continues to hold a strategic relevance for the completion of the overall exploration program, such an outpost can be not dismantled but left in place and a kind of recurrent unit, made with replica of the same elements, can be used for the next location. In essencem, just one "Itinerant Human Outpost", growing eventually in complexity and transforming itself, if needed, is utilized to perform a "Fantastic Journey" that touches, in a succession of different times, various locations. The basic architectural elements, that form the configurations of the various Human Outposts, are therefore, as much as possible, the same, even if they are used to build, at every selected different location, somehow different architectures. At each step of the journey the outpost, in addition to accomplish its mission, is utilized as technology and operations test bed, to prepare the "Next Step". The very interest in this kind of approach is that in the end it is expected to reduce the overall cost of the complex set of missions. In the front end such an approach requires more complex and costing solutions, because the design must take into account since the beginning very challenging requirements to guarantee the re-use in different destinations of the outpost elements that must therefore have long operative lives, and must be repairable, refurbishable and reconfigurable in orbit. Of course this requires larger design and development efforts in the front end. On the other hand, this approach allows the repeated utilization of the various architectural elements that implies fewer elements to be designed, developed and built. If looking at the whole scenario of exploration, including several missions to different targets, this would allow reducing the overall cost of the complex set of missions.

7.2.2 Scenario

The "Sequence of Locations" is initiated with a Human Outpost, positioned in an Equatorial Low Earth Orbit (ELEO) that is serviced mainly from equatorial launching bases that later on will support the entire exploration architecture. The deployment of the human outpost in LEO can be seen as the first logical step in a gradual path for the solar system exploration. Indeed, this would represent the closest destination to Earth and moreover a good knowledge of its environment has already been gained. Furthermore, having an equatorial orbit would guarantee an easier and less costing accessibility from Earth, relying on equatorial launching bases to be exploited even in following missions of the whole exploration program. The next destinations in the path of exploration are Earth-Moon Lagrangian points, from where the low lunar orbits and the near Earth asteroids region can subsequently be reached. As a matter of fact the Lagrangian points allow for low deep space accessibility costs, and therefore they are interesting locations where to depart from to accomplish missions towards further destinations, such as asteroids.

An outpost deployed in one of the Lagrangian points would provide a platform for the test of specific technologies, not testable in LEO (e.g. space radiations protection systems), without moving too far and with the advantage of an "easy" accessibility to/from Earth. Moreover, a cis-lunar infrastructure would allow more extensive science return from lunar robotic surface exploration, which is to be considered mainly for the exploitation of the resources available on the Moon. As a matter of fact, products obtained from ISRU activities on the Moon can be used as propellants for the next steps of the journey. From the cis-lunar regions (from Lagrangian Points) martian orbits will then be next attained. Martian Human Outposts are envisaged to be located in Low Mars Orbits (LMO) or on one of the Mars natural satellites: Phobos or Deimos. The outpost located in Mars orbit is foreseen to support manned operations

on the surface. The outpost can remain in orbit, while the astronauts spend some time on the surface performing specific exploration activities. One of the two Mars satellites can be considered as additional destination before the human expedition to Mars surface. In this case from the outpost deployed on one of the Mars satellites, tele-operations of robotics on the surface can be performed. As already introduced, descent missions on some asteroids, on the Moon and on the Mars surface, performed from the related nearby orbital locations where bases have been installed, complete the Scenario of what can be considered the “Grand Tour of the Earth Neighbors”, performed by the “Itinerant Human Outpost”.
A pictorial view of the target destinations of the “Itinerant Human Outpost” is reported in figure 7.5.

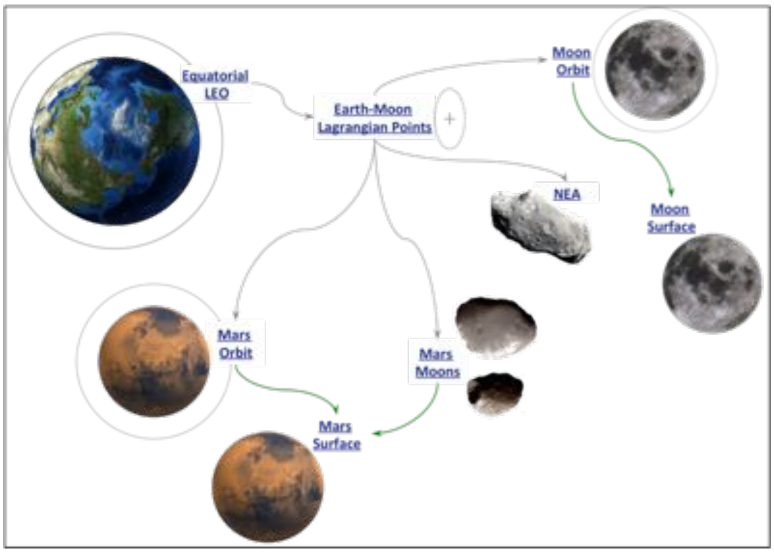


Figure 7.5: “Itinerant Human Outpost” destinations

The grey arrows in the picture indicate the transfers of the outpost to different destinations through the solar system. The green arrows indicate that the outpost does not perform the maneuver itself, i.e. it does not land on the surface of the Moon or Mars, but it remains in orbit to support the surface operations performed with other modules.
According to the scenario just described, the “Itinerant Human Outpost” is envisaged as an infrastructure, which travels through several destinations and supports different

missions. Therefore the design of such infrastructure shall be carried out taking into considerations the various issues related to the different destinations missions.

7.2.3 Itinerant human outpost: step-by-step approach overview

To support the "Grand Tour of the Earth Neighbors" the recourse to "Innovative Architectural Concepts" is needed. In designing each element of the "Itinerant Human Outposts", starting from the first one in Equatorial Low Earth Orbit, its successive multiple utilization and growth potential have to be properly taken into account. The possibility to benefit from the "Commonality" of hardware and software among the various elements, representing the Building Blocks (BB) of the architectures, has to be accounted for, since the early phases of the design. Whenever possible "Modularity" has to be extensively adopted in the design at all levels from parts, components, units, and subsystems to systems. The practice of introducing in the design the "Revolutionary Approach" of the "6R Space Systems", that is to say of the Repairable, Refurbishable, Replaceable, Reconfigurable, Retrievable and Reusable Space Systems, has to be as well adopted in order to optimize in the long term the costs of the outposts building up and operations. This means that these innovative space systems have to be:

- repairable in orbit and, eventually, on the celestial bodies surface, i.e. suited to be subject to ordinary and extraordinary maintenance and repair during their extended life;
- refurbishable through supply of consumables and perishable goods at planned intervals to renew the possibilities to continue the operations;
- replaceable, that is to say suitable to replacements over the years to accommodate parts more technologically advanced, capable therefore to accept with time more updated and improved high-tech products;
- reconfigurable, that is they shall be capable to accommodate in the course of their life also substantial changes of configuration and of mission, through the addition of new on board apparatuses and new payloads;
- retrievable, that is they shall be in condition to be returned to the Space Base from where have departed, for major changes to be introduced;
- reusable over and over again in multiple missions starting from the bases where they are housed to reach new destinations.

In the building-up of the various outposts' configurations, elaborated and innovative "In Orbit Assembly and Construction Techniques" shall be adopted, relying on Astronauts' Extra Vehicular Activities, largely adopted since the initial missions, and on the support, for the more complex activities, of "Dedicated Multipurpose Devices" that are aimed at facilitating the operations of construction. Also the activities of "Inspection", "Maintenance" and "Repair", needed to ensure a safe and long life to the "Inhabited Outposts", are to be eased by the combination of dedicated EVA and robotic systems. Consideration has to be given to the unusual situation of "Disassembling" of the elements requested for rapid changes of configuration occurring during the outpost growth and the transfers to the next locations. In the design of these space systems of next generation, it is important to rely on "ISRU", approach that moves the centre of the Construction, Operations and Logistics of the "Itinerant Human Outposts", from the Earth into space. The natural resources that can be found on the surface of the Moon, on the asteroids and on the surface of Mars have since long time been postulated to be useful for supporting the manufacturing and construction of the advanced space systems to be used for the intensive exploration of the Solar System; the new generation "Innovative and Advanced Space Systems" have to be largely based, especially for what concerns the consumables, on these extra-terrestrial resources. In particular propellants (Liquid Oxygen and Liquid Hydrogen) needed to support orbital maneuvers of the "Inhabited Outpost" and to guarantee the logistic connections have to be produced from materials found in place and stored in orbit in particular strategic locations. In this regard the outpost deployed in the EML points represent a favorable place where to collect and store propellants produced and retrieved from the Moon's surface to be used for further missions. The "Global Architecture" has to be conceived with an high degree of "Autonomy" not only in terms of resources, but as well "Command and Control Functions", with all the outposts working to form a "Net in Space" strictly connected, almost independent of the Earth.

For the various destinations of the journey a preliminary assessment of the mission objectives to be accomplished and identification of the needed building blocks are presented. Specifically, some details about the configuration of the outpost for each destination are reported in the following sections. However, the building blocks here described are defined at functional level, and the figures reported in the paper do not refer to specific design and sizing, but mainly to their functionalities. Besides the missions which will be discussed in the following sections, a certain number of additional missions are part of the whole scenario. Specifically the following types of mission will complete and support the journey:

- logistics missions, to resupply the station (e.g. resources);
- crew missions, which periodically are needed to perform experiments and maintenance as well as to outfit and reconfigure the outpost for the next destination;
- unmanned missions, to bring to the outpost additional modules, including transportation modules necessary for the outpost transfer towards other destinations;
- precursor robotic missions, aimed at bringing to a specific destination different elements to support the human exploration activities.

It must be underlined that the outpost is to be conceived and designed starting from a "small and easy" concept, but always taking in mind that it shall evolve in a "larger and more complex" configuration to include new elements, eventually needed for the following destinations missions. This means that a great level of flexibility is needed to reconfigure and adapt the outpost to the objectives of the following destinations missions. Being the outpost conceived as a reconfigurable spacecraft and being it not permanently inhabited, a great level of autonomy is needed for its modules. For example, it is very important to have automatic Rendezvous and Docking capability, especially for the docking/undocking operations of modules when the outpost is uncrewed. Furthermore, the outpost reconfiguration maneuvers are quite challenging and it is very likely to need a robotic system (like a robotic arm) to support these operations. This becomes particularly important for modules that nominally do not have an own propulsion system and for which berthing maneuvers can be envisioned. In addition, a robotic arm would be a very helpful support for the external maintenance activities.

7.2.3.1 Step 1: Equatorial Low Earth Orbit

The first step in the journey, as previously addressed, is an Earth equatorial orbit. In its initial operative life, the outpost placed in LEO shall be seen as an infrastructure to perform some research activities, as well as test of technologies to be used for the

following journey's steps . According to this, it can be conceived as a men-tended platform periodically visited by astronauts, to perform experiments as well as maintenance activities. Moreover, the visits of the crew shall be necessary to outfit and prepare the outpost to move to the following destination.
A schematic overview of the outpost configuration in its first step is shown in figure 7.6.

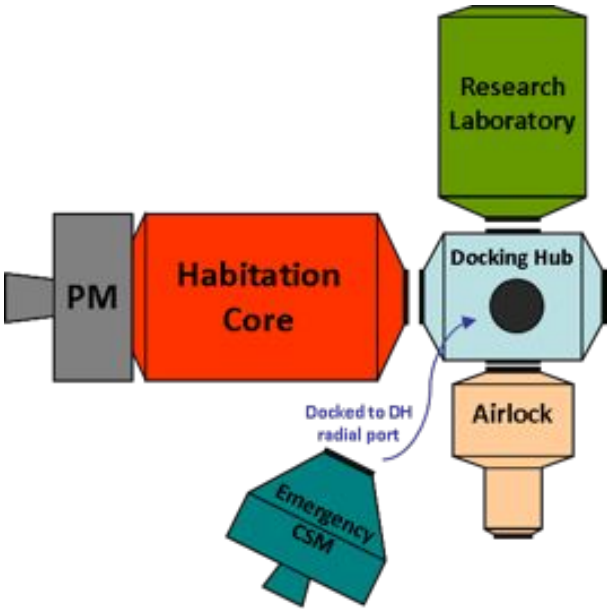


Figure 7.6: Equatorial LEO outpost configuration (not in scale)

The outpost shall include a habitation core, which at the beginning can be a single module, compatible with short permanence of the crew on board. Attached to the habitation module a propulsive module will be needed, mainly for orbit and attitude control. This module, which for this phase has limited capabilities, will not be enough for the following steps and different transportation modules may be envisaged to accomplish more demanding maneuvers. Anyway, the propulsion module will be part of the outpost during all the duration of the journey and therefore shall be refueled during its lifetime to execute additional maneuvers.
Another module to be included as part of the initial configuration is a research laboratory where to perform experiments and tests. It would be worth to have the lab as a separate element, since in this way it can be easily released as soon as it is not useful anymore according to the new destination's objectives.

An airlock is to be included as well, since it will be necessary to perform some external maintenance activities, with astronauts in EVA. This aspect becomes more and more significant, if consider the next steps of the journey, which will be characterized by long mission durations and distant trips.
In order to give the outpost a flexible architecture, the idea of introducing a docking hub in the overall spacecraft is to be considered, to provide several docking ports and guarantee to easily attach a "new" module as well as release an "old" one (not needed anymore).
Another crucial aspect is related to the necessity, for safety reasons, to have always available in the outpost a re-entry vehicle, specifically a capsule with attached its service module, to face any emergency situation (for example, in the case of failure occurrence to the capsule bringing the crew to the outpost). The Capsule and Service Module (CSM) system is docked at one of the radial ports of the docking hub, leaving only another free radial port.

7.2.3.2 Step 2: Earth Moon Lagrangian Points

The second step in the "Fantastic Journey into Space" is the cis-lunar space, and in particular one (or both) of the Earth-Moon Lagrangian points. At this level of the study, no selection between the two points has been done, that will be dependent on specific objectives still to be characterized for this destination. It can also be thought to "explore" both the two locations, in order to accomplish different exploration objectives (e.g. move to EML2 to support exploration activities on the far side of the Moon). To transfer the outpost in the new location, a specific transportation module shall be foreseen (no specific considerations are reported about this point).
Moving to this destination allows implementing, testing and validating specific technologies, not needed or only partially applicable in the LEO environment. One of the most significant examples is related to the space radiations issue. As a matter of fact, the environment beyond LEO, and specifically outside the Van Allen belts, becomes very critical for the exposure to space radiations. Therefore it will be mandatory to introduce in the outpost a dedicated shielding to protect the crew. A dedicated shelter shall be part of the overall spacecraft to provide a "safe heaven" in case of SPE occurrence. This module is envisioned to be docked to the last free radial port of the

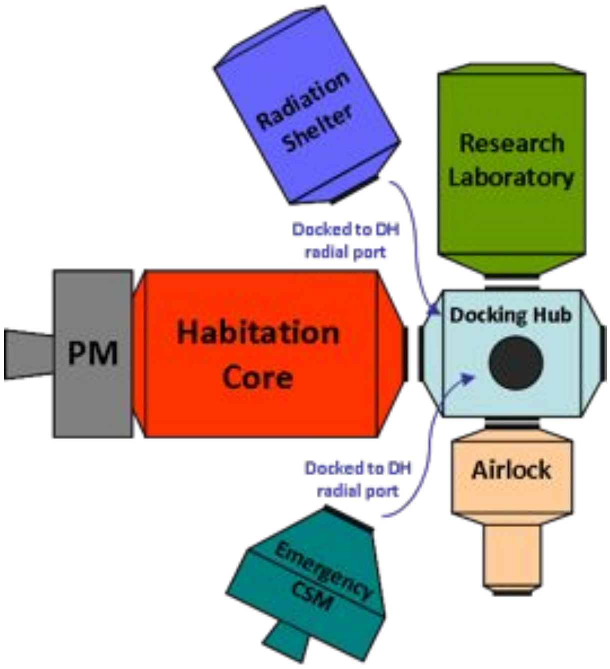


Figure 7.7: Earth-Moon Lagrangian Points outpost configuration (not in scale)

docking hub (see figure 7.7).

The outpost placed in cis-lunar still offers the possibility to perform some research activities as well as test of technologies. For this reason, the research laboratory module is moved to the Lagrangian point together with the remaining modules composing the outpost. One of the interesting tests that can be done in this second location is teleoperation of robotic assets deployed on the Moon's surface, which are assumed to be delivered to the target destinations, before human expeditions (some of the assessments about the outpost rely on this assumption).

The cis-lunar destination is also of interest for the training of the astronauts in a significant environment, for example for specific EVA operations. During the phase in cis-lunar, the outpost is still a men-tended infrastructure, and periodic missions are to be envisaged for the resupply of the station, as well as for maintenance activities performed by the visiting crew. Moreover, these missions are necessary to outfit and prepare the station for the following step.

Once the outpost is ready to move to the following step, the journey continues towards the next destination.

7.2.3.3 Step 3: Low Lunar Orbit

The third step is the Moon Orbit. To prepare the outpost for the third destination, some new modules need to be included.
Another docking hub is to be added, since with the present configuration only one axial port is still available, while at least two docking ports are needed: one is needed to dock the Lunar Lander and another one is necessary for the docking of the capsule carrying the crew and to allow the astronauts to move to the habitation core.
The outpost reconfiguration maneuvers are quite challenging and autonomous operations capabilities are necessary, as already addressed. Not all the modules move to the LLO, and in particular the crew capsule and its service module, attached to the second docking hub, remain in the Lagrangian point waiting for the crew to come back. As a matter of fact, this capsule is not needed during the Moon expedition and, moreover, an emergency vehicle is anyway included in the outpost architecture. The outpost configuration when in LLO is schematically depicted in figure 7.8, while the modules left in cis-lunar are shown in figure 7.9. The additional building block with respect to

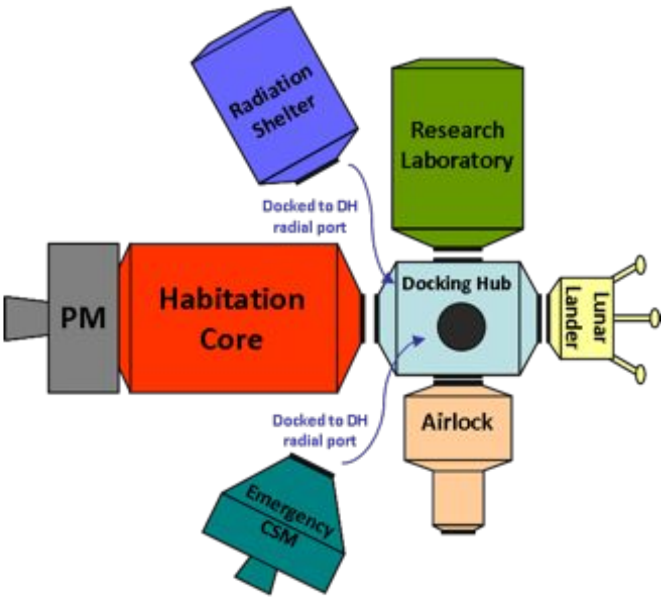


Figure 7.8: Low lunar orbit outpost configuration (not in scale)

the previous configuration is the lunar lander. Only one module is assumed to perform both landing and ascent maneuvers from the Moon's surface. The research laboratory

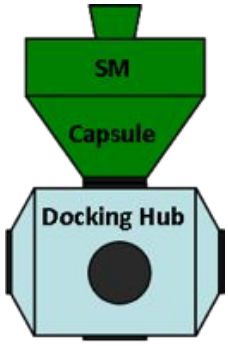


Figure 7.9: BBs "waiting" in cis-lunar during the Moon mission

gives the possibility to analyze the samples collected on the lunar surface before bringing them back to Earth for further analyses.
Some other modules to support the surface activities are assumed to be pre-deployed on the Moon with previous robotic missions. For example, one of the objectives of a mission on the Moon's surface is the production of propellants from ISRU activities. Accordingly, a dedicated ISRU plant can be deployed on the lunar surface prior to the crew arrival. The propellant produced on the Moon can be utilized to re-fuel the propulsion module, with the fuel needed for the maneuvers of the following steps. In particular, the propellants produced can be collected and stored in a dedicated depot, to be brought back and to have it available in cis-lunar for the refueling of the propulsion module at each new mission step. The module to collect the fuel is assumed to be brought to the Moon with a previous robotic mission, and it becomes part of the outpost only after having been filled with the propellant produced through ISRU activities. After the Moon's surface operations have been completed, the lunar lander is expended (not brought back to EML1/2), the fuel depot docks with the outpost and the outpost moves back to cis-lunar space. At this point, the crew transfers to the capsule, which was "waiting" in EML1/2, and re-enters on Earth.

7.2.3.4 Step 4: Near Earth Asteroid

A human mission to a Near Earth Asteroid (NEA) represents the next step in the "Fantastic Journey into Space" (missions that are considered as references are presented in [15, 44, 45]). The main objectives of this kind of mission are technological tests and

research (limited), but most of all the exploration activities in proximity and on the surface of the NEA. According to this, EVAs are to be performed, to explore the surface of the NEA and to collect samples to be brought back to Earth and analyzed.
A schematic overview of the configuration of the outpost for the NEA mission is shown in figure 7.10. Two new modules have been included in the outpost, which are a resources module and a health facility. As a matter of fact, due to the long duration of

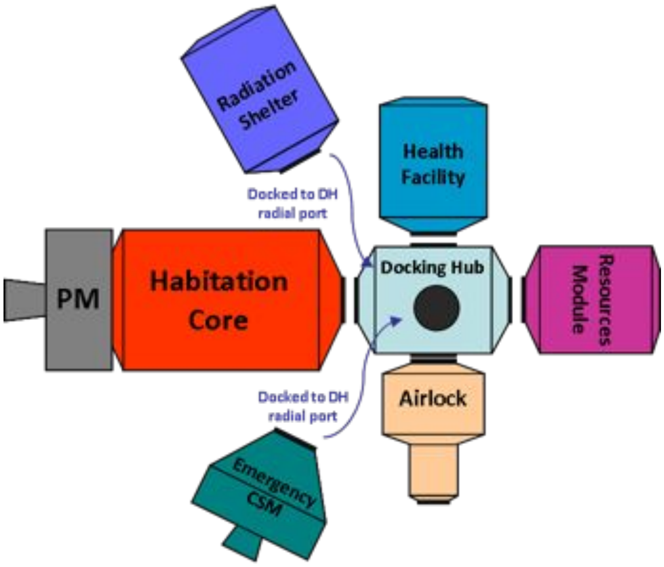


Figure 7.10: NEA outpost configuration (not in scale)

the mission additional resources are necessary for the crew and, given the impossibility to have logistic re-supply missions, a stowage module is included. Having a dedicated module for the resources rather than storing them in the main habitation core allows not limiting the free volume available for the astronauts. Moreover, this approach allows a good flexibility, since the resources module can be sized and organized according to specific mission requirements (e.g. depending on the selected target asteroid the mission duration can vary). Due to the long duration and the far distance of the NEA mission, a dedicated module for the health of the crew is considered. In particular, it has to be equipped with specific tools for the astronauts physical exercises to counteract the effects of prolonged microgravity exposure. Furthermore, medical equipment must be provided to face any emergency situation, including tele-medicine and in-flight surgery systems. Even in this case the outpost reconfiguration takes place in EML1/2, where the two additional modules are deployed and docked with the station. In particular the "health facility" is the first to be deployed, and in the period spent in cis-lunar, prior to departing for the NEA mission, specific tests and astronauts training are performed during the crew visits to the station. The configuration of the "waiting" modules left in cis-lunar is schematically shown in figure 7.11.

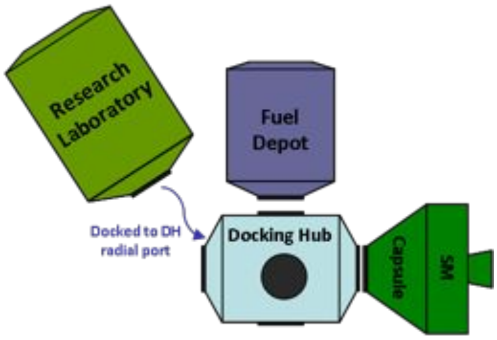


Figure 7.11: BBs "waiting" in cis-lunar during the NEA mission

7.2.3.5 Step 5: Low Mars Orbit

The fifth step envisages the transfer of the outpost to a Low Mars Orbit (LMO), to support the crew exploration activities on the Red Planet surface. The outpost moves to a LMO, where it remains for all the duration of the Mars exploration activities. A schematic overview of the configuration of the outpost as moved to LMO is shown in figure 7.12. No specific considerations about the elements necessary for the Mars surface activities are reported. In particular, for what concerns the surface elements (e.g. habitat, rover, ascent vehicle), it is assumed that they are deployed with a robotic mission (prior to the human expedition), and are already available on the surface. For what concerns the Entry, Descent and Landing (EDL) system, it has to be part of the human mission spacecraft, in order to allow the astronauts to perform the maneuvers to land on the red planet. However, the EDL has not been analyzed in details and it is simply considered as an additional building block to be part of the outpost (depicted with the dark blue BB in figure 7.12).

After Mars surface operations have been completed, the crew comes back to the outpost in LMO, the Mars elements are released and the trip back towards EML1/2 can begin. The configuration of the modules left in cis-lunar is analogous to the one described for

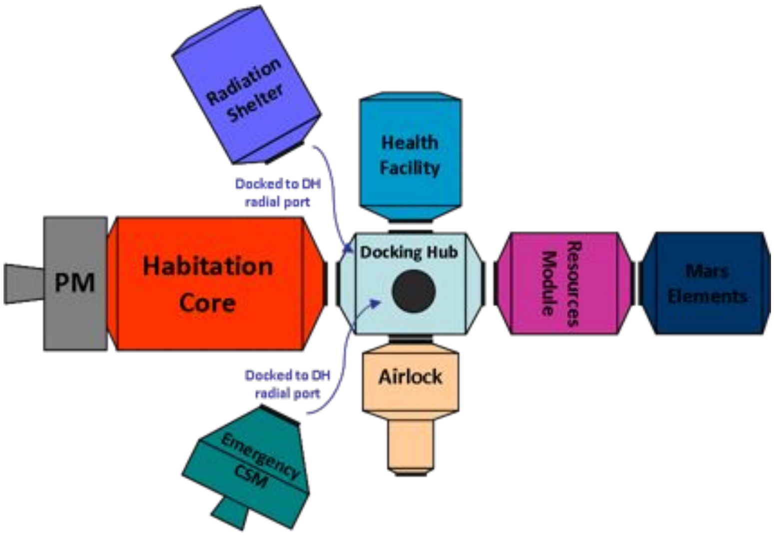


Figure 7.12: Low Mars orbit outpost configuration (not in scale)

the NEA mission step (figure 7.11). Once back at EML1/2, the crew transfers to the "waiting" capsule and re-enter on Earth. Even in this case, the resources module can be expended before the arrival in EML1/2, whenever it is not needed anymore.

7.2.3.6 Additional Step: Mars Moons

Another step can be included in the "Fantastic Journey into Space", which is a visit to one (or both) of the Mars Moons (Phobos or Deimos). A mission of this kind can be considered a preliminary step, before the human mission on the Mars surface, during which, tele-operations activities of robotic assets, pre-deployed on the Mars surface, are to be performed. A schematic overview of the configuration of the outpost as would be needed for a mission to a Moon of Mars is shown in figure 7.13.
In order to decide which of the two Moons is the most convenient target (or if both are to be included), further trade-off analysis shall be performed, according to the peculiar requirements identified for this destination (e.g. level of coverage of the Mars exploration sites, communications, ...). Both the Moons are quite small bodies and therefore, the systems for the surface operations (landing/anchoring systems) are similar to what would be needed for an asteroid. In this contest, no specific analysis about these elements has been carried out, but they are taken into consideration simply as

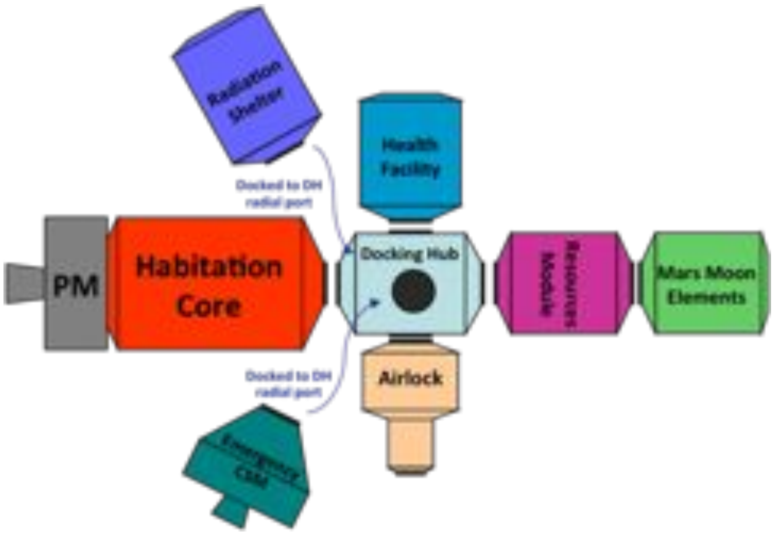


Figure 7.13: Mars Moon outpost configuration (not in scale)

an additional building block to be part of the outpost (depicted with light green BB in figure 7.13).

Moreover, the robotic assets placed on the Mars surface for surface exploration activities, which are to be tele-operated from the Mars Moon, are assumed to be already deployed on the surface of Mars through a previous robotic mission. After the planned exploration operations have been completed, the Mars Moon elements are released and the trip back towards EML1/2 can start. The configuration of the modules left in cis-lunar is analogous to the one described for the NEA mission step (see figure 7.11). Once back at EML1/2, the crew transfers to the "waiting" capsule and re-enter on Earth.

Analogously to what said for both NEA and LMO missions, the resources module can be expended before the arrival in EML1/2, whenever it is not needed anymore.

7.2.4 Itinerant human outpost: conclusions

The described "Itinerant Human Outpost" is conceived as an infrastructure to be used for the exploration of multiple targets, starting from a close location, that is an Equatorial Low Earth Orbit, arriving, as final destination, to the surface of Mars. The study here discussed focuses on the identification of the architectural configuration of the outpost as needed for the various destinations, part of the exploration scenario. The most important Building Blocks to be included into the station are identified, according to the main objectives and constraints deriving from the environments that the outpost has to withstand. The missions architectures and the related building blocks assessment is the result of a functional approach analysis, and no specific sizing is performed. All the main functions necessary for the outpost translate in a certain number of needed building blocks. By the way, the possibility to combine more BBs in only one module should be further investigated, in order to simplify the overall architecture. Just as an example, the resource module could be combined with the radiation shelter building block. The reconfiguration of the outpost, to make it comply with the new destination requirements, is performed relying on dedicated logistics and unmanned missions for the resupply of the station and the deployment of new modules in cis-lunar. In addition, periodic crew visits are envisaged to perform and support the reconfiguration operations.

The "Itinerant Human Outpost", as just described, represents an alternative approach for space exploration, with respect to what assessed in previous chapters, and in particular to the reference HSE scenario, built according to what described in chapter 4. However, the approach followed in the definition of the "Itinerant Human Outpost" missions and configurations, is perfectly in line with the philosophy behind the entire thesis work, that is to follow a stepwise approach in the human exploration of solar system, through successive destinations' missions, characterized by gradually improved complexity.

The main difference with respect to the reference HSE scenario described in previous sections is due to the reusability of the same modules; indeed, while the reference HSE scenario relies on the use of many recurrent units of modules, designed to accomplish more and more demanding requirements, with the "Itinerant Human Outpost", there is an initial core, including elements needed in all destinations, which is completely reused.

The rationale behind this kind of approach is that in this way the overall cost of the complex set of missions shall be reduced due to the repeated utilization of various basic architectural elements. This means that the outpost is conceived to have a great flexibility, allowing an incremental assembling of more and more complex architectures to accomplish more challenging missions.

However, this means that higher reliability and extended maintenance activities will be necessary.

8

Conclusions

This thesis summarizes the major results obtained in the frame of the PhD research activities on the topic “Space Exploration Systems, Strategies and Solutions”.
The work has been aimed at the development of a versatile methodology for the definition and analysis of human space exploration scenarios and missions; furthermore a detailed analysis of innovative technologies has been carried out.
The major results are represented by a reference exploration scenario, which has been built and characterized through the identification and definition of missions, concepts of operations, architectures and associated building blocks, and a large database, which collects the most important innovative and enabling technologies and provides an assessment of the development roadmaps.
The reference human space exploration scenario has been built pursuing a stepwise approach, which considers as final destination a human mission to Mars (as described by NASA DRA 5.0) and relies on several missions to intermediate destinations, selected on the basis of the number of implementable capabilities (high-level functions) with the aim of guaranteeing that all Mars' required capabilities are implemented along the scenario. In the present case-study six intermediate destinations concepts have been selected as the minimum number, necessary to gradually achieve the final reference human mission to Mars. Each concept, as it is defined, allows the demonstration of capabilities through correlated strategies, and common and evolutionary missions, architectures and elements. On the basis of design status analysis, it has been verified that the scenario, as conceived, actually guarantees to achieve the capabilities required

for the Mars expedition, allowing limited delta development effort while moving from one destination to the following one.

The obtained results can be a good starting point to take strategic decisions about future missions, possibly considering additional objectives. For example, the NEA mission concept does not represent a very high added value in the path of exploration if only the technological point of view is considered, even if it is very interesting to be considered as a rehearsal for the Mars mission, and moreover from the scientific and planetary defense standpoints. It is worth underlining that the results discussed in this thesis rely on specific assumptions, which have actually driven some of the choices. Of course, if some assumptions change, the methodology (and all the analysis steps) will still be valid and applicable, but the final results could potentially be different. For example, as the considerations behind the reference scenario have been driven by the assumption of having NASA DRA 5.0 mission to Mars as final target, nuclear propulsion has been implemented through various destinations; if a different final target mission were assumed, e.g. implementing cryogenic propulsion, cryogenic propulsion would be the solution to be chosen along the scenario.

The described methodology and results are based on a purely technical approach, which does not take into account cost considerations. Accordingly, the architectures for the various missions have been defined on the basis of qualitative assessment of different parameters and in such a way to guarantee a progressive achievement of technological capabilities. As a matter of fact, the final scope has been to analyze, from a merely technological point of view, which are the development needs to guarantee the feasibility of specific space missions towards successive destinations. Moreover, the obtained database can be a valid support to take strategic decisions and right place investments for technologies development.

The process followed in this study has some similar aspects with other techniques being studied by other research groups. For example, the MIT Space Architecture Group* is working on an approach to select the most interesting architecture for any given destination [46], based on the identification of a comprehensive set of possible mission design alternatives and their evaluations via assessment of cost proxy metrics. Even in this

*Metholdologies and results have been shared and discussed together during the MITOR 2012 project co-location at MIT.

study no direct calculations are done to estimate the cost, but the ranking of the architecture alternatives is performed on the basis of cost proxy metrics including the main drivers of cost, such as IMLEO and the number of development projects. Moreover, other parameters are taken into account and for them alternative options are considered and evaluated (e.g. number of crew, mission duration, etc.). The methodology described in this thesis has some similarities with the MIT Space System Architecture Group work, especially for what concerns the parameters considered for the architecture definition, as for instance the number of crew members and the mission duration. In addition, some quantitative assessments are part of the methodology for the definition of the architectures, mainly based on the estimation of the IMLEO. However, at the higher scenario level, choices cannot be made based on quantitative evaluations, but strategic decisions are to be taken to define the overall path for exploration before entering in the details of each step and deeper investigating every single mission. On the other hand, some differences hold, mainly due to the fact that in the present study the architecture selection is mainly driven by the final objective to get the capabilities required for the human mission to Mars, which not always allows for the most "cost-effective" solution. For example, if we limited to Cis-Lunar missions, the choice of nuclear propulsion would not be completely justified, and maybe conventional propulsion would be adopted. However, in view of the final mission to Mars, which relies on nuclear propulsion, it has been decided to implement this technology even in closer destinations, in order to achieve the Mars required capability in as gradual as possible way. This decision is therefore driven by the higher level scenario definition philosophy. Furthermore, the MIT work limits its evaluation to a single destination at a time focusing on two primary functions (habitation and transportation) which are then further decomposed, while for the present study the main objective is to build an overall scenario for exploration, considering multiple destinations and several elements classes in order to take into account the evolutions needed through the various steps.

Another crucial point of the PhD work has been the analysis of enabling technologies. Indeed, once defined the reference scenario in terms of missions, architectures and associated building blocks, it is important to further deepen the analysis of the enabling technologies to be implemented in the various systems and define opportune development roadmaps.

Identifying the most required technologies, which today limit the possibility to move forward in the exploration of the solar system, is a topic of interest of many industries, agencies and academic institutions. Moreover, once identified, it is essential to understand how to implement these technologies through several incremental steps, in order to test and validate them in less risky missions, thus, improving our knowledge to get ready for more challenging targets. According to this, a detailed database describing the most innovative technologies has been built and a tool to understand the level of applicability (required, applicable/demo) to various missions elements, at several deep space destinations and in specific timeframes has been developed.

As largely discussed, the results presented in this thesis have been driven by the assumption of a final human mission to Mars as defined by the NASA DRA 5.0. Although the mission as described by NASA DRA 5.0 is quite ambitious and has several weak points in its definition, all the considerations done within this study could be easily extended to other mission opportunities, which envisage a Mars Human mission as final target. The complexity and costs associated to this type of mission would be very high, thus, limiting the probability to accomplish such a mission by the end of 2030s. However, unlike the NASA DRA 5.0 mission (focusing on a direct mission to Mars), the idea behind the present study is that of following a gradual path in the expansion through the solar system, which can allow a stepwise technological development and capabilities achievement that can drastically reduce the risks and costs associated to a mission like the NASA DRA 5.0, making it a more realistic opportunity. The objective of this study has been, therefore, to demonstrate the importance and feasibility of developing a long-term strategy for capability evolution and technology development, when considering space exploration, and specifically to provide a general methodology to be followed for the identification of the needed technologies and to support the definition of opportune development roadmaps. Even if a different "easier" architecture or a different time opportunity (maybe a postponed time opportunity), were considered for the final mission to Mars, the considerations done in this study, and most of all the methodology developed, would still be valid and applicable. Furthermore, the methodology adopted in the definition of the tool is still valid if a different final target is considered, and in this regard the tool can be used as reference set of the most innovative and enabling technologies, for which their applicability to scenario elements is specified, to support decisions about future missions to whatever deep space destination of the solar system, up to a Mars mission. For example, considering as target a

cis-lunar mission, the technologies required for that destination are identified; moreover the tool allows verifying if each technology can be implemented in a previous mission, i.e. at the ISS. According to this information, it is possible to define an opportune roadmap for the technology in terms of its development and implementation on "easier" missions to validate it and have it ready for the cis-lunar missions. Finally, the obtained results are a good support to identify the most critical technologies that need to be developed, highlighting also the timeframe in which they are needed. This could be very helpful in order to well place investments in the development of specific systems in order to allow future space exploration missions. According to what just discussed, the obtained results have a good potentiality to assess which are the next destinations for the exploration of the space beyond LEO and to preliminarily define the missions' architecture, identifying the most significant needed elements and providing a valuable support for the assessment of innovative technologies roadmaps.
In the frame of the scenario definition, two space modules have been analyzed more in depth and their conceptual design has been developed. The first module is pressurized habitat envisaged to support astronauts deep space travel (Deep Space Habitat). The first unit is foreseen as a cis-lunar infrastructure for research and technologies demonstration, as well as staging post for farther missions. The habitat configuration and its major features have been analyzed, through specific trade-off and budgets analyses, in order to meet well-defined mission's requirements. Moreover, the innovative technologies needed to build up and operate the station have been analyzed.
The second analyzed space element is a propulsive vehicle (space tug) envisioned to support satellites servicing; it represents a precursor for vehicles needed to support future assembly of large spacecraft on orbit.
Some work has been devoted to the analysis of complementary and/or alternative missions/concepts, which could potentially be considered in addition to what obtained from the HSE reference scenario analysis.
Firstly, an interplanetary CubeSats mission has been investigated. One of the most significant aspects to consider when dealing with innovative space technologies is related to the in-orbit demonstration/validation. Indeed, new technologies need to be validated in orbit (to achieve a sufficient TRL level) prior to being implemented in actual missions. According to this, it has been interesting to evaluate the possibility of

exploiting an interplanetary CubeSats mission to support technological in-orbit demonstration (e.g. advanced telecommunication system).
Finally, in the last part of the research work, a vey preliminary study has been carried out to define an "itinerant human outpost", which can represent an alternative strategy for exploring space. The idea is to rely on the re-use of an infrastructure for the exploration of multiple targets, starting from a close location, that is an Equatorial Low Earth Orbit, arriving, as final destination, to the surface of Mars. In this case, there is an initial core, including elements needed in all destinations, which is completely reused, while opportune re-configuration activities have to be performed. The rationale behind this kind of approach is that in this way the overall cost of the complex set of missions shall be reduced due to the repeated utilization of various basic architectural elements. This means that the outpost is conceived to have a great flexibility, allowing an incremental assembling of more and more complex architectures to accomplish more challenging missions.

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