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Mission analysis for interplanetary Small-Sats

Support solutions for Martian Rovers' navigation

Relatrice e relatore:

Prof.ssa Sabrina Corpino

Prof. Fabrizio Stesina

Candidato:

Davide Marampon

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SUMMARY

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ABBREVIATION

AOCS	Attitude and Orbit Control System
ConOps	Concept of Operations
EMP	Electric Magnetic Pulse
EOL	End of Life
EOP	Early Orbit Phase
FoM	Figure of Merit
GNC	Guide Navigation and Control
GSD	Ground Sample Distance
MS	Multispectral
PAN	Panchromatic
SoA	State of Art
SoW	Statement of Work
STK	System Tool Kit
STM	Science Traceability Matrix
TBC	To Be Confirmed
TBD	To Be Defined
TRL	Technology readiness level
TTM	Technology Traceability Matrix

ABSTRACT

During the last years, more and more space missions have been choosing Mars as target for their research missions. Many different Space Agencies from all over the world are organizing their resources to reach the Red Planet and unravel the aspects which remain still mysterious to our eyes. Missions like Mars Sample Return and Perseverance are an example of the technological efforts we are taking in order to gather the highest amount of information about this world that, still now, represents a challenge on many levels and it could open ways we can barely imagine nowadays.

For this very reason, the present thesis offers the reader a report of the research activity which has the goal to plan a mission based on the State of Art of Small-Sat technology. In order to prove the feasibility of this solution, the mission has been studied under many points of view and the part of said study reported in this paper is focused on the design of the operative orbits and the project of the propulsion system. In the first pages, it is possible to find a summary of the preliminary passages of the mission analysis to clarify the objectives and the constraints that we need to follow. After this part is ended, a trade-off analysis has been carried out to determine the most important aspects of the mission.

The part of this report that concerns the design of the operative orbits and the definition of various potential scenarios has been carried out using the software STK by AGI. With the same methods and programming languages, it has been possible to analyse the answer of different classes of propulsion systems to choose and implement the most suitable one for the situation.

1 INTRODUCTION

The exploration on planet Mars has become more and more frequent in the last years and the possibilities to reach and explore the Red Planet with technologies which are already been developed are on the rise. Our mission has to achieve the goal to use the relatively young and recent technology of the Small-Sat in an environment which is not a Near Earth Orbit.

Considering how the interest for the application of this types of missions is increasing, several future missions to Mars are now in a development phase. For this project, we have considered the ESA (European Space Agency) mission Mars Sample Return, which has the purpose to take off from Mars and bring back to Earth some samples of Mars' soil to study the under-surface terrain compositions for the first time. Thus, we will be allowed to have a deeper understanding of the Mars' chemical and geological composition.

In order to achieve these goals, we must guarantee that the exploration of the planet's surface is as automatic as possible and independent from human error and control. Considering that the prohibitive distance makes impossible for the ground facility on Earth to control the motions of the rovers and the Satellites, an autonomous navigation system must be programmed and implemented into the on-board computer, plus, following this course of action, it will also allow us to increase the rovers' velocity of exploration.

The intent of the present mission is to make the in-situ operations faster, more precise and more accurate thanks to an optimized planning activity of the rover paths obtained through local and regional maps.

They will cover few square kilometers, but it will help us to highlight elements that are needed to be avoided, obstacles and threads due to the terrain morphology and physical characteristics. The main goal can be achieved through a constellation of Small-Sats. It is an innovative technology, but their readiness level has been increasing in these years and the various subsystems connected to them are keeping evolving and getting more efficient. We are specifically referring to aspects like propulsion, navigation, optical and multispectral payloads, and communication.

Moreover, the interplanetary missions impose a growing autonomy level of the systems. The autonomy is indeed made necessary by the average distance between Earth and Mars, giving no room to the chance to have any sort of communication in real time. Therefore, it is impossible for the ground segment on Earth to transmit commands or information of any kind, as it is impracticable to receive any downlink communication without a delay of at least 20 minutes. Rovers themselves are limited in their movements of exploration on Mars' surface.

The project that must satisfy said requirements is called SINAV ("Soluzioni Innovative per la Navigazione Autonoma Veloce" or "Innovative Solutions for Fast Autonomous Navigations") and, in the following report, we are here going to describe the goal of this project, which is to research the potential problems and peculiarities of the Small-Sat technology for this type of extra-terrestrial mission. ALTEC and TASI will participate as well, allowing the team members of this project to access their facilities, thus allowing them to use their simulators of Mars Surface to acquire images through the optical cameras chosen for the Small-Sats and verify their pointing capability and accuracy for the image quality which our research requires.

The workload has been divided among the team members to cover various aspects of the mission and propose a plausible and feasible solution.

In particular, the part entrusted to me focuses on the design and the definition of the operative orbits of the Small-Sats constellation, the choice and the project of the propulsion system and, lastly, the implementation of those thrusters on the satellites. The orbit definition is an extremely necessary aspect because it influences the possibility of the optical payload that was put on board to acquire images, necessary for the rover's navigation. The propulsion system will have to be chosen among the possibilities given by the current state of art of the Small-Sat's technologies and a research in that field is what will tell us whether the mission can be carried out or not.

The results shall be calculated using the AGI program STK (System Tool Kit) to verify the value of the algorithms, which were chosen for this peculiar case, and to simulate the operative environment in the most accurate way as possible.

In the following chapters, we are going to dive more deeply into these topics with the purpose to exemplify in a more precise way both the different aspects of the mission and the solutions we have come up with to solve the problems which we have encountered during the development of this architecture.

2 MISSION DEFINITION

2.1 Mission Statement

Before entering into a deeper understanding and explanation of the mission, here the Mission Statement is reported in order to highlight the ultimate purpose the mission is to achieve:

"To develop a space mission to support the autonomous navigation of Martian rovers providing unprecedented, periodically updated, information from the orbit over the rover site. This support mission is based on interplanetary Small-Sats with innovative miniaturized technologies."

The Mission Statement is a synthetic summary of the mission's reason of existence, the SINAV Project has the ambition to support the improvements of the Martian rovers' ability and performance to autonomously explore the surface of the planet through a quicker coverage of the explored surface.

Specific contour conditions, desired performance and constraints at mission and programmatic level (beyond that at system level) have been studied. A mother-daughter architecture is considered both for the release in orbit of the small-sat(s) and for the communication. The main performances are the revisit time of the site of interest, coverage areas, number and quality of the data / maps, limits on the operations, total cost, the definition of a launch window have been included.

2.2 Mission Objectives

Table 1: Mission objectives

ID	Type	Objective
OBJ1	Primary	To support Martian rovers to improve their autonomous navigation
OBJ2	Primary	To provide referenced-maps to Martian rovers
OBJ3	Primary	To validate new Small-Sats technologies in deep space environment
OBJ4	Auxiliary	To provide possible paths planning to Martian rovers

The goal of the mission is to gather information about Mars' surface and, after that, to process the images acquired by the satellite(s) orbiting around the planet in order to provide maps and, if any, path planning, and communicate them to Earth and to the rover. The obtained information would allow to identify and localize possible areas of interest, avoiding in the process dangerous or useless portions of the red planet, speeding up the exploration and optimizing paths and trajectories.

2.3 Mission Requirements

Mission / High level requirements have been derived in Table. They come from objectives and programmatic constraints.

Table 2: Mission requirements

MISSION REQUIREMENTS		**TRACEABILITY**
MR01	The mission shall be accomplished by Small-Sat(s)	OBJ01
MR02	The mission shall be provided raw Mars' map to performing navigation	OBJ02
MR03	The mission shall validate utility of Small-Sat(s) technology in deep space environment	OBJ03
MR04	The mission shall provide possible paths planning to Martian rovers	OBJ04
MR05	Mission duration shall be ≥ 2 years (extended mission duration shall be taken into account)	SoW
MR06	The Small-Sat(s) shall be injected in a stable orbit near Mars by the mothercraft	SoW
MR07	The total mass of Small-Sat(s) shall be at least 250 kg	SoW
MR08	The mission shall guarantee a coverage of at least 95% (TBC) of the Mars' surface where rovers operate	MR04
MR09	Mars shall be mapped in the visible spectrum to increase autonomous navigation of rovers' mission	MR02
MR10	Mars shall be mapped in the IR spectrum to increase autonomous navigation of Martian rovers	MR02
MR11	The launch date is schedule by 2030. Delays shall be taken into account	SoW
MR12	The mission shall end with de-orbiting maneuver to impact on Mars' surface	SoW
MR13	Low-cost technology (COTS with, at least, TRL 3 at 2022) shall be adopted in the mission	MR3
MR14	The mission shall cost less than 10M$	SoW
MR15	The mission shall guarantee a revisit time of TBD hours	SoW

The different parts in which this table is divided, have a twofold origin, they come from either the Statement of Work or the Mission Objectives. During the early phases of the project, we need to put reasonable limits of acceptance in order to consider this effort profitable and to ensure that the goals are achieved without risks or errors which could compromise the scientific truth and the objectives we need to satisfy. Guarantying these constraints are accomplished in the foreseen terms, the SINAV support mission is acceptable. On the other hand, those parts of the table which are strictly connected to the Mission Objectives, are there to remind the reasons why this mission is being developed and work as guidelines of the project.

2.4 STM and TTM

The Science Traceability Matrix (STM) and the Technological Traceability Matrix (TTM) are two important tools useful to define mission goals and requirements, starting from the main needs. Both follow the same logic: from the goals/objective it is possible to decompose the problem increasing details in order to derive measurement, instrument and mission requirements. While the STM aims to choose the suitable payloads for the spacecraft and to understand the main mission drivers, the TTM considers technology need to improve mission autonomy.

2.4.1 STM

Table 3: Science Traceability Matrix

Science need/question	Science goal	Science objective	Science requirements	Instrument/Technique	Measurements requirements	Instruments requirements	Mission requirements	Data products
How to mapping Mars surface?	To support Martian rovers in order to increase their autonomous navigation and definition of path planning	To produce maps of strategic areas in rovers missions in every condition of light	Mars shall be mapped in the visible spectrum in strategic areas for rovers missions	Optical imager	Spatial resolution ≤ 0,5 m	Camera TBD FOV: 4-8 deg Pixel: 12 Mpixel	Orbit altitude: 50-100 km (TBC)	High resolution Mars's surface map
		To characterize different terrain typology and their chemical composition	Mars shall be mapped in the spectral range of 0.4-4 μm	Hyperspectral camera	Spectral range ≤ 450-950 nm Spectral resolution ≤ 16 nm	HS Camera: HyperScout FOV: 31 [deg] Pixel: 12 Mpixel	Orbit altitude: 150 km (TBC)	Chemical composition and map of high interest areas

The principal scientific need is to map Mars' surface in order to support the autonomous navigation of martian rovers.

2.4.2 TTM

The technology's needs are to reduce the required propellant, in order to reach and maintain the desired orbit, and to determine the spacecraft position to develop a navigation system using landmarks. The navigation will be performed thanks to the use of the optical camera.

Table 4: Technology Traceability Matrix

Technology need/question	Technology goal	Technology objective	Technology requirements	Instrument/Technique	Measurements requirements	Instruments requirements	Mission requirements	Data products
How to minimize the required propellant for maintain Small-sat(s) in a stable orbit?	To reach the desired orbit	To provide ΔV needed for the operations	Part of the central volume shall be used to house the propulsion system	Electric/Electro-Magnetic Propulsion Thrusters	(-)	Average mass: 1-2 kg Average power: 100-1000 W	TBD (on STK)	ΔV budget
	To maintain the desired orbit	To correct possible errors in the real orbit due to perturbation and external disturbances	The Small-sat(s) shall have (criogenic) tank to store the fuel	Cold-Gas Thrusters	(-)	Average mass: 0,01-0,1 kg Average power: 1-10 W	TBD (on STK)	
How to define the Small-sat(s) position?	To perform navigation using landmark	To produce global mapping	Mars shall be mapped to obtain raw images for navigation purpose	Optical imager	Spatial resolution ≤ 1,5 m	Camera TBD FOV: 4-8 deg Pixel: 12 Mpixel	Orbit altitude ≤ 150 km TBC	Raw resolution Mars's surface map to identify landmarks
	To develop a navigation system based on optical instruments	To design a navigation system using land-marks obtained from the optical camera images	The GNC system shall be able to determine Small-sat(s) position through the use of reference points (landmarks)					Landmarks creation
				Reaction Wheels	Pointing accuracy ≤ 1 deg	Lifetime at least 5 years Speed Range TDB [rpm] Speed control accuracy TBD [rpm]	The Small-sat(s) position accuracy on three axes	Attitude determination

2.5 Mission design

2.5.1 Concept of Operations

The Concept of Operations explains how it is expected that the mission will work to satisfy Mission Requirements. Starting from the end of the interplanetary transfer and the arrival to Mars, the mission has been divided in four main phases, which have been further decomposed into multiple mission scenarios.

Table 5: ConOps

Mission Phases	Mission Scenarios		Duration
EOP	Deployment	Release from the mothercraft	30 minutes
	Detumbling	Detumbling in a stable orbit	2 hours
	Appendix deployment	Subsystems' appendix deployment	2 hours
Commissioning	Spacecraft checkout	Advanced technology checkout	1 hour
		Payloads checkout	2 days
On-orbit operative phase	Manoeuvre 1 - Transfer to operative orbit 1		TBD
	Global observations and mapping		3 months
	Manoeuvre 2 - Transfer to operative orbit 2		TBD
	Detailed observation and mapping		1,5 years
EOL	Active disposal (extra mission)		2 months
	Passive disposal		1 months

The mission starts with the deployment of the multiple Small-Sat(s) from the mothercraft to a stable orbit. After the detumbling phase is over, the commissioning phase starts and functional tests of subsystems and payloads are performed. The achievement of the desired instruments calibration leads to the transition to the On-orbit operative phase. In this phase the Small-Sat(s), in a stable orbit, map Mars for navigation purposes with an optical camera and characterize different terrain typologies and their chemical composition with hyperspectral cameras. After collecting images of Mars' surface, they will be later processed. After three months this process is completed and the Small-Sat(s) navigate autonomously using optical camera. Then, the Small-Sat(s) enter a second manoeuvre phase transferring to a lower orbit to perform optical remote sensing applications providing possible paths planning to Martian rovers. The end of this phase is expected after 1.5 years, when strategic areas' maps in rovers' missions are produce. A series of lowering and raising of altitude can be envisaged to collect different data of Mars' surface. In the end, an active disposal phase starts where Small-Sat(s) are still capturing images and data. Passive disposal phase provides a passivation of the Small-Sat(s) and a de-orbiting manoeuvre to impact on Mars' surface.

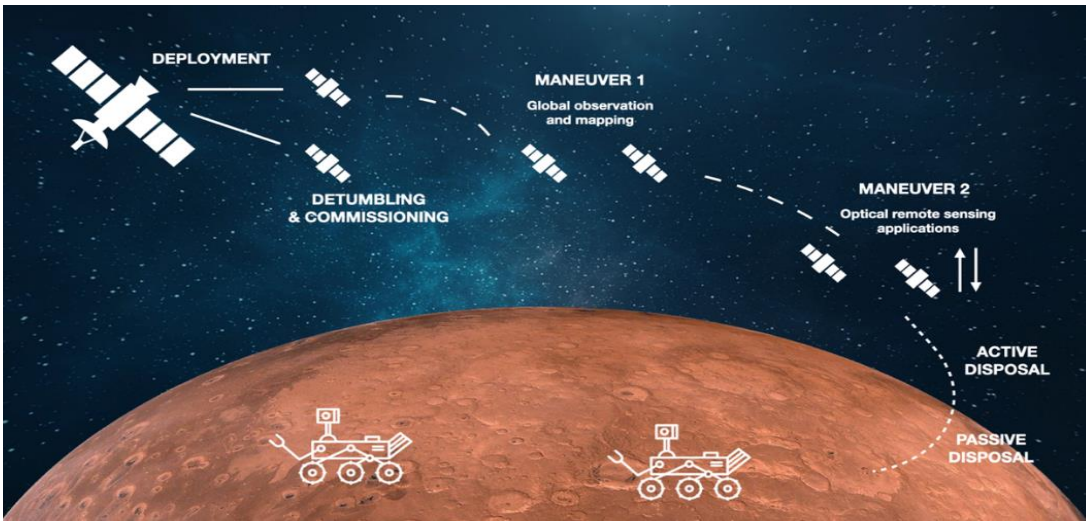


Figure 1: DRM

Table 6: Early Operation Phase

Characteristics	DEPLOYMENT	DETUMBLING	APPENDIX DEPLOYMENT
Initial condition	Mothercraft has reached a stable orbit	Small-Sat(s) are separated from the mothercraft	Small-Sat(s) have been stabilised in the right attitude
Final condition	Small-Sat(s) are separated from the mothercraft	Small-Sat(s) have been stabilised in the right attitude	Complete deployment of solar arrays and antennas
Environment	Mars space environment	Mars space environment	Mars space environment
Top Level Objectives	To separate Small-Sat(s) from the mothercraft	To prevent tumbling or spinning of the Small-sat(s) To manage the three-axis set-up of the Small-sat(s) To ensure the right attitude of the Small-sat(s)	To ensure the right deployment of subsystems's appendix
Required I/F with other systems	Mechanical, electrical and data I/F with mothercraft	Link with mothercraft for communication to/from Earth	Link with mothercraft for communication to/from Earth
General description	In this phase the Small-Sat(s) are released from the mothercraft after preliminary check	This phase provides the stabilization process of the Small-Sats' angular velocity after orbital insertion	This phase allows deployment of subsystems' appendix
Duration	30 minutes	2 hours	2 hours
Constraints	The constraints are defined in terms of: - Link with mothercraft for both communications to/from Earth - On orbit perturbations - Environmental conditions - Deployers	The constraints are defined in terms of: - Link with mothercraft for both communications to/from Earth - On orbit perturbations - Environmental conditions	The constraints are defined in terms of: - Link with mothercraft for both communications to/from Earth - Environmental conditions - Appendix deployment system
Potential Off-Nominal Events	Failure/impossibility to communicate with the Earth Failure of the deployer	Failure/impossibility to communicate with the Earth the correct attitude Failure of the AOCS system	Failure/impossibility to communicate with the Earth the correct deployment of appendix Failure of appendix deployment system

Table 7: Spacecraft checkout

Characteristics	ADVANCED TECHNOLOGIES CHECKOUT	PAYLOAD CHECKOUT
Initial condition	Complete deployment of solar arrays and antennas	The correct functioning of the primary systems has been checked
Final condition	The correct functioning of the primary systems has been checked	The correct functioning of the payloads has been checked
Environment	Mars space environment	Mars space environment
Top Level Objectives	To ensure the proper functioning of the subsystems	To ensure the proper functioning of the payloads To ensure the proper calibration of the payloads
Required I/F with other systems	Link with mothercraft for communication to/from Earth	Link with mothercraft for communication to/from Earth
General description	This phase provides functional tests of subsystems in order to verify that they working correctly	This phase provides functional tests of payloads in order to verify that they working correctly
Duration	1 hour	2 days
Constraints	The constraints are defined in terms of: - Link with mothercraft for both communications to/from Earth in order to communicate the status of all the primary systems - On orbit perturbations - Environmental conditions	The constraints are defined in terms of: - Link with mothercraft for both communications to/from Earth in order to communicate the status of all the payloads - On orbit perturbations - Environmental conditions
Potential Off-Nominal Events	Failure/impossibility to communicate with the Earth the correct functioning or failure of one/more than one subsystem Failure of one/more than one primary system Failure while booting up the functional test	Failure/impossibility to communicate with the Earth the correct functioning or failure of one/more than one payload Failure of one/more than one payload Failure while booting up the functional test

Table 8: On-Orbit Operative Phase

Characteristics	MANEUVER 1 - TRANSFER TO OPERATIVE ORBIT 1	GLOBAL OBSERVATIONS AND MAPPING	MANEUVER 2 - TRANSFER TO OPERATIVE ORBIT 2	DETAILED OBSERVATION AND MAPPING
Initial condition	The correct functioning of the payloads has been checked	Small-Sat(s) reached the desired orbit	All surfaces of interest have been mapped	All surfaces of interest have been mapped
Final condition	Small-Sat(s) reached the desired orbit	All surfaces of interest have been mapped	Small-Sat(s) reached the desired orbit	Small-Sat(s) reached the desired orbit
Environment	Mars space environment	Mars space environment in a circular orbit at the altitude of 150 km	Mars space environment	Mars space environment
Top Level Objectives	To complete the transfer to a stable orbit which is used to accomplish the primary objectives mission	To complete surface mapping for navigation purpose To characterize different terrain typology and their chemical composition	To complete the transfer to a stable orbit which is used to accomplish the secondary objectives mission	To complete the transfer to a stable orbit which is used to accomplish the secondary objectives mission
Required I/F with other systems	Link with mothercraft for communication to/from Earth	Link with mothercraft for communication to/from Earth	Link with mothercraft for communication to/from Earth	Link with mothercraft for communication to/from Earth

General description	The Small-Sat(s) reach the operative orbit for navigation purpose and definition of surface's chemical composition	This phase allows to collect data about the planet, its environment and to map its surface thanks to the optical payload in various frequencies of light. Small-sat(s) maintain a stable orbit with manoeuvres of station-keeping	The Small-Sat(s) reach the operative orbit for remote sensing applications	The Small-Sat(s) reach the operative orbit for remote sensing applications
Duration	TBD days	3 months	TBD days	TBD days
Constraints	The constraints are defined in terms of: - Link with mothercraft for both communications to/from Earth - On orbit perturbations - Environmental conditions - Attitude control - Proper functioning of the propulsion system	The constraints are defined in terms of: - Link with mothercraft for both communications to/from Earth - On orbit perturbations - Environmental conditions - Attitude control	The constraints are defined in terms of: - Link with mothercraft for both communications to/from Earth - On orbit perturbations - Environmental conditions - Attitude control - Proper functioning of the propulsion system	The constraints are defined in terms of: - Link with mothercraft for both communications to/from Earth - On orbit perturbations - Environmental conditions - Attitude control - Proper functioning of the propulsion system
Potential Off-Nominal Events	Failure of propulsion system Failure of AOCS system and impossibility to correctly point the thrusters	Failure of the visual camera used for navigation Failure of the hyperspectral camera used for study chemical composition of Mars' surface Failure of propulsion system for station-keeping purpose	Failure of propulsion system Failure of AOCS system and impossibility to correctly point the thrusters	Failure of propulsion system Failure of AOCS system and impossibility to correctly point the thrusters

Table 9: End of Mission

Characteristics	ACTIVE DISPOSAL (EXTRA MISSION)	PASSIVE DISPOSAL
Initial condition	Remote sensing applications has been completed	Start of space segment passivation
Final condition	Start of space segment passivation	Impact on Mars' surface
Environment	Mars space environment	Mars space environment, Mars surface
Top Level Objectives	To produce maps of strategic areas in rovers' missions for paths planning definition	To perform a de-orbiting manoeuvre to impact on Mars' surface To passivate the Small-Sat(s)
Required I/F with other systems	Link with mothercraft for communication to/from Earth	Link with mothercraft for communication to/from Earth
General description	Mission is complete, but Small-Sat(s) reduces its altitude up to 20 km (TBC) while still capturing images.	Small-Sat(s) ends his life in a de-orbiting manoeuvre to impact on Mars' surface
Duration	2 months (TBC)	1 months (TBC)
Constraints	The constraints are defined in terms of: - Link with mothercraft for both communications to/from Earth - Position of Martian rovers	The constraints are defined in terms of: - Link with mothercraft for both communications to/from Earth - Position of Martian rovers
Potential Off-Nominal Events	Failure of one or more subsystems Failure/impossibility to communicate with the Earth Failure of the visual camera used for remote sensing applications	Failure of one of more subsystems Failure/impossibility to communicate with the Earth

2.5.2 Mission Architecture

Considering the description of the ConOps, seven mission architectures are proposed. The main mission elements identified are:

- **Subject**: purpose of the mission, the same for all mission architectures
- **Space segment**: identified by the number of satellites capable of implementing the mission
- **Payload:** instruments able to perform mission requirements, in this case they are the same for all mission architectures
- **Mission geometry**: orbits (and, in case, the constellation configuration) in which satellites will operate
- **Operations:** define the number and types of the operations and the related efforts of facilities and human workload
- **Communication**: define the communication architecture
- **Ground segment:** it includes all the infra-structure on Earth used to exchange data with the interplanetary elements
- **Launch segment:** it highlights the activity related to the launch and achievement of the orbit to start the mission.

The last three elements (constraints of communication network) remain the same for all the choices, the ground station network of ALTEC and the mothercraft releases the Small-Sats after it reaches its stable orbit. Subjects are the areas of interest (where the rovers operate). Payload can include optic and hyperspectral elements that can be different in term of mass, volume and power but with the same performance requirements on spatial and spectral resolutions. The architectures are mainly differentiated by three elements: number of satellites, orbit configuration and the release method from the mothercraft. The differences between the seven architectures are listed below:

- **Mission Architecture 1**: a single Small-Sat operates at an altitude of 150 km in a circular orbit performing global observation and mapping operations. After they reach the altitude of 50-80 km, the Small-Sats start remote sensing applications for maps definition. In this case the Small-Sat has a mass of 150 kg, which allows to carry more propellent on-board. For this reason, it can be considered to perform more orbital maneuvers. Moreover, the volume and mass for payload is higher.

- **Mission Architecture 2**: two Small-Sats operate in a circular orbit of 150 km. The deployment type split up the mission architecture in two solutions:
  - **A.** The two Small-Sats are released together by the mothercraft and, with a gap of time from each other, they reach the altitude of 150 km. In this phase the two Small-Sats produce global observation and mapping. In the second operational phase the two Small-Sats reach a lower altitude, one of them at 80 km in a circular orbit and the other one at 50 km in a circular orbit.
  - **B.** The two Small-Sats are released in a distributed way, giving them a phase shift of 180° in the operative orbit at 150 km. Once they have completed the first operative phase, both perform the Hohmann Transfer at the same time to reach the altitude of 80 km in a circular orbit in order to perform the second operational phase. In the circular orbit of 80 km the two Small-Sats keep the different phase of 180°.

- **Mission Architecture 3**: three Small-Sats are in the same circular orbit of 150 km of the previous mission architectures. Also, in this case the deployment type provides two solutions:
  - **A.** The three Small-Sats are released together by the mothercraft and, with a gap of time from each other, they reach the altitude of 150 km in a circular orbit, where they produce global observation and mapping. In the second operational phase the three

Small-Sats reach a lower altitude, one of them at 50 km in a circular orbit and the other two at 80 km in a circular orbit with a phase shift of 180 deg from each other.

B. The three Small-Sats are released in a distributed way, giving them a phase shift of 120° in the operative orbit at 150 km. Once they have completed the first operative phase, all the Small-Sats perform the Hohmann manoeuvre at the same time to reach the altitude of 80 km in a circular orbit in order to perform the second operational phase. In the circular orbit of 80 km the three Small-Sats keep the different phase of 120 deg.

- **Mission Architecture 4**: four Small-Sats operating in the same circular orbit of 150 km. The two solutions resulting by deployment type are:

  A. The four Small-Sats are released together by the mothercraft and, with a gap of time from each other, they reach the altitude of 150 km in a circular orbit, where they produce global observation and mapping. After the Hohmann manoeuvre to reach a lower altitude in a circular orbit at 80 km, all Small-Sats perform the second operative phase with a phase shift of 90° from each other.

  B. The Small-Sats are released in a distributed way to have a different phase of 90° from each other at the altitude of 150 km. After the proper manoeuvre at the same time the four Small-Sats reach the circular orbit at 80 Km with different phase of 90° to perform the second operational phase.

In conclusion, all mission architectures are described in detail in the Tables.

Table 10: Mission architecture 1

MISSION ELEMENT	DESCRIPTION			
Space segment configuration	Number of Small-Sats:	1	150 kg/Small-Sat TBC	SINGLE DEPLOYMENT
Payload	Optical camera	1+	TBD	Optical imager shall be used to achieve remote sensing applications
	Optical camera	1+	TBD	Optical imager shall be used to achieve global observation and mapping of Mars for navigation purpose
	Hyperspectral camera	1	TBD	Hyperspectral camera shall be used to achieve ground chemical composition of Mars with 20,1 m GSD at 150 km.
Mission Geometry	Transfer to operative orbit 1			The Small-Sat shall move in a circular orbit at 150 km (TBC) after being released from the mothercraft (orbiting at 200 km).
	Global observations and mapping			The Small-Sat shall stay in a circular orbit at 150 km (TBC) to produce raw global mapping for navigation purpose with optical camera.
				In the same orbit at 150 km (TBC) data of chemical composition of high interest area are provided by hyperspectral camera.
	Transfer to operative orbit 2			The Small-Sat shall move from 150 km orbit to 50-80 km (TBC) circular orbit.
	Optical remote sensing applications			The Small-Sat shall stay in a circular orbit at 50-80 km (TBC) to produce high resolution map of Mars' surface by optical camera in VIS and NIR range.

Table 11: Mission architecture 2A

MISSION ELEMENT	DESCRIPTION			
Space segment configuration	Number of Small-Sats:	2	75 kg/Small-Sats TBC	SWARM DEPLOYMENT
Payload	Optical camera	1+	TBD	Optical imager shall be used to achieve remote sensing applications
	Optical camera	1+	TBD	Optical imager shall be used to achieve global observation and mapping of Mars for navigation purpose
	Hyperspectral camera	1	TBD	Hyperspectral camera shall be used to achieve ground chemical composition of Mars with 20,1 m GSD at 150 km.
Mission geometry	Transfer to operative orbit 1			The Small-Sat shall release together (with a gap of TBD minutes) by the mothercraft (orbiting at 200 km) and reach the operative orbit at 150 km (TBC).
	Global observations and mapping			The Small-Sats shall stay in a circular orbit at 150 km (TBC) to produce raw global mapping for navigation purpose with optical camera.
				In the same orbit at 150 km (TBC) data of chemical composition of high interest area are provided by hyperspectral camera.
	Transfer to operative orbit 2			The small-sats shall move in a lower orbit one by one.
	Optical remote sensing applications			One small-sat shall stay in a circular orbit at 80 km (TBC). The second one shall stay in a circular orbit at 50 km (TBC). Both satellites produce high resolution map of Mars' surface by optical camera in VIS and NIR range.

Table 12: Mission Architecture 2B

MISSION ELEMENT	**DESCRIPTION**			
Space segment configuration	number of Small-Sats:	2	100 kg/small-Sats TBC	DISTRIBUTED DEPLOYMENT
Payload	Optical camera	1	TBD	Optical imager shall be used to achieve remote sensing applications
	Optical camera	1	TBD	Optical imager shall be used to achieve global observation and mapping of Mars for navigation purpose
	Hyperspectral camera	1	TBD	Hyperspectral camera shall be used to achieve ground chemical composition of Mars with 20,1 m GSD at 150 km.
Mission geometry	Transfer to operative orbit 1			The Small-Sats shall release by the mothercraft (orbiting at 200 km) at different times. After one reach 180 deg of the operative orbit (150 km), the other one will be released.
	Global observations and mapping			The Small-Sats shall stay in a circular orbit at 150 km (TBC) with different phase of 180 deg to produce raw global mapping for navigation purpose with optical camera.
				In the same orbit at 150 km (TBC) data of chemical composition of high interest area are provided by hyperspectral camera.
	Transfer to operative orbit 2			The Small-Sats shall move in a lower orbit at the same time.
	Optical remote sensing applications			Both Small-Sats shall stay in a circular orbit at 80 km (TBC) with different phase of 180 deg. Both satellites produce high resolution map of Mars' surface by optical camera in VIS and NIR range.

Table 13: Mission architecture 3A

MISSION ELEMENT	DESCRIPTION			
Space segment configuration	Number of Small-Sats:	3	50 kg/small-Sats TBC	SWARM DEPLOYMENT
Payload	Optical camera	1	TBD	Optical imager shall be used to achieve remote sensing applications
	Optical camera	1	TBD	Optical imager shall be used to achieve global observation and mapping of Mars for navigation purpose
	Hyperspectral camera	1	TBD	Hyperspectral camera shall be used to achieve ground chemical composition of Mars with 20,1 m GSD at 150 km.
Mission geometry	Transfer to operative orbit 1			The Small-Sat shall release together (with a gap of TBD minutes) by the mothercraft (orbiting at 200 km) and reach the operative orbit at 150 km (TBC).
	Global observations and mapping			The Small-Sats shall stay in a circular orbit at 150 km (TBC) to produce raw global mapping for navigation purpose with optical camera.
				In the same orbit at 150 km (TBC) data of chemical composition of high interest area are provided by hyperspectral camera.
	Transfer to operative orbit 2			The Small-Sats shall move in a lower orbit one by one.
	Optical remote sensing applications			One small-sat shall stay in a circular orbit at 50 km (TBC). Two small-sats shall stay in a circular orbit at 80 km (TBC) with different phase of 180 deg. All satellites produce high resolution map of Mars' surface by optical camera in VIS and NIR range.

Table 14: Mission architecture 3B

MISSION ELEMENT	DESCRIPTION			
Space segment configuration	number of Small-Sats:	3	50 kg/small-Sats	DISTRIBUTED DEPLOYMENT
Payload	Optical camera	1	TBD	Optical imager shall be used to achieve remote sensing applications
	Optical camera	1	TBD	Optical imager shall be used to achieve global observation and mapping of Mars for navigation purpose
	Hyperspectral camera	1	TBD	Hyperspectral camera shall be used to achieve ground chemical composition of Mars with 20,1 m GSD at 150 km.
Mission geometry	Transfer to operative orbit 1			The Small-Sats shall release by the mothercraft (orbiting at 200 km) at different times. After one reach 120 deg of the operative orbit (150 km), the other one will be released. The last one will be released when first small-sat reach 240 deg on his orbit.
	Global observations and mapping			The Small-Sats shall stay in a circular orbit at 150 km (TBC) with different phase of 120 deg to produce raw global mapping for navigation purpose with optical camera.
				In the s orbit at 150 km (TBC) data of chemical composition of high interest area are provided by hyperspectral camera.
	Transfer to operative orbit 2			The Small-Sats shall move in a lower orbit at the same time.
	Optical remote sensing applications			All Small-Sats shall stay in a circular orbit at 80 km (TBC) with different phase of 120 deg. All satellites produce high resolution map of Mars' surface by optical camera in VIS and NIR range.

Table 15: Mission architecture 4A

MISSION ELEMENT	DESCRIPTION			
Space segment configuration	number of Small-Sats:	4	50 kg/small-Sats TBC	SWARM DEPLOYMENT
Payload	Optical camera	1	TBD	Optical imager shall be used to achieve remote sensing applications
	Optical camera	1	TBD	Optical imager shall be used to achieve global observation and mapping of Mars for navigation purpose
	Hyperspectral camera	1	TBD	Hyperspectral camera shall be used to achieve ground chemical composition of Mars with 20,1 m GSD at 150 km.
Mission geometry	Transfer to operative orbit 1			The Small-Sat shall release together (with a gap of TBD minutes) by the mothercraft (orbiting at 200 km) and reach the operative orbit at 150 km (TBC).
	Global observations and mapping			The small-Sats shall stay in a circular orbit at 150 km (TBC) to produce raw global mapping for navigation purpose with optical camera.
				In the same orbit at 150 km (TBC) data of chemical composition of high interest area are provided by hyperspectral camera.
	Transfer to operative orbit 2			The small-Sats shall move in a lower orbit one by one.
	Optical remote sensing applications			All small-Sats shall stay in a circular orbit at 80 km (TBC) with different phase of 90 deg to produce high resolution map of Mars' surface by optical camera in VIS and NIR range.

Table 16: Mission architecture 4B

MISSION ELEMENT	DESCRIPTION			
Space segment configuration	Number of Small-Sats:	4	50 kg/small-Sats	DISTRIBUTED DEPLOYMENT
Payload	Optical camera	1	TBD	Optical imager shall be used to achieve remote sensing applications
	Optical camera	1	TBD	Optical imager shall be used to achieve global observation and mapping of Mars for navigation purpose
	Hyperspectral camera	1	TBD	Hyperspectral camera shall be used to achieve ground chemical composition of Mars with 20,1 m GSD at 150 km.
Mission geometry	Transfer to operative orbit 1			The Small-Sats shall release by the mothercraft (orbiting at 200 km) at different times. After first satellite reach 90 deg of the operative orbit (150 km), the second one will be released. The third and the fourth will be released when first small-sats reach 180 deg and 270 deg
	Global observations and mapping			The Small-Sats shall stay in a circular orbit at 150 km (TBC) with different phase of 90 deg to produce raw global mapping for navigation purpose with optical camera.
				In the same orbit at 150 km (TBC) data of chemical composition of high interest area are provided by hyperspectral camera.
	Transfer to operative orbit 2			The Small-Sats shall move in a lower orbit at the same time.
	Optical remote sensing applications			All Small-Sats shall stay in a circular orbit at 80 km (TBC) with different phase of 90 deg to produce high resolution map of Mars' surface by optical camera in VIS and NIR range.

2.5.3 Preliminary ΔV analysis

The table containing the ΔV needed for each architecture is reported in

Table.

Table 17: Preliminary ΔV analysis

Architecture		Deployment phase from 200 km to 150 km	Global observations and mapping phase from 150 km to 80 km or 50 km		Remote sensing applications phase to 80 km or 50 km		ΔV_statellite [m/s]	ΔV_architecture [m/s]
1	ΔV [m/s]	-12,10	satellite 1	-29,51	satellite 1	-17,45	59,07	59,07
	Description	Hohmann Transfer	Orbit circularization and Hohmann Transfer		The satellite enters into the last orbit at 80 km			
2A	ΔV [m/s]	-24,21	satellite 1	-29,51	satellite 1	-17,45	71,17	157,59
			satellite 2	-37,09	satellite 2	-25,12	86,42	
	Description	Hohmann Transfer	Orbit circularization then: satellite1) Hohmann Transfer from 150 km to 80 km satellite2) Hohmann Transfer from 150 km to 50 km		Satellite 1 enters into the last orbit at 80 km Satellite 2 enters into the last orbit at 50 km			
2B	ΔV [m/s]	-24,21	satellite 1 satellite 2	-59,02	satellite 1 satellite 2	-34,91	118,14	118,14
	Description	Hohmann Transfer	Orbit circularization and Hohmann Transfer		The satellites enter into the last orbit at 80 km			
3A	ΔV [m/s]	-36,31	satellite 1 satellite 2	-59,02	satellite 1 satellite 2	-34,91	130,24	228,76
			satellite 3	-37,09	satellite 3	-25,12	98,52	
	Description	Hohmann Transfer	Orbit circularization then: satellite1+2)		Satellite 1+2 enters into the last orbit at 80 km Satellite 3 enters into the last orbit at 50 km			

			Hohmann Transfer from 150 km to 80 km satellite3) Hohmann Transfer from 150 km to 50 km					
3B	ΔV [m/s]	-36,31	satellite 1 satellite 2 satellite 3	-88,54	satellite 1 satellite 2 satellite 3	-52,35	177,20	177,20
	Description	Hohmann Transfer	Orbit circularization and Hohmann Transfer		The satellites enter into the last orbit at 80 km			
4A	ΔV [m/s]	-48,41	satellite 1 satellite 2 satellite 3 satellite 4	-118,05	satellite 1 satellite 2 satellite 3 satellite 4	-69,80	236,26	236,26
	Description	Hohmann Transfer	Orbit circularization and Hohmann Transfer		The satellites enter into the last orbit at 80 km			
4B	ΔV [m/s]	-48,41	satellite 1 satellite 2 satellite 3 satellite 4	-118,05	satellite 1 satellite 2 satellite 3 satellite 4	-69,80	236,26	236,26
	Description	Hohmann Transfer	The satellites enter into the last orbit at 80 km		The satellites enter into the last orbit at 80 km			

2.6 Preliminary coverage analysis

Here are reported four images showing the initial status of the chosen architectures. Along with that, an image of the ideal coverage of Mars' surface has been added along with the potential duration each architecture takes to cover at least 95% of the area of interest.

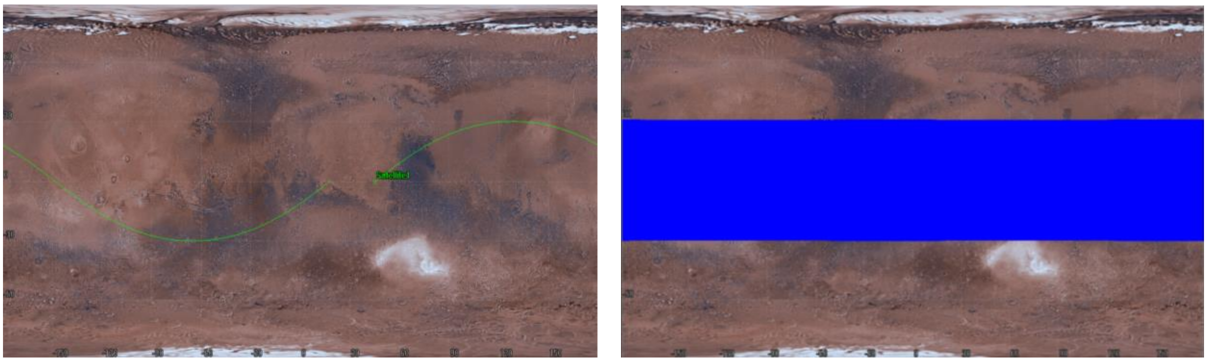


Figure 2: Coverage mission architecture 1 after 2 months and 24 days

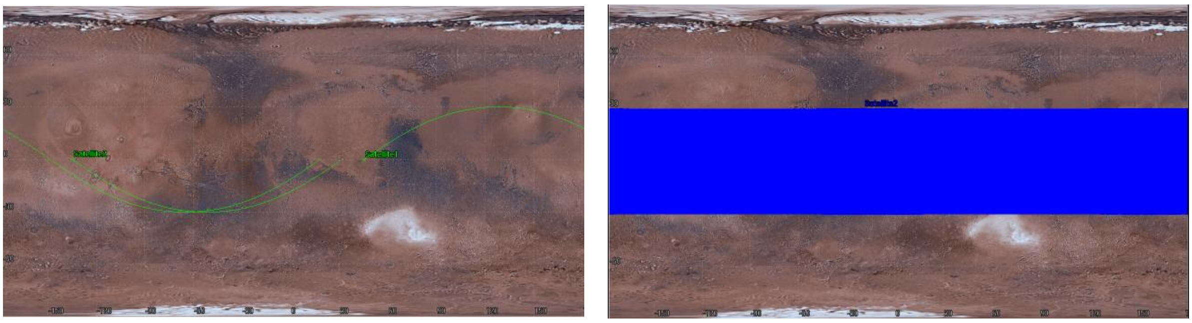


Figure 3: Coverage mission architecture 2B after 2 months and 4 days

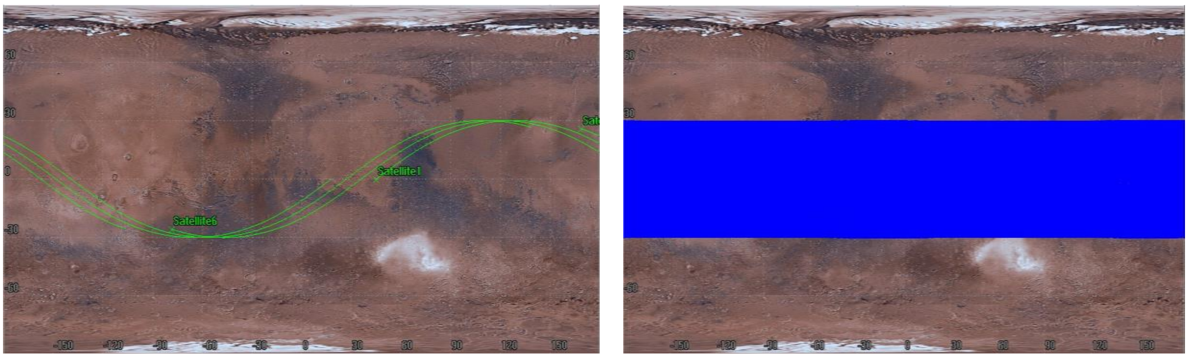


Figure 4: Coverage mission architecture 3B after 45 days

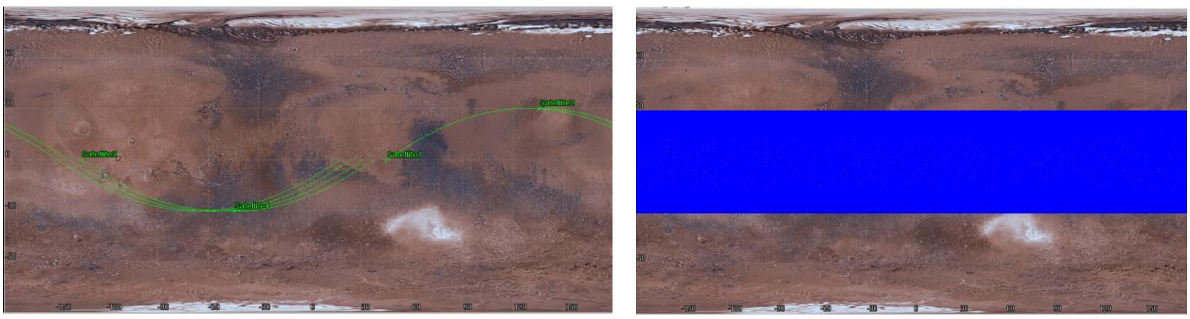


Figure 5: Coverage mission architecture 4B after 40 days

This is the first step to determine the global coverage of the satellite(s). In the previous images, it is possible to see an example of the ground track of the various Small-Sats, this helps us to understand how much time each architecture requires to guarantee a 95 % coverage of Mars' surface.

The next step we are going to take is use STK to put on board of each and every satellite a real sensor with parameters which are close to the ones of the real case. In the next paragraphs, there is a quick report of the research which the other SINAV team members have accomplished to decide what type of camera is the most suitable for the mission in exam. It is possible to find there the origin of the specific data that are inserted into STK in this chapter.

We will anticipate to the reader that the mission architecture that was chosen for this mission is the 4B architecture and this will be used as a base for our coverage analysis. In the next chapter, there will be a presentation of the work on STK that was carried on to program a simulation for said architecture. For this moment, we will consider having already the mission scenario 4B ready and operative on STK, therefore, we can proceed to insert "Coverage Definition" into our simulation and define the properties as the mission requires. There is the need to create a grid which is divided in various cells. Our target is to compute the accesses to each cell, it is possible to calculate how times a cell is covered by the satellite as the mission progresses, but, for now, it is sufficient to receive a yes-or-no report. Therefore, we will be able to tell what areas are flown over and what areas are not covered. So, the information about the number of passages on just one cell is an unwanted complication at this state of the research.

We will change the maximum and minimum latitude to 30° North and 30° South, thus STK will position and compute the cells only in that area the mission is interested in. This will make the grid more precise and it will not waste time and energy to create cells in areas that for sure are not covered by the satellites, like the polar regions for example. For the same reason, if we want a simulation that is as precise as possible, we will have to reduce the point granularity from 6° to 1°. The time interval to produce a result will be longer, but the final report will be more accurate and the analysis will be closer to the real situation of the mission.

Initially, we thought about giving some degress of margin for the maximum and minimum latitude inserting 35° N and 35° S, but this creates a grid that is less thick and, moreover, when you print the report the coverage percentage will not be representative of the real one because it will be a bit lower.

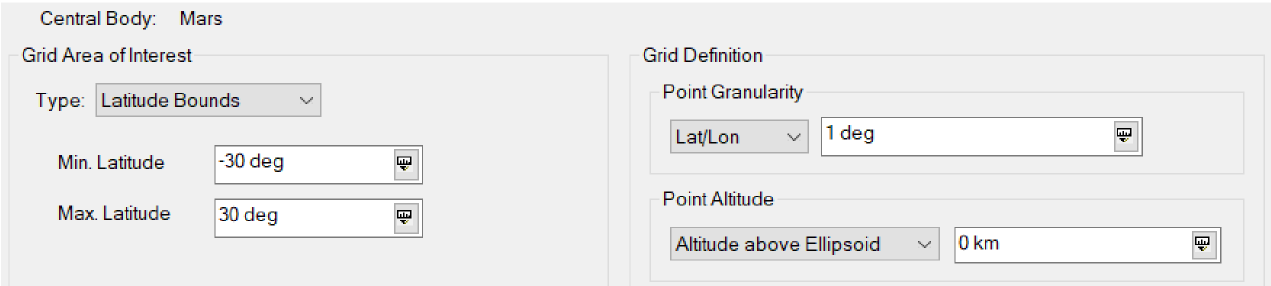


Figure 6: STK – Coverage Definition

The next step is to put on board of each satellite a sensor which represents the working camera during the mission. We will attach the sensor to the satellite and open its properties window on STK to adapt it to our case. We will choose to go with a rectangular Sensor Type with both Vertical and Horizontal Half Angle of 15.5°, then, remaining in the basic properties, we will set its resolution giving our sensor a focal length of 41.25 m. we will repeat these passages for each satellite.

Now we will insert of "Figure of Merit" and we will attach it to the "Coverage Definition". Again, we can open its properties to modify them. It is possible to go for a static access where STK highlights all the cells that the satellite has access to, without running the simulation. On the other hand, we can use a dynamic access where you see the various cells activate as the satellites fly over them during the mission simulation's animation.

In order to print the report of the total access and to find out the percentual coverage of the surface, we right-click on the "Figure of Merit" and select "Report & Graph Manager" and generate a report for the "Percent Satisfied".

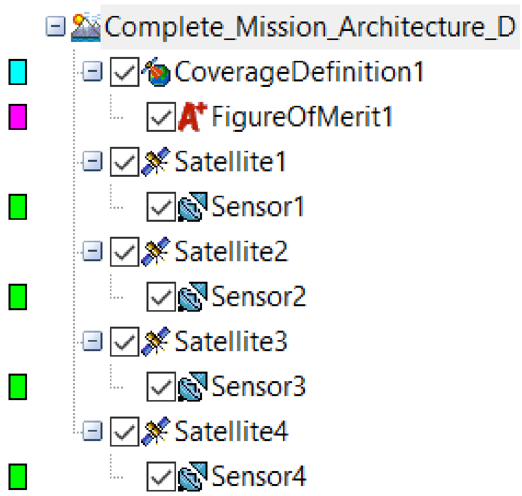
```
                         Coverage Properties
---------------------------------------------------------------------
Latitude Bounds Coverage
Min. Latitude: -30.0000 (deg)
Max. Latitude: 30.0000 (deg)
Grid Altitude: 0.0000 (km)
Ground Altitude set from grid altitude reference (On ellipsoid)
Resolution: 1.0000 (deg)
Number of Points: 18954
Assets required for a valid access: At Least 1
Assigned Assets:
    Satellite/Satellite1/Sensor/Sensor1:    Active
    Satellite/Satellite2/Sensor/Sensor2:    Active
    Satellite/Satellite3/Sensor/Sensor3:    Active
    Satellite/Satellite4/Sensor/Sensor4:    Active
Access Interval: 20 Feb 2022 11:00:00.000 to 28 Feb 2022 11:00:00.000
Regional Acceleration: Automatic
Light time delay: Ignored
Maximum Sampling Time Step: 360.000 secs
Minimum Sampling Time Step: 0.010 secs
Time Convergence: 5.000e-03 secs
Value Convergence: Relative 1.000e-08 - Absolute 1.000e-14


            FOM Properties
-------------------------------------
Simple Coverage FOM
Satisfaction: Greater Than 0.000000
FOM value range check is not enabled.


% Satisfied    Area Satisfied (km^2)
-----------    ---------------------
      99.73              72207151.65
```

We can see how the percentage of global coverage is over 99%.

Considering our requirements of quality and constraints we can say without problems that this is a fairly acceptable result and a very good percentage of coverage. In the case where we put a margin of 5° on the latitude limits, it is obvious that we would obtain a lower result. More precisely the final percentage would be about 91%. This happens because the satellites do not cover the margin areas.



In the image here reported, it is possible to see how we organized on STK in the "Object Browser" the various architecture elements, which we previously described and the reader can also see the colours code to better understand the Mars' map in the lower section.

Figure 7: Object Browser for Coverage

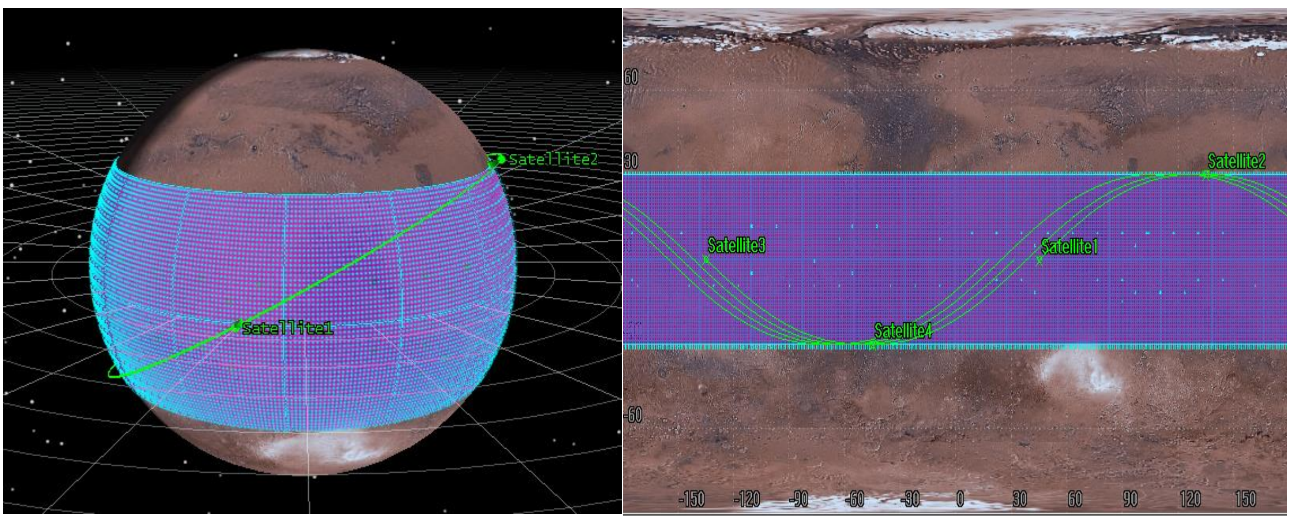


Figure 8: global Coverage Definition

2.7 Communication links

In the following section, a description of a basic Communication System will be presented to the reader. The mission has the goal to gather a large amount of data about Mars, its surface and the ground layers immediately under it, even for a mission like the futuristic Mars Sample Return that will have the duty to bring on Earth some samples of Mars' soil for the first time at the end of the operations, there will always be the need for the ability to have communications in both downlink and uplink between the three levels and altitudes: the rover on the planet, the satellites orbiting around it and the mothercraft on the highest orbit.

Through STK, it is possible to implement some tools to understand when and where the satellites have access to the rover or the mothercraft. In order to do that, we need a system based on an antenna and a receiver to evaluate the communication links.

Having clarified these aspects, we open the simulation on STK of architecture 4B, already prepared like it is described in chapter 3 of this report. If we have the four satellites in the right position in the starting orbit, the first step to make is to insert a rover on Mars' surface.

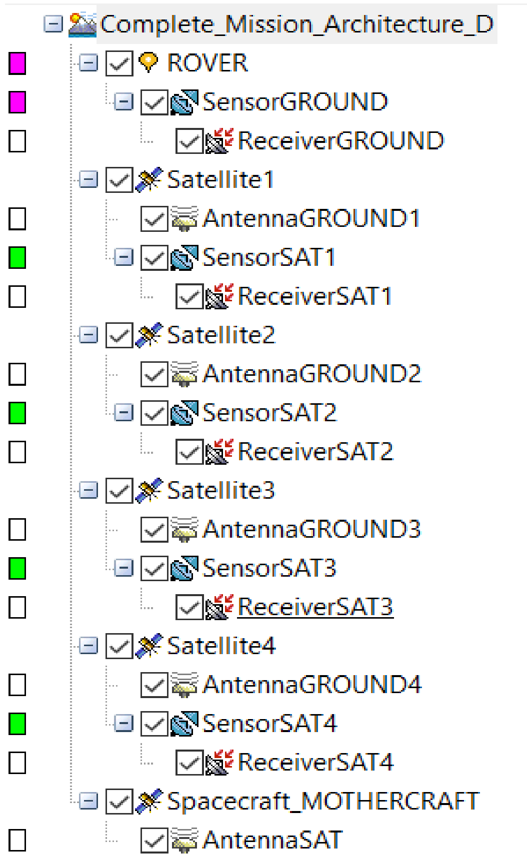
The first problem that we have to face is that STK does not contemplate a default possibility for a rover amongst the various options that can be inserted, the closest thing to a rover is a "Ground Vehicle"; however, rovers are typically extremely slow, the speed allowed by the current technology is about 10 m/hour, and, considering that we do not know the path, the rover will take during the real mission, at this point of the research a stationary point on the surface is simple enough to create a plausible simulation.

Therefore, we select "Place" and through the option "Define Properties", it is possible to select the coordinates that we desire. In this case, it is enough to change the latitude from the default value of 40,0386° N to 10° N to insert the "Place" into our area of interest. Considering that we are putting coordinates that for this phase of the research are completely random and that the longitude does not affect the fact the rover is in our area of interest, we will keep the default longitude value and we shall rename our place “ROVER”, following the hypothesis of approximation that we previously explained.

The next step is to attach a sensor to the rover, it will be kept as a simple conic type of sensor and its cone half angle is 25°. Another basic property that is going to be changed is the “Pointing Type”, and that is done selecting “Targeted”. We will then add as “Assigned Targets” the four Small-Sats of the architecture. Once we have finished to modify our sensor, we shall insert a receiver with a parabolic antenna and attach it to the sensor, we open its properties and begin the modifications to adapt it to the situation.

We will change the type from “Simple Receiver Model” to “Complex Receiver Model”, then we will change the antenna type that is by default “Gaussian” and we will make “Parabolic” with a Beamwidth of 50°.

Now, we shall insert an antenna on each of the satellites of the architecture, like we did before for the receiver, we shall modify it to have a parabolic antenna with a 50° beamwidth. These elements are what allows us to have communication between the ground compartment and the satellites. For the communication between the satellites and the mothercraft, we shall repeat the same passages, the only difference is that the receivers and the sensors will be attached to the satellites and the antenna will be a child to the mothercraft. Moreover, for every satellite, we will choose a “Assigned Target” the mothercraft.

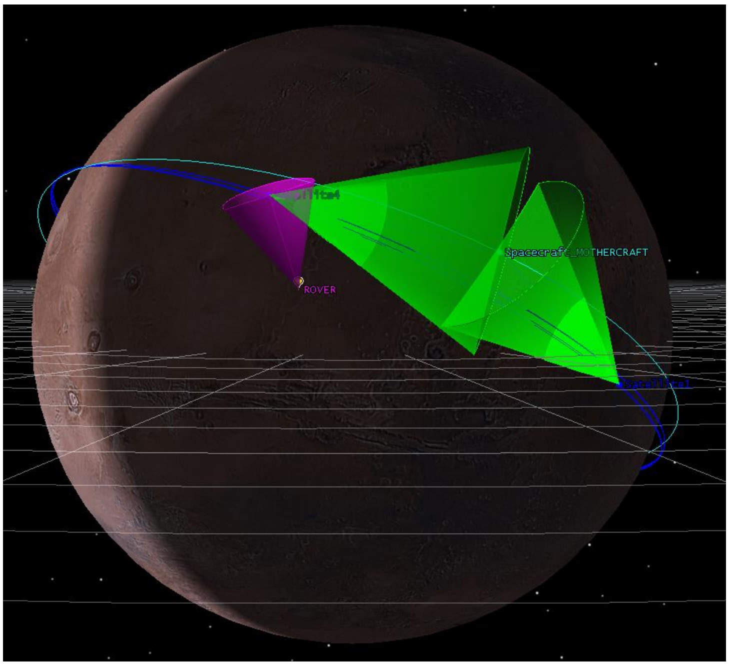


In order to differentiate the instruments for the two different communications, we will add "GROUND" to the name of each and every element involved in the communication between the four satellites and the ground segment, represented by the approximation of a rover.

On the other hand, we will add "SAT" to the name of the elements that are responsible for the downlink and uplink between the mothercraft and various Small-Sats of the architecture.

Figure 9: Object Browser for Communication links

In the following image will be presented a generic situation of communication between different elements of the constellation. The previous scheme is there to help the reader with the colour scheme used in this simulation to tell the different communication forms.



Satellite 1 and 4 are keeping a phase angle of 90° between each other.

The two green cones are the antennas' lines of sight of Satellite 1 and 4 and they are both communicating with the mothercraft on the higher orbit. The magenta cone is due to the fact the Satellite 4 is in the line of sight of the rover on Mars' surface.

Figure 10: generic Communication Links situation

The procedure described allowed STK to compute the accesses between two elements of the mission the rover on the surface, one of the four satellites and the mothercraft.

Like it was previously stated, we have two different communication types: Ground-to-Satellite and Satellite-to-Mothercraft. The inter-satellite link for this simulation was not considered because one satellite should never be able to see another one once they reach the operative orbits, because of the altitude and the phase angle between one another.

Now, in this section we are going to present the final extracts form the reports of the various accesses type.

The statistics of the various satellites will be reported in this section. However, it is necessary to make clear that the reports printed by STK gives the user the list and the characteristics of all accesses, but in real world mission we need to guarantee a minimum time of access between the transmitter and the receiver. If that limit is not reached, it is impossible to broadcast in an effective way.

For this reason, an access that lasts less than 200 seconds should not be considered, it would be like no communication was effected. If we have more stricter constraints about this topic, the time limit can rise to 250 seconds.

2.7.1 Ground-to-Satellite Communication

Considering the position that we have hypothesized for the "ROVER" and the dynamics of the four satellites of architecture 4B, STK gave as output the values of accesses for the global duration of the mission, from the release from the mothercraft to last phase, the disposal of the satellite on Mars' surface.

Now we shall present the reader the reports that presents the complete list of accesses, but they will be followed by the data that does not consider those which does not respect the minimum time constraint:

- **ROVER – Satellite1:** between these two elements, we have 3363 total passages

Global Statistics	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
Min Duration	3022	21 Jan 2024 00:07:04.188	21 Jan 2024 00:07:12.523	8.336
Max Duration	2	20 Feb 2022 20:00:28.716	20 Feb 2022 20:12:42.812	734.096
Mean Duration				383.695
Total Duration				1290367.072

We have 3013 passages that last more the 200 seconds

Min Duration	200.370s
Max Duration	734.096 s
Mean Duration	412.480 s
Total Duration	1242800.942 s

- **ROVER – Satellite2:** between these two elements, we have 3469 total passages

Global Statistics	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
Min Duration	2836	10 Nov 2023 19:09:48.229	10 Nov 2023 19:10:04.409	16.180
Max Duration	15	22 Feb 2022 20:23:31.476	22 Feb 2022 20:35:37.603	726.127
Mean Duration				392.356
Total Duration				1361083.362

We have 3127 passages that last more the 200 seconds

Min Duration	200.263 s
Max Duration	726.127 s
Mean Duration	420,802 s
Total Duration	1242800,942 s

- **ROVER – Satellite3:** between these two elements, we have 3140 total passages

Global Statistics	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
Min Duration	1539	19 Dec 2022 13:08:32.415	19 Dec 2022 13:08:40.533	8.118
Max Duration	15	22 Feb 2022 19:45:59.907	22 Feb 2022 19:57:52.076	712.169
Mean Duration				385.980
Total Duration				1211976.816

We have 2815 passages that last more the 200 seconds

Min Duration	200,085s
Max Duration	712.169 s
Mean Duration	415.216 s
Total Duration	1168824.361 s

- **ROVER – Satellite4:** between these two elements, we have 3254 total passages

Global Statistics	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
Min Duration	3048	30 Dec 2023 17:55:55.543	30 Dec 2023 17:56:03.902	8.359
Max Duration	14	22 Feb 2022 19:26:31.970	22 Feb 2022 19:38:51.274	739.303
Mean Duration				397.065
Total Duration				1292050.058

We have 2965 passages that last more the 200 seconds

Min Duration	200.173s
Max Duration	739.303 s
Mean Duration	423.347 s
Total Duration	1225222.724 s

2.7.2 Satellite-to-Mothercraft Communication

In this section, for the second type of communications, we need to highlight a difference, reporting the accesses between the mothercraft and the satellites while they are on the same orbit is not one of our points of interest.

Therefore, we need to print the various reports of the accesses from the moment the four satellites reach the 150 km altitude orbit to the end of the mission.

Like in the previous chapter, after the complete reports of all the accesses, we again report the accesses that are strictly superior than 200 seconds, showing the minimum, maximum, average and total duration.

The statistics of the various satellites are:

- **Satellite1-Mothercraft:** between these two elements, we have 2934 total passages

Global Statistics	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
Min Duration	2529	30 Jan 2024 20:53:35.938	30 Jan 2024 20:54:20.309	44.372
Max Duration	4	5 Mar 2022 11:24:53.953	6 Mar 2022 04:05:09.918	60015.965
Mean Duration				2426.229
Total Duration				7118555.705

We have 2900 passages that last more the 200 seconds

Min Duration	202.713 s
Max Duration	60015.965 s
Mean Duration	2453.268 s
Total Duration	7114477.902 s

- **Satellite2-Mothercraft:** between these two elements, we have 2848 total passages

Global Statistics	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
Min Duration	415	24 Aug 2022 04:17:29.234	24 Aug 2022 04:18:08.806	39.572
Max Duration	5	6 Mar 2022 00:12:20.090	6 Mar 2022 18:40:16.429	66476.339
Mean Duration				2622.403
Total Duration				7468603.456

We have 2816 passages that last more the 200 seconds

Min Duration	209.402s
Max Duration	66476.339 s
Mean Duration	2648.009 s
Total Duration	7464736.105 s

- **Satellite3-Mothercraft:** between these two elements, we have 2959 total passages

Global Statistics	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
Min Duration	2859	1 Apr 2024 05:52:01.095	1 Apr 2024 05:52:41.239	40.145
Max Duration	7	10 Mar 2022 13:42:03.830	11 Mar 2022 05:22:45.326	56441.496
Mean Duration				2389.011
Total Duration				7069083.688

We have 3013 passages that last more the 200 seconds

Min Duration	201,257 s
Max Duration	56441.496 s
Mean Duration	2416.007 s
Total Duration	7064404,982 s

- **Satellite4-Mothercraft:** between these two elements, we have 2818 total passages

Global Statistics	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
Min Duration	1097	31 Dec 2022 15:44:36.013	31 Dec 2022 15:46:01.828	85.815
Max Duration	6	8 Mar 2022 04:25:45.325	8 Mar 2022 23:01:45.648	66960.323
Mean Duration				2681.665
Total Duration				7556931.164

We have 2792 passages that last more the 200 seconds

Min Duration	202.405 s
Max Duration	66960.323 s
Mean Duration	2705.197 s
Total Duration	7552909.053 s

2.8 Trade off mission architectures

Trade-off analysis has been performed to identify the Mission Architecture baseline. The Figure of Merit (FoM) selected for this study are:

- **Coverage:** the amount of total area covered by Small-Sat(s) in relation to the timescales to cover it. Higher values are better.
- **Revisit time:** time between two passes of a satellite (one of the satellites of a constellation) over the area of interest (i.e rovers sites). Short re-visit time is better.
- **Amount of information:** amount of valuable data collected by space segment. High quantity and quality of data is better.
- **Δv:** total variation of space segment velocity to maintain the desired mission geometry. Low Δv is better because the mission is simpler and the quantity of propellant required is lower, higher volume for payload
- **Cost:** amount of costs foreseen for the mission, taking on 10M€ for a mission with one satellite and 2,5M€ for each satellite in a mission composed by a satellites' constellation
- **Operations:** includes all the activity required on ground (and onboard to perform the mission). Lower intervention from ground is better
- **Technology:** parameter that considers complexity of the on-board technologies and its TRL.

Once established the FoM weight through Analytical Hierarchy Process (AHP), the relative score of each architecture has been calculated, as reported in Table.

Table 18: Mission architectures trade-off

FoM	Weight [%]	1	2°	2B	3°	3B	4°	4B
Coverage	0,18	0,006	0,024	0,028	0,024	0,033	0,025	0,038
Revisit time	0,25	0,011	0,021	0,029	0,034	0,045	0,056	0,056
Amount of information	0,18	0,011	0,016	0,022	0,022	0,030	0,036	0,041
ΔV	0,07	0,018	0,011	0,014	0,006	0,010	0,006	0,006
Cost	0,07	0,010	0,014	0,014	0,011	0,011	0,005	0,005
Operations	0,18	0,027	0,036	0,022	0,033	0,019	0,029	0,012
Technology	0,07	0,018	0,012	0,011	0,009	0,008	0,008	0,007
Trade-off		0,101	0,133	0,140	0,139	0,156	0,165	0,166

For FoM such as coverage, revisit time and amount of information, the score values increase with the number of satellites; on the other hand, the FoM like ΔV, cost and technology the score values decrease. The missions with the highest number of satellites have the possibility of collecting more information and having a greater number of passages on point of interest, but they require more complexity to handle and communicate data. Mission architectures with swarm or distributed deployment are also considered because some FoM are influenced by this choice. Considering the final scores and the observation above, the mission architecture 4A and 4B can be chosen for the mission, but with further considerations, other parameters could be evaluated to analyze the mission architecture 3B, not far from the highest score obtained.

2.9 Payload Design

The optical payloads have a double scope within the mission because they serve both for mapping and for navigation. The reader can see how the research led to the identification of some COTS reported in Annex-Table 1. All the data in the table can be found in the product datasheets.

Mars shall be mapped in the visible spectrum in strategic areas for rovers' missions with an optical imager that guarantees a spatial resolution at least of 0,5 m at 80 km (TBC) of altitude, as specified in the various options of mission architectures. All cameras must guarantee the requirement for remote sensing applications at altitude of 80 km, that is accomplished for navigation purposes, where all cameras guarantee at least 0,6 m of at altitude of 150 km.

Mars shall be mapped in the spectral range to characterize different terrain typology and their chemical composition with a spectral range at least of 450-950 nm and a spectral resolution at least of 16 nm. For the operations related to the use of optical cameras a push-broom scanning technology has adopted, in order to decrease the technological and operations complexity and related costs.

2.10 Platform Design

The platform design is carried out on the basis of a well-established process involving functional analysis, definition of functional architecture and derivation of the product tree.

Payload deeply affects the spacecraft bus design. The management of images imposes capabilities in terms of quantity of data that should be handled and stored, computational power to execute image processing and stereoscopy algorithms, and data rate to transmit the image. Shooting images requires pointing accuracy to the target, pointing stability in order to avoid blurry pictures, and station-keeping capability to stay in selected hold points or virtual boxes. Moreover, payload mass, volume, and power consumption drive the subsystems sizing and the internal layout.

2.10.1 Block diagram

This section presents the high-level architecture diagram of the spacecraft. It can be considered as independent from the form factor, it holds both 12U platforms and higher solutions (Figure).

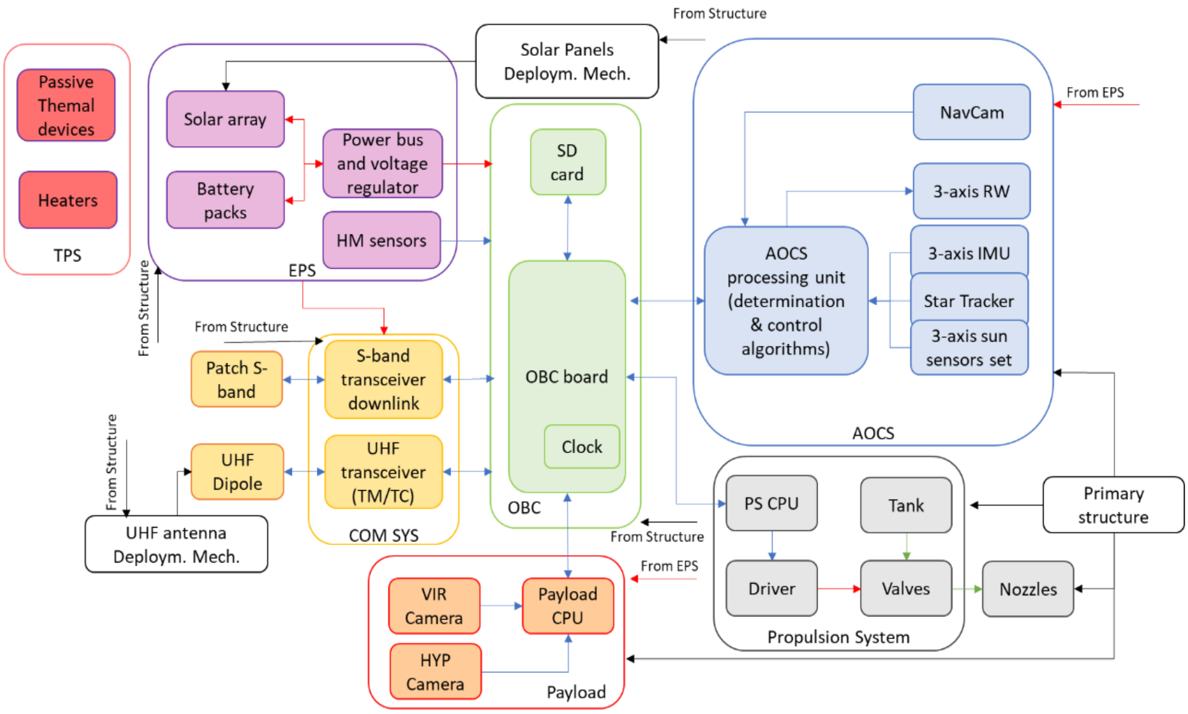


Figure 11: Satellite block diagram

2.11 Preliminary system design for the test bench

2.11.1 Use of Raspberry cameras in ALTEC / Thales Alenia Space Facility

The GSD value obtainable at an altitude of 10 m, assumed for the future tests in the ALTEC/TASI facilities, was calculated and reported in Annex-Table 6.

Table 19: Raspberry camera

RASPBERRY CAMERAS	GSD @ 10 m [m]	
Modulo camera V2	0,00441	
Videocamera HQ ufficiale	**Min**	**Max**
6 mm Wide Lens	0,00134	
16 mm Telephoto Lens	0,00059	0,00671
25 mm Telephoto Lens	0,00038	0,00429
35 mm Telephoto Lens	0,00033	0,00307
Zoom lens min	0,00117	
Zoom lens max	0,00019	

From the comparison of data in the table, the GSD calculated with the Raspberry cameras has a higher scale factor equal to 100 compared to the space cameras.

This scale factor is given by the size of the lens and the height from the ground. So, in test phase, it will be necessary to scale the objects and the distances identified from the Raspberry cameras by a scale factor equal to 100. It might be advisable to use a camera with a lens diameter around 30 mm to slightly reduce the difference between simulated environment and real environment.

In conclusion, Raspberry cameras are not spatial hardware, but they could be representative enough for the use in Thales Alenia Space facility.

2.11.2. System Architecture

Within the experimental framework of the SINAV program, a representative prototype of some of the main onboard subsystem is developed. In particular, it is of interest working on the image processing and payload management in order to test innovative algorithms to generate maps and, in case, paths planning.

Where possible, other aspects would be tested such as communication (I.e protocols to exchange info with rover and the mothercraft), and navigation (landmarking with real images).

To emulate the behaviour of a small sat in the ALTEC/TASI facilities, the prototype consists of:

Table 20: Subsystems

Subsystem	Components
Payload	Raspberry Cam (Hyperspectral Cam – TBC)
Onboard computer	Raspberry / Raspberry-based board
Com Sys	Wireless radio-module in S-band
Power System	Battery packs Power management Unit

3 OPERATIVE ORBITS DEFINITION

In this chapter, we shall present an overview of the work that has been done about the operative orbit study. An analysis of the mission's needs and requirements has been carried out in order to identify, among the possible architectures, which were introduced to the reader in the previous section of this report, the one that was more suitable for our scientific and technological objectives.

Following said Trade-off analysis, the architecture that has been chosen by the SINAV team is the architecture 4, further research and studies shall give deeper and more precise information about the choice between the "Architecture 4A" or "4B".

The following section of this paper will give an explanation of the study that was done, with particular emphasis on the software which was chosen for this work. The software STK (System tool Kit) by AGI gave us the opportunity to verify both the feasibility, the effectiveness and the efficiency of the solution that merged from the trade-off. However, before the analysis can be carried out, the STK software and the environment have to be prepared to possess the plausible and verified boundary and initial conditions. Only after that, the scenario can be programmed and the simulation can be launched.

Therefore, one of the first steps which needs to be made is preparing an accurate path of research with the purpose to gather the data we require about the environment of the simulation. In this case, of course, Mars is our target, this means there is the necessity to study its System, along with its gravity, its atmospheric aerodynamic resistance, the eventuality of the possible influence of a third body, such as its satellites, Deimos and Phobos, or of other massive objects in the Solar System near Mars, such as Jupiter, the Sun or Earth itself.

Once the data set and the simulation of the mission scenario are ready, we can begin our test and we can obtain the results we need to guarantee a possible future application of this technology in Mars environment. We will concentrate our Small-Sat distribution on the portion of Mars which constitutes our area of interest, in this case between a Latitude of 30° North and 30° South which leads to a fixed inclination i=30°.

STK will need a propagator which is made by gathering the characteristics previously exemplified, to have the initial conditions to solve the non-linear equations of the problem in exam, calculating through numerical methods the solution.

3.1 Chosen Architecture

3.1.1 implementation on STK

First of all, we need a better comprehension of the difference among every scenario, this will naturally lead to an implementation of all the possible architectures we have been chosen to analyse, progressing from the most basic to the most advanced. This is what allowed us to make our considerations about the various voices of the trade-off analysis, giving us the chance to develop our consideration about the parameters of interest of the mission; therefore, we now have a more precise idea of the values of Δv, Revisit Time, Coverage and amount of information of each mission, allowing us to understand which one is the most convenient to satisfy our constraints.

Now, we shall give the reader an overview of the possible scenarios along with the images of the constellation of Small-Sats around Mars both in a 3D map and in a 2D map.

The 3D images of Mars show all the orbits the Small-Sat will complete during the course of the mission. On the other hand, the 2D image, shows only the orbit the satellite is completing in that period. Moreover, the 3D map will be presented both with prospective view and with a top view, adding to said image an equatorial grid.

The first image shows the architecture 1

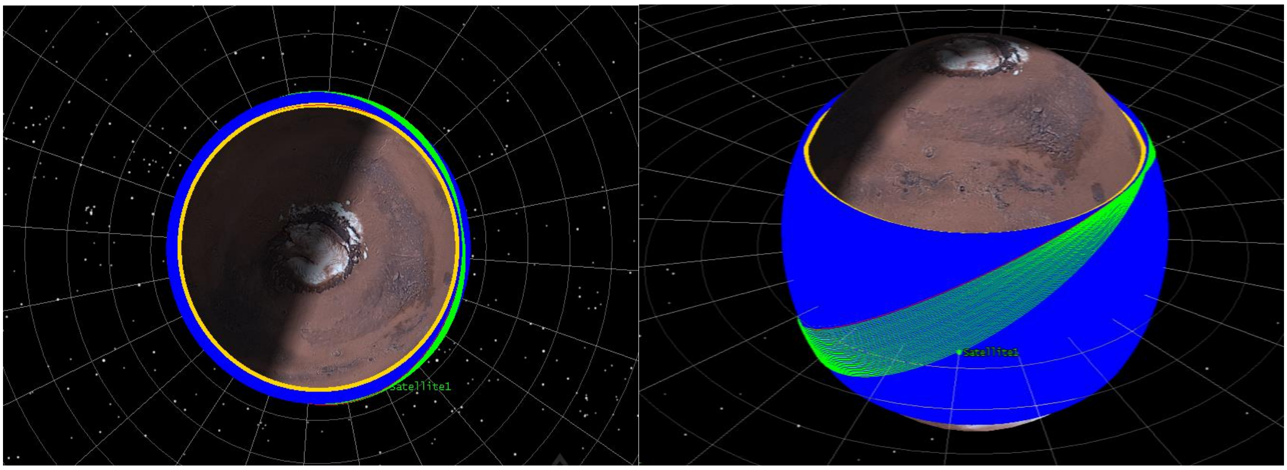


Figure 12: Architecture 1 – 3D maps

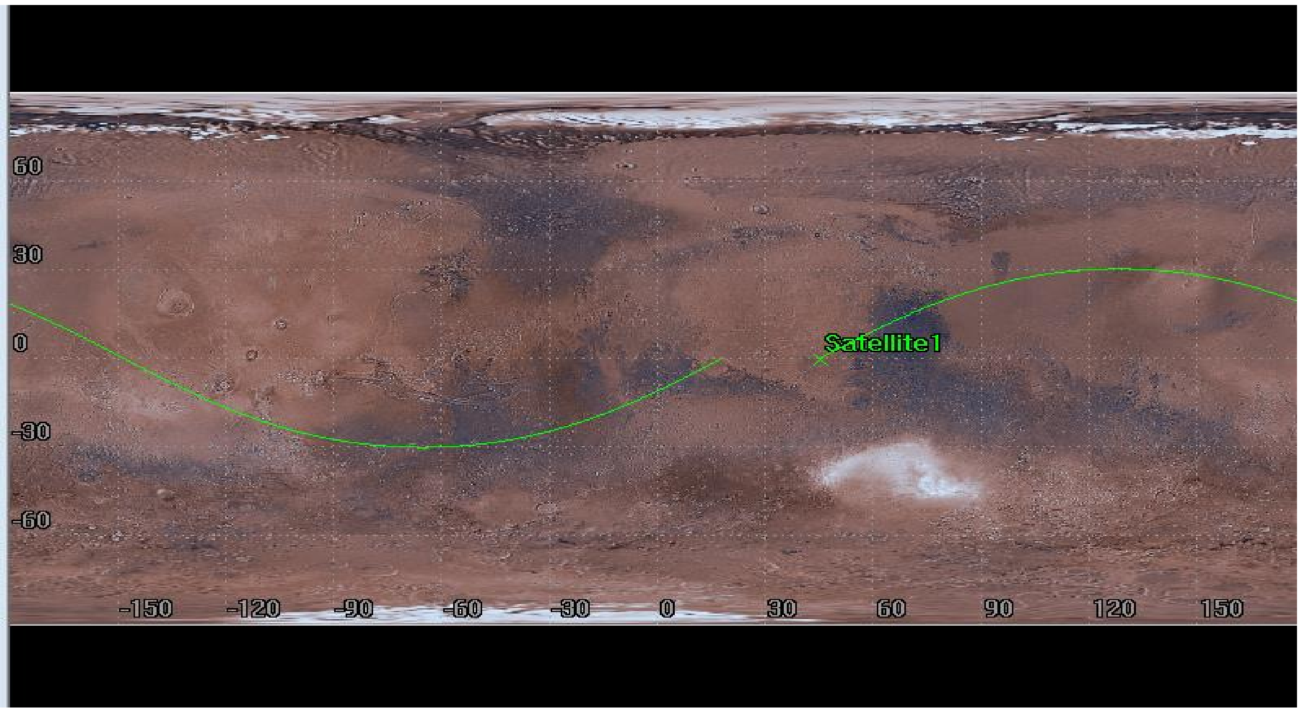


Figure 13: Architecture 1 – 2D map

Secondly, we report the images of the architecture 2A and 2B

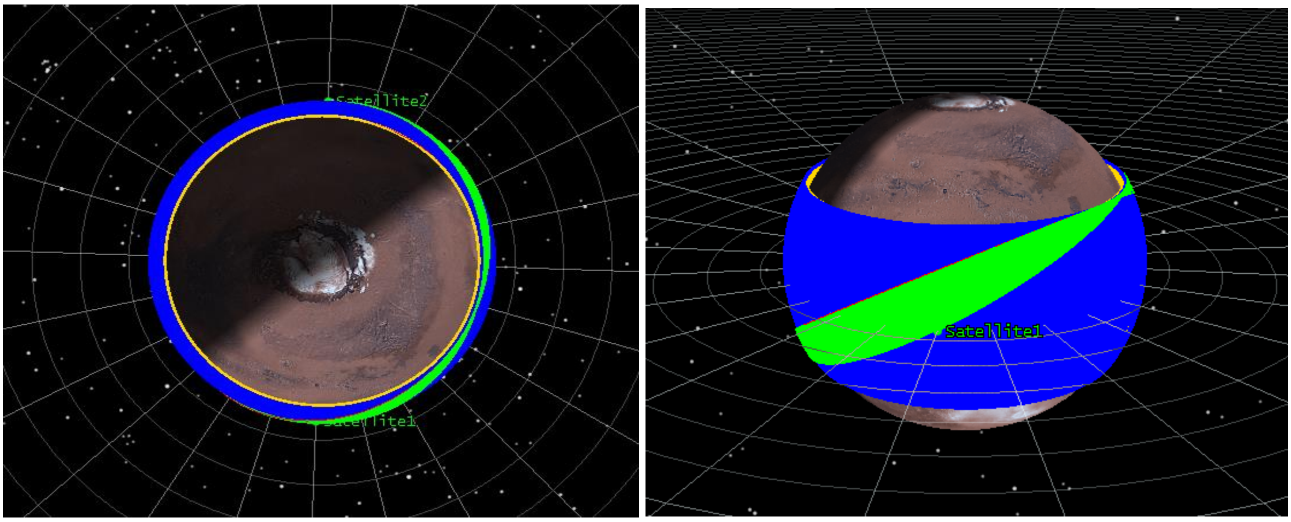


Figure 14: Architecture 2A and 2B – 3D maps

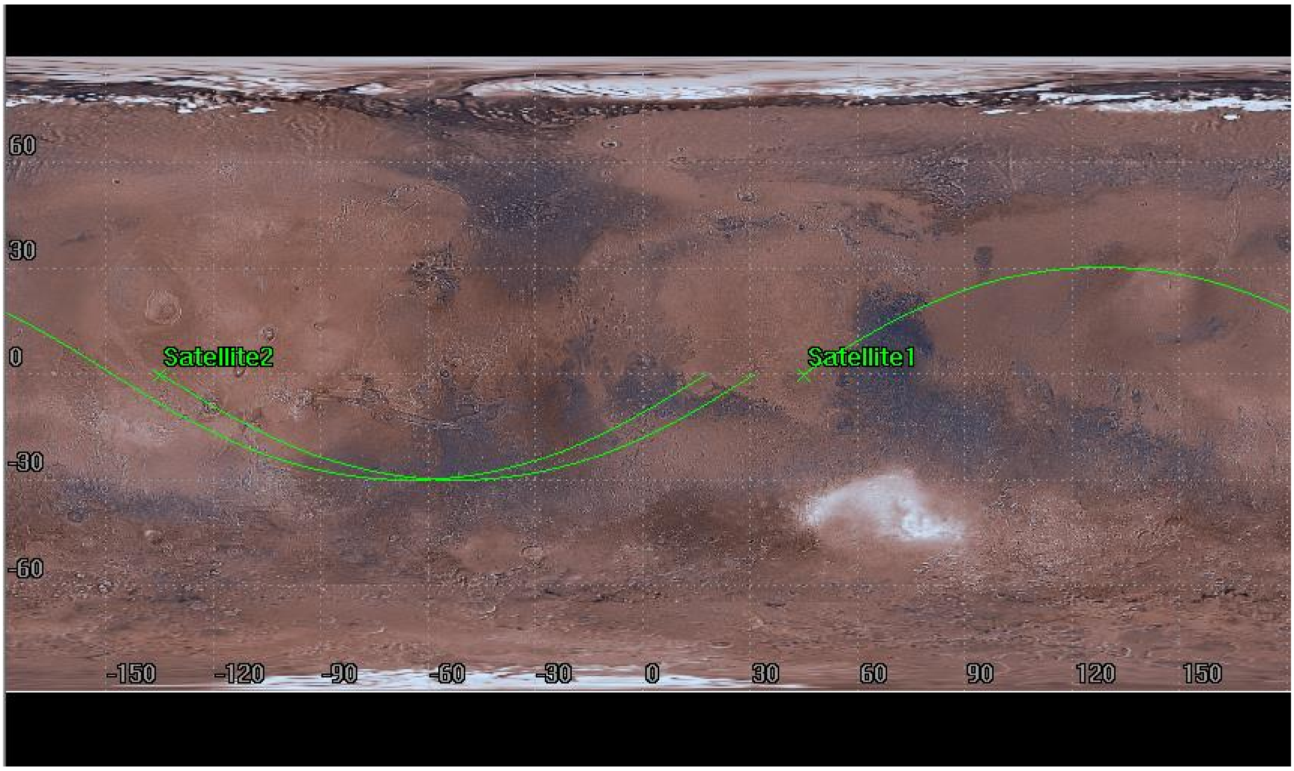


Figure 15: Architecture 2A and 2B – 2D map

Here, we report the images of the architecture 3A and 3B

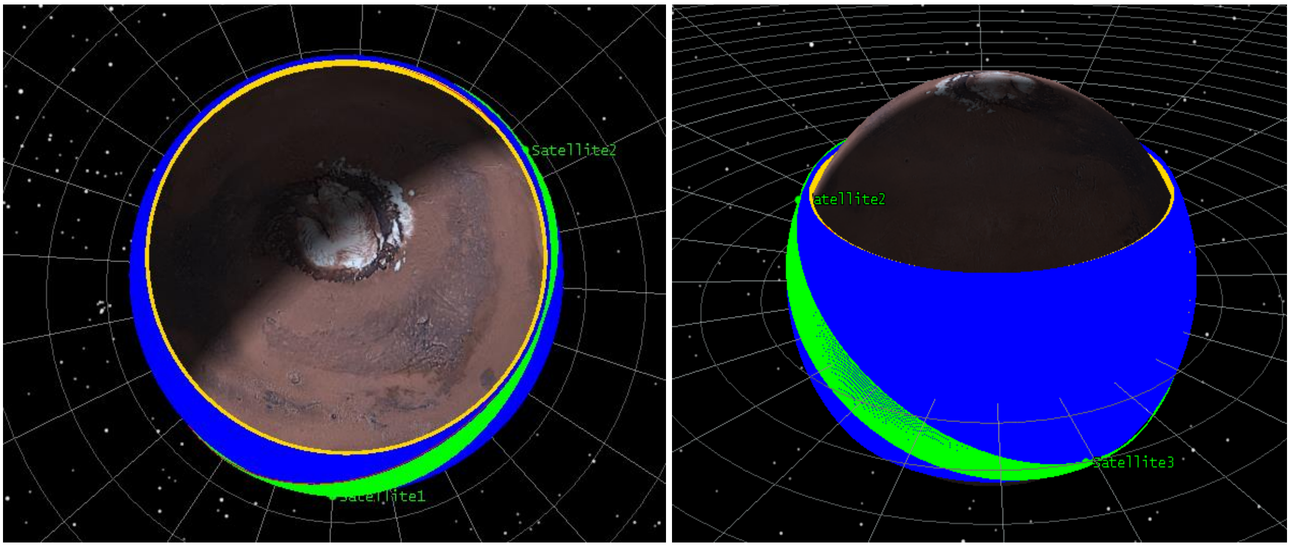


Figure 16: Architecture 3A and 3B – 3D maps

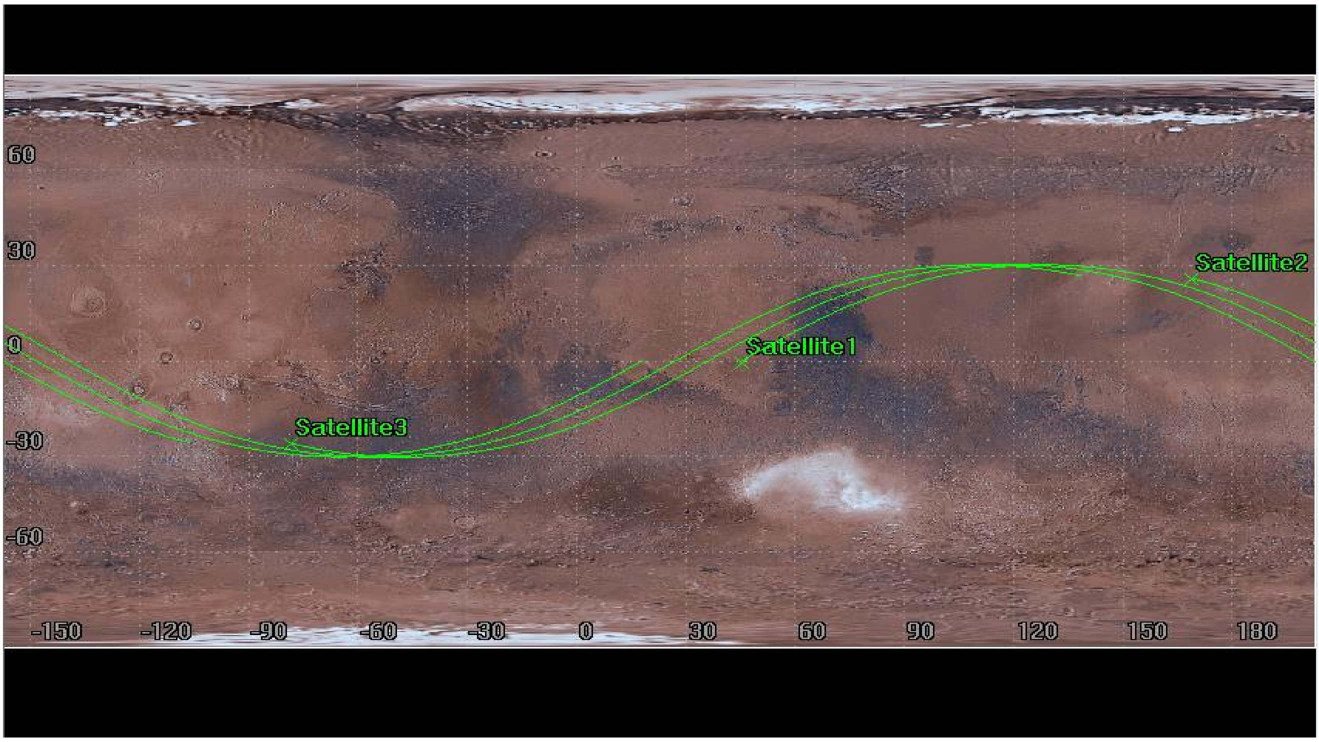


Figure 17: Architecture 3A and 3B – 2D map

And lastly, we report the images of the architecture 4A and 4B

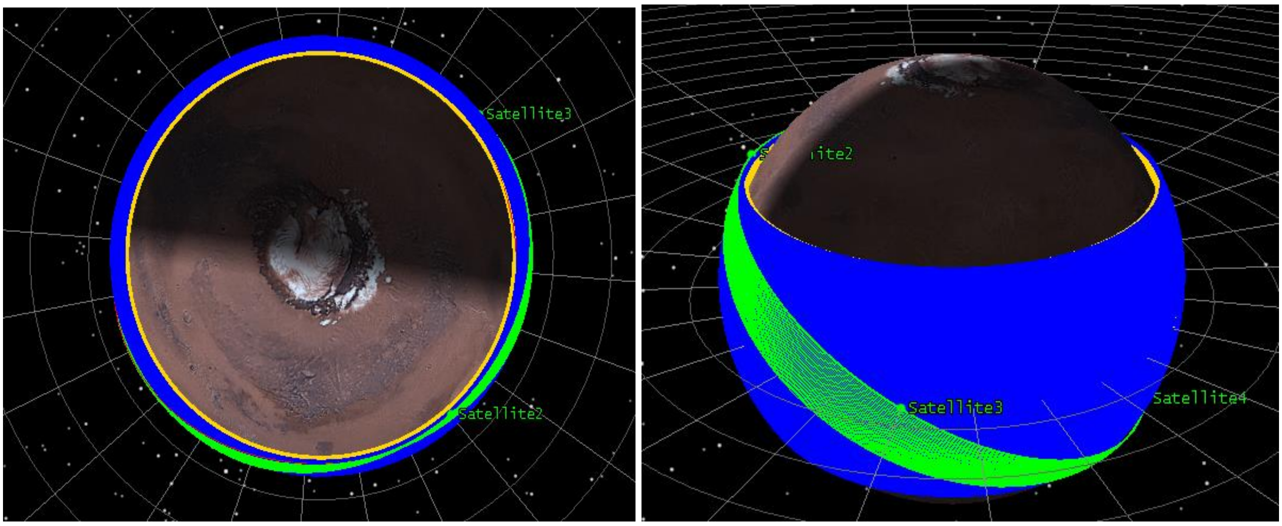


Figure 18: Architecture 4A and 4B – 3D maps

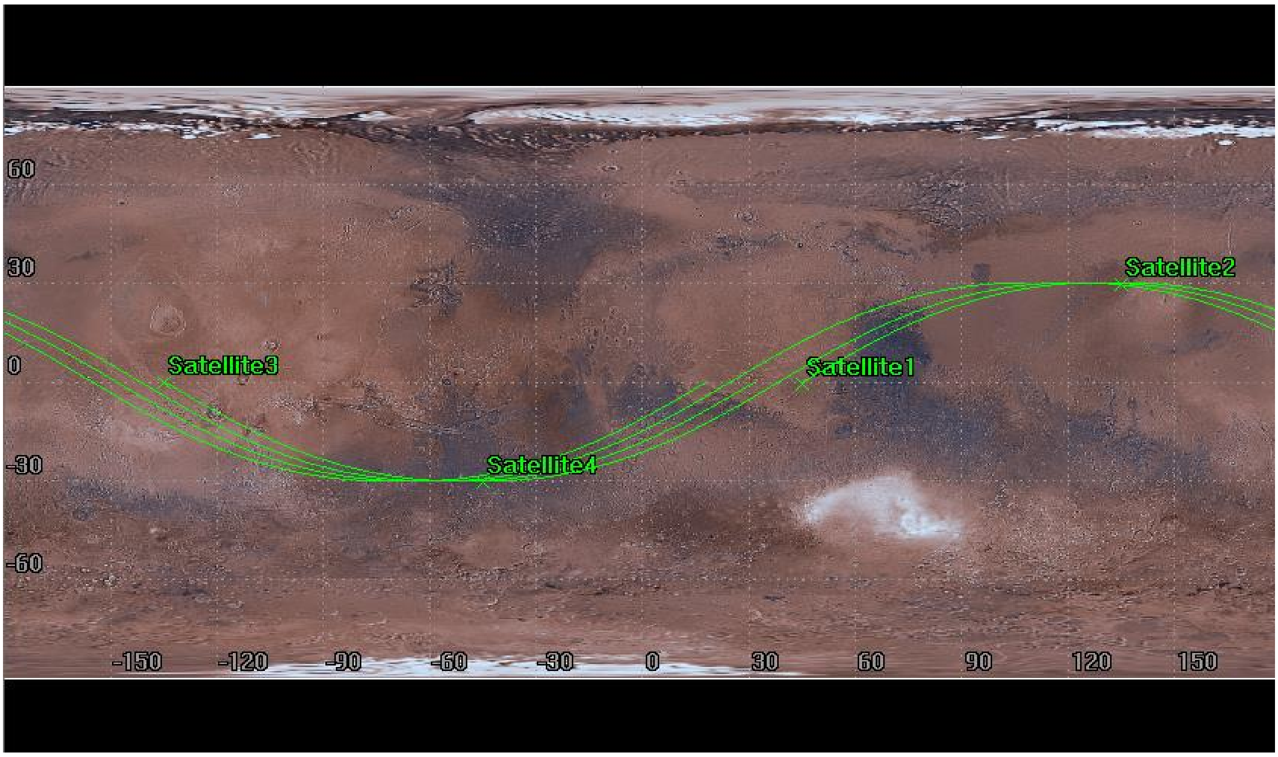
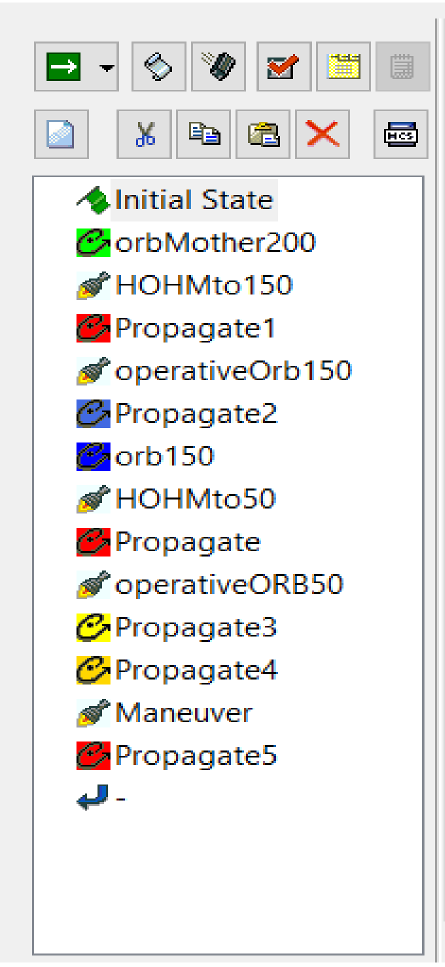


Figure 19: Architecture 4A and 4B – 2D map

From the 3D images, we can understand that it is divided into three stages, which corresponds to three different altitudes for the mission:

- Mothercraft altitude: 200 km
- First stage altitude: 150 km
- Remote sensing application altitude: 50 or 80 km



The initial orbit at 200 km is represented in "green".

The various Hohmann transfers between the various stages of altitude are coloured "red" and they are barely identifiable in the images because the duration of a transfer is incredibly briefer compared to the duration of each mission phase.

The orbit at 150 km, which corresponds to the revisit time phase, is coloured "blue" with differen shades.

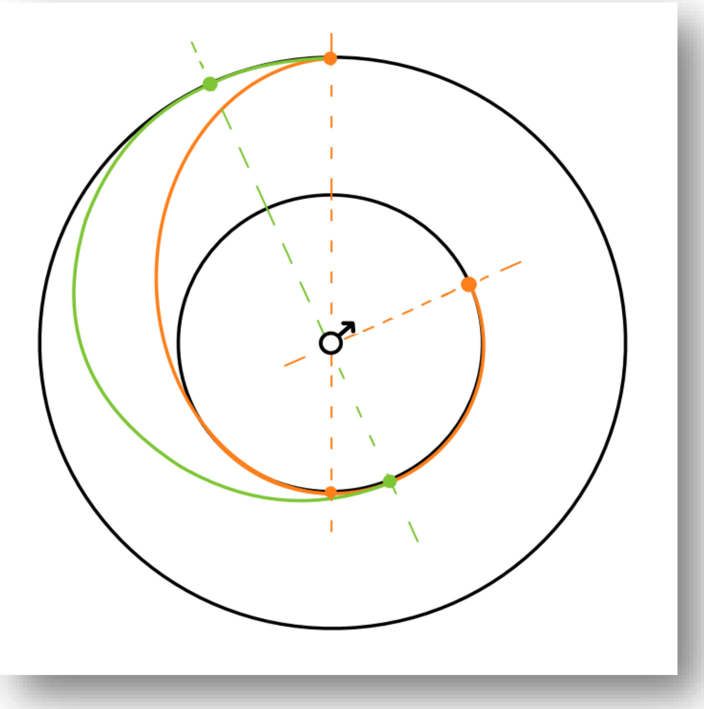
The orbit at 50 or 80 km, which corresponds to the remote sensing application, is coloured "yellow" with different shades.

Figure 20: Object Browser for Mission Architecture

3.1.2 Peculiarities of Architecture 4B

As it was reported in the chapter about the trade-off among the different architectures, the one that guarantees the most satisfying values of Coverage, Revisit Time and Amount of Information in the shortest period of time is Architecture 4, both 4A and 4B. The only notable divergence between these two mission solutions is how the satellites' deployment is organized.

In the architecture 4A, the mothercraft releases the Small-Sats at the same time all together, but they will perform the orbital transfer from an altitude of 200 km to 150 km in different moments in order to ensure an angle of phase $\varphi = \pi/2$ among all satellites. On the other hand, in the architecture 4B the mothercraft releases in the highest orbit the Small-Sats in different moments and, as soon as they are free to perform the Hohmann manoeuvre, they start descending towards the lower orbit. They are deployed from the mothercraft with the same angle of phase φ.



In this picture, we can appreciate a graphic representation of the 200 km altitude and the 150 km altitude orbit around Mars. This scheme does not show the dimensions in scale, because the two trajectories would have been too close to be understandable, considering they are significantly smaller than Mars Mean radius.

Figure 21: Phase Angle Scheme

- **Satellite A**
- **Satellite B**

These two satellites represent a generic system of artificial body that start off in the same place, we can say the mothercraft in this case, and then descend on a lower orbit with the same manoeuvre, but in different moments, therefore, once they arrive in the new orbit at 150 km of altitude, the two satellites will be out of phase. In order to calculate the phase angle φ, or "θ_B", as we will call it from now on, from the two points in the higher orbit from which the two Hohmann transfers start for the two satellites, we need to analyse the gravity system of the mission and his dynamics. Before we can begin, we need to understand that this will be an approximation and the hypothesis which we consider need be made explicit.

The first hypothesis is to consider both planet Mars and the orbits perfectly circular, this is clearly impossible, but the effects of the non-circularity have effect on a longer scale of time compared to the one of the transfers, therefore, they can be neglected. The second simplifying hypothesis is to assume that the various Δv for the manoeuvres is totally impulsive. To conclude, the last one is that no external effect is considered and does not affect the orbits and the transfers in any way. This allows us to neglect the any kind of influence from the Sun, from Mars satellites, Deimos and Phobos.

The starting point of the problem is when both satellites are together in the 200 km orbit and they are preparing to descend. We shall assume the problem has arrived at its conclusion when they are both in the 150 km orbit with difference of phase $\varphi = \pi/2$ rad.

t_{in}: initial moment $\quad$ t_{fin}: final moment $\quad$ $\Delta t = t_{fin} - t_{in}$: total duration of the problem

During a period of time which lasts Δt, Satellite A completes a Hohmann Transfer and treads an orbit portion that describes an angle equal to θ_A in a time "t_A". On the other hand, Satellite B, during the same amount of time, treads an angle of θ_B in the higher orbit in a time "t_B" and then descends.

Assuming that the time both satellites need to complete the Hohmann Transfer is the same and it is called "t_H", if we want Satellite B to arrive to the lower orbit when Satellite A is π/2 rad away then it is obvious that:

A) $\Delta t = t_H + t_A$

B) $\Delta t = t_H + t_B$

Therefore, the logical consequence is that:

$$t_H + t_A = t_H + t_B \quad \longrightarrow \quad t_A = t_B \quad \text{and} \quad \theta_A = \theta_B + \pi/2$$

$$t_A = \theta_A/\omega_2 \qquad t_B = \theta_B/\omega_1$$

ω_1 : angular velocity on the 200 km altitude orbit

ω_2 : angular velocity on the 150 km altitude orbit

$$t_A = t_B$$

$$\theta_A/\omega_2 = \theta_B/\omega_1 \quad \longrightarrow \quad (\theta_B + \pi/2)/\omega_2 = \theta_B/\omega_1$$

$$\theta_B (1 - \omega_2/\omega_1) + \pi/2 = 0 \quad \longrightarrow \quad \theta_B (\omega_2/\omega_1 - 1) = \pi/2$$

$$\longrightarrow \quad \theta_B = (\pi/2)(\omega_1/(\omega_2 - \omega_1))$$

In order to calculate the angle to calculate θ_B, a simple Matlab script can be prepared. It contains, as inputs, the data that are necessary to calculate the result:

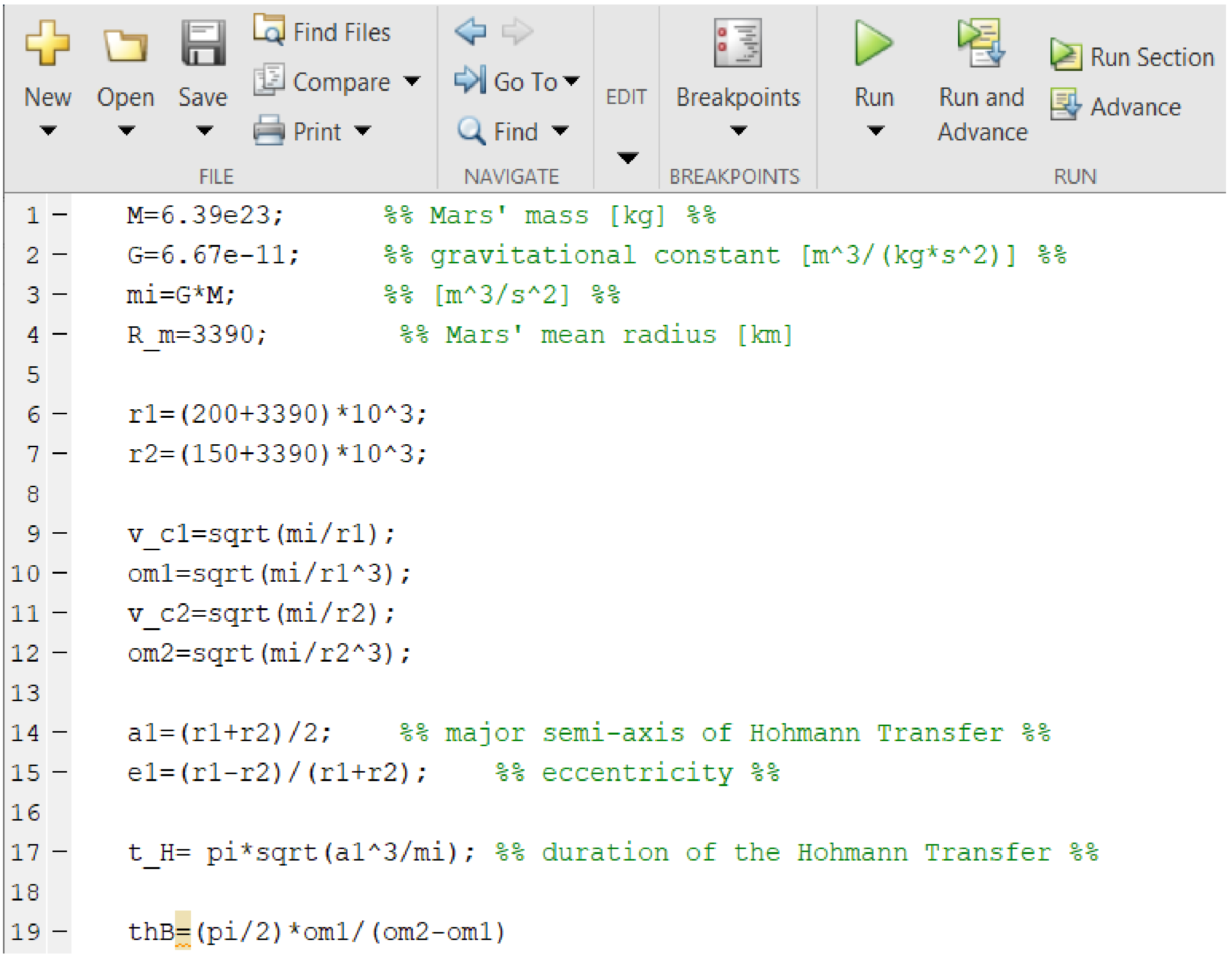


Figure 22: Phase Angle Matlab script

The result show that the phase angle which the Satellite B needs to have difference of phase required at the end, is :

$\theta_B = 4.7633$ rad

3.2 Mars Propagator – Mars HPOP

3.2.1 Propagator Functioning

STK gives the user the possibility to see the evolution of the satellite's trajectory as long as the mission progresses. Once you program the mission and you select the right environment when you open STK, to ensure the correct development of the calculation, we need to ensure the right propagator with the environmental data and the peculiarities of Mars star system, otherwise the solution of the differential equations won't converge to the result expected giving a sequence of local positions that is not related to any real, or even likely, situation.

The initial condition of STK has the basic orbit Earth Propagators "Earth HPOP Default v10", containing the atmosphere for modelling the resistance and lift, the file about the "Moon" as a third body for its gravity and many other aspects can be added and modified to make the simulation more and more precise. First of all, once we have selected Mars as an environment of work, in the main menu of STK, we need to go to the voice "Utilities" e select "Component Brower":

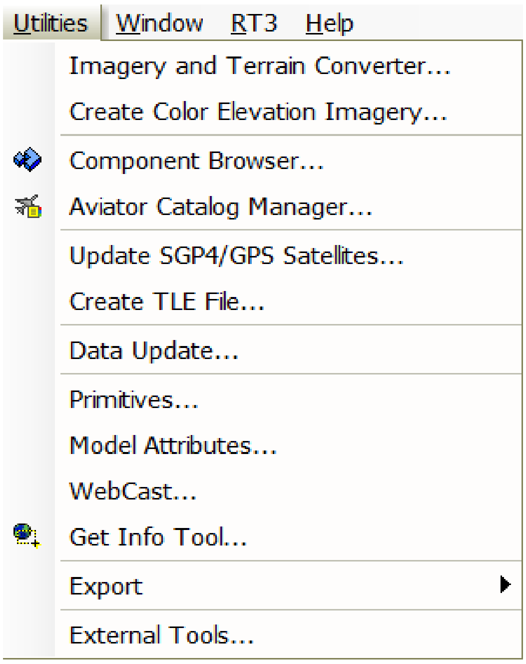


Figure 23: STK - Utilities

The "Component Browser" leads to a menu where many options of modification can be found and, utilizing them, one can adapt the mission to different circumstances.

This allows to modify the STK default settings for the propagators to prepare the numerical simulation and calculate the manoeuvres and the orbit which one is going to insert.

Starting from the propagator that si originally implemented on STK we begin our modification, deleting all the characteristics which are proper of the Earth gravity System and inserting the properties of Mars. In order to achieve this goal the first step we need to make is to duplicate the propagator “Earth HPOP Default v10”, give the copy a new label to rename and open it. In this case we have chosen as a new name “Mars HPOP”.

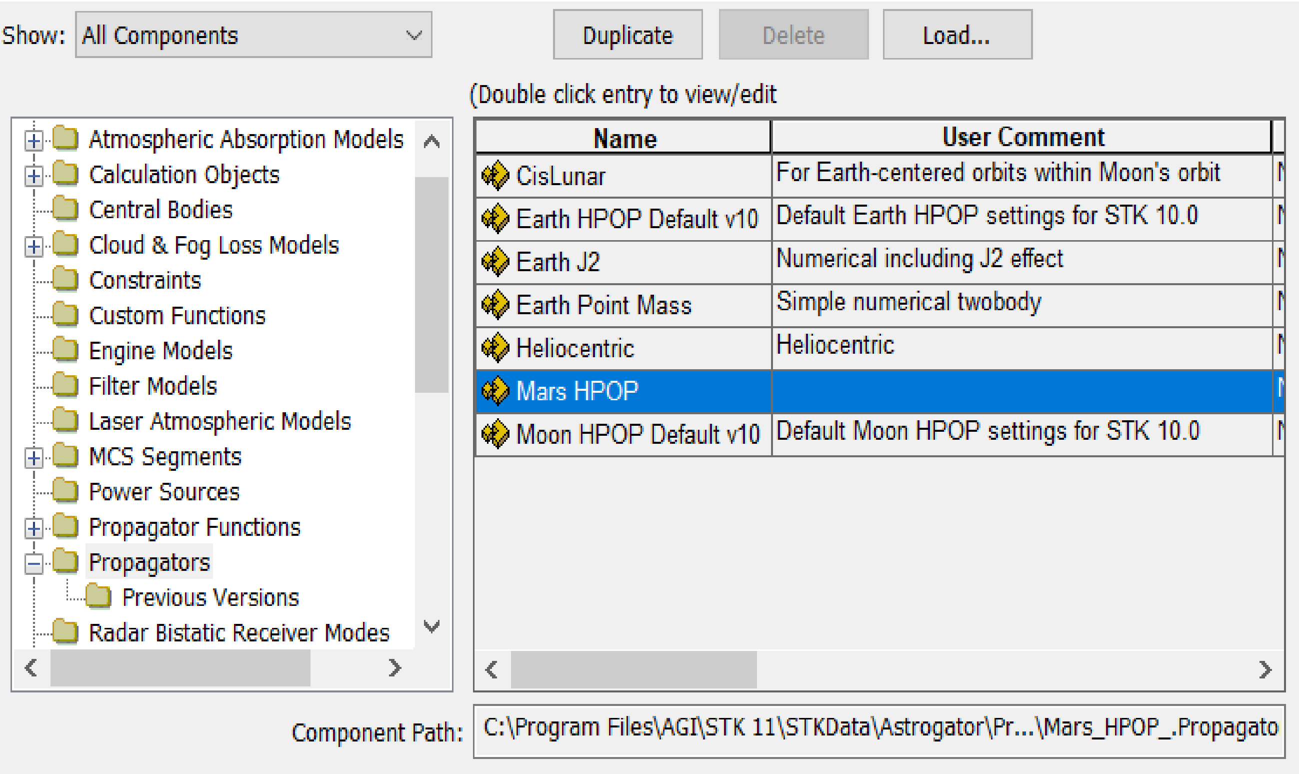


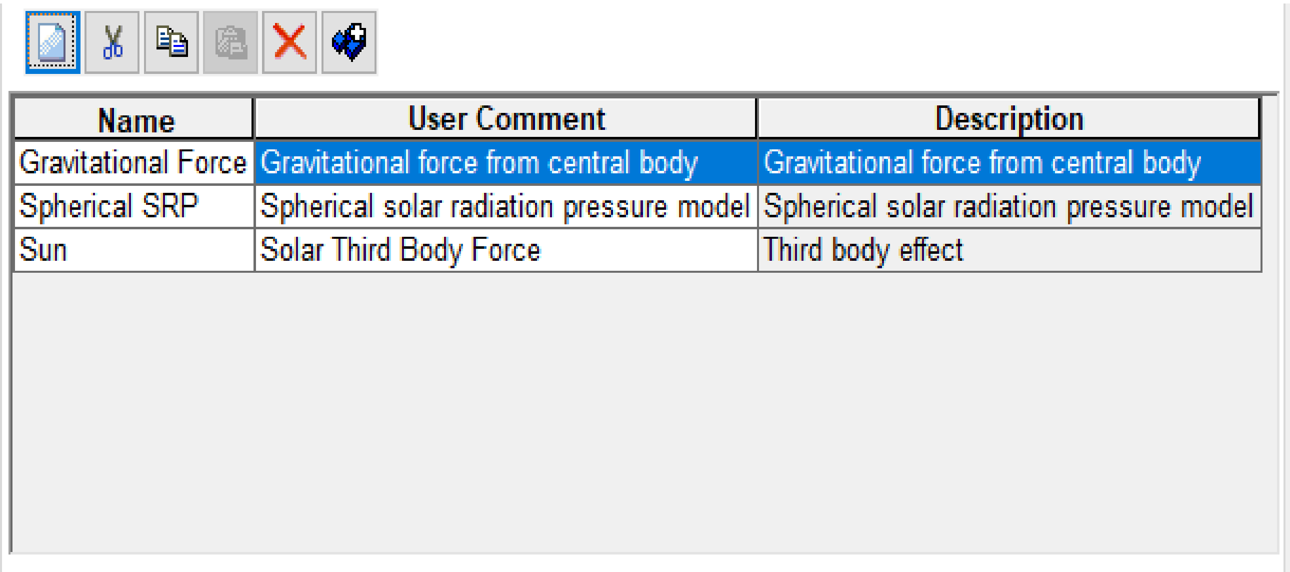
Figure 24: STK – Component Browser

After we have opened the copy and we have cleared the propagator’s copy from everything that is not strictly necessary for our mission, we can begin adding the elements that are required in order to achieve an acceptable simulation from the environment in which the mission moves.

3.2.2 Elements of Mars’ environment in the Propagator

In these circumstances, the most logic way to proceed is to begin with an easier and basic simulation that must contain only the elements that are strictly necessary to program a likely and plausible scenario. This will help the reader to focus on the targets that are the most important for the success we must expecting. Having a simpler, and yet still solid, base to build our process on, could be extremely helpful for many reasons.

One of them can be that, in case of failure or error of the system, it is easier to search for a mistake in the procedure if less elements are present. Moreover, this also represents a necessary, but not sufficient, condition for a right simulation. Once we are sure that everything proceeds as it has been foreseen, we can begin our further modifications and, as previously explained, considering that now we have proved the effectiveness of the basic propagator, if an error occurs from now on, we shall be sure that the problems will have their origin from now on.



Name	User Comment	Description
Gravitational Force	Gravitational force from central body	Gravitational force from central body
Spherical SRP	Spherical solar radiation pressure model	Spherical solar radiation pressure model
Sun	Solar Third Body Force	Third body effect

Figure 25: Mars HPOP

In order to add the elements, the simulation need, we shall go to the menu "new" where we can find many elements to upgrade the model with. This will help to create different types of scenarios for the same mission keeping into account various aspects. Moreover, the possibilities of modifications can change from the STK version that one is working with, the more recent the version is, the more elements can be found:

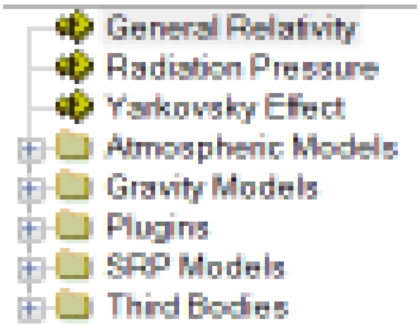


Figure 26: STK - Propagator

The possibility to consider effects that involves Relativity phenomena, Yarkovsky Effect allows us to have a much more precise idea of the mission.

The possibility to add various types of third bodies permits to consider their gravitational influence on the ideal orbit that were programmed.

It is undeniable that this complete guide of elements we can insert in our propagator ensures the possibility to define and foresee many aspects of the mission. However, the reader must understand that a downside is present. Not every information or specification is strictly necessary for the mission we are planning to complete.

An evidence of this, for example, is the fact that the relativity phenomena act on a scale of time that are completely different from the duration of our mission and it is at a level of precision superior to the one required for this paper. Another example is the Yarkovski effect, which is evident only for astronomic bodies with relatively small masses, generally asteroids and meteoroids are affected by this kind of phenomena considering that they have an inertia which allows them to be disturbed by variations of temperature and heat flow between the side enlightened by the Sun and the on in the dark. Also, this kind of bodies do not have any kind of atmosphere that distributes the heat on the surface.

With this statement in mind, the person who programs this type of mission must have clear what is the precision level we are seeking. And that every modification you make to your model will generally be translated into a longer amount of time required by STK to finish the simulation, this means a higher computational cost.

3.3 STK orbit analysis

When STK is opened, in order to have the right approach to the mission and the possibility to begin the research in exam in the correct environment, the first step one has to take is to go to "view" and to select "Planetary Options". Doing so, when the user arrives to the following page, they will be given the chance to choose the central body which shall be the origin of the local reference frame

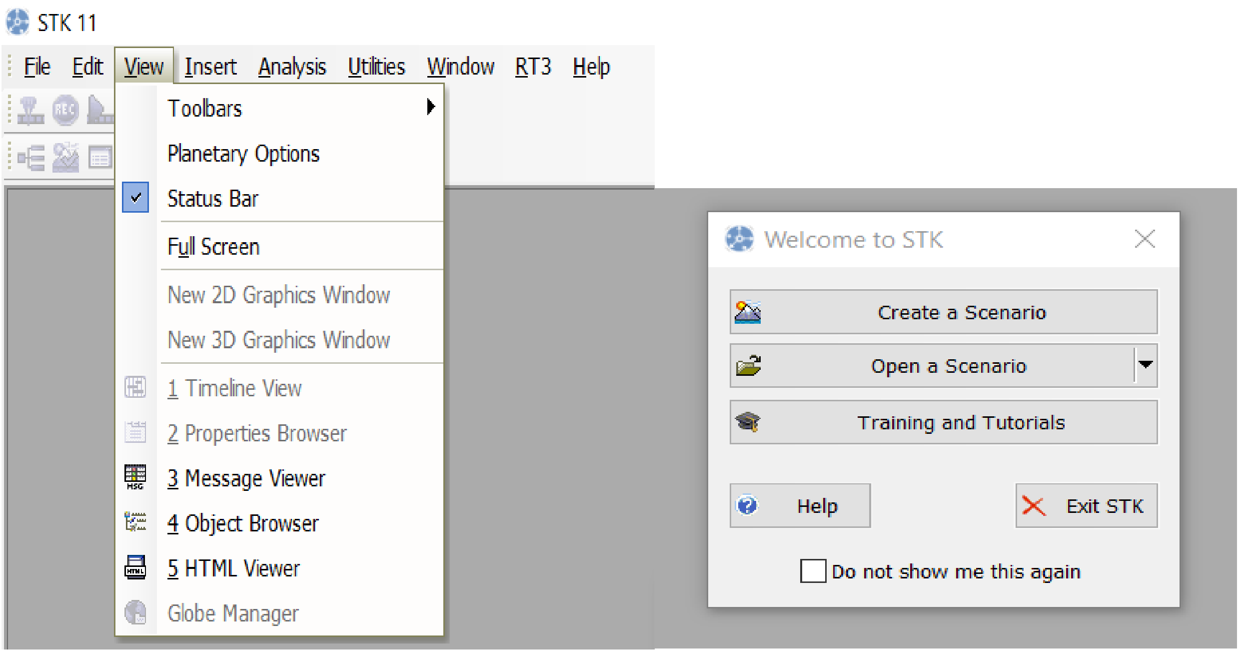
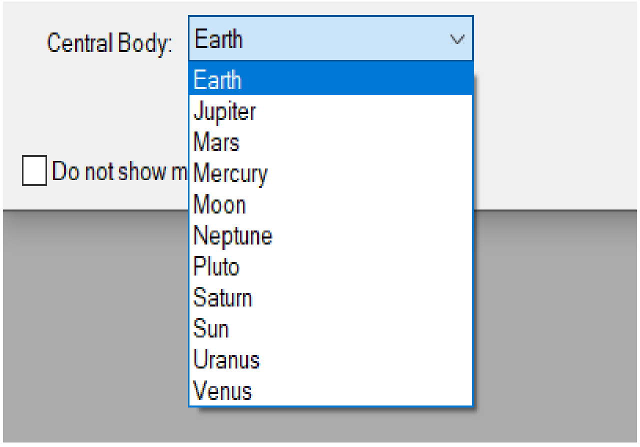


Figure 27: Creation of a scenario



As it is here reported, the version of STK used for this simulation allows to choose amongst every planet of the Solar System, adding to this scheme the Sun and the Moon as well.

Figure 28: Central body's definition

In this case, we are, of course interested in Mars as a "Central Body". However, if you needed to select a central body different from these, it is possible to insert their characteristics, such as mass, form, gravity, atmosphere where it is present along with other data, into the "Component browser".

Once the scenario is ready, STK allows the user to add a large number of tools, or "Objects", in the Work Environment through the command "Insert". It will now appear a window where the options are gathered and prepared for use.

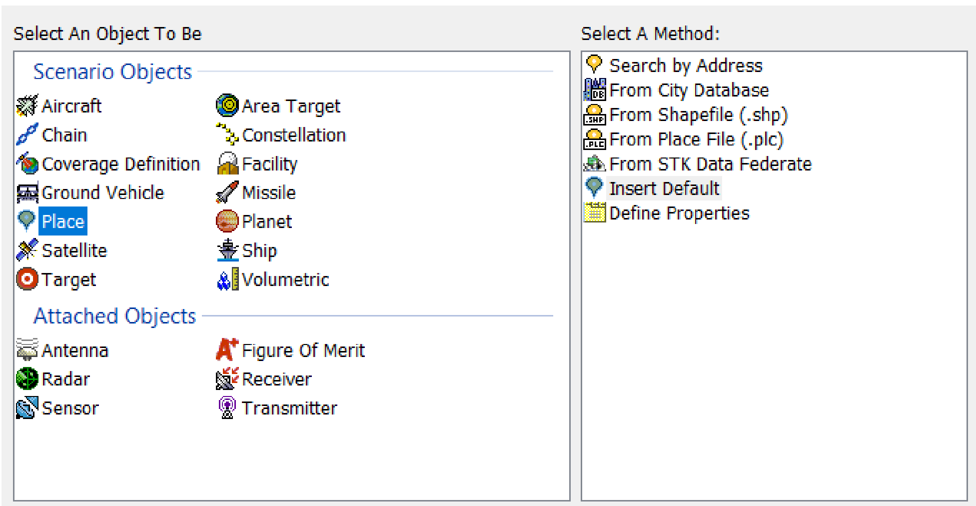


Figure 29: STK - Insert options

As it can be seen here, it is not only possible to select a scenario object like a place, a satellite or an aircraft, a "Scenario Object" in general, but also it gives the user the possibility to add elements to the object previously chosen.

This helps to get the results of more specific duties which you want the mission to accomplish. In the case which we are explaining in this report, we are interested in having a satellite or a group of satellites orbiting around the Red Planet, so we can proceed by selecting only one Satellite, for reasons we are going to unravel in the following paragraph. Once the Object Satellite is inserted, we can click on it and open the first voice "Properties" where we will be able to make the modification the mission requires.

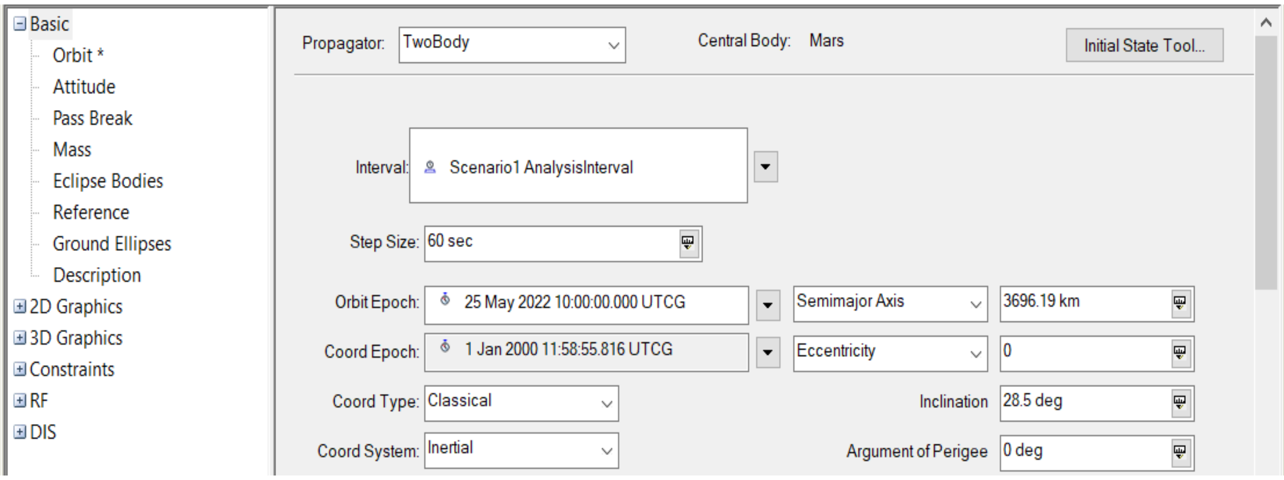


Figure 30: STK - Astrogator

You can see in right-hand side all the options that can be changed to the default setting of STK. Our first modification is to change the Propagator from "TwoBody" to "Astrogator". Only after this step, we can begin to program the steps of our mission.

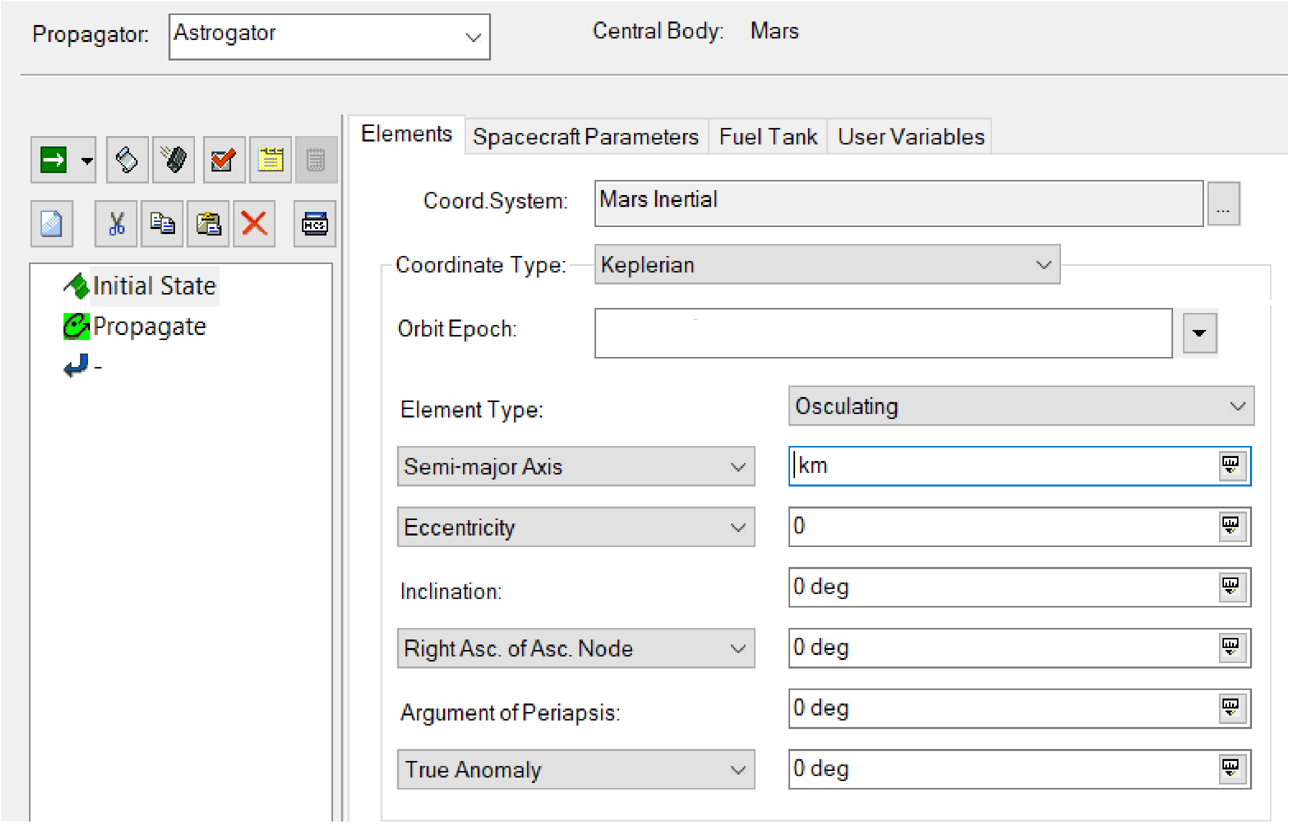


Figure 31: Insertion of the initial conditions

After that, we shall modify the “Initial State” in order to insert the six astronomical parameters that define the position and the asset of Satellite around the central body, the starting time, and other parameters like the initial mass of the Small-Sat, the propulsion system or the type of thruster, along with several other options. In the case we are considering the starting condition of the “Initial State” for the SINAV mission are:

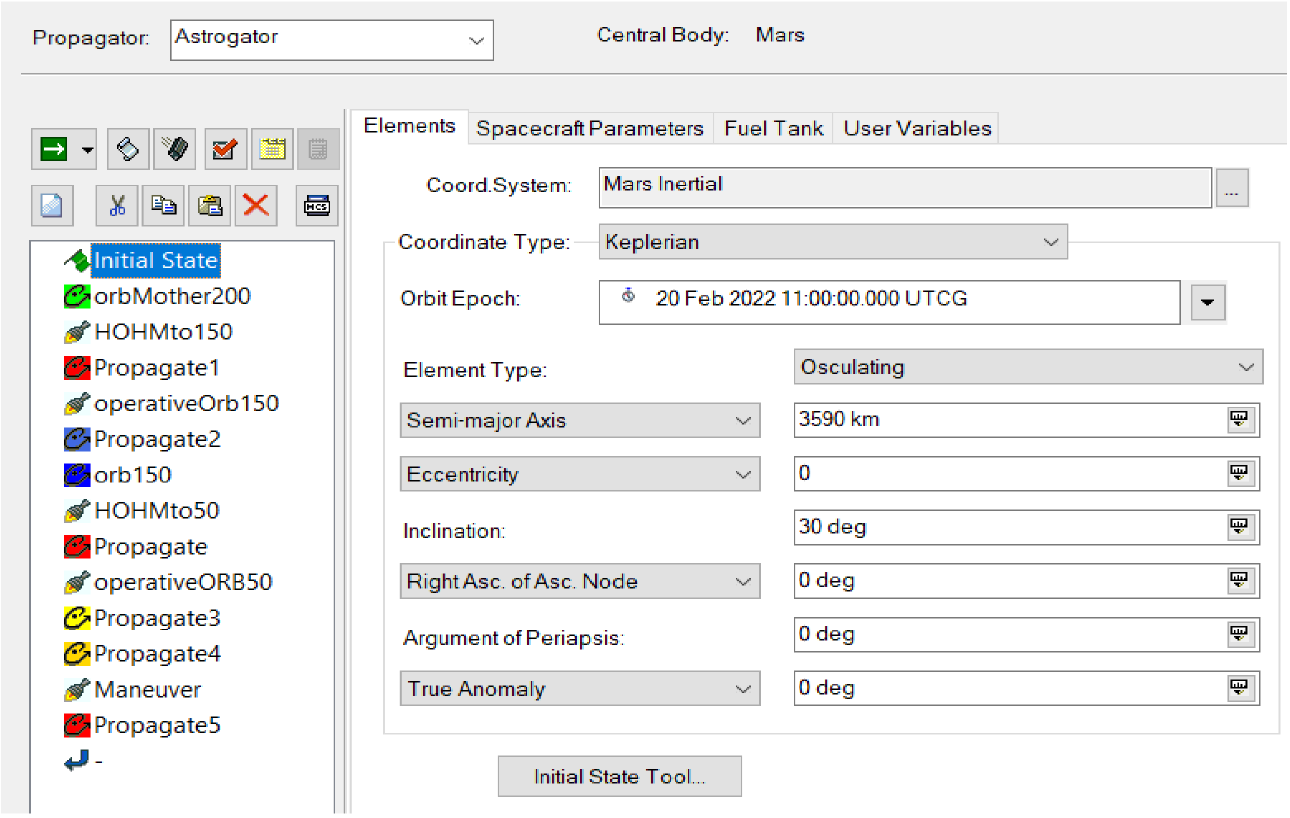


Figure 32: Mission Initial State

As it is shown in Figure 32, every manoeuvre is associated with a different colour and for every adding that was done, the user need to make sure to select Mars as a central body in an inertial System and to change the propagator from "Earth HPOP Default v10" to "Mars HPOP", otherwise the numerical methos implemented on STK shall not be able to solve the system to find the solution of the differential equations and the orbit will diverge normally from the trajectory that is supposed to follow.

If we stick to the rule previously explained of beginning with an easier element and to complicate the simulation later as the research progresses, it would be wise to start our research with only one satellite. This way of proceeding will help a lot because we shall see the trajectory which every satellite will follow during the mission; it does not really matter what is the architecture we have chosen at this point, because we need to have clear in mind the progression of the Small-Sat motion from the Deployment phase to the final Decommission phase and Disposal.

If we had immediately added all the Small-Sats from the Architecture 4 and launched the simulation, their trajectory would have overlapped one another. This would have made extremely difficult to understand at first sight the mission and it could have made more confusing to comprehend whether something was not working as we expected.

For the same reason, when I began programming the mission, I did not immediately use the actual duration for each phase, but I decided to go for a significantly smaller duration. I deliberately chose to begin using this method to ensure that I had the complete control over the STK programming language which the mission required and I could check if everything was going on as I wanted.

Considering that the period of a revolution of one satellite is much briefer than the rotation of the planet, the effect of the rotation of the planet beneath the satellite can be neglected, even in the worst circumstances (this means in the situation where the satellite is in the higher orbit and, therefore, when it has the longer period that it will ever have during the mission).

On the other hand, if I had used immediately the actual duration of the various phases of the mission, STK would have given back, as an output, all the orbits the satellite would have completed during the mission. The duration of an Hohmann Transfer would have been significantly smaller than the duration of just one phase and it would have been nearly invisible in the scheme, among many other orbits. Therefore, this is the reason why I decided to begin with a fictional mission which has each and every phase that lasts for an interval of time that is comparable with the duration of an Hohmann Transfer.

Doing so, I could be able to see exactly where and when the Hohmann Transfers began and ended, giving as an output in return a clear and immediately understandable interval between two different phases.

Only after a positive outcome, I corrected the duration by giving the real period of time of each phase, creating a more likely scenario, where it is possible to see all areas that will be covered by the mission, considering that, thanks a longer time of analysis, it is possible to see the effects of the rotation of the Red Planet under the satellites.

4 PROPULSION SYSTEM

Throughout the following chapter, an overview of the research work about the State of Art of the Small-Sat thrusters shall be given. Considering that in the previous section of this report he focus was on the orbit design, the next step which is logical to make is to look for a Propulsion System able to satisfy said thrust budget.

First of all, we need to reduce the wide research field of thrusters which are associated to artificial Satellites, taking into account the dimensions of the Space Segment that had been chosen for this mission, therefore a Small-Sat with a volume of 24 U. This leads to a first estimated mass that is about 50 kg.

After this initial evaluation, we shall present a brief overview that gathers the peculiarities and the characteristics of the thruster class chosen, considering that the propulsion system must not exceed certain dimensions and weight, otherwise the already limited space of the satellite will be mostly and exaggeratedly occupied by the thrusters and their control systems. In that unwanted situation, the volume budget which is left for the payload and the optical cameras, put there to satisfy the mission objectives, will be filled in a manner that it is not acceptable. This condition would drastically nullify the very reason of existence the mission.

Another aspect that needs to be considered in order to choose the right solution for the propulsion system is to check whether the thrust is adequate for the mission, this fact does not only mean that the thrust must be strong enough to perform the multiple manoeuvres the situation requires, but also that the propulsor must not have the possibility to provide too much high thrust than the one the mission needs, since a higher thrust capability generally means a larger, more complex thrusters. This would make the architecture more expensive than it would need to be in the ideal case.

It is true, though, that this report considers a situation that is simulated with STK, therefore it is, if course, an ideal and, until the moment the actual mission begins, a purely academical hypothesis. In the real world, an unexpected problem or an emergency can always occur.

In these circumstances, it is wise to have a reasonably higher thrust resource that can be used, compared to the one that is only strictly necessary, although it is obvious that this always represents the necessary, but not sufficient condition for the mission's feasibility.

4.1 Small-Sat Thrusters

Once we have established the proper mindset that is needed to highlight the constraints, the limits and the objectives of the research activity that is about to be explained in the following paragraphs, we shall present of recap of the mainly used thruster on board this type of Small-Sats.

Their technological peculiarities will be here reported with the purpose to see the differences and to analyse both their advantages and disadvantages of a possible implementation.

Before the report can begin, we ought to specify one thing that shall be the main guideline for the topic in exam. This is a summary of the research for the most fitting propulsion system solution; considering that this research was begun knowing only the exact valour of the various Δv needed during the mission, there was no clue of the total level of thrust the architecture required. The initial move was to consider each satellite separately, only after a positive answer, the total thrust budget was calculated from the beginning to the end.

Before we can continue, we have to specify that we are looking for a solution that satisfies two different types of propulsion. One of the major questions we have been pondering was if the same thruster set could be used to guarantee a correct response from both types or if two different solutions are required. It is obvious that the first method is simpler to implement, this means lower costs and lower complexity, on the other hand, the second is more specific in its purpose and function, this could be translated into a more precise solution.

The two types of propulsion that we are considering are:

- PRIMARY PROPULSION: this is the one that actively is responsible for the manoeuvres and orbital transfers. It is the one that modifies the instantaneous velocity of a spacecraft and it is the one which is more demanding considering his thrust levels. The primary propulsion's most efficient and effective solution is decided by the Δv of the mission which were reported in the previous chapters and were discovered through the STK orbit analysis
- SECONDARY "OR AUXILIARY" PROPULSION: if the previous type uses its thrust to be responsible for the Δv changes, which translates into a modification of one or more the orbit parameters, this second type of propulsion uses the thrust to oppose eventual disturbances or perturbations which causes unwanted and unforeseen modifications of the ideal orbit the spacecraft is supposed to follow.

It is also true that the secondary type has level of thrust that are drastically inferior to the one of the primary types and, moreover, the modulus of the Δv required for the auxiliary propulsion is significantly more contained, but there are certain aspects which are not to be underestimated.

One example of this fact is that, in this case, we have considered the hypothesis that the propulsion, both primary and secondary, is impulsive. This make it easier to calculate and dimension the level needed and it is less expensive to implement on STK, under the aspect of the computational cost. This is extremely useful to have an likely approximation of the moment in which the thrusters are active for the primary propulsion, knowing that the primary thruster will be active only in those specific temporal windows. This is what makes the primary propulsion easily predictable and simple to dimension, on the other hand the auxiliary type become active only in case of an emergency or when the spacecraft is not on the right path accepting certain error margins.

It is clear that these errors are unavoidable, the margins are simply something we should accept, because of course we do not want to keep the thrusters always turned on to correct each and every modification of the ideal orbit. Considering the satellite will have to move in a real world, it will encounter perturbances due to facts like atmospheric interference, that Mars is not a perfect sphere, Solar Wind or other types of electromagnetic phenomena, eventual meteoroids, the gravity influence of the Mars satellites; they are all events that are likely to happen as the mission progresses and, if some of them can be partially predictable with mathematical and numerical models, others cannot be, this translates into the most important difference between the two propulsion types. If the events which are the cause for the auxiliary propulsion to be switched on, are not predictable, so are the moments in which the auxiliary propulsion will be activated and considering the intensity of the phenomena is unknown, so it is the modulus of Δv required.

This makes the primary propulsion more demanding in Δv modulus every time it is activated, but the secondary more unpredictable and could be a major problem for the mission success. We do not just have to put a system that can perform the main propulsion, but also put some extra propellant in case of emergencies that could occur.

This is a problem that is needed to be faced, one of the questions that we have found most intriguing is whether we could solve both of these requirements with just one type of propulsion system or if we are going to be forced to use different solutions for the primary and secondary type. Carrying on a research activity, three classes of thruster types have been selected:

- Cold-gas thrusters
- Monopropellant thrusters
- Electromagnetic thrusters

Without further ado, in the following paragraphs the peculiarities of these different thrusters are presented.

4.1.1 Cold-gas Thrusters

This is one of the most structurally simple type of propulsion system. The system's complex dimensions are relatively small and its global weight is quite contained compared to other thrusters, this is the reason why they are mostly used on satellite with reduced dimensions, like Cube-Sats and Small-Sats. The propellant is an inert, this allows it to be non-toxic and it does not need a combustion process to generate thrust making it one of the safest propulsion systems. Usually, it can be used both a noble gas or Nitrogen N_2.

The propellant gas is stored in a pressurized tank, it can be both in liquid or in a high-pressure gas form. The thrust is generated by the propellant exiting the tank through a valve system, which regulate the outlet mechanism to have a control over the thrust modulus.

Table 21: Cold-gas Thrusters - performance

	Mass	Power	Thrust	I_{SP}
50-820 Triad	0,43 kg	9 W	52 – 105 N	-
58E163A	0,115 kg	1,3 W	10 – 120 mN	≈ 60 s
058-118	0,023 kg	1 W	10 – 120 mN	≈ 60 s
058E151	0,07 kg	1,5 W	10 – 120 mN	≈ 60 s
CGT	0,043 kg	-	28 N/bar (p = 1 – 6 bar)	>69 s
058E142A	0,016 kg	1 W	10 – 120 mN	≈ 60 s
058E146	0,04 kg	1 W	10 – 120 mN	≈ 60 s

Unfortunately, not all data are available on the SatCatalog, therefore, we have some unknown parameters for certain propulsors.

Another point that needs to be clarified is that some Cold-Gas thrusters can increment their thrust level if their mass increases too.

4.1.2 Monopropellant Thrusters

The Monopropellant Thrusters is another valid alternative for a small dimension spacecraft, like Cold-gas thrusters, these are also used for asset control and other manoeuvres that do not require a performance that is too demanding.

They are based on a relatively simple concept based on the composition of the propellant molecules, the energy that they need to generate thrust derives from a chemical reaction between a reducing agent and a catalyst.

The propellant that is mainly used for this type of propulsion is Hydrazine, N_2H_4, which acts, thanks to its properties, as a natural reducing agent. Then, it reacts with a metal oxide, typically an Aluminium Oxide, $Al_2\,O_3$. This allows to have higher thrust levels than Cold-Gas. However, unlike the previous category of propulsor, this technology involves a chemical reaction with nitrogen, therefore it could be a more impactful system on the environment and the products of this thruster could be more polluting, or worse, more poisonous.

Table 22: Monopropellant Thrusters - performance

	Mass	Power	Thrust	I_{SP}	I_{tot}
1N Monopropellant Hydrazine	0,29 kg	6 W	0,32 – 1 N	200 – 223 s	0,01 – 0,043 Ns
50N Monopropellant Hydrazine	0,4 kg	-	50 N	170 s	0,01 – 0,043 Ns
MR–104H 510N	2,4 kg	60,1 W	201 – 554,2 N	223 – 237 s	854 Ns
MR-107V 300N	1,01 kg	52 W	67 – 220 N	223 – 229 s	362,303 Ns
MRM–122 130N	0,66 – 0,76 kg	43 W	51 – 142 N	217 – 228 s	332 Ns
AJ10-200	1,95 kg	-	59,2 – 65,4 N	268 – 285 s	685 Ns
R-6F 22N	0,965 kg	-	22 N	307 s	89700 Ns

4.1.3 Electromagnetic Thrusters

The third class was considered has been classified with quite a generic name that can comprehend many different types of thrusters which works with slightly different ways. Under this voice we have gathered propulsion systems that can be classified as thermoelectric, ionic, electrostatic, electromagnetic or even Hall Effect thrusters.

It is clear that the thrust is quite low compared to the monopropellant thrusters. This must not surprise considering that said propulsors use a chemical reaction to develop the necessary force to move the satellite; in fact, usually, the chemical thrusters have higher thrust values than electromagnetic ones, but the undeniable advantage of these last type is the extremely high specific impulse I_{SP}. The higher it is, the smaller the propellant consumption is.

The thermoelectric thrusters use a resistance or, like in this case, an arcjet in order to increase the enthalpy of the propellant, without the need of a combustion.

The Hall Effect thruster are an evolution and modification of the normal and more general electromagnetic and ionic technology, moving the charged particles not only with a static electric field, but also with a magnetic field. This increases the time interval during which the particles remain in the chamber. Even in this case, it is often used an inert propellant. The Hall Effect thrusters give their best performance when Δv which has a value that is not too much elevated, are required.

Typically, the optimal operative range is 10 – 50 km/s, and the Δv required by the mission architecture 4B are perfectly contained in said interval.

Table 23: Electromagnetic Thrusters - performance

	Mass	Power	Thrust	I_{SP}	I_{tot}
BHT-100	1,16 kg	100 W	7 mN	1000 s	45,36 – 250 kNs
BHT-350	1,9 kg	300 W	17 mN	1244 s	250 kNs
BHT-600	1,1 kg	200 W	39 mN	1500 s	1,5 MNs
Halo Microelectric Thruster	0,65 kg	75 – 450 W	4 – 33 mN	1500 s	-
HT-400	0,9 kg	525 W	20 – 50 mN	1850 s	-
HT-100	0,44 kg	235 W	6 – 18 mN	1000-1600 s	-
MR-512 Low Power Arcjet	1,4 kg	1780 W	213 - 254 mN	>502 s	866,5 Ns
XR-5 Hall Thruster	12,3 kg	2 – 4,5 kW	117 – 290 mN	1676 – 2020 s	-

4.2 Trade-off Analysis

4.2.1 Performance

Now that we have an idea of the propulsion system options involved in our process of choice, we now have to decide the criteria which will allow us to proceed in the right direction, in order to identify the best solution to this problem.

Before we move forward, we need to set the parameters of a new Trade-off analysis based on the performance these thruster systems can offer and they are:

Mass “m”, Power “P”, Thrust “T”, Specific Impulse “I_{SP}”, Propellant Mass “m_P”, Complexity and Technology.

We will begin evaluating the importance of each parameter we have exemplified above, in order to understand which is the most relevant amongst them all.

Table 24: Trade off - Parameters

TRADE OFF - parameters	Mass	Power	Thrust	I_sp	m_p	Complexity	Technology		SUM	%
Mass	1	0,8	1	1	0,5	0,3	0,5		5,1	0,18214
Power	0,2	1	1	1	0,3	0,2	0		3,7	0,13214
Thrust	0	0	1	0,5	0,2	0	0		1,7	0,06071
I_sp	0	0	0,5	1	0,2	0	0		1,7	0,06071
m_p	0,5	0,7	0,8	0,8	1	0,2	0,5		4,5	0,16071
Complexity	0,7	0,8	1	1	0,8	1	0,5		5,8	0,20714
Technology	0,5	1	1	1	0,5	0,5	1		5,5	0,19643
TOT									28	1

In order to complete this and the following tables of the Trade-off analysis, we needed to organize a system of points to assign to various weight. In the diagonal of this symmetric matrix, we put “1”, and, after that we analyse every row and every column; when an element of a row is less important than the performance on the column considered, according to the criteria chosen for this analysis, we shall put a number strictly major than 0,5, which is the value that represents an equal importance. In any other circumstance we shall assign a point that is strictly minor than 0,5.

The inferior triangular part of the matrix is filled with elements that nothing are but the complement to one of the elements of the upper triangular matrix. With this system the elements that result more important are the Thrust “T” and the Specific impulse “I_{SP}”.

Among all of the parameters, it features the propellant mass “m_P”. therefore, in order to calculate said value, after we have completed the various tables, we need to make another Trade-off, one last analysis to determine the differences in the power consumptions of the various solutions. This will give us an idea of how the Small-Sat mass will change as the mission progresses, considering that we start from an initial mass of m = 50 kg.

In order to calculate this parameter, for a first approximation calculation, we can use the Tsiolkovsky Equation.

If F(i) as a generic force applied to a satellite of mass “m” which proceeds to a velocity “v”:

$$\sum_{i=1}^{N} F(i) = \frac{d(m\,v)}{dt}$$

$$= \frac{dm}{dt} v + m\frac{dv}{dt} = 0$$

$$a = \frac{dv}{dt} = \frac{T}{m} \approx \frac{dm\_P}{dt} * \frac{c}{m} \quad \rightarrow \quad dv = c * \frac{dm\_P}{m}$$

We know that: $m\,(t) = m\_in - \frac{dm\_P}{dt}\,t \quad \rightarrow \quad \frac{dm}{dt} = -\frac{dm\_P}{dt}$

Now, we integrate the equation “dv” between the initial and arbitrary final state:

$$\Delta v = \int_{v\_in}^{v\_fin} dv = \int_{m\_in}^{m\_fin} \left(-c * \frac{dm}{m}\right) = -c * \log(\frac{m\_fin}{m\_in})$$

$$m\_fin = m\_in * e^{\frac{-\Delta v}{c}} \quad \rightarrow \quad m\_P = m\_in * (1 - e^{\frac{-\Delta v}{c}})$$

Knowing that $c = g_0 * I_{SP}$ with $g_0 = 9{,}81$ m/s^2, we can calculate the Propellant Mass that each and every thruster consume when it completes the manoeuvres the mission requires.

4.2.2 Cold-gas Thrusters - Trade-off

With the same method explained in the previous chapter to calculate the Trade-off, the various parameters for the Cold-Gas thrusters to understand which is the one that it is the most suitable for our mission, had been organized in Trade-off Table, that can be found in the Annex. This will allow us to calculate an approximation of the weight of each parameter on the global mission.

This is a way to understand the importance of the chosen variables. Once that we have the weight associated to every Cold-Gas thruster, we combine it with the values that descend from the first table, which contains the relative importance of each variable.

Doing so, we come up with the following table, this procedure gives us a general idea of the thruster we might choose.

Table 25: Cold-gas Thrusters – Trade off

TRADE OFF Cold Gas Thruster	WEIGHT	50-820 Triad	58E163A	058-118	058 E151	CGT	058E142A	058 E146
Mass	0,18214	0,01268475	0,0175635	0,03415125	0,0227675	0,0276463	0,03968	0,02764625
Power	0,13214	0,009123952	0,018877143	0,028315714	0,017933286		0,028944952	0,028944952
Thrust	0,06071	0,003252321	0,010841071	0,010841071	0,010841071	0,003252321	0,010841071	0,010841071
I_sp	0,06071		0,010696524	0,010696524	0,010696524	0,007227381	0,010696524	0,010696524
m_p	0,16071		0,025254429	0,025254429	0,025254429	0,034437857	0,025254429	0,025254429
Complexity	0,20714	0,014795714	0,036989286	0,036989286	0,036989286	0,007397857	0,036989286	0,036989286
Technology	0,19643	0,017538393	0,033673714	0,033673714	0,033673714	0,010523036	0,033673714	0,033673714
TOT	1	0,057395131	0,153895667	0,179921988	0,15815581	0,090484752	0,186079976	0,174046226

In order to complete the Trade-off of the propellant mass consumption, in the following table a simple calculation was organized to find the total value of m_P for the complete duration of the mission, this will allow us to see the modifications of the global mass of the Small-Sat after each Hohmann Tranfer.

There are two tables because we have analysed both the case of mission in which the final orbit altitude is 50 km or 80 km. Of course, in the first case, considering that we need to get on a slightly lower orbit than 80 km, we have much higher consumption values corresponding to higher Δv values.

Table 26: Cold-gas Thrusters – Propellant consumption

Cold Gas Thrusters	h3=50km	50-820 Triad	58E163A	058-118	058 E151	CGT	058E142A	058 E146
Isp			60 s	60 s	60 s	69 s	60 s	60 s
1 - e^(-dv_i/c)	1)		0,0204	0,0204	0,0204	0,0177	0,0204	0,0204
	2)		0,0204	0,0204	0,0204	0,0178	0,0204	0,0204
	3)		0,0415	0,0415	0,0415	0,0362	0,0415	0,0415
	4)		0,0418	0,0418	0,0418	0,0364	0,0418	0,0418
mP_i	1)		1,02	1,02	1,02	0,885	1,02	1,02
	2)		0,999192	0,999192	0,999192	0,874247	0,999192	0,999192
	3)		1,991203532	1,991203532	1,991203532	1,746315259	1,991203532	1,991203532
	4)		1,922363467	1,922363467	1,922363467	1,692397534	1,922363467	1,922363467
TOT			**5,932758999**	**5,932758999**	**5,932758999**	**5,197959793**	**5,932758999**	**5,932758999**
m_i	0)		50	50	50	50	50	50
	1)		48,98	48,98	48,98	49,115	48,98	48,98
	2)		47,980808	47,980808	47,980808	48,240753	47,980808	47,980808
	3)		45,09896045	45,09896045	45,09896045	46,49443774	45,09896045	45,09896045
	4)		44,67239	44,67239	44,67239	44,80204021	44,67239	44,67239

Cold Gas Thrusters	h3=80km	50-820 Triad	58E163A	058-118	058 E151	CGT	058E142A	058 E146
Isp			60 s	60 s	60 s	69 s	60 s	60 s
1 - e^(-dv_i/c)	1)		0,0204	0,0204	0,0204	0,0177	0,0204	0,0204
	2)		0,0204	0,0204	0,0204	0,0178	0,0204	0,0204
	3)		0,0291	0,0291	0,0291	0,0253	0,0291	0,0291
	4)		0,0292	0,0292	0,0292	0,0255	0,0292	0,0292
mP_i	1)		1,02	1,02	1,02	0,885	1,02	1,02
	2)		0,999192	0,999192	0,999192	0,874247	0,999192	0,999192
	3)		1,396241513	1,396241513	1,396241513	1,220491051	1,396241513	1,396241513
	4)		1,360269341	1,360269341	1,360269341	1,19901668	1,360269341	1,360269341
TOT			**4,775702854**	**4,775702854**	**4,775702854**	**4,178754731**	**4,775702854**	**4,775702854**
m_i	0)		50	50	50	50	50	50
	1)		48,98	48,98	48,98	49,115	48,98	48,98
	2)		47,980808	47,980808	47,980808	48,24753	47,980808	47,980808
	3)		46,58456649	46,58456649	46,58456649	47,02026195	46,58456649	46,58456649
	4)		45,22429772	45,22429772	45,22429772	45,82124532	45,22429772	45,22429772

4.2.3 Monopropellant Thrusters - Trade-off

The same considerations previously done for Cold-Gas thrusters are also valid here, if we take into account the differences between the two propulsors types as we have described them in the previous paragraphs.

It is possible to find in the Annex the complete set of tables that report the research done for this class of thruster. Once we have finished them and obtained the weight for each and every propulsor, it is the moment to organise those values in the final table to understand the global weight of each thruster.

Table 27: Monopropellant Thrusters – Trade off

TRADE OFF Monopropellant Thruster	WEIGHT	1N Monoprop Hydrazine	50N Monoprop Hydrazine	MR-104H510N	MR-107V 300N	MRM-122 130N	AJ10-200	R-6F 22N
Mass	0,18214	0,0422825	0,036428	0,011709	0,0227675	0,0318745	0,014311	0,0227675
Power	0,13214	0,052856		0,033035	0,026428	0,019821		
Thrust	0,06071	0,0151775	0,009214911	0,003360732	0,006504643	0,00758875	0,007913982	0,010949482
I_sp	0,06071	0,009973786	0,011925179	0,009323321	0,009323321	0,009323321	0,008239214	0,002601857
m_p	0,16071	0,017792893	0,014062125	0,023245554	0,023245554	0,023245554	0,029559161	0,029559161
Complexity	0,20714	0,051785	0,031440893	0,011466679	0,022193571	0,0258925	0,027002179	0,037359179
Technology	0,19643	0,035427554	0,035427554	0,018590696	0,018590696	0,025255286	0,027359893	0,035778321
TOT	1	0,225295232	0,138498661	0,110730982	0,129053286	0,143000911	0,114385429	0,1390155

Like in the previous section we present here the table where the values of propellant mass consumed after each manoeuvre is reported for the two different cases where the final altitude is 50 or 80 km.

Again, here, the reader can see how the global mass of the Small-Sat is reduced after each orbital change.

Table 28: Monopropellant Thrusters – Propellant consumption

Monopropellant Thrusters	h3=50km	1N Monoprop	50N Monoprop	MR-104H 510N	MR-107V 300N	MR-122 130N	AJ10-200	R-6F 22N
Isp		211,5 s	170 s	230 s	226 s	222,5 s	276,5 s	307 s
1 - e^(-dv_i/c)	1)	0,0058	0,0072	0,0053	0,0054	0,0055	0,0045	0,004
	2)	0,0058	0,0073	0,0054	0,0055	0,0056	0,0045	0,004
	3)	0,012	0,0148	0,011	0,0112	0,0114	0,0092	0,0082
	4)	0,012	0,0148	0,0111	0,0113	0,0114	0,0092	0,0083
mP_i	1)	0,29	0,36	0,265	0,27	0,275	0,225	0,2
	2)	0,288318	0,362372	0,268569	0,273515	0,27846	0,2239875	0,1992
	3)	0,593060184	0,729308894	0,544130741	0,553912632	0,563690556	0,455869315	0,40672656
	4)	0,585943462	0,718515123	0,543037533	0,552599068	0,557264484	0,451675317	0,40831081
TOT		**1,757321646**	**2,170196017**	**1,620737274**	**1,6500267**	**1,67441504**	**1,356532132**	**1,21423737**
m_i	0)	50	50	50	50	50	50	50
	1)	49,71	49,64	49,735	49,73	49,725	49,775	49,8
	2)	49,421682	49,277628	49,466431	49,456485	49,44654	49,5510125	49,6008
	3)	48,82862182	48,54831911	48,92230026	48,90257237	48,88284944	49,09514319	49,19407344
	4)	48,24267835	47,82980398	48,37926273	48,3499733	48,32558496	48,64346787	48,78576263

Monopropellant Thrusters	h3=80km	1N Monoprop	50N Monoprop	MR-104H 510N	MR-107V 300N	MR-122 130N	AJ10-200	R-6F 22N
Isp		211,5 s	170 s	230 s	226 s	222,5 s	276,5 s	307 s
1 - e^(-dv_i/c)	1)	0,0058	0,0072	0,0053	0,0054	0,0055	0,0045	0,004
	2)	0,0058	0,0073	0,0054	0,0055	0,0056	0,0045	0,004
	3)	0,0083	0,0104	0,0077	0,0078	0,0079	0,0064	0,0058
	4)	0,0084	0,0104	0,0077	0,0078	0,008	0,0064	0,0058
mP_i	1)	0,29	0,36	0,265	0,27	0,275	0,225	0,2
	2)	0,288318	0,362372	0,268569	0,273515	0,27846	0,2239875	0,1992
	3)	0,410199961	0,512487331	0,380891519	0,385760583	0,390627666	0,31712648	0,28768464
	4)	0,411696449	0,507157463	0,377958654	0,38275165	0,392447299	0,315096871	0,286016069
TOT		**1,40021441**	**1,742016794**	**1,292419173**	**1,312027233**	**1,336534965**	**1,081210851**	**0,972900709**
m_i	0)	50	50	50	50	50	50	50
	1)	49,71	49,64	49,735	49,73	49,725	49,775	49,8
	2)	49,421682	49,277628	49,466431	49,456485	49,44654	49,5510125	49,6008
	3)	49,01148204	48,76514067	49,08553948	49,07072442	49,05591233	49,23388602	49,31311536
	4)	48,59978559	48,25798321	48,70758083	48,68797277	48,66346504	48,91878915	49,02709929

4.2.4 Electromagnetic Thrusters - Trade-off

The procedure described in the previous section remains unchanged for this case as well

Like before, we have finished the various Trade-off tables for the parameters of these propulsion system, it is the moment to organise those values in the final table to understand the global weight of each thruster. It is possible to find the Trade-off Tables in the Annex at the end of this report.

Table 29: Electromagnetic Thrusters – Trade off

TRADE OFF Electromagnetic Thruster	WEIGHT	BHT - 100	BHT - 350	BHT - 600	Halo Micro Electric Thruster	HT - 400	HT - 100	MR - 512 Low Power Arcjet	XR - 5 Hall Thruster
Mass	0,18214	0,024791278	0,018466972	0,023526417	0,030103694	0,025297222	0,034404222	0,02049075	0,005059444
Power	0,13214	0,02477625	0,017985722	0,021105694	0,021105694	0,017251611	0,018903361	0,005505833	0,005505833
Thrust	0,06071	0,011383125	0,010118333	0,00758875	0,010118333	0,00758875	0,008853542	0,002529583	0,002529583
I_sp	0,06071	0,008516264	0,006745556	0,005143486	0,005143486	0,003288458	0,006576917	0,012647917	0,012647917
m_p	0,16071	0,019195917	0,022544042	0,026561792	0,026561792	0,029909917	0,022544042	0,00669625	0,00669625
Complexity	0,20714	0,035961806	0,035961806	0,035961806	0,025317111	0,026180194	0,030495611	0,008630833	0,008630833
Technology	0,19643	0,033556792	0,033556792	0,033556792	0,023735292	0,027827583	0,027827583	0,008184583	0,008184583
TOT	1	0,158181431	0,145379222	0,153444736	0,142085403	0,137343736	0,149605278	0,06468575	0,049254444

Like in the previous sections, we present here the table where the values of propellant mass consumed after each manoeuvre are reported for the two different cases where the final altitude is 50 or 80 km.

Again, here, the reader can see how the global mass of the Small-Sat is reduced after each orbital change.

Table 30: Electromagnetic Thrusters – Propellant consumption

Electric Thruster	h3=50km	BHT-100	BHT-350	BHT-600	Halo Micro Electric Thruster	HT-400	HT-100	MR-512 Low Power Arcjet	XR-5 Hall Thruster
Isp		1000 s	1244 s	1500 s	1500 s	1850 s	1300 s	298,5 s	298,5 s
1 - e^(-dv_i/c)	1)	0,0012	0,00099123	0,00082213	0,00082213	0,00066664	0,00094855	0,0041	0,0041
	2)	0,0012	0,00099471	0,00082502	0,00082502	0,00066899	0,00095188	0,0041	0,0041
	3)	0,0025	0,002	0,0017	0,0017	0,0014	0,002	0,0085	0,0085
	4)	0,0026	0,0021	0,0017	0,0017	0,0014	0,002	0,0085	0,0085
mP_i	1)	0,06	0,0495615	0,0411065	0,0411065	0,033332	0,0474275	0,205	0,205
	2)	0,059928	0,049686201	0,041217086	0,041217086	0,033427201	0,047548855	0,2041595	0,2041595
	3)	0,12470018	0,099801505	0,08486005	0,08486005	0,069906537	0,099810047	0,421522144	0,421522144
	4)	0,129363967	0,104581997	0,084715788	0,084715788	0,069808668	0,099610427	0,417939206	0,417939206
TOT		**0,373992147**	**0,303631202**	**0,251899424**	**0,251899424**	**0,206474406**	**0,294396829**	**1,24862085**	**1,24862085**
m_i	0)	50	50	50	50	50	50	50	50
	1)	49,94	49,9504385	49,9588935	49,9588935	49,966668	49,9525725	49,795	49,795
	2)	49,880072	49,9007523	49,91767641	49,91767641	49,9332408	49,90502365	49,5908405	49,5908405
	3)	49,75537182	49,80095079	49,83281636	49,83281636	49,86333426	49,8052136	49,16931836	49,16931836
	4)	49,62600785	49,6963688	49,74810058	49,74810058	49,79352559	49,70560317	48,75137915	48,75137915

Electric Thruster	h3=80km	BHT-100	BHT-350	BHT-600	Halo Micro Electric Thruster	HT-400	HT-100	MR-512 Low Power Arcjet	XR-5 Hall Thruster
Isp		1000 s	1244 s	1500 s	1500 s	1850 s	1300 s	298,5 s	298,5 s
1 - e^(-dv_i/c)	1)	0,0012	0,00099123	0,00082213	0,00082213	0,00066664	0,00094855	0,0041	0,0041
	2)	0,0012	0,00099471	0,00082502	0,00082502	0,00066899	0,00095188	0,0041	0,0041
	3)	0,0018	0,0014	0,0012	0,0012	0,00095654	0,0014	0,0059	0,0059
	4)	0,0018	0,0014	0,0012	0,0012	0,00096132	0,0014	0,006	0,006
mP_i	1)	0,06	0,0495615	0,0411065	0,0411065	0,033332	0,0474275	0,205	0,205
	2)	0,059928	0,049686201	0,041217086	0,041217086	0,033427201	0,047548855	0,2041595	0,2041595
	3)	0,08978413	0,069861053	0,059901212	0,059901212	0,047763142	0,069867033	0,292585959	0,292585959
	4)	0,089622518	0,069763248	0,05982933	0,05982933	0,047955907	0,069769219	0,295789527	0,295789527
TOT		**0,299334648**	**0,238872002**	**0,202054128**	**0,202054128**	**0,162478251**	**0,234612607**	**0,997534986**	**0,997534986**
m_i	0)	50	50	50	50	50	50	50	50
	1)	49,94	49,9504385	49,9588935	49,9588935	49,966668	49,9525725	49,795	49,795
	2)	49,880072	49,9007523	49,91767641	49,91767641	49,9332408	49,90502365	49,5908405	49,5908405
	3)	49,79028787	49,83089125	49,8577752	49,8577752	49,88547766	49,83515661	49,29825454	49,29825454
	4)	49,70066535	49,761128	49,79794587	49,79794587	49,83752175	49,76538739	49,00246501	49,00246501

4.3 Chosen Solution

Considering the needs and constraints of the mission, at the end of the Trade-off analysis, we ought to find a compromise among the propulsor type which we think it is the most suitable, the availability of the data about it and the possibility to implement it in a successful way on STK to check the feasibility of the solution that we have come up with. Actually, these choice's criteria are the most effective path as long as the mission remains in the hypothetical and academic world.

At the moment when we bring this mission scenario into the real world, the final choice should also be influenced by more practical aspects like cost, safety, environmental impact and more.

Now that we have exemplified the approach to this problem, we can make the reader aware of the decision to eliminate the possibility to use the Monopropellant thrusters. It is undeniable that they are the one with the highest thrust level amongst the propulsion type we have taken into account, however, we do not require such level. If we go for that class of thrusters, it is likely that we would have exaggerated the power of the propulsion system and therefore wasting volume, cost and energy for something that is not strictly necessary.

Considering the Δv required by the expected manoeuvres and translating that on a thrust level we must apply to our satellite, electromagnetic thrusters are powerful enough to satisfy the goal we want to achieve, all of this without even stating that their extremely high I_{SP} allows them to have the lowest propellant mass consumption among all the thrusters which we have analysed.

We cannot avoid putting on board the Cold-Gas thrusters, because they are strictly necessary for the Reaction Wheels desaturation, alternatively, thanks to their small size and relatively light weight, they could even be used as another possible solution to control the attitude of the satellite, if do not choose to go with the Reaction Wheels. Of course, this would cause the mission to put on board greater tanks, due to the more massive amount of fuel, because that would be used not only for the secondary propulsion, but to control the rotational degree of freedom of the spacecraft.

Here, we shall now report the thrusters we have chosen for the other two classes along with their data and their parameters of interest:

- Cold-Gas thruster: 058E142A

Table 31: chosen Cold-gas Thrusters

	Mass	Power	Thrust	I_{SP}
058E142A	0,016 kg	1 W	10 – 120 mN	≈ 60 s

- Electromagnetic Thruster (Hall Effect thruster): BHT-600

Table 32: chosen Electromagnetic Thrusters

	Mass	Power	Thrust	I_{SP}	I_{tot}
BHT-600	1,1 kg	200 W	39 mN	1500 s	1,5 MNs

Once we have established our choices and we have highlighted their properties, we can proceed we the next section where we analyse whether we can use just one type of thrusters or if we have to implement a more complicated system. Moreover, if both the possibilities are confirmed as feasible, an evaluation of advantages and disadvantages has to be carried out to come up with the most performing system as possible.

4.3.1 Solution A: only Cold-gas Thrusters

The easiest and most immediate way to organise the propulsion system is to choose just one class of thrusters, in this case the Cold-Gas thrusters, to satisfy both the primary and the secondary propulsion requirements.

The following chapter will report the implementation of solution on STK to see if Cold-Gas thrusters are enough to guarantee the orbital control of the satellite. It is necessary to remind that we need to have the ability to completely handle the 6 Degrees of Freedom of the Small-Sats. Among these potential motions, 3 DoF are the linear accelerations along the direction of x,y,z in both positive and negative orientation and the other 3 DoF are the potential rotational movements about the same three axis.

We assume the axis “x” is in the direction of the motion, “z” is hypothesized to be Nadir-pointing, that means towards Mars’ surface, the last axis “y” is oriented following the Right-hand Law.

In order to guarantee the control over said movements, we need to implement on the satellite the various nozzles of the thrusters in certain strategical points. The total number of thrusters can be either 6, 8 or 12. Most Small-Sats implement 8 thrusters and they are localized in the 8 vertexes of the satellite with variable angles of inclination and orientation. For our case, we suggest a configuration of 12 nozzles, 2 for every face of the spacecraft.

Considering that the Cold-Gas thrusters are the one that are the simplest under a technological point of view, it is logical to assume that this solution is the most economically convenient.

On the other hand, a disadvantage that cannot be underestimated is that this is also the type of propulsor that has the highest values of propellant mass consumption, since the thrust does not come from a combustion, a chemical reaction or an electromagnetic phenomenon, the propellant mass is not very much energized. It is a technology that requires more mass to generate a force to move a much bigger mass, like the one of the satellites, compared to the one expelled from the nozzle. This leads to the fact that we are forced to put on board more mass, this means bigger tanks which need a more performing system to regulate the temperature and the pressure to keep the cryogenic propellant in the state of matter we consider to be more suitable.

According to our calculations, in the worst circumstances we need nearly 5,9 kg of propellant just for the primary propulsion, we would need to put that weight on a satellite that is 50 kg heavy. Without talking about the weigh of the tank which will have to be optimized depending on the form chosen and it will change according to the materials and the propellant chosen, we can assume the general weight of the other components of the Cold-Gas system.

In the case we deemed it to be unacceptably heavy, we could already state that it is impossible for this mission to work only thanks to the Cold-gas technology.

Here we report a scheme of a complete system that makes a multi-nozzle Cold-Gas thruster work:

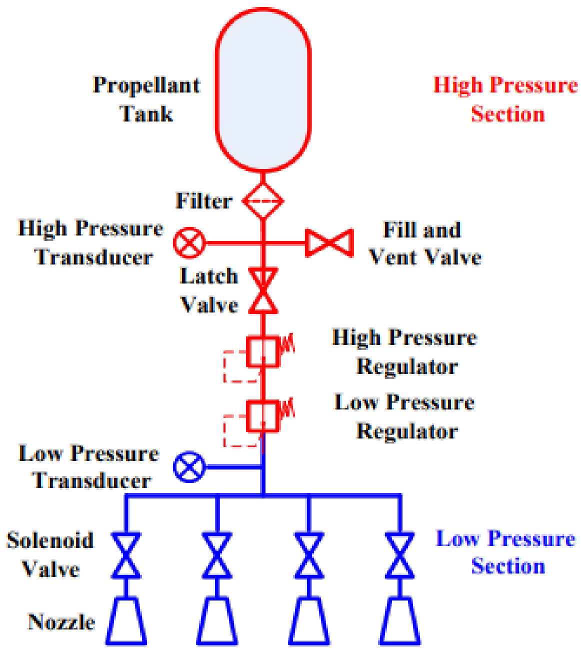


Figure 33: Cold-gas Thrusters – scheme

As we previously said, in this initial analysis we are not considering the tank weight because there is no certain way to estimate it in a precise way.

Table 33: Cold-gas Thrusters – mass analysis

COMPONENT	Mass [g]
Latch valve	≈ 50
Filter	≈ 38 – 40
Pressure regulator: • High pressure • Low pressure	 ≈ 110 ≈ 110
Nozzle + solenoid valve	≈ 16 – 20
Transducer • High pressure • Low pressure	 ≈ 28 - 30 ≈ 28 - 30
"Fill and vent" valve	≈ 48 – 50
Tubes, plates, brackets and fasteners	≈ 590 – 600
TOT	≈ 1020 – 1050

This System was designed to have 4 nozzles, this means that we need to have a system that is three times bigger than the result reported at the end of the table if we are planning to put 12 thrusters. This will bring us to have a propulsion system that will automatically have a mass greater than 10 kg.

4.3.2 Solution B: Cold-gas + Electromagnetic Thrusters

In this paragraph we will ponder the possibility of other considered architecture for the propulsion system, this new configuration combines the Cold-Gas thruster system with one or, more likely, two Hall Effect Thrusters.

Having a double propulsor allows us to control the motion along the x axis both in the positive and negative way. Thus, this permits to speed up and to slow down the Satellite in the direction of the motion.

Having a Hall-Effect thruster makes the system that deal with the primary propulsion much more efficient and less heavy. The tank and the other components of the system are not as wide the one from a Cold-Gas thruster; it is true that the thrust is on the size level of mN and the technology still has to be developed, but the application of this thruster on both great and small dimension spacecraft, proves their incredible versatility. We must also add that its density energy is an impressive advantage that easily surpasses the drawback of this technology.

However, in order to give the Small-Sat the proper furniture of thrust that the mission requires, we must consider that the Power Unit must provide the energy that ionize the propellant atoms. Typically, the propellant is a noble gas, in order to have a higher efficiency and greater energy density provided by the thruster per unit of mass we need a noble gas which has a higher Atomic Weight and a lower Ionization Potential. The ideal choice would be to use Xenon, but the drawback of this choice is that the Xenon is quite rare and expensive on Earth, making it more expensive than other elements which belongs to the Noble gasses too, like Krypton.

In fact, Krypton's lower cost and the ease it can be provided with, is one of the reasons why it is used more frequently for architecture where multiple small satellites are used to create constellations. Of course, if we go for Krypton, instead of Xenon, we have to accept that the global efficiency of thruster drops.

5 FINAL OPTIMIZATION

In this chapter we are going to present to the reader the work that has been carried on to implement the chosen solution for the propulsion System on the STK software. This will allow us to understand the feasibility of the solution we have come up with.

In the previous chapter, the analysis has been to decide the most suitable type of thruster for our Mission Architecture.

So, we shall begin to explain the upcoming topic reminding that we have used STK in order to simulate the mission scenario once we have calculated the Δv for the expected manoeuvres. Considering that, for a first approximation research activity, we did not change the default thrusters which the program has implemented, to begin our simulation, those values need to be changed and it is not enough to create an alternative set of data for our propulsor, but also there is the necessity to program the disposition of the nozzles in the structure of the Small-Sat.

That needs to be done because the vector of the thrust that acts on the satellite is not defined just by its modulus, but also by the direction of application of said force. If we apply two force vectors with the same intensity to the same body, but with different points of application and direction, the satellite's dynamics which comes out of this action is drastically different.

In the last section of this report, the configuration of the chosen thrusters was reported and it was made clear that we are planning to have different clusters of thrusters according to the type of propulsion system we are planning to put on board.

We have designed the solution that we deem to be most effective for both the Cold-Gas Thrusters and Hall Effect Thrusters, this will tell the one who will implement these propulsors the correct orientation. This further information will represent an addition to the performances' parameters found on SatCatalog about these pieces of technology, helping to formulate a realistic hypothesis of the behaviour of said technology as the actual mission progresses.

Now we report the procedural steps which have been completed to come up the solution of this problem.

The starting point of the following research is, of course, opening STK and go to check a generic satellite of the constellation, opening astrogator, it is possible to see in the Hohmann Manoeuvres blocks what needs to be changed.

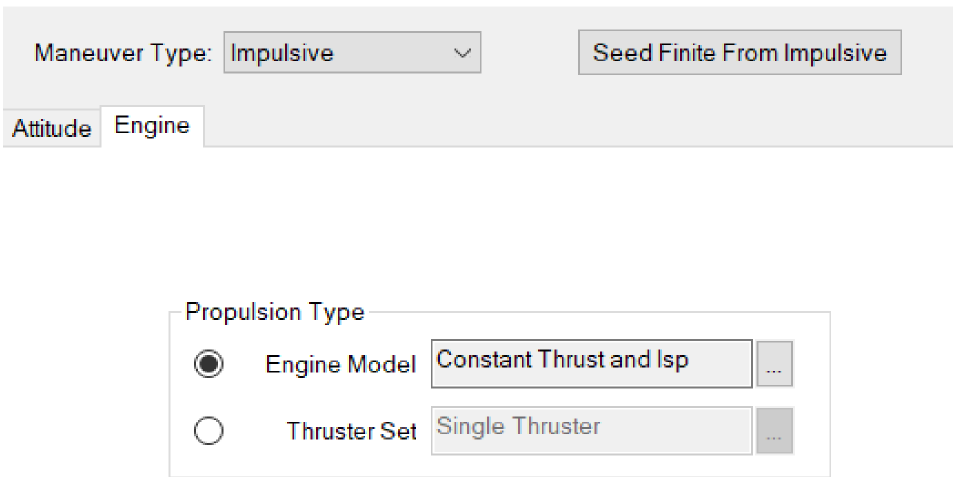


Figure 34: Default Propulsion System

We can see here the main aspects that can be controlled. This simulation began with the hypothesis of impulsive manoeuvres and that is still valid, therefore, it is not going to be changed. The other possibilities are what we must focus on, it is clear here what the default settings of the "Engine model" and the "Thruster Set".

In order to change those aspects, we need to open the "Component Browser" and apply the planned modifications under those voices.

5.1 Engine Models

The "Engine Models" is what controls the values of the performance parameters and enable the user to decide whether said values must be constant or not, and, in this case, how they vary with other control parameters.

In the following picture the various default possibilities between which we can choose, are reported:

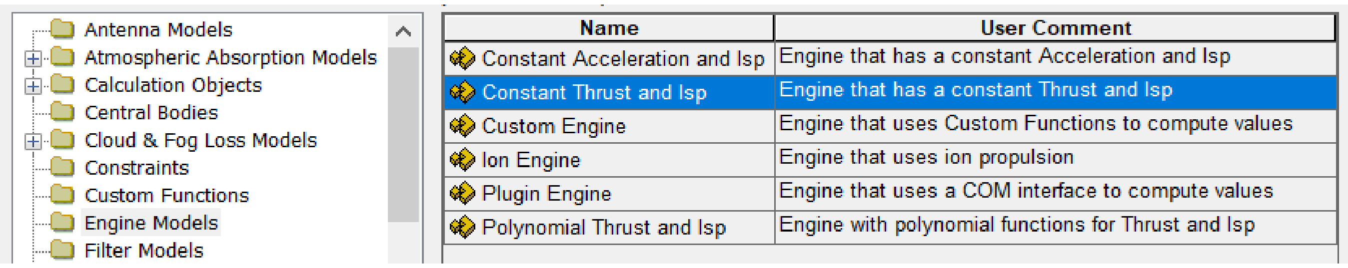


Figure 35: STK – Engine Models

STK is automatically set on "Constant Thrust and I_{SP}". At least for this actual stage of this project, we shall choose the same "Engine Model", both Thrust and I_{SP} will still be considered constant. In other circumstances, in order to have a more precise model, we suggest a session of experiments on the real engine to determine if the Thrust and parameters change their intensity when occur a change in certain conditions of the engine, or the environment.

Once we have found the voice "Constant Thrust and I_{SP}", we duplicate the file to enable the modification of the values of T and I_{SP}.

In the following image we report the unmodified initial values of the default model.

Name	Value	Description
ComponentName	Constant_Thrust_and_Isp	Name of the component
g	0.009806650000000000 km/sec^2	Earth surface gravity accel. for Isp conversions
Thrust	500.000000000000000 N	Thrust for this engine
Isp	300.000000000000000 s	Specific Impulse for this engine

Figure 36: Default Constant T and I_{SP}

We need to copy this file twice, because we need to insert the data of the Cold-gas thrusters and the Hall Effect thrusters. We shall rename them:

- T and I_{SP} – coldgas SINAV
- T and I_{SP} – EMP SINAV

Now that these files can be modified, they are going to be opened and values modified according to propulsor chosen in the previous chapter.

For the first model, which the one for the Cold-Gas Thruster, "T and I_{SP} – coldgas SINAV", the new values that are inserted, are reported in the following table:

Name	Value	Description
ComponentName	T_and_Isp_-_coldgas_SINAV	Name of the component
g	0.009806650000000000 km/sec^2	Earth surface gravity accel. for Isp conversions
Thrust	0.120000000000000 N	Thrust for this engine
Isp	60.000000000000000 s	Specific Impulse for this engine

Figure 37: Cold-gas thruster Constant T and I_{SP}

On the other hand, the model of the Hall-Effect thruster which has been chosen, T and I_{SP} – EMP SINAV, was implemented with the following value for its parameters:

Name	Value	Description
ComponentName	T_and_Isp_-_EMP_SINAV	Name of the component
g	0.009806650000000000 km/sec^2	Earth surface gravity accel. for Isp conversions
Thrust	0.039000000000000 N	Thrust for this engine
Isp	1500.000000000000000 s	Specific Impulse for this engine

Figure 38: Hall-Effect thruster Constant T and I_{SP}

An advertisement that needs to be made is that not all range of values are acceptable by STK to be used. When a new value is inserted for thrust and I_{SP}, STK also presents the user a lower and an upper limit that cannot be exceeded. Therefore, the value for the thrust parameter which can be inserted are comprised in a range between 0 and about $1.79*10^{308}$ N and the analogue for the specific impulse must be between 0 and about $1.79*10^{308}$ s.

Now that we have put the right values of thrust and specific impulse on the models we have created, the next step of this research is to organize these thrusters in a cluster with the desired orientation, so we shall go to the voice of the “Component Browser” that is deals with this aspect, which is “Thruster Sets”.

5.2 Thruster Sets

The procedure is the same which was explained in the previous chapter. This is the part of STK that deals with the orientation and the direction control of the thrusters and it shows how it influences the attitude control of the satellite.

The default version of thruster cluster or, how it is called “Single Thruster”, is organized as shown in the following picture:

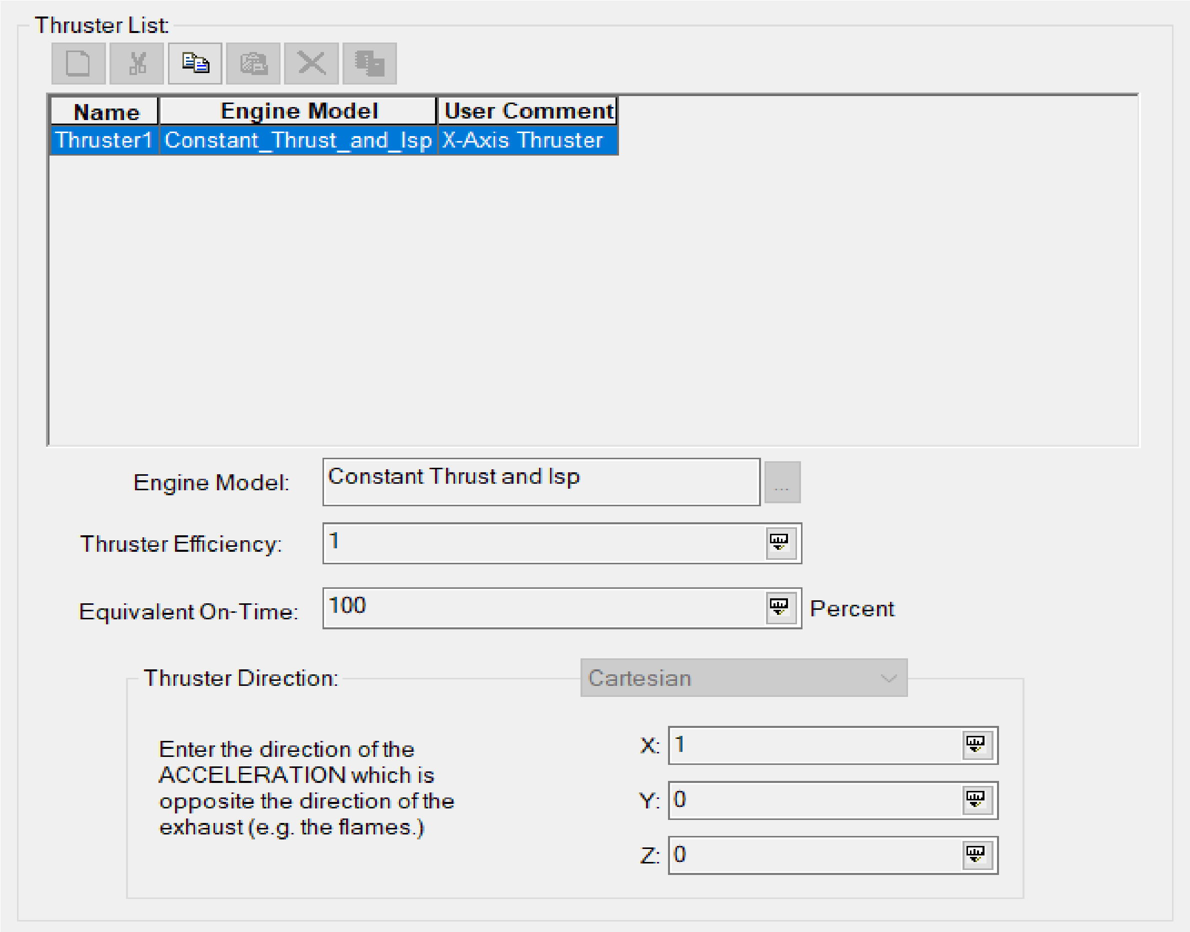


Figure 39: Default Thruster Set

It is clear that this "Single Thruster" represent an approximation of the real situation. This fact is understandable checking the thruster efficiency that is considered to be equal to 1, this is clearly beyond the most optimistic previsions since it is implicitly suggesting the thruster that is implemented by default is assumed as perfect, translating this fact into a complete absence of any kind of loss in the engine.

It is also possible to see how the "Engine Model" is the basic version we have begun our simulation with and we have modified as our work has progressed, this means the "Constant Thrust and Specific Impulse". In the lowest part of the panel, we see the commands to control the coordinates of the direction of the thrust vector, in the default version we have a single thruster pointing along the positive way of the x-axis direction which is the one we assume to be the direction of the average motion of the satellite.

In order to modify the model as we wish, it is possible to duplicate the default model and to delete or add now thrusters on the satellite, modify their orientation and Engine Model as we prefer, selecting the one we have created, as we explained in the previous section of this report. Now, we shall give the reader a report of the modification which we have made for the two types of thrusters that we have considered, treating them in separate ways to better dig into their peculiarities, their advantages and their downsides.

We present here the picture of the "Thruster Sets":

Name	User Comment	Description
Single Thruster	A collection of engine thrusters	A collection of engine thrusters
Thruster Set	A collection of engine thrusters	A collection of engine thrusters
Thruster Set - coldgas SINAV	A collection of engine thrusters	A collection of engine thrusters
Thruster Set - EMP SINAV	A collection of engine thrusters	A collection of engine thrusters

Figure 40: Modified Thrusters Sets

5.2.1 Cold-Gas Thruster Set

First of all, we need to make a choice about how many thrusters that we need to put on board the satellite. Considering that we need to ensure the control over 6 Degrees of Freedom (D.o.F.), the situation requires not less than 6 thrusters. The two most common architectures for this type of project present a cluster of either 8 or even 12 thrusters.

From this choice depends the disposition of the thrusters on the Small-Sats: if we choose 8 thrusters, they will be put on the eight vertexes of the satellite, typically their neutral inclination was determined by an angle of Azimuth of 45° and an Elevation Angle of 45°, so it will have an inclination of an angle of 135° with each face of the satellite. Another advantage of this architecture is that, if we organize the propulsor in such a way, they are more easily orientable and their position can be changed.

On the other hand, if we go with the configuration of 12 thruster, we will need to put two nozzles for each face. In the most common configuration of this type, the vector of the thrust force work in a direction that is perpendicular to the face where the thruster is located. This is a much more rigid architecture, it is more difficult to control and modify the direction of the thrust, but it is obvious that the higher number of thrusters reduces the disadvantages caused by this limitation and guarantee a more complete control over the various Degrees of Freedom. Of course, another disadvantage is that a higher number of thrusters make the propulsion system heavier. Before we can make the choice, we need to make sure of what downsides we are willing to accept and what action just represents just a pointless complication.

For the architecture in exam, it was chosen to go with the configuration of 12 thrusters because it results more complete and has another strong advantage: the greater number of nozzles is also extremely positive for the redundancy of the technology, if a problem occurs to one thruster, there is always another one for the same face of the Small-Sat. This means that this architecture or the propulsion has no blind spots that are not covered by an actuator. On the other hand, in the configuration with 8 thrusters, if the same problem occurs and there is a failure to one of the nozzles for any reason, that vertex of the satellite is completely uncontrollable; it would get extremely difficult to guarantee the complete control and the other thrusters will struggle to do that. A possible solution for this problem would be to put two thrusters for every vertex, but this is not something that can be suggested lightly because it would increase the global weight of the Propulsion System, even more than the 12 thrusters' configuration. That would also take away volume to the payload that it is planned to be put on board.

Considering all of aspects previously explained, eventually we chose to program on STK the 12 thrusters' configuration. We shall present now an image of how we have organized them on STK:

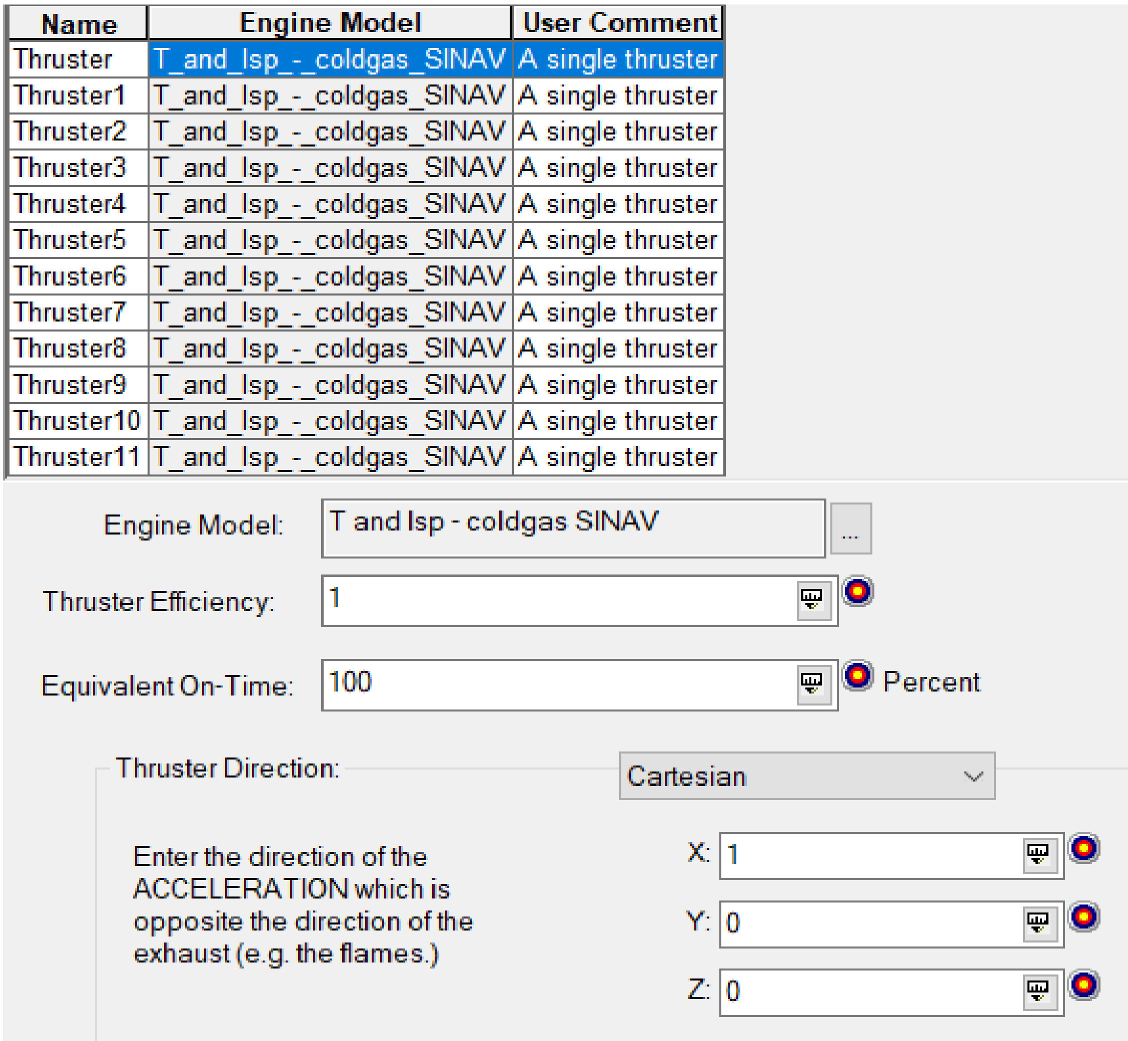


Figure 41: Cold-gas Thruster Thrusters Set

Modifying the X, Y and Z coordinates, we can program the direction of all the thrust vectors of the satellites. Two thrusters will be pointed at the positive way of one of the three directions, the other two will go to the negative way, this allows us to have two thrusters for every possible way, covering all possible movements.

Moreover, the Engine Model was also changed to use the one that corresponds to the Thruster Set of the Cold-Gas Thrusters.

5.2.2 Hall Effect Thruster Set

In this section, we will analyse the similar situation for the Hall-Effect thruster.

We have already talked about the fact that is better to put two thrusters instead of only one, one for the positive x-axis and the other one for the negative x-axis in order to guarantee the possibility to control the velocity of the spacecraft.

We shall present now an image of how we have organized them on STK:

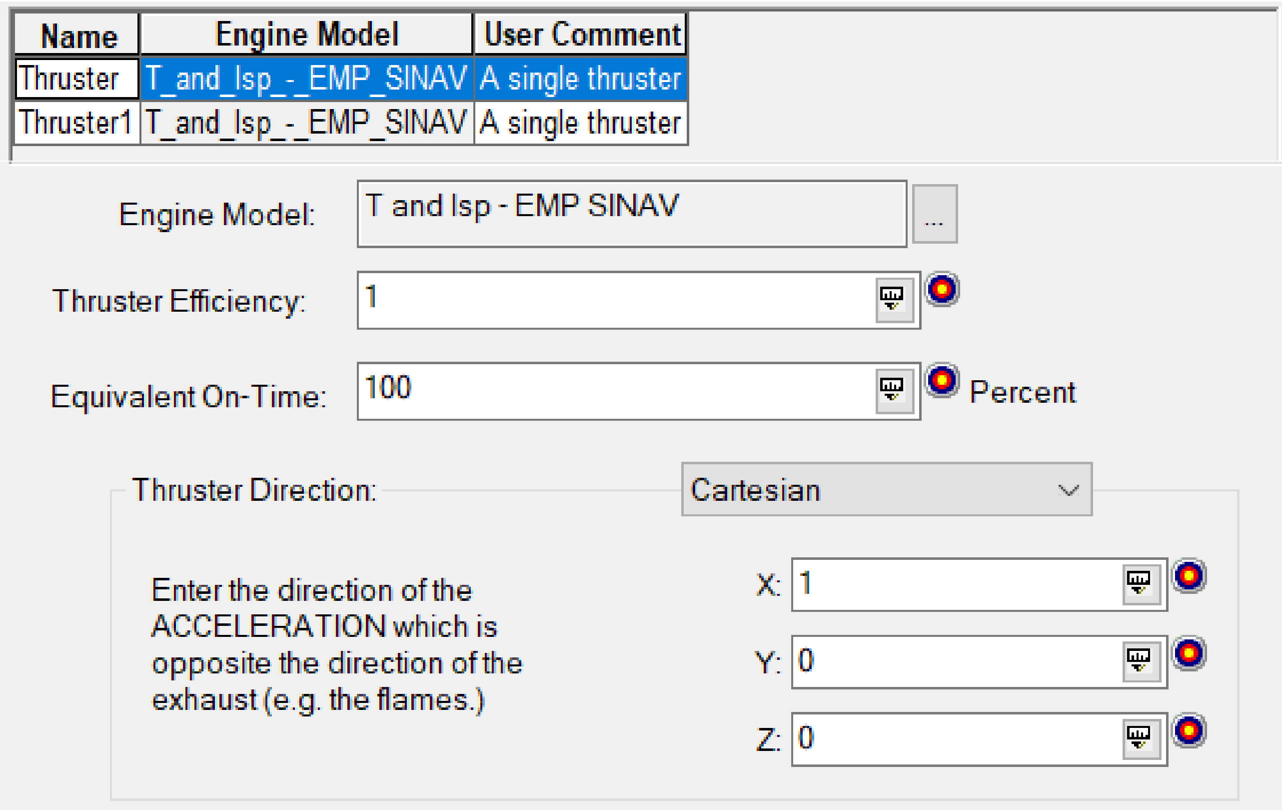


Figure 42: Hall-Effect Thrusters Set

Again, the Engine Model was adapted to the Thruster Set in exam, in this case it is, of course the one of the electromagnetic thrusters.

The reader might have noticed that in both cases the Thrust Efficiency was not changed and its value is kept equal to 1. This is due to the fact that, for this project, it was not carried out an experimental session on the real thruster in a predisposed testing facility. The lack of data about the actual behaviour of the thrusters in a real environment makes impossible to formulate hypothesis of this level for a first approximation analysis.

6 CONCLUSIONS

In this final part of the report, a quick overview and reminder of the research work that was done is presented. We started this analysis with the intention to prove the feasibility of the utilization of the Small-Sat technology in a Deep Space Environment.

In the first part of this paper, we have highlighted the parameters and the points of interest of this research, doing so, it has been possible to create a solid base to build our analysis on.

In the Chapter 3, through an STK analysis we have created a simulation for each and every possible mission scenario to determine which is the one that satisfies in the most efficient way the constraints and the objectives decided in the first chapter. After the choice of the Architecture 4B is taken, it was implemented on STK. This served as a base to study the global coverage of the area of interest and the communication link with the rover on Mars' surface and the mothercraft. Thanks to this procedure, STK has been able to print reports about these topics.

After that, it is presented a chapter that gives the reader an overview of the research that was done to choose the propulsion system to put on board of our satellite. The results that have been found tell us that the mission is feasible with both the solution which involves only the Cold-gas thrusters or the one that is a combined solution between the Cold-gas and the Hall-Effect thrusters. The final choice will only depend on the mass quantity we shall deem acceptable to put on board of the satellites. Through the software STK, we have verified the feasibility of each propulsor type and thruster set, with different thrust levels and positions.

In conclusion, in order to have a deeper understanding of the mission peculiarities, as the study will proceed and get more and more precise, it would be wise to conduct a session of experiments of the propulsors system to study in a more practical way the thruster performances and to see how they change with the environment parameters.

Another possible upgrade that could be made would be to modify the propagator that was used turning it into a more advanced and complex mechanisms to study various effects that has not been considered in this paper. This would allow us to expand ever further our comprehension of the mission in all of its aspects, in order to guarantee the safety that is necessary for a positive outcome from this mission.

7 ANNEX

7.1 Payload Annex

Annex-Table 1: COTS solutions

COTS	Producer	Mass [kg]	W x L x H [mm]	Power cons. [W]	GSD @ 500 km [m]	Swath @ 500 km [km]
iSIM-90 VNIR SWIR	Satlantis	<4	308 x 114 x 100	25,3	1,65	13
	Satlantis	<6	308 x 216 x 115	30,5	1,65	13-16
iSIM-170 VNIR SWIR	Satlantis	<8	593 x 276 x 308	25,3	0,8	13
	Satlantis	<15	593 x 471 x 308	30,5	0,8	13-16
Caiman Imager	Dragonfly Aerospace	1,8	100 x 100 x 245	<10	3 [PAN] 6 [MS]	12
Chameleon Imager	Dragonfly Aerospace	1,6	100 x 100 x 215	<10	3 [PAN] 6 [MS]	40
					Binned 20 or 40	20
Gecko Imager	Dragonfly Aerospace	0,4	100 x 100 x 650	2,6	39	80
DragonEye Imager	Dragonfly Aerospace	18	320 x x 920	<45	1,4 [PAN] 2,8 [MS]	20
Raptor Imager	Dragonfly Aerospace	45	450 x x 1200	<45	0,7 [PAN] 2,8 [MS]	11
Mantis Imager	Dragonfly Aerospace	0,5	100 x 100 x 650		16 [PAN] 32 [MS]	32
					32	
SpectraCAM	Redwire		50 x 50 x 47			
HyperScape100	SimeraSense	1,1	98 x 98 x176	7	4,75	19,4
MultiScape100 CIS	SimeraSense	1,1	98 x 98 x176	5,8	4,75	19,4
MultiScape200 CIS	SimeraSense	12,1	216 x 216 x 304	5,8	1,5	14
TriScape100	SimeraSense	1,1	98 x 98 x176	5,8	4,75	19,4 x 14,6
TriScape200	SimeraSense	12,1	216 x 216 x 304	5,8	1,5	14 x 10,5
IM200	AAC Clyde Space	0,059	29 x 29 x 70.7	700m		
HRVI-6HD	Berlin Space Technologies GmbH	10,45	540x320x170		4,6 [PAN] 9,2 [MS]	70
HRVI-2HD	Berlin Space Technologies GmbH	19	520 x 780 x 335		1,92 [PAN] 7,68 [MS]	15
SEEING 1.5-m / 0.75-m	Safran Reosc				1,5	
SEEING 10-m	Safran Reosc	8	180x180x250	30	10	
Monitor Imager	SCS Space	3			6,5	13
STREEGO	Media Lario	20	600 x 700 x 850	17	2,75	22
22mm Camera	KAIROSPACE Co., Ltd.	1	221 x 74 x 91		37 @600 km	125 @600 km
90mm Camera	KAIROSPACE Co., Ltd.	1,4	101 x 243		3 @400 km	9,95 @400 km
250mm Camera	KAIROSPACE Co., Ltd.	6	234 x 234 x 407		2,5 @600 km	25 @600 km
22mm Cluster Camera	KAIROSPACE Co., Ltd.	2,4	200 x 91 x 91	12-48		
HyperScout® 1	cosine Remote Sensing B.V.				67	280

HyperScout® 2	cosine Remote Sensing B.V.				67	280
C3D CubeSat Camera	XCAM	0,085	950 x 910 x 270	845m	360 @650 km	
Micro Camera System	Crystalspace	0,05				
Visible Spectral Camera	Chang Guang Satellite	29			<5	
JG-V1430G	Chang Guang Satellite	4,5			<1,9 [PAN] <7,6 [MS]	
JG-V1850G	Chang Guang Satellite	5			<2	
JG-V3200G	Chang Guang Satellite	40			0,7 [PAN] 2,8 [MS]	
JG-P275K	Chang Guang Satellite	5			<10 [PAN] <40 [MS]	
JG-P1100G	Chang Guang Satellite	6,2			<2,5 [PAN] <10 [MS]	
JG-P1430G	Chang Guang Satellite	4,5			<1,9 [PAN]	
JG-P1750G	Chang Guang Satellite	5			<1 [PAN] <4 [MS]	
JG-P3200G	Chang Guang Satellite	40			<0,7 [PAN] <2,8 [MS]	
JG-P3200G-S	Chang Guang Satellite	81			<0,76 [PAN] <3,1 [MS]	
JG-P4850K	Chang Guang Satellite	600			<1 [PAN] <4 [MS]	
JG-P10000G	Chang Guang Satellite	<1100			<0,5 [PAN] <2 [standard MS] <4 [extended MS] <5 [SWIR]	
ECAM-IR1	Malin Space Science Systems	0,330 W/O OP.	78 x 58 x 63	8,75		
ECAM-C30	Malin Space Science Systems	0,256 W/O OP.	78 x 58 x 44	2,5		
ECAM-C50	Malin Space Science Systems	0,256 W/O OP.	78 x 58 x 44	2,5		
DISC	Space Dynamics Laboratory	0,7		1		
SpaceViewTM 24 (SV-24)	L3Harris Technologies	10		10	0,9-1,1	
SpaceViewTM 35 (SV-35)	L3Harris Technologies	20-35		70-170	0,7-1,0	
SpaceViewTM 42 (SV-42)	L3Harris Technologies	25-40		70-170	0,5-0,75	
SpaceViewTM 50 (SV-50)	L3Harris Technologies	90-130		200-275	0,35-0,5	
SpaceViewTM 80 (SV-80)	L3Harris Technologies	150-225		250-350	0,22-0,35	

Annex-Table 2: GSD

	GSD @500 km [m]	GSD @150 km [m]	GSD @80 km [m]
iSIM-90 VNIR SWIR	1,65	0,50	0,26
iSIM-170 VNIR SWIR	0,80	0,24	0,13
DragonEye Imager	1,40	0,42	0,22
Raptor Imager	0,70	0,21	0,11
MultiScape200 CIS	1,50	0,45	0,24
TriScape200	1,50	0,45	0,24
HRVI-2HD	1,92	0,58	0,31
SEEING 1.5-m / 0.75-m	1,50	0,45	0,24
JG-V1430G	1,90	0,57	0,30
JG-V3200G	0,70	0,21	0,11
JG-P1430G	1,90	0,57	0,30
JG-P1750G	1,00	0,30	0,16
JG-P3200G	0,70	0,21	0,11
JG-P3200G-S	0,76	0,23	0,12
JG-P4850K	1,00	0,30	0,16
JG-P10000G	0,50	0,15	0,08
SpaceViewTM 24 (SV-24)	1,10	0,33	0,18
SpaceViewTM 35 (SV-35)	1,00	0,30	0,16
SpaceViewTM 42 (SV-42)	0,75	0,23	0,12
SpaceViewTM 50 (SV-50)	0,50	0,15	0,08
SpaceViewTM 80 (SV-80)	0,35	0,11	0,06

Annex-Table 3: Optical payload requirements

INSTRUMENT	REQUIREMENTS	Min	Max
VIS camera	GSD @500 km	0,35 m	1,92 m
	Swath @500 km	11 km	20 km
	Resolution	4096 x 3072 pixel	
	Number of spectral bands	4	8
	Scannig technique	Push-broom	
Hyperspectral camera	GSD @600 km	5,50 m	80,4 m
	Swath @600 km	19,5 km	280 km
	Resolution	4096 x 3072 pixel	
	Number of spectral bands	32	150
	Scannig technique	Push-broom	

Annex-Table 4: Subsystems

Subsystem	Components
Structure	Frames and Brackets
Attitude and Orbit Control System (AOCS)	ADC Processor
	3 Reaction Wheels
	2 Star Trackers
	1 Inertial Measurement Unit (IMU), including accelerometers and gyro
	NavCam
Command & Data Handling (C&DH)	C&DH Processor
	Data Storage Module
	Clock
Electric Power System (EPS)	Power Control and Distribution Unit (PCDU)
	Deployable Solar Panels
	Body Mounted Solar Panel
	Battery Pack(s)
Communication System	S-band (Antenna+Transceiver)
	UHF System (Antenna + Transceiver)
Thermal Control System	Heaters
	Passive components
Navigation System	NavCam - Visual based + Landmarking algorithms
	(optional RF / ISL with the Motehrcraft)
Propulsion System	Cold Gas thruster vs Electric propulsion (i.e. Resistojet or FEEP)

Annex-Table 5: Raspberry cameras

Product	Reso-lution	Image size	FOV[°]	FOV [rad]	f[mm]	f/ Max	f/ Min	d_L [mm]		λ [m]	H [m]	GSD [m] Min	GSD [m] Max	GSD [mm] Min	GSD [mm] Max	Swath_PB [m]
[Modulo camera V2]	3280 x 2464	-	62,20	1,09	3,04	2,0		1,52		5,50E-07	10,00	0,00441		4,41		12,06
[Videocamera HQ ufficiale]	4056 x 3040															
6 mm Wide Lens		1/2"	63,0	1,10	6	1,2		5,00		5,50E-07	10,00	0,00134		1,34		12,26
16 mm Telephoto Lens		1"	44,6	0,78	16	1,4	16	11,4	1,00	5,50E-07	10,00	0,00059	0,00671	0,59	6,71	8,20
		2/3"	30,0	0,52												5,36
		1/1.8"	24,7	0,43												4,38
		1/2"	21,8	0,38												3,85
25 mm Telephoto Lens		2/3"	20,2	0,35	25	1,4	16	17,8	1,56	5,50E-07	10,00	0,00038	0,00429	0,38	4,29	3,56
		1/1.8	16,5	0,29												2,90
		1/2"	14,5	0,25												2,54
35 mm Telephoto Lens		1"	20,9	0,36	35	1,7	16	20,5	2,19	5,50E-7	10,0	0,00033	0,00307	0,33	3,07	3,69
		2/3"	14,4	0,25												2,53
		1/2"	10,5	0,18												1,84
Zoom lens min		1/2.3"	45,0	0,79	8	1,4		5,71		5,50E-7	10,00	0,00117		1,17		8,28
Zoom lens max			5,35	0,09	50			35,71		5,50E-7		0,00019		0,19		0,93

Annex-Table 6: Swath

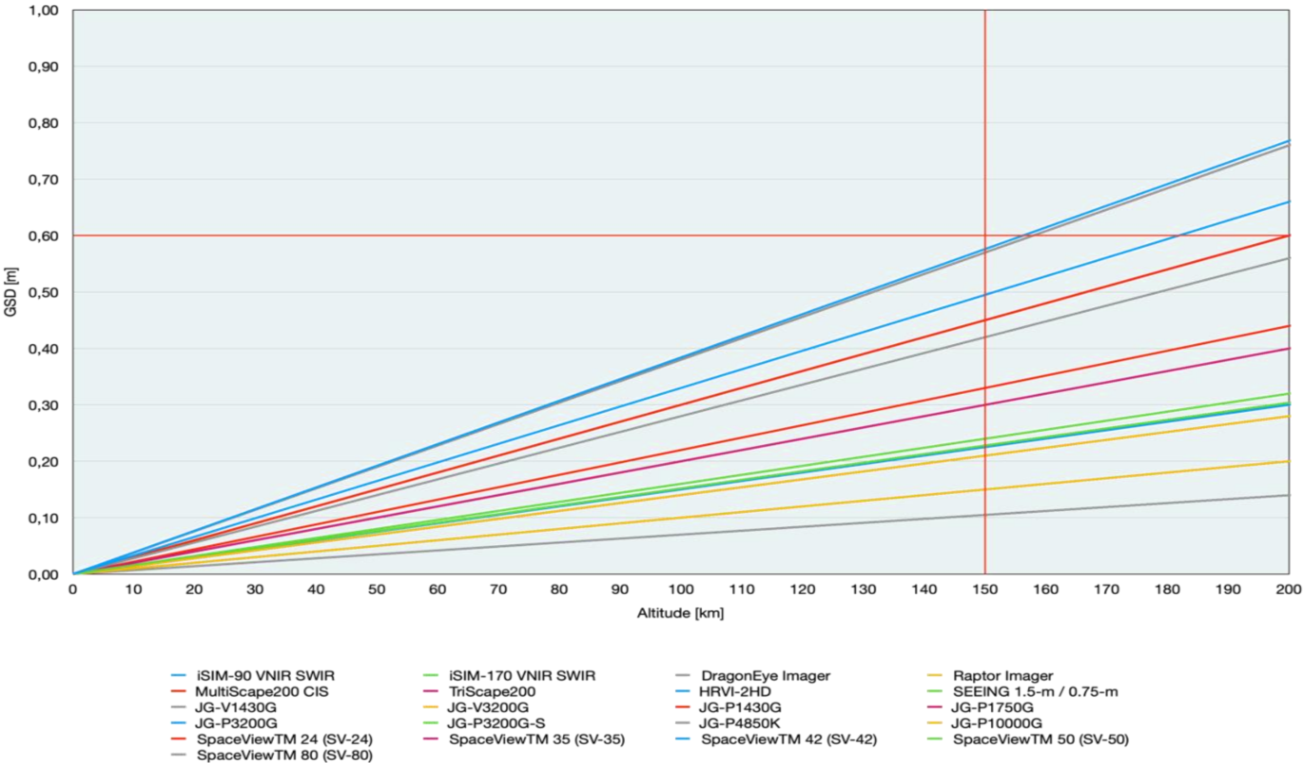
	Swath @500 km [km]	Swath @150 km [km]	Swath @80 km [km]
iSIM-90 VNIR SWIR	13,00	3,90	2,08
iSIM-170 VNIR SWIR	13,00	3,90	2,08
DragonEye Imager	20,00	6,00	3,20
Raptor Imager	11,00	3,30	1,76
MultiScape200 CIS	14,00	4,20	2,24
TriScape200	14,00	4,20	2,24
HRVI-2HD	15,00	4,50	2,40

Annex-Table 7: Space camera

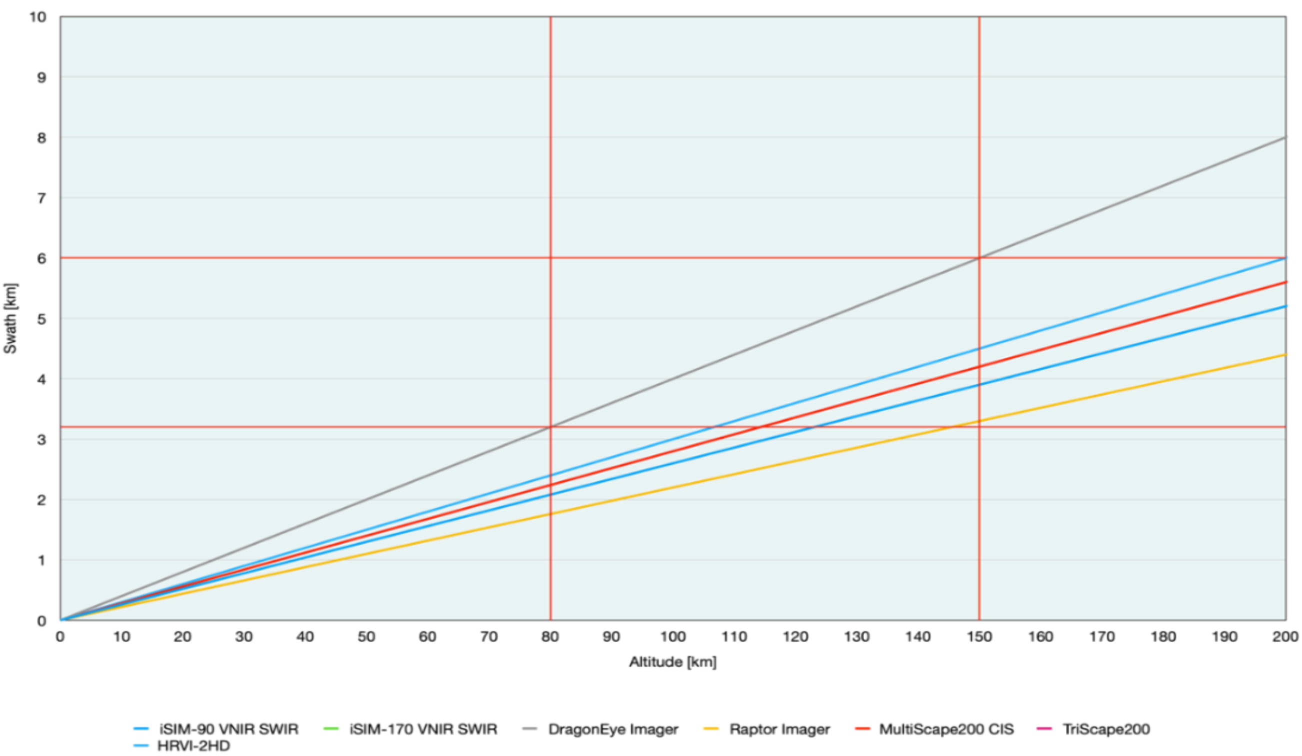
SPACE CAMERAS	GSD @500 km [m]	GSD @ 10 m [m]
iSIM-90 VNIR SWIR	1,65	0,0000330
iSIM-170 VNIR SWIR	0,80	0,0000160
DragonEye Imager	1,40	0,0000280
Raptor Imager	0,70	0,0000140
MultiScape200 CIS	1,50	0,0000300
TriScape200	1,50	0,0000300
HRVI-2HD	1,92	0,0000384
SEEING 1.5-m / 0.75-m	1,50	0,0000300
JG-V1430G	1,90	0,0000380
JG-V3200G	0,70	0,0000140
JG-P1430G	1,90	0,0000380
JG-P1750G	1,00	0,0000200
JG-P3200G	0,70	0,0000140
JG-P3200G-S	0,76	0,0000152
JG-P4850K	1,00	0,0000200
JG-P10000G	0,50	0,0000100
SpaceViewTM 24 (SV-24)	1,10	0,0000220
SpaceViewTM 35 (SV-35)	1,00	0,0000200
SpaceViewTM 42 (SV-42)	0,75	0,0000150
SpaceViewTM 50 (SV-50)	0,50	0,0000100
SpaceViewTM 80 (SV-80)	0,35	0,0000070

Annex-Table 8: GSD hyperspectral camera

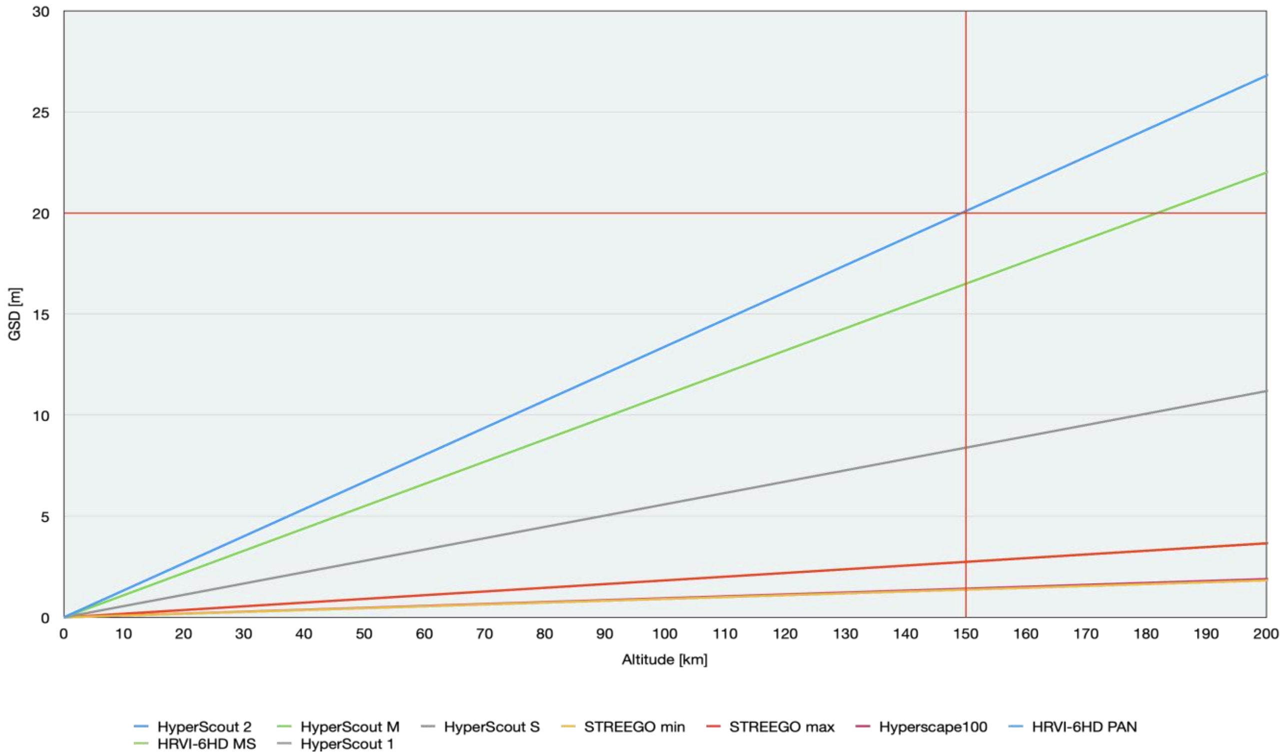
	GSD @600 km [m]	GSD @150 km [m]
STREEGO min	5,50	1,38
STREEGO max	11,00	2,75
Hyperscape100	5,70	1,43
HRVI-6HD PAN	5,52	1,38
HRVI-6HD MS	11,04	2,76
HyperScout 1	80,40	20,10
HyperScout 2	80,40	20,10
HyperScout M	66,00	16,50



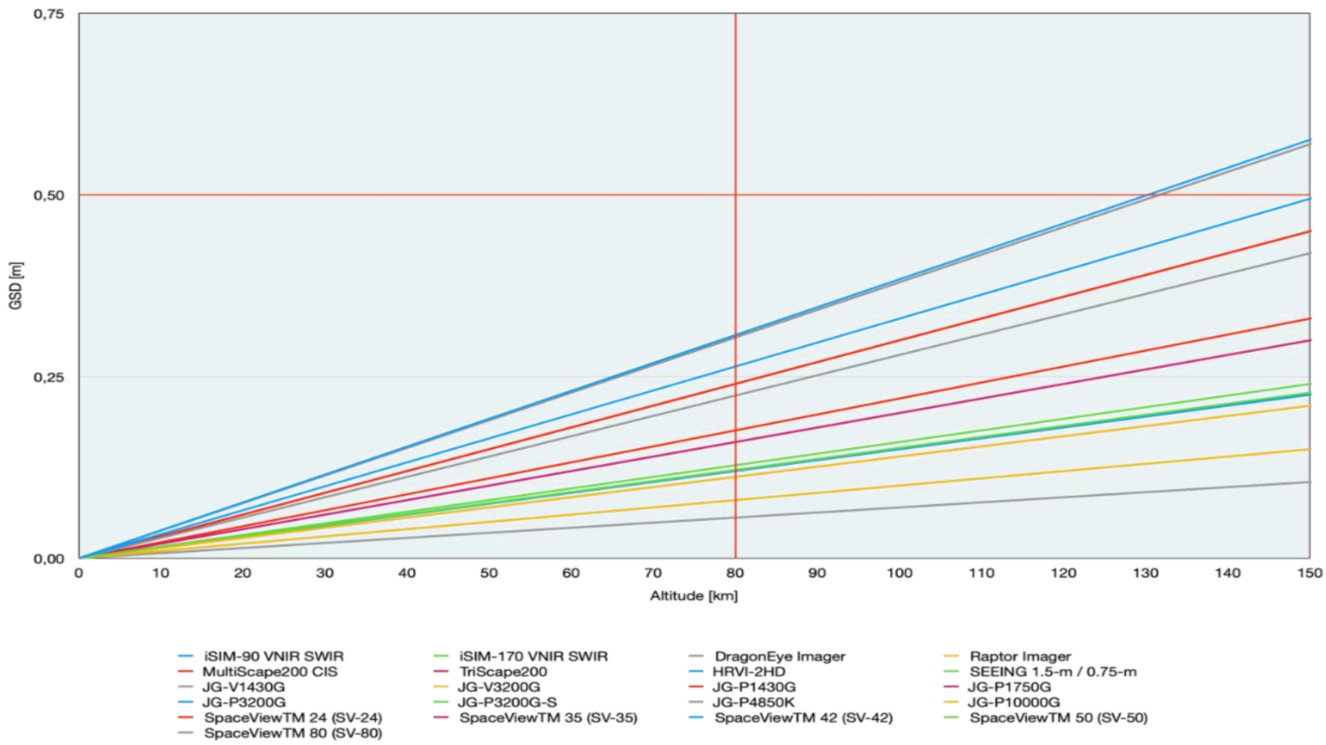
Annex-Figure 1: GSD for navigation purposes



Annex-Figure 2: Swath optical camera



Annex-Figure 3: GSD hyperspectral camera



Annex-Figure 4: GSD for remote sensing application

7.2 Propulsion Annex

7.2.1 Cold-Gas thruster

Annex-Table 9: Cold-gas thrusters - trade off - mass

Mass	50-820 Triad	58E163A	058-118	058 E151	CGT	058E142A	058 E146		SUM	%
50-820 Triad	1	0,3	0	0,25	0,2	0	0,2		1,95	0,06964
58E163A	0,7	1	0,2	0,3	0,25	0	0,25		2,7	0,09643
058-118	1	0,8	1	0,75	0,7	0,3	0,7		5,25	0,1875
058 E151	0,75	0,7	0,25	1	0,3	0,2	0,3		3,5	0,125
CGT	0,8	0,75	0,3	0,7	1	0,2	0,5		4,25	0,15179
058E142A	1	1	0,7	0,8	0,8	1	0,8		6,1	0,21786
058 E146	0,8	0,75	0,3	0,7	0,5	0,2	1		4,25	0,15179
TOT									28	1

Annex-Table 10: Cold-gas thrusters - trade off - power

Power	50-820 Triad	58E163A	058-118	058 E151	CGT	058E142A	058 E146		SUM	%
50-820 Triad	1	0,2	0	0,25		0	0		1,45	0,06905
58E163A	0,8	1	0,3	0,5		0,2	0,2		3	0,14286
058-118	1	0,7	1	0,8		0,5	0,5		4,5	0,21429
058 E151	0,75	0,5	0,2	1		0,2	0,2		2,85	0,13571
CGT										
058E142A	1	0,8	0,5	0,8		1	0,5		4,6	0,21905
058 E146	1	0,8	0,5	0,8		0,5	1		4,6	0,21905
TOT									21	1

Annex-Table 11: Cold-gas thrusters - trade off - thrust

Thrust	50-820 Triad	58E163A	058-118	058 E151	CGT	058E142A	058 E146		SUM	%
50-820 Triad	1	0	0	0	0,5	0	0		1,5	0,05357
58E163A	1	1	0,5	0,5	1	0,5	0,5		5	0,17857
058-118	1	0,5	1	0,5	1	0,5	0,5		5	0,17857
058 E151	1	0,5	0,5	1	1	0,5	0,5		5	0,17857
CGT	0,5	0	0	0	1	0	0		1,5	0,05357
058E142A	1	0,5	0,5	0,5	1	1	0,5		5	0,17857
058 E146	1	0,5	0,5	0,5	1	0,5	1		5	0,17857
TOT									28	1

Annex-Table 12: Cold-gas thrusters - trade off - I_{SP}

I_sp	50-820 Triad	58E163A	058-118	058 E151	CGT	058E142A	058 E146		SUM	%
50-820 Triad										
58E163A		1	0,5	0,5	0,7	0,5	0,5		3,7	0,17619
058-118		0,5	1	0,5	0,7	0,5	0,5		3,7	0,17619
058 E151		0,5	0,5	1	0,7	0,5	0,5		3,7	0,17619
CGT		0,3	0,3	0,3	1	0,3	0,3		2,5	0,11905
058E142A		0,5	0,5	0,5	0,7	1	0,5		3,7	0,17619
058 E146		0,5	0,5	0,5	0,7	0,5	1		3,7	0,17619
TOT									21	1

Annex-Table 13: Cold-gas thrusters - trade off – m_P

m_p	50-820 Triad	58E163A	058-118	058 E151	CGT	058E142A	058 E146		SUM	%
50-820 Triad										
58E163A		1	0,5	0,5	0,3	0,5	0,5		3,3	0,15714
058-118		0,5	1	0,5	0,3	0,5	0,5		3,3	0,15714
058 E151		0,5	0,5	1	0,3	0,5	0,5		3,3	0,15714
CGT		0,7	0,7	0,7	1	0,7	0,7		4,5	0,21429
058E142A		0,5	0,5	0,5	0,3	1	0,5		3,3	0,15714
058 E146		0,5	0,5	0,5	0,3	0,5	1		3,3	0,15714
TOT									21	1

Annex-Table 14: Cold-gas thrusters - trade off - complexity

Complexity	50-820 Triad	58E163A	058-118	058 E151	CGT	058E142A	058 E146		SUM	%
50-820 Triad	1	0	0	0	1	0	0		2	0,07143
58E163A	1	1	0,5	0,5	1	0,5	0,5		5	0,17857
058-118	1	0,5	1	0,5	1	0,5	0,5		5	0,17857
058 E151	1	0,5	0,5	1	1	0,5	0,5		5	0,17857
CGT	0	0	0	0	1	0	0		1	0,03571
058E142A	1	0,5	0,5	0,5	1	1	0,5		5	0,17857
058 E146	1	0,5	0,5	0,5	1	0,5	1		5	0,17857
TOT									28	1

Annex-Table 15: Cold-gas thrusters - trade off - technology

Technology	50-820 Triad	58E163A	058-118	058 E151	CGT	058E142A	058 E146		SUM	%
50-820 Triad	1	0,2	0,2	0,2	0,5	0,2	0,2		2,5	0,08929
58E163A	0,8	1	0,5	0,5	1	0,5	0,5		4,8	0,17143
058-118	0,8	0,5	1	0,5	1	0,5	0,5		4,8	0,17143
058 E151	0,8	0,5	0,5	1	1	0,5	0,5		4,8	0,17143
CGT	0,5	0	0	0	1	0	0		1,5	0,05357
058E142A	0,8	0,5	0,5	0,5	1	1	0,5		4,8	0,17143
058 E146	0,8	0,5	0,5	0,5	1	0,5	1		4,8	0,17143
TOT									28	1

7.2.2 Monopropellant thruster

Annex-Table 16: Monopropellant thruster - trade off - mass

Mass	1N Monoprop Hydrazine	50N Monoprop Hydrazine	MR-104H510N	MR-107V 300N	MRM-122 130N	AJ10-200	R-6F 22N		SUM	%
1N Monoprop Hydrazine	1	0,7	1	1	0,8	1	1		6,5	0,23214
50N Monoprop Hydrazine	0,3	1	1	0,8	0,7	1	0,8		5,6	0,2
MR-104H510N	0	0	1	0,25	0	0,3	0,25		1,8	0,06429
MR-107V 300N	0	0,2	0,75	1	0,3	0,75	0,5		3,5	0,125
MRM-122 130N	0,2	0,3	1	0,7	1	1	0,7		4,9	0,175
AJ10-200	0	0	0,7	0,25	0	1	0,25		2,2	0,07857
R-6F 22N	0	0,2	0,75	0,5	0,3	0,75	1		3,5	0,125
TOT									28	1

Annex-Table 17: Monopropellant thruster - trade off - power

Power	1N Monoprop Hydrazine	50N Monoprop Hydrazine	MR-104H510N	MR-107V 300N	MRM-122 130N	AJ10-200	R-6F 22N		SUM	%
1N Monoprop Hydrazine	1		1	1	1				4	0,4
50N Monoprop Hydrazine										
MR-104H510N	0		1	0,8	0,7				2,5	0,25
MR-107V 300N	0		0,2	1	0,8				2	0,2
MRM-122 130N	0		0,3	0,2	1				1,5	0,15
AJ10-200										
R-6F 22N										
TOT									10	1

Annex-Table 18: Monopropellant thruster - trade off - thrust

Thrust	1N Monoprop Hydrazine	50N Monoprop Hydrazine	MR-104H510N	MR-107V 300N	MRM-122 130N	AJ10-200	R-6F 22N		SUM	%
1N Monoprop Hydrazine	1	1	1	1	1	1	1		7	0,25
50N Monoprop Hydrazine	0	1	1	0,75	0,75	0,5	0,25		4,25	0,15179
MR-104H510N	0	0	1	0,25	0	0,3	0		1,55	0,05536
MR-107V 300N	0	0,25	0,75	1	0,5	0,3	0,2		3	0,10714
MRM-122 130N	0	0,25	1	0,5	1	0,5	0,25		3,5	0,125
AJ10-200	0	0,5	0,7	0,7	0,5	1	0,25		3,65	0,13036
R-6F 22N	0	0,75	1	0,8	0,75	0,75	1		5,05	0,18036
TOT									28	1

Annex-Table 19: Monopropellant thruster - trade off - I_{SP}

I_sp	1N Monoprop Hydrazine	50N Monoprop Hydrazine	MR-104H510N	MR-107V 300N	MRM-122 130N	AJ10-200	R-6F 22N		SUM	%
1N Monoprop Hydrazine	1	0,3	0,5	0,5	0,5	0,8	1		4,6	0,16429
50N Monoprop Hydrazine	0,7	1	0,7	0,7	0,7	0,7	1		5,5	0,19643
MR-104H510N	0,5	0,3	1	0,5	0,5	0,5	1		4,3	0,15357
MR-107V 300N	0,5	0,3	0,5	1	0,5	0,5	1		4,3	0,15357
MRM-122 130N	0,5	0,3	0,5	0,5	1	0,5	1		4,3	0,15357
AJ10-200	0,2	0,3	0,5	0,5	0,5	1	0,8		3,8	0,13571
R-6F 22N	0	0	0	0	0	0,2	1		1,2	0,04286
TOT									28	1

Annex-Table 20: Monopropellant thruster - trade off – m_P

m_p	1N Monoprop Hydrazine	50N Monoprop Hydrazine	MR-104H510N	MR-107V 300N	MRM-122 130N	AJ10-200	R-6F 22N		SUM	%
1N Monoprop Hydrazine	1	0,7	0,3	0,3	0,3	0,25	0,25		3,1	0,110714
50N Monoprop Hydrazine	0,3	1	0,25	0,25	0,25	0,2	0,2		2,45	0,0875
MR-104H510N	0,7	0,75	1	0,5	0,5	0,3	0,3		4,05	0,144643
MR-107V 300N	0,7	0,75	0,5	1	0,5	0,3	0,3		4,05	0,144643
MRM-122 130N	0,7	0,75	0,5	0,5	1	0,3	0,3		4,05	0,144643
AJ10-200	0,75	0,8	0,7	0,7	0,7	1	0,5		5,15	0,183929
R-6F 22N	0,75	0,8	0,7	0,7	0,7	0,5	1		5,15	0,183929
TOT									28	1

Annex-Table 21: Monopropellant thruster - trade off - complexity

Complexity	1N Monoprop Hydrazine	50N Monoprop Hydrazine	MR-104H510N	MR-107V 300N	MRM-122 130N	AJ10-200	R-6F 22N		SUM	%
1N Monoprop Hydrazine	1	1	1	1	1	1	1		7	0,25
50N Monoprop Hydrazine	0	1	1	0,75	0,75	0,5	0,25		4,25	0,151786
MR-104H510N	0	0	1	0,25	0	0,3	0		1,55	0,055357
MR-107V 300N	0	0,25	0,75	1	0,5	0,3	0,2		3	0,107143
MRM-122 130N	0	0,25	1	0,5	1	0,5	0,25		3,5	0,125
AJ10-200	0	0,5	0,7	0,7	0,5	1	0,25		3,65	0,130357
R-6F 22N	0	0,75	1	0,8	0,75	0,75	1		5,05	0,180357
TOT									28	1

Annex-Table 22: Monopropellant thruster - trade off - technology

Technology	1N Monoprop Hydrazine	50N Monoprop Hydrazine	MR-104H510N	MR-107V 300N	MRM-122 130N	AJ10-200	R-6F 22N		SUM	%
1N Monoprop Hydrazine	1	0,5	0,8	0,8	0,75	0,7	0,5		5,05	0,180357
50N Monoprop Hydrazine	0,5	1	0,8	0,8	0,75	0,7	0,5		5,05	0,180357
MR-104H510N	0,2	0,2	1	0,5	0,3	0,25	0,2		2,65	0,094643
MR-107V 300N	0,2	0,2	0,5	1	0,3	0,25	0,2		2,65	0,094643
MRM-122 130N	0,25	0,25	0,7	0,7	1	0,5	0,2		3,6	0,128571
AJ10-200	0,3	0,3	0,75	0,75	0,5	1	0,3		3,9	0,139286
R-6F 22N	0,5	0,5	0,8	0,8	0,8	0,7	1		5,1	0,182143
TOT									28	1

7.2.3 Electromagnetic thruster

Annex-Table 23: Electromagnetic thruster - trade off - mass

Mass	BHT-100	BHT-350	BHT-600	Halo MicroElectric Thruster	HT-400	HT-100	MR-512 Low Power Arcjet	XR-5 Hall Thruster		SUM	%
BHT-100	1	0,7	0,5	0,3	0,5	0,2	0,7	1		4,9	0,136111
BHT-350	0,3	1	0,3	0,3	0,25	0	0,5	1		3,65	0,101389
BHT-600	0,5	0,7	1	0,25	0,5	0,2	0,5	1		4,65	0,129167
Halo MicroElectric Thruster	0,7	0,7	0,75	1	0,7	0,3	0,8	1		5,95	0,165278
HT-400	0,5	0,75	0,5	0,3	1	0,25	0,7	1		5	0,138889
HT-100	0,8	1	0,8	0,7	0,75	1	0,75	1		6,8	0,188889
MR-512 Low Power Arcjet	0,3	0,5	0,5	0,2	0,3	0,25	1	1		4,05	0,1125
XR-5 Hall Thruster	0	0	0	0	0	0	0	1		1	0,027778
TOT										36	1

Annex-Table 24: Electromagnetic thruster - trade off - power

Power	BHT-100	BHT-350	BHT-600	Halo MicroElectric Thruster	HT-400	HT-100	MR-512 Low Power Arcjet	XR-5 Hall Thruster		SUM	%
BHT-100	1	0,7	0,75	0,75	0,8	0,75	1	1		6,75	0,1875
BHT-350	0,3	1	0,3	0,3	0,7	0,3	1	1		4,9	0,136111
BHT-600	0,25	0,7	1	0,5	0,8	0,5	1	1		5,75	0,159722
Halo MicroElectric Thruster	0,25	0,7	0,5	1	0,8	0,5	1	1		5,75	0,159722
HT-400	0,2	0,3	0,2	0,2	1	0,8	1	1		4,7	0,130556
HT-100	0,25	0,7	0,5	0,5	0,2	1	1	1		5,15	0,143056
MR-512 Low Power Arcjet	0	0	0	0	0	0	1	0,5		1,5	0,041667
XR-5 Hall Thruster	0	0	0	0	0	0	0,5	1		1,5	0,041667
TOT										36	1

Annex-Table 25: Electromagnetic thruster - trade off - thrust

Thrust	BHT-100	BHT-350	BHT-600	Halo MicroElectric Thruster	HT-400	HT-100	MR-512 Low Power Arcjet	XR-5 Hall Thruster		SUM	%
BHT-100	1	0,7	0,8	0,7	0,8	0,75	1	1		6,75	0,1875
BHT-350	0,3	1	0,75	0,5	0,75	0,7	1	1		6	0,166667
BHT-600	0,2	0,25	1	0,25	0,5	0,3	1	1		4,5	0,125
Halo MicroElectric Thruster	0,3	0,5	0,75	1	0,75	0,7	1	1		6	0,166667
HT-400	0,2	0,25	0,5	0,25	1	0,3	1	1		4,5	0,125
HT-100	0,25	0,3	0,7	0,3	0,7	1	1	1		5,25	0,145833
MR-512 Low Power Arcjet	0	0	0	0	0	0	1	0,5		1,5	0,041667
XR-5 Hall Thruster	0	0	0	0	0	0	0,5	1		1,5	0,041667
TOT										36	1

Annex-Table 26: Electromagnetic thruster - trade off - I_{SP}

I_sp	BHT-100	BHT-350	BHT-600	Halo MicroElectric Thruster	HT-400	HT-100	MR-512 Low Power Arcjet	XR-5 Hall Thruster		SUM	%
BHT-100	1	0,7	0,8	0,8	1	0,75	0	0		5,05	0,140278
BHT-350	0,3	1	0,7	0,7	0,8	0,5	0	0		4	0,111111
BHT-600	0,2	0,3	1	0,5	0,75	0,3	0	0		3,05	0,084722
Halo MicroElectric Thruster	0,2	0,3	0,5	1	0,75	0,3	0	0		3,05	0,084722
HT-400	0	0,2	0,25	0,25	1	0,25	0	0		1,95	0,054167
HT-100	0,25	0,5	0,7	0,7	0,75	1	0	0		3,9	0,108333
MR-512 Low Power Arcjet	1	1	1	1	1	1	1	0,5		7,5	0,208333
XR-5 Hall Thruster	1	1	1	1	1	1	0,5	1		7,5	0,208333
TOT										36	1

Annex-Table 27: Electromagnetic thruster - trade off – m_P

m_p	BHT-100	BHT-350	BHT-600	Halo MicroElectric Thruster	HT-400	HT-100	MR-512 Low Power Arcjet	XR-5 Hall Thruster		SUM	%
BHT-100	1	0,3	0,25	0,25	0,2	0,3	1	1		4,3	0,119444
BHT-350	0,7	1	0,3	0,3	0,25	0,5	1	1		5,05	0,140278
BHT-600	0,75	0,7	1	0,5	0,3	0,7	1	1		5,95	0,165278
Halo MicroElectric Thruster	0,75	0,7	0,5	1	0,3	0,7	1	1		5,95	0,165278
HT-400	0,8	0,75	0,7	0,7	1	0,75	1	1		6,7	0,186111
HT-100	0,7	0,5	0,3	0,3	0,25	1	1	1		5,05	0,140278
MR-512 Low Power Arcjet	0	0	0	0	0	0	1	0,5		1,5	0,041667
XR-5 Hall Thruster	0	0	0	0	0	0	0,5	1		1,5	0,041667
TOT										36	1

Annex-Table 28: Electromagnetic thruster - trade off - complexity

Complexity	BHT-100	BHT-350	BHT-600	Halo MicroElectric Thruster	HT-400	HT-100	MR-512 Low Power Arcjet	XR-5 Hall Thruster		SUM	%
BHT-100	1	0,5	0,5	0,8	0,75	0,7	1	1		6,25	0,173611
BHT-350	0,5	1	0,5	0,8	0,75	0,7	1	1		6,25	0,173611
BHT-600	0,5	0,5	1	0,8	0,75	0,7	1	1		6,25	0,173611
Halo MicroElectric Thruster	0,2	0,2	0,2	1	0,5	0,3	1	1		4,4	0,122222
HT-400	0,25	0,25	0,25	0,5	1	0,3	1	1		4,55	0,126389
HT-100	0,3	0,3	0,3	0,7	0,7	1	1	1		5,3	0,147222
MR-512 Low Power Arcjet	0	0	0	0	0	0	1	0,5		1,5	0,041667
XR-5 Hall Thruster	0	0	0	0	0	0	0,5	1		1,5	0,041667
TOT										36	1

Annex-Table 29: Electromagnetic thruster - trade off - technology

Technology	BHT-100	BHT-350	BHT-600	Halo MicroElectric	HT-400	HT-100	MR-512 Low Power	XR-5 Hall Thruste		SUM	%
BHT-100	1	0,5	0,5	0,75	0,7	0,7	1	1		6,15	0,170833
BHT-350	0,5	1	0,5	0,75	0,7	0,7	1	1		6,15	0,170833
BHT-600	0,5	0,5	1	0,75	0,7	0,7	1	1		6,15	0,170833
Halo MicroElectric Thruster	0,25	0,25	0,25	1	0,3	0,3	1	1		4,35	0,120833
HT-400	0,3	0,3	0,3	0,7	1	0,5	1	1		5,1	0,141667
HT-100	0,3	0,3	0,3	0,7	0,5	1	1	1		5,1	0,141667
MR-512 Low Power Arcjet	0	0	0	0	0	0	1	0,5		1,5	0,041667
XR-5 Hall Thruster	0	0	0	0	0	0	0,5	1		1,5	0,041667
TOT										36	1

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