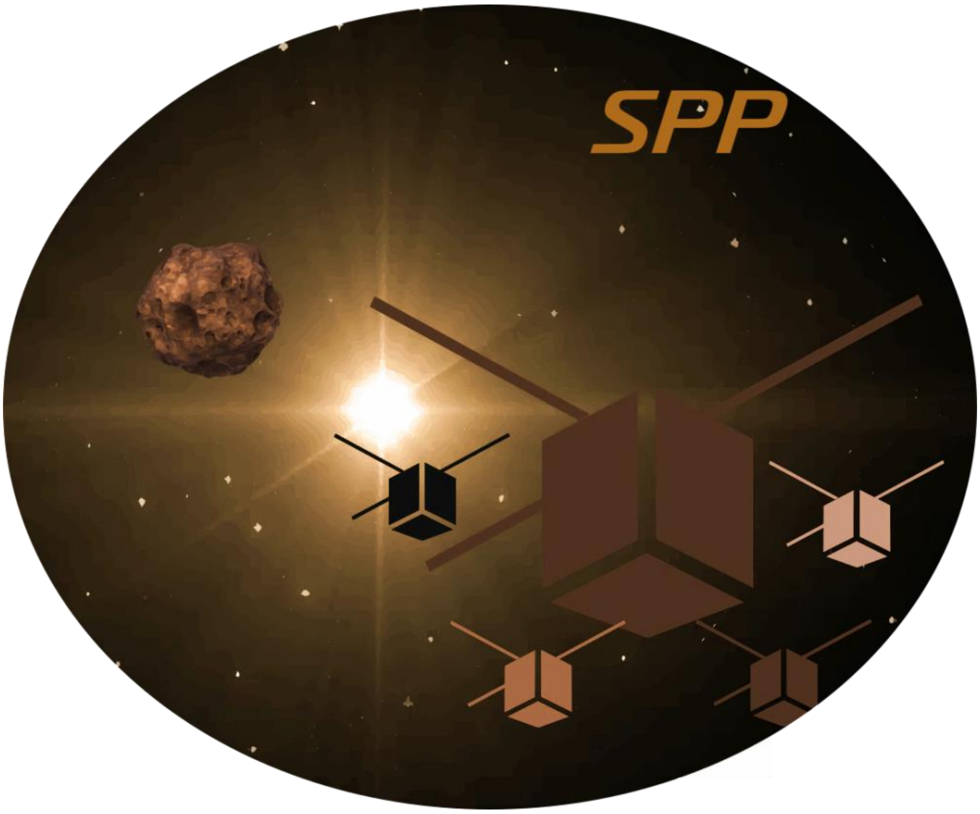
CDF Study Report

SPP NEO Inactive Body

Small Planetary Platforms Assessment

for NEO Inactive Bodies



concurrent design facility

CDF-178(A)
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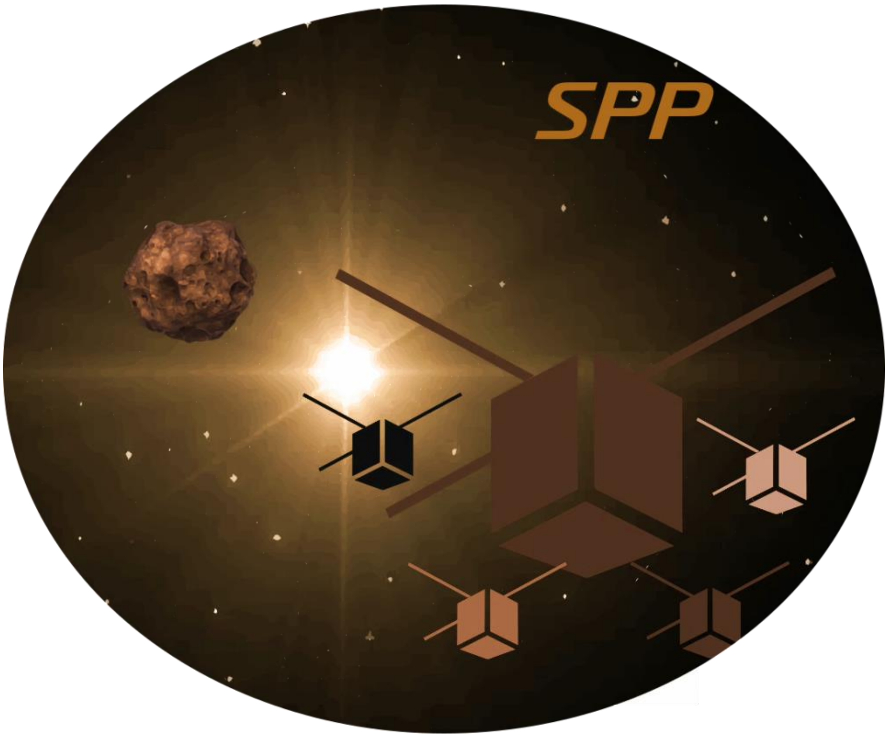

FRONT COVER

Study Logo showing satellite approaching an asteroid with a swarm of SmallSats

STUDY TEAM

This study was performed in the ESTEC Concurrent Design Facility (CDF) by the following interdisciplinary team:

TEAM LEADER			
AOCS		PAYLOAD	
COMMUNICATIONS		POWER	
CONFIGURATION		PROGRAMMATICS/ AIV	
COST		ELECTRICAL PROPULSION	
DATA HANDLING		CHEMICAL PROPULSION	
GS&OPS		SYSTEMS	
MISSION ANALYSIS		THERMAL	
MECHANISMS			

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With the technical support of:

Systems/APIES

Smallsats

Radiation

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1 INTRODUCTION

1.1 Background

Requested by SCI-FM and financed by GSP, the CDF Small Planetary Platforms (SPP) study carried out an assessment of small planetary mission concepts including a mothercraft and a swarm of smallsatellites. The study was organised in 8 design sessions, starting with a Kick Off on the 8th November 2017 and ending with an Internal Final Presentation on the 6th December 2017. An additional session with a reduced number of specialists took place at the end of January 2018 to look into the concept of a multi-asteroid tour with small satellites. The design team consisted of a multidisciplinary team of experts and included input from science and other directorates.

The concept studied was a proposal to perform multi-point (and possibly multi-target) measurements around small bodies (asteroids and comets), as well as Mars or Venus allowing the scientific community to gather information from different locations simultaneously. The potential interest in "multi-point measurement science", has been highlighted following missions like Rosetta.

1.2 Objective

The main goal was not to design a specific mission but to provide a “tool-box” of technical building blocks that the community can use to develop new planetary missions architectures, in reply to future science calls.

The objectives of the SPP study was to:

- Assess the feasibility of performing deep space planetary missions with an architecture consisting of a mothership spacecraft carrying a swarm of smallsats to be deployed for multi-point science observations.

1.3 Scope

The scope of the study was very wide ranging and rather than follow the traditional CDF study concept of trying to reduce the options and then studying a small number of them in detail, this study expanded the options to try to increase the potential usage of the toolbox.

- Highlight the main operational constraints (i.e. max communication range vs achievable data rates, communication links between the mothership and the swarm, max number of smallsats, etc.) imposed by the architecture, identifying technical solutions for a variety of scenarios including rendez-vous missions to small bodies, as well as missions around Mars and Venus.
- Identify any new specific technology developments enabling missions.
- Preliminarily design the mothercraft and the smallsats and perform parametric analysis to understand the flexibility/adaptability of the design to various environments.
- Assess the possibility of adding a lander asset on the surface of the small body.

- Provide a portfolio of potential transfers to small bodies for launches between 2024 and 2034.
- Define the programmatic approach, including the procurement of the smallsats as part of the payload complement.
- Assess the mission cost, with a target of 150M€ (i.e. fit in an "F class").

1.4 Document Structure

The layout of this report is different to a standard CDF Study, in that there are 3 main reports, one covering SPP for NEO Inactive Bodies (this document), one covering SPP for Main Asteroid Belt Active Bodies (CDF-178(B)) and an Executive Summary that compiles the main aspects of the two documents, the system-level and main sub-system level trade-offs and covers the top level synthesis (CDF-178(C)). Details of the study results can be seen in the Table of Contents. The details of each domain addressed in the study are contained in specific chapters.

Due to the different distribution requirements, only cost assumptions excluding figures are given in this report. The costing information is published in a separate document.

Note: In the drawings and figures included in this report sometimes the acronym NS is used to refer to the smallsats. NS and SS should be understood as one and the same thing.

2 PAYLOAD

2.1 Requirements and Design Drivers MC

It is assumed that the mother spacecraft contains no scientific payload for the purpose of this study.

2.2 Requirements and Design Drivers SS

The main scientific theme for this mission addresses the internal structure of an asteroid. The key instrumentation are surface penetrating radars that decipher the internal lithologies and stratigraphy. Depending on the geological history a variety of body structures may have been formed:

- Primordial agglomeration (uniform internal structure)
- Aqueous alteration, "mud ball" (structure undefined)
- Differentiated body (onion shell)
- Primordial rubble pile (primordial boulders of very different size agglomerated)
- Destructive rubble pile (fatal destruction by impact and re-aggregation)
- Formation of regolith layer (fine dust to meter-sized boulder, in principle all bodies, small bodies may have lost it).

In depth radar tomography will provide a complete understanding of the building formation and evolution of the target body. The surface topography and structures together with the mineralogical composition are intimately linked to the formation history.

The low-frequency radar (large depths) is located at two different spacecraft to enable bi-static measurements. The camera contributes to the development of the global shape model which is mandatory for proper interpretation of the radar data. The high-frequency radar (shallow depths) and IR spectrometer is distributed on another two spacecraft.

There is no strong scientific requirement to locate the high-frequency radar and IR spectrometer on separate spacecraft. They could be mounted on the mother spacecraft as well to fully achieve their scientific goals.

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
PAY SS-010	Low frequency radar penetration depth: 100s of meter	
PAY SS-020	Low frequency radar resolution: 20-40 meter	
PAY SS-030	High frequency radar penetration depth: 10s of meter	
PAY SS-040	High frequency radar resolution: 0.2 to 1.0 m	
PAY SS-050	Camera resolution: 10 cm @ 1 km distance	
PAY SS-050	IR spectrometer wavelength range: 500 – 2500 nm with 10-30 nm spectral resolution	

2.3 Assumptions and Trade offs SS

2.3.1 Assumptions

Assumptions	
1	The science themes are: (1) Internal structure of the body; (2) surface topography and structures; (3) surface mineralogy; (4) physical properties of target body
2	The strawman payload compliment is used to specify the resource requirements and operational requirements towards the spacecraft and mission operations.
3	The share of resources incl. data volume is an example only to test the feasibility of the mission design and will require further refinement on the basis of scientific justification.
4	Instrument examples were preferably taken from European sources. Exceptions are possible if justified by performance to meet the scientific goal.
5	A mass limit of 3.0 kg incl. 20% margin was set initially to the study.
6	A nominal operational distance of 5 km distance to surface was used.

2.3.2 Trade Offs and Selection of Instruments

In Table 2-1 the list of payload instrumentation is summarised per satellite. The chosen heritage instrument is also listed. A summary of all payload basic resource requirements is found in Table 2-4. A generic value of 0.25 kg per satellite for all instrument harnesses was assumed. This is part of the overall payload mass allocation of 3.0 kg per satellite.

Sat 1	Sat 2	Sat 3	Sat 4
Low frequency radar (DISCUS study)	Low frequency radar (DISCUS study)	High frequency radar (AIM D1 study)	IR spectrometer (BIRCHES/LunarIceSmall, NASA)
Camera (CUCorbiter)	Camera (CUCorbiter)	--	--
Radio Science	Radio Science	Radio Science	Radio Science

Table 2-1: Summary of instrumentation per satellite. The heritage instrument is identified

2.3.2.1 Analysis of deep interior structure

The low-frequency radar instrument has been studied in the context of the DISCUS mission study RD[17]. The basic concept foresees two smallsats (6U) carrying two radar units, a small camera and a drastically miniaturised laser altimeter. The radar will be used in a bi-static configuration in a similar manner as the CONSERT instrument of the Rosetta mission located on the orbiter spacecraft the and the Philae lander RD[19].

The radar works in a stepped frequency mode at a frequency centred at 20 MHz. The dipole antenna has a length of 2 x 3.75 m. The antenna relies on a self deployable tubular boom. A metal strip based boom of this length is a new development however has been a proven concept for shorter lengths up to 1 m. Table 2-2 shows the key characteristic of the low frequency radar according to RD[17].

Centre frequency	20 MHz
Antenna length	half-wave dipole lambda/2
Radar modulation	256 to2048 lines
Transmitting power	10 W
Receiver bandwidth	2 MHz

Table 2-2: Characteristic of the low-frequency radar

2.3.2.2 Analysis of shallow interior structure

The upper tens of meters of the asteroidal surface is typically covered by the regolith layer. This layer consists of nm to meter sized particles and boulders generated by micro and macro impact processes over the lifetime of the body. The internal structure has never been resolved on an asteroid by remote sensing or in-situ investigations.

The instrument is based on the WISDOM instrument developed for ESAs ExoMars 2020 mission RD[20]. A modified version has been studied in the context of the AIM mission study RD[21].

The instrument consists of an e-ebox and a monolithic static mounted antenna cube. The main emitted frequencies are between 300 and 800 MHz proposed to be operated in stepwise modulation. Operations at higher frequencies are of 2300-2400 MHz are considered.

The main processing of the SAR data will be performed on the data set transmitted to Earth.

2.3.2.3 Surface topography and structures by camera investigations

The camera provides images in the visible wavelength range. Images are used for surface characterisation, topographic map, crater record and development of the shape model. As baseline an advanced design of the ExoMars mission close-up imager, CLUPI, is used. A design study for adaptation as an orbiter camera was performed for the MarcoPolo-R asteroid sample return mission study RD[11].

The lens bases very compact instrument design fulfils the performance of a classic wide angle camera. It is based of a full colour APS. This specific detector is out of production and no longer commercially available. There is also no alternative product featuring the same characteristics currently on the market. Some adaptations of the instrument using up-to-date detector developments are required. Currently no filters are foreseen. Table 2-3 provides the basic characteristics of this camera.

Active pixel sensor (APS)	2652x1768x3 pixel (in colour), x 14 bit
Pixel size	7.8x7.8 µm
Spectral range	400-700 nm
Field of view	14°
Focal length	100 mm
Resolution pre pixel @ 5 km distance	39 cm

Table 2-3: Characteristics of the camera

On the commercial market, various camera designs are being developed for CubeSat application. COSINE (NL) has built a hyperspectral imager covering the wavelength range from 400 to 1000 nm at 42 wavelength bands. It provides a spectral resolution between 5-12 nm. The optical design would provide a spatial resolution of 67 cm per pixel at 5 km distance to surface RD[12]. A demonstrator model is ready for launch on the GOMx-4B cubesat. The launch is scheduled for 2019.

Skylabs (Slovenia) provides imager (NANOimager) in the vis/IR (450-1600 nm) and IR range (1000-2500 nm) with comparable performances and system resource requirements RD[13]. Another candidate is the ASPECT imaging system designed by VTT (Finland). This imaging system combines a three channel design with very low resource budgets to be integrated in a 3U standard cubesat. 2 channels have imaging capabilities in the visible and near infrared wavelength band while the 3rd channel is a actively cooled spot spectrometer with spectral range from 1600 to 2500 nm RD[14].

2.3.2.4 Surface mineralogy by IR spectrometry

Generally speaking the available resources are rather low for the integration of a high resolution IR spectrometer. Certainly no imaging spectrometer would currently fit into this category. Spot spectrometers can be used in pushbroom or pushwhisk mode stitching a uniform surface map together. A sufficient spatial resolution is pre-requisite for such an application.

A spot spectrometer is currently built for the NASA LunarIceCube mission which is scheduled for launch in 2019/20 time frame. This spectrometer covers a very large wavelength range from 1000 to 4000 nm with high spectral resolution of 5 nm and very appealing signal to noise ratio of >400. The spatial resolution would not be sufficient for an asteroidal target at 5 km distance. It would be only 500 m, which implies the whole object is covered by one pixel. A proper adaptation of the optical design is required. However, this would likely lead to an increase in mass and volume.

For ESA's SMART1 mission to the Moon (SIR) and for the Indian Chandrayan lunar explorer mission (SIR2) a commercial spot spectrometer by Zeiss (Germany) has been adapted to space environmental conditions and flown successfully RD[15], RD[16]. Both instruments would deliver an acceptable spatial resolution (6 m at 5km distance to surface) at a decreased wavelength range (940-2400 nm).

2.3.2.5 Physical properties by the Radio Science Experiment

The inter satellite communication link of the four spacecraft and the mother spacecraft can be used for the determination of precise orbit positions and subtle influences of the parent body. Currently no resources in addition to the standard spacecraft subsystems are foreseen for this experiment. Neither dedicated orbit operations have been reserved. A later sensitivity analysis shall be performed to achieve a deeper understanding of the possibilities and added scientific value.

It would be beneficial, also for other experiments, to add a simple lase altimeter for absolute measurements of the distance between spacecraft and surface. Given the close distance to the surface, no disturbing atmosphere and non-imaging performance a simple altimeter could be assumed. Although not existing yet rugged and low-resource design are available yet not adapted and space qualified RD[18].

2.4 Baseline Design SS

Table 2-4 contains the basic resource requirements per instrument.

	Mass [kg]	Power [W]	Volume lxbxh [mm]	Data rate or vol.	trl
Low f radar	Instrument 1.0 Antenna 0.5	40	120x150x50 3750x10 (deployed, 2x)	8 Gbit	radar 3 Antenna 2
High f radar	2.4 No power supply included	88.5 (averaged)	Antenna 372x372x273 E-box 280x145x90	300 kbs 39 Gbit whole mission (indicative)	3
Camera	1.0	15	225x100x120	67 Mbit per image no comp.	3
IR spectrometer	2.5	10	100x100x150	4 kbit per spot, no comp.	4
Harness per sat	0.250	--	--	--	--
Sat 1&2	**2.75**	**55**			
Sat 3	**2.65**	**88.5**			
Sat4	**2.75**	**10**			

Table 2-4: Basic resource requirements per instrument

2.5 Accommodation

The low frequency radar is a compact instrument despite their long deployable antennas. Based on the tubular boom technology the stowed volume is comparably to classic stiff boom based devices very low. The mounting location has to provide sufficient space for flawless deployment to both sides of the dipole antenna. Both spacecraft carrying the low frequency radars are complemented with one camera each.

The alignment of antennas with respect to the target and the view direction of the cameras (nadir) shall be coordinated.

The accommodation of the high frequency radar and the IR spectrometer is in principle straight forward since both instruments are the sole payload of their corresponding satellite.

An issue of severe concern is the current dimension of the high frequency antenna. Due to the large volume of currently 372x372x273 mm it appears as too large for accommodation on small (ie 6U or 12U) smallsats. Although not a viable option on this study an accommodation on the mother spacecraft should be considered for accommodation.

2.6 Operational Aspects

The basic assumptions forming the standard observation conditions are as follows:

- Target body diameter is 600 m
- 6 months of observation
- 5 km distance to surface for standard observation in a circular orbit
- No specific operations of special instruments demands were analysed.

It is understood that certain specific investigations would require a different configuration of the spacecrafts around the target. The full complexity of observations deviating from the standard scenario could not be assessed during this study.

The cameras and the IR spectrometer are operated nadir pointing. Deviations from this rule can be implemented with ease for example if any limb observations are required.

The camera is considered as the most sensitive factor defining a requirement towards the spacecraft pointing stability (RPE). It is assumed that 1\2 pixel smear over the integration time is acceptable. Further, an exposure time of 0.1 second and an instantaneous field of 0.12 mrad are assumed as instrument design parameter. This would lead to an RPE requirement of 12 arcsec over 0.1 second. Uncertainties within the instrument performance and albedo of the actual target as key design parameter have to be addressed at a later stage in much greater detail.

The phase angle S/C-Sun-target for camera observation is ideally 30 to 60 degrees.

The low frequency radar requires a distance from 5-10 km from the centre of mass of the body. The two spacecraft form an angle of 35 (±10%) degree during observation. For the subsequent orbit reconstruction the height above the surface and polar angles relative to the body shall be known with an accuracy of ±1.5 degree. The height accuracy obtained by the transmitting spacecraft is in the order of 15 m. The height information of the receiving spacecraft must come from a different, external source ideally by an (laser) altimeter. The post mission analysis requires the construction of a shape model based on camera data. A pixel resolution of 0.5 meter or better is sufficient.

Ideally the angle between the orbiting plane normal and the spin axis is 90 degree during the observations. It could be demonstrated that an angle down to 30 degree still provides a valuable set of data RD[22]. In a special observation configuration both spacecraft are opposite to each other while the radar beam penetrates the body.

In the current design the high frequency radar has an optimised operational distance between 1 km and 10 km from the surface (the object stays fully in the field of view). The approximate distance between spacecraft and asteroid surface within 100 m accuracy is essential to obtain prior the instrument operations. The wavelength intervals have to be selected accordingly. A relative velocity of approximately 1 m/s between spacecraft and target is currently assumed. In principle the instrument can adapt to different velocities by adapting and commanding different instrument parameter.

The acceptable deviation from pointing in nadir direction is ±10 degree.

2.7 Data Volume

For the estimate of the data volume produced by the camera a generic 2kx2k detector with 16 bit depth has been assumed. As a benchmark value 1500 images have been selected as sufficient to support complete imaging at different phase angles and for the reconstruction of the shape model RD[17]. A compression factor of 2 has been applied which leads to a data volume of 50.33 Gbit per camera.

The example IR spectrometer has in fact a too low spatial resolution. Since the generic data volume also strongly depends on the surface coverage a dummy value of 6 m surface coverage per investigated spot was set. Further, an area of 2x the actual surface and a compression factor of 1.8 are underlying the data volume calculation. This accumulates to a data volume of 3.43 Gbit.

The data volume of the low frequency radar and the high frequency radar are 8.0 Gbit and 39.0 Gbit, respectively. These numbers were adapted from the corresponding instrument studies RD[17] and RD[21]

Table 2-5 presents a summary of the data volume generated throughout the nominal mission life time.

active body	**SC 1** Low frequency radar Camera	**SC2** Low frequency radar Camera	**SC3** High frequency radar	**SC4** IR spectrometer	Total mission
					Σ
data vol [Gbit]	58.33	58.33	39	3.43	**(159.09)** **100.0**

Table 2-5: Data volume

The two identical cameras mounted on two spacecrafts very likely produce a significant amount of redundant data. Thus it appeared as appropriate to reduce the total amount of data to achieve a feasible value that eventually can be transmitted to Earth in most mission scenarios and not only under optimum conditions.

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3 MISSION ANALYSIS

The mission analysis tasks included the assessment of Delta-V required for different types of target, the analysis of launch strategies and optimisation of the transfer trajectory to the target. Trade-offs of dedicated vs. shared launch, and of the mission target were performed. The mission analysis concentrated more on the MC side. While some aspects of the close proximity operations were analysed, a more detailed analysis including the definition of the operational approach was performed by the GNC subsystem. Mass and Delta-V budgets, timelines and transfer geometry profiles, were provided during the CDF study as input for the design of the different subsystems.

3.1 Requirements and Design Drivers MC

3.1.1 Multipoint Mission

The MIS-010 requirement requires to look at simultaneous science around:

- Small bodies – at max distance of TBD AUs
- Mars/Phobos

3.1.2 Launch Vehicle

MIS-060 states that either a single launch with the Epsilon and/or Vega(-C) launchers or a shared launch on Ariane 6.2 should be the baseline. A comprehensive trade-off on the launch options has been conducted and is presented in Section 3.2.2.

3.1.3 Launch Date

Following MIS-070, the launch date shall be between 2024 and 2034. This affects the reachable targets since, depending on their periods, this timeframe might be favourable or not. Moreover, it adds constraint to the possibility of shared launch.

3.1.4 Transfer Duration

To reduce mission costs, excessive mission durations shall be avoided. The selected final target shall be reached after a maximum of 5 years (TBC) after launch (MIS-100). This requirement leads to a design driver on the type of transfer that should be analysed and feasible. Transfers of typically around 2-4 years were searched for, with a higher limit of 5 years.

3.1.5 Duration of Operations

According to MIS-110, 6 months (TBC) of science operations are foreseen after deployment of the Smallsats. Therefore, 6 months of propagation once at target were included in the different data and plots provided, in order to take into account this period. This is constraining the transfers in the sense that conjunctions are to be avoided during these months of operation. Moreover, in the case of a mission to a comet, one might want to coincide this period of operations with the perihelion to increase the scientific output.

3.2 Assumptions and Trade-Offs MC

3.2.1 Small Body Classification

The classification of small bodies is ambiguous and the reasons why an object is named asteroid, comet or something else is not broadly standardised and can lead to confusion. As well explained in RD[24]:

The classification of small bodies in the inner solar system as either asteroids or comets has historically been attempted by different scientists using different techniques and employing different criteria. Observational astronomers classify small bodies having transient, unbound atmospheres (usually made visible by the scattering of sunlight from entrained micron-sized dust particles) as comets. Bodies having instead a constant geometric cross-section are called asteroids. To planetary scientists, comets and asteroids are distinguished by their ice content or perhaps by their formation location. Comets are icy (because they formed beyond the "snow-line") while asteroids are not (supposedly because they formed at higher mean temperatures inside it). Lastly, to dynamicists, comets and asteroids are broadly distinguished by a dynamical parameter, most usually the Tisserand parameter measured with respect to Jupiter […].

For this CDF study, objects are called asteroid or comet, depending on which database they are listed by the International Astronomical Union (IAU) on the Minor Planet Center website RD[25].

3.2.1.1 Near Earth Objects (NEOs)

More than 17,000 objects are classified as NEOs. Most of them are shown in Figure 3-1. The only criterion is:

- $r_p < 1.3\ AU$

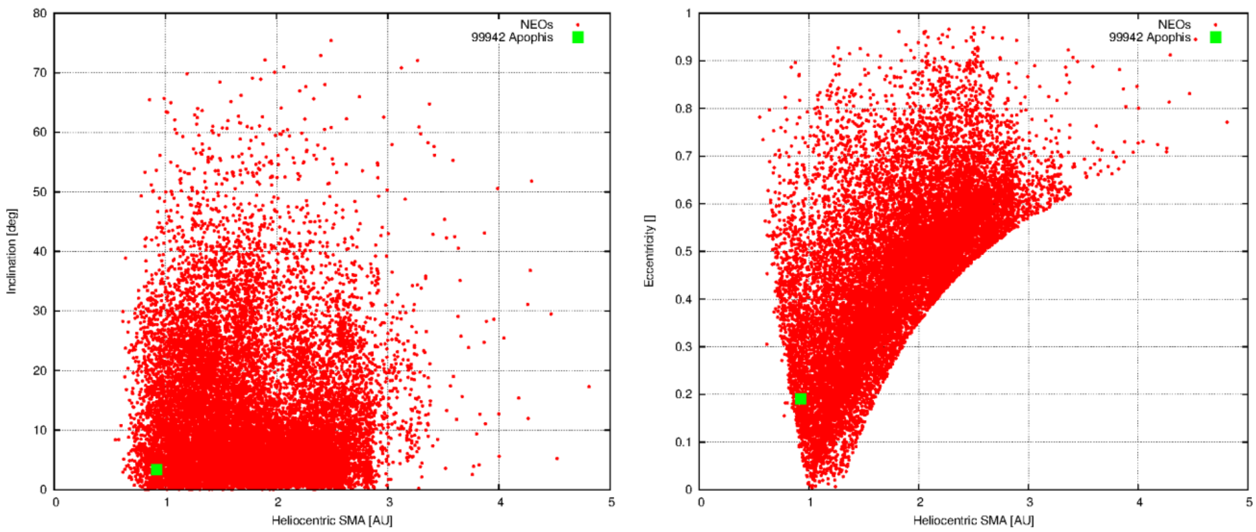

Figure 3-1: Distribution of most NEOs over SMA, inclination (left) and eccentricity (right) values

This database encompasses objects with a wide range of SMAs, eccentricity and inclination which leads to a wide range of Delta-V.

3.2.1.2 Main-Belt Asteroids (MBAs)

In total, around 700,000 objects have been discovered in the Main Asteroid Belt. Most of them are shown in Figure 3-2. The criteria are:

- $2\ AU < a < 3.2\ AU$
- $r_p > 1.666\ AU$

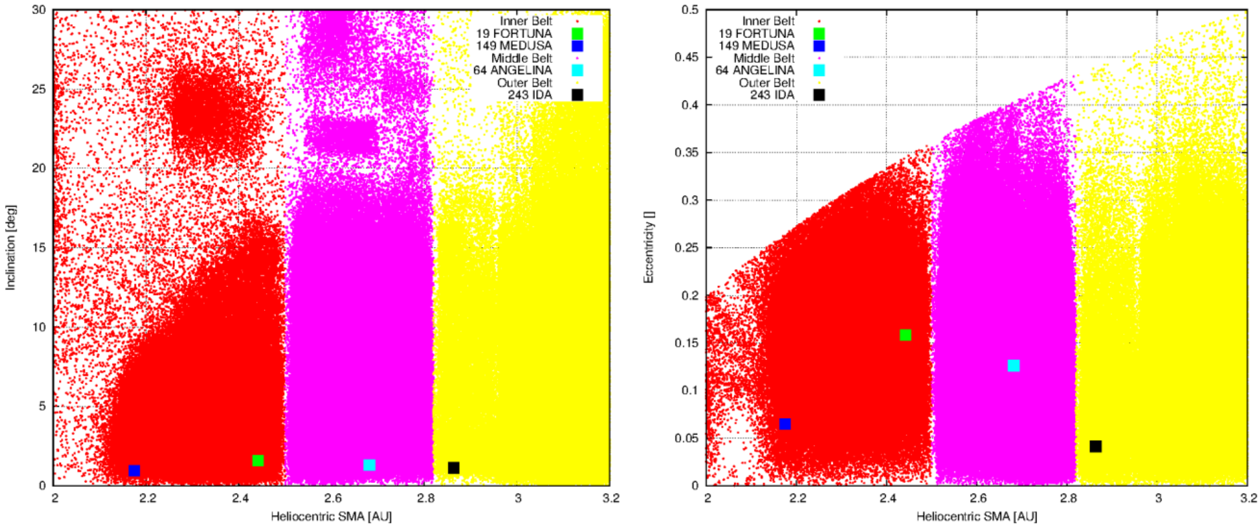

Figure 3-2: Distribution of most MBAs over SMA, inclination (left) and eccentricity (right) values

MBAs are divided in three main families separated by the Kirkwood gaps, depending on the value of their semi-major axis.

1. Inner Belt ($a < 2.5\ AU$)
2. Middle Belt ($2.5\ AU < a < 2.82\ AU$)
3. Outer Belt ($a > 2.82$)

The main belt contains some active bodies that are sometimes called Main Belt Comets.

3.2.1.3 Comets

There are currently ~940 objects in the comet database, distributed over a wide range of inclination and eccentricity values as can be seen in Figure 3-3.

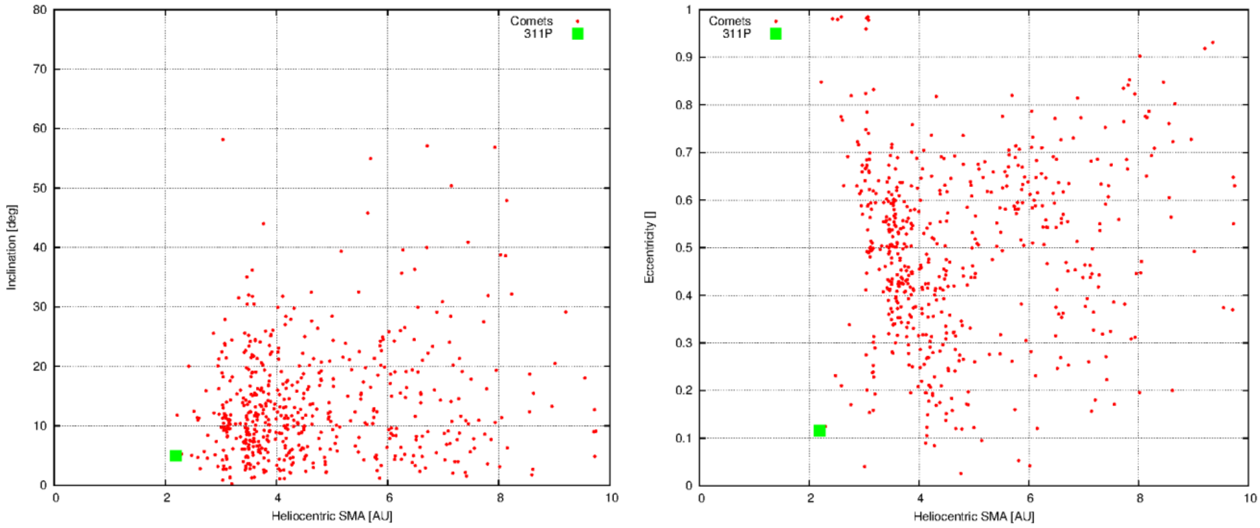

Figure 3-3: Distribution of a number of comets over SMA, inclination (left) and eccentricity (right) values

3.2.2 Launch Strategies and Trade-Offs

The trade space in terms of launch options is shown in Figure 3-4 with two main branches: dedicated and shared launch.

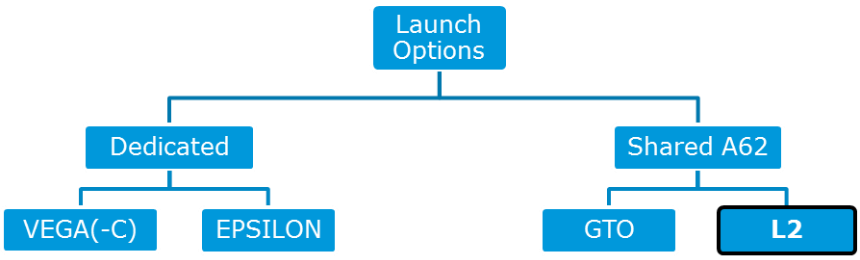

Figure 3-4: Launch options considered

3.2.2.1 Dedicated launch – Vega, Vega-C and Epsilon

The use of small launchers, the European Vega & Vega-C and the Japanese Epsilon, has been considered during initial trade-offs. For this scenario dedicated launch into LEO and Earth escape via the platform on-board propulsion has been regarded. For a first assessment a solution inspired by the LPF propulsion module is assumed. Thus a CP kick-off stage is assumed to provide the necessary burns (up to 7 to limit the negative effect of gravity losses) to reach the required escape infinite velocity.

The propulsion module is assumed to have a dry mass of 215 kg and be provided with a 450 N engine with a specific impulse of 320s. The dry mass is based on the LPF propulsion module which can be loaded with about 1200 kg of propellant. Reaching Earth escape requires larger Delta-V than what LPF required to transfer towards the SEL1 point, so that larger propellant mass might be needed. This might lead to larger propellant tanks. However, this has not been taken into account in this simplified analysis.

Vega:

The injection orbit for the launch with Vega is assumed the same as for LPF, an elliptic 200x1625 km orbit at 5 degrees inclination. For this orbit, very precise launcher performances data is available, resulting in a spacecraft separated mass of 1910 kg (RD[26]). The computation of the sequence of burns to achieve escape assumes a series of burns to raise apogee to an altitude of 300,000 km (higher apogee should be avoided due to large lunar perturbations), a burn close to apogee to turn the orbit plane and a last burn at the final perigee pass accelerating into the hyperbolic escape. The orbit plane change is needed to achieve declinations of the launch asymptote of up to 30 degrees such as not to penalise the transfer orbit with significant DSM Delta-V due to launching into a near-equatorial declination.

The optimisation of the apogee raise assuming a maximum of 5 burns to reduce the gravity losses and avoid too long and complex operations leads to a Delta-V of 2910 m/s. The plane turn manoeuvre at 300000 km to reach an inclination of 30 degrees requires 105 m/s. The final escape burn is computed to maximise the spacecraft mass at Earth escape.

The results for Vega are provided in Table 3-1, which shows the overall Delta-V including all gravity losses and the spacecraft escape mass including the kick-stage dry mass.

V∞ (km/s)	ΔV (m/s)	Propellant (kg)	SC Escape Mass (kg)
1.0	3176	1216	694
2.0	3313	1245	665
3.0	3544	1293	617
4.0	3874	1354	556
5.0	4310	1426	484
6.0	4856	1504	406

Table 3-1: Vega launch into LEO + escape with CP kick-stage

Vega-C:

For the upgraded Vega-C version, the assumption is to inject the spacecraft in the same orbit as with Vega (200x1625 km orbit at 5 degrees inclination) and to use exactly the same escape strategy. Currently the exact performance of Vega-C is not known. An educated guess is to assume 40% increase over the Vega performances, which leads to a separated spacecraft mass of 2674 kg.

The results for Vega-C are shown in Table 3-2. Due to the lower thrust-to-mass ratio, gravity losses are a bit larger in this case, 10-14% overall to be compared to 6-8% for the Vega case. Raising the apogee to 300,000 km requires 3053 m/s. The propellant mass being 50-80% more than for LPF implies that a larger kick-stage dry mass should be considered.

V∞ (km/s)	ΔV (m/s)	Propellant (kg)	SC Escape Mass (kg)
1.0	3323	1746	928
2.0	3461	1787	887
3.0	3698	1851	823
4.0	4043	1937	737
5.0	4510	2039	635
6.0	5097	2147	527

Table 3-2: Vega-C launch into LEO + escape with CP kick-stage

Epsilon:

The injection orbit for Epsilon is assumed: 250x500 km at an inclination of 31 degrees. The separated spacecraft mass in this orbit is 1200 kg (RD[27]). The inclination of the injection orbits allows reaching escape declinations between -31 and +31 degrees. Thus there is no need to change the inclination with the plane turn manoeuvre.

The results for Epsilon are shown in Table 3-3.

V∞ (km/s)	ΔV (m/s)	Propellant (kg)	SC Escape Mass (kg)
2.0	3379	791	409
3.0	3677	828	372
4.0	4002	865	335

Table 3-3: Epsilon launch into LEO + escape with CP kick-stage

The conclusion of the previous assessment, considering a fixed kick-stage dry mass of 215 kg and a reference escape velocity at infinity of 3 km/s is that Vega and Vega-C can deliver a 400 kg and 600 kg platform, respectively, while Epsilon can deliver a platform mass below 200 kg. From the first iteration with system such wet masses for the platform were deemed insufficient. Thus this LEO launch option + CP kick stage is found unfeasible.

3.2.2.2 Shared Ariane 62 launch into GTO

Shared launch into GTO was also regarded in the launch trade-offs. This scenario also assumes that the spacecraft is equipped with a CP kick-stage allowing it to perform the subsequent manoeuvres in order to reach escape. A strategy for GTO launch plus 5-burn-escape has been extensively analysed in previous CDF studies (RD[29], RD[30]).

The sequence of events, which covers a time span of 2 to 5 weeks, is as follows:

- Launch into GTO and separation from the launcher upper stage
- A sequence of 3 burns around perigee raises the apogee to at most 300,000 km (higher apogee altitudes to be avoided due to the strong lunar perturbations to the orbit). The apogee raising performed by 3 burns is sufficient to keep the gravity losses below 5%

- Near the apogee of this orbit a manoeuvre is applied to simultaneously change the orbit plane and rotate the line of apsides to achieve the correct orientation required for the escape
- Around the perigee of this pre-escape orbit, a final burn is applied to achieve insertion into the escape hyperbola.

A standard 246x35786 km, 7 degrees inclined GTO is assumed with an argument of perigee of 180 degrees. The Delta-V results provided below are taken from RD[30], which assumed a separated spacecraft mass of 3070 kg into the GTO. A 450 N CP engine with a specific impulse of 317 s is assumed. For the last plane change manoeuvre, it is assumed that a declination of the launch asymptote of +30 deg is needed (worst-case wrt -30 deg). The value of 30 degrees is consistent with the previous assessment for LEO launch.

The current performance estimation for Ariane 62 into GTO is 4500-5000 kg (RD[31]). This is including adapters and dual launch structure. In this preliminary assessment, 2000 kg are assigned to SPP. The results provided in Table 3-4 are thus conservative, because lower gravity losses are expected for this lighter spacecraft.

V∞ (km/s)	ΔV burns 1-3 (m/s)	ΔV burns 4-5 (m/s)	ΔV overall (m/s)	Propellant (kg)	SC Escape Mass (kg)
1.0	680	682	1362	709	1291
2.0	680	688	1368	717	1283
3.0	680	865	1545	785	1215
4.0	680	1250	1930	925	1075
5.0	680	1822	2502	1105	895
6.0	680	2539	3219	1290	710

Table 3-4: Ariane 62 shared GTO launch + escape with CP kick-stage

For 200 kg dry mass of the CP kick-stage, this preliminary assessment shows that a wet platform mass of 1000 kg can be delivered to escape with infinite velocity up to roughly 3 km/s.

3.2.2.3 Shared Ariane 62 Launch Into SEL2

The reader can find more information regarding this option in Section 3.3.3.

3.2.3 Preliminary Assessment of DIFFERENT MISSIONS

In the preparation work for the CDF study, a preliminary assessment of the Delta-V requirements of different mission concepts was carried out. This assessment was based on information already available from past missions and CDF studies. The results of the assessment are shown in Table 3-5. The data in this table was used in the preliminary system trades for launch options and CP versus EP.

Target	Earth V-inf (km/s)	CP ΔV (km/s)	EP ΔV (km/s)
Mars (4-sols)	3	1.65	3.86
Mars (300 km LMO)	3	~3	6.2
Mars – Phobos	**3**	**2.55**	**~5**
NEOs = Option 1	**5**	**2**	**2**
Main Belt Asteroid – Inner	5-6	4-5	4-5
Main Belt Asteroid – Main	6-7	5-6	5-6
Main Belt Asteroid – Outer	7-8	5-6	5-6
Comet Flyby	4	0	0
Comet RDV = Option 2	**4.5-7**	**4-6**	**4-6**

Table 3-5: Preliminary Earth escape velocity and Delta-V for different mission concepts

Presented in the table are two terms: the infinite velocity required at Earth and the Delta-V for the transfer, using either CP or EP. In a mission using direct escape launch, the infinite velocity at Earth is provided by the launch vehicle. Such a launch option typically requires a dedicated medium-size launcher and will not be available for SPP for the sake of reducing the mission cost. Therefore the Earth escape velocity will be provided by the SPP on-board propulsion. This increases the Delta-V required for the mission. The following cases were considered:

- Launch into Earth orbit (LEO or GTO) then escape using CP, likely with a kick-stage module. Escape is achieved via a complex series of manoeuvres around perigee and possibly a plane turn manoeuvre at the last apogee before the final escape burn. The manoeuvres close to Earth are subject to gravity losses. The required Delta-V is thus strongly dependant on the characteristics of the platform, mainly thrust-to-mass ratio and specific impulse, and has to be optimised for each case. Some results for this case have been obtained for the launch trade-offs in Section 3.2.2.
- Launch into Earth orbit and escape using EP. This leads to long spiralling times in Earth orbit, which significantly increases the radiation dose. The initial thrust-to-mass ratio significantly impacts the duration of the spiralling, but the Delta-V to reach the Escape condition (near-zero infinite velocity) remains basically the same. The following two cases were simulated to assess the required Delta-V:
  - Escape from a 700 km circular LEO: Delta-V 7.0 km/s
  - Escape from standard GTO (250x35786 km): Delta-V 3.9 km/s
- In addition to the Delta-V to reach escape, the on-board EP system also has to increase the Earth relative velocity at departure with the value that is provided in the table. Therefore, the overall EP mission Delta-V is composed by the Delta-V to achieve escape + the required Earth infinite velocity + the required EP transfer Delta-V.

- Departure from SEL2 point. In this case it is assumed that the SPP uses its on-board propulsion, CP or EP, to leave the vicinity of SEL2 and start the transfer. Due to the large distance to Earth the gravity losses can be neglected for both propulsion systems and as a first guess the Delta-V to depart from Earth is assumed equal to the required infinite velocity. Thus the overall mission Delta-V is composed of the required Earth infinite velocity + the required transfer Delta-V (CP or EP).

3.3 Baseline Design MC

3.3.1 Target Selection

Four NEO candidates were preselected based on scientific interest. The main orbital elements and characteristics of these targets are listed in Table 3-6.

Body	SMA [AU]	ECC	INC [deg]	Period [y]	Comments
Apophis	0.92	0.19	3.34	0.89	Typical accessible NEOs
2001 WN5	1.71	0.47	1.92	2.24	Might be reachable too
1999 AN10	1.46	0.56	39.93	1.76	High inclination and eccentricity
Ganymede	2.66	0.53	26.69	4.34	High inclination and eccentricity

Table 3-6: Orbital characteristics of 4 pre-selected potential targets

Out of these four preselected targets, two were found potentially reachable within the assumed range of Delta-V for the mission, by looking at the porkchop plots between 2024 and 2034 (cf. Figure 3-5): Apophis and 2001 WN5.

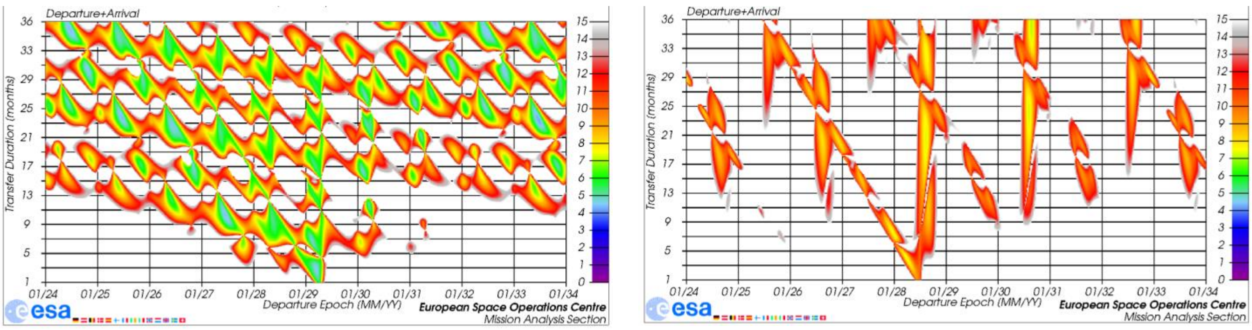

Figure 3-5: Porkchop plots for Apophis (left) and 2001 WN5 (right), departure + arrival velocity

Finally, Apophis was chosen as baseline target for the study.

3.3.2 Apophis

Apophis is a 370 m radius asteroid that has previously been studied by ESA as a potential target for the Don Quijote mission. It is mostly known for the high score it reached on the Torino impact hazard scale before the probability of impact with the

Earth in 2029 and 2036 was recalculated to be lower. The orbit of Apophis as well as the currently estimated orbit after the close encounter are depicted in Figure 3-6.

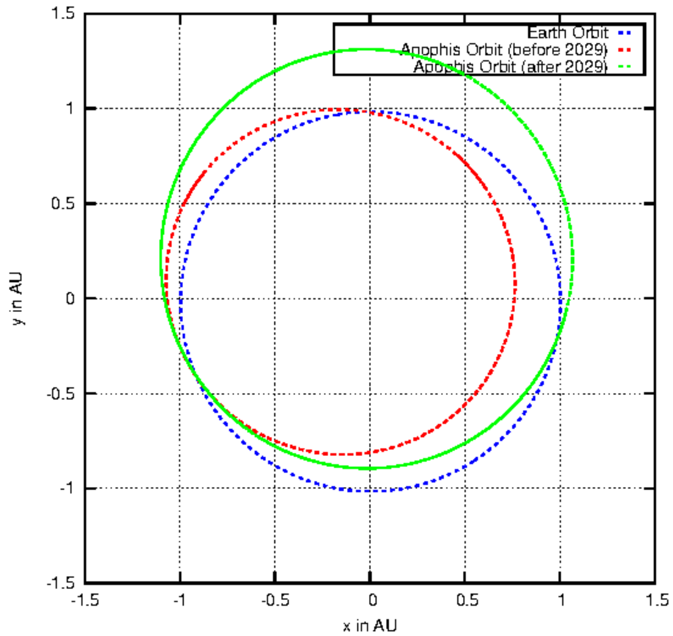

Figure 3-6: Projection of Apophis orbit in the ecliptic plane, EME2000

A close encounter of Earth by Apophis will occur in April 2029, significantly changing its orbital elements. Therefore, the main impacts are major changes in the orbital elements of the asteroid, but also an increase in the uncertainties on the ephemeris of the asteroid. Although the precise orbit of the asteroid will be known with increased exactness thanks to observations at the time of the encounter, a margin on the propellant mass should be added.

If the rendezvous is happening within a few months before the encounter, additional constraints on operations are foreseen because of the Earth gravitational attraction will produce a strong perturbation to the close proximity orbits. This will impact the operations of MC and SS during the encounter.

A comparison of the orbital elements and perihelion/aphelion values for the orbits before and after encounter is done in Table 3-7.

Time	SMA [AU]	Perihelion [AU]	Aphelion [AU]	ECC	INC [deg]
Before	0.923	0.746	1.099	0.192	3.337
After	1.104	0.895	1.313	0.189	2.208

Table 3-7: Orbital elements of Apophis orbit before and after close encounter

This change in orbit will obviously impact the launch windows to reach the target as well as the whole transfers. The porkchop plots for the Apophis orbit before and after encounter are presented Figure 3-7 and Figure 3-8. The diagonal black lines show approximately the transfers with arrival in April 2029.

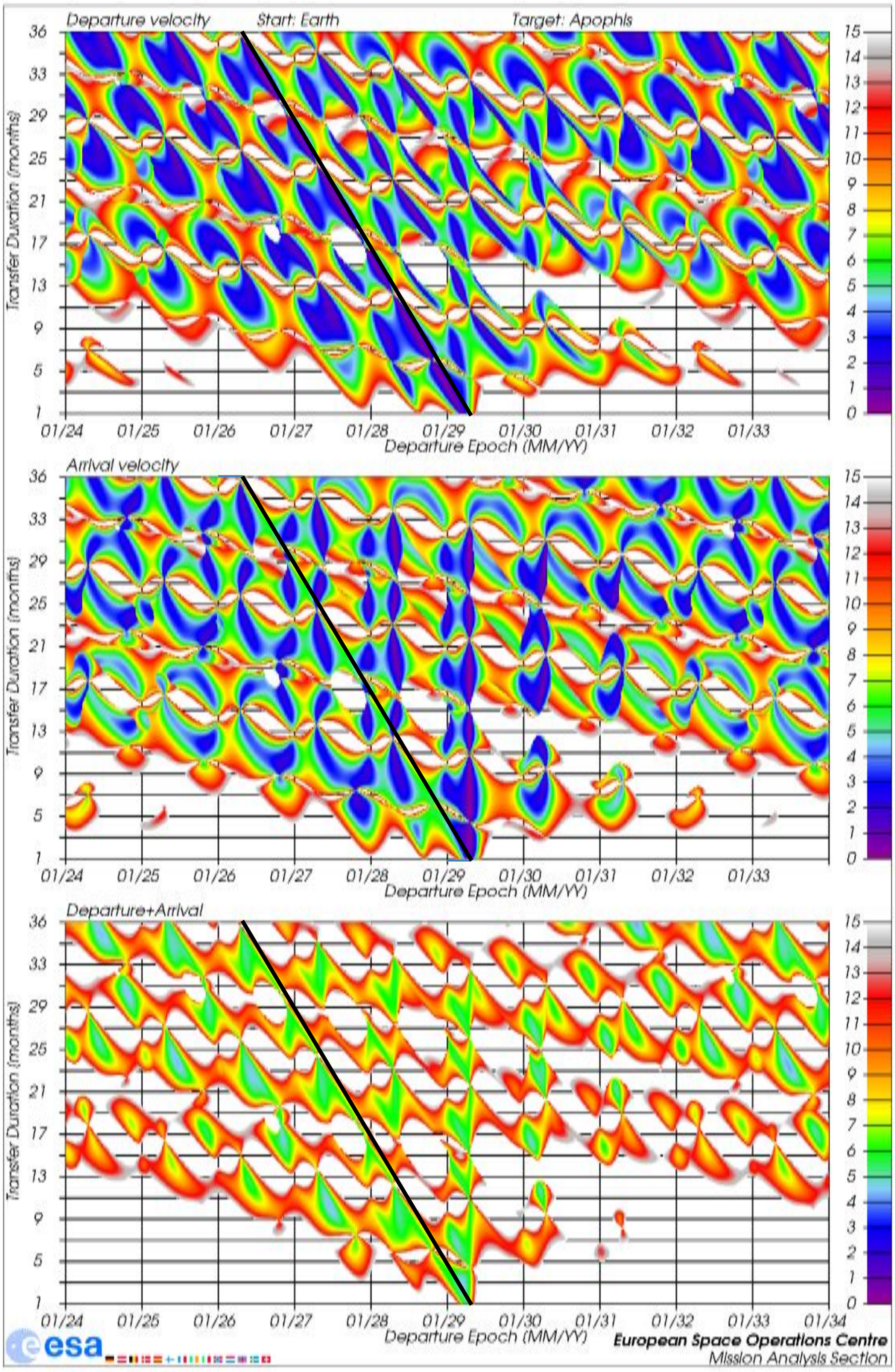

Figure 3-7: Porkchop plot for CP transfers to Apophis with departure between 2024 and 2034 – Valid until 04/29

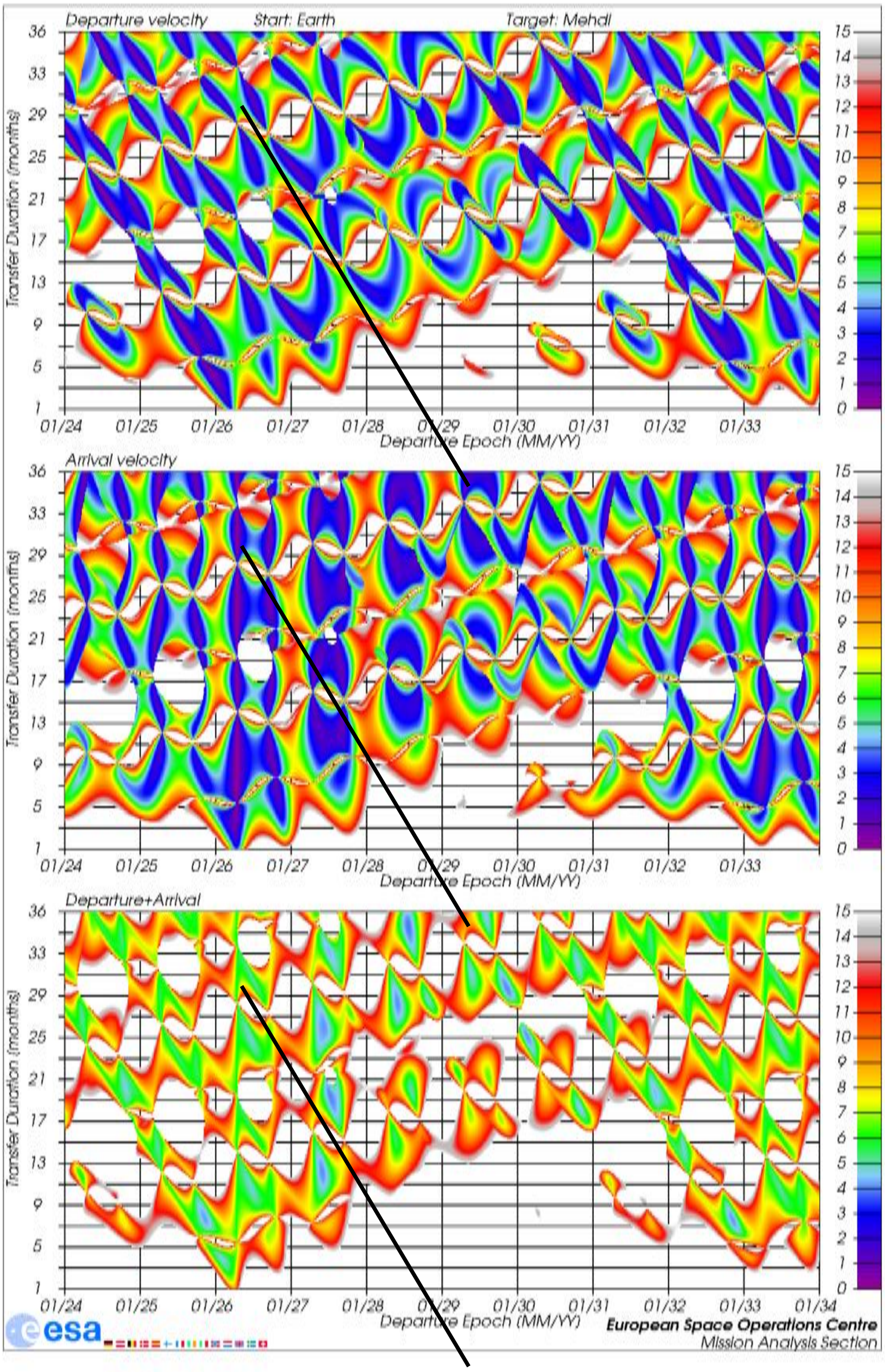

Figure 3-8: Porkchop plot for CP transfers to Apophis with departure between 2024 and 2034 – Valid after 04/29

In the context of this CDF study, analysing transfers to Apophis after the close encounter is tantamount to looking at a completely different target as its orbit is completely different. However, the impact of these changes on the transfer is mentioned herein below and a transfer reaching Apophis on its new orbit has also been computed.

The evolutions of distances and angles of Apophis with regards to the Sun and the Earth throughout the mission timeframe are shown in Figure 3-9. The main difference is then identified to be the range of distances to the Sun which will have an impact on the thermal and power subsystem.

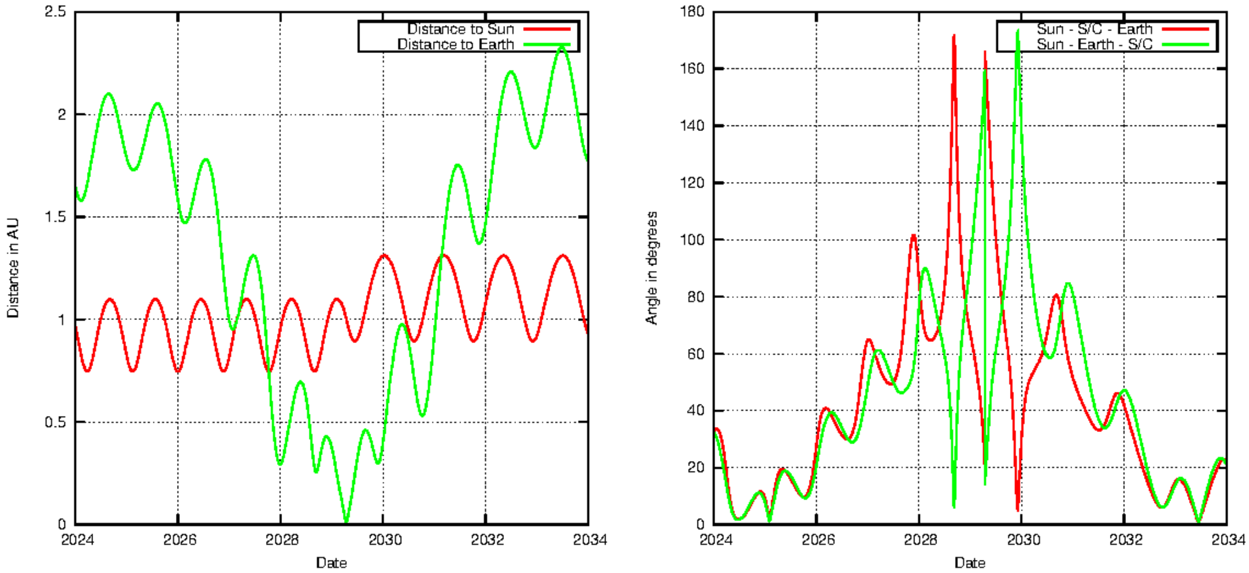

Figure 3-9: Evolution of Apophis distances and angles wrt to Sun and Earth from 2024 to 2034

3.3.3 Launch Scenario

The baseline launch scenario selected for this mission is a shared launch to SEL2. In this context, some Delta-V should be budgeted for transfer, possible station keeping and navigation.

In addition to the Delta-V required to reach the actual targets, some propellant has to be accounted for in order to reach SEL2. Moreover, since the mission is designed on a shared opportunity, the spacecraft will have to wait in SEL2 for a suitable low-thrust transfer window to the target. Approximate values for the propellant needed to reach and stay at SEL2 can be taken from another SEL2 mission.

3.3.3.1 Transfer to SEL2

The following results and values are based on Euclid[1] CReMA RD[23] and experience, in order to provide coherent values that can be used at this stage. No analyses were done for SPP as these are beyond the scope of the CDF study.

There will be no deterministic Delta-V for the transfer to SEL2, but stochastic components are foreseen to correct launcher dispersions and perigee velocity. These

[1] Euclid is an ESA mission to map the dark universe from a Quasi-Halo orbit about the Sun-Earth libration point 2 (SEL2) with departure foreseen in 2020.

corrections can be gathered inside the Transfer Correction Manoeuvre (TCM), a very critical manoeuvre of up to 45 m/s if manoeuvre is achieved on day 2 into the mission.

In case the TCM is delayed, an amplification factor has to be applied. This factor increases with execution delay due to the dynamics on a parabolic escape trajectory and is presented in Figure 3-10. The blue curve shows a nominal evolution while the green curve shows an evolution when influenced by the moon. The magenta vertical bar indicated the nominal correction 24 hours into the mission, the following two black bars represent day 2 and day 5, respectively. As an example, at day 2, the factor of 8.5 leads to a 45 m/s Delta-V.

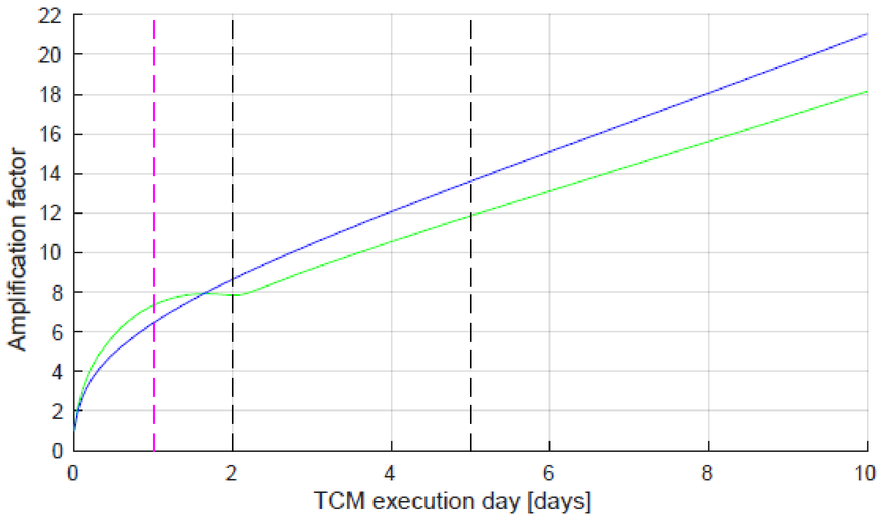

Figure 3-10: Correction Delta-V amplication factor as function of manoeuvre time, [1]

In the case of a mission with EP, there are several unknowns that would need to be determined at further stages of the mission development, namely:

- What is the delay before the TCM can be done in the case of EP?
- The Delta-V will be higher with EP, but by how much?

3.3.3.2 Station keeping at SEL2

The following results and values are based on Euclid CReMA RD[23] and experience, in order to provide coherent values that can be used at this stage. No analyses were done for SPP as these are beyond the scope of the CDF study.

An order of magnitude for the amount of propellant needed to maintain the S/C at SEL2 while waiting for the proper EP departure window can be retrieved, based on the following assumptions:

- Spherical thrust and attitude controlled S/C
- Particularly low non-gravitational accelerations on the S/C.

For Euclid, the maximum yearly station-keeping Delta-V can be as high as 7 m/s, depending on the frequency of the manoeuvres. In the context of the SPP mission, a realistic estimation is 3.5 m/s assuming the following:

- No bias

- Only bare minimum attitude control performed.

3.3.4 Transfers to Apophis

The reference case studied is a transfer to Apophis in around two years. The porkchop plots in Figure 3-7 and Figure 3-8 show multiple potential consecutive launch windows.

Initially, a departure in 2026 was analysed but later on, in order to be less constraining in the case of a shared launch with a certain primary payload, a departure in 2027 was investigated. Transfers in 2028 are also possible as shown a bit further down.

Arriving slightly before or slightly after the close encounter will have effect on the analysis, the operations and the propellant budget, as already mentioned above.

The transfers to Apophis were optimised considering the variable thrust and Isp model for the T6 and/or PPS-1350-E engine provided by the EP expert. The model for the T6 used assumes 1 kW is available for the EP system as input to the PPU at a Sun distance of 2.5 AU, which allows operating one T6 thruster at reduced power. The model for the PPS used assumes 1.5 kW is available for the EP system as input to the PPU at a Sun distance of 1.1 AU. The power at PPU input is scaled with $1/R_S^2$, where R_s is the distance to the Sun, and the thrust and Isp are obtained from polynomials fitting the T6 and PPS performance.

The contact time with the ground stations during transfer and close proximity operations has not been computed during the study and should be assessed individually for each transfer and each target.

3.3.4.1 Transfer in 2026

Transfers to Apophis starting in 2026 with the same T6 engine designed for option 2 were computed, with a duration of either 1.5 or 2 years. Results are compared in Table 3-8.

Option	T6	
Power [kW]	1.0 @ 2.5 AU	1.0 @ 2.5 AU
Departure date	23/07/2026	27/05/2026
Initial mass [kg]	900	900
Delta-V [km/s]	4.15	6.23
Average Isp [s]	4001	4005
Propellant mass [kg]	90	132
Duration [days]	715	502
Arrival date	07/07/2028	11/10/2027

Table 3-8: Transfer to Apophis in 2026, comparison of options

Such a T6 engine for a target closer to the Sun such as Apophis is clearly oversized. On the other hand, it might allow the S/C to reach a higher number of targets.

Complete results for the 2-year transfer are shown herein below. Projection of the trajectory in the ecliptic frame is presented in Figure 3-11, the thrust level in Figure 3-13 and the distances and angles with regards to the Sun and the Earth, in Figure 3-12, Figure 3-14 and Figure 3-15.

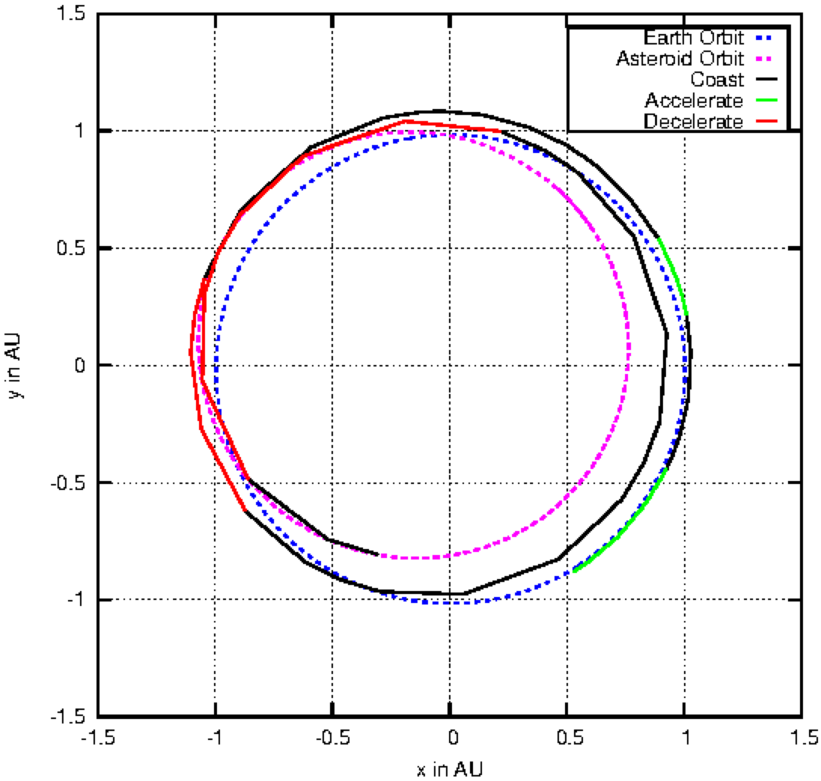

Figure 3-11: Projection of the transfer to Apophis starting in 2026 in the ecliptic plane

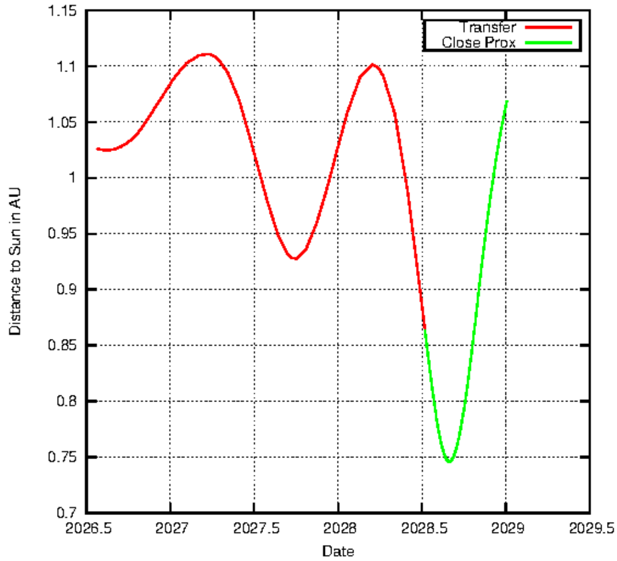

Figure 3-12: Transfer to Apophis 2026 – Distance to Sun

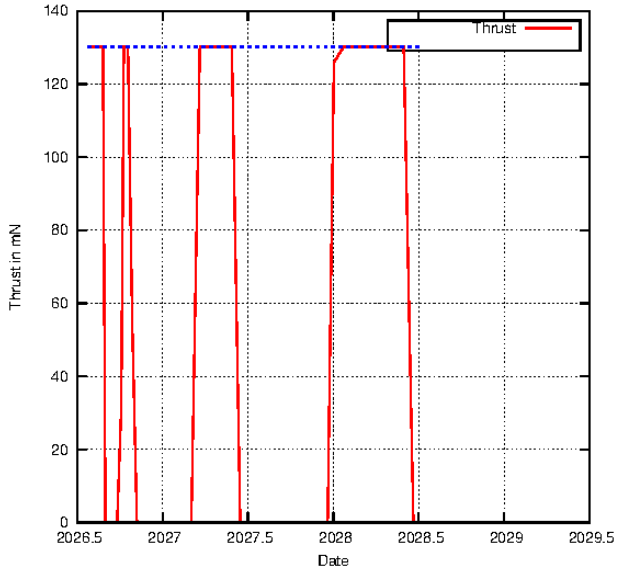

Figure 3-13: Transfer to Apophis 2026 – Thrust level

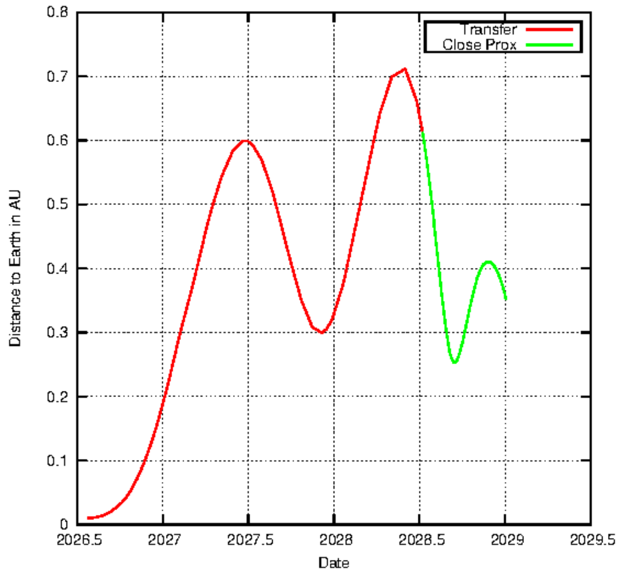

Figure 3-14: Transfer to Apophis 2026 – Distance to Earth

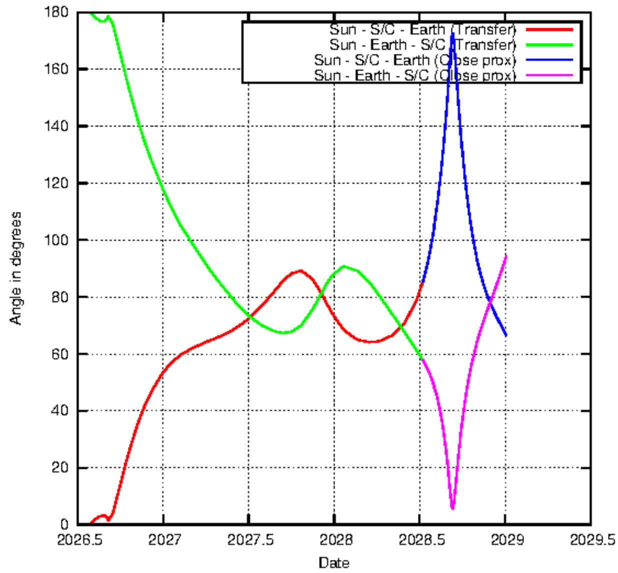

Figure 3-15: Transfer to Apophis 2026 – Angles wrt Sun and Earth

3.3.4.2 Transfer in 2027 (Reference Case)

A departure in 2027 would be interesting in order to envisage a shared launch with the Ariel mission (RD[28]). The current mission design of Ariel is based on 2 possible launch windows in 2026, one around April and the second around October

Transfers to Apophis starting in 2027 with the same T6 engine designed for option 2 but also 2 versions of PPS-1350-E were computed, with a duration of 2 years. Results are compared in Table 3-9. All these transfers arrive "at target" after April 2029, date of the close encounter of Apophis with Earth, which means that Apophis is on another orbit and the rendezvous is jeopardised.

Option	T6	PPS	
Power [kW]	1.0 @ 2.5 AU	1.5 constant	1.5 @ 1.1 AU
Departure date	29/04/2027	23/05/2027	22/05/2027
Initial mass [kg]	900	900	900
Delta-V [km/s]	4.00	4.70	4.53
Average Isp [s]	4000	1640	1792
Total impulse [MNs]	3.42	3.67	3.59
Propellant mass [kg]	87	228	204
Duration [days]	738	729	730
Arrival date	06/05/2029	20/05/2029	21/05/2029

Table 3-9: Transfer to Apophis in 2027, comparison of options

Complete results for the 2 years transfer with the PPS-1350-E engine are shown herein below. Projection of the trajectory in the ecliptic frame is presented in Figure 3-16, the thrust level in Figure 3-18 and the distances and angles with regards to the Sun and the Earth, in Figure 3-17, Figure 3-19 and Figure 3-20.

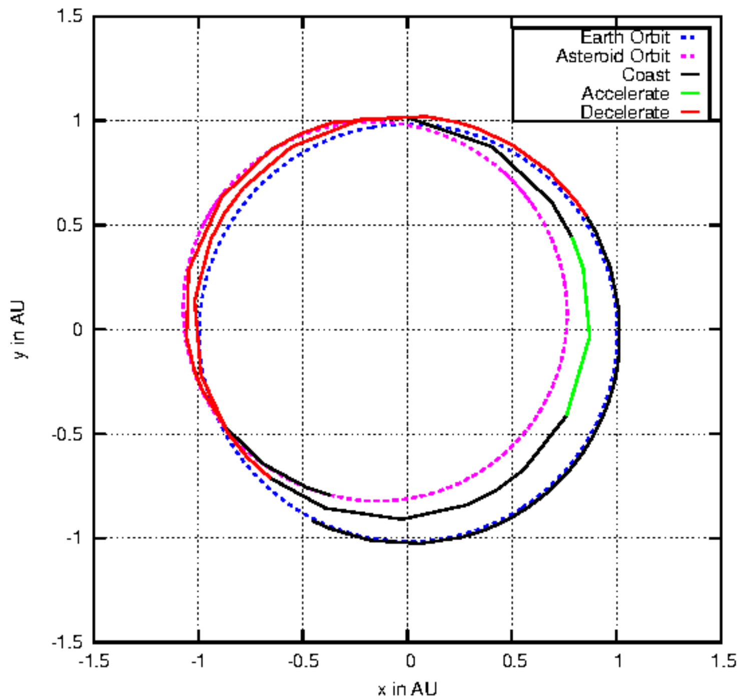

Figure 3-16: Projection of the transfer to Apophis starting in 2027 in the ecliptic plane

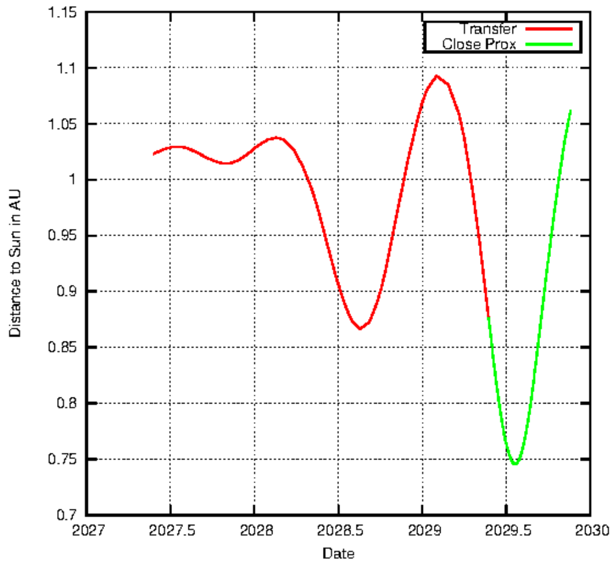

Figure 3-17: Transfer to Apophis 2027 – Distance to Sun

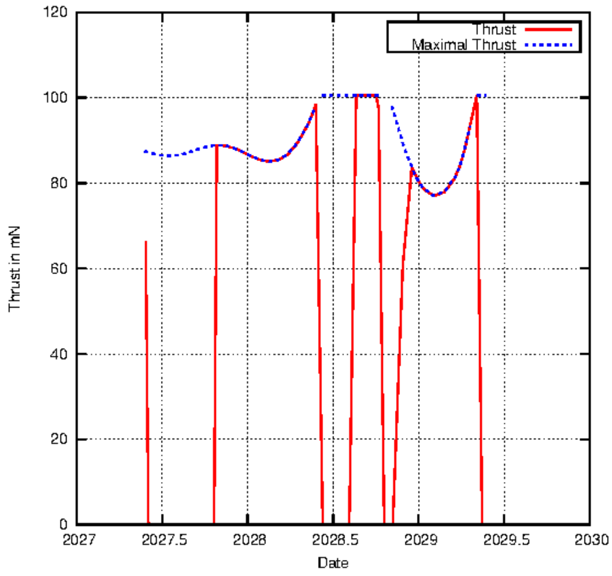

Figure 3-18: Transfer to Apophis 2027 – Thrust level

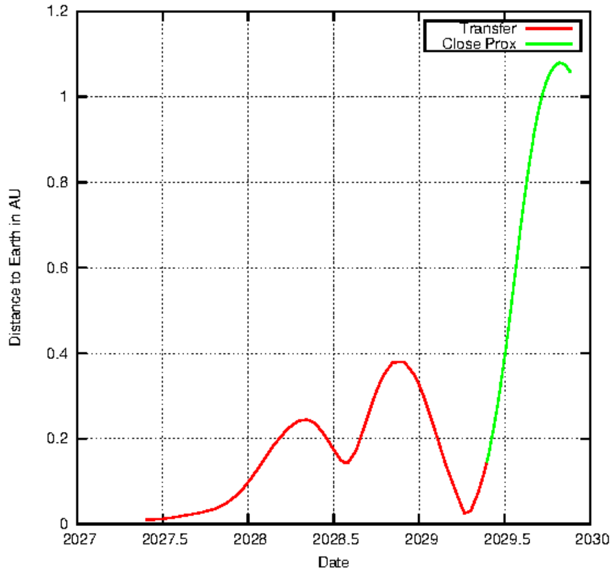

Figure 3-19: Transfer to Apophis 2027 – Distance to Earth

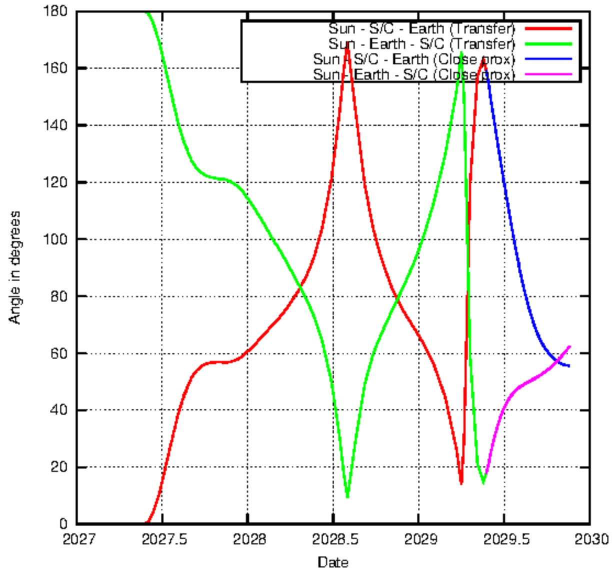

Figure 3-20: Transfer to Apophis 2027 – Angles wrt Sun and Earth

Since this transfer is arriving after the close encounter, it is a theoretical transfer.

3.3.4.3 Transfer in 2028

A departure in 2028 would lead to an arrival at the orbit of Apophis after the close encounter, i.e. to a completely different orbit than before. Such a transfer was also analysed briefly, mainly to study the impact it has on the baseline design.

Results are compared in Table 3-10.

Option	PPS
Power [kW]	1.5 @ 1.1 AU
Departure date	2028/06/30
Initial mass [kg]	900
Delta-V [km/s]	3.52
Average Isp [s]	1716
Total impulse [MNs]	2.86
Propellant mass [kg]	170
Duration [days]	877
Arrival date	2030/11/25

Table 3-10: Transfer to Apophis in 2028, comparison of options

Complete results for the 2-year transfer with the PPS-1350-E engine are shown herein below. For this transfer, the gravity of the Earth was not included because mid-way through the transfer, a flyby of the Earth is foreseen. That will most probably have a beneficial impact on the Delta-V required,but this needs to be assessed in detail and is out of the scope of this CDF study. It is safe to assume that current CDF baseline design would largely cope with the modified transfer requirements.

Projection of the trajectory in the ecliptic frame is presented in Figure 3-21, the thrust level in Figure 3-23 and the distances and angles with regards to the Sun and the Earth, in Figure 3-22, Figure 3-24 and Figure 3-25.

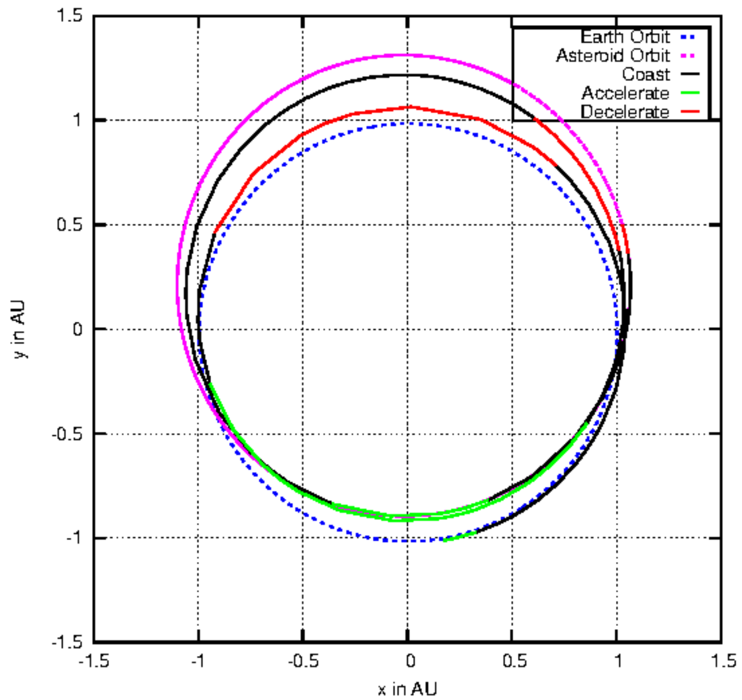

Figure 3-21: Projection of the transfer to Apophis starting in 2028 in the ecliptic plane

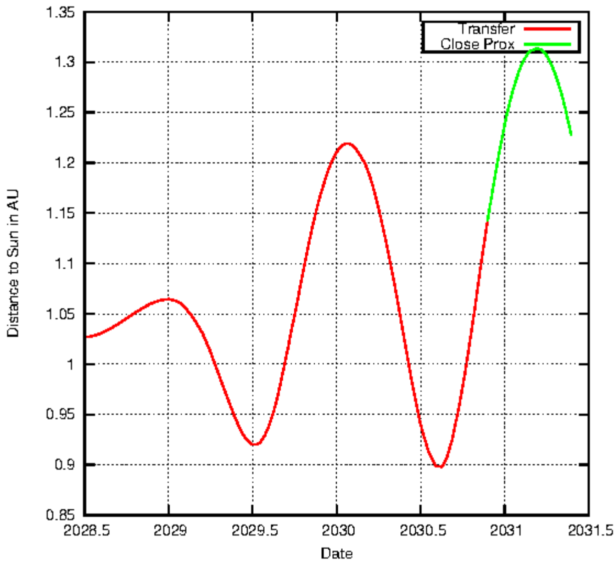

Figure 3-22: Transfer to Apophis 2028 – Distance to Sun

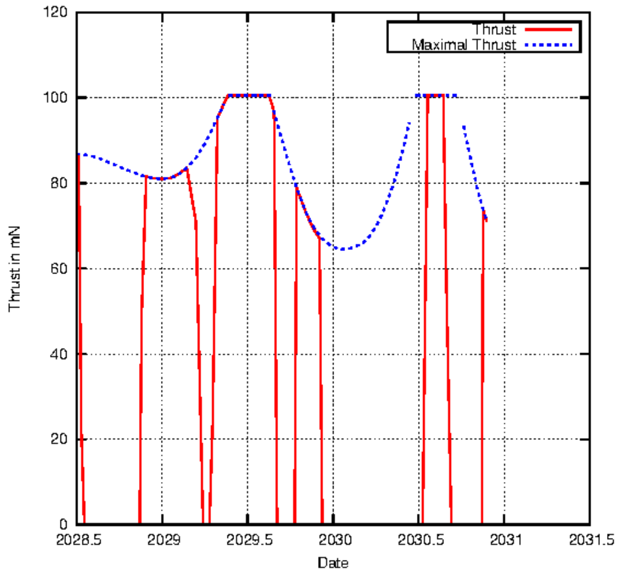

Figure 3-23: Transfer to Apophis 2028 – Thrust level

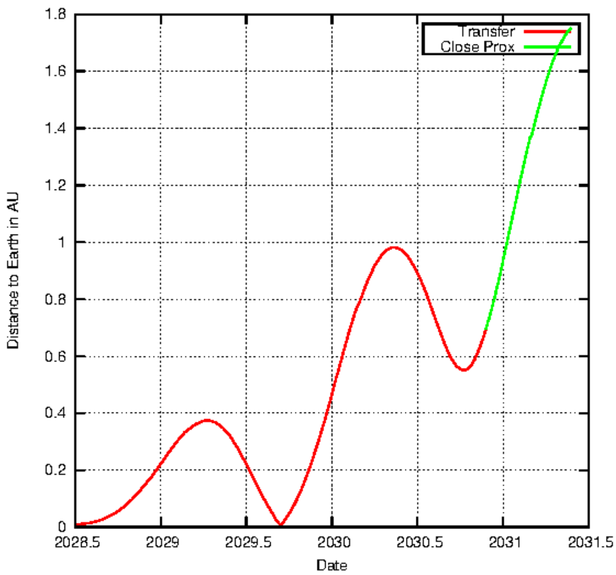

Figure 3-24: Transfer to Apophis 2028 – Distance to Earth

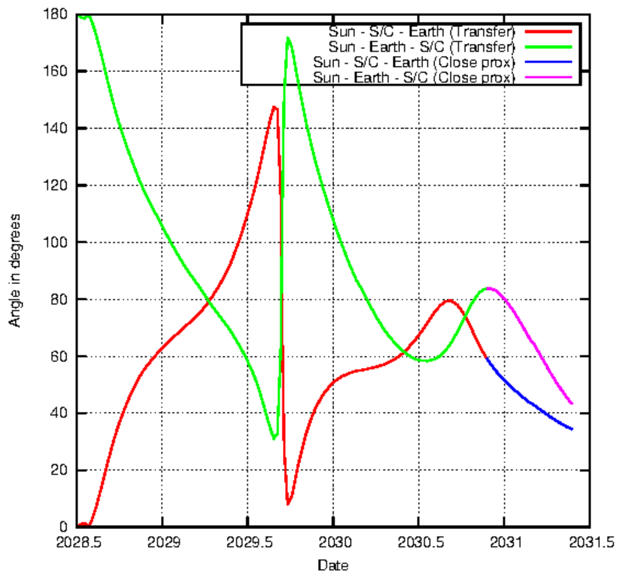

Figure 3-25: Transfer to Apophis 2028 – Angles wrt Sun and Earth

3.3.4.4 Margins for navigation

Since it is out of the scope of this study to perform a full navigation analysis, the orders of magnitude for navigation Delta-V margins and thruster accuracy are discussed herein below, mostly based on the experience of another EP mission.

The following results and values are based on BepiColombo experience, in order to provide educated guesses and orders of magnitude. No analyses were done for SPP as these are beyond the scope of the CDF study.

The assumed margin value on the Delta-V for BepiColombo for navigation correction purposes is 10%, assuming that:

- There is one ground station pass every week for interplanetary arcs (coast and thrust) with a typical pass duration of less than 8h. Range data are sampled once every 60 minutes and Doppler data at a rate of 1 measurement every 10 minutes. Moreover, a delay of 14 days between the measurements processing and control law upload (conservative assumption)
- The absolute pointing error of the thrust vector during thrust arcs shall be lower than 1.5° half-cone angle.

3.4 Sensitivity Analysis for MC: What if?

3.4.1 Launch into LEO + EP Assisted Escape

As part of the sensitivity analysis, the EPSILON launch was further investigated. Looking only into this smaller launcher is a worst case because Vega(-C) have higher performances. From the launcher performance data provided in Figure 3-26, the initial injection orbit was retrieved.

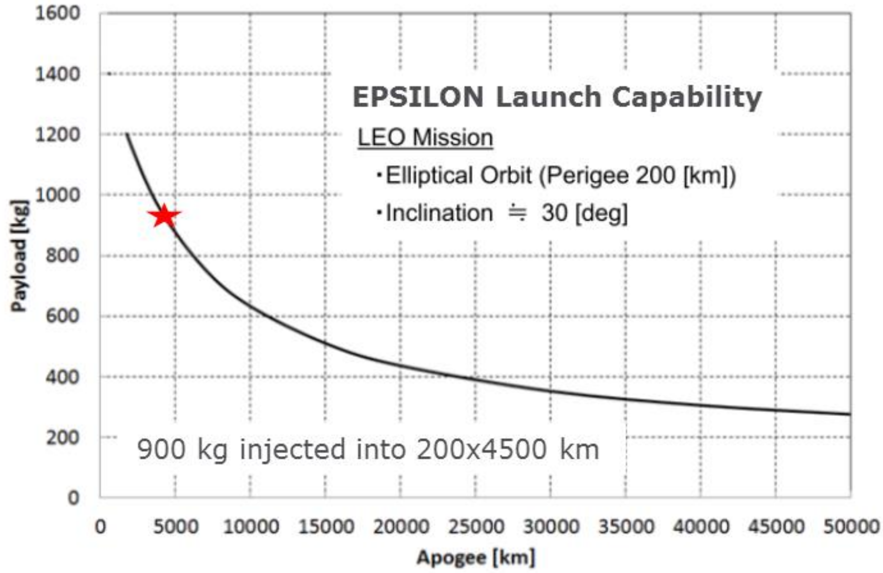

Figure 3-26: Epsilon launch capabilities for LEO missions

The spiralling out from LEO has been computed with the 2 different engine options and the results are summarised in Table 3-11. The total duration is 15 months to ~2 years depending on the EP option and the main concern is on the large radiation doses accumulated in the Van Allen belts[2].

[2] Assumed frontier of Inner Van Allen at 13000-14000 km altitude

Engine	T6	PPS-1350
Thrust [mN]	145	84
Isp [s]	4048	1640
Delta-V [m/s]	6306	6314
Propellant [kg]	132	292
Time < 20000 km [days]	**207**	**339**
Time to escape [days]	**467**	**719**

Table 3-11: Spiral out from LEO with 2 different engines

3.4.2 Other NEOs Targets

A sensitivity analysis is required in order to assess what other NEOs could be reached with the current design. There is no simple way to compute a high numbers of EP transfers even without optimising. Therefore, another approach has to be imagined.

There is currently more than 17,000 objects in the NEOs database from the MPC website, daily updated by the IAU. A first filter can be applied to this database, based on orbital elements:

- $0.74\ AU < a,\ r_a < 2.6\ AU$
- $0.74\ AU < r_p < 1.3\ AU$
- $i < 5°$

The lower bound for a, r_a and r_p as well as the higher bound for a and r_a have been chosen in order to include the two potentially reachable targets that were mentioned in Section 3.3.1. The higher bound for r_p is simply the definition of a NEOs. The limit on the inclination can be seen as an arbitrary limit but looking at Figure 3-27, one can see that the minimum Delta-V[3] that could be needed rise quickly with the inclination.

[3] Not really the minimum value since this represents the value to change inclination if the change was done at the orbit of Earth. But it would be cheaper to do it at higher SMA.

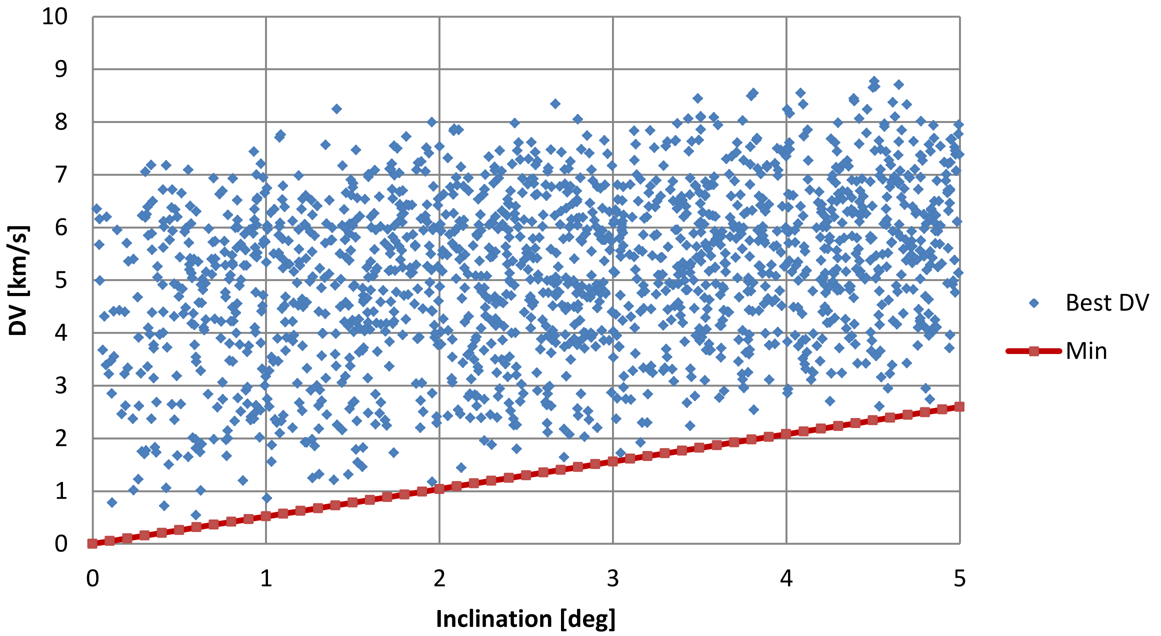

Figure 3-27: Distribution of best Delta-V values depending on inclination of target

This filter yields a total of 1633 NEOs[4] for which was then computed the best CP Delta-V value in the 2024-2034 time window, using the following assumptions:

- 2 CP manoeuvres: departure and arrival
- Departure date: from 2024 to 2034
- Transfer duration: from 1 to 3 years.

The result is a list of best Delta-V values that can be classified in a histogram as shown in Figure 3-28. In Table 3-12 are listed the number of targets that have a best CP Delta-V below respectively 3 to 7 km/s.

[4] Up to 3118 NEOs if inclinations up to 10 degrees are allowed .

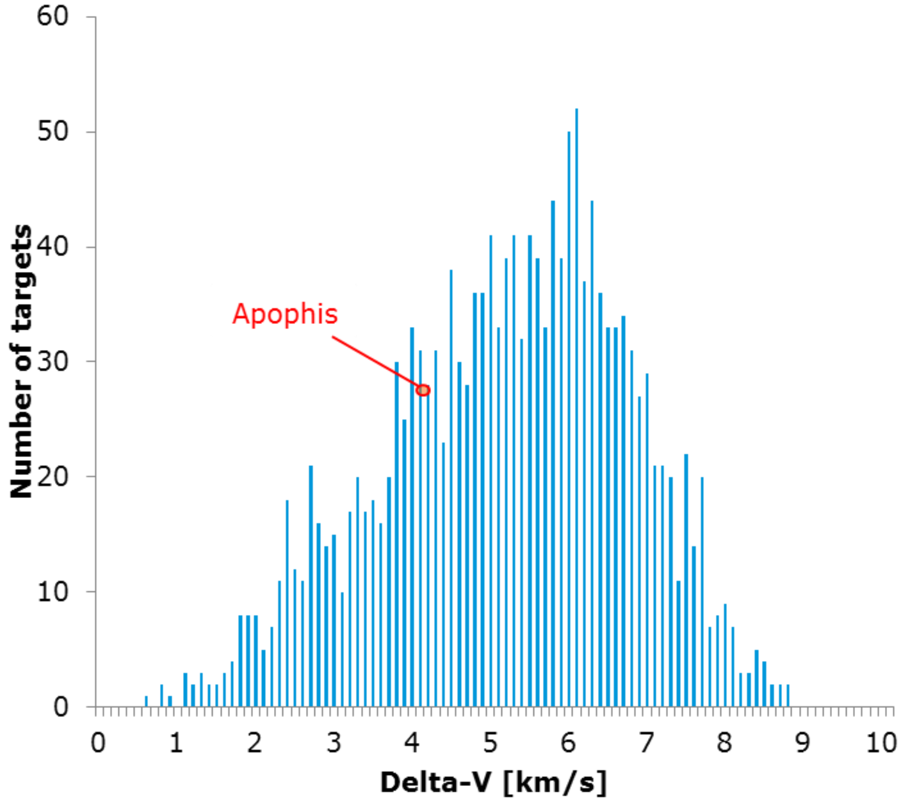

Figure 3-28: Histogram of best Delta-V values to reach different asteroids between 2024 and 2034

Best CP ΔV [km/s]	<3	<4	<5	<6	<7
# Asteroids	162	350	664	1007	1423

Table 3-12: Numbers of asteroids theoretically reachable with less than 3 to 7 km/s

However, it has to be noted that for EP transfers, Delta-V values are typically slightly higher but allow for more flexibility. This is mainly due to gravity losses at departure from SEL2 but also to the inherent idea of low thrust. With CP, the manoeuvre can be placed at the optimal orbit point but with EP, the Delta-V is spread out around this point, thrusting most of time outside of the optimal point. This is even more true when the thrust-to-mass ratio is low and the EP solution tends to apply the thrust continuously over the entire transfer.

On the other hand, EP transfers sometimes enable some economy by thrusting when it is the most efficient; at the nodes, for instances. Moreover, with EP, some transfers might not be feasible (depending on engine and thrust duration), because the duration of transfer might be too short to fit in all the thrust arcs required to achieve the overall Delta-V.

Therefore the Ideal CP or best CP Delta-V mentioned in this assessment represent lower bound and are not necessarily “guaranteed”. A dedicated transfer trajectory optimisation has to be carried out for each target to find out the real Delta-V that is required.

3.4.3 Mission to Phobos

The assessment of implications for a mission with target at Phobos was performed during the study. Mission analysis looked into transfers to Mars and performed a high-level analysis of how close proximity operations at Phobos can be accomplished.

A reference transfer to Mars with departure from SEL2 in 2028 was optimised for the study. The results are shown in Table 3-13. The transfer is constrained so that the departure relative velocity wrt SEL2 and the arrival velocity wrt Mars are both zero. The EP system with 1xT6 engine was considered with the design point of 1 kW at PPU input at 2.5 AU from the Sun. Such a system might be oversized for a transfer to Mars for which Sun distances up to 1.67 AU are expected. Even with such EP assumptions the transfer to Mars is expected to take 1.9 years, and 2.75 years if the spiralling down to the Phobos orbit is included.

Option	T6
Power [kW]	1.0 @ 2.5 AU
Departure date	2028-09-17
Initial mass [kg]	900
Delta-V [km/s]	6.38
Average Isp [s]	3956
Propellant mass [kg]	136
Duration [days]	677
Arrival date	2030-07-26
Days after SEP < 5 deg	44

Table 3-13: Transfer to Mars in 2028 with 1xT6 engine operating at 1 kW @ 2.5 AU

Projection of the trajectory onto the ecliptic plane is presented in Figure 3-29. Distances and angles wrt the Sun and Earth and the thrust level are shown in Figure 3-30 to Figure 3-33. These figures show the transfer continued by the operations around Mars assuming that they take 18 months, 12 months for reaching Phobos and 6 months for the close proximity operations.

An operational aspect that has to be considered is that arrival at Mars will require very precise navigation accuracy. This is typically achieved by using DDOR (Delta-Differential One-Way Ranging) measurements during the Mars approach navigation campaign that will extend for about 1 month. This needs to be considered in the approach for operations. Once in orbit around Mars the navigation accuracy provided by the Doppler measurements will be sufficient.

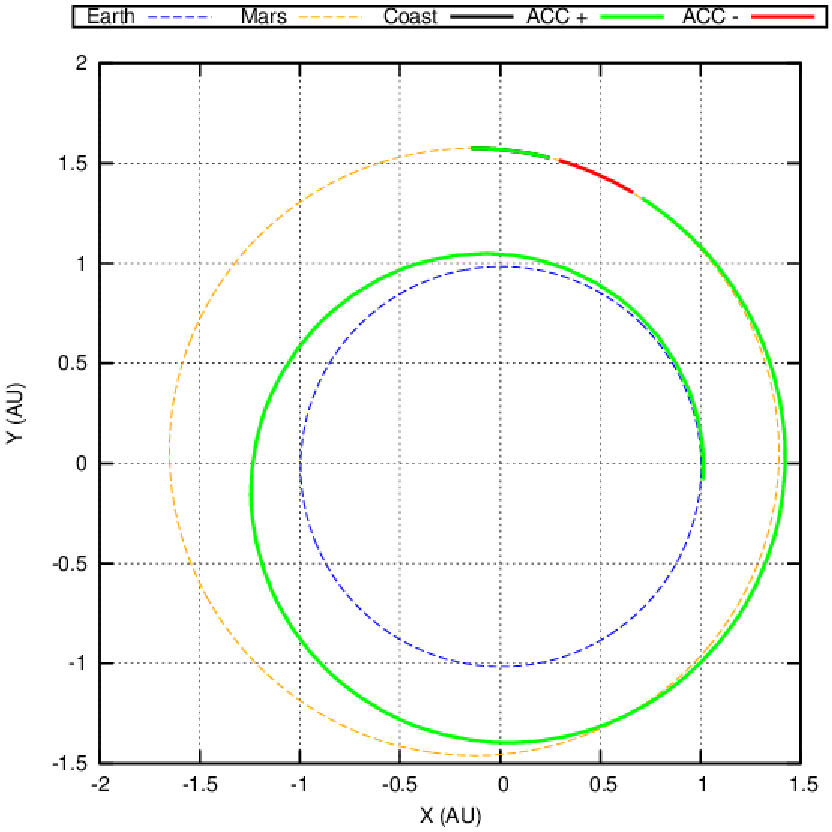

Figure 3-29: Ecliptic projection of the transfer to Mars starting in 2028

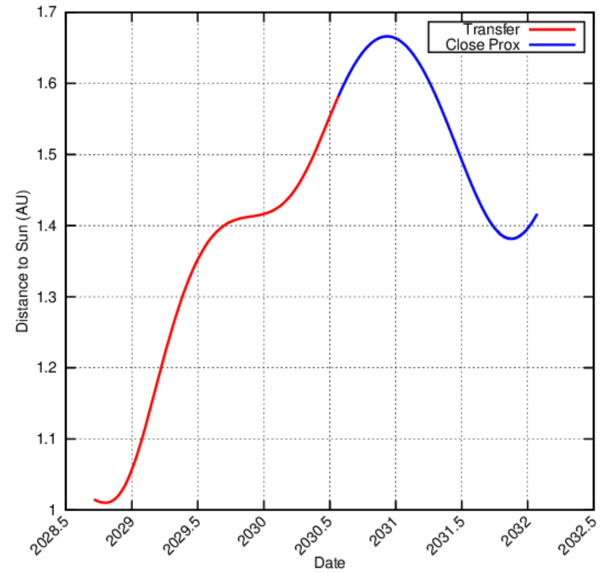

Figure 3-30: Transfer to Mars 2028 – Distance to Sun

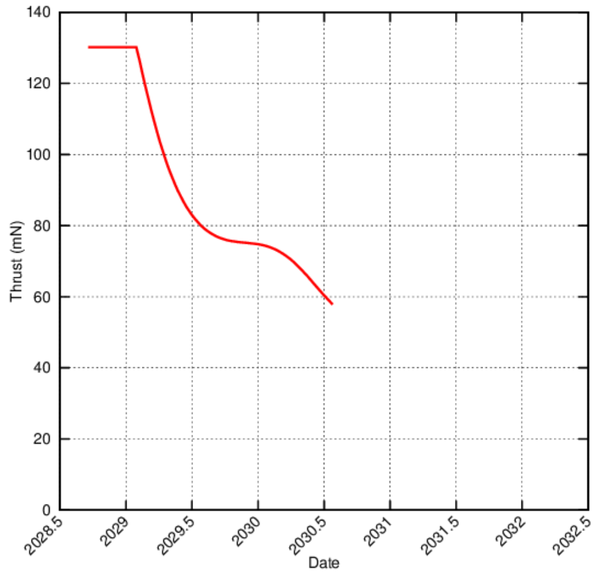

Figure 3-31: Transfer to Mars 2028 – Thrust level

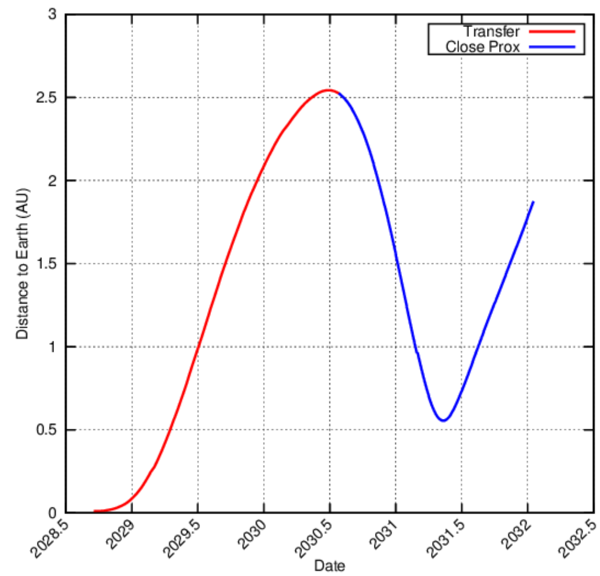

Figure 3-32: Transfer to Mars 2028 – Distance to Earth

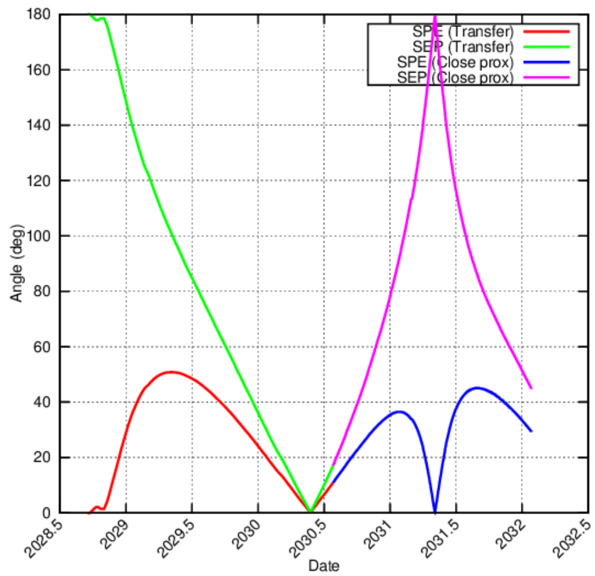

Figure 3-33: Transfer to Mars 2028 – Angles wrt Sun and Earth

After arrival to Mars the MC will keep using the EP engine to decelerate and spiral down to the orbit of Phobos (semi-major axis 9378 km). This mission phase has been computed assuming a continuous operation of the EP engine with thrust along the anti-velocity direction. Results of the spiralling are shown in Table 3-14. Arrival to the vicinity of Phobos occurs around mid-May 2031.

Option	T6
Power [kW]	1.0 @ 2.5 AU
Initial date	2028-09-17
Initial mass [kg]	764
Delta-V [km/s]	2.0
Average Isp [s]	3430
Propellant mass [kg]	43
Duration [days]	330
Arrival date	2031-05-19

Table 3-14: Spiral down close to Phobos orbit with 1xT6 engine operating at 1 kW @ 2.5 AU

Stable Keplerian orbits around Phobos are not feasible, as its sphere of influence lies below its surface. Quasi-satellite orbits (QSO) can be used for performing the close science observations of Phobos. QSO is a special orbit around Mars with semi-major axis close to that of Phobos, inclination close to the Laplace plane and mean eccentricity slightly larger to that Phobos's orbit. Thus the QSO remains in proximity of Phobos, although being subject to strong orbit perturbations due to the presence of Phobos, its orbital elements undergo periodic changes. Converted to the Phobos-Mars rotating frame a QSO appears to have an elliptic and retrograde trajectory, with significant variations of the orbit shape.

Extended investigations via numerical integration of QSO trajectories have been carried out in the frame of previous and current studies missions to Phobos (RD[32], RD[33], RD[34], RD[35]). QSO that remain stable for periods within 1-4 weeks without orbit corrections have been numerically obtained. However, 6-month operations close to Phobos are expected to require station keeping manoeuvres. The analysis of the station keeping was out of the scope of the current CDF study. Some investigations seem to indicate that station keeping manoeuvres of few cm/s per week are sufficient to control these orbits (RD[35]), but this result requires independent validation by ESA.

In terms of eclipses in the QSO, two kinds of eclipses are possible: Mars shadow passes and Phobos shadow passes. Eclipses due to Mars can be assumed equal to those experienced by Phobos, as a spacecraft in QSO remains close to Phobos. Such eclipses are unavoidable and occur in seasons that last 14 months reaching a maximum eclipse duration of 55 minutes. In between of the eclipse seasons, there is an eclipse-free period of 7 or 9 months, alternatingly.

Eclipses due to Phobos have to be studied through numerical simulation of the QSO. Long eclipses of up to 3 hours are known to be possible for certain geometry conditions. Mitigation measures to constraint the QSO and reduce this eclipse duration are possible, but have implications for the orbit inclination and increase the range between minimum and maximum Phobos distances, which may not be favourable for the science observations (RD[32]).

Much more extensive analysis of the close-proximity operations in the QSO will be needed if this mission option is further considered for next project phases.

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4 SYSTEMS

4.1 System Requirements and Design Drivers

4.1.1 Mission and System Requirements Tree

In order to have a better visualization of all the mission requirements a requirements tree was built, organising the entries at mission and subsystem level. The main requirements were identified for each branch. The requirements tree can be seen in Appendix A.

4.1.2 Mission Requirements Update

After the study was completed, it was possible to fill in the missing information and values from the initial study requirements list.

Mission Requirements		
Req. ID	**Statement**	**Parent ID**
MIS-010	The mission shall be able to perform multi-point and simultaneous science measurements around: small bodies (at a maximum distance of 1.1 AUs), or Phobos.	
MIS-020	The mission architecture shall consist of a mothercraft carrying a fleet of at least 4 smallsats.	
MIS-030 (goal)	The mission should be designed as a "multi-object tour" mission featuring flyby(s) of small bodies before reaching the selected final target(s).	
MIS-040 (goal)	The maximum flyby velocity should be limited to 2 km/s (TBC) to allow for meaningful science observations of the targets.	
MIS-050 (goal)	The mission should be designed to deploy at least one landed element on the surface on the final selected target if this is a small body (i.e. not on Mars or Venus).	
MIS-060	The mission shall be compatible with a single launch with the Epsilon and/or Vega-C launchers and a shared launch on Ariane 6.2.	
MIS-070	The mission shall be compatible with a launch date between 2024 and 2034.	
MIS-080	The mission shall be designed such that the encounter with the selected final target occurs when the distance between Earth and the body is equal or less than 1.1 AU.	
MIS-090 (goal)	The mission should be designed such that any flyby occurs at a maximum distance from Earth of 1.1 AU.	
MIS-100	The mission shall be designed such that the selected final target can be reached after a maximum of 5 years after launch.	
MIS-110	The mission lifetime shall be of 5.5 years (maximum) from	

	launch to end of life, including at least 6 months of science operations after deployment of the smallsats around the selected final target.	
MIS-120	The maximum distance between the mothercraft and the smallsats shall not exceed ~ 5 km.	
MIS-130	The mission shall have the following phases: • Launch • LEOP • Commissioning • Cruise • Flyby(s) Operations (goal) • Science Operations at the selected final target • Disposal	
MIS-140	The mission shall incorporate the following scientific payloads: • Low frequency radar • High frequency radar • Camera • IR spectrometer	
MIS-150	The mission shall be sized to support a science data volume return of 159 Gbits over 6 months.	
MIS-160	The total mission cost shall be below 150 MEuro.	

Table 4-1: Mission Requirements

Mothercraft System Requirements		
Req. ID	**Statement**	**Parent ID**
MC-010	The mothercraft shall be able to carry the smallsats to the selected final target.	
MC-020	The mothercraft shall be able to provide the data relay function to ground for the smallsats' TM/TC and science data.	
MC-030	The mothercraft shall be able to maintain the smallsats (and their scientific payload) within their operational and non-operational temperature range up to their deployment.	
MC-040	The mothercraft shall be able to provide the smallsats (and their scientific payload) with W average power up to their deployment.	
MC-050	The mothercraft shall have a data and power interface to the smallsats.	
MC-060	The mothercraft shall have the capability to do ranging to the smallsats using Inter Satellite Link (ISL).	
MC-070	The mothercraft shall provide a data relay function of the smallsats TM/TC and payloads to ground.	
MC-080	The mothercraft shall be capable of activating and commanding the smallsats before deployment including	

	payload activation, navigation sensors, software upload and health status monitoring.	
MC-090	The ISL shall be omni-directional and continuously available for mothercraft to smallsat communications.	
MC-100	The mothercraft shall be able to communicate simultaneously with all of the deployed smallsats.	
MC-110	The mothercraft shall be able to use the ISL to send clock corrections to the smallsats.	
MC-120	The propulsion system of the mothercraft shall be able to provide 4540 m/s delta-V.	
MC-130	The mothercraft shall have AOCS capabilities for reaching the final target, station keeping and release of the smallsats.	
MC-140	The mothercraft shall be able to communicate with ground using X-band.	
MC-150 (goal)	The mothercraft should be capable of performing science operations with the scientific instruments and the ISL and X-band communication packages operating simultaneously.	
MC-160	The mothercraft shall be designed to command the deployment of all the smallsats simultaneously and individually.	
MC-170	The mothercraft shall be capable of deploying the smallsats with a speed of 2-5 cm/s (TBC) ± 1 cm/s	
MC-180	The mothercraft shall ensure zero rates during smallsats deployment.	
MC-190	The mothercraft shall support reception of commands from ground control at a minimum data rate of 2kbps (TBC).	
MC-200	The mothercraft shall have on-board data storage for its own TM/TC and housekeeping data.	
MC-210	The mothercraft shall have on-board data storage for the smallsats' TM/TC and payload data.	
MC-220	The mothercraft's data handling system shall be sized to store all science data generated for 6 months.	
MC-230 (goal)	The mothercraft shall not accommodate scientific payload.	
MC-240	The mothercraft shall be designed according to the standard CDF margin philosophy: For equipment, the following mass margins shall be used: - 5% for off the shelf items - 10% for off the shelf items requiring minor modification - 20% for new developments or items requiring significant modification	
MC-250	A 20% system margin shall be accounted for in the	

	mothercraft's design	
MC-260	For calculation of the mothercraft's propellant mass, the following margins on the effective mission delta-V shall apply: • 3% for deterministic manoeuvres • 100% for attitude control manoeuvres • no additional margin on the delta-v specified for navigation manoeuvres	
MC-270	The nominal mothercraft's propellant mass shall be calculated based on its own dry mass including all margins, the wet mass of the smallsats and the delta-v including margin.	
MC-280	A 2% propellant margin shall be added on top of the nominal propellant mass to account for residuals.	
MC-290	The mothercraft design shall be compatible with a storage phase on ground of at least 3 years (TBC).	

Table 4-2: System Requirements

Smallsatellites System Requirements		
Req. ID	**Statement**	**Parent ID**
SS-010	Each of the smallsats shall be able to accommodate at least 3 kg of scientific payload.	
SS-020	After deployment from the mothercraft, during science operations, each of the smallsats shall be able to provide at least 117 W of average electrical power to the scientific payload.	
SS-030	Each of the smallsats shall provide a 5 V (TBC) power interface to the scientific payload.	
SS-040	All the smallsats shall have identical interfaces towards the mothercraft and towards the scientific payload.	
SS-050	After deployment from the mothercraft, each of the smallsats shall be able to maintain the scientific payload within their operational and non-operational temperature range.	
SS-060	The smallsats shall be capable of performing science operations with all the scientific instruments and the ISL communications package operating simultaneously.	
SS-070	The design of the smallsats shall guarantee an un-obstructed FoV for the scientific payload when operating.	
SS-080	The propulsion system of the smallsats shall be able to provide 10 m/s delta-V.	
SS-090	The smallsats shall have AOCS capabilities for station keeping after deployment from the mothercraft.	
SS-100	The smallsats shall be non-inertially pointing having the means to maintain a line of sight to point of interest.	
SS-110	Each of the smallsats shall provide an Absolute Pointing Error	

	(APE) of better than 0.5 deg (TBC).	
SS-120	Each of the smallsats shall provide a Relative Pointing Error (RPE) of 20 arcsec over 0.1 s (TBC).	
SS-130	Each of the smallsats shall provide an Absolute Knowledge Error (AKE) of TBD deg.	
	The smallsats shall be designed according to the standard CDF margin philosophy: For equipment, the following mass margins shall be used: - 5% for off the shelf items - 10% for off the shelf items requiring minor modification - 20% for new developments or items requiring significant modification	
SS-140	A 20% system margin shall be accounted for in the smallsats' design	
SS-150	For calculation of the smallsats' propellant mass, a 100% margin on the attitude control delta-v shall be taken into account.	
SS-160	The nominal smallsats' propellant mass shall be calculated taking into account the dry mass including margin and the delta-v margin	
SS-170	A 2% propellant margin shall be added on top of the nominal propellant mass to account for residuals.	
SS-180	The smallsats design shall be compatible with a storage phase on ground of at least 3 years (TBC).	

Table 4-3: Smallsatellites Requirements

4.1.3 Mission Design Drivers

The CDF study identified the following aspects as the main drivers for the SPP and each subsystem:

- Multi-target assessment: the design should be adaptable to different targets and provide a "tool-box" platform solution
- Architecture: design single platform (maximising resources e.g., power)
- Lifetime: the transfer time to target can be up to 5 years (≤ 3 years as goal) and the operations time is of 6 months
- Reliability/Procurement strategy: the SS equipment needs to be adapted to science standards
- Systems architecture: assessment of the possibility of using a kick stage and launcher accommodation
- Environment: extra shielding needed for configurations launched into LEO and orbit raising with EP, the SS need to be shielded during the transfer and the equipment compatible with the environment at target
- Delta-V: total impulse sizing on propulsion trade-off

- Mass: minimise overall mass (scientific payload mass < 3 kg per SS)
- Volume: accommodation of four SS on the mother to be deployed at target for distributed measurements
- Communications: the MC is used as a relay satellite for SS science, the ISL communications package is operating simultaneously and there is a high scientific data volume to download
- Power: the generation of electric power at the target is critical (eclipses)
- Thermal: need to adapt to the variable environment with limited power at target and limited space for radiator accommodation is limited
- Navigation: line of sight navigation required at target
- Operations: the deployment sequence is critical, minimum maintenance manoeuvres and autonomy required, aiming for minimum complexity, dependence on target
- Cost: M-class mission with the goal to have a total cost inferior to 150 Meuro.

4.2 System Baseline Design

4.2.1 Target Selection and Strawman Payload

From the NEOs catalogue, four potentially scientifically interesting targets were considered. Apophis – a 370 m non-active asteroid within 0.75 AU to 1.1. AU - was selected for its low inclination and eccentricity that make it the easiest target to reach.

Body	Semi-Major Axis (AU)	Eccentricity	Inclination (°)	Period (years)	Comments
Apophis	**0.92**	**0.19**	3.34	0.89	Typical accessible NEOs
2001 WN5	1.71	0.47	1.92	2.24	Might be reachable too
1999 AN10	1.46	0.56	39.93	1.76	High inclination and eccentricity
Ganymede	2.66	0.53	26.69	4.34	High inclination and eccentricity

Table 4-4: NEA target selection

The main focus of the scientific payload is radar tomography. For this purpose, different payload configurations were considered for the four smallsats with the goal to design a single platform that fits all possibilities.

Configuration 1	Configuration 2	Configuration 3
Low frequency radar	High frequency radar	IR spectrometer
Camera		

Table 4-5: Scientific payload configurations

	Configuration 1	Configuration 2	Configuration 3	Strawman Payload
Mass	2.75 kg	2.65 kg	2.75 kg	2.75 kg
Power	55 W	88 W	10 W	88 W
Data Volume	59 Gbit	39 Gbit	3.43 Gbit	59 Gbit
Volume	1U + 2U	3 U	1 U	3 U

Table 4-6: Different payload configurations envelope values and resulting strawman payload

4.2.2 Transfer, Rendezvous and Operations Overview

After considering the different launch options, the shared Ariane 62 to L2 with full EP platform was selected as the baseline. The escape from L2 is with relatively low V∞ and thrusting with EP to reach Apophis. One of the drivers to select the shared launch, was the opportunity to allocate the launch with ARIEL (Atmospheric Remote-sensing Exoplanet Large-survey) - one of the three candidate missions selected by the European Space Agency (ESA) for its next medium-class science mission due for launch in 2026.

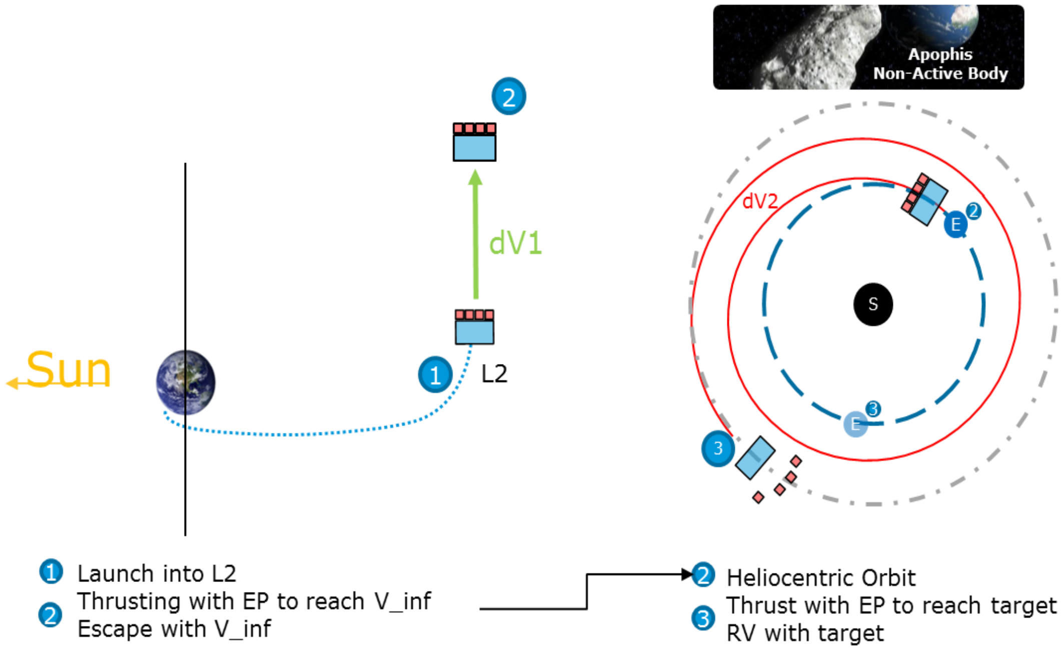


Figure 4-1: Overview of the transfer from L2

The different parameters of the transfer are summarised in Table 4-7. The selected EP system is the PPS1350-E engine with 1.5 kW at 1.1 AU and the wet mass assumed at departure from L2 is of 900 kg. Additionally, a 90% Duty cycle was applied to thrust (navigation, outages, contingencies…).

Power (kW)	Departure	Initial Mass (kg)	Delta-V (km/s)	Average Isp (s)	Total impulse (MNs)	Prop. (kg)	Duration (days)	Arrival
1.5 @ 1.1 AU	2027-05-22	900	4.53	1792	3.59	204	730	2029-05-21

Table 4-7: Summary of the different parameters of the Apophis baseline mission transfer

Once the target is reached, the MC will be inserted into a stable orbit in a plane between the Earth and the target. This will result from a slow stepped approach of four to six weeks. The SSs will be deployed individually in a sequence with the MC in the stable orbit. Each SS will be operated by the Principal Investigator responsible for the scientific payload of the SS.

The SSs will manoeuvre to the operational distance to the target. The MC will stay in a 'ping-pong' hyperbola of 7 day arcs (pericentre ~12 km, maximum distance ~20 km) maintaining visibility of the whole constellation and the SSs will be in 4-3-4-3 day hyperbolic arcs (3-day arc: pericentre ~5 km, max distance ~12 km; 4 day arcs: pericentre ~5 km, max distance ~16 km).

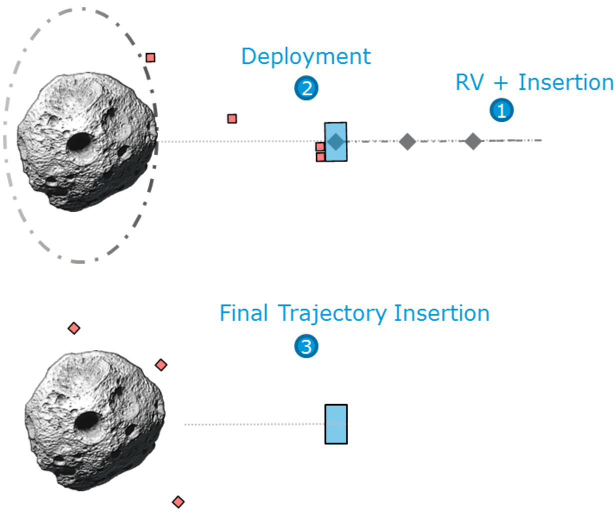


Figure 4-2: Target rendezvous, deployment of the smallsats and insertion in final trajectory

4.2.3 Baseline MC Design Summary

The Space Segment of SPP is a 2.0 x 2.0 x 2.2mother spacecraft carrier and a swarm of 4 0.26 x 0.23 x 0.45smallsats capable of delivering 10 m/s of low-thrust delta-V at target. It has power-optimised solar arrays, a 2 m X-Band high gain antenna for deep space communications and 2 S-Band LGA to support the ISL and it is easily adaptable to different payloads that enable it to carry out diverse missions for multi-point science observations. A summary of the system design is presented in Table 4-8 and Table 4-9.

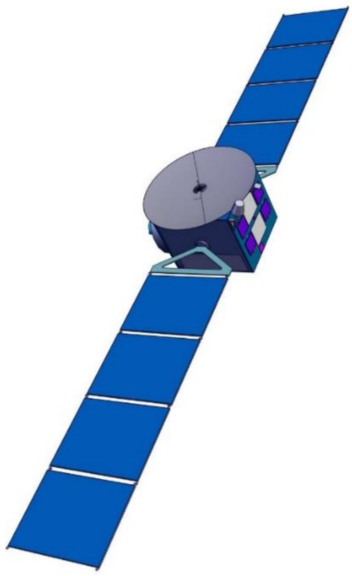
Mother Spacecraft		
Dimensions (m)	2.0 x 2.0 x 2.2	
Dry Mass incl. margin (kg)	554.48	
Wet Mass incl. margin (kg)	784.36	
Power available to Electric Propulsion System at 1.1 AU (kW)	1.9	
Thrust level at 1.1 AU (mN)	84	
Specific Impulse at 1.1 AU (s)	1640	
Delta-V (m/s)	4530 for the transfer (2 years) 10 at target + RW desaturation	
Payload	-	
AOGNC	Sensors: IMU | STR | SUN | NAV CAM	
	Actuators: RW | CG | Gimbal EP	
Communications	Earth link: X band 2m HGA – 8h of contact with Ground Station	
	ISL: 2 S-band LGAs	
Data handling	OBC: Rad-hard components	
Mechanisms	SADM | EP Gimbal | 4 Smallsats deployer	
Electric Propulsion	2 propellant tanks by Orbital ATK of each 135 kg Xe storage capability, 1 high pressure regulator	
	2 HET PPS thrusters (variable thrust and ISP), 1 thruster pointing mechanism, 2 Xenon flow controllers, 2 PPU, 2 EFU, 1 Pressure Regulation Electronic Card	
	1 Cold Thruster assembly	
Power	2 solar arrays with a total area of 8.3 m^2 with power generation optimised by SADM (MEC)	
	20 kg PCDU and 10.26 kg battery (ABSL manufacture)	
Structures	81kg	
Thermal	Radiators – 0.83 m^2	
	Kapton Multi Layered Insulation, loop heat pipes	

Table 4-8: MC Design Summary

4.2.4 Baseline SS Design Summary

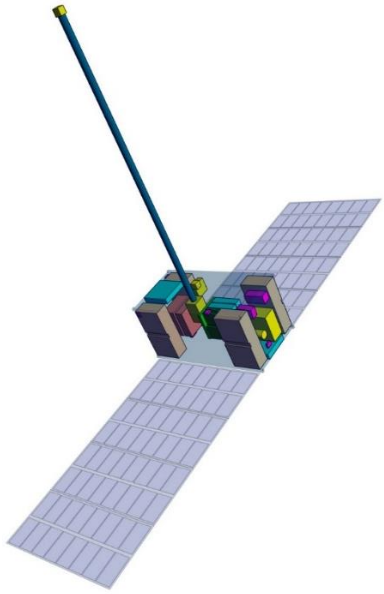
Smallsat (x4)		
Dimensions (m)	0.26 x 0.23 x 0.45	
Dry Mass incl. margin (kg)	28.87	
Wet Mass incl. margin (kg)	29.04	
Power generation at 1.1 AU (W)	117	
Delta-V (m/s)	10 at target	
Payload	Low frequency radar High frequency radar Camera IR spectrometer 159 Gbit expected data return	
AOGNC	Sensors: IMU | STR | SUN | NAV CAM	
	Actuators: RW | CG | Gimbal EP	
Communications	ISL: 2 S-band LGAs	
Data handling	OBC: Rad-tolerant components	
Mechanisms	SADM	
Chemical Propulsion	Butane Cold gas system ~520 g Cold gas system	
Power	2 solar arrays with a total area of 0.64 m^2 with power generation optimised by SADM (MEC)	
	0.86 kg battery	
Structures	16U SmallSat of the shelf Structure 2.25kg	
Thermal	Black MLI chosen to maximise absorption at the target	
	Radiators 0.33 m^2 – deployable radiators needed	

Table 4-9: SS Design Summary

4.2.5 Propellant Budget

For the calculation of the needed propellant during the whole mission, the margin policy RD[36] has been adapted accordingly:

MAR-DV-010	5% or 10 m/s delta-v margin (whichever is highest) shall be applied to deterministic, accurately calculated manoeuvres (trajectory manoeuvres as well as detailed orbit maintenance manoeuvres) documented in a MAG/CreMA. Note: 10 m/s are to be compared to 5% of the sum of all deterministic delta-v terms and added only once if higher.	

Table 4-10: Adoption of Delta-V margins

Under this consideration, the Delta-V Budget looks like Table 4-11.

Manoeuvres	Delta v (m/s)	Margin	Total (m/s)	Delta v (m/s)	Margin	Total (m/s)
System	Mother spacecraft			Smallsat		
Transfer	4119	10% (EP)	4530	1	100%	10
Orbit maintenance at target	2.4	-	10	6.7	5%	
Pointing & Attitude control	0.5	-		0.2	100%	
Total		-	4540		-	10

Table 4-11: Delta-V Budget for Mother Spacecraft and Smallsat

4.2.6 Product Tree

The OCDT model architecture was defined with the Space segment containing the Motherspacecraft (MC) and the 4 Smallsats (SS). No Elements were defined for the Ground and Launch segment in the model. The Product Tree is depicted in Figure 4-3.

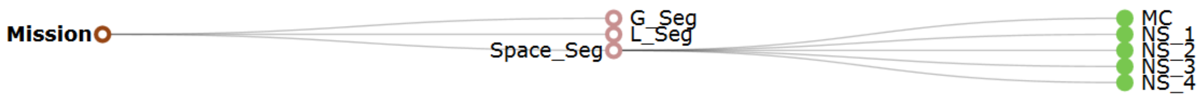


Figure 4-3: Mission Model description showing the Space-Segment containing the Motherspaceraft and the 4 Smallsats

The Product-Tree for the Motherspacecraft and the Smallsat can be seen in Figure 4-4 and Figure 4-5.

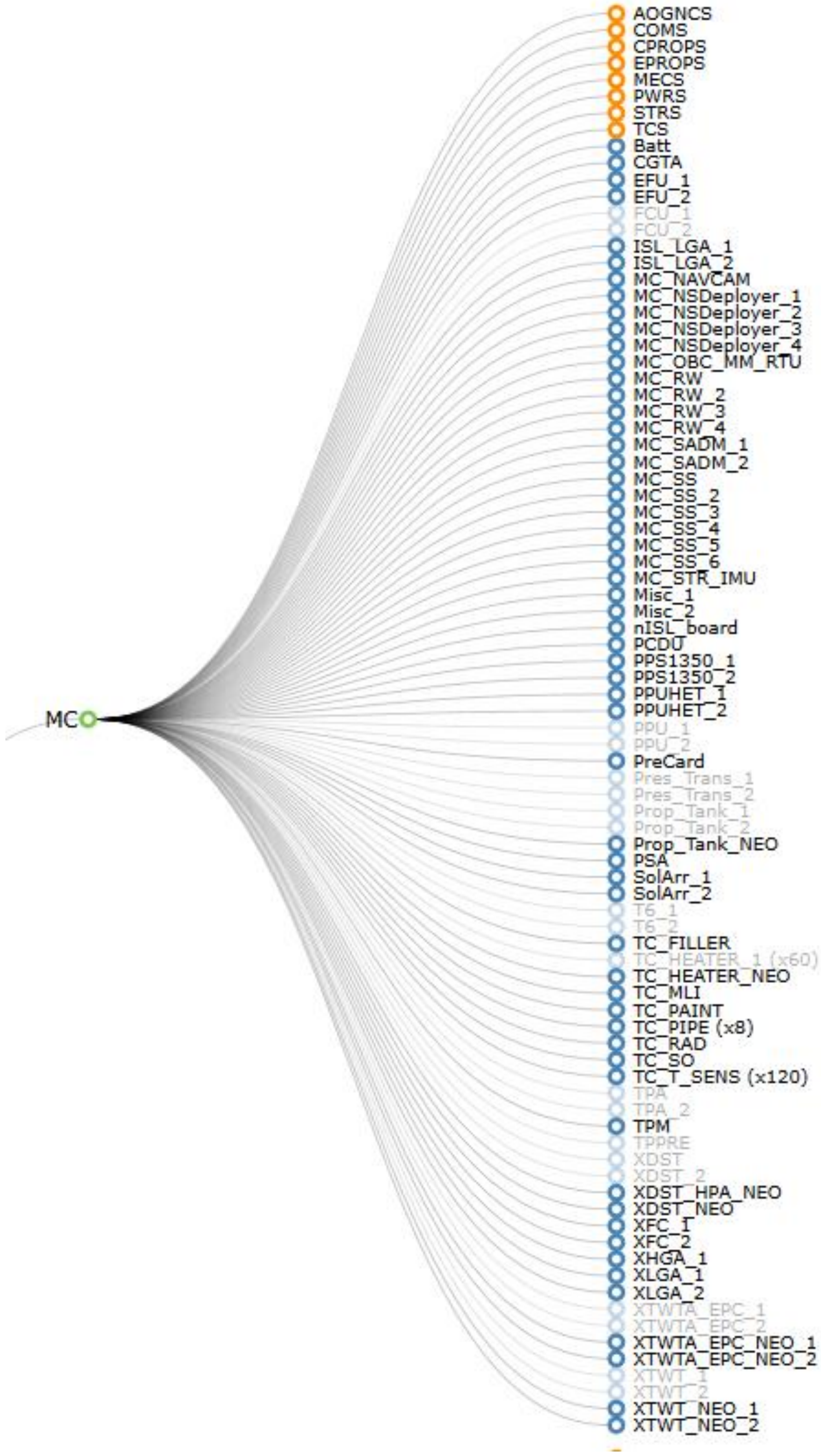


Figure 4-4: Model representation of the Motherspacecraft. Equipment for other options shown as greyed element usages

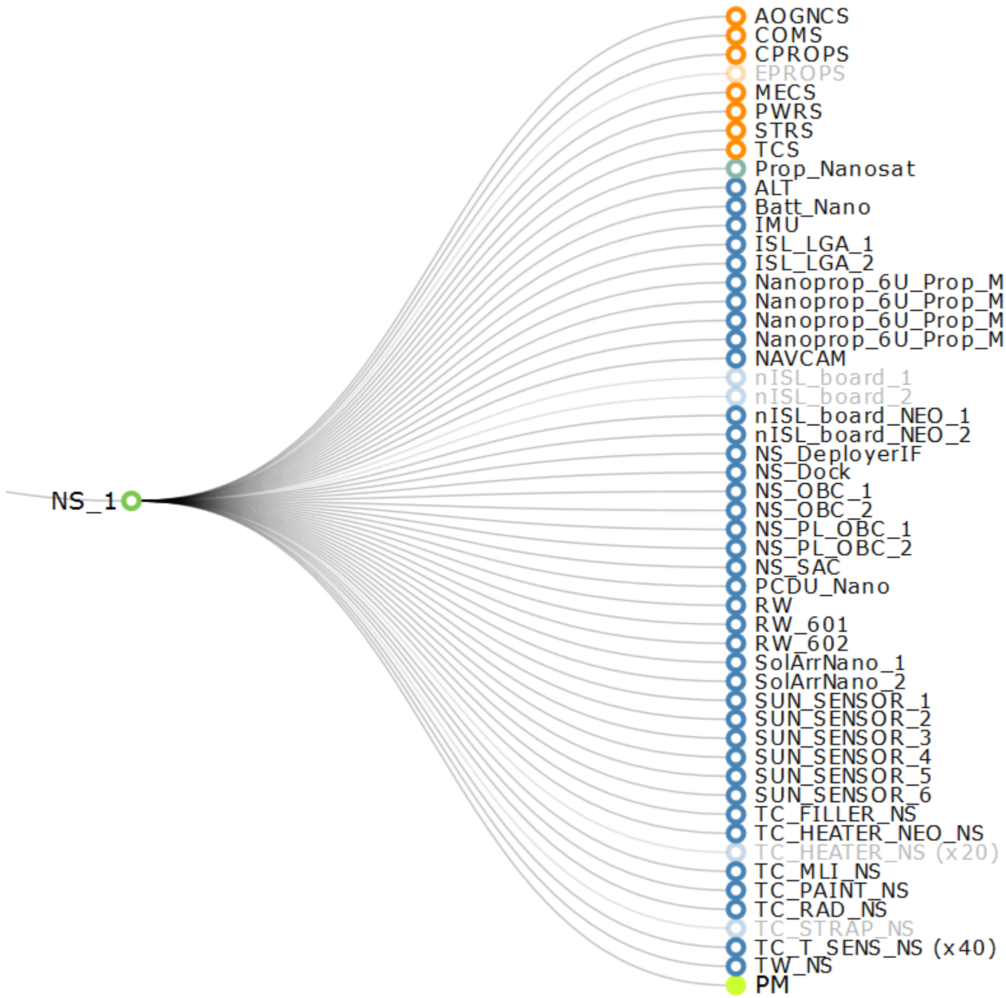


Figure 4-5 : Model representation of the Smallsat. Equipment for other options shown as greyed element usages

4.2.7 Modes and Phases

For the whole mission the following phases were considered:

Launch – Commissioning – Transfer –Rendezvous and deployment – Operations – Disposal

They are also presented in Figure 4-6.

Figure 4-6: Mission Phases

The following modes were considered for Power and Thermal analysis:

Motherspacecraft:

- Launch (Launch)
    - Mode during launch when the System gets disconnected from ground till the separation from the launcher
- Low Earth Operation (LEOP)
    - Operations and Commissioning in Low Earth Orbit
- Low Earth Safe (LESAFE)
    - Safe mode in Low Earth Orbit
- Electric Propulsion at Earth (EPROP_EARTH)
    - Mode for using the Electric Propulsion Subsystem in the proximity of Earth (~1AU)
- Communication (COMS)
    - Telecommunications with ground using the HGA
- Electric Propulsion at Target (EPROP_TARGET)
    - Mode for using the Electric Propulsion Subsystem in the proximity of the Target (~1.1AU)
- Relay Communication (RELAY)
    - Telecommunication with ground using the HGA and with the Smallsats using the ISL
- Stand-by/Safe (SAFE)

Smallsat:

- Sleeping/Hibernation (SLEEP)
    - Mode for hibernation during the transfer to the target
- Operational (OPS)
    - Operational mode at target using the Payload
- Stand-by/Safe (SAFE)
- Start-up/boot (BOOT)
    - Mode for commissioning the Smallsats at Target

4.2.8 Smallsat Mass Budget and Equipment List

The final Smallsat mass including 0.52kg of Propellant located in the 4 Smallprop 6U Modules and a system margin of 20% is 29.40kg. An overview of the Equipment and its masses is shown in Table 4-12 and a total mass budget of the whole Smallsat separated in subsystems in Table 4-13.

	Nr.	Mass per Unit (kg)	Mass margin (%)	Mass incl. margin per Unit (kg)	Total Mass incl. margin (kg)
AOGNC					**0.725**
Jenoptik DLEM Laser Rangefinder	1	0.034	5	0.036	0.036
Memsense IMU 3020	1	0.02	5	0.021	0.021
Hyperion Technologies IM200	1	0.059	5	0.062	0.062
GomSpace SmallTorque GSW-600	3	0.18	10	0.198	0.594
Hyperion Sun Sensor SS200	6	0.002	5	0.002	0.011
COM					**0.48**
smallISL LGA	2	0.05	20	0.06	0.12
smallISL Electronics	2	0.15	20	0.18	0.36
CPROP					**3.083**
Smallprop 6U PropModule	4	0.77	0.1	0.771	3.083
DH					**0.309**
Dock Board	1	0.0742	10	0.082	0.082
Platform OBC	2	0.04	0	0.04	0.08
Payload OBC	2	0.07	5	0.074	0.147
INS					**3.3**
StrawMan Payload	1	2.75	20	3.3	3.3
MEC					**1.02**
Deployer Interface	1	0.1	20	0.12	0.12
Solar Array Control Unit	1	0.75	20	0.9	0.9
PWR					**5.424**
Battery	1	1	20	1.2	1.2
Power Control and Distribution Unit	1	0.6	20	0.72	0.72
Solar Array	2	1.46	20	1.752	3.504
STR					**2.7**
Primary Structure	1	2.25	20	2.7	2.7
TC					**5.874**
Thermal Filler	1	0.0012	20	0.001	0.001
Heater	40	0.005	10	0.0055	0.22
Multi-Layer Insulation	1	0.405	20	0.486	0.486
Paint	1	0.162	20	0.194	0.194
Radiator Panel	1	3.96	20	4.752	4.752
Temperature Sensor	40	0.005	10	0.0055	0.22

Table 4-12: Equipment List of SmallSat

SmallSat Mass Budget	Mass [kg]
Attitude, Orbit, Guidance, Navigation Control	0.725
Communications	0.48
Chemical Propulsion	3.08
Data-Handling	0.31
Instruments	3.30

SmallSat Mass Budget		Mass [kg]
Mechanisms		1.02
Power		5.42
Structures		2.70
Thermal Control		5.87
Harness	5%	1.15
Dry Mass w/o System Margin		**24.06**
System Margin	20%	4.81
Dry Mass incl. System Margin		**28.87**
Propellant Mass		0.52
Propellant Residual	2%	0.01
Total Wet Mass		**29.40**

Table 4-13: Mass Budget of SmallSat

4.2.9 Mother Spacecraft Mass Budget and Equipment List

The final Mother spacecraft mass including Smallsats and 225.38kg of Propellant is 784.36kg. The dry Mother spacecraft mass including a system margin of 20% and without Smallsats is 436.86kg. The propellant mass is calculated with the assumption of a total wet mass of 900gk. An overview of the Equipment and its masses is shown in Table 4-14 and a total mass budget of the whole Smallsat divided in subsystems in Table 4-15.

Row Labels	Nr.	Mass per Unit (kg)	Mass margin (%)	Mass incl. margin per Unit (kg)	Total Mass incl. margin (kg)
AOGNC					**10.196**
TSD DVS Navigation Camera	1	2.4	5	2.52	2.52
MW1000 Reaction Wheel	4	1.44	5	1.512	6.048
Mini FFS Sun Sensor	6	0.05	5	0.053	0.315
DTU uASC Star Tracker and IMU	1	1.25	5	1.313	1.313
COM					**23.07**
smallISL LGA	2	0.05	20	0.06	0.12
smallISL Electronics	1	0.15	20	0.18	0.18
X-Band DST built-in HPA (Allocation)	1	0	0	0	0
X-Band DSTRASP	1	3.7	10	4.07	4.07
X-Band HGA	1	8	10	8.8	8.8
X-Band LGA	2	1	20	1.2	2.4
X-Band TWT	2	2	5	2.1	4.2
X-Band TWTA EPC	2	1.5	10	1.65	3.3
DH					**3.6**
OBC MM RTU	1	3	20	3.6	3.6
EPROP					**86.97**
Cold Gas Thruster Assembly	1	3.05	10	3.355	3.355
Electric Filter Unit	2	0.7	5	0.735	1.47

Miscellaneous Piping, Harness, Sensors	2	3	20	3.6	7.2
HET Thruster PPS1350-E	2	4.35	10	4.785	9.57
PPU for HET	2	11.8	5	12.39	24.78
Pressure Regulation Electronic Card	1	1.3	10	1.43	1.43
Propellant Tank	1	20.4	5	21.42	21.42
High pressure regulator – Propellant Supply Assembly	1	4.5	5	4.725	4.725
Thruster Pointing Mechanism	1	10.6	5	11.13	11.13
Xenon Flow Controller	2	0.9	5	0.945	1.896
MEC					**38.684**
Small Satellite Deployer	4	6.484	20	7.781	31.124
Solar Array Deployment Mechanism	2	3.6	5	3.78	7.5
PWR					**67.116**
Battery	1	9.78	20	11.736	11.736
Power Control and Distribution Unit	1	17	20	20.4	20.4
Solar array	2	15.9	10	17.49	34.98
STR					**81**
Primary Structure	1	67.5	20	81	81
TC					**36.73**
Thermal Filler	1	1.35	20	1.62	1.62
Heater	60	0.01	10	0.011	0.66
Multi-Layer Insulation	1	5.6	20	6.72	6.72
Paint	1	5.4	20	6.48	6.48
Heat Pipe	8	0.75	20	0.9	7.2
Radiator Panel	1	9.96	20	11.952	11.952
Stand Offs	1	1.2	20	1.44	1.44
Temperature Sensor	120	0.005	10	0.006	0.66

Table 4-14: Equipment List of Mother Spacecraft

MC Mass Budget		**Mass [kg]**
Attitude, Orbit, Guidance, Navigation Control		10.20
Communications		23.07
Data-Handling		3.60
Electric Propulsion		86.97
Mechanisms		38.68
Power		67.12
Structures		81.00
Thermal Control		36.73
Harness	5%	17.36
Dry Mass w/o System Margin		**364.72**
System Margin	20%	72.944
Wet Mass Small Sat	**4.00**	**117.61**
Dry Mass incl. System Margin		**555.27**

MC Mass Budget		Mass [kg]
Propellant Mass		225.38
Propellant Residual	2%	4.51
Wet Mass		**785.16**

Table 4-15: Mass Budget of Mother Spacecraft

4.2.10 Power Budget

In order to simplify the power budget development for the Mother spacecraft, the EPROPS equipment is mostly absent. Instead, all power made available to EPROPS is done via the PPPUHET (PPU for HET). Internally, the EPROPS adjusted the power as needed. For Power and Thermal analysis, the power level applied to the PPPUHET can vary depending on the distance to the Sun. Therefor the two sizing cases EPROP at Earth and EPROP at Target model the sizing cases. According to RD[36], there is no margin on the consuming mean power of the PPPUHET. The results are shown in Table 4-16.

	P_on	P_stby	#	LAUNCH	LEOP	LESAFE	EPROP EARTH	COMS	EPROP TARGET	RELAY	MS_SAFE
TSD DVS Navigation Camera	13	0	1	0.00	0.00	6.50	6.50	0.00	6.50	6.50	6.50
MW1000 Reaction Wheel	35	2	3	0.00	11.90	11.90	11.90	11.90	11.90	11.90	11.90
DTU uASC Star Tracker and IMU	5.2	5.2	1	0.00	5.20	5.20	5.20	5.20	5.20	5.20	5.20
smallISL Electronics	10	2.3	1	0.00	0.00	0.00	0.00	0.00	0.00	6.15	0.00
X-Band DST built-in HPA	20	0	1	10.00	10.00	10.00	0.00	10.00	0.00	10.00	10.00
X-Band DSTRASP	15	10	1	10.00	15.00	0.00	0.00	15.00	0.00	15.00	0.00
X-Band TWT	200	0	1	0.00	25.00	0.00	0.00	200.00	0.00	200.00	0.00
X-Band TWTA EPC	10	0	1	0.00	0.00	0.00	0.00	10.00	0.00	10.00	0.00
MC OBC MM RTU	6.3	0	1	6.30	6.30	6.30	6.30	6.30	6.30	6.30	6.30
Cold Gas Thruster Assembly	1	0	1	0.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00
PPU for HET	2000	20	1	0.00	0.00	0.00	1500.00	0.00	1500.00	0.00	0.00
Solar Array Deployment Mechanism	3	0	2	0.00	3.00	3.00	3.00	3.00	3.00	3.00	3.00
Power Control and Distribution Unit	10	10	1	10.00	10.00	10.00	10.00	10.00	10.00	10.00	10.00
Heaters	100	0	1	70.00	100.00	97.62	0.00	0.00	0.00	0.00	97.62
Total w/o Margins				**106**	**214**	**178**	**1571**	**299**	**1571**	**312**	**178**
Losses (PCDU + Harness)			**3%**	3	6	5	47	9	47	9	5
Total S/C				109	221	184	1618	308	1618	321	184
Margin			**20%**	22	44	37	24	62	24	64	37
Total w/ Margins				**131**	**265**	**220**	**1641**	**370**	**1641**	**385**	**220**

Table 4-16: Mother spacecraft mean Power by modes

The power consumption of the Smallsat is design to provide the maximum available power of 117W. Therefore, the mean Power of the strawman Payload is adjusted to fit in the Power budget and not to exceed the 117W that can be provided to the System. The results are shown in Table 4-17.

	P_on	P_stby	#	SLEEP	OPS	SAFE	BOOT
Jenoptik DLEM Laser Rangefinder	1.8	0.01	1	0.00	0.19	0.00	0.00
Memsense IMU 3020	0.5	0	1	0.00	0.50	0.50	0.50
Hyperion Technologies IM200	0.7	0	1	0.00	0.07	0.07	0.00
GomSpace SmallTorque GSW-600	0.3	0.3	3	0.00	0.30	0.30	0.30
Hyperion Sun Sensor SS200	0.04	0.0025	6	0.00	0.04	0.04	0.04
smallISL Electronics	10	2.3	1	0.00	6.15	6.15	6.15
Smallprop 6U PropModule	2	0	2	0.00	0.50	0.50	0.50
Dock Board	0.1	0	1	0.10	0.10	0.10	0.10
Platform OBC	0.6	0	1	0.60	0.60	0.60	0.60
Payload OBC	30	0	1	0.00	5.00	0.00	1.00
StrawMan Payload	137	0	1	0.00	81.50	0.00	68.50
Solar Array Control Unit	0.75	0	2	0.00	0.38	0.38	0.00
Power Control and Distribution Unit	0.5	0.5	1	0.50	0.50	0.50	0.50
Heater	51.39	0	1	0.00	0.00	51.39	0.00
Total w/o Margins				**1.20**	**97.50**	**62.20**	**79.49**
Margin			**20%**	0.24	19.50	12.44	15.90
Total w/ Margins				**1.44**	**117.00**	**74.64**	**95.39**

Table 4-17: Smallsat mean Power by modes

4.2.11 Structural Assumptions

The structural mass of the Mother spacecraft has been assumed to be 15% of the target dry mass of 450kg leads to a structural mass of 67.5kg. For the Smallsat structure a of the shelf product was selected. The average structural mass of a 16U SmallSat is 2.25kg. Because of these assumptions, a DMM of 20% is foreseen.

Figure 4-7: 16-Unit CubeSat structure by ISIS

4.3 System Options

The total wet mass of the system is 784.36 kg, with a launcher performance for the chosen baseline of 900kg, there is a difference of 115.64kg of mass. This allows options with more than 4 Smallsats. In Table 4-18 an overview of these options is given. The accommodation of additional SmallSats leads to the need of additional Small Satellite Deployer (7.78kg each) on the Mother spacecraft. Additionally there will be a growth in Harness mass and System Margin Mass. The growth of propellant mass needed for the options is not considered since the baseline propellant mass is already computed with the assumption that the wet mass of the system is 900kg.

The baseline option can be modified to host up to 6 SmallSats within the target Mass.

		Mass [kg]			
S/C Mass Budget		**4**	**5**	**6**	**7**
Dry Mass MC		**364.05**	**372.22**	**380.39**	**388.56**
System Margin	20%	72.81	74.44	76.08	77.71
Sum Small Sat Wet Mass		**117.61**	**147.02**	**176.42**	**205.82**
Dry Mass incl. System Margin		**554.48**	**593.68**	**632.89**	**672.10**
EPROP Propellant Mass		225.38	225.38	225.38	225.38
EPROP Propellant Residual	2%	4.51	4.51	4.51	4.51
Total Wet Mass		**785.164**	**823.57**	**862.77**	**901.98**
Target Wet Mass		900.00	900.00	900.00	900.00
Above Target Mass by		**114.836**	**76.43**	**37.23**	**-1.98**

Table 4-18: Mas Budget for options with different number of Smallsats

5 CONFIGURATION

5.1 Requirements and Design Drivers MC

The following requirements apply to the configuration of the Mother Spacecraft.

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
CON-010	The configuration shall fit within the constraints of the EPSILON, VEGA-C or in a dual launch configuration in an Ariane 6.2 launcher.	MIS-060
CON 020	The interface to the launcher shall be compatible with either a 937 or a 1194 standard adapter.	
CON-030	The configuration shall accommodate all Payload and Equipment required for the mission objectives and requirements.	
CON-040	The configuration shall accommodate the volume of 4 Small-Spacecraft defined in the Mission Objectives.	MIS-020, MC-010
CON-050	The Configuration shall accommodate Mechanical, Thermal, Power interfaces including a Deployment Mechanisms for 4 Small-Spacecraft.	MIS-020, MC-010, MC-030, MC-040, MC-170
CON-060	The Configuration shall take into account constraints and limitations due to AIV requirements.	
CON-070	The Configuration shall provide an unobstructed field of view for all instruments and equipment.	
CON-080	The Configuration shall provide an unobstructed deployment window for the 4 Small-Spacecraft.	
CON-090	The Configuration shall provide unobstructed position for the thrusters to fulfil the mission requirements without contamination of relevant parts of the spacecraft.	

5.2 Requirements and Design Drivers SS

The following requirements apply for a generic configuration of a Small Spacecraft.

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
CON SS-010	The SS Configuration shall be compatible with the dimensions of the SmallSat family of spacecraft.	
CON SS-020	The SS Configuration shall accommodate the Instruments and Equipment required for the objectives of the mission.	
CON SS-030	The SS Configuration shall comply with the Mechanical, Thermal and Power interface requirements of the MS.	
CON SS-040	The SS Configuration shall provide an unobstructed field of	

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
	view for all instruments and equipment.	
CON SS-050	The SS Configuration shall take into account constraints and limitations due to AIV requirements.	

5.3 Assumptions and Trade offs MC

The MC configuration is based on a small to medium satellite. This is mission driven and also by the requirement to provide volume support for all the instruments, equipment and Smallsats including interface hardware.

5.4 Assumptions and Trade offs SS

For the purpose of the study, the SS is based on existing definitions for CubeSats. These type of spacecraft are defined by a standard with what is called a 1U CubeSat which has the following dimensions: 100 x 100 x 100 mm. These dimensions are for the smallest body of the Cubesat, and do not include extra structure in one direction for interface purposes with the launcher dispenser (adding up to 113.5 x 100 x 100 mm). These dimensions provide building block dimension for larger than 1U CubeSats. Due to the initial expected payload and equipment in the SS, the dimensions of the SS are a multiple of the above-mentioned building block, and set for 16U (see further explanation in paragraph 5.6.

The rationale to assume the SmallSat to be a 16U Cubesat is based on standard dimension and available Dispensers for the deployment of the SmallSat. Ultimately the decision can be made to go for bespoke designs of SmallSats and Dispensers, to better fit the needs and objectives of a future mission. This would however come with additional development and qualification costs.

5.5 Baseline Design MC

This paragraph describes the Mother Spacecraft [MC]. Figure 5-1 shows the MC for both the stowed and the deployed configuration, and its main dimensions.

5.5.1 Description of the MC Spacecraft

Clear spacecraft features a two large Solar Array panels, a large High Gain Antenna, and on top the doors for the dispensers of the 4 SmallSats.

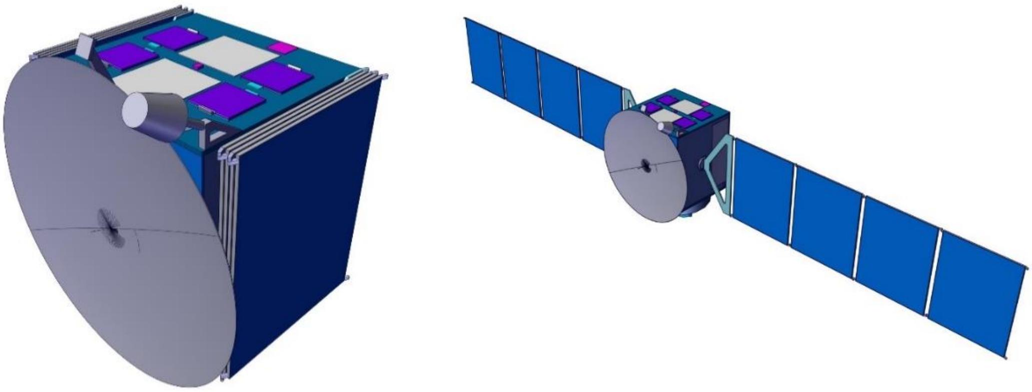


Figure 5-1: Mother Spacecraft stowed and deployed

Figure 5-2 shows the SPP spacecraft in a VEGA-C fairing. The volume available in the fairing of the VEGA-C provides sufficient space for the initial stowed configuration of the SPP spacecraft. An adapter will be required to interface with the launcher. Initially a standard 1194 adapter is foreseen, but due to the large electrical propulsion engines an additional interface ring will be needed between the Launcher 1194 to the Satellite interface.

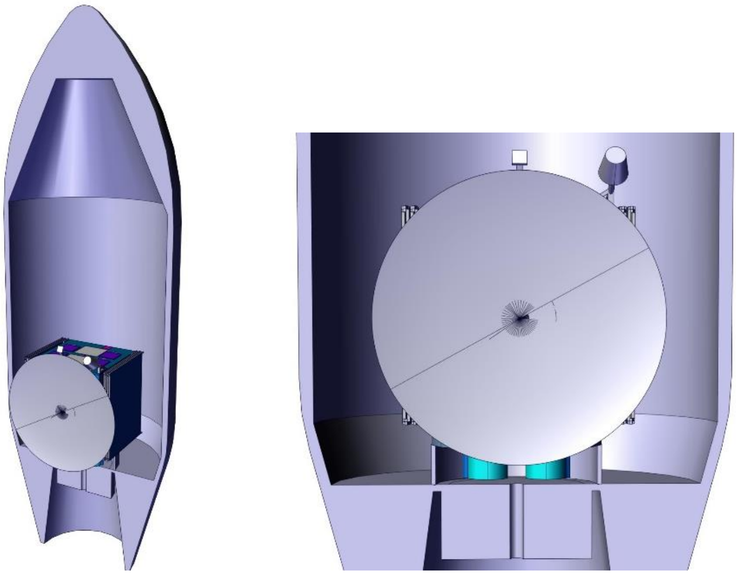


Figure 5-2: SPP spacecraft in a VEGA-C launcher fairing

Figure 5-3: shows the SPP spacecraft in the Ariane 6.2 dual launch fairing. Ariane 6.2 is the larger launcher (largest of the three considered in this study). It has more volume and as a result provides ample space for the spacecraft in stowed configuration. The 1194 standard payload adapter will be able to interface with the spacecraft directly.

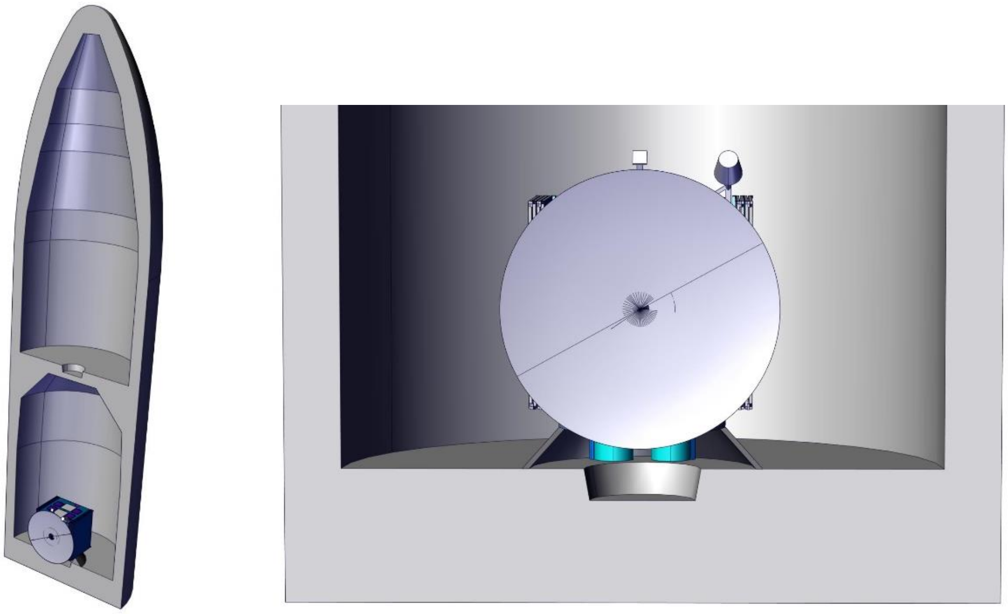


Figure 5-3: SPP spacecraft in an ARIANE 6.2 launcher fairing

Figure 5-4 shows the SPP spacecraft in the Epsilon launcher fairing. The available space in this fairing is the most challenging to comply with, relative to the VEGA-C and ARIANE 6.2. The conceptual configuration as proposed and shown for the two other Launchers will not fully fit in the EPSILON launcher fairing.

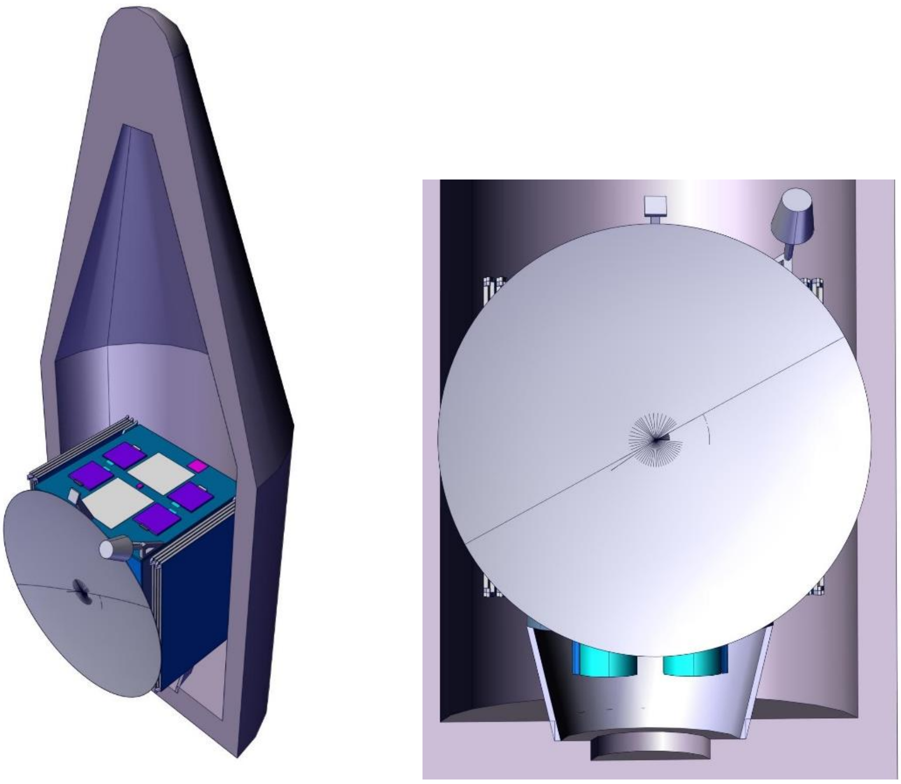


Figure 5-4: SPP spacecraft in an EPSILON launcher fairing

Figure 5-5 shows that there is an interference between the fairing and the 2 meter diameter High Gain Antenna [HGA]. This means that the first conceptual design for the spacecraft will not comply with the volumetric requirements of the EPSILON launcher.

Figure 5-5: Configuration issue for the EPSILON launcher fairing

The conceptual design can be adapted to accommodate the HGA in a different way than foreseen for the configuration for the VEGA-C and ARIANE 6.2 launcher. This would require a repositioning of various other elements, especially the SmallSat deployers.

Figure 5-6 shows an alternative configuration addressing the constraints for the HGA, as well as a new configuration location for the SmallSat deployers on the lateral panels of the spacecraft.

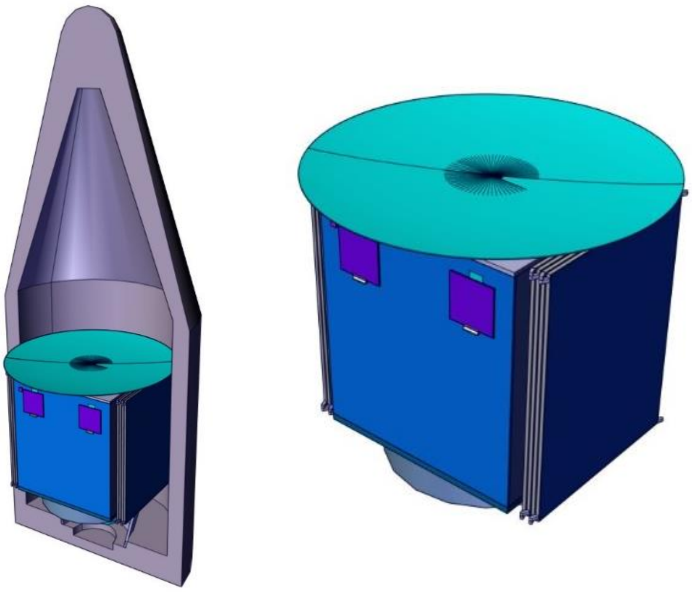


Figure 5-6: Alternative configuration for EPSILON launcher fairing fit

This option has not been studied in more detail, but seems feasible as an outcome for the study. This report details the initial conceptual design as shown in Figure 5-1 and Figure 5-7.

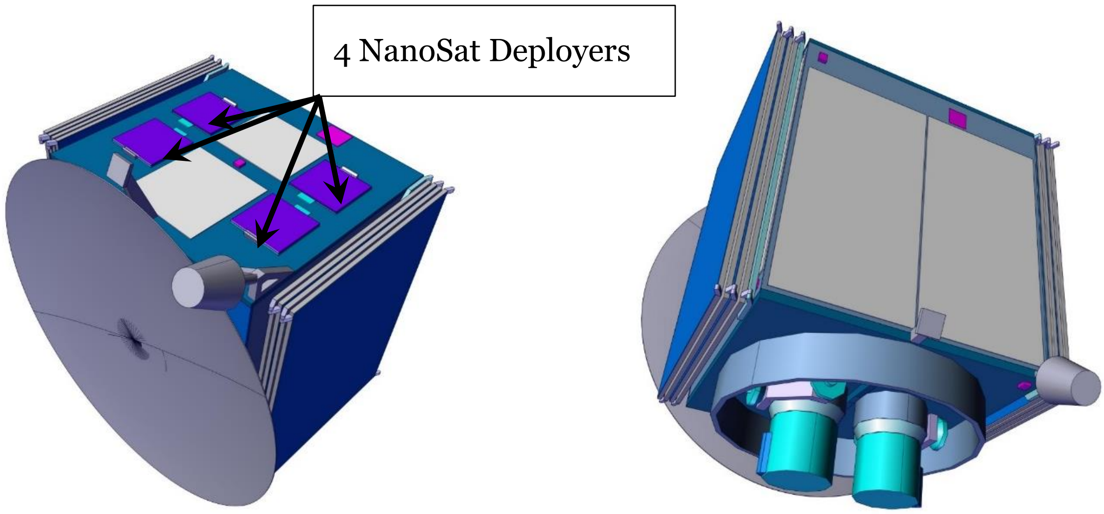


Figure 5-7: Conceptual design in stowed configuration

The configuration shown in Figure 5-7 shows a compact design, with the large HGA to the left front side, the solar arrays in stowed position to each side next to the HGA. Furthermore the main engines are shown on the bottom side (side with the interface to the launcher) and the four SmallSat deployers on the top.

Figure 5-8 shows the spacecraft in deployed configuration. The large solar panels are a result of the mission that requires the farthest distance to the Sun. In case of closer distance to the Sun, the number or size of the solar panels can be reduced. There is sufficient space in the preliminary design to add a panel in case more power is required.

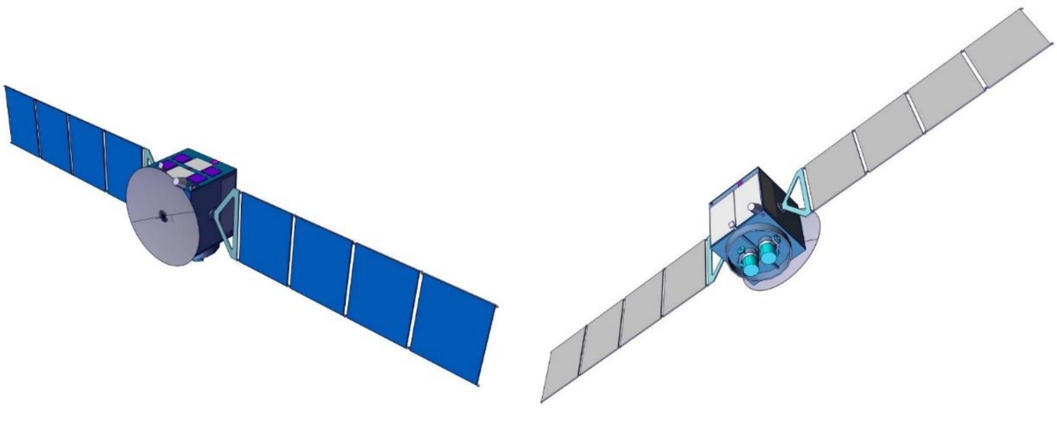


Figure 5-8: SPP deployed spacecraft

Figure 5-9 shows the major appendages to the spacecraft. In the left image the Solar panels are shown in exploded view. In the right image, the HGA and the Electrical Engines are shown in exploded view.

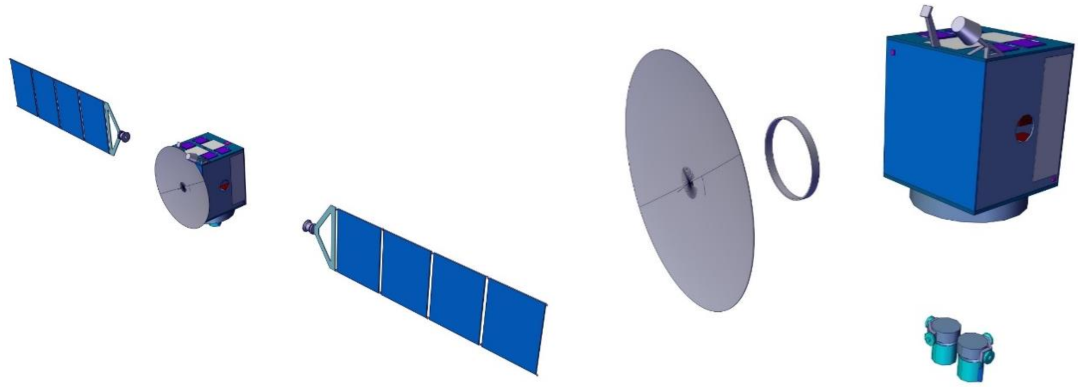


Figure 5-9: Solar Panels , HGA and Electrical Engines

At the core of the design is a compact body for the spacecraft. There are different elements attached to the outer panels of the structure of the body. The radiator panels can be seen in Figure 5-10.

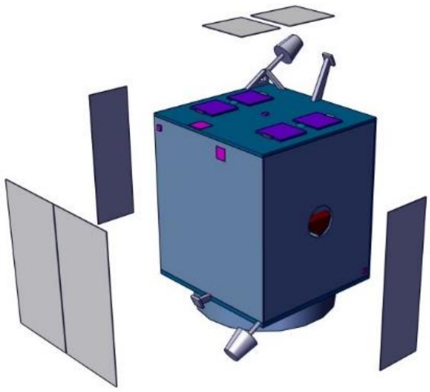


Figure 5-10: Radiator panels on the external panels of the spacecraft

In the exploded views of Figure 5-11 the external panels have been removed to show the internal layout of the spacecraft. In the left image the 4 SmallSat deployers are visible and still attached to the primary structure of the spacecraft. In the image on the right the deployers have been raised to show the location more clearly, in addition showing the four 16U SmallSat above (as if ejected from the deployers).

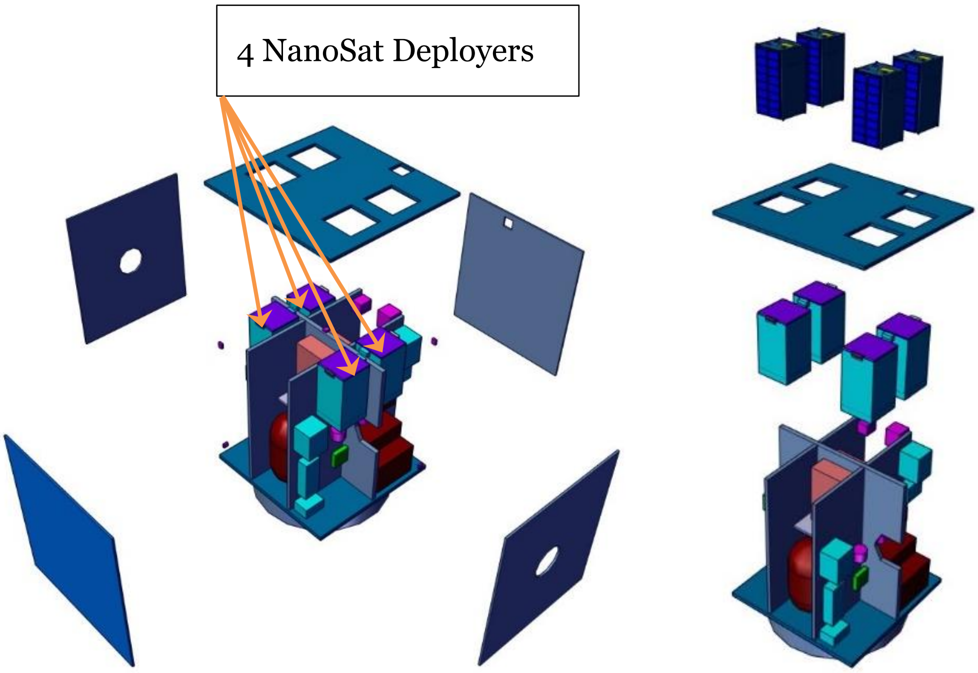


Figure 5-11: Exploded view of the spacecraft body

Before deploying the SmallSats, the doors of the deployers will open, so that the SmallSats can start their part of the overall mission. Figure 5-12 shows the opened doors of the deployers. This is a styled representation, since for most deployers of SmallSats the deployment is synchronous with the opening of the door.

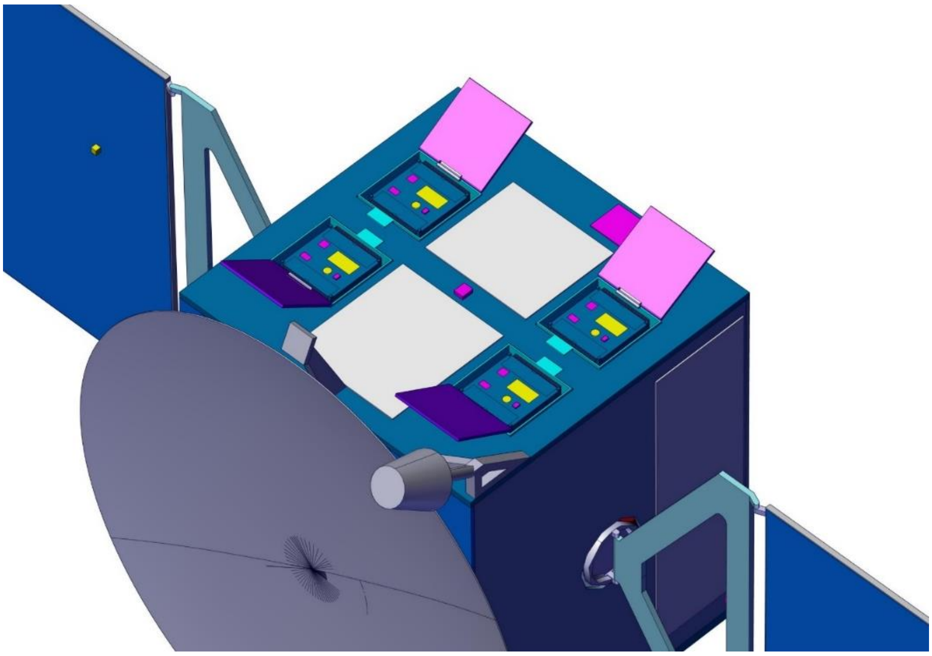


Figure 5-12: Opening of the SmallSat deployers

The deployment of the SmallSats is shown in Figure 5-13. The sequence of the deployment shall be decided on the mission requirements.

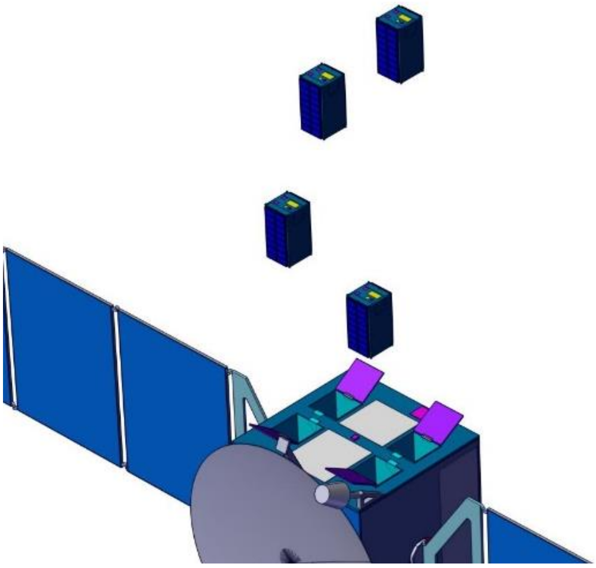


Figure 5-13: Deployment of the SmallSats

An initial overview of the equipment of the MotherCraft is given in Figure 5-14.

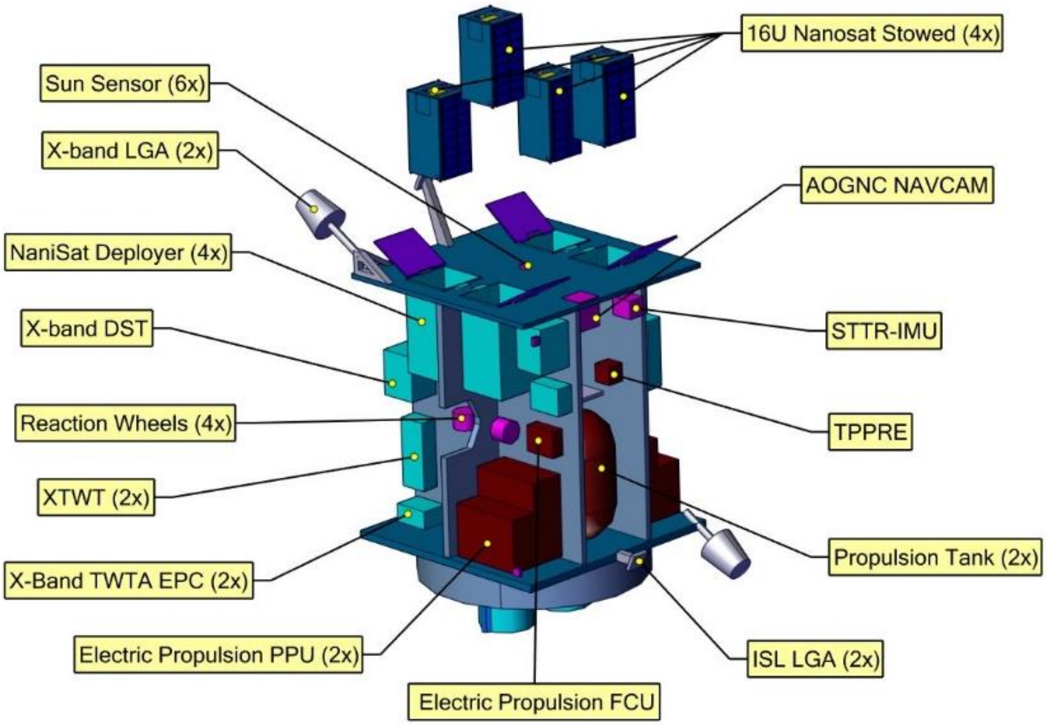


Figure 5-14: Equipment inside the MotherCraft

5.5.2 Main Dimensions of the MC spacecraft

The following images show the initial basic dimensions of the MS spacecraft. Figure 5-15 shows the outer dimensions of the spacecraft in stowed configuration.

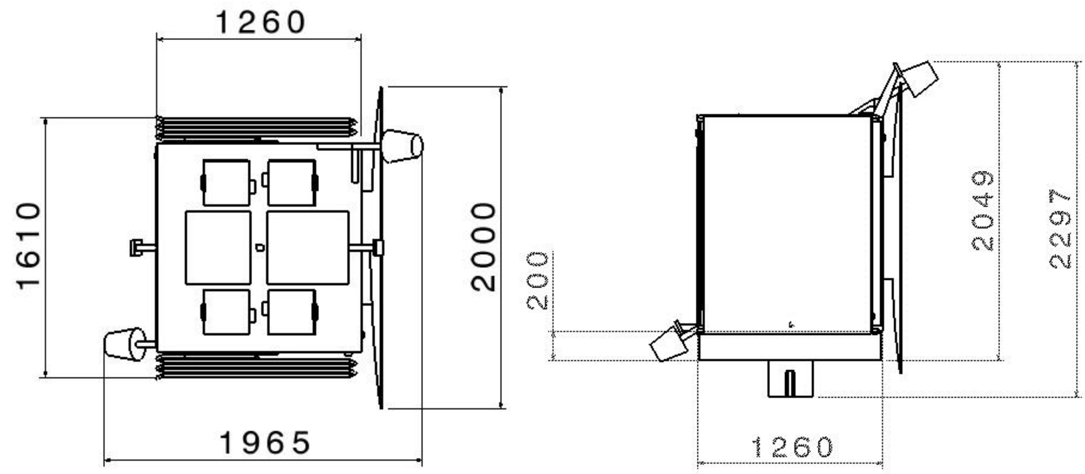


Figure 5-15: MC main dimensions in stowed configuration

Figure 5-16 shows the main dimension for the deployed configuration. Nothing changed for the main body of the spacecraft, except the deployment of the Solar Arrays.

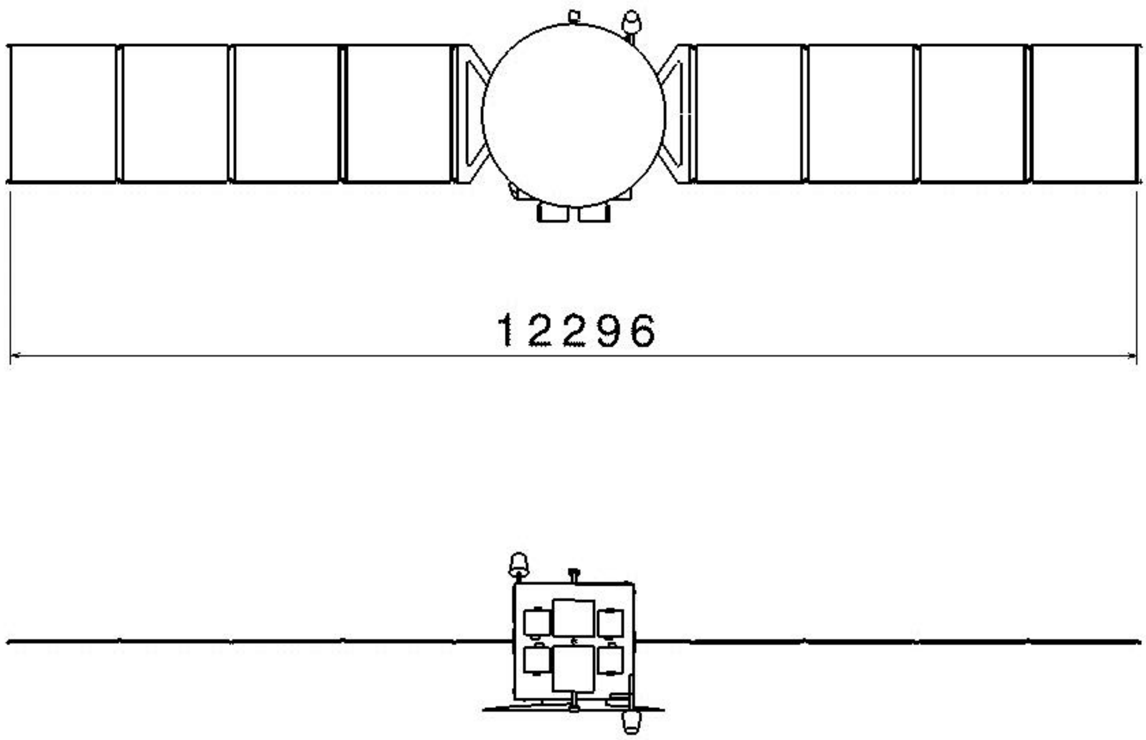


Figure 5-16: MC main dimension for the deployed configuration

5.6 Baseline Design SS

This paragraph describes the Small-Satellite [SS], and the dimensions.

5.6.1 Description of the SS Spacecraft

The SmallSats for this study have different "packaging" options for the instruments which are based on the mission requirements. The accommodation exercise studied and presented here is based on SAT 2 (see payload chapter 2.4). This SAT 2 configuration consists of the following payload:

- Mass Spectrometer
- Magnetometer
- Ion/Electron Spec
- Radio Science

Using the required equipment and instruments and their initial dimensions, the preliminary sizing of the SmallSat resulted in a 16U SmallSat design. The stowed and the deployed configuration are shown in Figure 5-17.

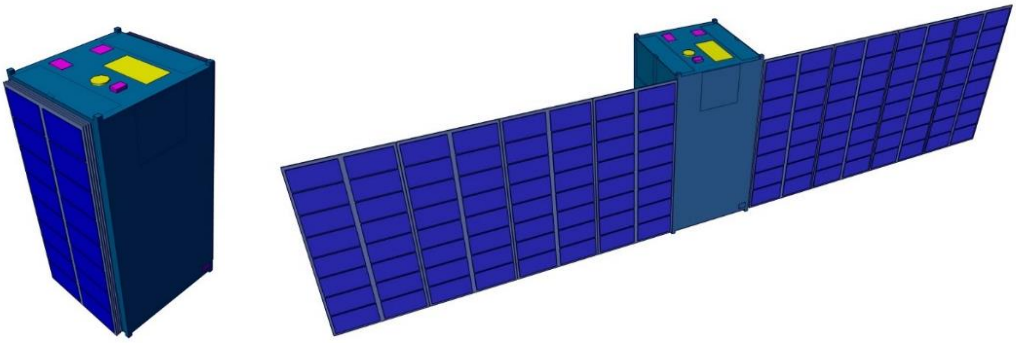


Figure 5-17: Selected option for SmallSat design

Figure 5-18 shows the deployed boom for the Magnetometer.

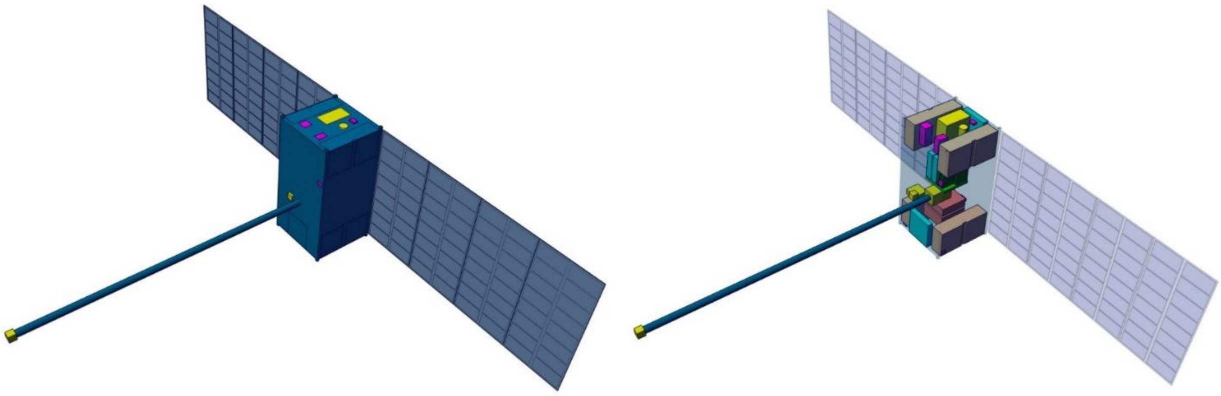


Figure 5-18: Deployed SmallSat, with deployed boom and Solar Arrays

Figure 5-19 labels different Instruments and equipment for the SmallSat. Not all space has been filled. The details of the structure of the SmallSat depend on the possible use of a COTS Frame. In addition the possible need for a Solar Array mechanism for rotating it into the best Sun position may require an at this time not sufficiently defined volume. Ultimately, when detailing the SmallSats, a different size than the standard 16U selected for this study may be an option, for example a 12U SmallSat.

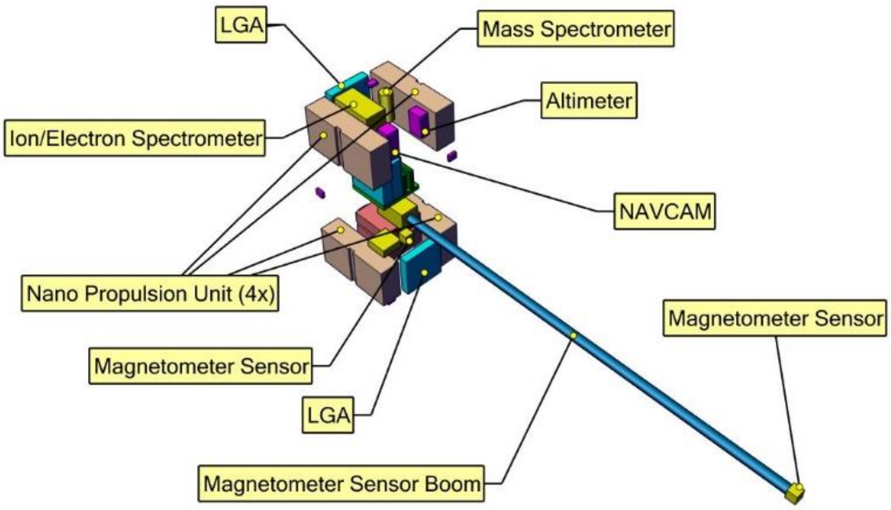


Figure 5-19: Instruments/Equipment for the SmallSat (Sat2)

5.6.2 Main Dimensions of the SS Spacecraft

Figure 5-20 shows the preliminary main dimensions of the SmallSat spacecraft. The dimensions for the SS are dependent on the size and type of SmallSat selected and required for the mission. These measures coincide with the choice to study the feasibility and use a standard 16U SmallSat as foundation. The Solar Arrays are based on preliminary dimensioning. Further study will have to show if the selected Deployer can accommodate the current stowed position of the solar arrays.

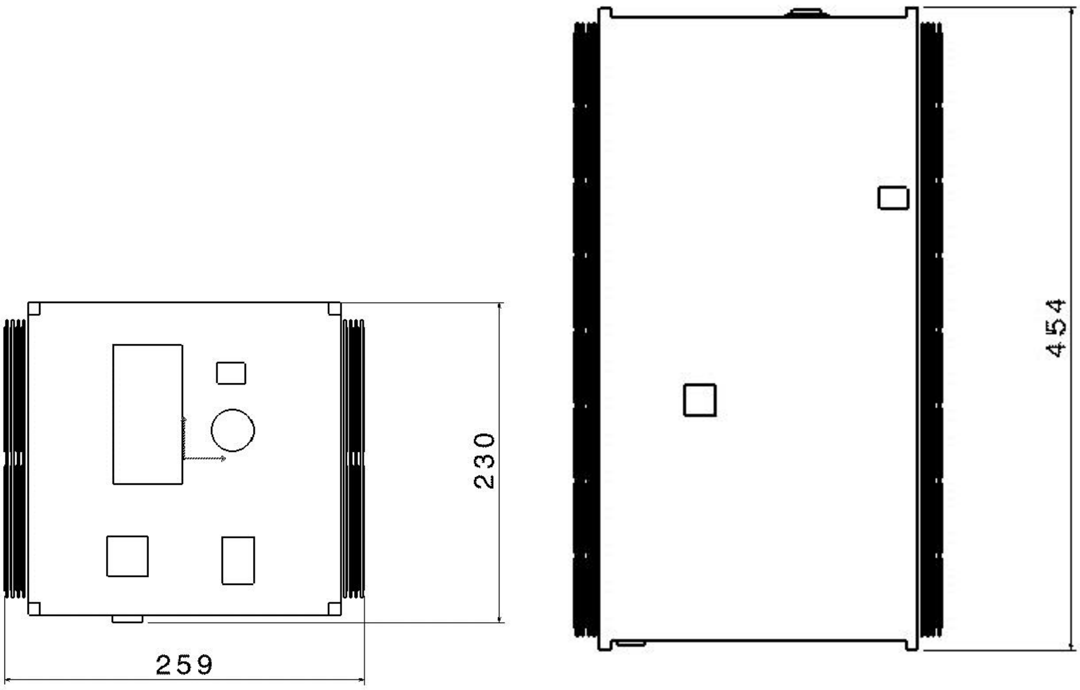


Figure 5-20: SS main stowed dimensions

When deployed the spacecraft will have a "wingspan" in the order of 1.8 meters, which is shown in Figure 5-21.

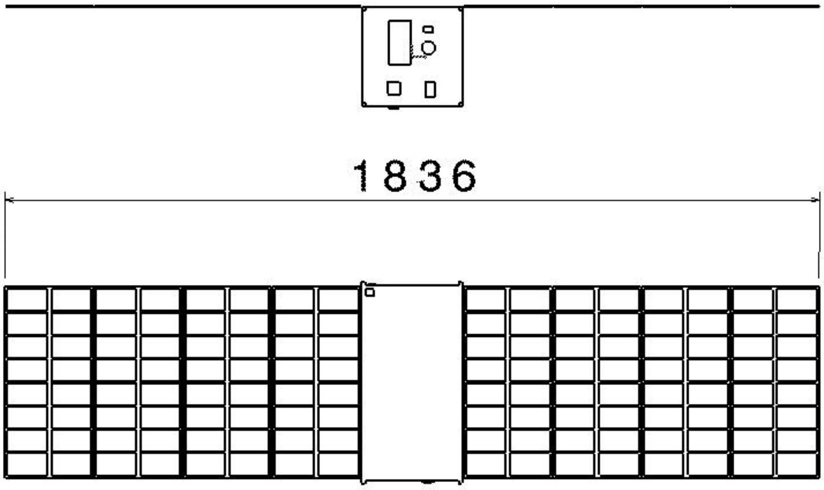


Figure 5-21: SS main deployed dimension

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6 MECHANISMS

6.1 Requirements and Design Drivers MC

The following System and mechanical subsystem requirements are applicable to the design of the MC mechanisms for the SPP mission, in addition to the generic mechanisms requirements within ECSS-E-ST-33-01C.

System & SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
MC-010	General requirement, see system section	
MC -050	General requirement, see system section	
MC-160	General requirement, see system section	
MC-170	General requirement, see system section	
MEC-010	The MC shall include 2 Solar Array Drive Mechanisms (SADMs) for 1 axis Solar Array pointing to the Sun	
MEC-020	The MC SADMs shall provide the capability to transfer up to 1200 W of power at target	

Table 6-1: Requirements applicable to the mechanisms of the MC

6.2 Requirements and Design Drivers SS

The following System and mechanical subsystem requirements are applicable to the design of the SS mechanisms for the SPP mission, in addition to the generic mechanisms requirements within ECSS-E-ST-33-01C.

System & SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
SS-040	General requirement, see system section	
MEC SS-010	The SS shall include 2 Solar Array Drive Mechanisms (SADMs) for 1 axis Solar Array pointing to the Sun	
MEC SS-020	The SS architecture shall be compatible with a SmallSat type configuration	

Table 6-2: Requirements applicable to the mechanisms of the SS

6.3 Assumptions and Trade Offs MC

6.3.1 Assumptions

To facilitate the selection and initial sizing of a mechanisms concept for the SPP MC the following assumptions have been made:

- It will be possible to accommodate an adequate number of hold down points between the solar array (SA) and MC to support the use of existing Solar Array Drive Mechanism (SADM) configurations
- The SA deployment mechanisms will be integrated into the SA

- No antenna pointing mechanisms are required as the MC can point adequately to Earth
- The reaction wheels and thruster pointing assemblies will be off the shelf items qualified for a relevant environment, thus not requiring mechanisms development options to be investigated herein.

6.3.2 Trade Offs

The following MC mechanisms are considered as part of the SPP study:

- MC SADMs
- SS deployers

Trade-offs for both these mechanisms are provided below.

For the SADMs the NEO target under assessment herein provides the opportunity to consider a smaller design than would be possible for a main asteroid belt target, as the max power generated by the SA and transferred via the SADMs is expected to be approximately 2.25 kW for both wings. Thus, SADM options are compared below.

Model	SEPTA 31 from Ruag	SEPTA 32 from Ruag
Mass (inc connectors)	3.6 kg	4.4 kg
Max Power transfer per SADM	2.2 kW	3.3 kW
Max Loads – Radial	333 N	500 N
Max Loads – Axial	200 Nm	250 Nm
Qualification life	84 000 cycles	100 000 cycles

Table 6-3: MC SADM trade-off

As shown in Table 6-3 the SEPTA 31 is lighter and meets the power transfer need, thus a SEPTA 31 class SADM is selected for the baseline.

Deployment strategy	Individual SmallSat type low velocity deployers	Integrated custom HDRMs system in MC
Mass	-	+
Compatibility with toolbox approach	++	-
Need for additional features on MC (e.g. shielding)	++	--
Reuse of existing technology	++	+
Ability to achieve low speed	++	++
Constraints on SS form	--	++
Summary	+++++	+++

Table 6-4: SS Deployer trade-off

As shown in Table 6-4 the approach of using individual SmallSat type deployers is considered as most suitable for the SPP mission and is therefore selected for the baseline.

It is notable that the off the shelf SmallSat deployers are generally compatible with deployment speeds down to 0.5-2.0 m/s, which is significantly above the SPP requirement of 0.05-0.07 m/s. It is therefore beneficial to divide the SmallSat deployment into its basic functions:

- Ejection out of the deployer
- Separation from the deployer with predefined speed.

Each function has its own dedicated mechanism. This has the advantage that both ejection and separation actuators can be sized independently to ensure compliance to the deployment velocity requirement while still achieving compliance to the general ECSS mechanisms requirements.

A market survey has shown that the only low velocity deployer available in Europe expected to be compliant with the release velocity requirement is the LV-POD from ISIS as considered for the AIM mission, thus this is the model which the deployer will be based on.

The sequence of stowage, ejection and separation is shown below.

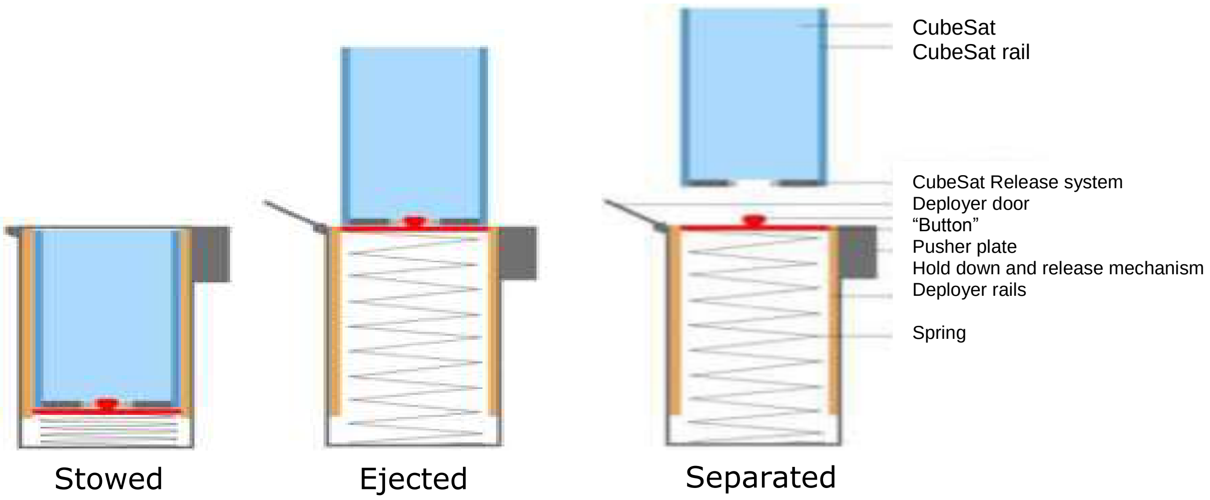

Figure 6-1: SS deployer functions

6.4 Assumptions and Trade Offs SS

6.4.1 Assumptions

To facilitate the selection and initial sizing of a mechanisms concept for the SPP SS the following assumptions have been made.

- Adequate synchronisation of the solar panel deployment can be achieved by tuning the hinges or HDRMs, so a dedicated synchronisation system will not be necessary
- The SA deployment hinges will be integrated into the SA
- The payload mechanisms are incorporated within the payload developments proposed, and thus not needing further mechanisms development to be discussed herein

- The reaction wheels will be off the shelf items qualified for a relevant environment and thus not needing further mechanisms development to be discussed herein.

6.4.2 Trade Offs

The following SS mechanisms are considered as part of the SPP study:

- SS SADMs

To select the most appropriate product for the SS SADM a market survey was performed as summarised in Table 6-5.

	IMT (I)	Honeybee (USA)	MMA Design (USA)
TRL	3	8-9	8-9
Mass (<6U)	< 300	ca. 180	ca. 250

Table 6-5: Solar Array Control trade-off

Note: A US company (SolAero) has started offering a roll out deployable SA called COBRA, however the TRL is unknown and the technology is not expected to initially be SADM compatible in the size needed.

Thus as the IMT SADM is the only known viable European option it is selected for the baseline.

6.5 Baseline Design and List of Equipment MC

The selected MC mechanisms for the baseline design are summarised below.

- SADM: SEPTA 31 class from Ruag x2
  - (TRL 8-9: No modification planned)

Figure 6-2: Septa 31 from RUAG

- Deployer: LV-POD from ISIS x4
  - (TRL 6: with the following modifications
    - Increase the size to accommodate larger SS
    - Move the separation release device to remain on the MC side

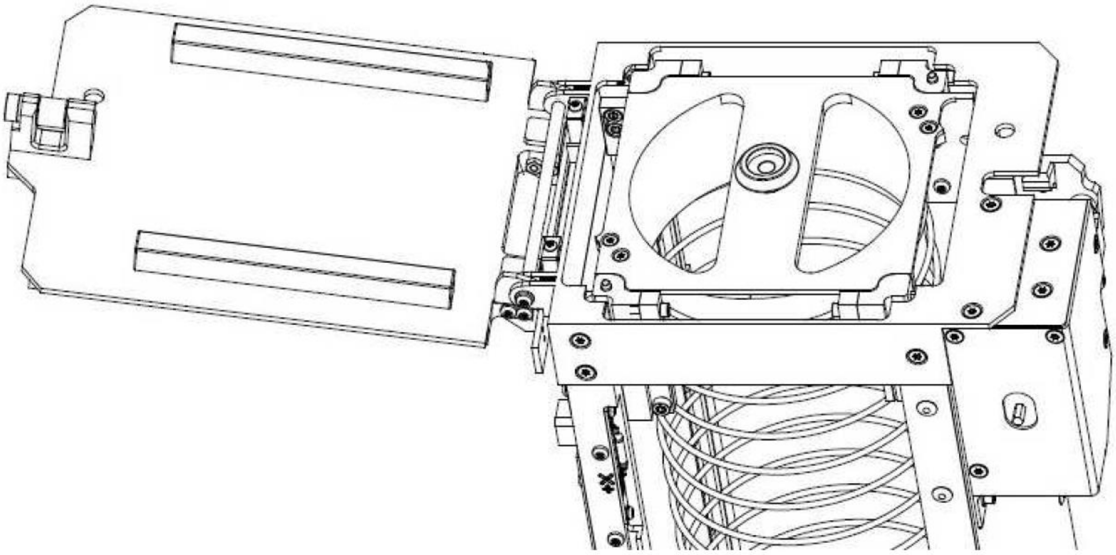

Figure 6-3: LV-POD from ISIS

Equipment	No. Off	Mass per item (kg)	Margin (%)	Total Mass Inc margin (kg)	Power On per item (W)	Power Off (W)
SADM	2	3.6	5	7.56	2.4	0.0
LV-POD	4	6.48	20	31.1	100 W (1 s)	0.0
Total				38.66		

Table 6-6: Summary of MC mechanisms equipment

6.6 Baseline Design and List of Equipment SS

The selected SS mechanisms for the baseline design are summarised below.

- SADM: SAC from IMT
    - TRL 3: with the following modifications
        - Wider design for 2U wide platforms and deployers

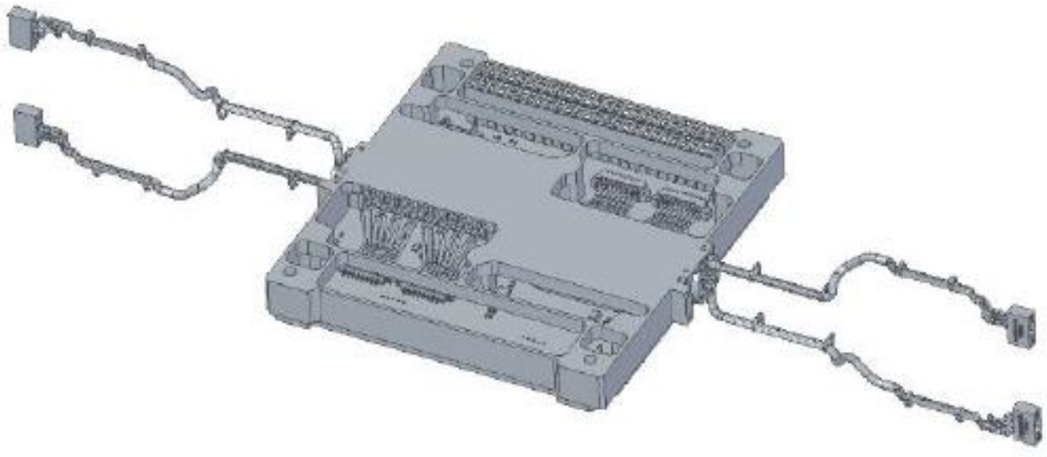

Figure 6-4: SAC from IMT (6U version shown)

Note: Although this solution has a low TRL it is highlighted that it is the only European option, it was also considered for the M-ARGO mission and is currently under development with the aim of reaching TRL 6.

Equipment	No. Off	Mass per item (kg)	Margin (%)	Total Mass Inc margin (kg)	Power On (W)	Power Off (W)
SAC	1	0.75	20	0.9	0.5	0.0
Deployer IF	1	0.1	20	0.12	0	0
Total				1.02		

Table 6-7: Summary of SS mechanisms equipment

6.7 Sensitivity Analysis for MC

The two MC mechanisms are very different and exhibit sensitivities to different factors. Regarding the SADM whilst it is clear that significantly increasing or decreasing the size of the SA can result in a corresponding increase or decrease in the size of the SADM selected. The capabilities of available SADMs are well documented and a change in mass would not be expected unless a significant change causes a change up or down a size, so this won't be further discussed here.

In the case of the SS deployers, these represent a significant mass and it is understandable that the size and mass will be sensitive to the size and mass of the SS, but little information is available so the sensitivity of the deployer mass to the size of the SS SmallSat configuration was further investigated. A parametric scaling of the SS deployer mass estimate based on the equivalent SmallSat max mass & surface area was created for some standard size SS and checked against commercially available deployers, as shown below:

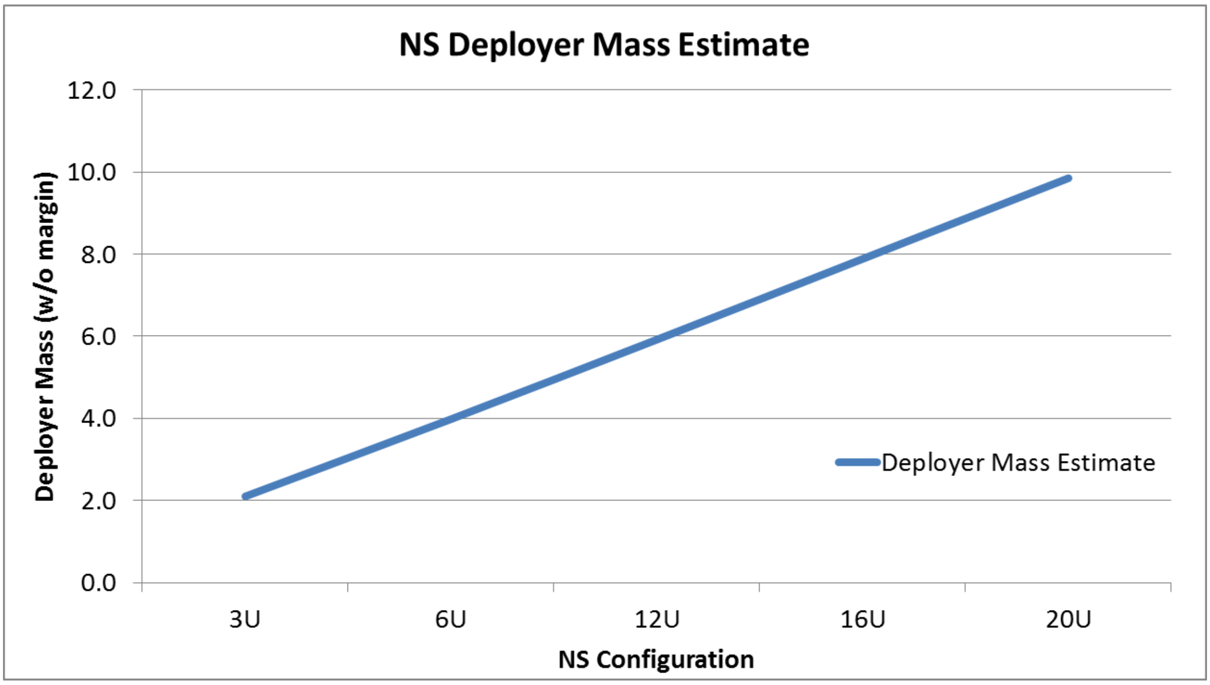

Figure 6-5: Sensitivity of the deployer mass to the SS configuration

As can be seen in the plot above the mass of the deployer will increase with an increasing size of the SS. This relationship is expected to be approximately linear for the SmallSat configurations shown but it is notable that other SmallSat form factors would deviate from this linear relationship.

6.8 Sensitivity Analysis for SS

The only mechanism utilised by the SS is the SADM and as there is only one European technology under development able to meet the need in this area there are limited options to reasonably assess possible sensitivities. The intention of the on going SADM development activity is to create a flexible SADM suitable for a wide variety of SmallSat applications. Thus, to leverage the benefits of this existing development it is important to stay within its capabilities, these include:

- Power: 90 W, with a target of 120 W in LEO
- Size compatibility: 6U/12U (interpreted as 1U/2U wide)
- Maximum rotational speed 0.4 °/s

See section 6.11 for areas of caution on this topic.

6.9 Sensitivity to Target: What if Phobos?

In the case a larger target is selected (e.g. Phobos) and the deployment speed requirement (MC-170) can be relaxed to approximately > 1 m/s a standard single stage deployer may be used. This would give the benefit of a lower number of separation devices to command and would thus improve the reliability. The baseline SS deployer mass could also be reduced by approximately 0.2 kg per deployer. A schematic of this logic is shown below.

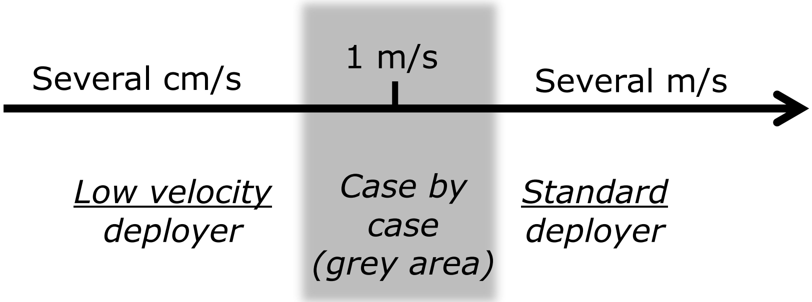

Figure 6-6: Sensitivity of the deployer type to the ejection velocity

6.10 Architecture Sensitivity Lander

Due to the range of possible lander cases it is beneficial to breakdown the assessment into two possible scenarios, as outlined below:

- Lander Scenario 1: **uncontrolled landing** (impact)
  - No obvious change to the mechanisms architecture
- Lander Scenario 2: **controlled landing**
  - Total re-design of SS mechanisms architecture including
    - Probable removal of the deployed Sas from any SS used as a lander to avoid control issues and damage due to the loads.
    - Additional landing equipment for a soft touch down or self-righting (such as for the MASCOT-2 mission) may be needed.

6.11 Major Design Constraints: CAUTIONS!

It is highlighted that the SS Solar Array accommodation within the deployer is at the limit of the number of panels which can be accommodated between the outside of the SS and the inside of the deployer wall.

The baseline 4 folded panels per wing are considered to be a very challenging configuration to fit within a SmallSat style deployment POD, and thus will need to use existing developments of thin solar panels. The available volume for the stowed SA inside a standard deployer is shown schematically in the figure below, this volume would need to be 94aximized during the necessary LV-POD re-sizing activities to accommodate the larger SS.

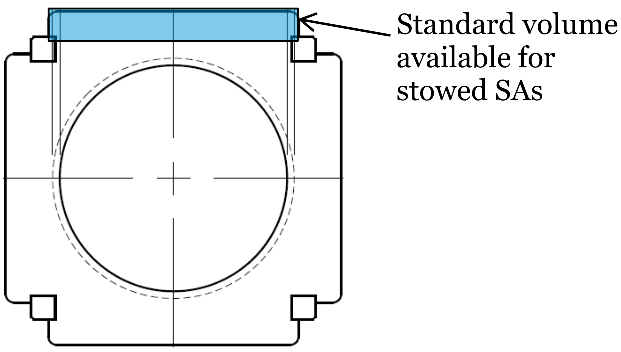

Figure 6-7: SmallSat allowable Volume cross section

It is also notable that a higher number of deployable panels per wing results in a higher degree of difficulty to reliably predict the deployment dynamics due to variations in the deployment parameters like friction in the hinges. With four panels per wing there is a potential risk of clashes, thus the deployment hinges would need to be carefully designed. Possible developments necessary could include: different sized hinges on the different panel hinge lines, or the use of lateral panels.

Any increase in the required number of panels would require a non standard stowed envelope and an additional synchronization mechanism, which would not be compatible with the use of a normal SmallSat style deployer.

6.12 Technology Requirements

The following technologies are required or would be beneficial to this domain:

Equipment and Text Reference	Technology	Suppliers and TRL Level	Additional Information
LV-POD	Low velocity SmallSat Deployer	ISIS (NL) TRL 6	The low velocity technology is expected to be used in flight on the RemoveDEBRIS mission, however this will require modification to accommodate the larger SS for the SPP Mission
SAC	SS SADM	IMT (I) TRL 3	An activity is on going to increase the TRL level to 6, however this may require modification to accommodate and equivalent of 4 panels on a 2U wide platform as these are not strict requirements for the development.

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7 CHEMICAL PROPULSION

7.1 Requirements and Design Drivers SS

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
PROP SS-010	Required Δv=10 m/s. This includes the overall pointing as also velocity change of the spacecraft.	
PROP SS-020	Lifetime of several years (passive) + several months (active)	
PROP SS-030	No general direction requirement for the S/C	
PROP SS-040	Low complexity and mass optimised system	
PROP SS-050	Smallsat frame if possible	

7.2 Assumptions and Trade offs SS

7.2.1 Assumptions

The following assumption table includes the information taken for the Smallsatellite propulsion system. Since the corresponding requirements were so general the table is also seen as a first starting point for a detailed assessment for other missions.

Assumptions	
1	ECSS compliance not to be strictly followed. This means that overall propulsion systems build from sub-equipments (tanks, thruster, pipes, valves, …) were not assessed in detail.
2	The spacecraft does not need the propulsion system for any kind of safe mode or for any kind of fine pointing. This implies that the direction in which the thrusters are mounted is not so critical as the spacecraft could rotate itself to enable the thruster to fire in the right direction.
3	The propellant mass of the entire spacecraft can be used in different manoeuvres. Since the system is build up from different single propulsion systems for Smallsats, the corresponding tanks mounted inside are not connected to each other. This means that, if the entire propellant has to be used, first one module has to be used until the propellant is empty and then another one is to be used after rotating the Spacecraft. If there would be any kind of time constraint this has to be checked against the specifications of the module.
4	Usage of the propulsion system in relation to any other spacecraft at launch site does not impose additional safety impacts. This is seen as not as critical due to the chosen system but cannot be ensured entirely (pressure vessel used). Any additional impact has to be assessed in detail for a given mission.
5	The Propulsion System can be monitored and maintained in temperature during passive mode. This means that during the dormant mode of the Smallsat attached to the mother Spacecraft the corresponding temperature and possible leakage of the system can be monitored and any FDIR is done within the mother spacecraft.
6	For the system no dedicated thrust requirement was provided. Therefore, and to maintain the possibility of using fine delta v firings, the mN thrust range was chosen to be appropriate.

7.2.2 Trade Offs Kick-Stage Propulsion System for MC

7.2.2.1 Kick-Stage propulsion systems

During the first CDF session, a kick-stage based on chemical propulsion was discussed. In session 2, this kick-stage was discarded due to mission constraints (passenger and therefore maximum mass capability) and cost impacts based on preliminary assessments. A corresponding summary is included as it is strongly recommended to reconsider the appropriateness of a kick-stage, based on the specific mission targets and constraints applicable..

These kick-stages are addressed for solid and also for liquid propulsion systems.

7.2.2.1.1 Lisa Pathfinder Propulsion Module

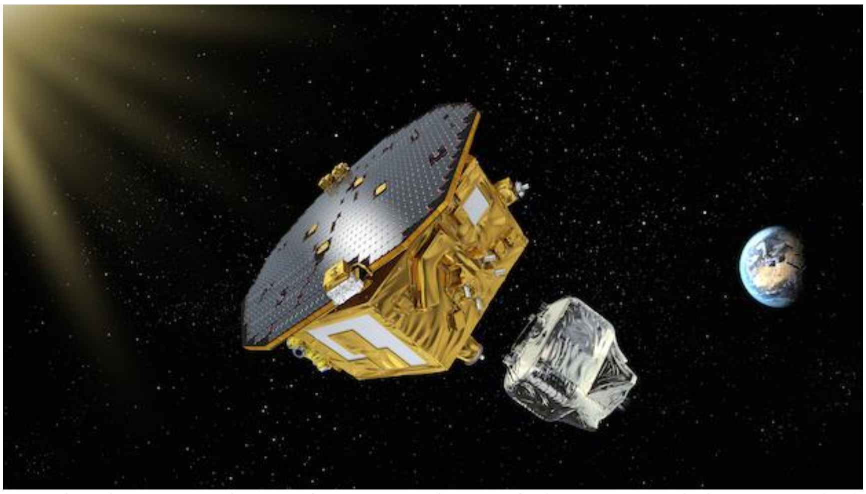

Figure 7-1: Artist impression of separation of the Propulsion module from Lisa Pathfinder (http://sci.esa.int/lisa-pathfinder/57156-lpf-propulsion-module-separation/)

The Lisa Pathfinder propulsion module was used to move the spacecraft into the L1 orbit from the earth orbit it was inserted into. The main parameters of this kick-stage were:

Propulsion module dry mass [kg]	Average Isp [s]	Propellant mass [kg]
220	320	1250

Table 7-1: Lisa Pathfinder propulsion module parameters

7.2.2.1.2 Mars Sample Return kick-stage

This MSR kick-stage was assessed within a CDF study and is delivering the following main characteristics:

Propulsion module dry mass [kg]	Average Isp [s]	Propellant mass [kg]
360	313	1491

Table 7-2: Mars Sample Return kick-stage parameters

7.2.2.1.3 The advantage of a liquid kick-stage is the overall thrust accuracy and the possibility of having several firings. As can be seen from the Lisa Pathfinder Propulsion module, the overall specific impulse is in the order of 320s. In general, about 25% of the overall propellant can be assumed as dry mass of the propulsion system.

7.2.2.1.4 Solid rocket motors as usage as possible kick-stages

The following table lists overall the solid rocket motors available from ATK.

Name	Burn Time [s]	Total Impulse [Ns]	Average Thrust [N]	Mass Total Loaded [kg]	Mass Propellant [kg]	Burnout [kg]	Isp [s]
Star 12GV	13.9	91940	6472	42	33	9	279
Star 13B	14.8	115876	7598	47	41	6	278
Star 15G	33.3	223345	6539	94	80	13	279
Star 17	17.6	197946	10943	79	70	9	282
Star 17A	19.4	319382	16014	126	112	12	282
Star 20	27.4	772033	24465	300	273	27	250
Star 24	29.6	560476	18549	218	200	16	280
Star 24C	28	613854	20684	239	220	18	269
Star 26	17.8	616078	33362	269	231	38	263
Star 26B	17.8	635028	34625	261	238	23	264
Star 26C	16.8	621861	35007	263	232	30	259
Star 27	34.4	950985	25444	361	334	24	267
Star 30BP	54	1461040	26623	543	505	33	290
Star 30C	51	1672953	32472	630	591	34	286
Star 30C/BP	51	1704869	32917	632	591	36	290
Star 30E	51	1812872	35141	674	631	37	289
Star 37GV	49	2823552	56937	1085	974	104	292

Table 7-3: Solid rocket motors from ATK (https://www.orbitalatk.com/flight-systems/propulsion-systems/docs/2016%20OA%20Motor%20Catalog.pdf)

The red marked motors are ones seen as a good starting point for this mass class. Any kind of additional impact (spin-rate due to high thrust in the order of 20kN and higher) must be assessed on top. Additionally, gimballing or thrust vector control by other means was not assessed and shall be analysed for every mission.

7.2.2.2 Smallsat Propulsion system trade-off

Table 7-4 shows feasible propulsion systems which were taken under consideration. The focus was set on current available and in-development European Smallsat propulsion systems. Since there is no European hydrazine smallsat propulsion available, the MPS-120 by Aerojet was included in the list to assess also the potential of having a monopropulsion system with an equivalent Isp for this propellant. For the corresponding class of mission (mass, complexity, delta-v requirement, …) the development build up from commercial off the shelf units (COTS) was not considered. If the corresponding parameters change significant this assessment has to be done again.

As a result of the corresponding safety impacts and the monitoring issue for the entire lifetime, a cold gas system was favoured for the corresponding mission application.

	Gomspace Nanoprop 3U	Gomspace Nanoprop 6U	Aerojet MPS-120: 1U	Aerojet MPS-120: 2U	Hyperion PM200	Nanoavionics EPSS	Tethers Unlimited: HYDROS-C
Propellant	Butane	Butane	Hydrazine	Hydrazine	N2O/Propene	ADN-blend	Water
Nominal Thrust [N]	0.001/0.04	0.001/0.04	0.25 – 1.25	0.25 – 1.25	0.5	0.3	1.2
Specific Impulse [s]	60-110	60-110	206 - 217	206 - 217	285	220	310
Max. Firing Time	-	-			10	60	
Dry Mass	0.3	0.77	1.06	1.36	1.1	0.6	1.02
Total Impulse			0.3				2151
Useable Propellant	0.05	0.13	0.38	0.98	0.3	0.2	0.74
TRL	6	6	3	3	4	7	6
Characteristics			Non-European Component	Non-European Component	No Flight qualification now		No flight qualificiation now

Table 7-4: Currently available and in-development European Smallsat Propulsion Systems

7.3 Baseline Design SS

The Baseline Design of the Smallsat contains 4 times the Gomspace Smallprop 6 U Unit.

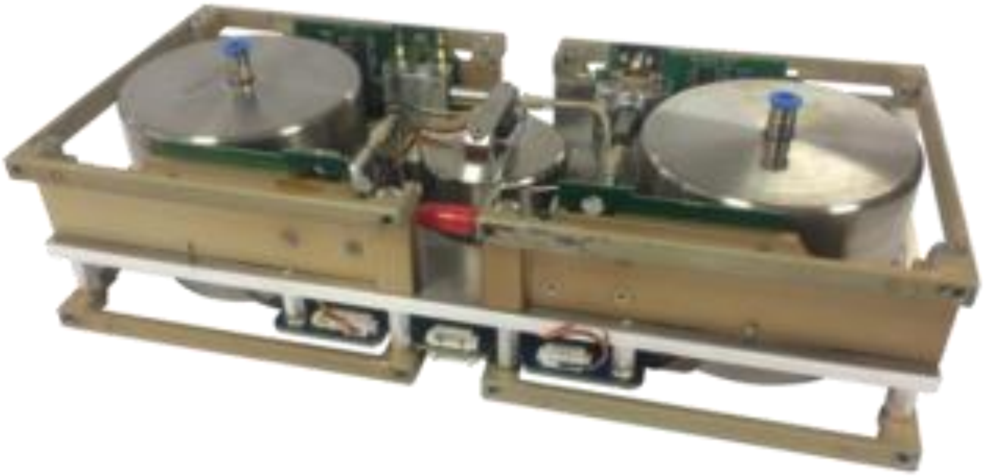

Figure 7-2: Gomspace Smallprop 6U Equipment (https://gomspace.com/Shop/subsystems/propulsion/nanoprop-6u-propulsion.aspx)

The technical features of one unit are:

Configuration:

- 4 individual thrusters
- separate main tanks
- Closed-loop thrust control
- Real time thrust measurement
- Propellant: Butane
- Propellant safety barriers: Min. 2

Specifications:

- Thrust: 1mN or 10mN (4x)
- Thrust resolution: 10µN or 100µN
- Specific impulse: 60-110sec
- Total impulse 80Ns
- Power consumption < 2W (average)
- Operating pressure: 2-5bar
- Temperature range 0° to 50°C

Interfaces:

- Communication: CAN, I2C
- Protocal: CSP (optional)
- Supply voltage: 5 VDC and 12 VDC
- Maximum Current: <1.5 A

Mechanical Features:

- Dimensions 200 x 100 x 50 mm^3, (including electronics board)
- Interface: 8x M3 (PC/104 spec)
- Mass (dry/wet) 770/900g

To achieve the mission requirements of a delta-v of 10m/s and to perform attitude control manoeuvres, four individual systems are used on the S/C. In the current baseline, no connection between the different tanks is foreseen, but could be introduced. Also a system with bigger, but shared tanks can be taken under consideration, which could reduce the number of units and allow greater flexibility.

For the performed calculations, the Isp from the lower end of the spectrum was taken. During the qualification process, higher levels will be aimed, so the propellant needs would decrease or the performance capabilities would increase.

The baseline system is not qualified for deep space environment. Impacts from e.g. radiation have to be investigated and the design has to be adapted. Also the long passive lifetime with constant temperature monitoring has to be taken into account.

7.4 List of Equipment SS

The list of Equipment for the Smallsat option is to have 4 times the Gomspace Nanoprop 6U unit built into the spacecraft.

7.5 Sensitivity Analysis for SS: What if?

Corresponding sensitivities are addressed:

- Higher delta v requirements:
  - The corresponding chosen system can be adapted in terms of tank sizes for the propulsion module. Care must be taken that the corresponding thruster and the performance has to be assessed in detail for the higher throughput.
- Thrust control
  - Currently, there is no dedicated requirement for thrust vectors and therefore thrusters were accommodated in the easiest way for configuration. If there is a special need for thrust vectoring, corresponding adaptations of the thrusters or the system will have to be investigated.
- Thrust range
  - If the thrust range has to be increased significantly, the overall approach would be to look for different modules or to qualify the corresponding module for higher thrust ranges.

7.6 Sensitivity to Target: What if Phobos and Lander

No sensitivity assessment was done for the Phobos and the Lander scenario since they do not differ from each other.

7.7 Major Design Constraints: CAUTIONS!

The major design constraint for this type of propulsion system for the Smallsat propulsion system is the ECSS compliance of the corresponding system. Any potential impact (dormant mode, reliability, ...) has to be assessed in detail and what kind of impact the corresponding system can have on the MC. For example, inadvertent firing of the thruster of the Smallsat propulsion systems would affect the overall Mother spacecraft due to the same order of thrust.

7.8 Technology Requirements

The following technologies are required or would be beneficial to this domain:

Included in this table are:

- Technologies to be (further) developed
- Technologies available within European non-space sector(s)
- Technologies identified as coming from outside ESA member states.

Equipment and Text Reference	Technology	Suppliers and TRL Level	Technology from Non-Space Sectors	Additional Information
Kick-stage applications	Water propulsion for kick-stage applications	-	no	This technology would be beneficial in terms of kick-stage application. Since the corresponding system have the potential of increasing the overall specific impulse the performance of the kick-stage can be improved.
High Performance Smallsat Propulsion System	e.g. Mono-/Bipropellant System	See Table 7-4	No	
Deep Space Qualification for Smallsat Propulsion Systems	-	-	-	

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8 ELECTRIC PROPULSION

8.1 Requirements and Design Drivers MC

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
EPROP-010		MIS-100
EPROP-020		MC-120
EPROP-030		MC-260
EPROP-040		MC-270
EPROP-050		MC-280
EPROP-060	The use of the Electric Propulsion subsystem shall not generate charging of the satellite or any of its parts, this includes solar arrays, reflectors, etc.	
EPROP-070	It shall be possible to reconfigure the propulsion subsystem after failure of one thruster, by isolating the failed thruster.	
EPROP-080	The Propulsion Subsystem shall include all propellant components and assemblies associated with storing, conditioning, routing, controlling and expelling propellant, as required to meet the mission requirements, from the moment of separation from the launch vehicle up to the End-of-Life.	
EPROP-090	The design and layout of the xenon feeding system (pipework, valves and regulators) shall ensure that during operations the xenon flow does not exhibit instabilities due to xenon change of state, by operating above the xenon critical temperature.	

8.2 Requirements and Design Drivers SS

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
EPROP SS-010		SS-080
EPROP SS-020		SS-150
EPROP SS-030		SS-160
EPROP SS-040		SS-170
EPROP SS-050	The use of the Electric Propulsion subsystem shall not generate charging of the satellite or any of its parts, this includes solar arrays, reflectors, etc.	
EPROP SS-060	The Propulsion Subsystem shall include all propellant components and assemblies associated with storing, conditioning, routing, controlling and expelling propellant, as required to meet the mission requirements, from the moment of separation from the launch vehicle up to the End-of-Life.	

8.3 Assumptions and Trade offs MC

8.3.1 Assumptions

The selection between different electric propulsion subsystems is based on a compromise between the need of systems capable to provide adequate thrust to reduce mission duration, maximising the specific impulse (to minimise the propellant mass requirements), and the need to reduce the EPROP power demand (to minimise the power generation system mass). Further, due to the relatively high total impulse expected as a consequence from the high demand in change in velocity, lifetime constraints of the individual thrusters are considered in the trade-off as well.

In addition to the above considerations, the performance capabilities of the available technology have to be taken into account to avoid additional development costs wherever possible.

	Assumptions
1	Wet mass of Mother S/C is 900 kg
2	Full-electric transfer from L2 to NEO target (Δv = 4-5 km/s w/o margin)
3	Power available to EPROP subsystem at Earth may exceed 5 kW
4	10% of Δv as margin with average I_{sp} (derived from MA analysis)
5	Thruster performance adjusted according to power level available
6	Deviation to nominal and demonstrated performance to be kept low
7	Equipment used on other missions is suitable for near-Earth transfer
8	Baseline architecture is single-point failure tolerant

8.3.2 Trade Offs

Three electric propulsion subsystems have been evaluated during the SPP study:

- A subsystem, based on a 1Nominal+1Redundent high-power Gridded Ion Engine (T6 by QinetiQ), developed and under qualification for BepiColombo and AlphaBus-NG programme.
- A subsystem based on a 2N+1R medium-power Hall Effect Thruster (PPS1350-G by Safran), flown on SMART-1 and qualified for AlphaSat.
- A subsystem based on a 1N+1R medium-power NextGeneration Hall Effect Thruster (PPS1350-E by Safran)

For the evaluation of the thruster subsystems, the performance was scaled to power according to the empirical functions derived from the qualification and performance testing. The additional PPS1350-G is a necessity to fulfil the total impulse requirement derived from the assumed change in velocity and wet mass at launch. The next-generation thruster PPS1350-E has an expected higher total impulse capability, and therefore requires only 1 thruster to fulfil the requirement.

To allow for a comparison of the propulsion architectures, the mass of the solar array has been included in the trade-off. Since the T6 PPU input is about 3 times higher than for the Hall effect thruster considered here, the S/A mass is considerably higher. The power subsystem is sized in the Hall thruster case to provide 1.5 kW to the PPU at the NEO target.

Redundancy has been considered for the thruster head w/ FCU, the PPU, and the thrust vectoring mechanism. No internal redundancy was considered.

In Figure 8-1, the results of the trade-off are summarised. While the propellant consumption for the Hall effect thruster options is considerably larger due to the lower average specific impulse, the heavy redundancy and the larger solar array for the ion engine compensates the propellant increase. By and large, the T6 and PPS1350-E subsystem masses (including S/A) are comparable. It was, however, expected that the lower power of the PPS1350 architecture would lead to less stringent requirements on other subsystem components (power, thermal, S/A mechanisms and structures, etc.), that could potentially lead to lower masses than compared to the T6 architecture. Therefore, the PPS1350-E architecture is baselined.

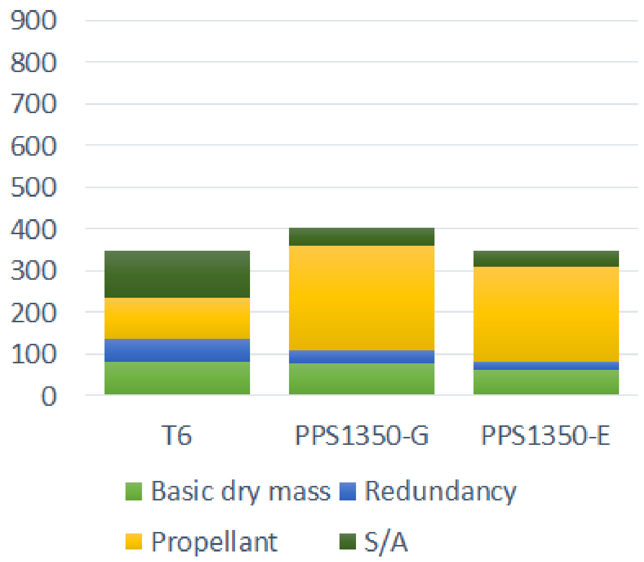

Figure 8-1: Comparison of subsystem wet masses for the considered trade-off

8.4 Assumptions and Trade offs SS

8.4.1 Assumptions

Assumptions	
1	Power available to EPROP subsystem > 50 W
2	Wet mass of Smallsatellite is 20 kg

8.4.2 Trade Offs

Resulting from the high power demand of one of the scientific instruments on board the SS, the power architecture is sized for more than 100 W and therefore could provide a substantial amount of power to the EPROP subsystem since the duty cycles of both propulsion and scientific instrument could be organised to not overlap.

Depending on the required thrust level and change in velocity, different technologies could be considered for a trade-off. Since cold gas propulsion is baselined, a trade-off for an EPROP subsystem was not conducted. Candidates might include PPTs, FEEPs, and resistojets.

8.5 Baseline Design MC

Based on the above considerations, a Propulsion System using the Hall Effect Thruster PPS1350-E by SAFRAN-Snecma (France) has been proposed for the EP transfer. The PPS1350-E is based on the PPS1350-G, which was the main propulsion source on SMART-1 and which was qualified for the Alphabus platform. The PPS1350-E offers higher power (up to 2.5 kW), providing a significant increase in thrust (+ 50%) and specific impulse RD[41].

In 2015 the PPS1350-E was selected by Space Systems/Loral to equip its telecommunication platforms. With upcoming flight opportunities the thruster will soon have the level of maturity suitable for long-term interplanetary missions, and is shown in (Figure 8-2).

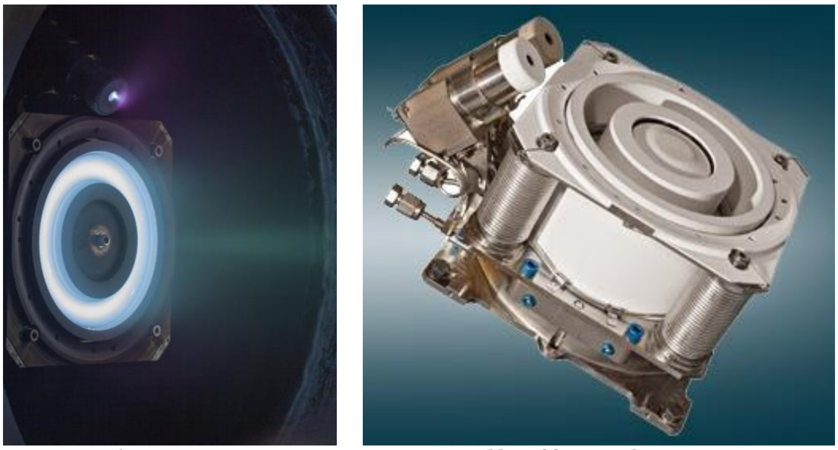

Figure 8-2: PPS1350-E Hall Effect Thruster

The SPP EPS architecture (shown in Figure 8-3) consists of:

- 1 nominal and 1 redundant PPS1350-E Hall Effect Thruster
- 1 xenon storage and feed system, comprising storage tank, valves, filters, 1 pressure regulator with its driving electronics and pipework
- 2 Thales Mk II PPUs with switching capability to command/control both thrusters and their neutralisers, necessary to counterbalance the positive charges of the ions expelled from the thrusters; the PPU provides a single interface to the spacecraft avionic and power subsystem
- 2 XFC flow control units from Safran-Snecma, one for each thruster, to deliver the required flow rate at each thrust level commanded
- 1 Thruster Pointing Mechanism (TPM) from RUAG Space Austria suitable to accommodate 2 thrusters
- Harness between PPUs and thrusters

- 4 nominal and 4 redundant SVT01 cold gas thruster from NAMMO UK.

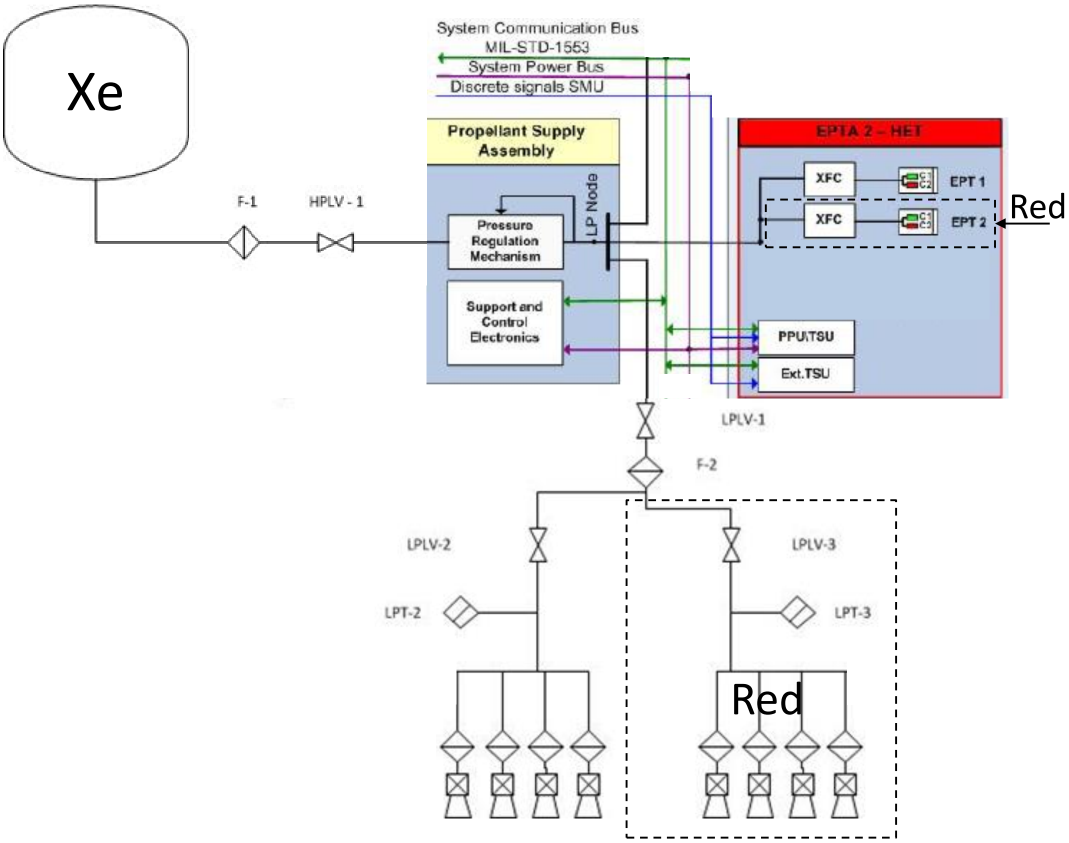

Figure 8-3: PPS1350-E subsystem as baseline for transfer to NEO

In order to achieve the thrust range and life time capability required by the SPP mission a system of 2 thrusters is baselined, one nominal and one redundant. Each thruster and Xenon Flow Controller (XFC) is commanded / controlled by a Power Processing Unit PPU (Figure 8-4), conceived as the only electrical interface to the satellite avionics. The Thales PPU Mk2 is a higher-power development of the flight-proven Mk1 of which more than 20 flight models had been delivered and flown RD[42]. The Mk2 has been qualified, and several flight models are in production for upcoming telecommunications satellites.

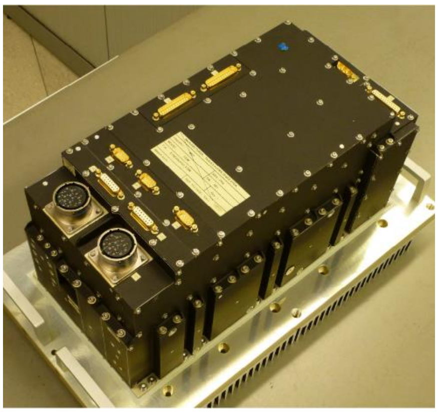

Figure 8-4: PPU Mk2

Both thrusters are mounted on one pointing mechanism typically used NSSK on telecom satellites. This mechanism is used mainly to correct the thrust vector due to CoG evolution over the mission life.

Xenon is contained in an Orbital ATK tank 80458-1, capable of storing up to 223 kg of xenon at 186 bars.

With the empirical performance functions, MA derived a propellant mass for the main transfer of 204 kg xenon with an average specific impulse of 1792 s. For pointing, AOGNC requires 0.5 m/s + 100% margin that is handled by the cold gas system with a specific impulse of 25 s (w/ Xe), thus, another 3.7 kg of propellant. Finally, 17.7 kg are added to reflect a 10% margin on the calculated change in velocity with the aforementioned average specific impulse. Therefore, **225.4 kg** of xenon propellant are to be expected for this example mission (excluding residuals). Since the propellant amount is highly sensitive to the initial wet mass, iterations between SYS and MA are typically required to yield a final propellant estimation. In the framework of the SPP study, this was, however, neither possible nor required, so the values presented here are to be considered examples for such a satellite sizing. Therefore, the tank size has not been modified to account for the exceeding propellant mass since it is expected to change with further iterations.

8.6 Baseline Design SS

N/A

8.7 List of Equipment MC

Table 8-1 reports the complete list of equipment and the estimated dry mass budget of the Electric Propulsion Subsystem.

Equipment	Qty	Unit mass (kg)	Margin (%)	Total mass (kg) w/ margin
PPS 1350-E thruster	2	4.35	10	9.57
Power processing unit	2	11.80	5	24.78
High pressure regulator assembly	1	4.50	5	4.73

Xenon flow controller	2	0.90	5	1.89
Electric filter unit	2	0.70	5	1.47
Pressure regulation electronic card	1	1.30	10	1.43
Thruster pointing mechanism	1	10.60	5	11.13
Tank	1	20.41	5	21.43
Harness/pipes	2	3.00	20	7.20
Cold gas thruster assembly	1	3.05	10	3.35
TOTAL dry mass of the subsystem (kg)				86.97

Table 8-1: Electric Propulsion Subsystem estimated mass budget

8.8 List of Equipment SS

N/A

8.9 Option MC

The following summarises potential alternative EP subsystems proposed for further future assessment:

- High-power Gridded Ion Engine T6 operated at a lower power level
  - Developed and qualified by QinetiQ (UK)
  - Heritage: BepiColombo (to be launched)
  - Low-power operation of T6 to reduce S/A mass to be investigated
  - Higher propellant mass, longer transfer due to reduced thrust and specific impulse levels
- 2N+1R medium-power Gridded Ion Engine T5
  - Developed and qualified by QinetiQ (UK)
  - Not assessed, but likely longer transfer
  - Optimised for PPU input power < 1 kW
  - Heritage: GOCE
  - If power allows, thrusters can be fired in parallel
  - Thrust level between 0.5 and 25 mN
  - I_{sp} > 3000 sec
  - Lifetime: 3MNs per engine
- Similar in performance, a 1N+1R high-power Gridded Ion Engine RIT2X could replace the T6 architecture.
  - Developed and under qualification by Ariane Group (DE)
  - Nominal power between 2.3 and 5 kW
  - Thrust levels between 80 and 205 mN
  - I_{sp} > 3800 sec

- Estimated lifetime >10 MNs
- Similar in performance, the T5 ion engine could be replaced by a 2N+1R RIT 10 EVO ion engine architecture
  - Developed and qualified by Ariane Group (DE)
  - Not assessed, but likely longer transfer
  - Thrust level between 5 and 25 mN
  - I_{sp} > 3400 s
  - Lifetime: 1.1 MNs per engine

8.10 Options SS

N/A

8.11 Sensitivity Analysis for MC: What if?

- What if the available power level changes?
  - If the available power at target is lower than 1.5 kW, a lower thrust is available and the transfer time will increase. For this, more propellant will be needed and thus the wet mass will be higher.
  - If the available power at target is higher than 1.5 kW, a higher Isp will be available and it follows that less propellant will be needed. Thus, there will be a higher thrust and the transfer time will decrease.
- What if the initial wet mass changes? (e.g. different launcher, different starting point, kickstage option, smaller or larger satellite in general)
  - If the wet mass is higher, the propellant storage architecture needs to be adjusted to reflect the increased need in propellant as the current projected propellant consumption is at the limit of the tank capacity
  - If the wet mass is lower, the propellant tank is larger than necessary, but this will be no issue per se. An adjustment of the size to a smaller capacity can be easily achieved with the propellant tank families of the supplier, and subsystem mass can be saved consequently.

8.12 Sensitivity Analysis for SS: What if?

N/A

8.13 Sensitivity to Target: What if Phobos?

Compared to a NEO target, the transfer to Phobos follows similar design decisions for the EPROP subsystem. Resulting from the increased change in velocity required to perform the transfer, the necessary propellant mass will increase which will result in potential changes to the propellant management system (see 8.11). To process the increased propellant mass, additional thrusters might be required to fulfil the total impulse requirement. Therefore, the trade-off (see 8.3.2) needs to be revisited to verify whether the current baseline remains the preferable option or whether an ion engine architecture becomes superior for this scenario.

8.14 Architecture Sensitivity Lander

Since the main functionality task of the EPROP subsystem is the transfer to the target, no substantial impact on the Mother S/C is to be observed if one of the smallsatellites is replaced by a lander. However, since the lander release and post-release operation of the Mother S/C might require additional propulsive tasks by GNC, a revisit to the demanded propellant mass and/or thruster performance is recommended.

8.15 Major Design Constraints: CAUTIONS!

- General
  - Propellant masses require additional iterations w/ MA and SYS, and optimised trajectories – potential change in propellant tank design
  - Lifetime of thrusters compared to calculated prop masses – additional thrusters potentially required when increasing propellant amount
  - No European supplier for variety of OTS xenon tanks in the considered size
- Hall effect thruster
  - PPS1350-G flight-tested for transfer to Moon (SMART-1) and used for stationkeeping on telecom satellites, however, PPS1350-E not yet fully qualified, and not assessed for 1.1 AU.

8.16 Technology Requirements

Equipment and Text Reference	Technology	Suppliers and TRL Level	Technology from Non-Space Sectors	Additional Information
PPS1350-E	Hall effect thruster	Safran-Snecma	NO	Ongoing qualification for stationkeeping purposes. To be assessed whether delta qualification required for transfer to NEO
PPU Mk2	PPU	TAS Belgium	NO	To be assessed whether delta qualification required
Xenon tanks	Tank	MT Aerospace	NO	Potential European supplier; preliminary design exists
HPR & FCU	Propellant management	AST / Smallspace	NO	Low-mass developments alternative to baseline equipment

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9 GNC

The GNC system main functions are to provide the required orientation of the spacecraft during the entire mission and to estimate on-board the relative position and velocity of the spacecraft with respect to the asteroid in order to prevent collision and to point properly the navigation and/or science instruments. Note that this high level functionalities apply both to the mothercraft and the smallsats. Given the mission requirements, the mothercraft and the smallsatellites are 3-axis stabilised platforms.

The GNC differences between mothercraft and smallsats are significant and therefore the GNC systems will be analysed separately. Some commonalities will be highlighted in order to reduce the technology development cost.

9.1 Requirements and Design Drivers MC

For the MC the main design drivers are listed below.

- Multiple years of interplanetary EP transfer.
  Certain level of autonomy would be desirable in order to reduce the number of ground contacts (on-board estimate of thruster performance, on-board monitor of trajectory evolution)
- Arrival to a faint target.
  ROSETTA experience is applicable and re-use of procedures is advisable. EP transfers should produce arrival trajectory with low relative velocity and phase angle close to 90deg. Small, faint targets may pose some additional detectability problems, a good practice is to check the ground observability windows before arrival in order to reduce the ephemerides uncertainty (easier detection of target).
- Stay in safe orbits close to target
  To reduce complexity of proximity operations a good practice is to maintain the SC in passively safe trajectories. The preferred solution is fly hyperbolic arcs with a safety margin in the pericenter radius and velocity.

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
GNC-010	The GNC sub-system shall provide hardware and associated on-board software to acquire, control and measure the required spacecraft attitude during all phases of the mission, and to control and monitor all the necessary Delta-V for the complete mission according to the specified system requirements.	
GNC-020	The MC spacecraft shall be 3-axis stabilised.	
GNC-030	For all mission phases, the MC spacecraft shall have the autonomous capability to maintain the required attitude and to perform attitude manoeuvres, including when contact with ground is not available or ground response time is inadequate.	
GNC-040	The GNC sub-system shall be able to maintain, during Safe mode, the solar arrays pointing to the Sun using a minimum of	

	the on-board resources ensuring power generation and ground communication.	
GNC-050	The AOGNC shall detumble the MC after launcher separation in less than 20 minutes, for a worst-case tip-off rate of 5 deg/sec along any spacecraft axis.	
GNC-060	During thrust arcs performed with electrical propulsion, the contribution of the GNC to the APE of the thrust vector shall not exceed 1.5 deg (TBC) half cone (95% confidence level).	
GNC-070	During communication windows, the contribution of the GNC to the APE of the HGA shall not exceed 0.5 deg half cone (95% confidence level).	
GNC-080	In asteroid proximity, the MC position relative to the asteroid shall be known on-board to an accuracy better than 20% of the distance to the asteroid, with a 99.7% confidence level in every axis (each axis independent of the rest).	
GNC-090	As a goal, the wheel offloading should not take place more often than once per week during the close proximity operation phase.	

9.2 Requirements and Design Drivers SS

For the SS the main design drivers are listed below.

- Minimum distance to surface 5 km.
  This needs to be compatible with the duration of the arcs, the gravity parameter knowledge and the performances of the manoeuvre execution
- A priori knowledge ~100 m (distance to surface) at pericenter.
  This cannot be achieved with the low cost approach but is feasible with the on-board navigation. Some more analysis shall be done to understand the implications in the instrument operations (interaction between on-board GNC and the payload calibration and/or operation)
- Limited delta-V capability (10 m/s).

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
GNC SS-010	The GNC sub-system shall provide hardware and associated on-board software to acquire, control and measure the required spacecraft attitude during all phases of the mission, and to control and monitor all the necessary Delta-V for the complete mission according to the specified system requirements.	
GNC SS-020	The SS spacecraft shall be 3-axis stabilised.	
GNC SS-030	For all mission phases, the SS spacecraft shall have the autonomous capability to maintain the required attitude and to perform attitude manoeuvres, including when contact with ground is not available or ground response time is inadequate.	
GNC SS-	The GNC sub-system shall be able to maintain, during Safe	

040	mode, the solar arrays pointing to the Sun using a minimum of the on-board resources ensuring power generation and communication with the MC (no direct to Earth communication needed).	
GNC SS-050	The AOGNC shall detumble the SS spacecraft after separation from MC in less than 10 minutes, for a worst-case tip-off rate of 15 deg/sec along any spacecraft axis.	
GNC SS-060	In asteroid proximity, the SS position relative to the asteroid shall be known on-board to an accuracy better than 20% of the distance to the asteroid, with a 99.7% confidence level in every axis (each axis independent of the rest).	
GNC SS-070	As a goal, the wheel offloading should not take place more often than once per 3 days during the close proximity operation phase. *Note: to avoid perturbing the hyperbolic arcs and combine the wheels off-loading with the delta-V manoeuvres.*	
GNC SS-080	The APE during science operations and optical navigation imaging shall be better than 0.5 deg with 95% probability and 90% confidence level. *Note: the objective is to have the asteroid in the FoV of the NAVCAM and the science sensors*	
GNC SS-090	The APE during science operations and optical navigation imaging shall be better than 0.5 pixel over 0.1 s with 95% probability and 90% confidence level.	

9.3 Assumptions and Trade offs MC

9.3.1 Assumptions

Assumptions	
1	EP gimbal to reduce CP propellant during cruise To reduce the propellant mass required for angular momentum management and torque perturbation compensation, a gimbal of EP is assumed (Isp of cold gas systems is very low and would lead to an unacceptable propellant mass for long interplanetary transfers). This will cancel the thrust misalignment during EP thrust arcs (pitch and yaw). Depending on the number of EP thrusters and their location, roll control might also be possible. For very short interplanetary transfers the benefits of the gimbal system must be traded against RW+CP system.
2	CP used during proximity operations. The passively safe trajectories require very small delta-V. The total delta-V is very low and the chemical propellant mass is small. The delta-V at the intersection of the hyperbolas shall be split to ensure that interruption of these manoeuvres will always result in a hyperbola of higher energy (the risk of collision is always lower than in the final trajectory). EP might be used for proximity operations but then the margins on the trajectories and the execution of delta-V shall be reassessed considering slews,

Assumptions	
	power, thruster uncertainties, wheels off-loading…
3	Prox. Ops. Hyperbolic arcs with Vpericentre > 1.4 Vescape. The margin of 40% is based on ROSETTA experience with a reduction due to the lack of outgassing affecting the trajectory and the navigation sensors (mainly the STR)
4	Far distance to avoid perturbations & simplify operations The minimum distance to the target is defined to be able to execute one manoeuvre per week with a safe trajectory and considering higher uncertainties than in ROSETTA mission due to the simplified flight dynamics.
5	On-board autonomy only for collision avoidance and NAVCAM pointing To reduce cost only these functions are performed on-board (similar modes were implemented in ROSETTA for camera pointing during asteroid fly-bys)
6	SS deployment not changing baseline operations (no dedicated flyby) A major driver of GNC and ground operations was found in AIM to be the deployment of passive lander (MASCOT-2) a la Philae. Therefore, the deployment of the SS will be done in the final orbit of the MC for its proximity operations.
7	Link with SS via omnidirectional antenna No dedicated slews to point inter-satellite antennas towards the SS.

9.3.2 Trade Offs

One trade-off in case a cost reduction is desired is the use of STR instead of NAVCAM for optical navigation purposes. The following must be analysed:

- The approach phase (detectability of the target vs performances), Hayabusa used the STR for approach phase, ROSETTA used the NAVCAM.
- The LoS measurements computation and performances during prox. Ops. (see next chapter for description of navigation algorithm). The STR shall be able to provide a full raw image (snapshot) to be processed on-ground or on-board (during prox. Ops.)

Since the MC does not have stringent pointing stability requirements, another trade-off that might be done in some missions is the use of RW vs CP (also considering the gimbal of the EP thruster). This trade-off was mentioned in the assumptions and should include the complexity of the operations and on-board system.

9.4 Assumptions and Trade offs SS

9.4.1 Assumptions

Assumptions	
1	SS inserts itself in operational orbit after deployment from MC The SS shall be able to execute delta-V to inject in the operational hyperbolas
2	Assumption 2
3	Same safety margins for prox. Ops. Orbits as MC Passively safe hyperbolas with the same constraints as MC (but some parameters

	are different due to different platform)
4	At most 2 delta-V per week (3-4-3 day trajectory arcs pattern) In line with ESOC low cost operations strategy.
5	Maintain target in FoV of imagers/spectrometers Pointing accuracy not very demanding and compatible with navigation requirements (obtain images of the target for orbit determination).
6	Pointing stability not driving design RPE similar to MARGO study (10 arcsec over 100 ms), no perturbation during the ballistic flight (RW desaturation performed simultaneously to the delta-V for arc insertion)

9.4.2 Trade Offs

The selection of the science orbit is based on passively safe hyperbolas as in the MC, but considering the science requirements. The insertion in closed stable orbits (e.g. Self-Stabilised Terminator Orbits (SSTO)) might be analysed considering the asteroid size (gravity) and the solar radiation pressure. Regions of stable SSTO can exist that satisfy the science objectives.

The delta-V budget depends on the distance at pericenter and also on the frequency of manoeuvres (duration of each hyperbolic arc). An example of a potential trajectory from AIM is presented in Figure 9-1. In this case in each ground cycle 2 delta-V are computed and executed. The design proposed hereafter includes the autonomous functions to perform such manoeuvres, however additional analyses would be required to adapt to a specific mission.

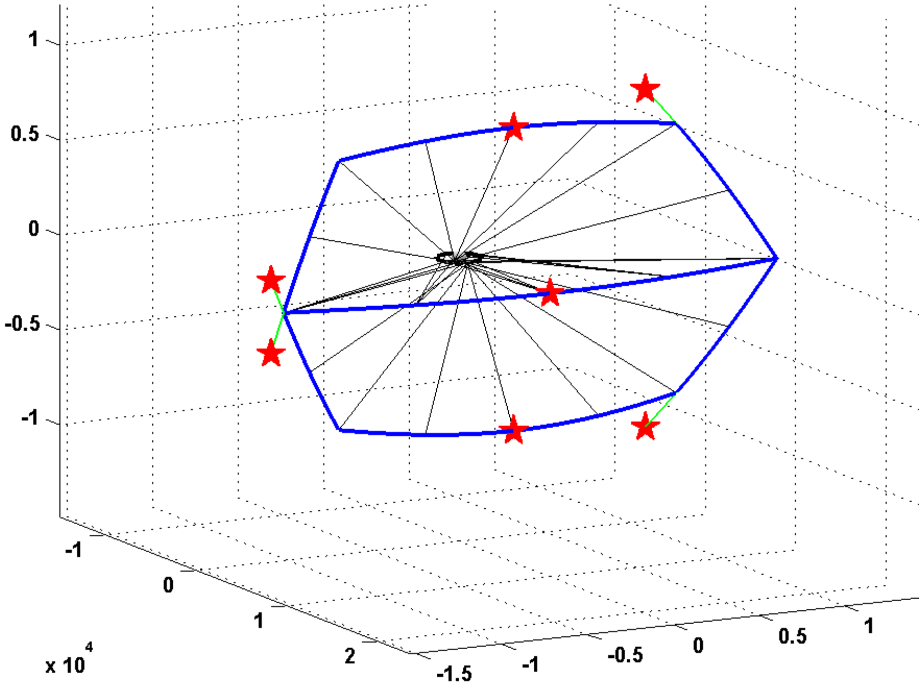


Figure 9-1: Example of trajectories with 3-1-2-1 day arcs (in each operation cycle of 3-4 days there are 2 delta-V executed)

9.5 Baseline Design MC

The MC hyperbolas are designed to minimise slews for ground comms (fixed HGA), imaging asteroid (navigation), and to provide optimum power generation continuously.

In order to simplify the ground operations, 1 delta-V per week is preferred. The minimum pericenter distance compatible with this requirement is analysed. The results presented in Figure 9-2 consider uncertainties compatible with the low cost approach of the platform and the ground operations. The pericenter distance for the MC shall be above 12 km. Each delta-V is around 10 cm/s in total considering the split delta-V. That is the delta-V budget per week in 'orbit' around a 500 m target. The maximum distance to the asteroid reached during this time is slightly above 20 km.

The Wheels Off-Loading is simultaneous to these manoeuvres. The RW capacity must ensure that perturbation torques do not saturate any wheel during that time.

To make the operations as simple as possible, the hyperbola can lay fixed with the pericenter in the line between the Earth and the asteroid (like Hayabusa), that minimises the amplitude of the slews to point the HGA to Earth or to take pictures of the asteroid for navigation. The axis of the solar arrays should be kept as perpendicular as possible to the Sun-asteroid-Earth plane in order to maximise the power generation.

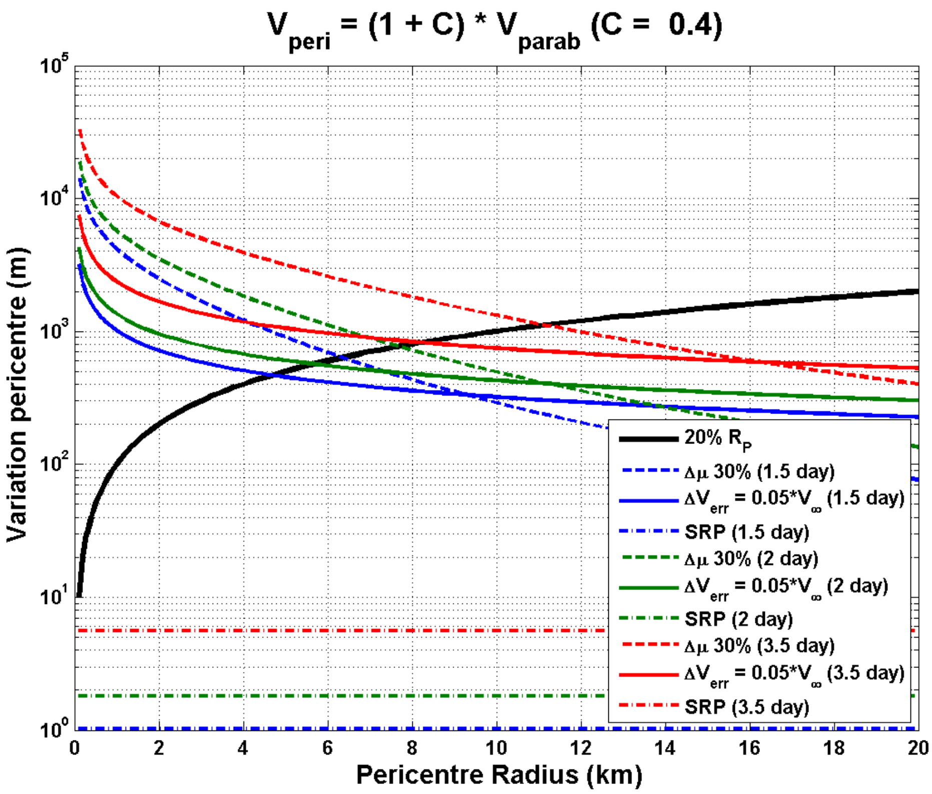


Figure 9-2: Safe pericenter radius considering major perturbations in the hyperbola

The basic GNC architecture is depicted in Figure 9-3. This has been optimised for low-cost considering the ground and space segments (from AIM studies). The share of responsibilities are:

- Ground-based manoeuvre plan (translational guidance)
  - EP and proximity operations
  - Pre-planned collision avoidance manoeuvres table
- On-board relative navigation for attitude pointing during prox. Ops.
  - Compensate trajectory deviations to ensure proper imaging and monitor collision risk
- On-board attitude determination and control (standard platform services).

It is important to highlight that the prox. Ops. Hyperbolas are ballistic (no thruster activation). Therefore there is no need of ground or autonomous orbit control (in case of safe mode triggering, the trajectories are intrinsically safe and no specific autonomous measures are needed).

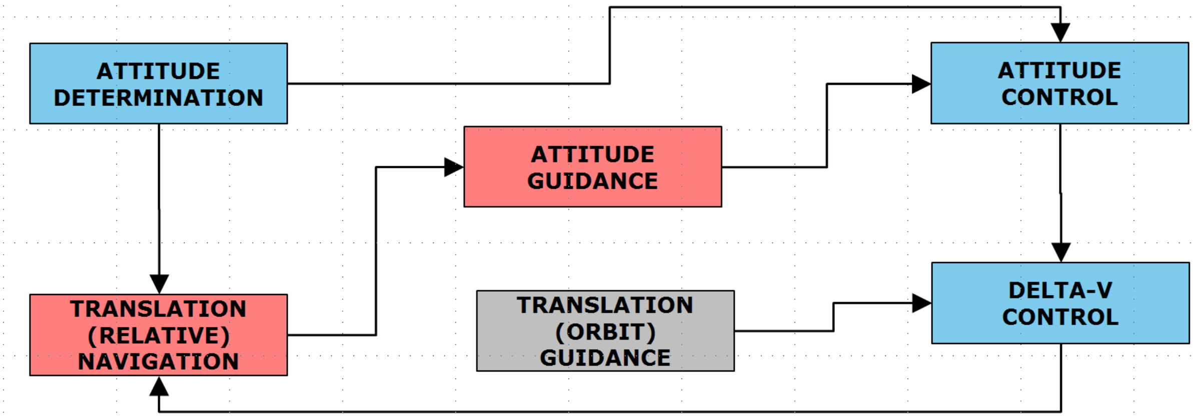


Figure 9-3: GNC architecture

The proposed algorithms to implement the relative navigation used for pointing and collision risk assessment are based on the 'low-cost' approach analysed during AIM study. It is based on the use of the NAVCAM images for vision-based navigation with two main components:

- 'Simple' centroiding image processing algorithm (see Figure 9-4)
- Unscented Kalman Filter for data fusion and uncertain parameter estimation (gravity, shape, delta-V).

The typical on-board knowledge of the relative position is below 100 m, usually 10 times better than the a priori ground prediction error.

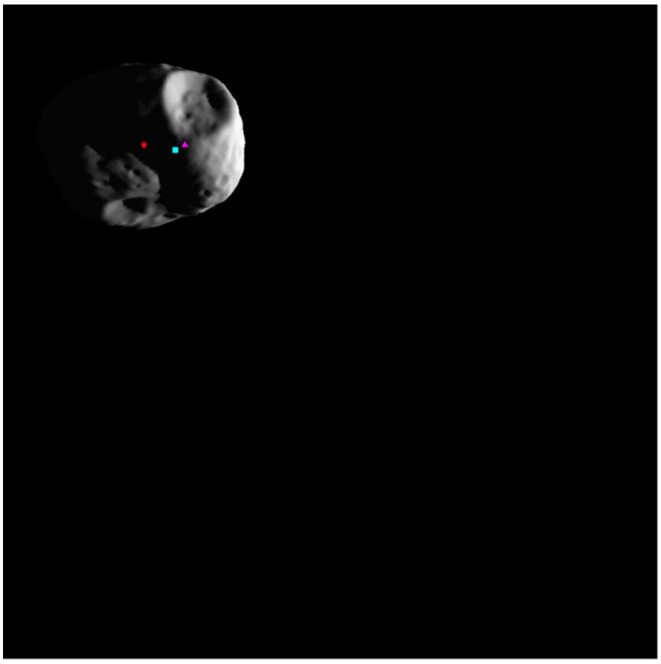


Figure 9-4: Example of IP and navigation results

9.6 Baseline Design SS

The same assumption as for the MC in prox. Ops. Are considered here. In this case however, in order to reach the low pericenter altitude, a 4-3-4-3 day hyperbolic arcs are required. The baseline trajectories are presented in Figure 9-5 and Figure 9-6. The main characteristics are listed below:

- 3-day arc pericenter @ 5 km
- 38 cm/s per week (4.4 m/s for 6 months operations).

The distance and phase angle are depicted in the figures below. It must be noted that the phase angle can be changed (in this case the pericenter of the 3-day arc is in the Sun-asteroid line, phase 0deg). The location of the points can be rotated wrt the Sun-asteroid line in order to observe the surface with different illumination conditions.

It must be noted that the accommodation of the payload shall be compatible with thermal requirements. For instance in the plots below, the payload is pointing to the asteroid and might interfere with the accommodation of the radiators.

If the trajectories are rotated 90 deg (the pericenter of the 3-day arc is now at 90deg phase angle), then the accommodation of the payload can be in a different side than the cold side of the SC. It is assumed that the solar arrays axis can always be almost perpendicular to the Sun-SC line in order to maximise the power generation.

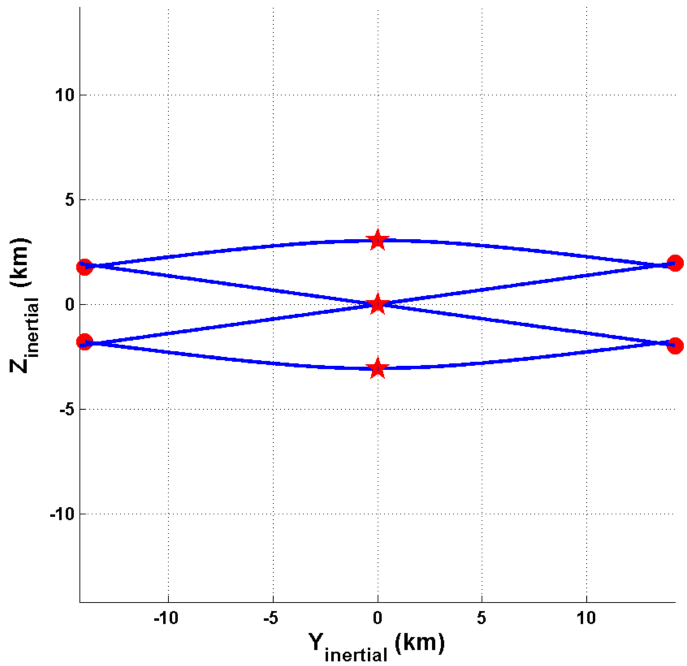


Figure 9-5: Possible science trajectories viewed from the Sun direction towards the asteroid (Z axis points in the direction of the asteroid orbital momentum)

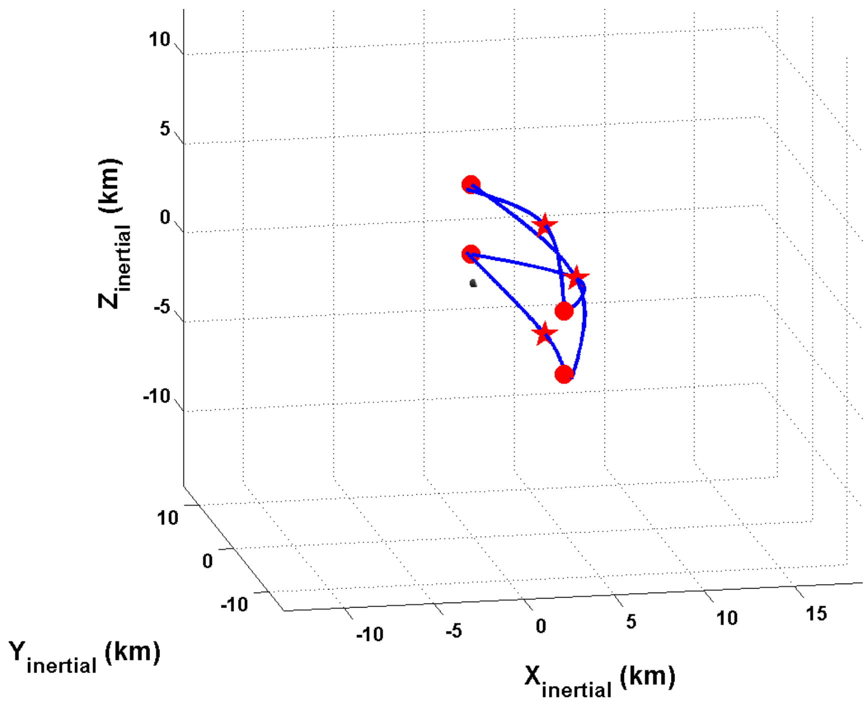


Figure 9-6: Possible science trajectories (X axis points in the direction to the Sun)

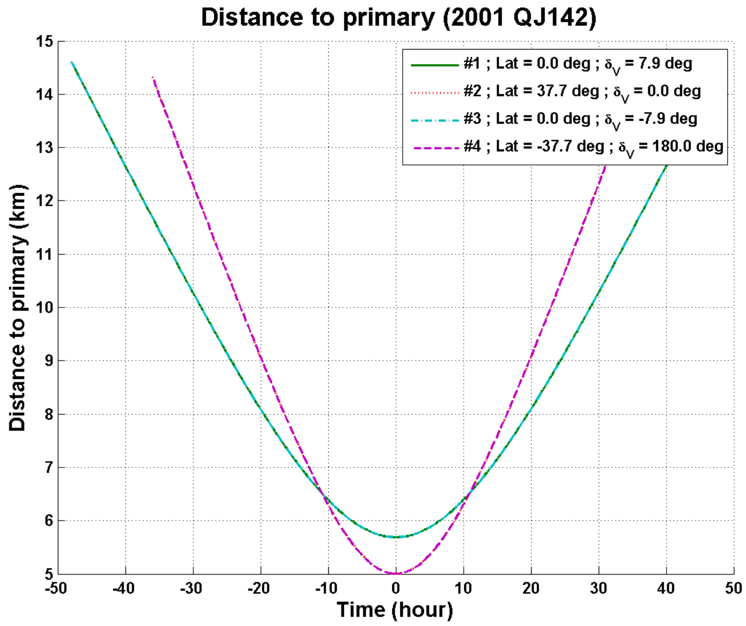


Figure 9-7: Distance to asteroid for SS (time origin is the pericenter of each hyperbola)

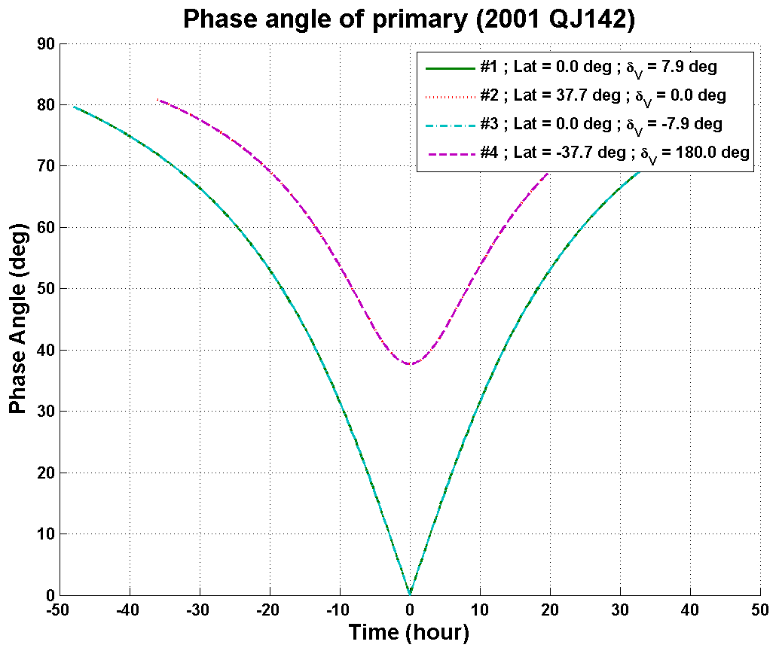


Figure 9-8: Sun-asteroid-SC angle for SS (time origin is the pericenter of each hyperbola)

9.7 List of Equipment MC

A list of space-qualified, off-the-shelf equipment suitable for a low-cost mission is provided based on previous missions like PROBA-3. It must be assessed for each particular mission, whether these equipment fulfils all particular mission requirements.

Quantity	GNC Equipment	Unit Weight	Total Weight	With 5% margin
4	Reaction Wheels	1.44 kg	5.76 kg	6.05 kg
1	Star Tracker and IMU	1.25 kg	1.25 kg	1.31 kg
1	Visual Navigation Camera	2.4 kg	2.4 kg	2.52 kg
6	Sun Sensors	0.05 kg	0.3 kg	0.315 kg
	Total		9.71 kg	10.12 kg

Table 9-1: Mass Budget for MC

9.7.1 Reaction Wheels

The selected RW are MSCI MicroWheel in tetrahedral configuration for redundancy. These RW were flown in PROBA-2. The main characteristics of the wheels are:

- Maximum Torque: 0.03Nm
- Momentum Storage: 1.1Nms
- RW Mass: 1.5kg x 4
- RW Power: 9W x 4

Should higher capacity wheels be required, an alternative could be AFW 250. These wheels have lower TRL (TRL6). The main characteristics are:

- Maximum Torque: 0.1Nm
- Momentum Storage: 4Nms
- RW Mass: 2.7kg x 4
- RW Power: 24W x 4

9.7.2 Star Tracker and IMU

The selected STR is DTU μASC (Advanced Stellar Compass) which has been flown on missions including deep-space. The μASC is composed of the following elements:

- Two Camera Head Units (CHU): this elements comprises the optics and the detector (0.4Mpixels)
- Redundant Digital Processing Unit (DPU)
- Two baffles: this is a passive element intended to reduce straylight from Earth/sun and asteroid.

The main characteristics of the micro-ASC are:

- DPU: 0.57kg, CHU: 0.30kg
- 5.2W (total)
- DPU: 124x100x41.5mm^3 , CHU: 50x50x57.5mm^3
- NEA: 1arcsec/8arcsec RMS

The IMU that is selected as baseline for the mission is the μMIRU from DTU. This unit is a MEMS-based IMU with moderate performance and has the strong advantage that it is integrated in the star tracker's CHU for limited additional mass and power consumption. Moreover, no additional data/power interfaces are required:

- CHU+40g
- CHU+130mW
- TRL6
- Accelerometer
  - Resolution: 2.77e-4 g
  - Random walk: <0.053m/s/√hr
  - Bias stability (@300s): 0.16mm/s²
- Gyros
  - Resolution: 8.75e-3 °/s
  - Random walk: 1.16°/√hr
  - Bias stability: 6.2°/hr

Figure 9-9: DTU µASC star tracker (left: CHU – Camera Head Unit, middle: DPU – Digital Processing Unit, right: baffle)

9.7.3 Visual Navigation Camera

The baseline visual camera for the AIM S/C is based on the DVS (Digital Video System) camera from TSD (Techno System Developments/Italy). This camera was used for the PRISMA mission. The PRISMA DVS offers a suitable detector, but the FoV is 28° and thus needs to be adapted to the needs of the MC (5 deg):

- 2.4 kg
- 13 W (imaging)
- 140x130x160mm³ (TBC)
- 2048x2048 detector
- 5° Field-of-View
- TRL6

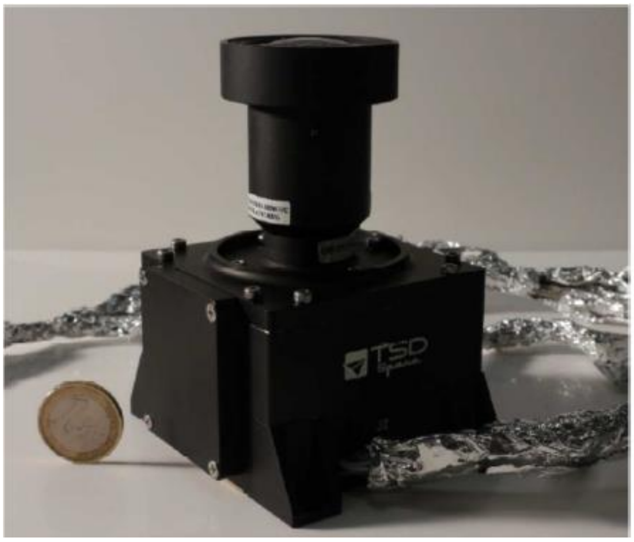


Figure 9-10: DVS camera from TSD

9.7.4 Sun Sensors

The mini-FSS is a fully passive analog Fine Sun Sensor, based on a quadrant photo detector device, with two-axis measurement capability. This sensor is the baseline for instance for ExoMars 2020 mission:

- 50 g
- FoV 128x128 deg
- Without any ground calibration, accuracy < 1.5° (3 s) in the whole FoV.
- With on-board implementation of a look-up table, accuracy <0.5° (3 s) per axis.

9.8 List of Equipment SS

A preliminary selection of equipment has been carried out, which allowed identifying suitable COTS solutions for all the sensors and actuators.

Quantity	GNC Equipment	Unit Weight	Total Weight	Margin	With margin
1	IMU	0.02 kg	0.02 kg	5%	0.021 kg
6	Sun Sensors	0.002 kg	0.012 kg	5%	0.013 kg
1	Altimeter	0.034 kg	0.034 kg	5%	0.036 kg
1	Optical NavCam	0.059 kg	0.059 kg	5%	0.062 kg
3	Reaction Wheels	0.18 kg	0.54 kg	10 %	0.59 kg
	Total		0.665 kg		0.725 kg

Table 9-2: Mass Budget for SS

Details of the selected equipment are provided in the following subsections.

9.8.1 Inertial Measurement Unit

A possible IMU is the US-built MS-IMU/3020 by Memsense, shown in Figure 9-11.

Figure 9-11: Memsense MS-IMU/3020

The IMU has the following performance characteristics:

- Bias Instability: 0.84 °/h
- Angle random walk: 0.29 °/√h

An alternative could be a European IMU by Sensonor, with the former being chosen as baseline, as it weighs less and it consumes less power. The Sensonor sensor, however, provides better performance.

A comparison of the specifications is provided in Table 9-3.

	Memsense MS-IMU/3020	Sensonor STIM-300
ARW	0.29 deg/sqrt(h)	0.15 deg/sqrt(h)
Bias instabiity	0.84 deg/h	0.5 deg/h
Mass	20 g	55 g
Power	0.5 W	2 W
Dimension	28x28x10 mm	44.8x38.6x21.5 mm

Table 9-3: Comparison of baseline and option IMU

Note that the European IMU benefits from flight heritage of similar products by the same vendor (NASA AeroCube-4 in 2012) and has been selected for NASA Raven and NEO-scout missions. As a consequence, the TRL of the European IMU (TRL 7) is higher than that of the American IMU (TRL 6).

9.8.2 Sun Sensors

Potential Sun sensors are the Bison-64 by Lens R&D, shown in Figure 9-12.

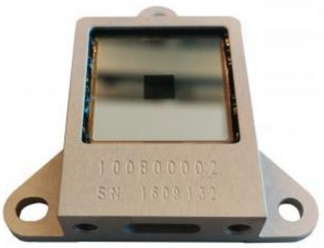


Figure 9-12 : LENS R&D Bison-64

The Sun sensor has the following performance characteristics:

- Accuracy between 0.5 deg and 3.5 deg
- FoV: 64 degrees

These Sun Sensors have been subject to extensive qualification tests and possibly only minor delta-qualification would be needed for interplanetary mission.

An alternative sun sensors are Hyperion SS200 which are much lighter (2 grams). The drawback of these sun sensors is that the TRL is lower and there might need extensive qualification campaign to meet the environmental conditions of the interplanetary mission.

- 2 grams
- 2.5 mW – 40 mW
- 20 x 15 x 6 mm
- FoV 110 deg

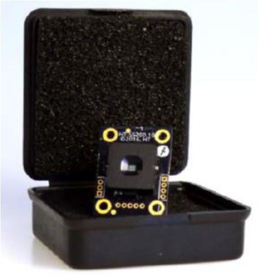


Figure 9-13: Hyperion Technologies SS200 Sun Sensor

9.8.3 Altimeter

The selected altimeter is the DLEM laser range finder by Jenoptic, shown in Figure 9-14.

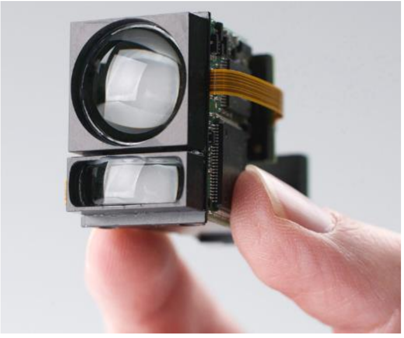


Figure 9-14: Jenoptic DLEM laser rangefinder

The altimeter has the following performance characteristics:

- Total measuring range: 0 m to 5000 m
- Accuracy: better than 1 m.

9.8.4 Optical Navigation Camera

The IM200 relative navigation imager by Hyperion Technologies has the following specifications:

- Mass: 59 g
- Power Consumption: 700 mW
- Dimensions: 29 x 29 x 70.7 mm
- Pixels: 4 MP
- Focal length: 16 mm (F1.2) or 50 mm (F2.0).

Figure 9-15 : Hyperion Technologies IM200

9.8.5 Reaction Wheels

A possible RW for Pico and Small Satellites is RW 1 from Astro-und Feinwerktechnik Adlershof GmbH. There are two different rotation masses available that provides different performances.

- Angular momentum @ 8000 rpm: Type A 5.8.10-4 Nms ; Type B 1.0.10-4 Nms
- Max. rotation speed: 16.000 rpm
- Nominal torque Type A 23.10-6 Nm ; Type B 4.10-6 Nm
- Mass Type A 20 g ; Type B 12 g
- Power Max 0.72 W.

Figure 9-16: Astrofein RW1

9.9 Sensitivity Analysis for MC: What if?

9.9.1 Impact of Change Target Size

The MC should not enter into low altitude orbit since it only needs to deploy the SS and relay data from SS to Earth. However, depending on the target size the distances for communication with SS might be too large and insertion on stable orbit or higher frequency of manoeuvres might be required.

For more information about impact of target size please see next section.

9.10 Sensitivity Analysis for SS: What if?

If target size is larger, and the minimum distance to the surface is maintained at 5 km, then:

- The delta-V is larger (Table 9-4), which implies a larger impact of delta-V error in the trajectory if the duration is maintained, and
- The impact of the gravity parameter uncertainty in the trajectory is also larger.

With the current baseline of low-cost operations and platform, target size above 1 km usually requires insertion into a stable orbit (see SSTO in the figures below) to keep the 5 km minimum distance.

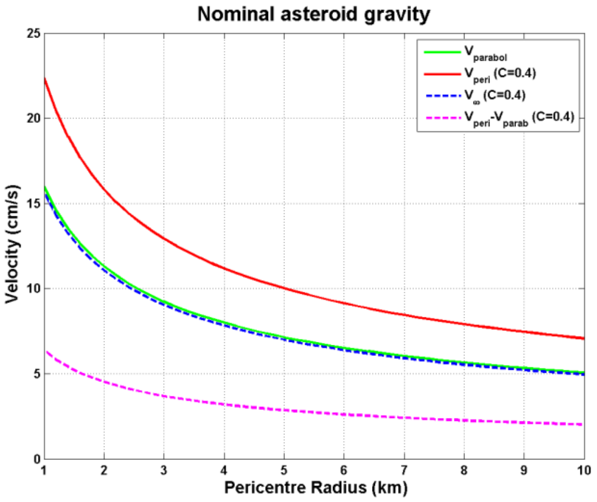


Figure 9-17: Typical velocities for 500 m asteroid

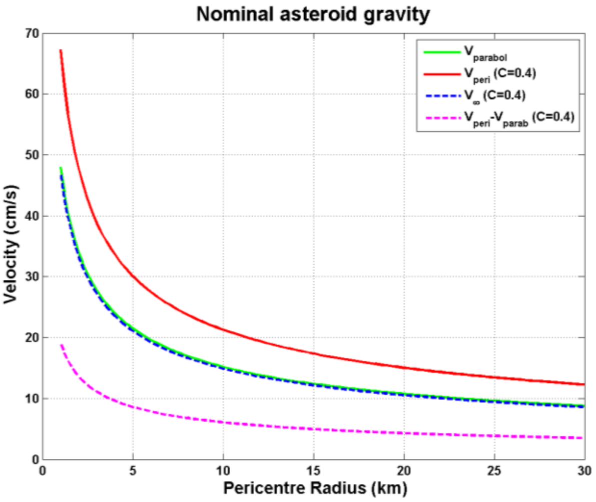


Figure 9-18: Typical velocities for 1 km asteroid

Depending on the mission dynamical parameters characterisation, the minimum and maximum distances of the hyperbolas shall be defined in a case by case basis. In particular, the frequency of delta-V (3-4 day arcs) maybe not compatible with distance requirements. If the minimum distance can be traded, possible alternative trajectories with higher pericenter and lower maximum distance can be found that keeps the 3-4-3-4 day arcs.

There might be possibilities to have shorter duration hyperbolic arcs (1 day) as in AIM (Figure 9-1) but then the operations are a little bit more complicated. In that case, the navigation knowledge maybe not compatible with low-cost approach.

For the 500 m target, if a 1 km altitude fly-by is required with the passive safe constraints that have been described, then:

- 10% uncertainty of gravity knowledge is needed (at the end of the nominal mission this might be feasible since the gravity parameter is a by-product of the orbit determination process)
- The pericenter must be reached 6 hours after the execution of the manoeuvre (pericenter velocity might be higher than 1.4 times the parabolic velocity for such altitude).

This fly-by requires dedicated operations not compatible with the routine 3-4-3-4 day arcs but seems feasible after several months of nominal operations.

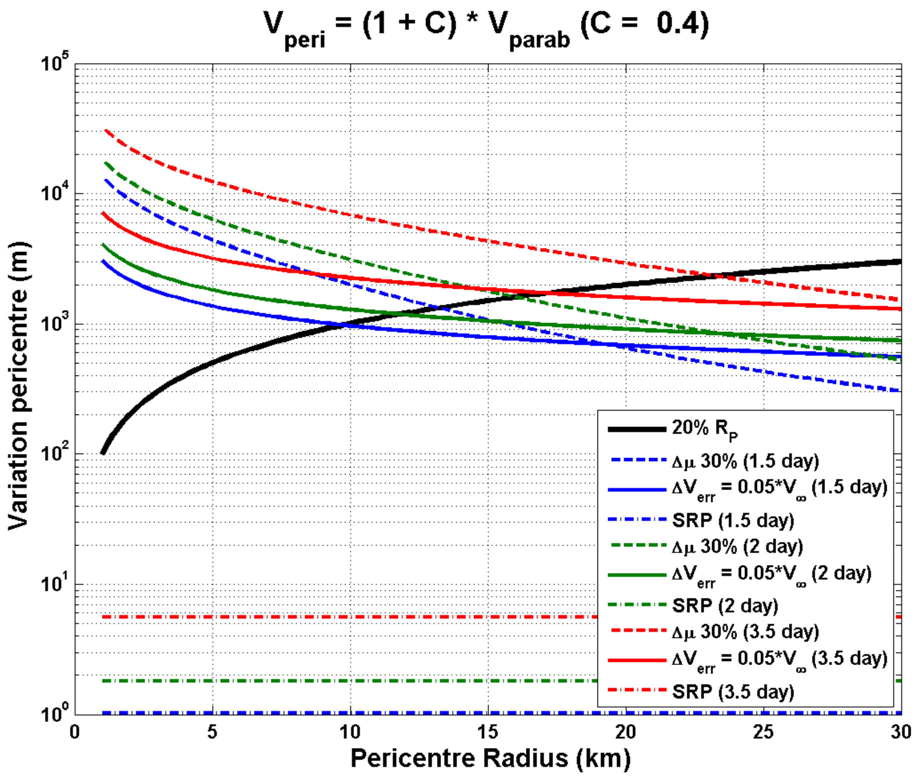


Figure 9-19: Safe pericenter radius considering major perturbations in the hyperbola (500 m target)

Target Size	Delta-V (m/s per week)	Minimum Distance (km)	Maximum Distance (km)
500 m (nominal orbit)	0.3	5	16
500 m (1 km flyby)	0.45	1	22
1 km (nominal orbit)	0.85	5 (TBC)	30

Table 9-4: Delta-V and typical distances in hyperbolic arcs

Stable photo-gravitational orbits or Self-Stabilised Terminator Orbits (SSTO) might be feasible for targets larger than 1 km (see Figure 9-20 and Figure 9-21). These orbits are perpendicular to Sun-asteroid line always. The SSTO orbital plane is slightly displaced wrt the center of the asteroid (a little bit behind the terminator). The stable orbits exist for a certain radius interval (stable means few weeks without manoeuvres after insertion) depending on the distance to the Sun, asteroid gravity, and the spacecraft area and mass.

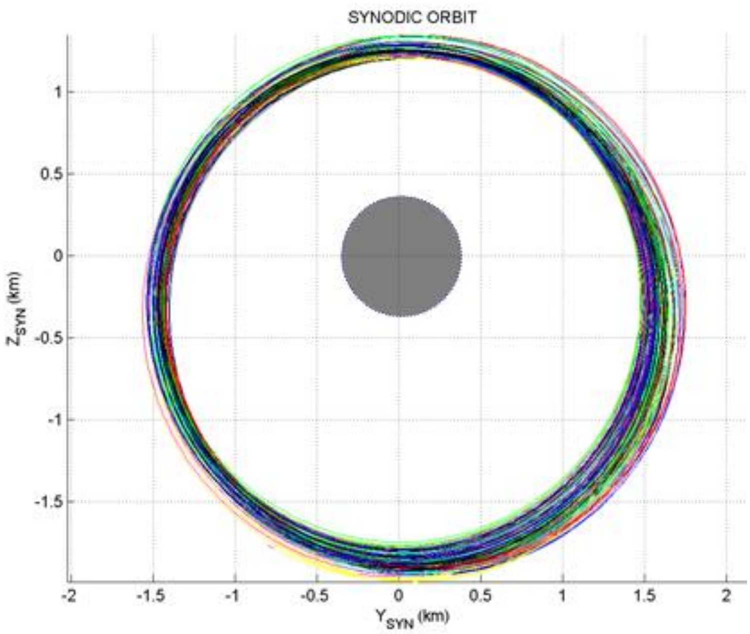


Figure 9-20: SSTO seen from the Sun direction

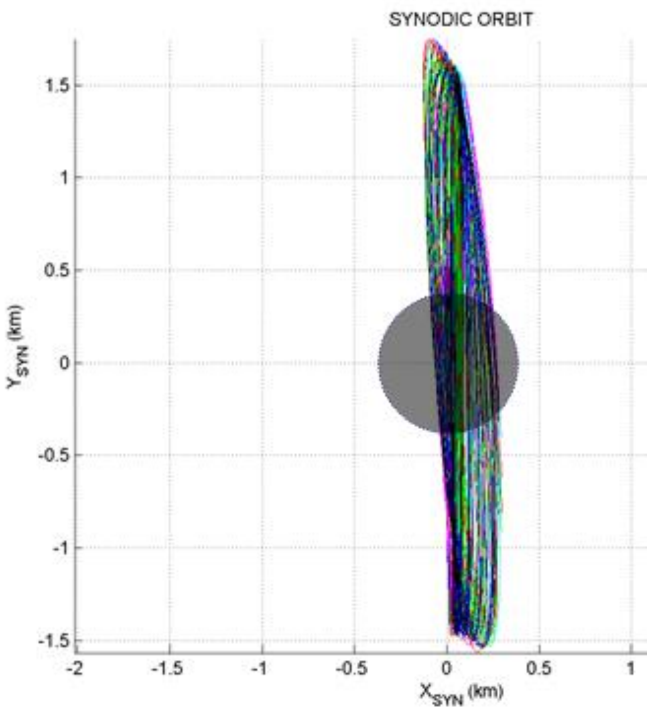


Figure 9-21: SSTO seen from the circumferential direction (aligned with the asteroid velocity in case of circular orbit around the Sun)

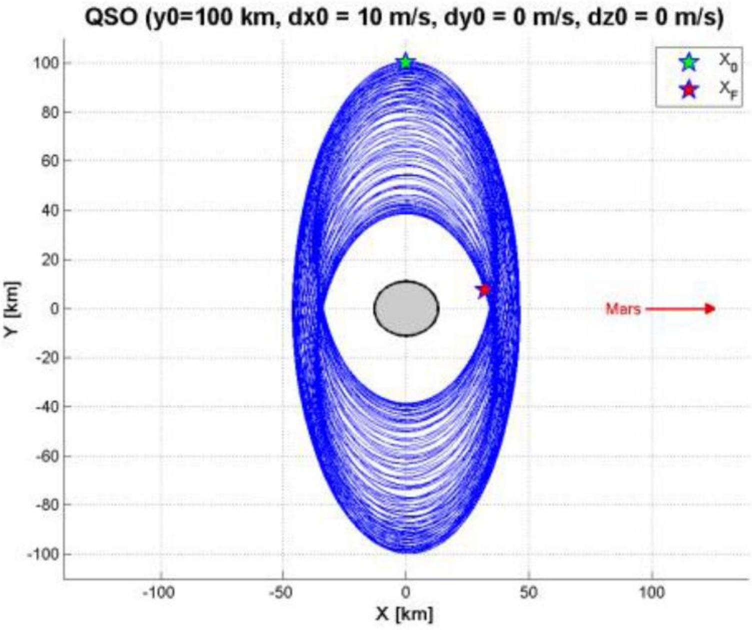
9.11 Sensitivity to Target: What if Phobos?

The Prox. Ops. Around Phobos are completely different to the asteroid case (this was largely investigated in different GNC studies in MREP for Phobos Sample Return mission). In this case the SC orbits around Mars (closed orbits not hyperbolas). These orbits are close to Phobos and requires different orbit insertion and correction strategy. An example is the Quasi-Satellite Orbits (QSO) presented in Figure 9-22. The MC can be inserted in a far QSO from which the SS are released and they insert themselves in lower amplitude QSO.

The main differences in the prox ops are:

- Inter-satellite Distances (MC-SS) increases significantly (~10-100 times)
- Delta-V increases 10-100 times
- Different navigation strategy to observe Mars and Phobos alternatively (NAVCAM with larger FoV needed ~20 deg)

Another strategy could be to place MC in a far QSO or resonant orbit to deploy SS for low-altitude fly-by while staying in this safe orbit. In that case the altitude of the SS flyby might be lower but the duration of the science observations is limited (delta-V limitations might prevent to perform multiple fly-bys with a single SS).



(a) 100 x 50 km planar QSO

Figure 9-22: Example of Phobos observation orbit

The navigation relative Mars and Phobos cannot probably be performed with the same algorithms as in the asteroid case (centroiding). Other more complicated techniques like limb-fitting, shape-matching or landmark matching are needed (Figure 9-23). In addition, the navigation filter shall be changed because the primary is Mars and Phobos is a perturbing body (third body like the Sun) but is the target of the attitude.

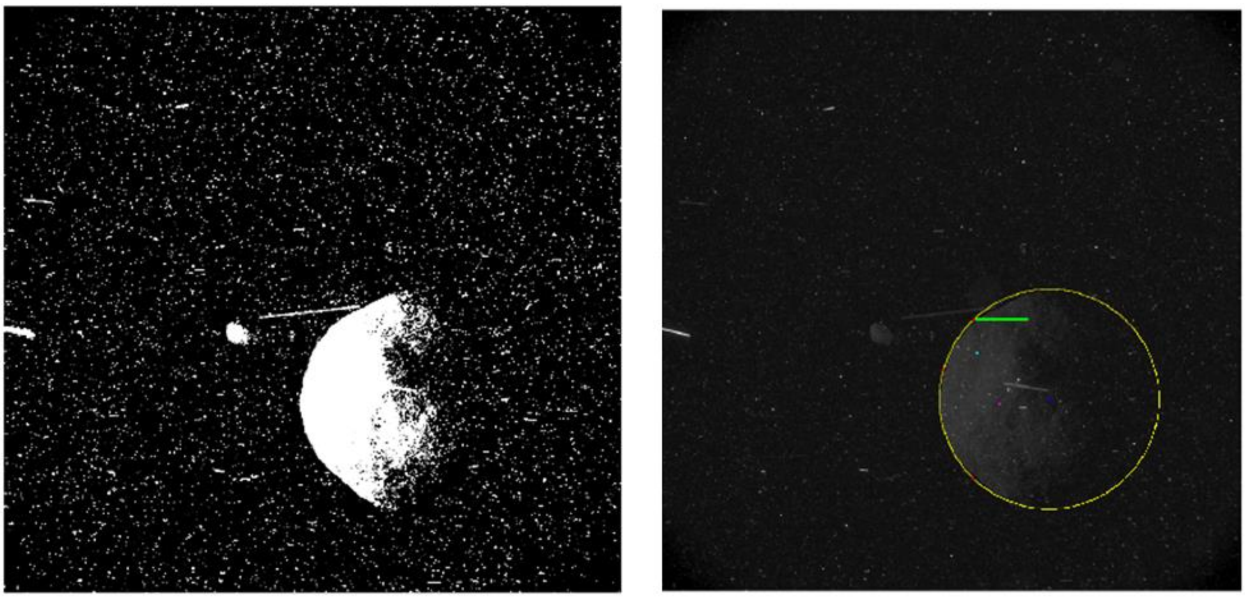


Figure 9-23: Example of limb fitting to derive LOS

9.12 Architecture Sensitivity Lander

There has been several GNC development activities in the frame of Marco Polo to land on NEO asteroids of similar size than the target. The major design drivers are presented below.

- Landing accuracy improves with higher landing velocity. However, the higher the landing velocity, the higher risk of bouncing or tip over. There must be a trade-off between the maximum acceptable touch-down velocity and the landing dispersion (large landing dispersions also introduce landing risk due to terrain hazards).
- It is preferred to design a short descent with few manoeuvres that lands on the illuminated site (30deg Sun phase)
- The autonomous GNC is needed to achieve the landing conditions with the low-cost operation approach for the MC (open-loop performances would not permit landing)
- Additional autonomous navigation mode based on unknown feature tracking is required. The use of the altimeter cancels the drift in transversal position and vertical velocity observed in Marco Polo and AIM due to the scale factor uncertainty (see Figure 9-24). A straight descent in quasi-inertial frame is preferred to maximise the track length of the detected features.
- The rotation period and the size are critical for the touch-down velocity and the navigation performances. For large, fast rotating asteroids the control authority demand might require larger thrusters. An analysis of the centrifugal velocity is needed. There might be limitations in the reachable latitudes (equatorial regions not accessible due to required acceleration larger than available thrust).
- The Delta-V for ~1 week mission (no inspection hyperbolas) is ~5-10 m/s
    - Assuming the SS deployed on a hyperbola with same safety margins than usual.

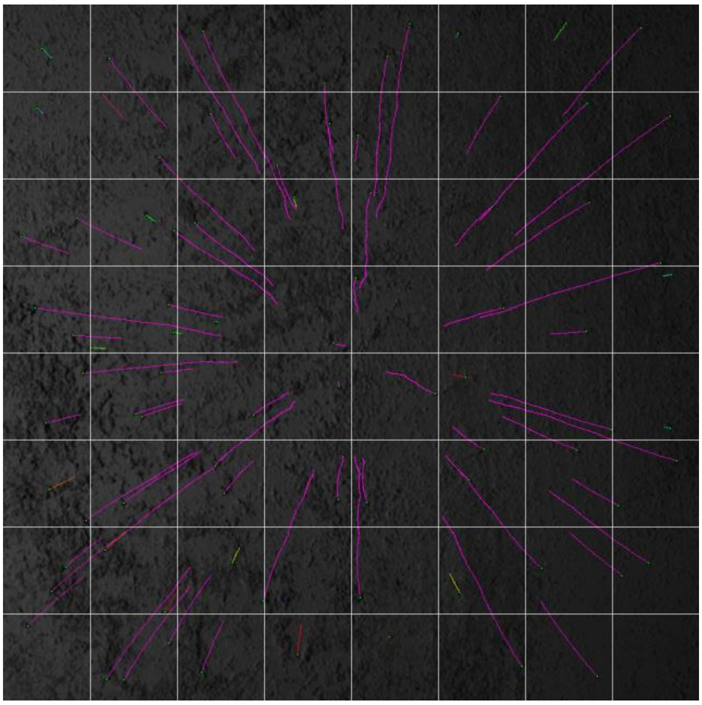


Figure 9-24: Example of IP performances for unknown feature tracking for a quasi-vertical descent (HW-in-the-loop tests in robotic facility with a mockup of asteroid Itokawa)

9.13 Major Design Constraints: CAUTIONS!

- The Line-Of-Sight based navigation for instrument pointing and CAM needs to be merged with the traditional ground based attitude guidance (semi-autonomous guidance).
- The trajectory a priori knowledge is limited by low-cost operations (high uncertainty in the gravity parameter) and the manoeuvre execution error of low-cost platform (a critical parameter that depends mainly on the thruster errors and GNC control errors).
- The ΔV budget depends mainly on the minimum altitude (science requirement) & ΔV frequency (operation pattern).
- The existence of 4-3-4-3 days hyperbolic arcs depends on the minimum altitude and arc duration.
- SmallSat equipment required for relative navigation might need delta-qualification in particular detectors of optical sensors and the electronics. Some measures to increase the radiation tolerance might be needed (e.g. binning of oversample images to minimize impact of SEU).

9.14 Technology Requirements

The following technologies are required or would be beneficial to this domain:

Included in this table are:

- Technologies to be (further) developed

- Technologies available within European non-space sector(s)
- Technologies identified as coming from outside ESA member states.

Equipment and Text Reference	Technology	Suppliers and TRL Level	Technology from Non-Space Sectors	Additional Information
9.5	Semi-autonomous attitude guidance based on LOS navigation in asteroids	ADS, GMV (TRL-4)	N/A	Activity pre-development for AIM
9.11	Semi-autonomous attitude guidance based on LOS navigation in Phobos mission	ADS, GMV (TRL-4)	N/A	Limb-detection for spherical bodies implemented in JUICE
9.12	GNC for asteroid landing	ADS , GMV (TRL-5)	N/A	Developments carried out for MarcoPolo and MarcoPolo-R

10 POWER

10.1 Requirements and Design Drivers MC

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
EP-010	When in sunlight at 1.1Au, the solar array shall be able to provide 1.5kW (+0% margin) of EP power and the platform power (+20% margin)	
EP-020	The battery shall be able to provide all of the energy (+20% margin) from launch up until successful solar array deployment and Sun pointing, including a safe mode routine	
EP-030	The power system shall provide a regulated bus 100V to the EP	
EP-040	The power system shall provide a 28V unregulated to the platform.	

10.2 Requirements and Design Drivers SS

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
EP SS-010	When in sunlight at 1.1Au, the solar array shall be able to provide 104W (+20% margin).	
EP SS-020	The battery shall be able provide energy (+20% margin) for a safe mode routine.	

10.3 Assumptions and Trade offs MC

10.3.1 Assumptions

To minimise the mass of the solar array Sun pointing is necessary, therefore a 3-axis stabilised spacecraft with 2 wings has been assumed for the baseline design. The high efficiency 3G30 cell with the standard CMX 100μm AR coverglass has been selected. A low solar array mass calculation factor of 4kg/m^2 has been used to calculate the SA mass. The solar array sizing has considered 2 strings failed, 1% harness loss, 80% effective cell area, and the 3% losses for power conversion (in the PCDU). The solar array has been sized with worst case degradation to provide at 1.1Au, 1.5kW (+0% margin) of EP power and the platform power (+20% margin).

There are usually only two manufacturers for batteries of this energy, SAFT and ABSL. ABSL designs for the same energy are usually lower mass, so it is assumed that ABSL would be selected as the manufacturer. For sizing the battery, 2 strings failed, 99% efficiency, and 2% capacity fade has been assumed.

For the PCDU it is assumed that the design would be based on BepiColombo MTM which has characteristics of high power conversion capability, low mass, high efficiency and low power dissipation.

Assumptions	
	SOLAR ARRAY
1	2 Wings, 0° Sun Aspect Angle
2	3G30C Cell with 100AMR coverglass
3	Low radiation environment, 2.5E14 @ fluence 1MeV (e/cm2)
4	Low mass of 4.5kg/m2 including mechanisms
5	2 strings failed
6	80% effective cell area coverage
7	1% harness losses
	BATTERY
8	Lower mass manufacturer assumed (ABSL, 18650NL cell)
9	2 strings failed
	PCDU
10	Design is based on BepiColombo MTM.
11	10W consumption
12	97% solar array power conversion efficiency

Notes: The SA sizing model includes temperature effect.

10.3.2 Trade Offs

In Figure 10-1 a block diagram of the baseline EPS design is shown. This topology has been selected for high efficiency and low dissipation. To generate the 100V bus for the EP power, a boost MPPT converter is used (heritage from BepiColombo MTM). A boost converter is advantageous for this application because it has high efficiency and the step up topology means that the solar array must be designed so the maximum voltage is always below about 90V, avoiding the potential problems of high voltage solar arrays. MPPT tracking enables the maximum power to be extracted from the SA in all temperature and solar flux conditions.

The BepiColombo MTM EP system required an unregulated bus, but for the EP of the SPP a 100V regulated bus is required. The EP power is much higher than the platform power, so the SAR generates directly the 100V bus for the EP. There is a problem that when the EP power and platform is off there is no power on the 100V bus so it may be difficult for the SAR to achieve regulation. To solve this problem a start-up load could be added that is on when the EP system and platform is off.

For the platform, an unregulated bus is selected due to a higher overall efficiency and lower mass and dissipation compared to a regulated bus (these advantages are because there is no battery discharge regulator stage). The battery is charged by a buck converter which draws its power from the primary 100V bus.

For simplicity the control, data handling and distribution aspects of the PCDU are not shown in the block diagram.

10.4 Assumptions and Trade offs SS

10.4.1 Assumptions

To minimise the mass and area of the solar array, Sun pointing is necessary, therefore a 3-axis stabilised spacecraft with 2 wings has been assumed for the baseline design. The high efficiency 3G30 cell with the standard CMX 100μm AR coverglass has been selected. Solar array mass calculation is based on the scaling up of an off-the-shelf item from Andrews Space. The solar array sizing has considered 2 strings failed, 1% harness loss, 79% effective cell area, and 10% losses for power conversion (in the PCDU). The solar array has been sized with worst case degradation to provide 104W (+20% margin) at 1.1Au.

The battery has been based on the off the shelf item available from GOM Space. This battery is using the 18650 cell which is the same form of cell that ABSL use. For sizing the battery, 1 strings failed, 99% efficiency, and 5% capacity fade has been assumed.

The PCDU is based on an off-the-shelf item from GOM Space to give approximate values for power consumption, efficiency and mass.

Assumptions	
	SOLAR ARRAY
1	2 Wings, 0° Sun Aspect Angle
2	3G30C Cell with 100AMR coverglass
3	Low radiation environment, 2.5E14 @ fluence 1MeV (e/cm2)
4	Mass of 300g per 16 cells (based on Andrews Space 6u SA)
5	2 strings failed
6	79% effective cell area coverage
7	1% harness losses
	BATTERY
8	Based on GOM Space BPX (18650 cell, 62.5g per cell)
9	1 string failed
	PCDU
10	Based on SmallSat GOM Space P31u x 6
11	0.5W consumption
12	90% power conversion efficiency

Notes: - The SA sizing model includes temperature effect.
- 90% is a high efficiency for a small power system.

10.4.2 Trade Offs

In Figure 10-2 a block diagram of the baseline EPS design is shown. An MPPT buck topology has been selected so that the losses of the SA series diode is minimised due to 4 series cell design of the solar array. The off-the-shelf power supply from GOM Space is an MPPT boost converter which has a slightly higher efficiency than a buck converter, but because the SA voltage must be lower than the battery voltage, the losses of the SA series diode is higher. If a boost converter is used then the battery voltage must always be higher than the solar array voltage. However, because of the small number of cells needed for the required energy, a SA boost topology may constrain the battery to being oversized and can also remove the possibility for tolerance to loss of strings.

For the baseline, the battery is 4 strings of 2 cells in series and the 3.3V and 5V are generated by buck converters. If higher voltages are required, topologies such as boost, forward or flyback could be used. If isolation is required for the secondary voltages the efficiency will be lower. In the block diagram a forward converter is used to generate +15V and -15V.

It should be noted that the conversion efficiency is varying with the SA and load currents and in some conditions may be lower than 90%, down to about 80% in the worst case.

Off-the-shelf designs for SmallSats may not be acceptable for ESA missions because they are generally not following critical ECSS standards for radiation tolerance, qualified processes and components, and failure tolerant designs. In the block diagram in Figure 10-2 all of the power conversion elements are shown to be redundant.

For simplicity, the control, data handling and distribution aspects of the PCDU are not shown in the block diagram.

10.5 Baseline Design MC

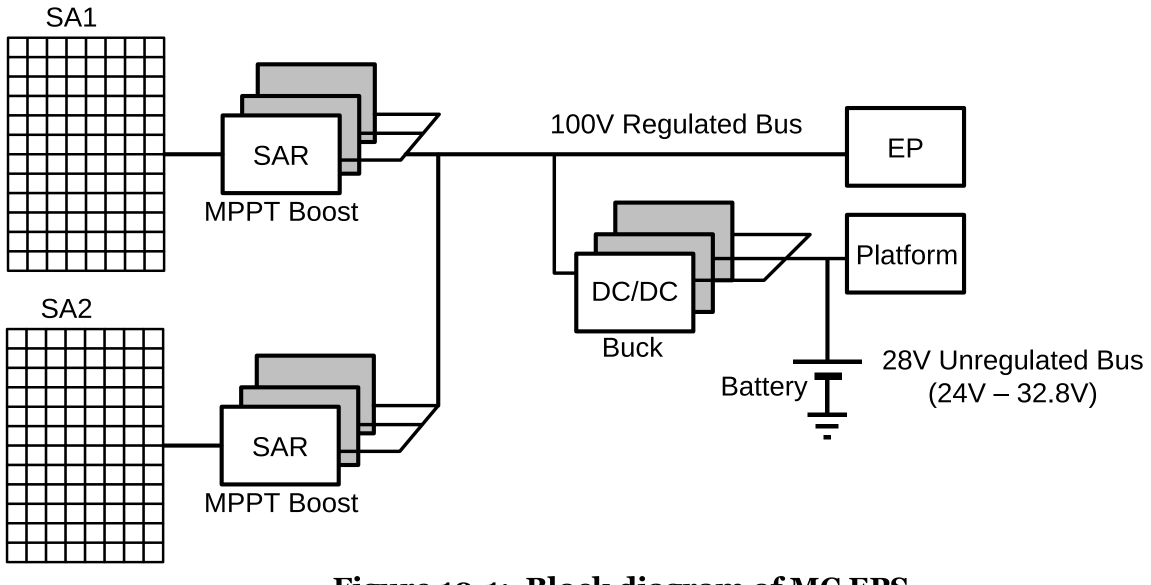


Figure 10-1: Block diagram of MC EPS

10.6 Baseline Design SS

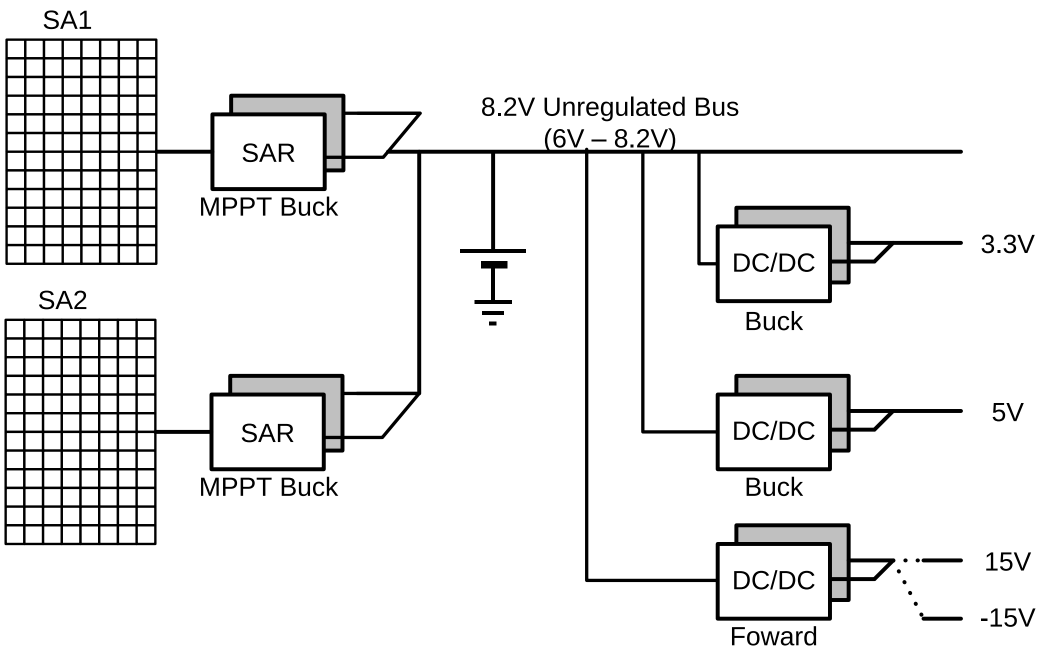


Figure 10-2: Block diagram of SS EPS

10.7 List of Equipment MC

PCDU

- Mass: 17kg

SA

- WC MB Power Generation at 1.1Au: 1657W
- WC MB Power Generation at 1Au: 1897W
- Mass: 31.8kg (15.9kg per wing)
- Area: 7.1m² (3.05m² per wing)

Battery

- Required Energy + 20% Margin: 886Wh
- Nameplate Capacity: 1152Wh
- Mass: 9.78kg

Quantity	GNC Equipment	Unit Weight	Total Weight	Margin	With margin
1	Battery	9.78 kg	9.78 kg	20%	11.736kg
2	Solar Array	15.9 kg	31.8 kg	10%	34.98 kg
1	PCDU	17 kg	17 kg	20%	20.4 kg
	Total		58.58 kg		67.116 kg

Table 10-1: Mass Budget for MC

10.8 List of Equipment SS

PCDU

- Mass: 0.6g

SA

- WC MB Power Generation at 1.1Au: 127W
- WC MB Power Generation at 0.746Au: 231W
- Mass: 2.92kg (1.46 kg per wing)
- Area: 0.64m² (0.32m² per wing)

Battery

- Required Energy + 20% Margin: 85Wh
- Nameplate Capacity: 115.2Wh
- WC Capacity: 94.8Wh
- Mass: 1kg

Quantity	GNC Equipment	Unit Weight	Total Weight	Margin	With margin
1	Battery	1 kg	1 kg	20%	1.2 kg
2	Solar Array	1.46 kg	2.92 kg	20%	3.5 kg
1	PCDU	0.6 kg	0.3 kg	20%	0.72 kg
	Total		3.06 kg		5.42 kg

Table 10-2: Mass Budget SS

11 DATA HANDLING

This chapter presents the design description of the On-Board Data Handling subsystem for the Small Planetary Platform mission for both Mother Craft (MC) and Small Satellite (SS).

11.1 Requirements and Design Drivers MC

The following requirements are directly applicable to Mother Craft Data-Handling Subsystem:

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
DH MC-050	The mothercraft shall have a data and power interface to the smallsats.	
DH MC-080	The mothercraft shall be capable of activating and commanding the smallsats before deployment including payload activation, navigation sensors, software upload and health status monitoring.	
DH MC-200	The mothercraft shall have on-board data storage for its own TM/TC and housekeeping data.	
DH MC-210	The mothercraft shall have on-board data storage for the smallsats' TM/TC and payload data.	
DH MC-220	The mothercraft's data handling system shall be sized to store all science data generated for TBD days.	

Additionally, during the course of the study the following design drivers were derived:

- DH subsystem shall provide a mass memory of 10 Gbit EoL. Note: Value of 10Gbit is derived from data budget provided by COMM sub-system
- To increase reliability, DH subsystem should be manufactured using Rad-Hard components
- To increase reliability DH should be fully redundant, including redundant CAN Bus
- DH should provide computational power for platform processing. No payload processing is foreseen
- For cost reduction purposes, DH should try to follow the trend of 'miniaturized' avionics (i.e. MASCOT-2).

11.2 Requirements and Design Drivers SS

Only one system requirement (SS-040) is directly applicable to Small Satellite Data-Handling Subsystem.

Over the course of the study the additional requirements were identified:

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
DH SS-010	DH shall provide computational power for platform, payload and GNC processing	
DH SS-020	DH shall provide capability to store TM&TC/Scientific data for TBD days	
DH SS-030	DH shall support CAN as a main avionics bus.	
DH SS-040	DH shall provide interfaces allowing communication with other subsystems, payloads, sensors and actuators.	

The following design drivers were applied to Small Satellite DH Subsystem:

- DH shall be compact, i.e. SmallSat format
- Where it is possible, commercial-of-the-shelf products should be considered
- As SPP will be more exposed to high energy particles, to ensure reliability, latch-up immune components and redundant solutions should be considered.
  Note: This involves redundancy in both sub-system level (i.e. two OBC in the design) and component level (i.e. two chips of the same memory type per OBC).

11.3 Assumptions and Trade offs MC

11.3.1 Assumptions

Taking into consideration requirements and design drivers, no assumptions were made.

11.3.2 Trade Offs

For platform data-handling, as the only heritage 'miniaturized' avionics available is MASCOT-1 (and updated MASCOT-2), no trade-off was performed. MASCOT-2 design was taken as a baseline, although it is clear that redesign and delta qualification is needed. Moreover, looking at usage of GR712 processor (core component of MASCOT-1/2) in small satellites targeting Moon/Mars/Jupiter [ADCSS presentation], it is clear that presented approach in line with current trend.

In the CDF sessions, the topic of compression of scientific data has been discussed. Two possible scenarios have been considered:

Scenario A	Data is compressed on SS, and then sent to the MC
Scenario B	Data is first sent to the MC, and then compressed there

The conclusion from the discussions was that scenario A is the best, as it is assumed that the communication efforts are larger than the compression efforts. This is true for both star and mesh topology (Figure 11-1 and Figure 11-2). Worth noting is that the communication efforts are larger in the case of mesh topology, since each data packages could be sent more than once to reach MC. The amount of data is expected to be high (~Gbits) due to the low/high frequency radars.

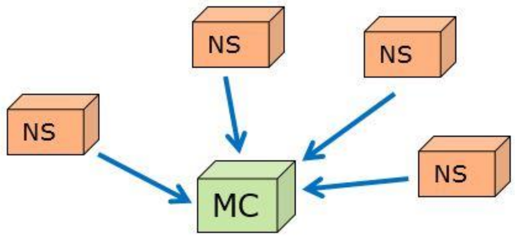


Figure 11-1: Star topology

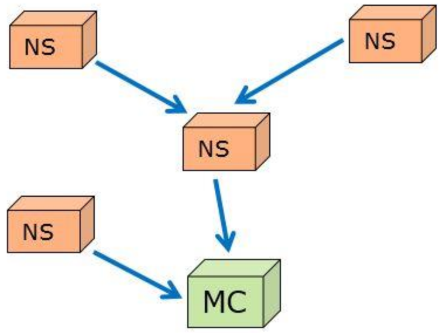


Figure 11-2: Mesh topology

Note: Data compression on-board MC would be feasible, but compression on SS is considered baseline.

11.4 Assumptions and Trade offs SS

11.4.1 Assumptions

The following assumptions were taken into consideration when proposing baseline design.

	Assumptions
1	Most of the off-the-shelf SmallSats sub-systems have only one CAN bus available. Moreover, the subsystems should have I2C available as a possible back-up to CAN bus although I2C is not considered in current baseline for platform bus.
2	It is assumed; that computational power needed by GNC, payload and platform processing should be fulfilled by dual core SoC with capabilities comparable to Xilinx Zynq platform (double A9 core, each running up to 866 MHz). This is in line with design driver that highly integrated data handling design should be capable of both platform management and performing GNC calculations (i.e. Vision Based Navigation in the case of SS becoming a lander).
3	It is assumed that temperature acquisition will be partially done by each subsystem (that is the case for most of the SmallSat solutions i.e. COMM, EPS). Any other needed sensor acquisition will be done in digital way (either using I2C or by digital I/O).

11.4.2 Trade Offs

Radiation hardened components are reliable, and resistant to both latch-ups and SEU. This would be necessary for the mothercraft, as the active lifetime would be longer than for the smallsats. The availability of the mothercraft is also mission critical. For the smallsats, an option would be to use rad-tolerant components, as it lowers the cost significantly. Rad-tolerant components are latch-up resistant, but the SEU protection can be covered by other means (EDAC/Scrubbing, TMR, software FDIR, sub-system level redundancies). During the discussion it was decided that rad-tolerant components should have preference for Small Satellite.

With the requirements, design drivers and assumptions presented above, the following of-the-shelf solutions were investigated (as a part of previous studies):

- Modular Avionics from GomSpace (Denmark) RD[43]
  (Integrated design with good flight heritage, radiation characteristics not fully known)
- On Board Computer from ISIS (The Netherlands) RD[44]
  (Good flight heritage, borderline performance, radiation characteristics not fully known)
- Data-Handling solutions from C3S (Hungary) RD[45]
  (Disruptive design, low radiation tolerance and no space heritage)
- Heterogeneous Computing Module from Unibap (Sweden) RD[46]
  (Good performance, no radiation data)

Having in mind the above pros and cons of investigated solutions, it was decided to baseline the SmallSat design on a solution from GomSpace which is modular, small sized and has enough capabilities for future computational needs (i.e. if VBN would be considered).

11.5 Baseline Design MC

The design for the MC is based on the MASCOT-2 design [SpW Article], including:

- OBC running LEON3FT, GR712 (fully redundant)
- I/O module with mass memory and RTU, 2GB BoL storage, 32 + 32 interfaces for thermal/separation sensors (fully redundant)
- CAN network for platform (redundant)
- Set of interfaces for communicating (RS422/SpW etc…).

The proposed communication with SS before deployment is point-to-point RS422 link (4 links, one for each SS). The estimated total mass would be below 3 kg and the assumed total power consumption would be below 6.5 W.

Figure 11-3: MC Data Handling Boards

11.6 Baseline Design SS

The following baseline solution for the SS is proposed:

- Docking board capable of hosting 4 expansion boards (Figure 11-4, left).
- OBC Unit (fully redundant), new development using upcoming RT microcontrollers, 1GB Flash per board (available COTS version Figure 11-4 right).
- Payload Processing Unit (fully redundant) (Figure 11-4, centre). Existing solution has the following characteristics:
  - Xilinx Zynq 7030 Programmable SoC with Dual ARM Cortex A9 (800 MHZ),
  - 1 GB DDR3 RAM and 4 GB storage (32 GB option),
  - FPGA module – 125k logic cells.

The proposed solution has the following properties:

- Mass: 2x40g (new OBC) + 2x70g (Zynq) + 74.2g (Dock) = 295g
- Power: 0.6W (OBC) + 2.3W-30W(Z7000, depending on usage)
- Size: 0.3 Unit.

Note: for purpose of power consumption estimation, for newly developed OBC Cortex-M0+ microcontroller UT32M0R50 from Cobham Geisler is assumed. [IPC] contains summary of ongoing developments for space graded microcontrollers.

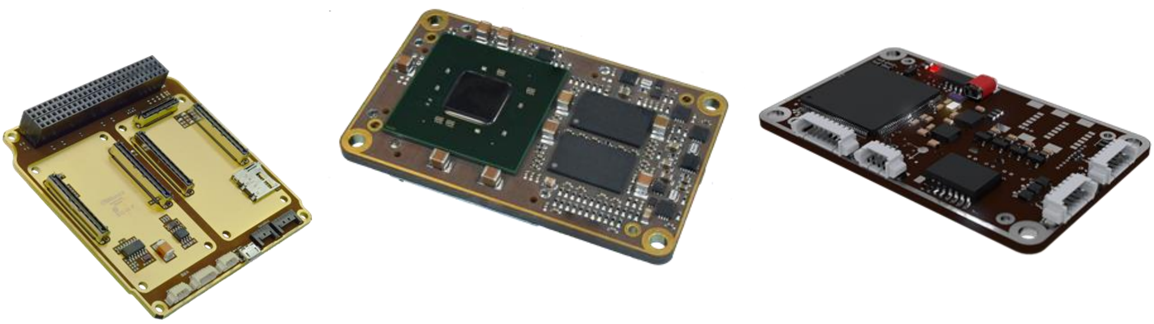


Figure 11-4: SS Data Handling Components

11.7 List of Equipment MC

Equipment	#	Mass	Mass w/ Margin	Power (Typ.)	Dimensions	Temp.	TRL	Rad. Dose
OBC Module	1	3 kg	3.6 kg	6.3 w	0.2x0.2x0.2 m³	-40/85 °C	6	

Table 11-1: DH equipment list for MC

11.8 List of Equipment SS

Equipment	#	Mass kg	Mass w/ Margin	Power (Typ.)	Dimensions	Temp.	TRL	Rad. Dose
Docking board	1	0.074 2	0.08162 kg	0.1 W	100x100x10 mm	-40/85 °C	6	20 krad
OBC	2	0.04	0.04 kg	0.6	50x50x10 mm	-40/85 °C	3	20 krad
Xilinx Zynq	2	0.07	0.0735 kg	2.3 – 30*	50x50x10 mm	-40/85 °C	6	20 krad

* Depending on duty cycle

Table 11-2: DH equipment list for SS

11.9 Sensitivity Analysis for MC: What if?

No sensitive cases are identified for the data handling system.

11.10 Sensitivity Analysis for SS: What if?

No sensitive cases are identified for the data handling system.

11.11 Sensitivity to Target: What if Phobos?

No sensitive cases are identified for the data handling system.

11.12 Architecture Sensitivity Lander

The data handling system will be able to handle VBN if there is a need for it.

11.13 Major Design Constraints: CAUTIONS!

No specific design constraints are identified for the DHS.

11.14 Technology Requirements

Referring to mission requirement MIS-070, the launch date is estimated in the time frame of 2024 and 2034. In the upcoming years, a breakthrough in terms of space certified microcontrollers is expected. Updated technology will result in increased performance for a lower cost.

12 TELECOMMUNICATIONS

12.1 Requirements and Design Drivers MC

The following requirements are directly applicable to Mother Craft Telecommunications Subsystem:

- MIS-080
- MIS-090
- MIS-110
- MIS-120
- MIS-150
- MC-020
- MC-060
- MC-090
- MC-110
- MC-140
- MC-150
- MC-190

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
COM-010	Hot redundancy shall be provided for telecommand (uplink) and cold redundancy for telemetry (downlink)	
COM-020	Ability to receive commands shall be possible at all times (except for close solar conjunction, if any)	
COM-030	Link-budget calculations shall be in accordance with ECSS standards	
COM-040	The TT&C subsystem shall implement ranging	
COM-050	The ISL shall support time transfer from the MC to the SS	

The following design drivers are considered.

- A total of **159.09Gb** of scientific data will be produced by the fleet of smallsats, split as follows: 58.33Gb, 58.33Gb, 39Gb and 3.43Gb.
- The design shall maximise the connection time between MC and SS (safe operations).
- The Comm design shall minimise the mass of the overall S/S (TT&C and ISL).
- Full redundancy with high-reliability components on the MC is assumed due to:
  1. The TT&C functions are in use throughout the missions and also the only means of communications for the SS with Earth.
  2. ISL on the MC is a central node for the communications among the SS and Earth.

12.2 Requirements and Design Drivers SS

The following requirements are directly applicable to SS Craft Telecommunications Subsystem:

- MIS-050
- MIS-080
- MIS-090
- MIS-110
- MIS-120
- MIS-150
- SS-040
- SS-060
- SS-180

On the SS only the ISL system is present and it is well covered as design drivers by the MC section; clear difference is the ICD (Mechanical and Electrical).

12.3 Assumptions and Trade offs MC and SS

The assumptions and trade offs for the MC and MS are common to both designs and are given below.

12.3.1 Assumptions

The following assumptions are considered.

Assumptions	
1	Baseline G/S is any ESTRAK 35m, option is the SRT
2	RF power output up to 100W (today SoA for SCI Missions)
3	HGA can be body mounted (not steerable/deployable) on the MC
4	The need for an MGA depends on the selected HGA diameter, specific mission profile and CONOPS
5	Ka-Band only for TTC is not possible due to missing support by ESA G/S
6	Dedicated PDT on Ka-band is not considered due to the Mass and Cost penalty
7	Cruise and Proximity operations Earth-MC distances as per Mission Analysis computations
8	Volumes to be transferred from SS network to Earth via TTC is: TOTAL of SCI + 10 Gb for NAV + SS platform HKP @ 1kbps constant + an overall 20% overhead for protocol. The assumed total is 221.5Gb. MC platform HKP contribution during Proximity operations not accounted and TBD.
9	Scientific data is generated linearly over time (time scale >> seconds)
10	SS can have either a cooperative or a non-cooperative attitude for ISL communications (pointing towards the MC vs keep any other pointing mode)
11	MC during Proximity operation can ensure constant pointing towards the centroid of the target minor body

Assumptions	
12	MC-SS relative geometry: MS-Target 12-20km, SS-Target 5-16km
13	For ISL Link usage computations: 1. Star topology (all SS communicates only to the MS directly) 2. Full time geometrical visibility (MS to SS)
14	Solar conjunctions outage (relevant windows of comms blockage) not considered, if any
15	Data latency (time from generation by Sensor/Payload to Ground delivery) not taken in to account

12.3.2 Trade Offs

12.3.2.1 TT&C Link trade Offs

The Frequency band for the direct to Earth TT&C Link present on the MC is constrained to be on X-Band but a number of parameters need to be trade-off, in particular the HGA gain (diameter), RF power output (only constrained by design driver to be less than 100W).

Typical architecture for TT&C in deep space are well known. For the specific case of SPP, the wide range of parameters, first of all the maximum slant range impacts the dimension of the HGA; depending on the needed gain by the HGA an intermediate step between the LGA, namely an MGA may be needed for safe operations (the higher the HGA gain the more demanding is the S/C pointing capability as precondition); as consequence, a frozen architecture with or without MGA, meant to close the gap between LGA and very directive HGA cannot be defined at present.

Some parametric rate estimation are done with following degrees of freedom:

- HGA diameter: 1.5m and 2m
- RF power output: 35W, 70W and 100W
- G/S: 35m, SRT

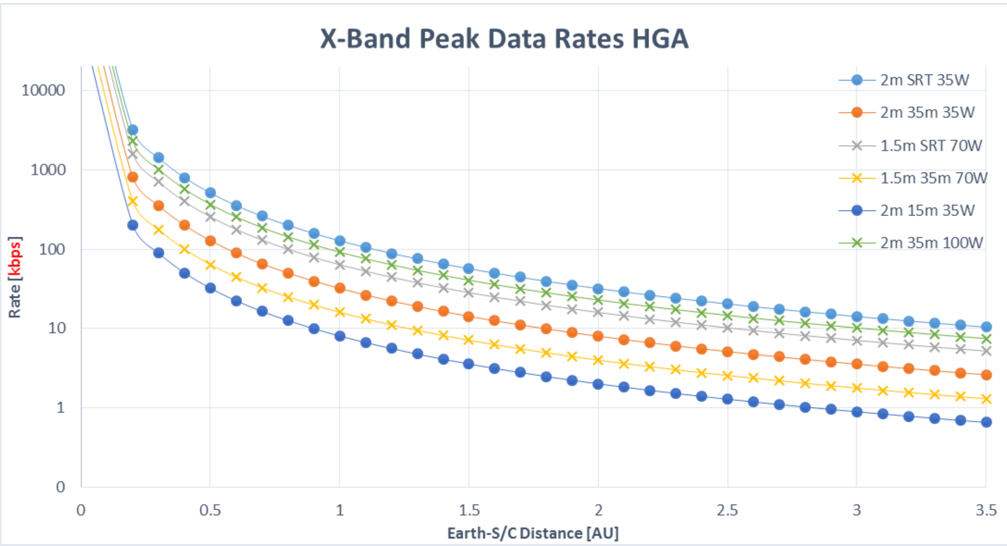


Figure 12-1: TT&C TM rates as a function of key parameters

100W RF output provides an improvement of about 40% with respect to the 70W output.

Achievable rates via onboard LGA are not shown but computed to be already limited (600bps) at distances in the order of 0.2AU.

Achievable information rate and ultimately data volumes depends on the number of hours a day available for transmission as well as the number of day per week and due to different trajectory profile also on the launch date.

Different launch dates will results in different arrival time and date of Proximity operation.

In the following figures, four communications scenarios (differences are the weekly amount of TX time and the diameter of the onboard HGA) are presented: as a function of the arrival date (X-axis) the amount of data that will be possible to download in the next 6 months of Proximity operations are computed (Y-axes).

The horizontal line marks the compliance with the data volume threshold and it is evident that for higher resources (larger HGA diameter and more contact time) the compliance is achieved for wither launch window.

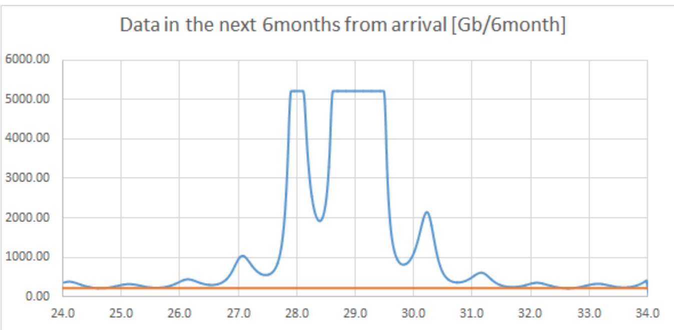


Figure 12-2: 2m HGA, 35m G/S, 100W: 16h/day, 7days/week

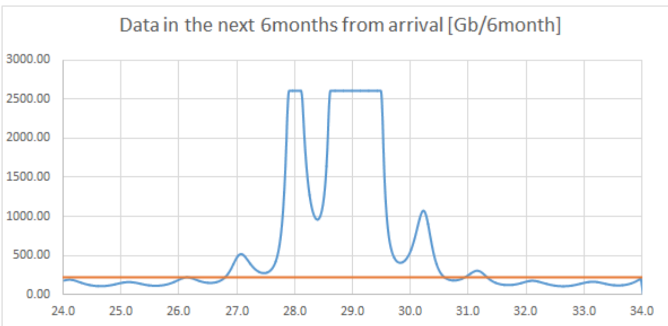


Figure 12-3: 2m HGA, 35m G/S, 100W: 8h/day, 7days/week

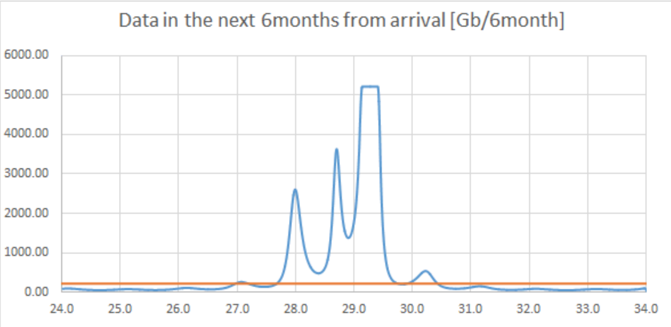


Figure 12-4: 1.5m HGA, 35m G/S, 100W: 16h/day, 7days/week

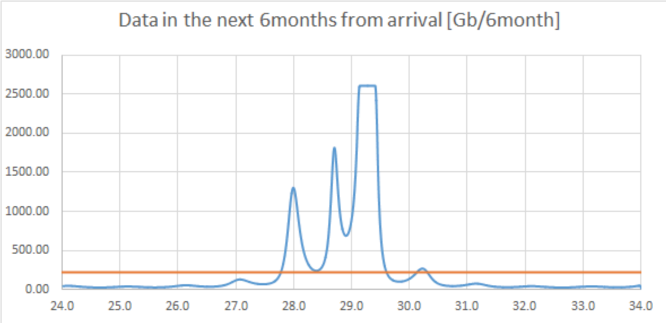


Figure 12-5: 1.5m HGA, 35m G/S, 100W: 8h/day, 7days/week

Figure 12-2 shows the widest compliant window, uninterrupted even though in some cases marginally over the full considered period, while worst case is shown in Figure 12-5 with compliance restricted to within 2027.5 and 2029.8.

In all the shown cases a plateaux is reached (flattening of the plot): that is due to limitation by ITU of the signal bandwidth that limits the maximum bitrate, despite that at short S/C-Earth distance from link budget higher rate could have been used.

12.3.2.2 ISL Trade Offs

The ISL system requires first a definition of the basic architecture. Two cases of interest are define and shown in Figure 12-6, Figure 12-7 both assuming star network topology (MC at the centre and SSs each as an end-node). More elaborated topologies such as a mesh network, even dynamically established, can improve coverage and data restitution at the price of a more complicated protocol to handle communications over the ISL. However not being identified as a stringent need, only simple star-topology is considered for SPP.

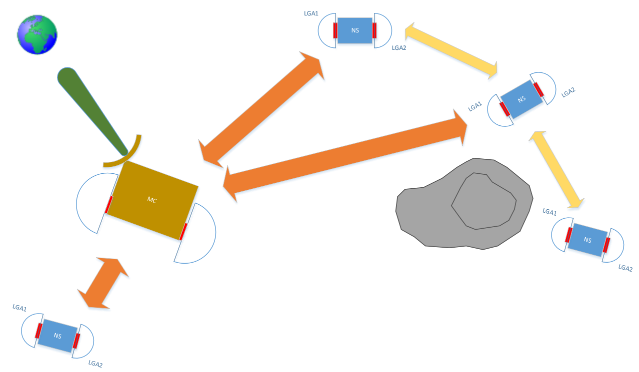


Figure 12-6: MC & SS with omni-coverage for ISL (LGA only)

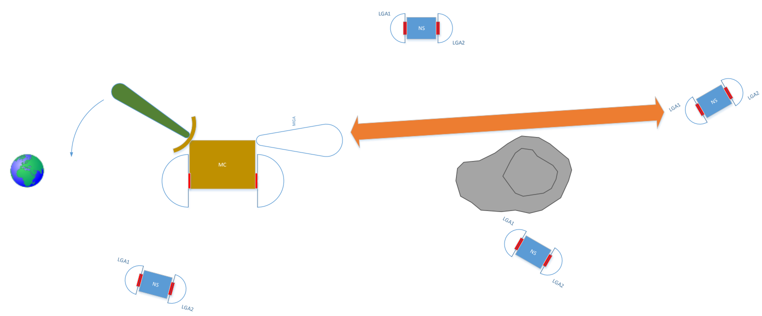


Figure 12-7: MC & SS with omni-coverage + MC with directive coverage for ISL (LGA + MGA)

An ISL system with only LGA on the MC can ensure basic communication regardless of relative orientation and distances (up to a max range).

An ISL that on the MC foresees LGA+MGA can add performance boost when MGA sees SS in main lobe, however due to MC manoeuvre for direct to Earth communications (TT&C HGA pointing towed Earth) some duty cycle among high-rate ISL and Direct to Earth comms).

Applying the foreseen geometries for Proximity operations, the max angle among MC-Target and MC-SS will stay below 45deg, therefore an LGA can ensure some useful gain; this means that for SPP it is sufficient to exploit the gain around the LGA boresight to achieve a communication boost without the need for actual MGA.

Selected baseline is *LGA only with MC pointing toward the center of the target, SS may point at the MC if/when needed.*

12.4 Baseline Design MC

Standard *TTC* System (LGA + HGA), MGA as an option.

Standard Deep Space TTC System (redundant TAS-I DST and TWTA); classic Parabolic Reflector can be used or more light weight solution (derived for example by TelecomSat) can be used with a high TRL already.

The Trasponder is equipped with a built-in 5W RF amplifier (to be use in alternative to the external TWTA) in cases where link budget does not requires very strong RF emissions (namely LEOP): this will allow to save energy and avoid any noncompliance with ITU regulation during the initial phase of the mission.

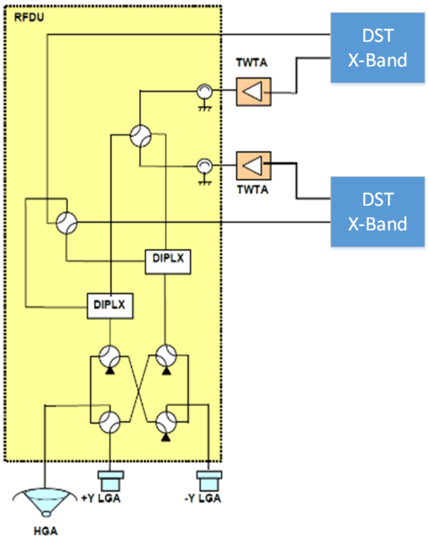


Figure 12-8: SPP TT&C baseline architecture

ISL based on 2 LGA both on MC and SS (to exploit antenna alignment for performance boost).

The baseline is CLASS 3 Proba-3 derived GamaLink. To be added to the baseline is adaptive rates capability, increased rate granularity, low power modes in stand-by and improve RNG for radio-science applications if needed. Most of the upgrades are SW/Firmware activities. The TMTC I/F to bridge communications among protocol used for SS and the one on the MC are still to be consolidated.

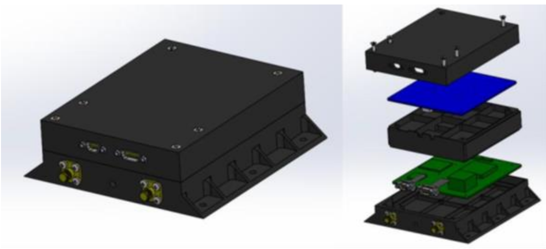


Figure 12-9: Proba-3 ISL

Table 12-1 shows per SS the achievable volumes that can be transferred to the MC with the various rates estimated to be achievable within the corners of the defined link geometry during proximity operations. In the case that all the SS are constantly at lowest rate (10kbps), the total load on the ISL exceeds the full capacity (102%): this is seen as an over pessimistic condition and not as a show stopper.

#NS	Volume (Gb)		% @ 600kbps	% @ 60kbps	% @ 30kbps	% @ 10kbps
1	58.33	3.8	0.6	6.3	12.5	37.5
2	58.33	3.8	0.6	6.3	12.5	37.5
3	39	2.5	0.4	4.2	8.4	25.1
4	3.43	0.2	0.0	0.4	0.7	2.2
Total=	159.09	10.23	1.7	17.0	34.1	102.3

Table 12-1: Duty cycle (time %) to meet Volume requirement

On top of the ability to exchange data over the ISL, also line of sight distance measurements are possible with the foreseen ISL system. 1D accuracy is as of today in the order of 50cm-1m but can be improved if needed; 3D position knowledge can be achieved thanks to combining multiple 1D measurements (1D distance measurements against MC and all the other SSs). Time transfer from MC to each of the SS will be performed over the ISL too to ensure that even in the event of a reset by any SS absolute time knowledge will be available (MC will perform time synchronization/correlation with Earth over the TT&C link).

12.5 Baseline Design SS

As mentioned already, the SS design is closely linked to the one on the MS, limited to the ISL system.

As for the MC, it is foreseen the same ISL system derived from Proba-3 however without redundancy and boxing, fulfilling (if needed) smallsat form factor.

12.6 List of Equipment MC

The list of equipment on the MC foresees a complied TTC&C (redundant) and completed ISL system (internal redundancy).

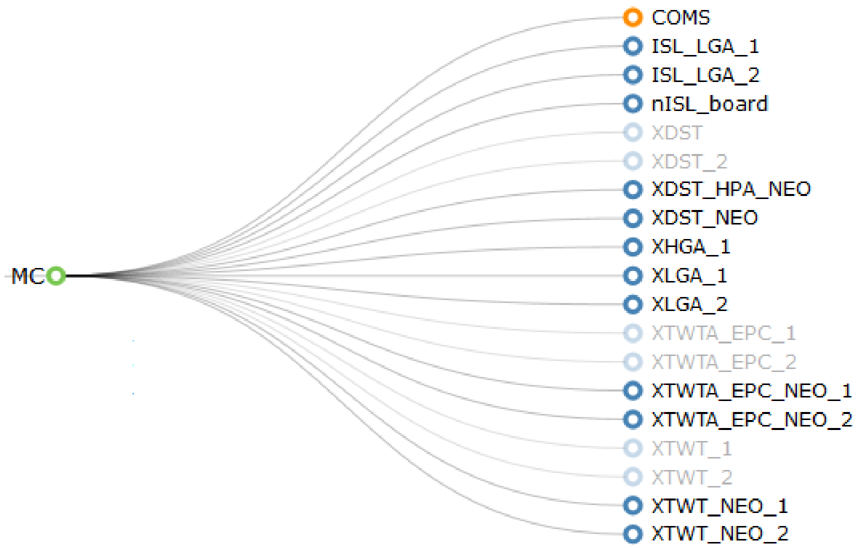


Figure 12-10: MC List of equipment

	Nr.	Mass per Unit (kg)	Mass margin (%)	Mass incl. margin per Unit (kg)	Total Mass incl. margin (kg)
COM					**23.07**
nanoISL LGA	2	0.05	20	0.06	0.12
nanoISL Electronics	1	0.15	20	0.18	0.18
X-Band DST built-in HPA (Allocation)	1	0	0	0	0
X-Band DSTRASP	1	3.7	10	4.07	4.07
X-Band HGA	1	8	10	8.8	8.8
X-Band LGA	2	1	20	1.2	2.4
X-Band TWT	2	2	5	2.1	4.2
X-Band TWTA EPC	2	1.5	10	1.65	3.3

Table 12-2: Mass Budget of MC

12.7 List of Equipment SS

The list of equipment on the NC foresees completed ISL system (without internal redundancy), equal on any of the SS.

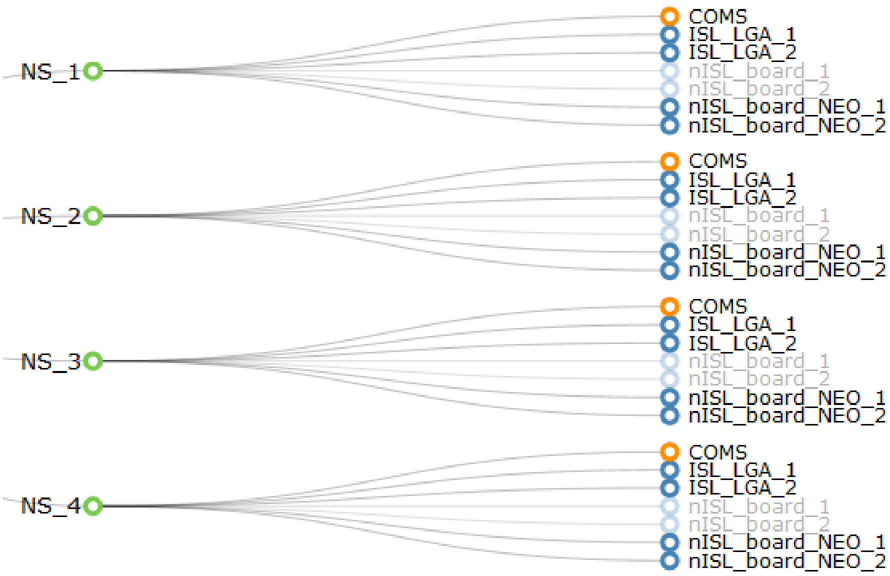


Figure 12-11: SS List of equipment

	Nr.	Mass per Unit (kg)	Mass margin (%)	Mass incl. margin per Unit (kg)	Total Mass incl. margin (kg)
COM					**0.48**
nanoISL LGA	2	0.05	20	0.06	0.12
nanoISL Electronics	2	0.15	20	0.18	0.36

Table 12-3: Mass Budget of SS

12.8 Sensitivity Analysis for MC: What if?

The design is quite sensitive to volumes and range variations. In trade-offs chapter (12.3.2) the sensitivity with respect to key parameters and achievable performance is shown. Good flexibility can be achieved thanks to non HW variations (contact time or used G/S) or modular variation (TWTA with different Power output).

12.9 Sensitivity Analysis for SS: What if?

Volumes that can be circulated over the ISL network in Star topology (MS as center), with selected baseline, are derived from corner cases: if such given geometry boundaries changes, the adequacy of an LGA-only concept on the MC may not be suitable anymore.

12.10 Sensitivity to Target: What if Phobos?

TT&C

Data budgets and sizing the link are to be derived when an actual mission is defined.

ISL

Link geometry to be evaluated for data volume restitution/DC power (Peak consumption and Duty Cycle) to best derive the most suitable concept of the ISL (LGA vs LGA+MGA) and needed RF Power to close the Link.

To be consolidated the amount of data to be transmitted. This may have implication on the TT&C as well.

12.11 Architecture Sensitivity Lander

TT&C

Data budgets and sizing the link are to be derived when an actual mission is defined.

ISL

Link geometry to be evaluated for data volume restitution/ DC power (Peak consumption and Duty Cycle) to best derive the most suitable concept of the ISL (LGA vs LGA+MGA) and needed RF Power to close the Link.

Extra loss in the link budget to be accounted for by the possible interaction of the surface with the antenna. Possibly a revision of the antenna network may be needed with 2 configurations (one prior landing, one after landing).

To be consolidated the amount of data to be transmitted. This may have implication on the TT&C as well.

12.12 Major Design Constraints: CAUTIONS!

To be consolidated the expected Spacecraft-Earth range over cruise and Proximity operation: such figure drives the, as shown, transmittable data volume, and is strongly linked with launch date. Onboard HGA sizing and/or duty cycles on the TTC link (G/S usage included, for cost) is depending on that.

12.13Technology Requirements

Some evolution of the Proba-3 ISL system is needed to introduce flexibility, in particular the capability of adapting rates accordingly to the link conditions without the need for pre-planned configuration but on the bases of the estimated real-time link conditions this will increase the achievable transferred volume of data.

As already mentioned, more sophisticated network topology compared to the baseline star networking can be developed allowing the ISL system to be able to further perform thanks to multiple packets hope. This is a technology, well established on ground networks that can be beneficial to be developed also for space ISL.

Accuracy of the 1D (3D) ranging estimate performed by the ISL can be improved if need (for example due to execution of scientific experiment based on that): it is mainly requested to improve the characterization/calibration of some key RF/Signal Processing parameters and some non-critical design modifications.

Regarding the direct to Earth link from the MC (the TT&C link), the availability of lighter, less power consuming and cheaper X-Band Transponder is of general interest. This is seen as a possibility being SPP different from usual Scientific Planetary missions for life time (SPP is a relatively short mission) and does not mandatory imposes Class-1 components (normally mandatory for ESA Deep Space missions).

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13 THERMAL

13.1 Requirements and Design Drivers MC

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
TH-MS-010	The TCS shall maintain all satellite sub-systems within their operating range while in operation and within their survival temperature range during all other mission times.	
TH-MS-020	The TCS shall maintain the propellant tank and feed lines temperatures in the following range for the whole duration of the mission: [+20°C; +50°C].	
TH-MS-030	The TCS shall minimise the use of active thermal control techniques.	
TH-MS-040	The TCS shall ensure the small-satellites to not exceed the TBC temperature range during mission up until deployment.	

Table 13-1: MC Thermal SubSystem Requirements

The design of the MC spacecraft is mainly driven by the varying thermal environments in earth orbit and at the target Apophis, which are described in more detail in section 1.1.1.

Throughout the mission duration the S/C has to cope with environmental heat fluxes. The external heat loads and available power, and therefore most times the thermal dissipation, increases as the spacecraft gets closer to the Sun. So the dimensioning case for radiator sizing will be at 0.75AU, which would then lead to an increased power demand in Earth orbit or at 1.1 AU to compensate for the large radiator panels.

Due to the varying distance to the Sun the solar thermal environmental heat loads are increased by about 180% at 0.75AU and reduced by about 20% compared to the Earth orbit. Operation at target perihelion at 0.75AU will influence the material selection due to high surface temperatures resulting from the high incident solar flux. The cold case to determine the required heater power is defined by environmental thermal heat loads and reduced available power, which is equivalent to low thermal dissipation.

Therefore the MC Thermal Control System (TCS) has to find a compromise to be suitable for all mission phases.

1.1.1 S/C Mission Thermal Environment

The environmental heat fluxes at the target asteroid Apophis were assessed. The environmental heat fluxes are solar heat flux from the Sun, and albedo as well as infrared heat fluxes. Figure 13-1 shows how the solar heat flux changes based on the orbit eccentricity for the target Apophis. Fluctuations in the solar heat flux due to annular and long-term solar activity are neglected in this assessment.

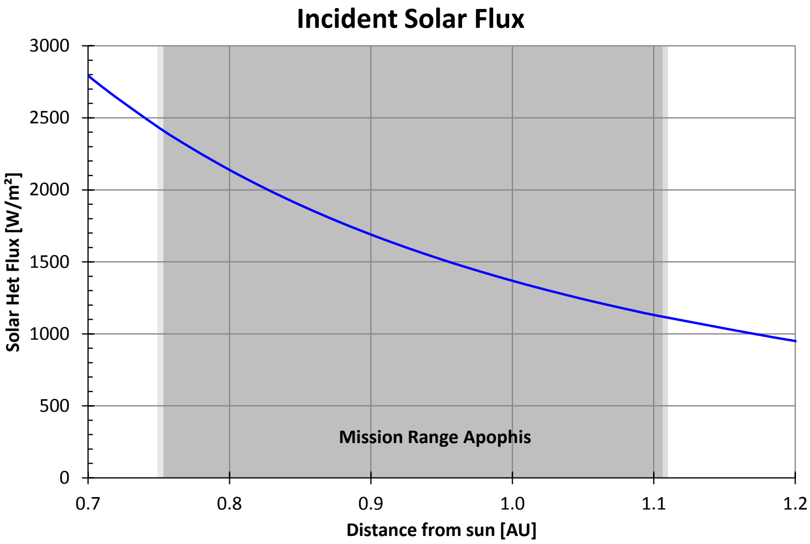


Figure 13-1: Solar Incident heat fluxes at the target Apophis

The albedo and infrared heat fluxes depend on target object properties. For target Apophis an albedo of 0.23, an emissivity of 0.9 and a diameter of 320 m was assumed. The target was assumed to be perfectly spherical and is considered to be in thermal equilibrium at all times. In reality, the surface temperature which drives the infrared heat flux will depend on the material properties of the surface, their thermal inertia and the rotation speed of the target. As such, the infrared heat flux is underestimated in the following figures. Figure 13-2 shows solar (Q_sol), albedo (Q_alb) and infrared (Q_IR) heat fluxes at the perihelion of target Apophis. The calculated target temperature at the perihelion is ~263 K. The spacecraft was assumed to be a cube to decouple the qualitative heat fluxes from the final spacecraft configuration. It was assumed that one side of the cube is facing the target object. This means that one spacecraft cube side has a large view factor to the target object and the four lateral cube sides have a smaller view factor to the target object. The rear side of the spacecraft cube is assumed to have no view factor to the target object. The heat fluxes in Figure 13-2 are plotted over the distance between a spacecraft and the surface of the target asteroid.

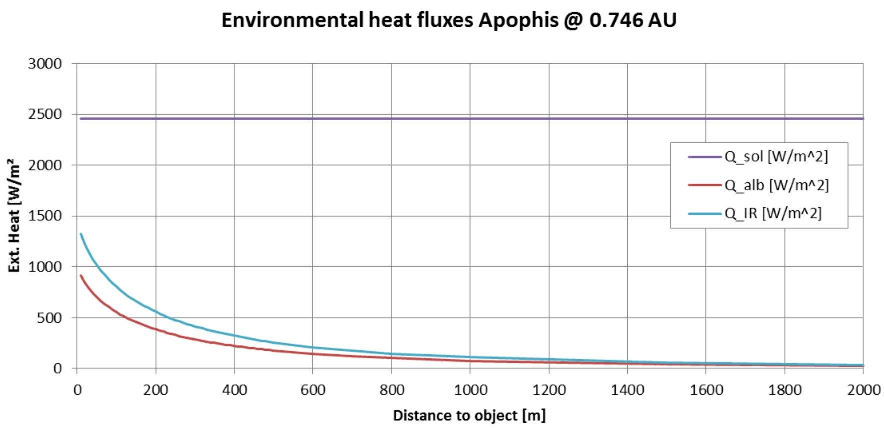


Figure 13-2: Environmental heat fluxes on a spacecraft at the target Apophis at perihelion

The same fluxes at target aphelion are shown in Figure 13-3. At the target aphelion a target temperature of ~318 K was calculated. The heat fluxes in Figure 13-3 are plotted over the distance between a spacecraft and the surface of the target asteroid.

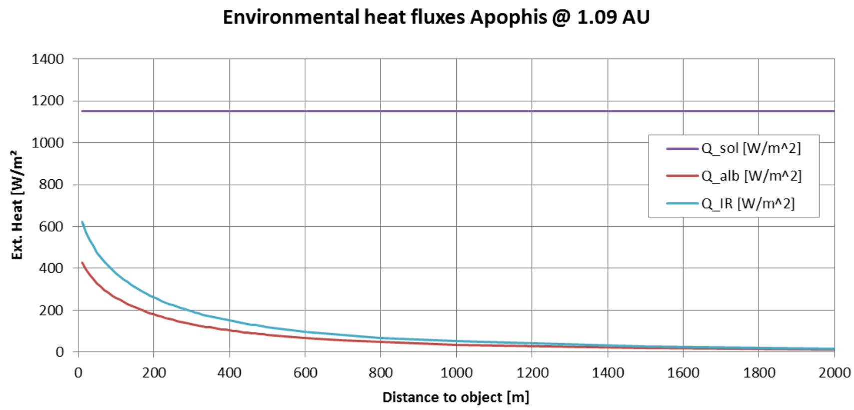


Figure 13-3: Environmental heat fluxes on a spacecraft at the target Apophis at aphelion

It can be concluded from looking at the environmental heat fluxes in Figure 13-2 and Figure 13-3 that the infrared and albedo heat fluxes are negligible for orbits above 2000 m of altitude.

13.2 Requirements and Design Drivers SS

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
TH SS-010	The TCS shall maintain all components of the SmallSatellite within their operating range while in operation and within their survival temperature range during all other mission times.	
TH SS-020	The TCS shall maintain the cold gas propellant subsystem in the following TBC non-operational temperature range: [-10°C; +50°C].	
TH SS-030	The TCS shall maintain the cold gas propellant subsystem in the following TBC operational temperature range: [0°C; +50°C].	
TH SS-040	The TCS shall minimise the use of active thermal control techniques.	

Table 13-2: SmallSatellite Thermal SubSystem Requirements

13.3 Assumptions and Trade-offs MC

13.3.1 Assumptions

Assumptions	
1	Cubic shape 1.5 x 1..5 x 1.5 m^3
2	Max. 250W thermal dissipation at earth orbit
3	S/C internal unit temperatures represented by an averaged core temperature.
4	Average core temperature limits are +20°C to +30°C (flow down from TH-MS-020).
5	A temperature gradient of 10K is assumed between the averaged core temperature and the radiator temperature.
6	Radiator efficiency of 90%
7	No external thermal loads on radiators for hot case sizing.

13.3.2 Trade Offs

To prevent the MC from overheating, radiator area or a radiator panel will be required. The sizing case for the radiator definition is the closest distance to the Sun including the highest thermal dissipation. In contrast, at Apophis perihelion the radiator will be a major contributor to the heat losses. The reduced thermal dissipation, caused by the reduced available power has to be compensated.

There are different concepts possible:

TCS Tech.	Basic Principle	Comments
Radiators & Heaters	Low alpha / High eps + compensation heating by electrical heaters	"Classic" & easiest approach Suitable for one design (max. power) case Requires compensating heating during low power cases
Radiators & RHUs	radio-isotopic thermal sources, radiators Flexible to variable ext. heat fluxes	Classical design but using RHUs for compensation heating to be independent from available power RHUs are not in line with the common European mission
Variable emissivity radiator & Heaters	Reduction of radiator performance by a) reducing active are or b) modifying optical properties	Power savings in cold case a) Mechanism required e.g. for louvers or deployable radiator Increased mass, Less efficient radiator in hot case b) Thermo-Chromics or Electro-Chromics with low TRL and limited performance
Heat switch & Radiator / Heaters	Variable heat conductance to radiator allowing to decouple the radiator, e.g. by using VCHPs, LHPs, MPLs or other form of heat switch.	Requires LHPs, VCHPs, MPLs or other form of heat switch Power savings in cold case

Table 13-3: MC radiator concepts

Figure 13-4 shows the radiator area required to dissipate 250W to deep space, using a radiator without external thermal loads as sun illumination. This provides the smallest required radiator area but leading to constraints on the attitude of the spacecraft. If there is some thermal backload on the radiator e.g. by solar, albedo or earth fluxes the radiator area will be higher than depicted in Figure 13-4.

For comparison the necessary area for a louvered radiator is also depicted in Figure 13-4. It can be seen that a radiator including louvers requires more area. This is because the louver mechanism as well as the louver fins partially cover the radiator, leading to a less efficient use of radiator area. The decrease in radiator area efficiency for louvered radiators is covered in this analysis by a reduced effective infrared emissivity leading to a required area increase of about 15%.

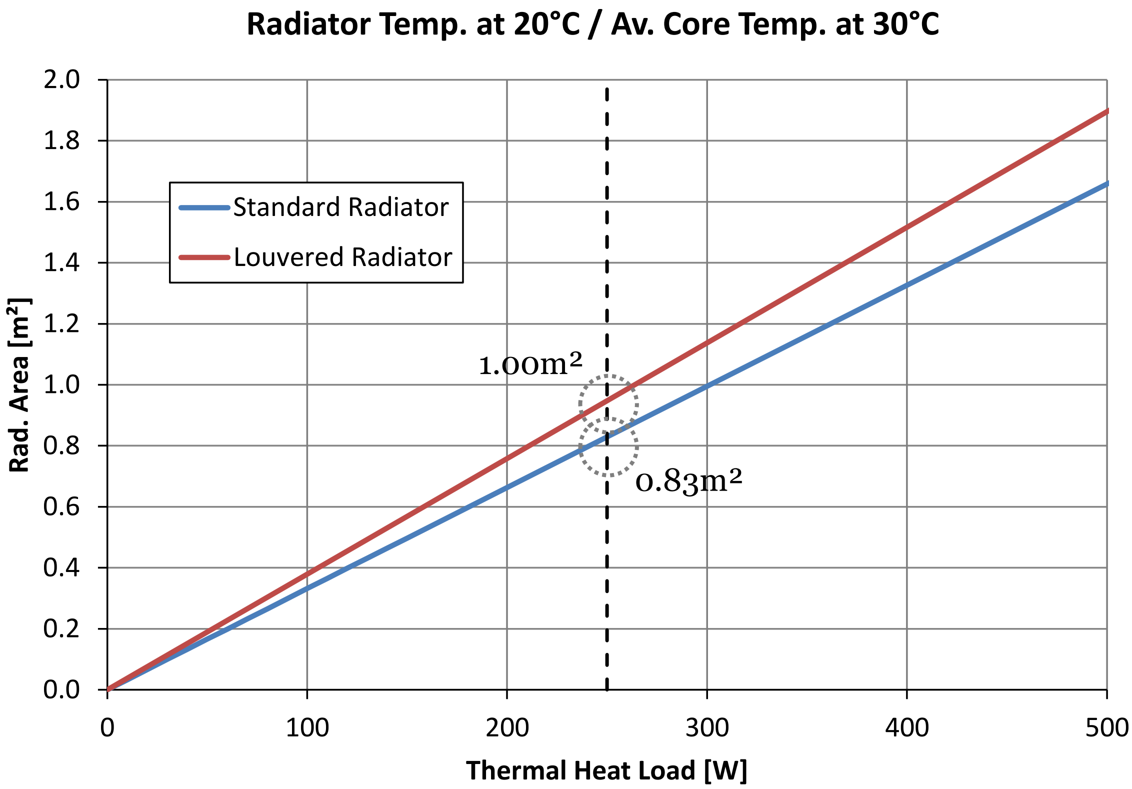


Figure 13-4: Required radiator area over thermal heat load

In conjunction with sizing the radiator for the hot case, the cold case must be taken into account to determine the required heater power. Figure 13-5 shows the heat losses of a potential MC. To guarantee a thermal environment within the limits these heat losses have to be either compensated by thermal dissipation of the installed equipment or electrical heater power.

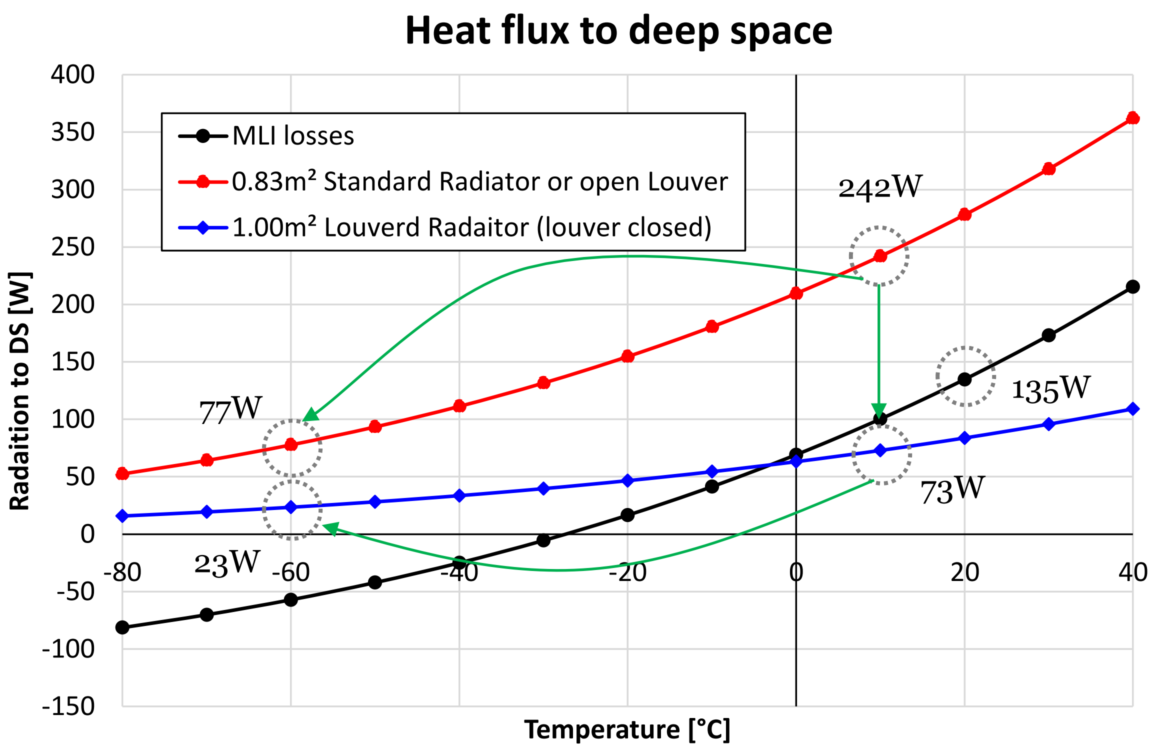


Figure 13-5: Heat losses to deep space for heater power estimation at 1.1AU

The black line indicates the heat losses of the MC though the MLI insulation at the target (1 side Sun illuminated). To hold the average core temperature above 20°C a heater power (w/o margin) of 135W is required.

A standard radiator of 0.83m² or an open louvered radiator would require about 242W of thermal/heater power to be maintained at 10°C, so 10K below the required 20°c average spacecraft core temperature. By closing the louver the required thermal/heater power can be reduced to 73W.

An alternative is the implementation of a heat switch. A variable heat switch functionality can be achieved by using variable conductance HPs (VCHPs) or Loop Heat Pipes (LHPs). Such a functionality would allow to decouple the radiator from the MC internal compartment. This allows an increased gradient between the average satellite core temperature and the radiator temperature. The minimum allowable radiator temperate would be determined by the freezing point of the working fluid of the heat transport system. The working fluid is Ammonia in most space flight applications. Other working fluids, e.g. Propylene, could be considered. But the advantage of having a lower freezing point goes together with the drawback of lower TRL level and a reduced heat transport capability.

Considering Ammonia as working fluid, a radiator temperature of about -60°C could be accepted in the cold case before survival heating has to be applied. This would reduce the required thermal/heater power to the radiator from 242W to 77W.

Combining the louvered radiator and the heat switch approach would lead to a total reduction of the required thermal/heater power on the radiator to about 23W.

The final design has to be determined by a system trade-off. If a thermal power (thermal dissipation and heater combined) of about 212W would be available the standard radiator plus a heat switch would be the solution. Therefore the louver mechanism, the increased radiator size and thereby mass could be avoided. If the outcome of the system trade-off is that the main mission driver is the available power and not mass and complexity, a combined solution of a louvered radiator and a heat switch is proposes.

13.4 Assumptions and Trade offs SS

13.4.1 Assumptions

	Assumptions
1	Cubic shape of 16U = 0.2 x 0.2 x 0.4m³
2	Max. 99.6W thermal dissipation
3	S/C internal unit temperatures represented by an averaged core temperature.
4	Average core temperature limits are +20°C to +30°C (flow down from TH-SS-020).

13.4.2 Trade Offs

Due to its relatively small size, there are two thermal design principles possible for the SS TCS, both providing advantages and disadvantages:

TCS Tech.	Basic Principle	Comments
Insulation and radiators	The SS is insulated from the environment Heat disposal is done via dedicated radiators or radiator faces	+ Most flexible TCS + Most efficient TCS + Thermal multi-zone design possible / high special thermal environment control performance + heater power reduction e.g. in safe mode - Restrictions on attitude - more complex TCs design - integration difficulties on SS in pods
One thermal zone	SS temperature trimming through choosing the right thermal coating (solar absorbance vs. IR emissivity). SS in radiative equilibrium with the environment	+no preferred sun illuminated side - one thermal environment / low special thermal environment control performance - risk of large temperature gradient across the SS due to the SS size - no heater power reduction e.g. in safe mode - TCS to ensure heat exchange between all SS sides

Table 13-4: SS TCS concepts

Figure 13-6 show the heat losses for SS dependent on the distance from the Sun. It can be seen that the heat losses at the mission range from 0.75AU to 1.1AU are relatively stable. This indicates that it might be possible to find one design, which could cover the full mission range.

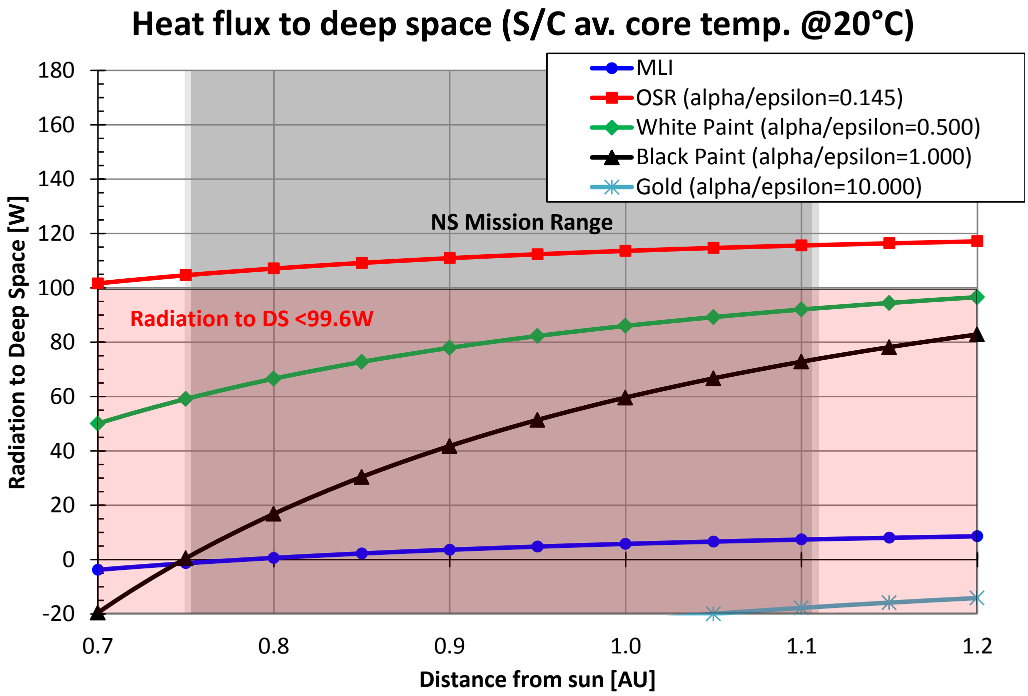


Figure 13-6: Heat losses to deep space for different thermal finishes

The SS thermal dissipation is shown as a transparent red area in Figure 13-6. A SS covered entirely in MLI already results in a heat flux to deep space through the MLI between -3.4W to +18W. This is only up to 20% of the total thermal dissipation. So to prevent a MLI insulated SS from overheating a radiator area or a radiator panel will be required to radiate 103W to deep space in total (thermal dissipation + environmental heat load) at 0.75AU.

On the other hand, a theoretical solution of a SS fully covered with OSRs leads to heat fluxes to deep space between 102W and 115W, which is in the same order of magnitude as the thermal dissipation. Therefore this design would be not feasible respectively very marginal.

From Figure 13-7 it can be seen that only relying on one thermal zone and trimming the SS internal temperature by adapting the alpha/epsilon ratio of the optical properties is extremely marginal. There are nearly no trimming options between a perfect black and a perfect OSR coated SS – not considering a real design including e.g. instrument openings, a non-perfect heat distribution and all other inaccuracies.

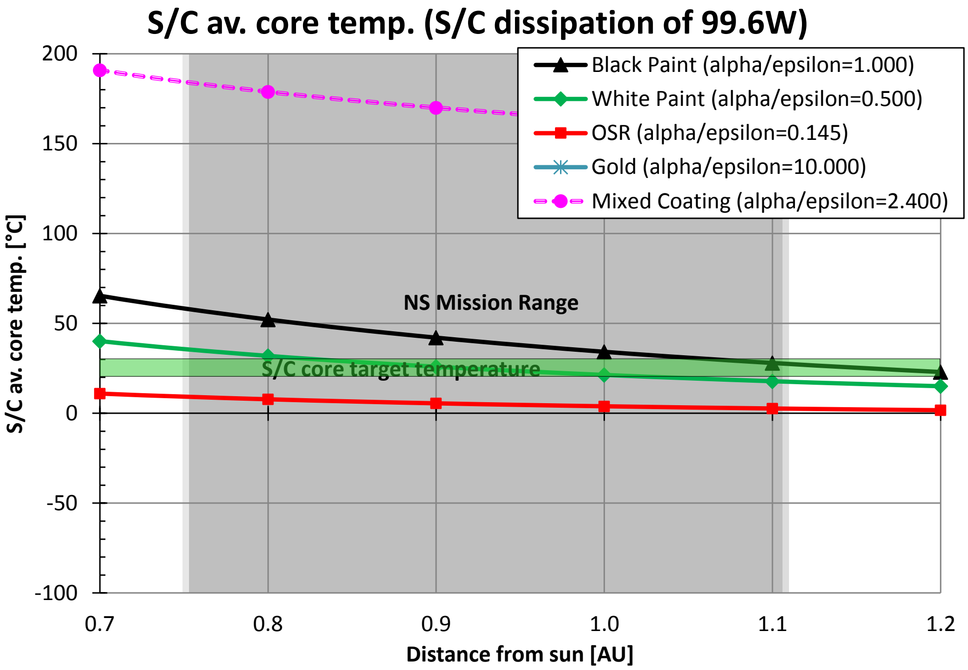


Figure 13-7: SS average core temperature for different thermal finishes

To gain design freedom other measures have to be taken to improve the situation of this marginal design. Possibilities are the reduction of the thermal dissipation, e.g. by a limited operational duty cycle, the operation of the SS at a hotter temperature level or the provision of additional radiative areas.

The first two options, reducing the thermal dissipation or operating hotter, cannot be discussed here, because they are dependent on the mission requirements and the hardware qualified to the specified temperature range.

Because the SS has to fit into the transport and launch pods (i.e. deployers) an extension in size to gain radiative area is not possible. Therefore this additional area has to be deployed during the commissioning phase of the SS.

To assess different possibilities of radiator concept the minimum radiator area (not considering environmental heat loads on the radiator) can be extracted from Figure 13-8.

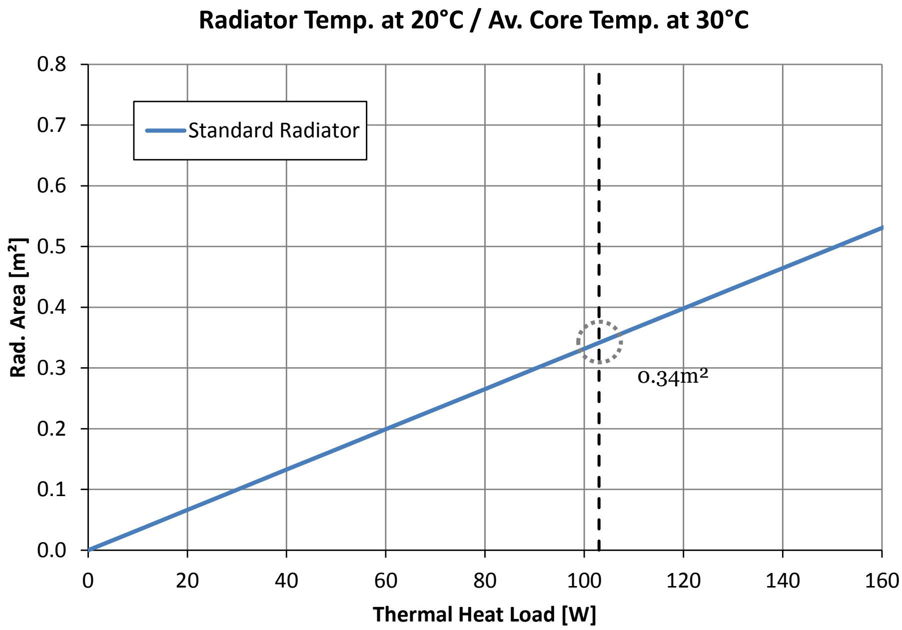
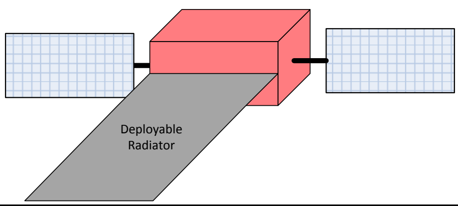
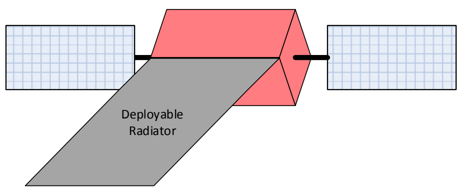


Figure 13-8: Required radiator area over thermal heat load

To radiate 103W of heat (SS thermal dissipation and environmental heat loads) to deep space a radiator are of about 0.34m² is required. This area assumes that there are no environmental heat loads from the Sun, albedo or infrared sources on the radiator. This underlines the fact it would be marginal to only use the SS body areas as radiator (0.34m² of radiator size required vs. 0.4m² SS total surface area).

Due to the fact that the most likely required deployable radiator will be fixed to the SS body the SS has to be stabilised towards the Sun direction to shadow the radiator environment. In case external solar fluxed on the radiator areas cannot be avoided, the required radiator area will increase. To reduce mainly the absorbed solar heat flux the SS Sun pointing side shall be insulated with MLI or covered with SSM or OSRs.

Different radiator concepts can be considered. They range from relying fully on the deployable radiator and insulating the SS body with MLI to a combination of SS body radiators and a deployable radiator as presented in Table 13-5.

Number of full body radiators			Deployable Radiator Area [m²]
0			0.34

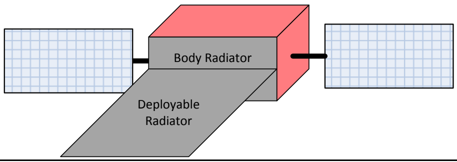
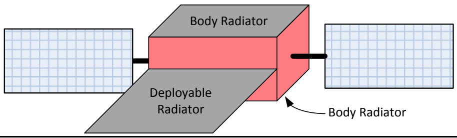
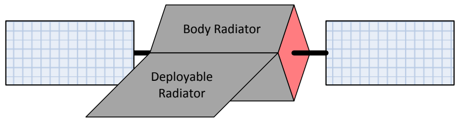
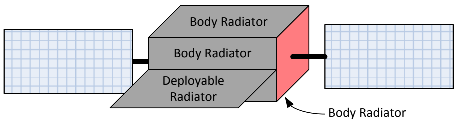
1			0.26
2			0.18
3			0.10

Table 13-5: Different combinations of deployable radiators and body radiators

A final design choice can only be made if more details of the SS design are available.

13.5 Baseline Design MC

Due to the early stage of this development no detailed mission and S/C design exists. Many questions, e.g. as the orbit attitude towards the Sun are still open and not fully defined.

The final TCS design will be dependent on several other constraints, e.g. attitude, available power, qualified temperature ranges of equipment.

Instead of a detailed thermal design and a TCS definition, basic assessments and a toolbox of different potential TCS measures are presented. In this way boundary conditions and the feasibility of this mission can be shown.

The TCS should make use as much as possible from standard thermal hardware so as:

- Multilayer-Insulation (MLI),
- Thermal coatings,
- Thermal washers and fillers,
- Thermal straps,
- Thermal doublers,
- Heat pipes (HPs),
- Electrical heaters for compensation and survival.

In addition, some special thermal hardware might be required to cover the wide range of the thermal environment. These measures are e.g.

- High temperature MLI
- Radiator shades / baffles
- Radiator parabolic reflectors
- Louvres based on fins or shutter
- Heat switches
- Variable conductance heat pipes (VCHPs)

- Loop heat pipes (LHPs)
- Mechanical pumped loops (MPLs).

But also measures are available, e.g.

- Favourable attitude control towards the Sun
- Thermal multi-zone design.

A high level example for the TCS design of the MC is shown in Figure 13-9.

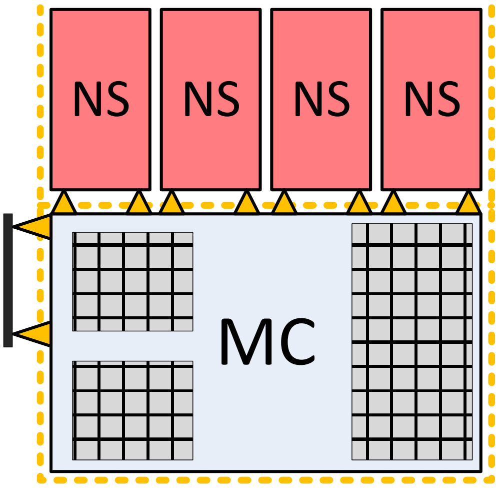


Figure 13-9: Potential TCS design principle of MC

The MC including the pod for the SS are wrapped in MLI (indicated with dashed orange line) to minimise the heat losses to deep space as well as the environmental thermal heat loads via the S/C body sides. MLI or even high temperature MLI has to be selected to withstand the high external thermal loads at 0.75AU. Sun trapping shall be avoided by the design.

To minimise the heat losses and environmental thermal heat loads via the SS pods as much as possible, the pods shall be mounted to the MC main S/C using thermal standoffs (sketched as orange triangles).

Dissipated heat will be radiated to deep space via dedicated radiator panels (gridded grey areas). These panels should point to deep space to reduce the incoming solar heat loads. In case this cannot be avoided by mission constraints, these radiators should be covered with OSRs or even parabolic reflector fins.

This example for a MC baseline thermal design would have the following key facts:

- 0.83m² radiator panels
- 242W thermal power (dissipation and heater power combined) required at the target
- Radiator mass ~12kg

13.6 Baseline Design SS

The NC thermal design should be as simple as possible. Appendages should be as much as possible avoided to fit the SS into the transport and launch pods.

Therefore the thermal design of the SS would only consist out of standard thermal equipment as far as required.

- Multilayer-Insulation (MLI),
- Thermal coatings,
- Thermal washers and fillers,
- Thermal straps,
- Thermal doublers,
- Heat pipes (HPs),
- Electrical heaters for compensation and survival.

Section 13.4.2 showed that selecting as concept to have only "one thermal zone" and to trim the SS temperature by adapting the ratio of thermal coatings (alpha/epsilon ratio) is extremely marginal.

Based on the mission boundary conditions, uncertainties, current requirements, and the project status it is expected that such a design would be too marginal. Due to the limited surface area of the SS and the high thermal dissipation, an external radiative area has to be provided through a deployable radiator. The deployable radiator could be combined with no or several body radiators. Several concepts have been presented and discussed in section 13.4.2. Areas which are not used for radiating heat shall be covered with MLI (or even high temperature MLI if required) to achieve insulation from the environment. The final design solution could only be made as soon as more information is available about the detailed SS design and orbit attitude control performances.

A high level example for this TCS design of the SS is shown in Figure 13-10 on the left.

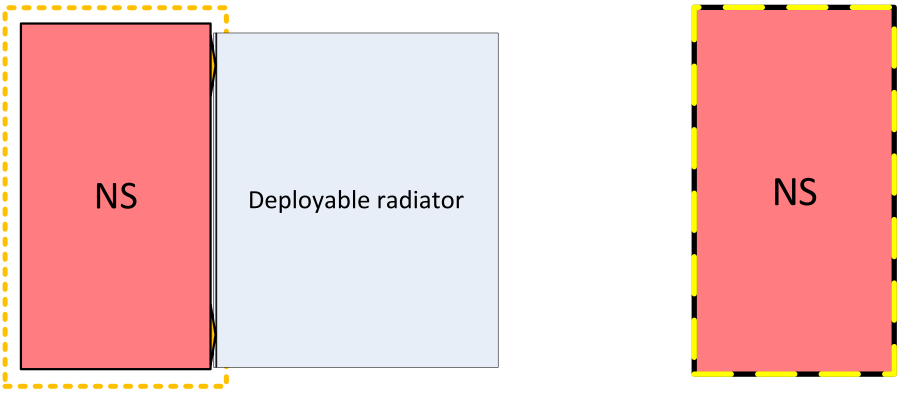


**Figure 13-10: Potential TCS design principle of SS
left: insulted and body and deployable radiators
right: body radiators vial alpha/epsilon trimming**

On the other hand, the marginal design of trimming the SS temperature by the right ratio of thermal coatings is the simplest TCS design available for the SS. During the detailed design phase it would be worth to investigate in more depth if this TCS design would be sufficient or too marginal. In case this TCS design suits the needs or can be adapted to the needs (e.g. limited duty cycle of payload) it will be the first choice due to its simple design and reduced limitations to SS attitude towards the Sun.

In this case special care has to be taken to distribute the heat inside the compartment as evenly as possible. All S/C wall have to have a good coupling to each other to avoid large temperature gradients between the Sun illuminated and the shaded side walls. If this cannot be achieved by normal thermal conduction HPs might be required.

13.7 List of Equipment MC

Table 13-6 shows the list of thermal equipment plus the mass budget for the Option 1 Mothercraft.

Thermal Hardware	Comments	Components	Mass [kg]	Mass (incl. margin) [kg]
MLI	Mass includes MLI, stand-offs and grounding straps	Assumed to covers all parts of the mothercraft except for radiator area.	5.60	6.72
Paints	black & white paints (mass includes primer and paint)	Outer surfaces of electronic boxes and inner surface of compartments	5.40	6.48
Thermal Washer	Vetronite & ceramic washers	Electronic units; Propulsion units: Payload elements;	1.20	1.44
Thermal Filler	Sigraflex thermal filler sheet (thickness = 0.2mm)	Between electronic boxes / payloads and respective structure	1.35	1.62
Temp. Sensor & Harness	PT1000, NTC 15 kOhm or NTC 10 kOhm as required / supported by data handling	Electronic boxes Analysis units Piping Radiators	0.60	0.72
Heat pipes		Distribution of heat inside the mothercraft; heat sources to radiator	6.000	7.200
Radiator	radiator surface area 0.83m²; includes louvers, includes SSM Tape	Outer surface of the mothercraft	9.96	11.95
Heater		Spread across the motherhcraft	0.6	0.66
Total thermal h/w mass			**30.71**	**36.73**

Table 13-6: List of thermal equipment and masses for Option 1 Mothercraft

13.8 List of Equipment SS

Table 13-7 shows the list of thermal equipment plus the mass budget for the Option 2 Smallsatellite.

Thermal Hardware	Comments	Components	Mass [kg]	Mass (incl. margin) [kg]
MLI	Mass includes MLI, stand-offs and grounding straps	Assumed to covers all parts of the Smallsatellites.	0.40	0.48
Paints	black & white paints (mass includes primer and paint)	Outer surfaces of electronic boxes.	0.16	0.19
Thermal Washer	Vetronite & ceramic washers	Electronic units; Propulsion units: Payload elements;	0.01	0.01
Thermal Filler	Sigraflex thermal filler sheet (thickness = 0.2 mm)	Between electronic boxes / payloads and respective structure	0.0012	0.0014
Temp. Sensor & Harness	PT1000, NTC 15 kOhm or NTC 10 kOhm as required / supported by data handling	Electronic boxes Analysis units Piping Radiators	0.20	0.22
Heat pipes		n.a.	0.000	0.000
Radiator	Body fixed and deployable radiator	Two sides of the Smallsat plus deployable radiator used as radiators.	3.96	4.75
Heater		Spread across the motherhcraft	0.2	0.22
Total thermal h/w mass			**4.93**	**5.87**

Table 13-7: List of thermal equipment and masses for Option 1 Smallsats

13.9 Major Design Constraints: CAUTIONS!

Thermal design constraints:

- Spacecraft configuration
  - Location of dissipating units
  - Location of temperature critical units
  - Radiator area accommodation
  - Heat losses via external I/Fs
  - Internal heat distribution (SS option 2)

  - No general design possible (S/C attitude unknown)
- Operational modes
  - Variety in dissipated heat loads
- Orientation and distance with respect to Sun and target body
  - Environmental heat fluxes.

13.10 Technology Requirements

The following technologies are required or would be beneficial to this domain:

Included in this table are:

- Technologies to be (further) developed
- Technologies available within European non-space sector(s)
- Technologies identified as coming from outside ESA member states.

Equipment and Text Reference	Technology	Suppliers and TRL Level	Technology from Non-Space Sectors	Additional Information
Smallsat	Deployable Radiator	TRL3 in Europe TRL6 for US companies	-	GSTP initiated

14 RADIATION

14.1 Requirements and Design Drivers

The basic requirement for the spacecraft, as for any other space system, is the proper functioning of the system when exposed to the space environment.

The Space Environment can cause severe problems for any space system. Proper assessment of the potential effects is an essential part of the engineering process and it is important that this is taken into account from the earliest phases of the project. This chapter gives an assessment of the space environment seen on interplanetary missions and its effects on the system. It is intended to assist the developers of the spacecraft and its instruments in assessing the effects of the space environment on their systems.

In general, the environments that need to be considered for a space system are the following RD[48]:

- Gravitation
- Geomagnetic fields
- Solar and Earth electromagnetic radiation
- Neutral Earth atmosphere
- Plasmas
- Energetic particle radiation
- Particulates
- Contamination.

The energetic particle radiation is considered the most important in the interplanetary environment, and the following analysis has therefore been restricted to this environment. This leads to the following specific requirements:

- The degradation/damage due to the energetic particle radiation shall be kept at acceptable levels
- The effects of radiation background in the instrumentation shall be kept at acceptable levels.

Consequently, the design drivers are the various possible mitigation measures. Examples of mitigation measures are:

- Shielding
- Radiation hardness of components
- Operational measures
- Earth escape trajectory selection.

The ECSS standard RD[48] shall apply to all space environments and effects analyses. This defines appropriate analysis methods and models, including the ones employed here.

14.2 Assumptions and Trade offs

The baseline design assumes that the entire mission will occur in interplanetary space, and the effects of the trapped radiation belts will not be considered as these trapped radiation belt effects are strongly dependent on the transfer trajectory selected. It is generally to be expected that for a direct injection trajectory the trapped environment will be a second order contributor to the overall mission radiation environment.

Thus, the principle contribution to the radiation environment for the mission is expected to arise from solar energetic particle events and galactic cosmic rays. Both of these environments are dependent on the phase of the solar cycle, and so the higher conditions for the two effects are considered. This assumption, though, does not imply worst-case conditions. Further, the duration of the mission plays a significant role in the total dose effects and must be considered, from MIS-100 and MIS-110 specify a maximum of 5 year transfer phase and 6 month operation phase, a combined 5 ½ year mission duration.

Ultimately, though, it is to be expected that with a direct injection, i.e. no Electric Orbit Raising escape trajectories, that the radiation environment for the mission will be no worse than a contemporary geostationary mission.

The MC and SS radiation environments will effectively be identical, as their separation will be insignificant on a heliospheric scale.

14.3 Baseline Design

14.4 Energetic Particle Radiation

In general, the energetic particle environment consists of geo-magnetically trapped charged particles, solar protons and galactic cosmic rays. It is the penetrating particles that pose the main problems, which include upsets to electronics, payload interference, degradation and damage to components and solar cells (see also RD[48]). The main components of the radiation environment are:

14.4.1 The Radiation Belts

The radiation belts encircle the Earth and contain electrons and protons that are trapped in the geo-magnetic field. An inner relatively stable belt contains mostly protons with energies up to several hundred MeVs that varies with the solar cycle, with higher levels encountered during solar minimum. An outer, highly dynamic, belt consists primarily of energetic electrons with energies up to a few MeVs.

This radiation source is not relevant for interplanetary missions, except for the Earth escape phase.

14.4.2 Solar Particle Events

Events of strongly enhanced fluxes of primarily protons originate from the Sun, usually with a duration on the order of a couple of days. These events occur randomly and mainly during periods of solar maximum (~7 years of the 11 year solar cycle). The events are also accompanied by enhanced fluxes of heavy ions. The geo-magnetic field can provide an element of shielding of these particles in equatorial zones at lower altitudes, but is irrelevant for interplanetary missions.

14.4.3 Galactic Cosmic Rays

A continuous flux of very high energy particle radiation is received from outside the heliosphere. Although the flux is very low, they include heavy ions capable of causing intense ionisation as they pass through matter. Although their contribution to the total dose is insignificant, they are important when analysing single event effects. The geo-magnetic field can provide an element of shielding of these particles in equatorial zones at lower altitudes, but is irrelevant for interplanetary missions.

14.4.4 Radiation Effects

Table 14-1 gives the parameters that are used for quantification of the various radiation effects. In the following, predictions of these basic parameters are discussed together with the information on how they have been derived and which models have been used.

The effects fall into two main groups:

1. those dependent on integrated doses
2. those dependent on peak fluxes or single event phenomena

The SPENVIS system RD[50] is used to determine the radiation environment and its effects on spacecraft.

Radiation effect	Parameter
Electronic component and material degradation	Total ionizing dose.
Material (bulk damage), CCD, sensor and opto-electronic component degradation	Non-ionizing dose (NIEL).
Solar cell degradation (power output)	NIEL & equivalent fluence.
Single-event upset (SEU), latch-up, etc.	LET spectra (ions); proton energy spectra; explicit SEU/SEL rate of devices.
Sensor interference (background signals)	Flux above energy threshold and/or flux threshold; explicit background rate.

Table 14-1: Parameters for quantification of radiation effects

14.4.5 Method

To obtain the radiation environment over the mission a 1 AU interplanetary orbit is selected.

For the solar proton events the ESP statistical solar proton model RD[53] is used with a 95% confidence level and assuming the 5½ year mission is during a period of solar maximum activity, providing a worst case scenario, see Figure 14-1.

Dose is then calculated using the SHIELDOSE-2 model RD[52] and solar cell degradation calculated using the AzurSpace 3G30 (21 eV SR-NIEL) EQFLUX models RD[54], RD[55], see Figure 14-2 and Figure 14-3, respectively.

The galactic cosmic rays (GCR) will be significant for Single Event Effect (SEE) and instrument background/noise analyses, but due to its low flux, it has been ignored for the dose calculation. The GCR ion spectra have been calculated for interplanetary space during both quiet (normal) and active (solar energetic particle event) conditions with a nominal spacecraft shielding of 1 g/cm². These spectra have been combined into a Linear Energy Transfer (LET) spectra, which is the standard input to the SEE analysis tools, see Figure 14-4 and Figure 14-5.

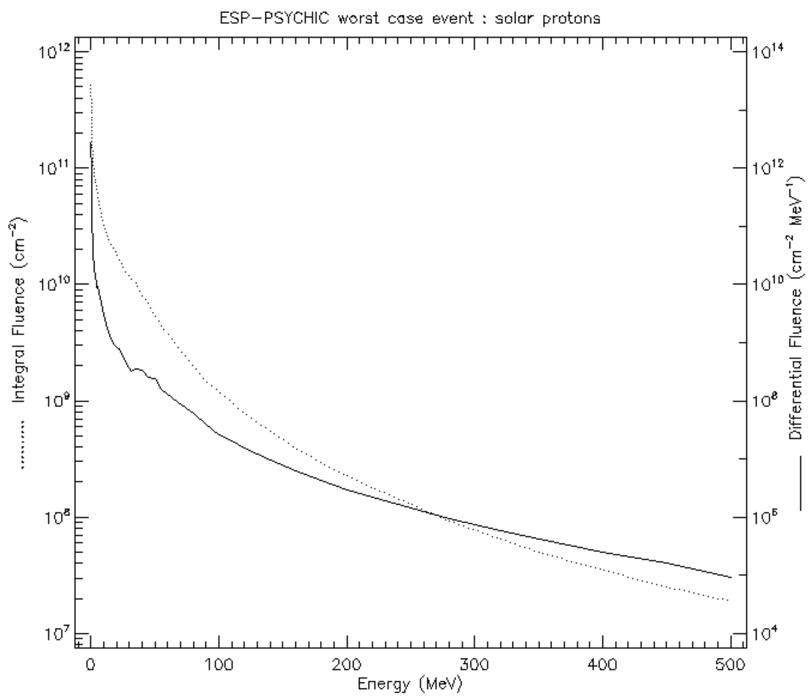


Figure 14-1: Mission solar proton fluence spectra

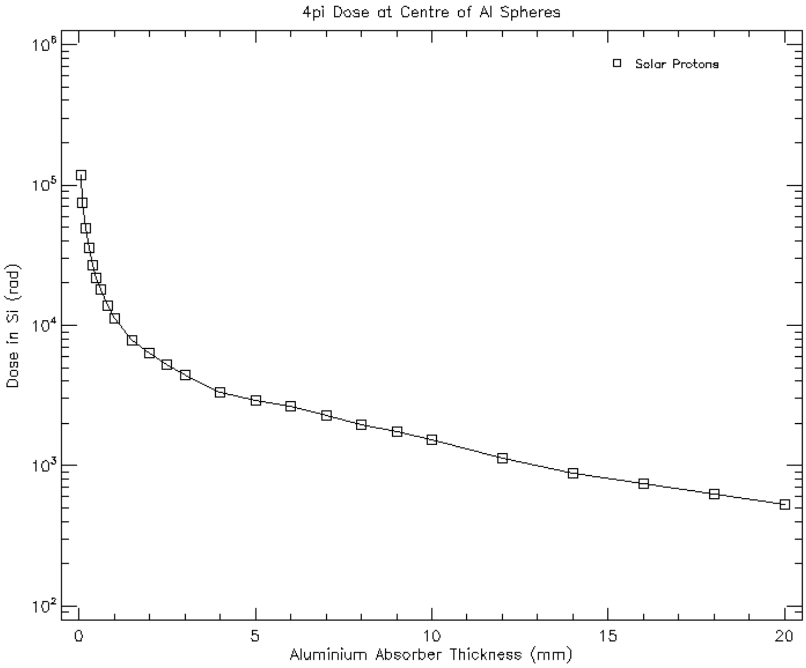


Figure 14-2: Mission total ionising dose as a function of solid sphere aluminium shielding

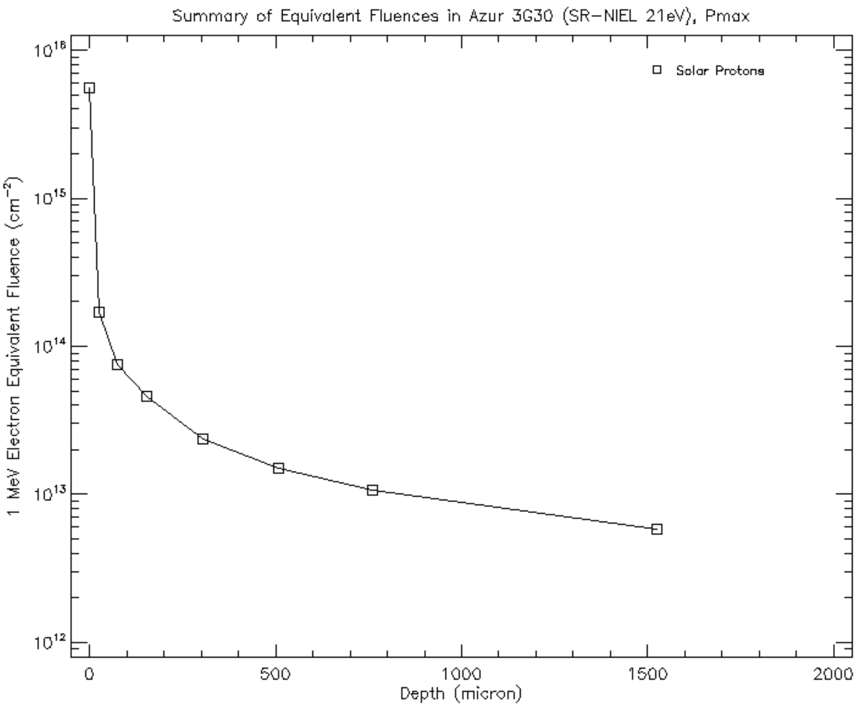


Figure 14-3: Mission Azur 3G30 (SR-NIEL) solar cell equivalent 1 MeV electron maximum power fluence as a function of cover glass thickness

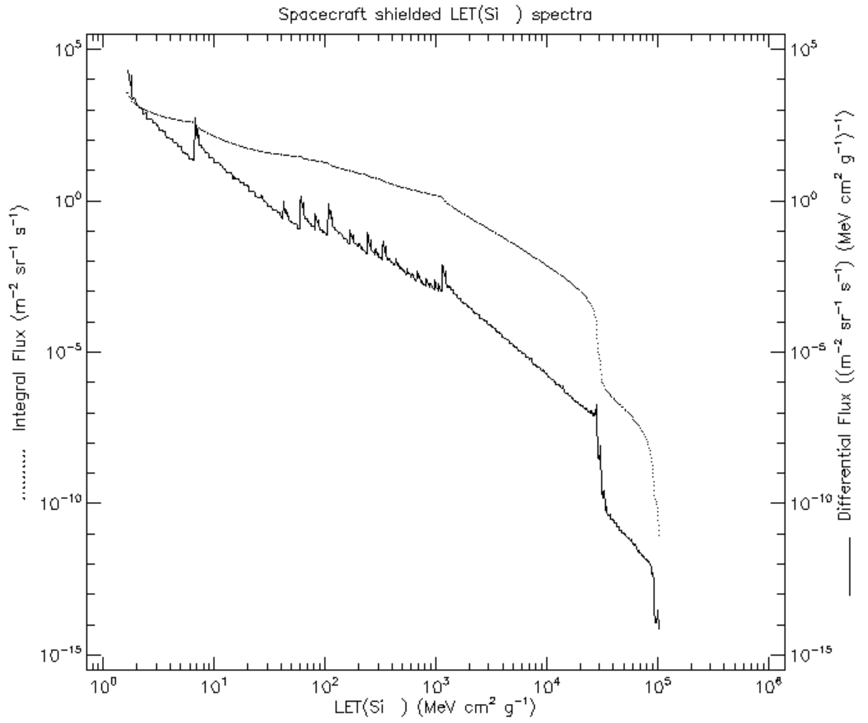


Figure 14-4: GCR Linear Energy Transfer flux spectrum – Quiet conditions

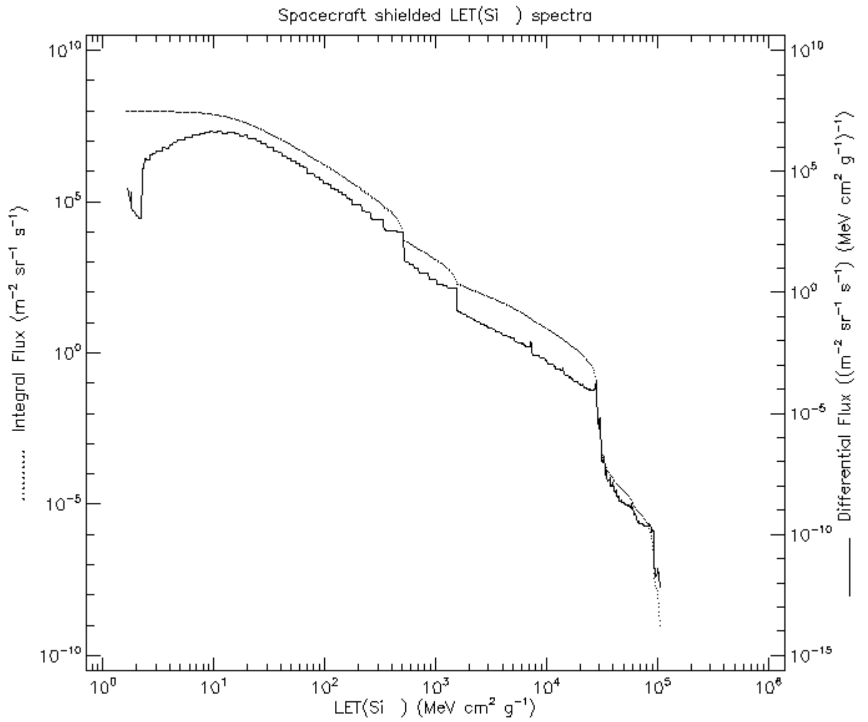


Figure 14-5: GCR Linear Energy Transfer flux spectrum – Solar particle event conditions

14.5 Sensitivity to Target: What if Phobos?

There are no radiation belts around Mars, and so the environment remains "interplanetary" in nature. The only variability will be due to the Mars and Phobos solid angle shielding of galactic cosmic rays and solar particle. This solid angle shielding can largely be ignored: the spacecraft must be designed to operate in the radiation environment when the shielding is not provided; and when the shielding is available the total effect is expected to be considerably smaller than the uncertainties in the environment models.

14.6 Major Design Constraints: CAUTIONS!

As previously stated, the Earth escape phase radiation environment must be considered during the planning phase and the mission concept is more mature.

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15 GROUND SEGMENT AND OPERATIONS

15.1 Requirements and Design Drivers MC

The ground segment and operations infrastructure for the Mission Operations Centre (MOC) of the future SPP missions will be set up by ESA/ESOC and it will be based, as far as possible, on the extension of the existing ground segment infrastructure.

The preparation of the GS&Ops Concept for future missions using the SPP Tool-Box Study is mainly driven by the cost-effective concept. Mission Characteristics

Missions using the Small Planetary Platform, like Rosetta, will perform multi-point remote measurements around small bodies allowing the scientific community to gather information from different locations simultaneously; the SPP mission concept includes a mothercraft (MC) and a swarm of small-satellites (SS).

15.1.1 Mission Timeline Overview MC

Launch	2024-2034
Near Earth Commissioning	~ 2 weeks
Cruise Phase	~ 3 years
Rendezvous and deployment	~ 10 weeks
Stay at Asteroid Duration	6 months
Distance to Earth	0.75-1.1 AU
Disposal Phase	< 2 weeks.

15.1.2 On-Board Autonomy MC

The operations should be kept simple:

- Should have simple operating modes, and simple GNC modes.
- There should not be complex on-board autonomy.

Having complex and many spacecraft operating modes is directly proportional to the ground operations complexity: the more complex on-board operating modes the more complex will be the ground operations. Similar for on-board autonomy, complex on-board autonomy implies more ground testing and it does not always imply simple ground operations.

15.2 Requirements and Design Drivers SS

15.2.1 Mission Timeline Overview SS

Launch	With MC
Near Earth Commissioning	n/a
Cruise Phase	As per section 15.1.1.
Rendezvous and deployment	
Stay at Asteroid Duration	

Distance to Earth	
Disposal Phase	

15.2.2 On-board autonomy SS

As per section 15.1.2.

15.3 Assumptions and Trade-Offs MC

Many of the Ground Segment and Operations assumptions are based on the operations concept that was envisaged for AIM and for AIM-Next due to the similar mission characteristics and to the GS&Ops cutbacks foreseen for AIM-Next.

15.3.1 LEOP MC

Several options were considered for the launcher, see MIS-060. Hereafter are the operational considerations to take into account for all launcher options.

Low Earth Orbit Phase operations end with the first successful launcher dispersion correction manoeuvre at 2 to 3 days after launch.

The following operations consideration should be taken into account during this phase:

- No long LEOP durations, ~ 48 hours
- No complex Earth departures strategies (e.g. as Lisa PathFinder)
- No complex LEOP operations: critical manoeuvres, deployments, etc.

For a shared-launch option, the LEOP activities, including Separation Sequence, should be synchronized with the co-passenger operations.

15.3.2 Near Earth Commissioning MC

Commissioning of two to three weeks will be performed after LEOP.

It is assumed that the sub-systems to be commissioned are not complex; the duration shall be analysed on a case by case basis.

The following operations are envisaged during this phase:

- Electrical propulsion system requires long term operations in order to gain confidence in uninterrupted operations
- Commissioning of the MC platform.

15.3.3 Cruise Phase MC

The duration of the cruise to the asteroid is around 3 years.

The Electric Propulsion system for the Cruise Phase is the baseline for this study. After an initial period of one year after launch, weekly coverage is compatible with the electric propulsion, but it still requires a dedicated control and monitoring effort, and constant orbit determination. A highly reliable propulsion system will probably simplify operations versus a poor performance electric propulsion system.

A complete Chemical Propulsion transfer is operationally beneficial versus an electric propulsion transfer. It will require less "baby-sitting" and it assumed that it will reduce the transfer duration.

It is assumed that, during the cruise phase:

- Minimum P/L checkouts
- No pointing requests
- No strange modes, e.g. hibernation, spin, etc.
- No swing-bys
- Non-contact periods for "passive" cruise should be in the order of 7 (EP)-14 days; anything above/below is likely to cause major impacts on the spacecraft ground segment design.

15.3.4 Operations at the Asteroid MC

Approach phase: a precise tracking campaign is required involving dual DDOR and Doppler and Ranging measurements over a duration of 4 weeks. The required duration will be analysed on a case by case basis and it will depend on the knowledge of target body before arrival. The ground contact periods will increase accordingly.

Asteroid operations and Science Phase: the MC will fly and release the SS. The Ground contact frequency will depend on the ground visibility analysis, the data downlink volume and the data latency requirements.

Ground communications will be via the HGA permanently pointed to ground, and MC will communicate with the SS via the ISL LGAs or MGAs.

Processing of all in-flight data to determine masses, shape, landmarks is not an operational task. It is assumed that it is still FDS task to do so as part of the reconstruction process. FDS are equipped for it from Rosetta, an extension to deal with the destination body system will be needed.

Note that the following does not mean that it will not be possible to select landmarks, construct maplets, determine shape and rotation state of body. What it means is that the operations strategy, and hence the operational distances, will not require doing so in the operations cycle:

- Navigation ground based
- Pyramid-like strategy at possible distances (ideally mans every 7 days)
- Operational optical navigation based only on body centroiding measurements meaning:
  - No operational need for landmarks
  - No operational need for maplets
  - No operational need for shape reconstruction
  - No operation need for body rotation state knowledge
  - At most, body total mass need is used for operations
- Loose navigation, just to allow imaging; possible distance TBC given the assumptions above, AIM between 10-20 km. Wide camera FoV will be beneficial

- No attempts to precise navigate for lander delivery
- No attempts of bound orbit
- No close fly-bys
- Regular daily passes
- Ground reaction time at best-effort basis.

15.3.5 Disposal Phase MC

Although this phase was not detailed during the study, it is assumed that no special operations are required for MC disposal phase.

15.4 Assumptions and Trade-offs SS

As per section 15.3.

15.4.1 LEOP SS

n/a

15.4.2 Near Earth Commissioning Phase SS

n/a

15.4.3 Cruise Phase SS

The SS will be in a sleep mode with infrequent unit check outs, minimum checkouts are assumed during this phase.

15.4.4 Operations at the Asteroid SS

During the approach phase and near the Asteroid, the SS will be again checked-out before being released.

SS Commissioning phase, it is assumed there will be a short SS commissioning phase after release, including the Reaction Wheels restart after a long off period during the cruise phase. As explained later in section 15.6.1, it is assumed that they are operated as any other instrument, after separation Mission Operations Centre (MOC) and MC will simply act as bent-pipes.

The orbit control of the SS is not considered under MOC responsibilities and will need to be agreed with ESA/ESOC. If ESOC/ESA shall perform the orbit control of the SS, there will be additional support needed to operate all the units as independent flying satellites, including the development of a representative SS Simulator.

15.4.5 Disposal Phase SS

As per section 15.3.5.

15.5 Baseline Design MC

15.5.1 Mission Operations Concept MC

The MC shall be operated by ESA/ESOC within the Interplanetary family of missions.

The mission operations are based on strictly pre-planned operations. All operations will be conducted by ESA/ESOC according to procedures included in the FOP (Flight Operations Plan).

The MC mission operations will comprise:

- Spacecraft Operations, consisting of mission planning, spacecraft monitoring and control, and orbit and attitude determination and control. Planning of the spacecraft trajectory and attitude will be fully under MOC responsibility, these will be exposed to the science community as inputs during the relevant planning cycles.
- Science instruments are not foreseen in the MC, however, the MC will have a NAVCAM, which can always be used as a Science instrument. If so, the Science Plan can be developed by Project Scientist and SWT and implemented by the MOC.

Mission Operations of the MC will commence at separation of the satellite from the launcher and will continue until the end of the mission, when the ground contact to the spacecraft will be aborted. Mission Operations will comprise the following tasks:

- Mission Planning, minimum planning tasks
- Spacecraft status monitoring; anomalies will be normally detected with delay
- Spacecraft control, based on monitoring and according to procedures contained in the FOP (Flight Operations Plan). Nominal spacecraft control will be 'off-line' with SPACONs checking the correct performance of the operational steps, and applying predefined procedures in case of minor problems. In case of important problems an engineer is called. Ground automation will be used, as far as possible, of similar flying missions at that time; manual operations will be needed when the criticality requires.
- Offline performance analysis
- Orbit determination and control using tracking data and implementation of orbit manoeuvres
- Attitude determination and control based on the processed attitude sensors data in the spacecraft telemetry and by commanded updates of control parameters in the on-board attitude control system
- On-board S/W maintenance
- Maintenance of ESA ground facilities
- Data dissemination and archiving.

A 3 year period is assumed for mission preparation (as per AIM-Next). The preparation phase includes in particular the following verification activities:

- Mission Sequence Tests
- SVTs (System Validation Tests)
- RF Compatibility Test (RFCT)
- Simulation Campaign.

No additional simulation campaign is foreseen in preparation of the operations at the target body, due to the limited GS&Ops support envisaged for this mission.

15.5.2 Ground Segment Design Overview MC

The ESA/ESOC ground segment will consist of:

- ESTRACK
  - Ground Stations
  - Communications Network
  - ECC (ESTRACK Control Centre)
- Flight Control Team (Multi-mission) supported by hardware/software:
  - MCS (SCOS 2000 or EGS-CC Mission Control System)
  - Mission Planning System and Ground Automation of similar flying missions will be used as far as possible
  - Simulator. For such a mission, the simulator development is assumed to be complex; the cost on the simulation development will only cover essential functionalities: The Simulator will support LEOP, Cruise and approach phase by simulating the MC
  - OBSW (On Board Software Maintenance) tools
- Flight Dynamics
  - Mission Analysis, for mission preparation
  - Flight Dynamics team, for mission operations phase
  - Respective computer hardware
- Data Systems and Infrastructure
  - Procedure tool (MOIS Mission Operations Infrastructure System or similar)
  - Archive and DDS (Data Distribution System)
  - The MCS (Mission Control System will be based on latest available developed system within the Solar and Planetary Missions Division and the cost in development has been considered low (albeit dependent upon level of customisation necessary). Mission specific software will be developed wherever absolutely necessary; the intention is that customisation will be minimum.
  - OPSLAN (operational LAN) and interface hardware/software
  - Development, Launch support and Maintenance for all mission data systems.

15.5.3 Ground Station Coverage Concept MC

All ground communications with MC are via X-Band.

The Deep Station allocation will be decided once there is a final target selection and it will be based on the ground station coverage performed by Mission Analysis. The LEOP ground station coverage will be quasi-continuous and will have to be analysed once the final launcher is assigned, and it will need to consider the co-passenger strategies if the launch is shared with another mission. Non-contact periods for “passive” cruise should be in the order of 7 (EP)-14 days; anything above/below is likely to cause major impacts on the spacecraft ground segment design. Before arrival and during the target body operations, the ground visibility will increase according to the downlink data volume, data latency requirements and critical operations execution.

The ground station handle up- and downlink as well as spacecraft tracking, as defined in the ESA Tracking Stations (ESTRACK) Facilities Manual (EFM), RD[56].

There are redundant communication lines to the ground stations.

The ESTRACK Control Centre (ECC) schedules and requests the respective stations. The station pointing is controlled based on inputs from Flight Dynamics. The ECC is also responsible for the TM/TC links to and from the ground stations (and in case of need any data retrieval of data stored at the ground station).

15.6 Baseline Design SS

15.6.1 Mission Operations Concept SS

The SS Operations Concept is similar to the MC, the text of this section it is similar to section 15.5.1 but specific for SS, it is recalled here for sake of readability.

The SS mission operations will comprise:

- Spacecraft Operations, the Pis are for the operations of their instrument (routine operations, software changes, anomaly investigations) with the help/support of ESA/ESOC.

  If this should not be the case, and the SS will be operated by ESA/ESOC as additional spacecraft units, the Concept of Operations will then be similar to the MC and it will have to discussed and agreed with ESA/ESOC and the cost will adapted accordingly.
- Science operations fairly static and simple, well defined in advance and not likely to change much. Science pointing is defined by MOC via inputs received from the science community.

Mission Operations of the SS will commence during transfer where there will be limited SS check-ups and will continue until the end of the mission, when the ground contact to the spacecraft will be aborted. Mission Operations will comprise the following tasks:

- Mission Planning, minimum planning tasks
- Spacecraft status monitoring (Anomalies will be normally detected with delay)
- Spacecraft control, based on monitoring and according to procedures contained in the FOP (Flight Operations Plan). Nominal spacecraft control will be ‘off-line’ with SPACONs checking the correct performance of the operational steps, and applying predefined procedures in case of minor problems. In case of important problems an engineer is called. Ground automation will be used, as far as possible, of similar flying missions at that time; manual operations will be needed when the criticality requires.
- Maintenance of ESA ground facilities
- Data dissemination and archiving.

The following tasks are expected to be performed by the Pis:

- Offline performance analysis
- Orbit determination and control using tracking data and implementation of orbit manoeuvres
- Attitude determination and control based on the processed attitude sensors data in the spacecraft telemetry and by commanded updates of control parameters in the on-board attitude control system
- On-board S/W maintenance.

A 3 year period is assumed for mission preparation (as per AIM-Next). The preparation phase includes in particular the following verification activities:

- Mission Sequence Tests
- SVTs (System Validation Tests)
- RF Compatibility Test (RFCT): n/a for SS
- Simulation Campaign, n/a for SS as the baseline simulator will not be fully represent the SS operations.

15.6.2 Ground Segment Design Overview SS

The ESA/ESOC ground segment will consist of:

- Flight Control Team (Multi-mission dedicated) supported by hardware/software, integrated within the MC Flight Control Team, see section 15.4.4. With the following exceptions:
    - Simulator. For such a mission, the simulator development is assumed to be complex; the cost on the simulation development will only cover essential functionalities: payload models and SS will be very simple and the payload TM/TC interface will be functionally simulated.
    - OBSW tools, n/a for SS as it is assumed that the SS will be operated by the Pis with ESA/ESOC support.
- Flight Dynamics, n/a for SS, see section 15.4.4.
- Data Systems and Infrastructure, integrated within the MC Data Systems and Infrastructure, see section 15.5.2.

15.6.3 Coverage Concept SS

All communications with the SS are via MC thru the ISL.

15.7 Sensitivity Analysis for MC: What if?

- Poor Electrical Propulsion performance during transfer phase:
    - Continuous restart of electric propulsion unit
    - Daily ground station coverage
    - Team will need to be expanded to cover the continuous Ods and operations.

15.8 Sensitivity Analysis for SS: What if?

- Target Body Size does not allow hyperbola-like strategy, impact on cost affected by:

  - Body knowledge for operations, previous assumptions no longer valid
  - Planning of the spacecraft trajectory and attitude will be fully under MOC responsibility and known in advance
  - Simulator with representative SS operations (TBC).

15.9 Sensitivity to Target: What if Phobos?

All previously mentioned considerations should be taken into account, with the addition of the following assumptions that have a direct impact on the manpower support:

- Mars Orbit Insertion implies execution of a critical manoeuvre
- Spiral down: long duration, uneven orbits that will require passes any time of the day (passes should take place during working hours)
- Quasi Stationary Orbit is unstable and requires regular correction manoeuvres
- Eclipses, conjunctions, attitude management in case spacecraft are sensitive to albedo.

The advantages are the known ephemerides and the possible usage of flying in-orbit relays.

15.10 Architecture Sensitivity Lander

Precise Lander Delivery

- For AIM Next, the precise navigation for lander delivery was discarded. Because:
  - Exhausting activities from the operations side: Flight Control Team, intensive flight dynamics support, fly-bys, elliptical orbits
  - It also implies rehearsals, simulations, go-nogo status.

The FASTMOPS Study covers the lander delivery timeline and requirements.

- The option of autonomous lander was mentioned during the Study but details were not provided and the operations execution seems negligible, on the other hand, the design and test of this autonomy will be arduous.

A lesson learned from Rosetta is that landers shall not be treated as payloads/instruments and ESA should be more involved in the design and in the operations. It should assess whether this could be applied for this mission without incrementing the existing resources, Lander only or also the SSs.

No Precise Lander Delivery

- Operations will depend on the separation and descend strategy
- Ensure the reception of the Science data by the MC.

15.11 Major Design Constraints: CAUTIONS!

- Kick-stage assumptions:
  - KS operations should not introduce additional complexity to the MS operations mentioned in this Report.

15.12 Technology Requirements

There are ground technologies beneficial to the Ground Segment and Operations for operating SPP missions. All those that will improve and reduce the limitations imposed by cost constraints and that will not add additional work to the overall mission design.

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17 ACRONYMS

Acronym	Definition
alpha	UV-absorbtance (solar absorbtance)
AIT/V	Assembly, Integration and Test/Validation
AIV	Assembly, Integration and Validation
AKE	Absolute Known Error
AOCS	Attitude, Orbit Control System
APE	Absolute Pointing Error
APS	Active Pixel Sensor
AST	Advanced Space Technologies GmbH
AU	Astronomical Unit
BoL	Beginning of Life
CAM	Collision Avoidance Manoeuvre
CAN	Controller Area Network
CCD	Charge Coupled Device
CDF	Concurrent Design Facility
CFRP	Carbon Fibre Reinforced Plastic
CHU	Camera Head Units
CoG	Centre of Gravity
COTS	Commercial off-the-Shelf
CP	Chemical Propulsion
DDOR	Delta Differential One Way Ranging
DDS	Data Distribution System
DH	Data Handling
DPU	Digital Processing Unit
DSM	Deep Space Manoeuvre
DST	Deep Space Transponder
DVS	Digital Video System
ECC	ESTRACK Control Centre
ECSS	European Cooperation on Space Standardisation
EDRS	European Data Relay Satellite

Acronym	Definition
EGEP	Enhanced Galileo Electric Propulsion
EoL	End of Life
EP	Electric Propulsion
EPC	Electrical Power Conditioning
EPS	Electrical Power Systems
epsilon	IR-emissivity
ESP	Emmission of Solar Protons – Solar Proton Model
FCU	Flow Control Unit
FD	Flight Dynamics
FDIR	Failure Detection, Isolation and Recovery
FEEP	Field Emission Electric Propulsion
FM	Flight Model
FOP	Flight Operations Plan
FoV	Field of View
FPGA	Field Programmable Gate Array
FSS	Fine Sun Sensor
GCR	Galactic Cosmic Radiation
GFRP	Glass Fibre Reinforced Plastic
GIE	Gridded Ion Engine
GMM	Geometrical Thermal Model
GNC	Guidance, Navigation and Control
G/S	Ground Station
GSP	General Studies Programme
GTO	Geostationary Transfer Orbit
HDRM	Hold Down and release Mechanism
HGA	High Gain Antenna
HKP	Housekeeping
HP	Heat Pipe
HPA	High Power Amplifier
HPR	High-Pressure Regulator
HW	HardWare

Acronym	**Definition**
HWIL	Hard Ware In The Loop
I2C	Inter Integrated Circuit
ICD	Interface Control Document
I/F	Interface
IAU	International Astronomical Union
IMU	Inertial Measurement Unit
IP	Image Processing
IR	Infra Red
ISL	Inter Satellite Link
ISO	International Organisation for Standards
ITU	International Telecommunications Union
LAN	Local Area Network
LET	Linear Energy Transfer
LEO	Low Earth Orbit
LEOP	Launch and Early Operations Phase
LGA	Low Gain Antenna
LHP	Loop Heat Pipe
LoS	Line of Sight
LPF	Lisa PathFinder
LV	Launch Vehicle
MAB	Main Asteroid Belt
MBA	Main Belt Asteroid
MC	MotherCraft
MCS	Mission Control System
MEMS	Micro Electrical Mechanical System
MGA	Medium Gain Antenna
MLI	Multilayer-Insulation
MM	Memory Module
MOC	Mission Operations Centre
MOIS	Mission Operations Infrastructure System
MPC	Minor Planet Centre

Acronym	Definition
MPPT	Maximum Power Point Tracker
NEA	Near Earth Asteroid
NEO	Near Earth Object
NIEL	Non Ionizing Dose
nT	Nano Tesla
OBC	On-Board Computer
OBSW	OnoBoard SoftWare
OCDT	Open Concurrent Design Tool
OD	Orbit Determination
OSR	Optical Solar Reflector
OTS	Off The Shelf
P/L	Payload
PCDU	Power Conditioning and Distribution Unit
PDT	Payload Data Transmitter
PPT	Pulsed Plasma Thruster
PPU	Power Processing Unit
QSO	Quasi-Satellite Orbit
Rad-Hard	Radiation Hardened
Rad-Tol	Radiation Tolerant
RDV	Rendezvous
RFCT	Radio Frequency Compatibility Tests
RIT	Radiofrequency Ion Thruster
RMS	Root Mean Square
RNG	Ranging
RPE	Relative Pointing Error
RTU	Remote Thermal Unit
RW	Reaction Wheels
S/C	Spacecraft
SA	Solar Array
SAC	Solar Array Controller
SADM	Solar Array Drive Mechanism

Acronym	Definition
SCOS	Spacecraft Control and Operations System
SEE	Single Event Effect
SEL	Single Event Latchup
SEL2	Sun Earth Libration point 2
SEP	Sun-Earth-Probe angle
SEU	Single Event Upset
SMA	Semi-Major Axis
SoC	System on Chip
SPENVIS	Space Environment Information System
SPP	Small Planetary Platforms
SS	SmallSats
SSM	Secound Surface Mirror
SSTO	Self-Stabilised Terminator Orbit
STR	Star Tracker
SVT	System Validation Test
SWT	Science Working Team
TAS	Thales Alenia Space
TBD	To Be Determined
TCM	Trajectory Correction Manoeuvre
TCS	Thermal Control System
TM/TC	Telemetry/ Telecommand
TMM	Thermal Mathematical Model
TPM	Thruster Pointing Mechanism
TRL	Technology Readiness Level
TT&C	Telemetry, Tracking and Command
TWTA	Travelling Wave Tube Amplifier
VBN	Visual Based Navigation
VDA	Vapour-Deposited-Aluminium
VNC	Visual Navigation Camera

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A REQUIREMENTS TREE

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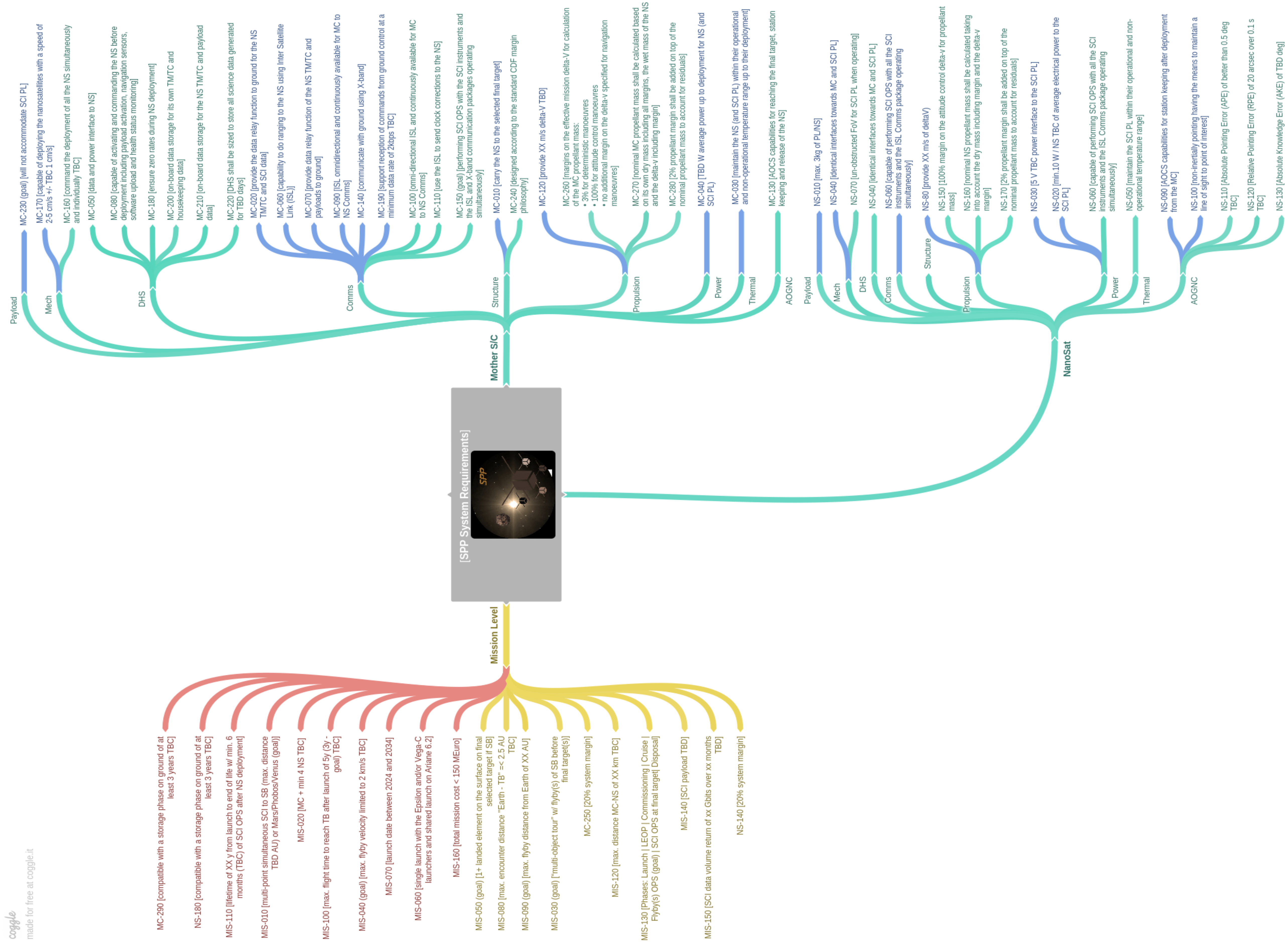
[SPP System Requirements]
SPP
Mother S/C
Payload
MC-230 (goal) [will not accommodate SCI PL]
Mech
MC-170 [capable of deploying the nanosatellites with a speed of 2-5 cm/s +/- TBC 1 cm/s]
MC-160 [command the deployment of all the NS simultaneously and individually TBC]
DHS
MC-050 [data and power interface to NS]
MC-080 [capable of activating and commanding the NS before deployment including payload activation, navigation sensors, software upload and health status monitoring]
MC-180 [ensure zero rates during NS deployment]
MC-200 [on-board data storage for its own TM/TC and housekeeping data]
MC-210 [on-board data storage for the NS TM/TC and payload data]
MC-220 [DHS shall be sized to store all science data generated for TBD days]
Comms
MC-020 [provide the data relay function to ground for the NS TM/TC and SCI data]
MC-060 [capability to do ranging to the NS using Inter Satellite Link (ISL)]
MC-070 [provide data relay function of the NS TM/TC and payloads to ground]
MC-090 [ISL omnidirectional and continuously available for MC to NS Comms]
MC-140 [communicate with ground using X-band]
MC-190 [support reception of commands from ground control at a minimum data rate of 2kbps TBC]
MC-100 [omni-directional ISL and continuously available for MC to NS Comms]
MC-110 [use the ISL to send clock corrections to the NS]
MC-150 (goal) [performing SCI OPS with the SCI instruments and the ISL and X-band communication packages operating simultaneously]
Structure
MC-010 [carry the NS to the selected final target]
MC-240 [designed according to the standard CDF margin philosophy]
Propulsion
MC-120 [provide XX m/s delta-V TBD]
MC-260 [margins on the effective mission delta-V for calculation of the MC propellant mass:
• 3% for deterministic manoeuvres
• 100% for attitude control manoeuvres
• no additional margin on the delta-v specified for navigation manoeuvres]
MC-270 [nominal MC propellant mass shall be calculated based on its own dry mass including all margins, the wet mass of the NS and the delta-v including margin]
MC-280 [2% propellant margin shall be added on top of the nominal propellant mass to account for residuals]
Power
MC-040 [TBD W average power up to deployment for NS (and SCI PL)]
Thermal
MC-030 [maintain the NS (and SCI PL) within their operational and non-operational temperature range up to their deployment]
AOGNC
MC-130 [AOCS capabilities for reaching the final target, station keeping and release of the NS]
NanoSat
Payload
NS-010 [max. 3kg of PL/NS]
Mech
NS-040 [identical interfaces towards MC and SCI PL]
NS-070 [un-obstructed FoV for SCI PL when operating]
DHS
Comms
NS-040 [identical interfaces towards MC and SCI PL]
NS-060 [capable of performing SCI OPS with all the SCI instruments and the ISL Comms package operating simultaneously]
Structure
Propulsion
NS-80 [provide XX m/s of deltaV]
NS-150 [100% margin on the attitude control delta-v for propellant mass]
NS-160 [nominal NS propellant mass shall be calculated taking into account the dry mass including margin and the delta-v margin]
NS-170 [2% propellant margin shall be added on top of the nominal propellant mass to account for residuals]
Power
NS-030 [5 V TBC power interface to the SCI PL]
NS-020 [min.10 W / NS TBC of average electrical power to the SCI PL]
NS-060 [capable of performing SCI OPS with all the SCI instruments and the ISL Comms package operating simultaneously]
Thermal
NS-050 [maintain the SCI PL within their operational and non-operational temperature range]
AOGNC
NS-090 [AOCS capabilities for station keeping after deployment from the MC]
NS-100 [non-inertially pointing having the means to maintain a line of sight to point of interest]
NS-110 [Absolute Pointing Error (APE) of better than 0.5 deg TBC]
NS-120 [Relative Pointing Error (RPE) of 20 arcsec over 0.1 s TBC]
NS-130 [Absolute Knowledge Error (AKE) of TBD deg]
Mission Level
MC-290 [compatible with a storage phase on ground of at least 3 years TBC]
NS-180 [compatible with a storage phase on ground of at least 3 years TBC]
MIS-110 [lifetime of XX y from launch to end of life w/ min. 6 months (TBC) of SCI OPS after NS deployment]
MIS-010 [multi-point simultaneous SCI to SB (max. distance TBD AU) or Mars/Phobos/Venus (goal)]
MIS-020 [MC + min 4 NS TBC]
MIS-100 [max. flight time to reach TB after launch of 5y (3y - goal) TBC]
MIS-040 (goal) [max. flyby velocity limited to 2 km/s TBC]
MIS-070 [launch date between 2024 and 2034]
MIS-060 [single launch with the Epsilon and/or Vega-C launchers and shared launch on Ariane 6.2]
MIS-160 [total mission cost < 150 MEuro]
MIS-050 (goal) [1+ landed element on the surface on final selected target if SB]
MIS-080 [max. encounter distance "Earth - TB" =< 2.5 AU TBC]
MIS-090 (goal) [max. flyby distance from Earth of XX AU]
MIS-030 (goal) ["multi-object tour" w/ flyby(s) of SB before final target(s)]
MC-250 [20% system margin]
MIS-120 [max. distance MC-NS of XX km TBC]
MIS-130 [Phases: Launch | LEOP | Commissioning | Cruise | Flyby(s) OPS (goal) | SCI OPS at final target| Disposal]
MIS-140 [SCI payload TBD]
MIS-150 [SCI data volume return of xx Gbits over xx months TBD]
NS-140 [20% system margin]
coggle
made for free at coggle.it