CDF Study Report

AIM3P

Asteroid Impact Mission – Payload Assessment

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Further information and/or additional copies of the report can be requested from:

A. Gálvez
ESA/HQ/TEC-SF
8-10 Rue Mario-Nikis
75738 PARIS CEDEX 15
France
Tel: +33-(0)1-53697623
Fax:
Andres.Galvez@esa.int

For further information on the Concurrent Design Facility please contact:

M. Bandecchi
ESA/ESTEC/TEC-SYE
Postbus 299
2200 AG Noordwijk
The Netherlands
Tel: +31-(0)71-5653701
Fax: +31-(0)71-5656024
Massimo.Bandecchi@esa.int

FRONT COVER

Study logo showing DART Impactor about to strike secondary asteroid of the Didymos Binary

STUDY TEAM

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

TEAM LEADER	M. van Pelt, TEC-SYC		
AOCS	I. Heurtas, TEC-ECN B. Brito	POWER	M. Tourloukis, TEC-EPM H. Carbonnier
COMMUNICATIONS	P. Concari, TEC-ETC J. Lizarrage Cubillos	PROGRAMMATICS/ AIV	O. Brunner, TEC-MXC D. Hagelschuer A. Böerngen
CONFIGURATION	D. de Wilde, TEC-MSS	PROPULSION	T. Langener, TEC-MPA R. Schonenborg
COST	G. Reinbold, TEC-SYC G. Oger	RISK	F. Restagno, TEC-QQD
DATA HANDLING	C. Monteleone, TEC-EDD	MISSION ANALYSIS	M. Khan, HSO-GFA W. Martens
GS&OPS	D. Patterson, HSO-OSA	STRUCTURES	T. Cardone, TEC-MSS H. Fischer
INSTRUMENTS	M. Leutzner, TEC-SF A. Gálvez	SYSTEMS	D. Binns, TEC-SYE J. Vennekens A. Cotuna
O/B AUTONOMY	Y. Yushtein, TEC-SWE M.Perrotin, C. Honvault	THERMAL	M. Giacomazzo, TEC-MTT A. Pfaltzgraff
		SIMULATION	B. Garcia Gutierrez, TEC-SWG

Under the responsibility of:

A. Gálvez, TEC-SF, Study Manager

I. Carnelli, TEC-SF, AIM Project Manager

With the scientific assistance of:

P. Michel Observatoire de la Cote D' Azur. Scientific Advisor

With the technical support of:

T. Dreischer, RUAG, Optical Link

Z. Sodnik, TEC-MME, Optical Link

I. Zayer, HSO-GSS, Optical Link

A. Herique, LAOG, Radar Instruments

Z. Fletcher, JHU APL Systems

M.Kueppers, SRE-OOR, Science Operations & Remote Sensing

R. Moissl SRE-OOR, Remote Sensing

F. Romera, TEC-MSM, Mechanisms

M. Toso, TEC-MSM, Mechanisms

The editing and compilation of this report has been provided by:

A. Pickering, TEC-SYE, Technical Author

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1 INTRODUCTION

1.1 Background

ESA has been studying the role of space missions with respect to the NEO impact hazard in the framework of the General Studies Programme since 2002, with projects such as the Don Quijote mission study. In line with the Director General's strategy laid out in ESA´s "Agenda 2015", work now focuses on international cooperation, precursor missions and In Orbit Demonstration (IOD) of enabling technologies and operations for ESA science and exploration missions.

The Asteroid Impact Mission (AIM) is part of the joint Asteroid Impact & Deflection Assessment (AIDA) project of ESA, DLR, Observatoire de la Côte d´Azur (OCA), NASA, and John Hopkins University Applied Physics Laboratory (JHU/APL). The intention of AIDA is to send two spacecraft to the binary asteroid 65803 Didymos (1996 GT):

- An asteroid impactor - the NASA/APL DART (Double Asteroid Redirection Test) Mission led by the Applied Physics Laboratory in the US
- An Asteroid rendezvous and observation spacecraft - the ESA AIM mission.

Requested by TEC-SF (General Studies Programme Unit) and funded by GSP, the AIM-3P study described in this report is a second iteration of the AIM-3 study performed during May and June 2014. This second iteration is to incorporate into the design all aspects related to the payload instruments and experiments and was carried out in the ESTEC CDF in four sessions between September and October 2014. The Internal Final Presentation took place on October 29, 2014. The current AIM-3 and AIM-3P studies followed the previous CDF AIM study performed in March and April 2012 RD[1], the results of which proved to be a valuable starting point.

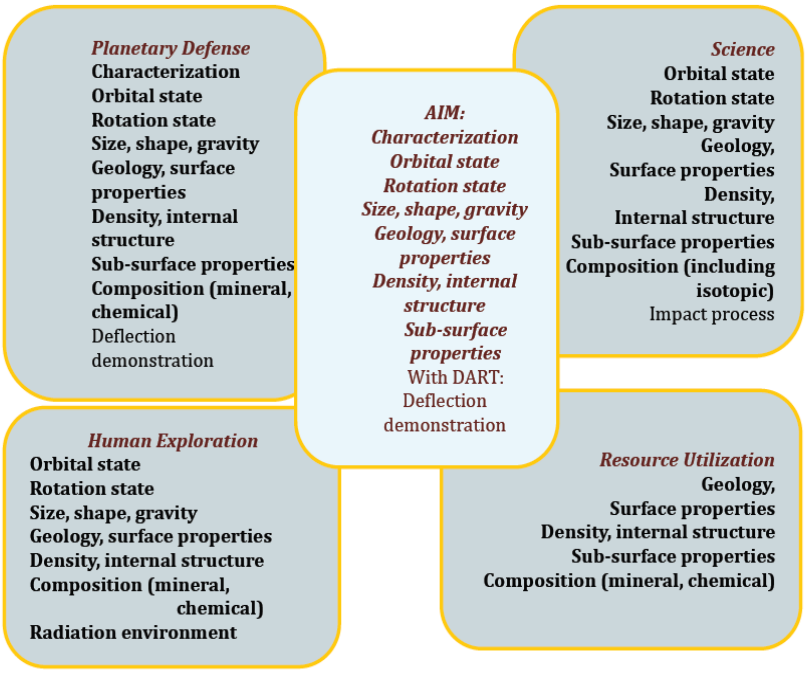

Figure 1-1: Knowledge gain from the AIM mission

1.2 Scope

AIM-3P, being the third in a list of three possible AIM mission options, is envisaged to be a small, low-cost spacecraft for characterising the binary asteroid Didymos (1996 GT). The main mission objectives are to determine the binary asteroid orbital and rotation dynamics as well as the mass, geophysical properties and surface and subsurface structure, both before and after the DART impact. In addition AIM-3P is to demonstrate novel spacecraft technologies and operations to advance science and exploration missions, particularly optical communications and the use of an Inter-Satellite Link (ISL) with CubeSats in interplanetary space. It would be the first mission to a binary asteroid. Note this report covers both the work done in the previous AIM-3 study as well as the second iteration AIM-3P study.

The AIM-3P mission would involve:

- Reaching the binary asteroid Didymos (1996 GT)
- Determining the binary asteroid's orbital and rotation dynamics, as well as its physical properties by remote sensing, using as a minimum a Visual Imaging System (VIS) and a Thermal IR Imager (TIRI), and in addition a Monostatic High Frequency Radar (HFR) and a Bistatic Low Frequency Radar (LFR) on the Platform.
- Placing a lander (including a Bistatic Low Frequency Radar and possibly other instruments) on the Secondary, i.e. the smaller of the two asteroids
- As a goal, observing the impact of the DART spacecraft on Didymos Secondary, i.e. the smaller of the two asteroids (not a formal requirement, but nonetheless deemed valuable while having very little impact on the overall mission design)
- Observing the impact crater and ejecta plume generated by the impact
- Performing a second characterisation campaign after the impact, to study the change in the Didymos' orbital and rotation dynamics.

And as Technology Demonstrations:

- Demonstrate Optical Laser Communications in deep space
- Deploy a number of CubeSats in deep space and maintain an ISL between them and the mothercraft
- Deploy a lander on the surface of the Secondary.

The task for the CDF involved:

- Evaluating the launch date and trajectory options for reaching the asteroid and completing its characterisation before the arrival of the DART mission in October 2022
- Designing a (low-cost & low mass) asteroid rendezvous spacecraft able to carry and deploy the entire payload for determining the asteroid dynamics, geophysical properties and surface and subsurface structure, as well as the OPTEL-D Optical Laser Communication terminal, a lander and the CubeSats for technology demonstration purposes.
- Determining the AIM-3P technology needs and options, budgets, development plan, risks and costs

- Incorporate into the design all aspects related to the payload instruments and demonstration experiments.

1.3 Document Structure

The layout of this report of the study results can be seen in the Table of Contents. The Executive Summary chapter provides an overview of the study; 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.

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2 EXECUTIVE SUMMARY

2.1 Study Flow

The study was a second iteration of the 8-sessions AIM-3 study carried out between May and June 2014 and involved 4 CDF sessions (and various splinter meetings), starting with a kick-off on September 4, 2014 and ending with an Internal Final Presentation on October 29, 2014. This report considers the outcome of both studies as the AIM-3P study was an iteration of the previous study to specifically look at the payload instruments and experiments. The first few sessions of the AIM-3 study focussed on mission analysis, the definition of the payload, and the study of the feasibility of a Vega launch using a standard solid propellant kick stage. After the latter was found unfeasible, the study then focussed on a mission involving a Soyuz launch from Kourou and the iteration of the related spacecraft design. The AIM-3P study was specifically related to defining and incorporating a more mature set of payload instruments and experiments, and ensuring that their end-to-end performance and operations and their system impacts (mass, power, pointing accuracy and stability, thermal, mechanical stability, contamination, data handling, ground segment etc.) are compatible with the AIM-3 design. The original AIM-3 spacecraft design had to be adapted to incorporate the more consolidated payload suite and the more refined payload requirements and constraints. The platform design was further optimised and detailed w.r.t. the earlier AIM-3 study, especially in the area of Power, TT&C, Thermal, Structures and Mission Analysis, while a new, detailed spacecraft orbit and attitude simulation was added to the CDF output.

2.2 Requirements and Design Drivers

The main study requirement was to provide a mission concept and spacecraft definition for AIM-3 as a small, low-cost spacecraft for characterising the binary asteroid Didymos (1996 GT). The main mission objectives were the determination of the binary asteroid's orbital and rotation dynamics as well as the mass, geophysical properties and surface and subsurface structure, both before and after the DART impact. In addition AIM-3 is to demonstrate novel spacecraft technologies and operations to advance science and exploration missions. It would be the first mission to a binary asteroid.

The CDF was requested to:

- Evaluate the launch date and trajectory options for reaching the asteroid and completing its characterisation before the arrival of the DART mission in October 2022.
- Design a (low-cost & low mass) asteroid rendezvous spacecraft able to carry and deploy a suite of instruments (Visual Imaging System and a Thermal IR Imager, and in addition a Monostatic High Frequency Radar and a Bistatic Low Frequency Radar on the Platform, plus a Lander) for determining the asteroid dynamics, mass, geophysical properties and surface and subsurface structure.
- Incorporate as technology demonstration payload a laser communication terminal, a lander and a number of CubeSats.

- Determine the AIM-3 technology needs and options, budgets, development plan, risks and costs.

The main mission constraints were imposed as follows:

- Launch on an Arianespace launcher.
- Arrive at Didymos (1996 GT) at least 2 months before the DART impact (leading to the need to launch before the end of 2020). Based on further mission timeline evaluations during the study, this was further refined as a need to arrive at the asteroid by the end of May 2022, assuming a DART impact on October 6, 2022.
- Technologies shall have a TRL of at least 6 by the end of Phase B1.

A more detailed list of the mission requirements and constraints can be found in the System chapter.

2.3 Mission

A summary of the AIM-3 mission baseline and spacecraft concept is presented in Table 2-1 and Table 2-2:

Mission description		
Launch	Launcher	Soyuz, Kourou
	Launch date	October - November 2020
Orbit	Orbit type	Interplanetary, 1 AU perihelion, 2.2 AU aphelion
Operations	Ground stations	ESTRACK (possibly with support from NASA facilities)
	Mission operations centre	ESOC
Lifetime	1.5 year transfer + 5.5 months of operation at the asteroid (+ up to 2.5 months extension).	
Overall system characteristics		
Mass	Dry mass	425 kg, including margins, excl. adapter
	Wet mass	717 kg, including margins, excl. adapter 827 kg, including margins, incl. adapter
Delta-V	1350 m/s (1250 m/s for transfer + 100 m/s at asteroid).	
Dimensions	Stowed	1804 x 2050 x 2102 mm^3
	Deployed	7740 x 3278 x 2102 mm^3

Table 2-1: AIM-3 Mission and system summary

Payload and subsystems	
Payload	• 2.4 kg Visual Imaging System

	• 3.6 kg Thermal IR Imager • 1.7 kg monostatic High Frequency Radar • 1.2 kg bistatic Low Frequency Radar receiver • 13.0 kg microlander incl. a 2.0 kg Low Frequency Radar emitter • 39.3 kg Optical Communications Terminal • 13.2 kg for 2 x 3U CubeSats incl. dispensers resulting in a total payload mass of 74.4 kg including margins. Note: VIS Camera is also used for navigation. Minimum data generated to meet mission objectives, incl. lander, exl. mission extensions: 21.3 Gbit
GNC	3-axis stabilised spacecraft Actuators: 4 reaction wheels (Teldix 12-75/60) Sensors: • 2 Star Tracker Optical Heads + 2 Electronics Units (Hydra) • 4 Sun Sensors • 2 IMU • 1 VIS Camera (also used as a payload instrument)
Communications	2 miniDST X-Band 2 HPA - TWTA 1 RFDN 2 LGA 1 HGA – metasurface, 1.15 m Microlander interface: • 1 ISL electroc box • 2 ISL antenna
DHS	1 CDMU based on AS250 OSCAR computer (Seosat, Cheops), with 16 Gbit uncompressed data storage. 1 RTU based on AS250 from Astrium or AFIO (Advanced Flexible I/O) from RUAG Sweden.
Power	Solar panels: 2 deployable, fixed wings of 2.8 m^2 each, 2 panels each, GaAs cells 1 Battery, Li-ion, 15 Ah 1 PCDU, 28V unregulated bus
Propulsion	MON/MMH bipropulsion with • 4 x Airbus DS OTS 31/0 propellant tanks • 1 pressurant tank • 24 x 10 N thrusters
Structure	Aluminium honeycomb panels, tank support brackets and launcher

	interface ring.
Thermal	MLI on external surfaces Black paint on internal surfaces OSR radiators on anti-Sun side and Solar Array side panels Heating lines Thermistors, thermal washers, thermal fillers

Table 2-2: AIM-3 platform overview

2.4 Technical Conclusions and Options

The CDF study has shown that an AIM-3P concept based on a Soyuz launch from Kourou can result in an overall robust mission design without immediate showstoppers or unmanageable risks, with a satisfactory asteroid observation scenario, and with sufficient margins at various levels. The mission can be based on mostly high-TRL equipment, while incorporating a limited amount of innovative equipment for demonstration purposes (for which high-TRL alternatives are anyway available, with minor impacts on the overall spacecraft and mission design).

3 MISSION RATIONALE

ESA's AIM has a triple character. It demonstrates technology, provides science and it is also linked to SSA-NEO´s objectives i.e. asteroid impact risk reduction.

AIM as an ESA Technology Mission of Opportunity addresses among other telecommunications technology for science and exploration spacecraft.

These aspects of the mission are described below.

3.1 Background

ESA's Director General's strategic plan for the next four years, Agenda 2015 (*Director General, ESA (2011). ESA BR-303 Agenda 2015. A Document by the ESA Director General*), calls for "initiating a planetary defence mission (possibly in cooperation with non-European partners) [which] would increase Europe's competitiveness since such a mission would require the development of new technologies also relevant to other missions."

The asteroid impact risk is low but the potential consequences to our society can be very severe. Small bodies are continually colliding with the Earth, however, the vast majority of these objects are very small and pose no threat to human activity. Larger impacts are more rare but, when they occur, can lead to a major natural catastrophe. For instance the energy released from the Tohoku earthquake in Japan (3rd March 2011) was estimated to be approximately 45 megatons; this natural disaster caused an estimated economic loss of over $200 billion according to the World Bank. The effects of an asteroid impact on Earth depends on many factors such as e.g., location, asteroid trajectory or physical properties, but a small 150 m object could release several times the amount of energy released in Tohoku.

In the last decade, ESA has supported work on systems addressing the Near-Earth Asteroid (NEA) Earth impact risk:

- Since 2002 and in the context of the Agency's General Studies Programme (GSP) mission studies have been conducted on the "Don Quijote" concept (Galvez & Carnelli, 2006) and other asteroid probes, in close cooperation with the directorate of Scientific Programme and its Cosmic Vision plan.
- Since 2009, in the Space Situational Awareness (SSA) Preparatory Programme, on the ground system and user community interfaces.
- In the frame of ESA's technology programmes relevant concepts for In-Orbit technology Demonstration have been considered; also, applicable techniques in spacecraft autonomy have been investigated but have not yet been flight-tested.

In the US, the call of the National Research Council Report for developments in asteroid demonstration missions and in particular kinetic impact tests (NRC Committee, 2010); has been followed by the Double Asteroid Redirection Test (DART) (Cheng, 2012) mission definition work conducted by the John Hopkins University Applied Physics Laboratory (JHU/APL) and supported by several NASA centres.

In this context, ESA's Future Preparation and Strategic Studies Office has initiated work on the Asteroid Impact Mission (AIM) study, which has the objective of defining an affordable and independent mission element that ESA would contribute to a joint asteroid impact test characterisation campaign.

The two projects, AIM and DART are therefore combined in the Asteroid Impact & Deflection Assessment (AIDA), a collaboration between ESA, JHU/APL, NASA, the Cote d'Azur Observatory (OCA) in France and the DLR in Germany.

3.2 The DART (Double Asteroid Redirection Test) Concept

To date, only the NASA Deep Impact mission has delivered a kinetic impactor, which targeted the Jupiter Family Comet Tempel-1, but which yielded no measurable deflection of the target. The capability and technologies to deflect an asteroid's trajectory have never been demonstrated. Moreover, even if the amount and direction of deflection needed to mitigate a given impact hazard are known, it is not possible to predict accurately how large a spacecraft impact (how much momentum and energy) is needed in order to accomplish this deflection. This uncertainty results from both the poorly understood dynamics of hypervelocity impacts and the largely unknown physical properties of asteroids.

The Double Asteroid Redirection Test (DART) main goal will be to demonstrate asteroid deflection. In order to demonstrate an impact hazard mitigation, this mission must not only deflect the trajectory of an asteroid, but it must measure the deflection to within 10%. Moreover, this mission is required to meet a stringent cost target of $150M including launch on a Minotaur V. The DART project is undertaken by the Johns Hopkins Applied Physics Laboratory with support from members of NASA centres including Goddard Space Flight Centre, Johnson Space Centre, and the Jet Propulsion Laboratory.

The DART mission will use a single spacecraft to impact the smaller member of the binary near-Earth asteroid (65803) Didymos (1996 GT) in October, 2022. Didymos is an already well-observed radar and optical binary system, consisting of two objects orbiting each other. The impact of the >300 kg DART spacecraft at 6.1 km/s will change the mutual orbit of these two objects. By targeting the smaller, 150 m diameter member of a binary system, the DART mission produces an orbital deflection which is both larger and easier to measure than would be the case if DART targeted a typical, single near-Earth asteroid so as to change its heliocentric orbit. It is important to note that the target Didymos is not an Earth-crossing asteroid, and there is no prospect that the DART deflection experiment would create an impact hazard.

The DART asteroid deflection demonstration targets the binary asteroid Didymos in Oct 2022, during a close approach to Earth. The DART impact will be observable by ground-based radar and optical telescopes around the world, providing exciting opportunities for international participation in the mission, and generating tremendous international public interest, in the first asteroid deflection experiment.

As opposed to earlier mission concepts (e.g. ESA´s Don Quijote), in DART ground-based observations can be used to make the required measurements of the orbital deflection, by measuring the orbital period change of the binary asteroid. The DART impact will change the period by 0.5% - 1%, and this change can be determined to 10% accuracy within months of observations. The DART target is specifically chosen because it is an eclipsing binary, which enables accurate determination of small period changes by ground-based optical light curve measurements. In an eclipsing binary, the two objects pass in front of each other (occultations), or one object creates solar eclipses seen by the other, so there are sharp features in the lightcurves which can be timed accurately.

The DART mission additionally will provide data needed to determine the momentum transfer efficiency from the DART impact. DART will carry a high resolution visible imager, based on the LORRI instrument from the New Horizons mission, in order to return detailed images of the target object as well as its binary companion. DART will autonomously guide itself to impact using this imager. The high resolution images returned to Earth shortly before impact will enable determination of the precise impact point (even in the absence of another spacecraft witnessing the impact) as well as fundamental assessments of the target body geology and surface physical properties. This information could answer questions such as, is the target object a rubble pile asteroid (a gravitationally re-accumulated assembly of fragments from a catastrophic collision)? Is there a mobile regolith surface? This information greatly enhances physical interpretation of the momentum transfer efficiency using the measured binary orbital change, and will be the first step to gaining understanding of, and eventually becoming able to predict, the results of hypervelocity kinetic impacts for asteroid deflection.

The DART mission uses a simple, high-technology-readiness, and low-cost spacecraft to intercept Didymos. DART hosts no scientific payload other than the imager for targeting and data acquisition. The spacecraft is single string, and most of the components are either rebuilds of previous designs or commercial off-the-shelf equipment. Terminal guidance to the target asteroid is accomplished using the LORRI telescope for optical navigation and using autonomous guidance algorithms based on APL experience in development of the Standard Missile. A monopropellant propulsion system is used for all delta-v burns. Three-axis attitude control is performed using thrusters as on New Horizons.

The spacecraft wet mass is estimated to be 235 kg with 30% dry mass margin and with delta-v capability of 100 m/s. Ballast mass would be added to reach the launch capability of 330 kg. Power is estimated at 202 W. The spacecraft has a fixed 1-meter high gain antenna with X-band telemetry. The mechanical layout of the spacecraft is optimised for the terminal navigation phase, with fixed geometries for the imager, high gain antenna, and solar arrays. The spacecraft has no gimbals or deployables.

3.3 AIM as a part of the AIDA Cooperation

When AIM, a Didymos asteroid-observing spacecraft is operated simultaneously with DART, the knowledge on physical parameters relevant to the characterisation of the study is boosted. These data are key in areas beyond asteroid threat mitigation. Theories and tools developed to understand collision and impact physics are extensively used in an industrial context such as in the oil and gas industry, in addition to being a field of major interest for space-related activities such as Solar System science, exploration and resource utilisation. The AIDA Mission Rationale document RD[2] provides a first description of the opportunities that a joint AIM and DART impact test could bring for technology R&D, science and the cooperation between long-standing partners on a problem of global dimensions.

In order to establish the specific goals of the AIM mission and the added value for the ESA-led project of the joint operation of the two spacecraft, ESA has commissioned a study by European experts on different aspects of the impact dynamics and planetary research, under the coordination of the Observatoire de la Côte d´Azur (OCA). The final report of the RD[3] has been the basis for the definition of the requirements of AIM as a key element in a complete impact experiment and characterisation test.

3.4 AIM as an ESA Technology Demonstration

AIM as a stand-alone ESA mission is also an Technology Mission of Opportunity focusing on telecommunications technology for science and exploration spacecraft.

One of the key technology innovations of the mission is the use of a deep-space optical link with Earth that, if demonstrated operationally, will dramatically increase the data returned on the Didymos system and pave the way for its utilisation in future ESA missions. This would also have significant impacts on the Ground Segment and operations concept. AIM will carry such an optical terminal, called Optel-D, a design by RUAG. The company has carried out a first, yet very detailed, assessment, in the frame of an ESA contract which run in parallel and supported the AIM3P CDF study. A specific design solution has been proposed with a modular architecture that could easily be adapted to address future missions needs depending on e.g. mass, power, AOCS design drivers. The nominal AIM mission provides a very relevant operational scenario for the Optel-D demonstration, with space-to-ground downlink distances between 0.1 and 0.5 AU (16 million km to 75 million km) and multi –Mbits data rates; testing link distances of up to 3.2 AU -relevant to missions operating at Venus, Mars orbit and beyond- is an option. The lifetime of the Optel-D system shall be at least 2 1/2 years in an interplanetary space environment, with a demonstrated operational lifetime of more than 400 h. The goal of the demonstration is to complete the downloading of 10 Gbits of mostly VIS data within one month, with a goal of two weeks.

In addition to the optical communication demonstration, AIM will take ESA Inter-Satellite Links development one step beyond and demonstrate networking capabilities between the mother spacecraft, the microlander and the CubeSat payloads, with a course ranging capability (few m) as an additional goal.

The technology demonstrations are meant to provide a very valuable support to the AIM research mission.

Optel-D could be used as a complement to a scientific payload; with minor modifications it could perform functions such as high-resolution NIR imaging, precise time-of-flight and spacecraft ranging, and altimetry for small body topographic mapping (as in Bepi Colombo´s BELA).

MASCOT-2´s scientific potential is clear due to its heritage. As a technology demonstration, MASCOT-2 will be the smallest microlander to be deployed on the surface of a Solar System body and survive autonomously several months. The challenge will be to try to match Rosetta´s Philae feat landing on an object that is about forty times smaller than comet 67P/Churyumov-Gerasimenko, and with a escape velocity of only a few cm/s. In addition to this, as a platform for science, investigations of the surface and subsurface properties, especially those relevant to the determination of the ¨beta factor¨, like the surface mechanical properties and structure, could be addressed in-situ. The lander would also carry one of the two components of the Low-Frequency Radar with the other in the AIM s/c.

The specific objectives of the CubeSat Opportunity Payload Intersatellite Network System (COPINS), either as individual sensors or working as a system, will be the subject of an Announcement of Opportunity and thus are not defined yet. These could include e.g. collectively gathering remote-sensing or in-situ data on the asteroid system and the results of DART impact.

Finally, all these demonstrations will be completed with the use of the AIM platform for the qualification of new components, either in the telecommunication subsystem or in other ones.

If should also be noted that while all these technology payloads will be used to collect information on the asteroid system, in a similar but inverse way, the instruments (VIS, TIRI, HFR..) would provide information relevant to the S/C navigation (imaging for search and collision avoidance, ranging, altimetry etc). AIDA also offers opportunities for joint technology demonstration. For instance in the field of optical communications, where interoperability of future systems will be a key issue for future science and exploration missions. Two different ground stations scenarios would be considered, a baseline assuming use of ESA´s 1-m aperture OGS in Tenerife, and an optional extension if a larger facility becomes available. Cooperation with NASA/APL´s DART scenarios could also involve optical link between AIM and DART etc.

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4 PAYLOAD INSTRUMENTS

According to the mission objectives, AIM shall serve both the goals of technology demonstration and science. While AIM is expected to contribute to the scientific understanding of binary asteroid systems, it is the capability to characterise the dynamical state and the physical properties of an asteroid along with a change of these parameters following a high-velocity collision with an external object, that shall be demonstrated. The following payloads have thus been selected for their ability to provide the required measurements or as demonstrators for an enabling technology.

- Visual Imaging System (VIS)*
- Thermal Imager (TIRI)
- High Frequency Radar (HFR)
- Low Frequency Radar (LFR)
- Optical Link for Communication (Optel-D)
- MASCOT-2 asteroid lander
- CubeSat Opportunity Payloads (COPINS)

*The VIS Imaging System is to be used as a navigation camera also, and its design is driven by GNC needs.

The strawman payload of the AIM3P S/C concept involves the baseline design outlined within this chapter.

4.1 Payload Requirements

The payload design has to take into account performance requirements stemming from the mission objectives and constraints imposed by the S/C and mission design. Top-level requirements for each individual payload are given below along with a rationale for its selection.

4.1.1 Visual Imaging System (VIS)

The objective of the Visual Imaging System is to acquire visual images of the Didymos binary system that will enable the extraction of the following information:

- The determination of mass, volume, size of the secondary component of the system
- The determination of the binary system orbital period and the secondary asteroid rotational period with an accuracy of 10%
- The characterisation of shape, topography, the granularity of surface material and the secondary asteroid.

A full coverage of the secondary asteroid body shall be achieved with a resolution of at least 1 m during the course of the AIM asteroid observation phases. As required for navigation and collision risk mitigation purposes, during the observation phases the VIS shall keep the primary and secondary components within its FOV. In addition to this the VIS limiting magnitude shall be compatible with the expected heliocentric orbit uncertainty (due to both ground navigation and Didymos position uncertainties).

As a consequence, the requirements for the VIS instrument are the following:

- Pixel IFOV: 50 µrads
- Limiting magnitude: 12-15 TBD based on requirement during approach phase
- Spectral wavebands: 450-650 nm (visible)
- Dynamic range: TBD compatible with all asteroid observation phases
- Maximum sampling rate: 10 Hz or higher.

No multi-band photometric requirement has been defined at this stage. The sampling rate has not been defined considering e.g. impact ejecta real-time observations. Both aspects could be addressed in future iterations as more precise data becomes available.

4.1.2 Thermal Imager (TIRI)

The Thermal Imager shall obtain data on the thermal properties of the asteroid surface that are relevant to the characterisation of the surface material. The main objectives to be addressed by TIRI are:

- The determination of the surface thermal inertia
- The characterisation of the granularity and aggregation estate of surface material of the secondary asteroid
- The characterisation of the shallow (few m) subsurface material exposed by the DART impact crater.

A full coverage of the secondary asteroid body shall be achieved with a resolution of at least 20 m during the course of the AIM asteroid observation phases. As a requirement for navigation and collision risk mitigation purposes, during the observation phases the TIRI should, simultaneously to the VIS, keep the primary and secondary components within its FoV.

No requirements have been assumed at this stage in relation to the dust or ejecta generated by DART´s impact.

These objectives translate into the following design requirements for the TIRI instrument:

- Pixel IFoV: 1 mrad / 10 m pixel size
- Spectral band: 7-14 µm
- Number of channels: 1 (radiometer)/9 or more (spectrometer)
- Passive cooling.

4.1.3 High Frequency Radar (HFR)

The main objective of the High Frequency Radar shall be to characterise the asteroid surface layer from which ejecta will be generated by the DART impact. The study of the propagation of the electromagnetic signal will provide data on the range to the observed object, and dielectric properties and aggregation state of the material constituting the objects down to the signal penetration depth. In particular the objectives are to:

- Provide data on the geophysical properties and structure of the secondary asteroid subsurface (several locations or globally, few cm to few m)

- Discriminate between different levels of porosity (<20%, 20-50%, 50%-80%, >80%),
- Contribute to the characterisation of the secondary object surface topography and the crater generated by DART´s impact.

Since the impact crater depth of DART is assumed to be 10-20 m, the HFR will have to penetrate at least this deep into the asteroid surface layer. A vertical resolution of 1 m is expected after post-processing. Although the target for HFR measurements will be mostly the secondary asteroid component, the primary component shall be investigated also as a goal.

4.1.4 Low Frequency Radar (LFR)

In order to characterise the asteroid deep interior in terms of its structure (homogeneity, discontinuities, size of “building blocks”) a radar system is required operating on lower frequencies than the HFR to penetrate into and propagate through the secondary asteroid. A bistatic radar concept similar to Rosetta´s COSERT is identified. The sampling rate and SNR shall be sufficient to distinguish between macroscopic changes in density within the asteroid interior.

4.1.5 OPTEL-D

AIM shall carry out the demonstration of the deep space operation of an optical link for communication. The Optel-D shall transmit at least a measurement set consisting of a data volume of 10 Gbits.

A measurement set corresponds to the operation of the instrument, nominally VIS, in a way that images are taken every 90 seconds (480 images during an entire secondary asteroid / binary orbital period i.e. 12 h).

The test shall be undertaken at Earth-spacecraft range between 75 M km and 16 M km, and as soon as 2 weeks after asteroid operations start, with a target of operating at larger ranges enabled by the interplanetary cruise phase, (not considered in detail).

4.1.6 MASCOT-2 Lander

AIM shall conduct the deployment of the MASCOT-2 lander on the required landing site on the surface of the secondary. The MASCOT-2 lander mission shall be designed to prevent the lander bouncing off after touchdown. It shall be fully operational on the secondary asteroid after deployment for at least 3 months.

In-situ measurements will be conducted on the asteroid surface. The lander shall serve as a platform for one of the bistatic LFR components and carry additional payloads.

4.1.7 CubeSat Opportunity Payloads (COPINS)

AIM provides the CubeSat Opportunity Payload Intersatellite Network-System (COPINS), an opportunity for the demonstration of CubeSat technology used in deep space missions, since the environmental requirements for such missions might often be more demanding on CubeSat hardware than for LEO missions. The application of such a standard widely used across Europe nowadays might facilitate the quick and simple integration of experiments in missions.

- The AIM S/C shall carry two 3U CubeSat dispensers including CubeSats for this kind of demonstration. The two deployment pods will operate independently and provide data and power connectivity with up to three separate CubeSats each
- The CubeSats shall be wither 1-U, 2-U or 3-U
- They shall be qualified for deep-space operation for a period of at least 1 month
- Their lifetime should be compatible with the AIM mission and expected deployment date i.e. more than 2 years.

4.2 Payload Specification

4.2.1 Visual Imaging System (VIS)

The baseline design for the AIM Visual Imaging System (VIS) includes dedicated optics, a CCD detector and electronics. VIS would be capable of providing monochromatic images with a GSD of 0.85 m from a distance of 10 km from the target. With a FoV of 10.3° both components of the asteroid binary system are observable simultaneously. The S/C would have to perform an attitude control adjustment at least every half-period (6h) to track the secondary around the primary.

The AOCS must achieve a pointing accuracy of at least ±0.5° in order to keep both asteroids within the VIS footprint at 10 km distance. This is assuming an orbital positioning knowledge of at least 50 m. The AOCS should also provide sufficient pointing stability to achieve the limiting magnitude required to detect Didymos on arrival (e.g. 1 degree over 10 minutes).

The baseline design for the AIM3P assumes an adaptation of SMART-1´s AMIE RD[6] with modified optics and a 2048 x 2048 pixel CCD sensor. A sampling depth of 10 bit is assumed. The VIS will include a data compression unit which would employ a standard lossless compression algorithm (compression factor of 2) Further reduction in image data volume (up to 89%) would be achieved by a truncation algorithm to selecting data of interest (asteroid images vs. empty space). Taking these considerations into account the VIS will produce approximately 1.26 Gbit of data while imaging one entire rotation of the secondary asteroid in 12 h. These data rates would be very manageable by the optical link and have been assumed in the CDF phase as minimum required to achieve mission goals but might need to be revisited in case of more demanding science requirements.

Sufficient clearance in line with the VIS camera FoV for unobstructed viewing of the observed scene is required. Key specifications for the VIS are given below.

Dimensions [mm]	200 x 150 x 50 (cuboid)
Mass [kg] (including 10% margin)	2.4
Peak Power consumption [W]	9
Max. data rate generated [kbit/s]	251.64
Data volume over lifetime [Gbit]	7.55 (minimum to meet goal)
Required pointing accuracy [°]	+- 0.53
Required positioning knowledge of S/C [m]	+- 50
Max. / Min. operational temperature [C°]	40 / - 30
Max. / Min. non-operational temperature [C°]	60 / - 40

Table 4-1: VIS specifications

4.2.2 Thermal Imager

The baseline design assumes an evolution of the MERTIS instrument on the BepiColombo ESA mission RD[4]. Such design for the AIM TIRI features a radiometer and, as an option, a spectrometer based on a 160x120 pixel uncooled bolometer array. Both sensors would be sampled with a bit depth of 8 bit and will share one optical path. The instrument would be operated by the push-broom principle.

The dimensions of the AIM3P TIRI are those of MERTIS (RD[4]). It consists of the electronics box housing the imaging system, a long cylindrical baffle and a smaller nozzle-like baffle. The longer main baffle is mounted on the optical port used for the main scientific measurements, while the short baffle is mounted to the deep space sensor calibration port. Both optical ports must remain unobstructed by any S/C structural elements. MERTIS is depicted in Figure 4-1.

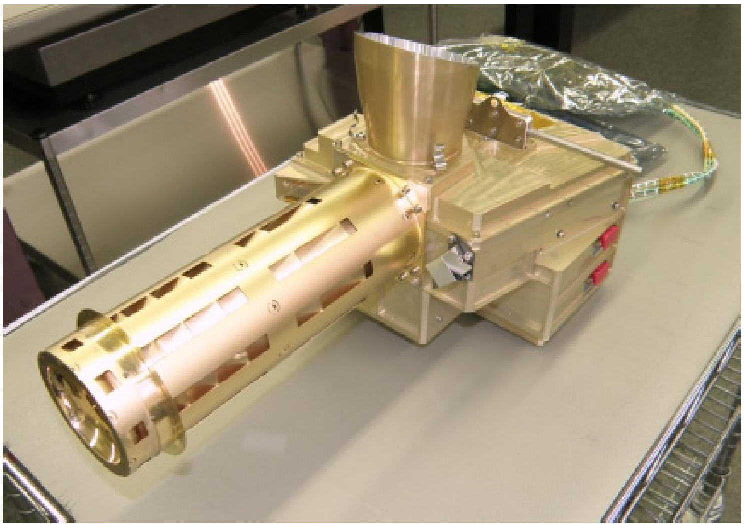


Figure 4-1: MERTIS structure model RD[4]

Key specifications for the TIRI are given in Table 4-2.

A lossless compression algorithm would be employed with a compression factor of 2. Measurements can be conducted from a fixed observer position relative to the asteroid system with each measurement lasting 12 h in order to capture one entire rotation of the secondary. The GSD from a distance of 10 km is 14.2 m. Data rate and volume figures are obtained by assuming a frame rate of one image every 1183 s.

Dimensions [mm] RD[1]	Box: 180 x 180 x 130 Baffles: 200 x Ø75 +90 x Ø75
Mass [kg] (including 10% margin)	3.6
Peak Power consumption [W]	14
Max. data rate generated [kbit/s]	0.06
Data volume over lifetime [Gbit]	0.017 (6 measurement sets)
Required pointing accuracy [°]	+- 0.53
Required positioning knowledge of S/C [m]	+- 50
Max. / Min. operational temperature [C°]	30 / - 30
Max. / Min. non-operational temperature [C°]	45 / - 30

Table 4-2: TIRI specifications

4.2.3 High Frequency Radar

The AIM high-frequency radar (HFR) is based on the FANTINA-B design RD[5] proposed for the Marco Polo mission which in turn is based on the WISDOM instrument design developed for the ExoMars mission. The AIM HFR is a monostatic Step Frequency (SF) radar operating from 300 MHz to 3 GHz. It would consist of two identical Vivaldi antennas fixed next to each other on a single cuboid antenna structure and an electronic unit which is to be accommodated inside the S/C. In order to meet the penetration depth and resolution requirements an entire rotation of the target asteroid must be mapped from at least three viewing angles to ensure global coverage.

During measurements, the boresight of the HFR antennas must be continuously oriented towards the target with a pointing accuracy of ± 5°. Key specifications for the HFR specifications are given in Table 4-3. Dimensions, mass and power are taken from the FANTINA-M RD[5] preliminary design. The data volume over lifetime is given for the entire measurement scenario to map the three full rotations of the secondary asteroid from different observation positions as previously described.

Dimensions [mm] RD[2]	EU: 145 x 155 x 50 Antennas: 260 x 620 x 240
Mass [kg] (including 10% margin)	1.7
Peak Power consumption (full power mode) [W]	16
Max. data rate generated [kbit/s]	80
Data volume over lifetime [Gbit]	10
Required pointing accuracy [°]	+- 5
Required positioning knowledge of S/C [m]	+- 100
Max. / Min. operational temperature [C°]	50 / - 40
Max. / Min. non-operational temperature [C°]	50 / - 50

Table 4-3: HFR specifications

4.2.4 Low Frequency Radar

The baseline design heritage stems from Rosetta´s CONSERT transmission radar experiment, which is similar to the FANTINA-B RD[5] instrument proposed during the Marco Polo-R mission assessment phase. Since the radar operates in a bistatic operational mode it consists of two subsystems, one on the main S/C and one on the lander. The radar signals will be transmitted by the lander and received by the AIM S/C through the secondary´s interior. The lander component of the LFR is included on the MASCOT-2 lander. The LFR system operates within a 20 MHz bandwidth at a center frequency of 60 MHz.

A priori, the LFR operation preparation requires a shape, motion and orbitography model. The accuracy of this model should be 10 m for the shape and 100 m for the orbit of the secondary asteroid. The lander and S/C LFR component are both each composed of an electronics unit and an antenna structure. The baseline orbiter antenna design consists of four monopole antenna elements mounted at the corners of the spacecraft in plane as depicted in Figure 4-2. The antennas will be deployed by memory alloy springs fixed at the ends of the rod antennas.

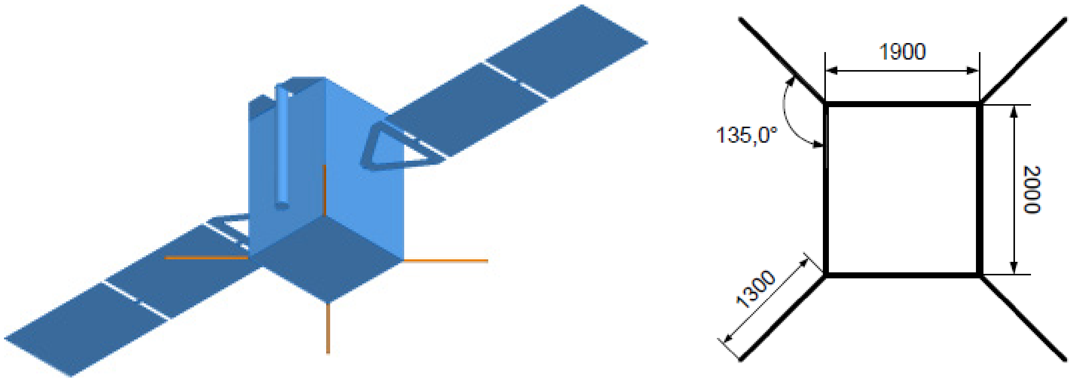


Figure 4-2: LFR antenna configuration on AIM S/C RD[5]

During measurements the antenna structure must be oriented towards the target asteroid with the precision given in Table 4-4. Key specifications for the LFR are given below.

Dimensions of Electronic Unit (EU) [mm]	220 x 220 x 60
Mass of S/C components [kg] (including 10% margin)	1.2
Mass of lander components [kg] (including 10% margin)	2.0
Power consumption on S/C [W]	10
Power consumption on lander [W]	4
Max. data rate generated on S/C [kbit/s]	20
Data volume over lifetime (accumulated on S/C) [Gbit]	0.64
Required pointing accuracy (APE) from S/C [°]	+- 5
Required positioning knowledge of S/C [m]	+- 10
Max. / Min. operational temperature [C°]	50 / - 40
Max. / Min. non-operational temperature [C°]	50 / - 45

Table 4-4: LFR specifications

4.2.5 OPTEL-D

The OPTEL-D RD[3] is a modular optical downlink system that is an evolution of the RUAG Optel-μ equipment RD[4] designed for Earth Orbit applications. The downlink system shall be capable of transmitting data to a ground stations using a modulated (16-

PPM) laser beam (1550 nm, TBC). The footprint of the laser beam is kept on the GS during communication by a fine pointing mechanism (FPM) and a coarse pointing assembly (CPA) included in the OPTEL-D system. A robust pointing of the laser beam during transmission is achieved by detecting a beacon signal, emitted by the ground station, and using this as an optical reference for the space terminal steering. The OPTEL-D system includes active closed loop correction of S/C jitter.

The OPTEL-D terminal provides the functionality to buffer on-board data of the AIM S/C. Furthermore, the following assumptions are made for the OPTEL-D operations:

- The ground segment is based on terminals with a 1 m aperture (ESA´s OGS)
- Data will be transmitted by the OPTEL-D for a maximum of 1 hour (TBC).

OPTEL-D system is composed of the Electronic Unit (EU), the Optical Head Unit (OH), the Laser Unit (LU) and its harness. The function of the OH is to provide the beam steering capabilities, while the LU generates the laser beam and the EU provides the signal modulation, reference beam detection and other functions. Figure 4-3 shows the configuration of the OPTEL-D system within the AIM S/C. Its volume is approx. 5.6 l.

Key specifications for the OPTEL-D are given in Table 4-5.

Mass [kg] (including 20% margin)	39.3
Power consumption (full operation) [W]	129
Required pointing accuracy (APE) from S/C [arcsec]	+- 70
Pointing knowledge (APK) to be provided to the payload by S/C [arcsec]	+- 10
RPE spectrum f>10 Hz [rad] (0 to peak, 2σ)	9.5E-07
RPE spectrum f>100 Hz [rad] (0 to peak, 2σ)	6.3E-07
RPE spectrum f>200 Hz [rad] (0 to peak, 2σ)	4.75E-07
RPE spectrum f>300 Hz [rad] (0 to peak, 2σ)	3.3E-07
RPE spectrum f>500 Hz [rad] (0 to peak, 2σ)	1.6E-07
Max. / Min. operational temperature [C°]	60 / - 15
Max. / Min. non-operational temperature [C°]	60 / - 40

Table 4-5: OPTEL-D specifications RD[6]

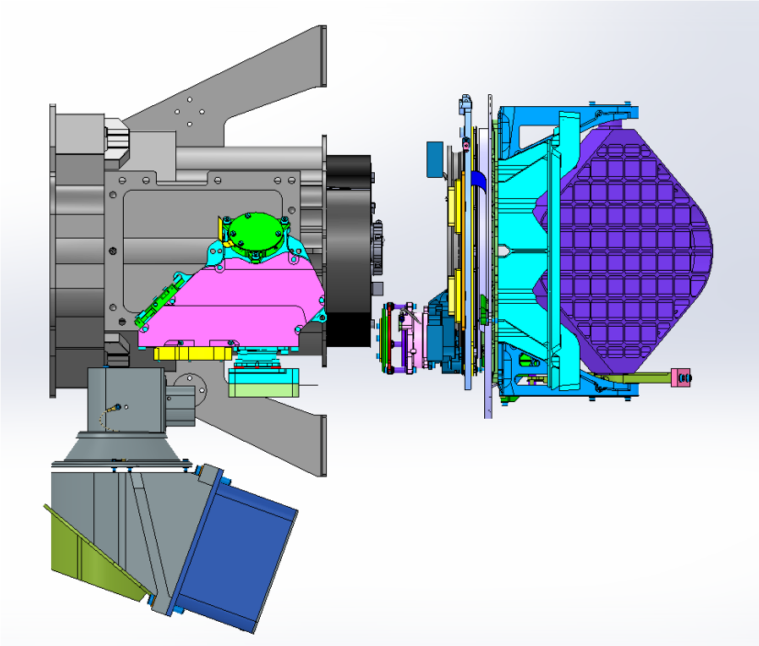


Figure 4-3: OPTEL-D Configuration on AIM S/C RD[6]

4.2.6 MASCOT-2 Asteroid Lander

The baseline for the MASCOT-2 lander is an evolution of its predecessor, MASCOT, developed by DLR for JAXA´s Hayabusa-2 mission to be launched in December 2014. MASCOT-2 design is also largely based on the Marco Polo-R FANTINA lander design RD[5]. As in the case of FANTINA, it would accommodate the second component of the bistatic LFR, a visual camera and an accelerometer. All communications between MASCOT-2 and the ground station would be relayed by the S/C using an Inter-Satellite Link that is part of the S/C telecommunication design. To prevent the depletion of the primary batteries of MASCOT-2 when it is mounted within the S/C, the carrier shall keep the battery charged. This will require a power supply of approx. 1 W, from the time of integration into the S/C until deployment.

MASCOT-2 would be deployed on the secondary component of the asteroid system, on a landing area providing safe landing and operation conditions e.g. speed well below escape velocity right after touchdown, and illumination conditions that alternate sun and shadows at regular intervals. As a baseline the landing site was assumed within a ±15° latitude band from the equator on the secondary asteroid. The following issues, concerning the MASCOT-2 design are still TBD:

- Possible mobility or “reorientation” capability of MASCOT-2 on landing site
- MASCOT-2 – S/C communication system
- MASCOT-2 landing velocity on Secondary
- S/C manoeuvre for deployment of MASCOT-2
- Surface illumination conditions (energy generation and thermal control).

The MASCOT-2 system would be deployed by the Mechanical and Electrical Support Structure (MESS). The MESS acts as housing for the Lander and is thus the main load bearing element between the lander and the AIM S/C structure. It includes a Separation & Push-off Mechanism consisting of a non-explosive actuator releasing a bolt which mounts the lander to the main spacecraft. A spring pushes the lander away upon deployment. The trigger mechanism is assumed to require 60 W for less than 1 s upon activation.

The MASCOT-2 lander in its stowed configuration housed inside the MESS is depicted in Figure 4-4.

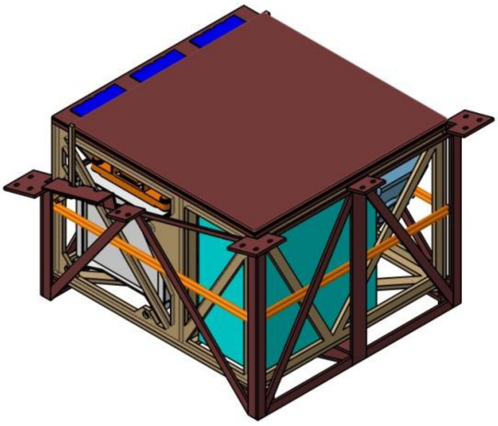


Figure 4-4: MESS including stowed MASCOT-2

Key specifications for the MASCOT-2 lander and MESS are given in Table 4-6.

Dimensions [mm] (MASCOT-2+MESS including margin)	337 x 301 x 200
Dimensions [mm] (MASCOT-2 including margin)	300 x 300 x 200
Total Mass [kg] (MASCOT-2+MESS)	13.0
Data volume over lifetime (to be relayed to GS through S/C) [Gbit]	1
Required pointing accuracy (at deployment) [°]	+- 0.25
Required positioning knowledge of main S/C (at deployment) [m]	+- 50
Max. / Min. operational temperature [C°]	30 / - 30
Max. / Min. non-operational temperature [C°]	50 / - 35
Operational lifetime on landing site [months]	3

Table 4-6: MASCOT-2/MESS specifications

4.2.7 CubeSat Opportunity Payloads (COPINS)

Two 3U CubeSat deployment systems based on an evolution of existing developments with flight heritage RD[8] will be mounted on the AIM spacecraft. They would carry the CubeSat Opportunity Payload Intersatellite Network-System (COPINS) during the AIM mission. The deployment systems would hold a TBD configuration of CubeSats with a total volume of 2 x 3U. The CubeSats will interface with the main spacecraft pre-deployment using an umbilical via the dispenser and CubeSat external port. This will be for battery charging and functional checkout / on-board software maintenance and will require 1W power. All communications between COPINS and the ground station are relayed by the S/C using an Inter-Satellite Link that is part of the S/C telecommunication design. The following issues, concerning the CubeSats are still TBD:

- CubeSat mission scenario (free-flying, flying with attitude and/or orbit control or landing on primary or secondary asteroid, type of payload, cooperative or independent missions etc)
- Configuration of CubeSats (1-U, 2-U or 3-U units).

Key specifications for the CubeSats and their 2 deployment devices (PODs) are given in Table 4-7.

Dimensions [mm]	2 * (414x130x180)
Total Mass [kg] (including 10% margin)	13.2
Power required during cruise for battery charging [W]	1
Data volume over lifetime (to be relayed to GS through AIM s/c) [Gbit]	2
Required pointing accuracy [°]	+- 0.25
Required positioning knowledge of S/C [m]	+- 50
Max. / Min. operational temperature [C°]	30 / - 30
Max. / Min. non-operational temperature [C°]	50 / - 35

Table 4-7: CubeSat + deployer specifications RD[8]

4.3 Payload Summary

4.3.1 Payload Operations

This section describes in more detail the temporal sequence that was assumed for the payload operations in the AIM3P study.

During the interplanetary cruise phase, only an early check-up and commissioning of the instruments is foreseen. The possibility of performing a long-range test with the Optel-D has been identified as realistic but not analysed in detail. The actual payload operations start two weeks prior to the arrival at the asteroid the AIM S/C would undergo a final approach phase in which the functionality of the payload will be checked again and final preparations for the measurement phases are performed. According to the mission timeline the scheduled arrival at the asteroid is between 22nd May and 1st July. After the approach the following phases would be executed:

- Early Characterisation Phase (ECP) / 2-6 weeks, depending on early or late arrival date: A *minimum* of one single VIS measurement set (1-10 Gbits) is acquired from 36 km distance from the asteroid and transmitted via the Optel-D data downlink system as a technology test. As a backup and for test validation purposes, the data is transmitted with the RF system. It is also in this period and through the VIS operation, plus possibly the operation of TIRI that the landing site selection starts.
- Detailed Characterisation Phase Period 1. (1. DCP) / 4 weeks: The S/C will move to a position 10 km from the asteroid and conduct a *minimum* of one measurement set with the VIS, one set with the TIRI and one with the HFR. The generated data is transmitted to the GS via the RF link and possibly also by the OPTEL-D. The

daily data volume downlink is maximised but always below the constraints set by the link performance and the DHS storage capacity (i.e. 16 Gbits).

- Lander phase / 2 weeks: The MASCOT-2 lander is deployed on the secondary asteroid to a point of the surface within a +/- 15 deg latitude band (TBC) that can guarantee power and thermal conditioning and safe MASCOT-2 operation for 3 months. In the same period the CubeSat Opportunity Payloads (COPINS) are also deployed and start operating for at least 1 month (this is just an initial assumption, as deployment timing and exact mission duration will depend on the exact mission that they should perform).
- Detailed Characterisation Phase Period 2. (2. DCP) / 4 weeks: A *minimum* of two measurement sets are conducted with the TIRI and VIS. The HFR and LFR will both be operated. This will also be the case for any other payloads carried by MASCOT-2 in addition to the LFR. This data is transmitted to GS during this time period. Detailed instrument operation plan is defined by the daily datalink and mass memory storage. Both the RF and Optel-D are used to maximise daily data downlink.
- Impact phase: The S/C retreats to a safe position one week prior to the DART impact (100 km from asteroid and in a plane 90 deg from the impact direction). The VIS, TIRI and possibly HFR and LFR are used to observe the ejecta resulting from the impact without putting at risk the spacecraft (the COPINS might be supporting this phase if operational).
- Detailed Characterisation Phase Period 3. (3. DCP): Following the DART impact the S/C will move to an observation point at 10 km distance from the asteroid. It will then conduct *at least* two VIS measurement sets and three TIRI measurement sets. The HFR and LFR will both be operated. The data is transmitted to the GS within the measurement phase.
- Possible extension: The mission may be extended for 3-4 weeks until 1st February.

Table 4-8 provides an illustration of the basic payload operation timeline. Temporal constraints for the operation of the LFR have not been defined within the AIM3P study. Any payload operations during DART impact have also not been specified.

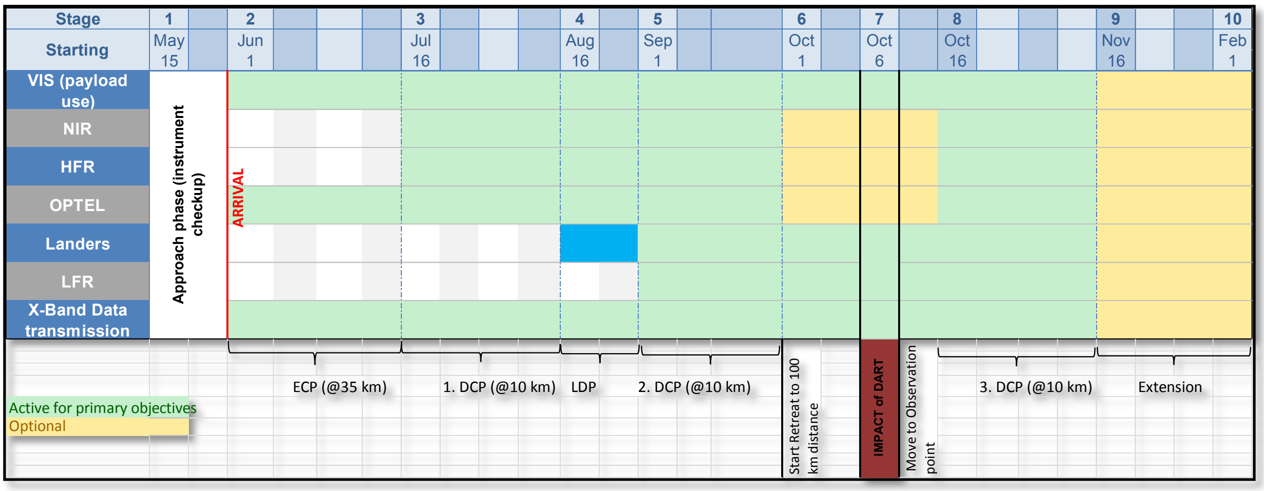


Table 4-8: AIM observation timeline

Table 4-9 presents a summary on the *minimum* data volume that would need to be produced during the mission to meet the asteroid observation goals.

Full use of downlink capacity by daily contacts compatible with the baseline ground stations i.e. ESA DSN and OGS, and the efficient use of on-board data storage, together with a suitable payload operation plan, are assumed to maximise data return.

Observation phase	Data volume [Mbit]	Measurements
Early Characterisation Phase	1258.	1 measurement set of VIS
Detailed Characterisation Phase Period 1.	11261	1 measurement sets of VIS + 1 measurement sets from TIR + HFR
MASCOT-2 Deployment Phase	TBD	
Detailed Characterisation Phase Period 2.	2523	2 measurement sets of VIS + 2 measurement sets from TIR
Impact Phase	TBD	
Detailed Characterisation Phase Period 3.	3784	2 measurement sets of VIS + 3 measurement sets from TIR
TBD	3690	Minimum data volume (MASCOT-2 inc. LF) + COPINS
Total	21260	

Table 4-9: Generated data volume by AIM payloads

4.3.2 Payload Mass Budgets

Table 4-10 shows the mass budget for the AIM payloads.

Payload	Mass [kg]	Margin [%]	Mass [kg] incl. margin
VIS	2.2	10	2.4
TIRI	3.3	10	3.6
HFR	1.4	20	1.7
LFR	1.1	10	1.2
OPTEL	32.7	20	39.3
MASCOT-2	11.8	10	13.0
COPINS	12.0	10	13.2
Total	64.5		74.4

Table 4-10: Payload mass budget

5 MISSION ANALYSIS

The target of the AIM mission is asteroid 65803/Didymos, an Apollo-type NEO with a perihelion that is just below the aphelion radius of the Earth orbit. Didymos is a binary body; the primary has a radius of around 750 m, rotation with a period of 2.259 h while the secondary had a diameter of around 170 m and rotates round the primary at a distance of 1.18 km once every 0.4958 days RD[10].The bulk density of the primary is given in RD[10] as 1.7 ± 0.4 g/cm³. Values for all these properties published by different authors vary slightly due to the scarcity and limitations of existing observations.

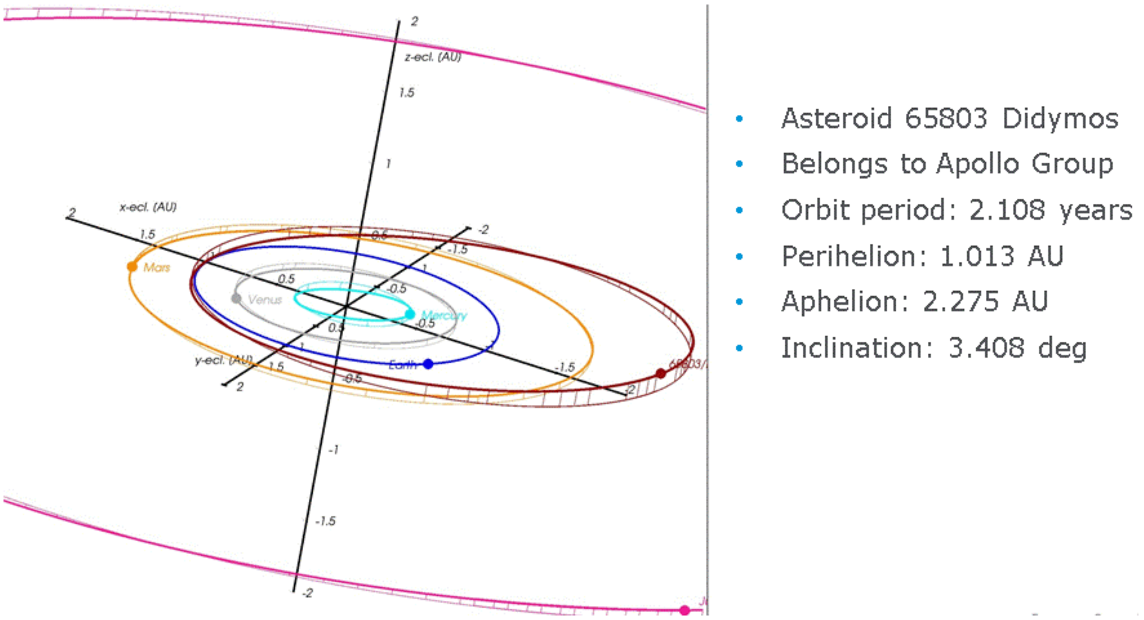


Figure 5-1: Orbit of Asteroid 65803/Didymos

In September/October 2022 the asteroid will pass its perihelion when the Earth is at a very close distance. This is an opportunity for a low-cost fast-impact mission that can be launched into the path of the asteroid, which will be overtaking the Earth at a relative speed of 6.25 km/s. The impactor will be targeted at the secondary body with the aim of creating a measureable difference in its orbital period around the primary.

The mission analysis objective is to design a complementary spacecraft mission that can observe and characterise the Didymos system before, during and after the high velocity impact. The Mission Analysis design produces vital input data for the other modules involved in the CDF study process.

5.1 Requirements and Design Drivers

The prime requirement is related to the mission timeline, which is driven by the relative geometry of Earth and asteroid 65803/Didymos and the resulting mission design of the DART high velocity impact mission in October 2022. AIM is required to take up its first observation station at a defined location 35 km distant from the asteroid no later than 2022/5/22. Launch shall not take place earlier than late 2020 in order to leave a realistic time span for all project phases leading up to the launch.

Figure 5-2 shows the hyperbolic Earth escape velocity (above) and relative arrival velocity (below) for direct, ballistic (i.e., manoeuvre-free) transfers from the Earth to asteroid 65803/Didymos as function of departure date and transfer duration. There is an opportunity in late 2020 with a duration of around 19 months (i.e., arriving in mid-2022) that leads to a theoretical Earth departure velocity of approximately 5 km/s (colour-coded in yellow) and an arrival velocity of about 1 km/s (purple-blue). This opportunity is the only one that is consistent with the stringent timeline requirements. Compliance with this mission opportunity was the principal driver for the mission analysis design.

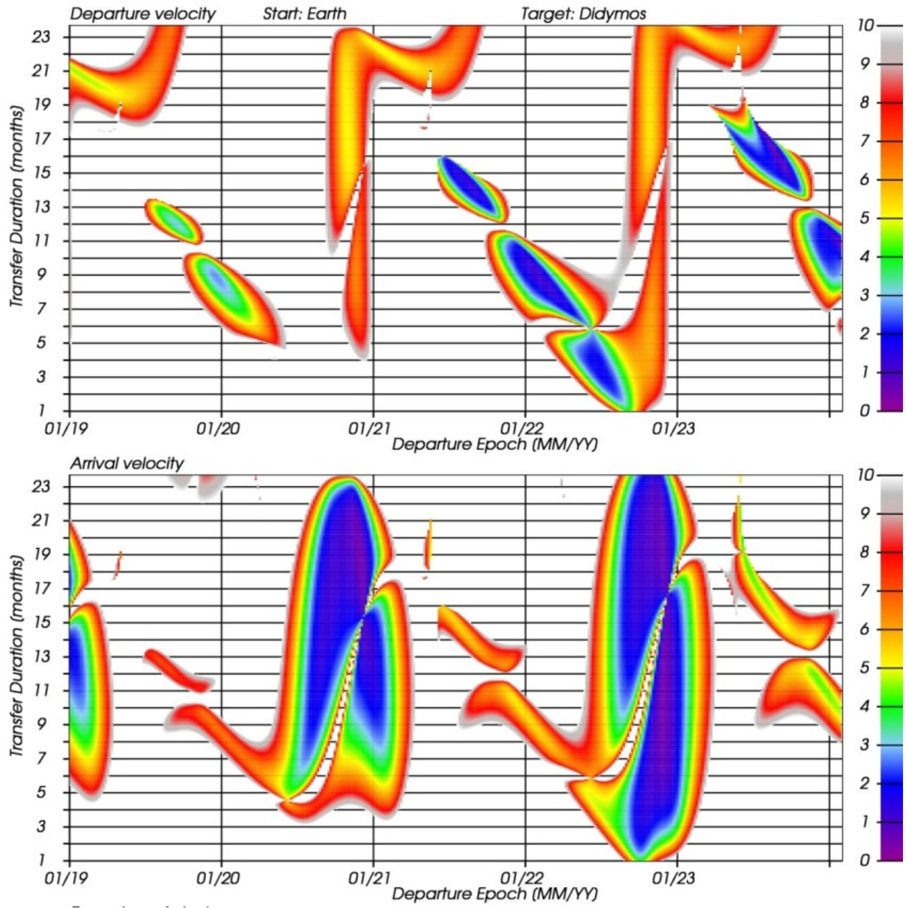


Figure 5-2: Pork-Chop Plots for Direct Earth-Didymos Transfers

5.2 Assumptions and Trade-Offs

5.2.1 Launch and Earth Escape

In the present study, only launch with a Soyuz 2.1b / Fregat MT from Kourou was considered. The launch is assumed to insert into a circular, intermediate LEO inclined at 70 deg and with an altitude of around 200 km. The Fregat stage performs a first manoeuvre to insert into the circular parking orbit. The high inclination ensures that the third stage falls into the North Atlantic far from any land mass. The node module (composite of Fregat MT upper stage and AIM spacecraft) coasts on parking orbit until the appropriate time for start of the 2nd Fregat manoeuvre. This inserts into the escape hyperbola, ensuring that the spacecraft leaves the Earth at the correct hyperbolic velocity and direction.

The ascent trajectory was computed numerically via parametric optimization using software available within the Mission Analysis section at ESOC. *The feasibility of the ascent trajectory, compliance with far range safety constraints and also the obtained performance value must be confirmed by the launcher authority*. Figure 5-3 shows the obtained ground track with the drop zones for the boosters, fairing, core stage and third stage, respectively. Using this parking orbit, for an escape C_3 of 27 km^2/s^2 (corresponding to an escape velocity of 5.2 km/s, a performance of 900 kg was obtained. Subtracting the adapter mass of 110 kg (for a 937 mm diameter) and 50 kg of launchability margin, the available wet mass at that escape velocity is obtained as 740 kg.

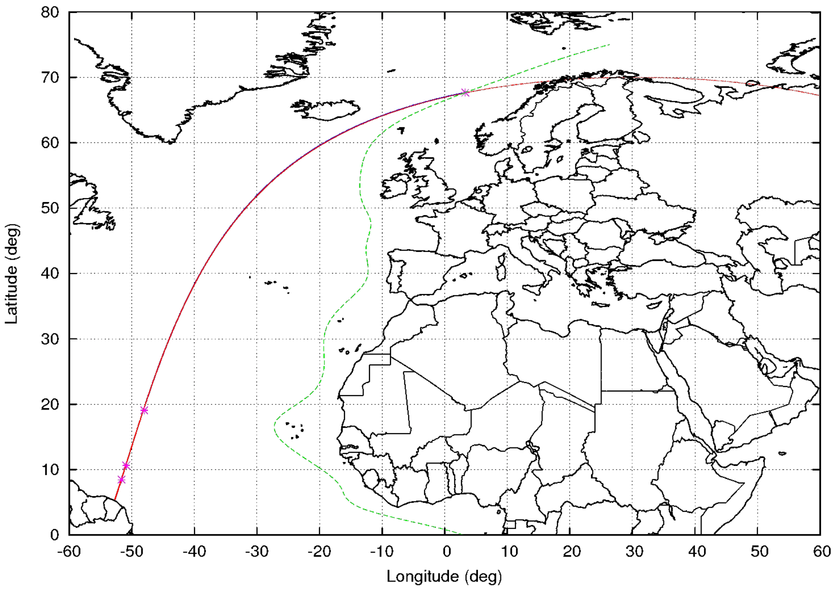


Figure 5-3: Ground Track for Soyuz Fregat Ascent Trajectory into Parking Orbit

5.2.2 Lander Targeting

For the deployment of the lander, targeting of the primary vs. the secondary body was traded on the basis of engineering considerations. Furthermore, deployment involving hovering close to the designated surface location was traded against deployment from hyperbolic flyby.

5.3 Baseline Design

5.3.1 Launch Period and Cruise

A 21 day launch period was assumed for the mission. To maximise the design efficiency, the total deterministic delta-v (i.e., not containing margins or provisions for navigation), was constrained to the same maximum value throughout the window. The maximum escape velocity encountered is 5.191 km at LPC, which becomes the driving case, though the value reached at LPO is almost equal. This is slightly less than 5.2 km/s, for which a feasible wet mass of up to 740 kg was established in Section 5.2.1.

Launch date	2020/10/17	2020/10/22	2020/10/27	2020/11/1	2020/11/6
Escape velocity [km/s]	5.191	5.042	4.994	5.126	5.154
Escape declination [deg]	22.8	24.3	25.9	26.8	28.9
DSM date	2020/12/20	2020/12/16	2020/12/20	-	2020/12/29
DSM size [m/s]	95	97	118	0	227
Asteroid arrival	2022/4/22				
Duration [d]	554	549	544	539	534
Arrival manoeuvre [m/s]	1155	1153	1132	1129	1023
Total Delta-v [m/s]	**1250**	**1250**	**1250**	**1250**	**1250**

Table 5-1: Launch Period Characteristics for Soyuz-Launched Mission

Asteroid arrival has been constrained to April 22, 2022. There is a large arrival manoeuvre of up to 1155 m/s. In this respect, LPO is the worst case in the launch period.

This manoeuvre will be split into a number of small sub-manoeuvres, spreading out over a period of weeks. This has been studied in Section 5.3.2. The breakdown into a multi-manoeuvre insertion sequence does not increase the overall cost, but it does delay the time at which the final observation station is reached. The arrival epoch assumed in the transfer design was chosen such that ample time remains available to allow reaching the initial observation station by 2022/5/22.

Figure 5-4 shows the interplanetary transfer, which will lead the spacecraft to a maximum Sun distance of up to 2.2 AU and a maximum Earth range of 3.2 AU. A superior conjunction in Oct/Nov 2021 will interrupt communications but does not interfere with critical operations.

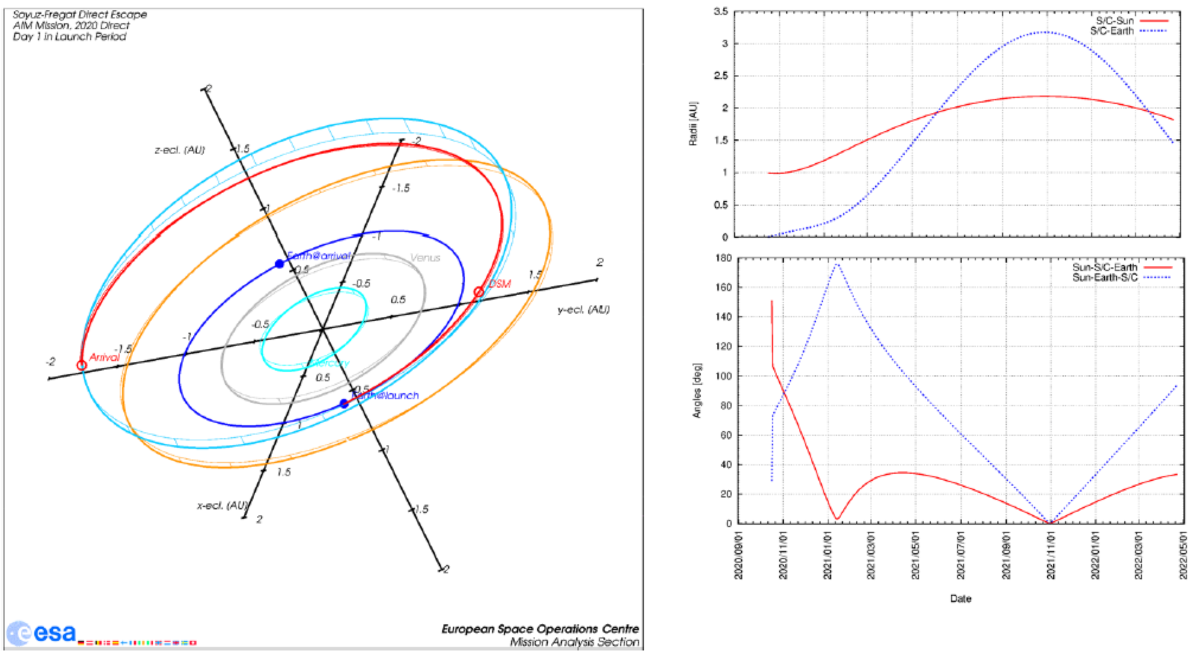


Figure 5-4: Interplanetary Transfer to the Asteroid, LPO Case

Figure 5-5 shows the coverage from the Estrack stations Cebreros, New Norcia and Malargüe during interplanetary cruise for the LPO case.

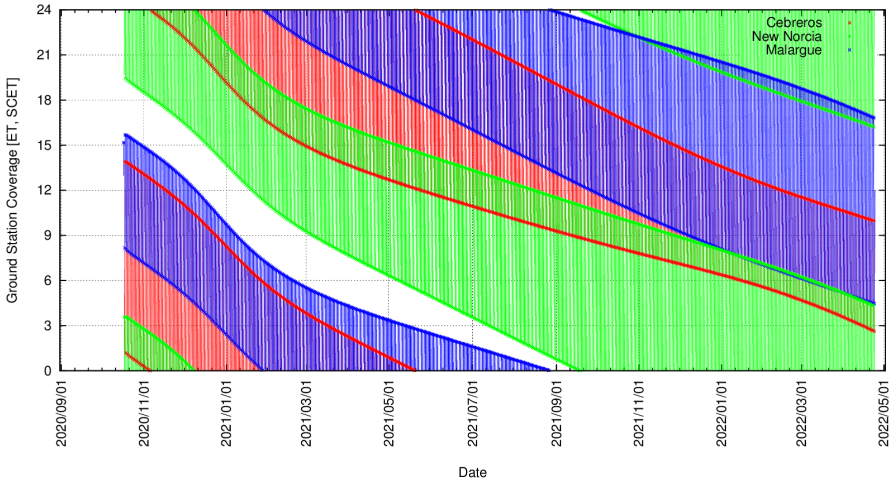


Figure 5-5: Estrack Station Coverage during Interplanetary Cruise, LPO Case

5.3.2 Asteroid Insertion Sequence

A five-manoeuvre insertion sequence was simulated for the LPO case. The sequence is designed such that the manoeuvres become consecutively smaller. The manoeuvres are spaced by seven days, which leaves ample time for orbit determination, thruster performance calibration and preparation of the next manoeuvre in the sequence. The target point of the insertion sequence is the 35 km formation flying station described in Section 5.3.4.

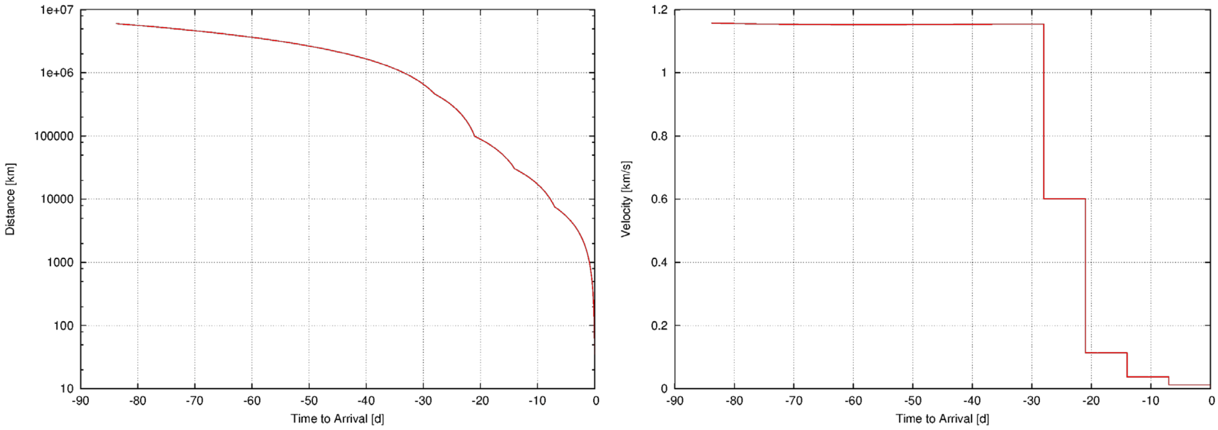


Figure 5-6: Relative Distance and Velocity wrt. Asteroid During Approach

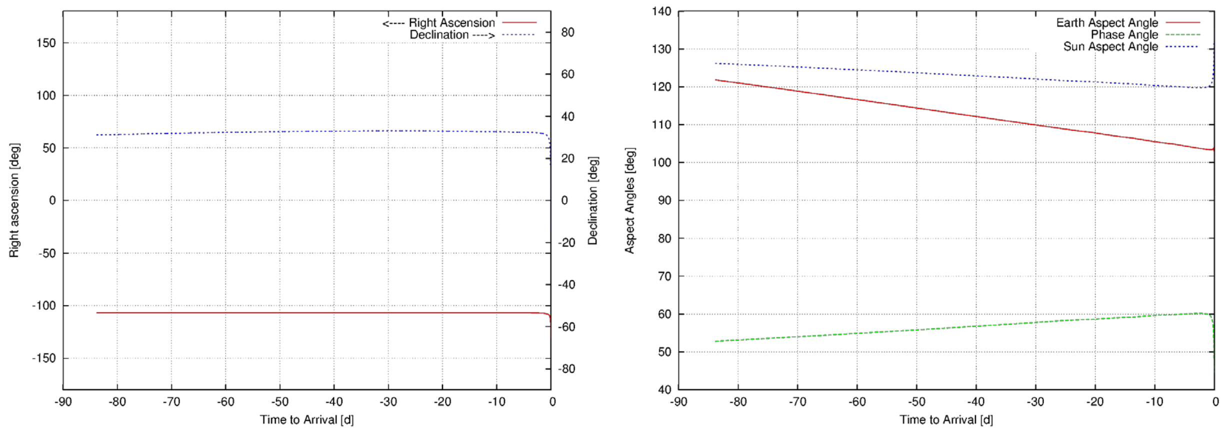


Figure 5-7: Asteroid Direction and Aspect Angles During Approach

Manoeuvre #	Date	Nominal size [m/s]	Distance [km]	Approximate ast. brightness
1	2022/4/19	554	5.3E5	+8.5mag
2	2022/4/26	488	1.2E5	+5.2mag
3	2022/5/3	75	3.4E4	+2.5mag
4	2022/5/10	25	9E3	-0.5mag
5	2022/5/17	12.5	35	

Table 5-2: LPO Manoeuvre Sequence Outline

Table 5-2 summarises the assumed five-manoeuvre insertion sequence and also gives the relative distance to the asteroid and the apparent magnitude at which it will be visible in the spacecraft camera, based on the distance from the Sun and the phase angle. There is some uncertainty in this assessment because of the unknown surface properties, so the given magnitude values should be seen as indicative.

During arrival, the spacecraft will search for the asteroid with its camera. The relative direction in which the asteroid will be visible hardly changes throughout the approach, as shown in Figure 5-7. It is likely that there will have been ground-based observation campaigns in preparation for the AIDA missions, so the asteroid orbit should be fairly well known by that time. In combination with the expected magnitudes listed above, identification of the asteroid in the images should be unproblematic. As of early May, the asteroid is likely to be the brightest object in the observed star field.

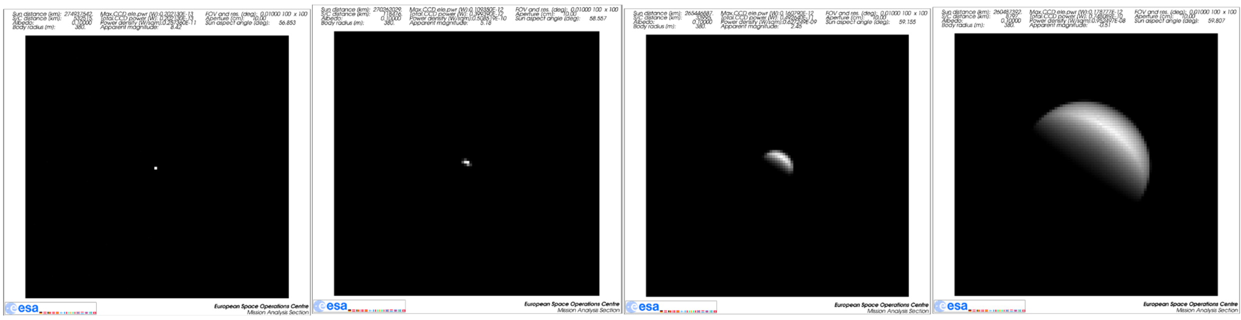


Figure 5-8: Simulated Camera View of Asteroid 65803/Didymos at Manoeuvre Times 1-4

Figure 5-8 shows simulations of what a camera with a FOV of 0.01 deg and a resolution of 100x100 pixels (or a 100x100 cut-out of the image of a narrow angle camera with a wider FOV would show, assuming a spherical, featureless primary body at the times of manoeuvres 1 through 4. The actual asteroid-Sun-Spacecraft geometry and thus also the illumination conditions have been correctly taken into account. These images were produced as part of the numerical simulation that led to the apparent magnitude assessment.

5.3.3 Asteroid-Earth-Sun-Geometry

The geometry of asteroid Didymos with respect to the Earth and Sun is of critical importance for the design of the spacecraft and near-asteroid operations. Arrival occurs at an Earth distance of around 1 AU and a Sun distance of around 1.6 AU. Both values then decrease until the time of DART impact, which occurs close to perihelion on the asteroid orbit. After that, both distances rise again.

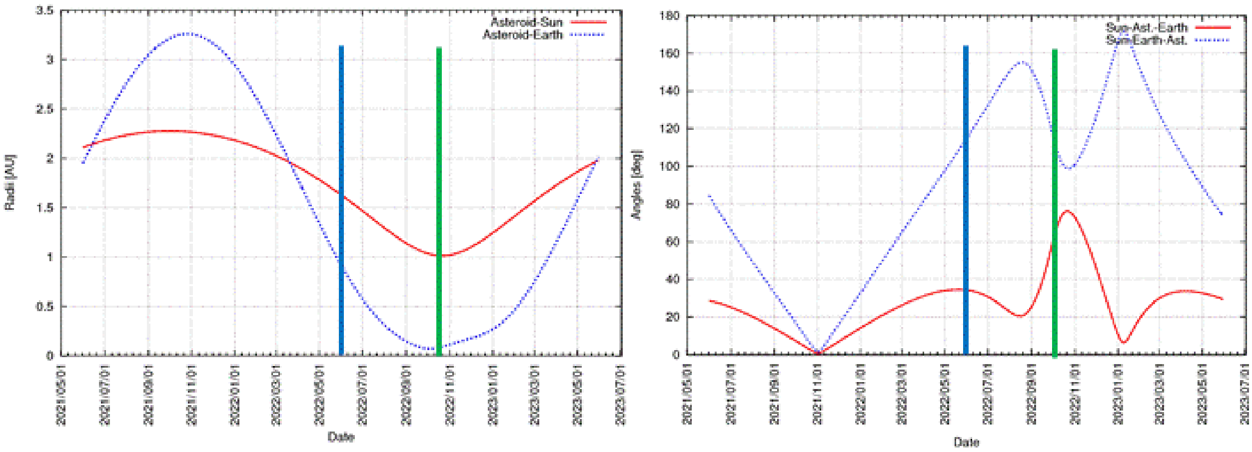


Figure 5-9: Didymos-Sun-Earth Geometry during Period of Interest

Especially in the case of a spacecraft design with fixed arrays, it is important to analyse the respective directions of Earth and Sun in order to reconcile the requirements of data relay and power supply.

Figure 5-10 shows the orbits of the Earth and asteroid Didymos from 2022/5/1 to 2023/3/1 projected into the ecliptic plane. Not only the inertial directions, but also the angular difference between both directions change significantly, the latter in the range between 0 and 90 deg. In addition, there is a small added effect due to the inclination of asteroid Didymos. This is not shown here.

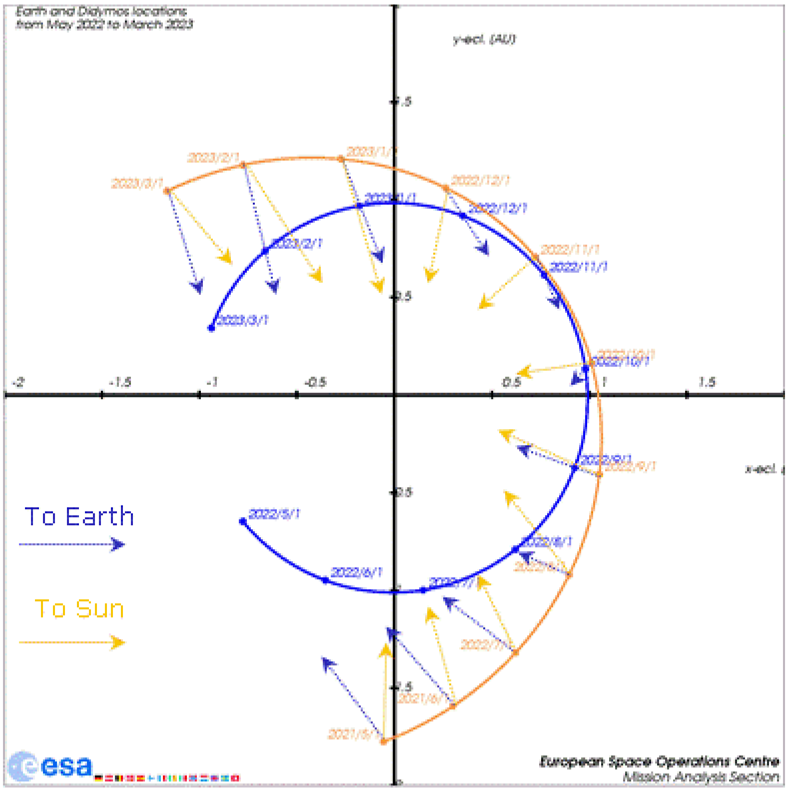


Figure 5-10: Directions from Didymos to Earth and Sun during Period of Interest

The orientation of the rotation axis of the primary and the axis of the orbit plane of the secondary is not entirely known. Two solutions were found RD[11] with significant residual uncertainties. One solution places the rotation axis near the ecliptic plane, the other near the ecliptic pole. In consequence, the spacecraft must be designed to handle both possibilities.

Solution:	1	2
N.P. Ecl. R.A. [deg]	165	290
N.P. Ecl. Decl. [deg]	20	-70

Figure 5-11: Possible Ecliptic Axis Directions for 65803 / Didymos

The illumination conditions as function of the date were studies for either of the above solutions over the time span during which the AIM spacecraft will be near asteroid Didymos. Keeping in mind that the exact orientation of the pole is unknown and further observations are necessary, it appears not unlikely that one of the poles will be permanently dark up to or beyond the DART impact, so the AIM spacecraft will not be able to perform a full characterisation of the surface by visual means.

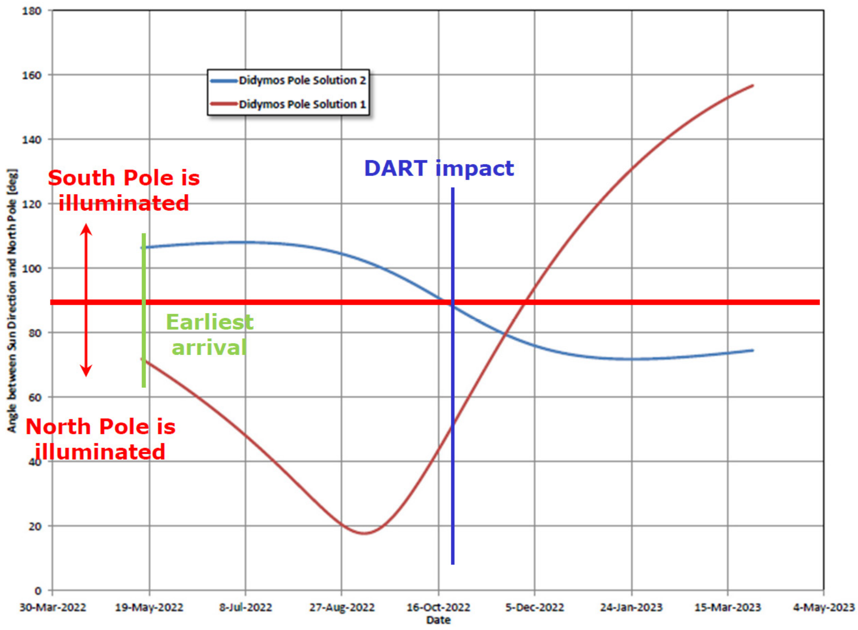


Figure 5-12: Angle between Didymos North Pole and Sun Direction for Both Solutions

5.3.4 Near-Asteroid Operations at the 35 km Station

Observation of the Didymos system is assumed to take place initially from a formation-flying orbit in a distance of around 35 km, within the plane of the asteroid around the sun but offset by around 45 deg from the Direction towards the Sun. Thus, the spacecraft would be permanently observing the asteroid from above the illuminated side, but never from above the subsolar point where the contrast of surface features would be reduced due to the absence of shadows.

The spacecraft position is assumed to be controlled within deadbands of ±1.5 km in the radial, normal and transverse directions. The orbit maintenance was numerically simulated taking into account gravitational and solar radiation pressure perturbations.

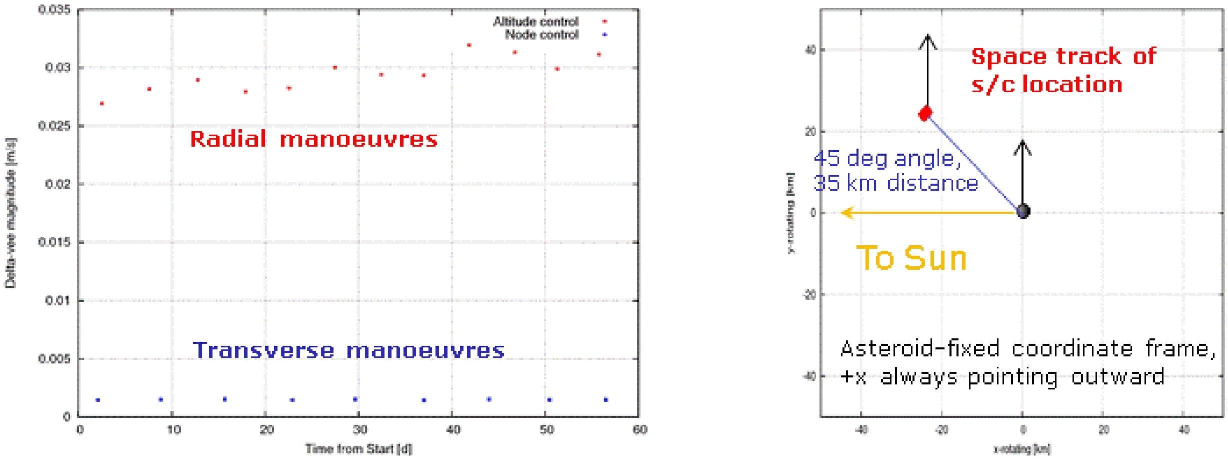


Figure 5-13: Formation Flying Manoeuvre Size and Frequency

Figure 5-13 shows the expected frequency and size of control interactions over a simulation period of 60 days. When combining transverse and radial manoeuvres, a frequency of one manoeuvre per five days appears realistic. The total manoeuvre budget for station keeping is low – here around 0.3 m/s per month were obtained. However, accurately imparting very small individual manoeuvres may constitute a challenge.

In view of the low manoeuvre frequency and the slow rate of change in location, control of the formation flying orbit using camera observations of the surface appears feasible. Use of additional instruments for navigation purposes would be a bonus but are not essential.

Temporary loss of command access is unlikely to lead to an uncontrolled surface impact or a prolonged exposure to eclipse conditions. It is unlikely that an uncontrolled trajectory will intersect the surface.

5.3.5 Near-Asteroid Operations at 10 km Stations

After the initial observation phase at the 35 km station, the spacecraft will be moved to one or several 10 km observation stations. These are also positioned at a 45 deg angle with respect to the subsolar meridian, but unlike the 35 km station, are not necessarily within the asteroid orbit plane. Three locations, positioned 45 deg above and below the orbit plane and inside the orbit plane have been studied.

It was found that the stationkeeping cost at any 10 km station does not exceed 1.1 m/s per month. The stations outside the orbit plane also require inclination control manoeuvres while the in-plane station does not. The manoeuvre frequency increases to around one manoeuvre every 2-3 days.

5.3.6 Transfers between Stations

The transfers between the observation stations are assumed to take no longer than day or less. This means that the spacecraft will be on very hyperbolic orbits during the transfers. These can in close approximation be described as straight lines with constant velocity. The transfer delta-v can thus be approximated by dividing the distance to be covered (in meters) by the transfer duration (in seconds). This gives the cost of initiating the transfer. The same delta-v must then be applied when arriving at the new target to stop the drift, so the obtained value must be multiplied by two.

The transfer from the "35 km in-plane station" to the 10 km above-plane station", with its distance of 28,800 meters will thus cost a total of 1.33 m/s if the transfer is accomplished in 12 hours. Slower transfers cost less; fast transfers cost more.

Additionally, transfers must in all cases be fail-safe. The transfer direction may never be such that it intersects either of the two asteroids in the system or their shadow cones. If there is such a possibility, a transfer must be replayed by two individual arcs that trace a detour but are safe. This adds to the cost and timeline.

5.3.7 Lander Deployment Strategy

The current baseline calls for deployment of a lander on the surface of the smaller secondary body using a hyperbolic fly-by manoeuvre (see 10.2.2.4).

An option is to deliver the lander from a hovering orbit in close proximity, having started out from a 10 km observation station. This increases the risk, the complexity and the delta-v cost for the spacecraft but minimises the landing dispersion and the risk of landers rebounding from the surface.

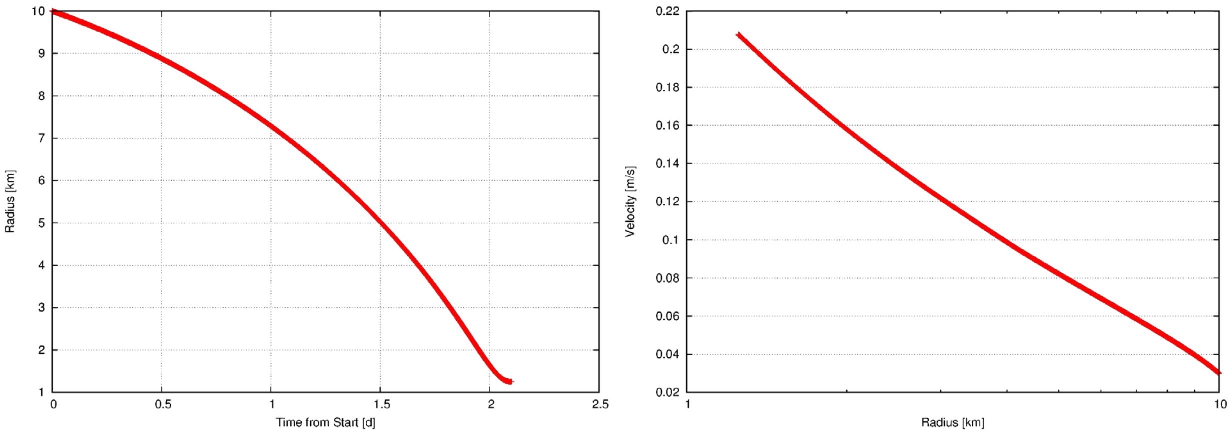


Figure 5-14: Approach from 10 km Observation Station

This approach trajectory has been simulated numerically. Figure 5-14 shows the approach characteristics *in coordinates centered in the primary body* down to a radius of 1.2 km, which corresponds to the orbital radius of the secondary. Starting out from 10 km distance with a 3 cm/s manoeuvre, the spacecraft will take 2 days to reach the secondary's orbit. The right hand diagram shows the increase of the velocity during approach (here the x-scale is logarithmic!).

The velocity with respect to the primary will then have increased to 21 cm/s. If the mass of the primary is at the high or the low end of the expected range, the velocity will also be higher or lower than shown here, respectively. Additionally, there is a slight acceleration due to the gravitational attraction of the secondary body, though this has only around 1% of the mass of the primary.

The lowest relative velocity is obtained if the approach is prograde, with the spacecraft rear-ending the secondary. The highest is obtained for a retrograde or head-on approach. The orbital velocity of the secondary is 17 cm/s. This constrains the relative velocity range to between 4 cm/s and 38 cm/s.

To slow the spacecraft down to a hover above the secondary's surface, the applicable delta-v (which must be in this range) must be applied. Further manoeuvres are required for hovering. After lander deployment, the spacecraft must insert into a safe departure orbit that leads away from both asteroids and does not intersect any shadow cones. This also requires some delta-v.

In addition to the obvious requirement of avoiding an impact with the relatively hot asteroid surface, it must also be ensured that the lander impact velocity is such that the risk of the lander modules rebounding at speeds exceeding the escape velocity is minimised.

5.4 Budgets

Salient budget-relevant values for the baseline mission have been extracted from the preceding sections and are listed below. No margins have been added.

- Spacecraft wet mass at launch: up to 740 kg
- Deterministic delta-v budget: 1250 m/s
- Navigation budget: 25 m/s for launcher insertion correction and 15 m/s for arrival navigation
- Delta-v requirement for orbit control in near-asteroid phase: low. In formation flight, up to ca. 1.1 m/s/month for station keeping
- Station keeping manoeuvre frequency: 1 per 5 days at 35 km station, 1 per 2 days at 10 km station
- Typical manoeuvre size: 3 cm/s radial, 1 mm/transverse at 35 km station; 2.5 cm/s radial, 5 mm/s transverse, 1 cm/s normal at 10 km station
- Minimum solar range during cruise: 1 AU
- Maximum solar range during cruise: 2.2 AU
- Solar range at asteroid arrival: ~1.75 AU
- Earth range at asteroid arrival: >1.1 AU
- Maximum Earth range during cruise: 3.2 AU (during superior conjunction event)
- Earth range during DSM: 0.25 AU
- Superior conjunction event: Mid-Oct – Mid-Nov. 2021
- Sun distance at DART impact: 1 AU
- Earth distance at DART impact : <0.2 AU
- Eclipses during mission: None.

5.5 Options

5.5.1 Targeting from 10 km to the Primary Body

Targeting for a landing on the primary instead of the secondary works in analogy to the strategy described in Section 5.3.7. The initial delta-v of 3 cm/s is targeted in a direction that is different by around 45 deg.

The descent takes just over 1.5 days. The arrival velocity is 40 cm/s; this is also the size of the braking manoeuvre that initiates hovering. The short transfer may be a problem, because correction manoeuvres are likely to be necessary and the time to compute and implement them is very short.

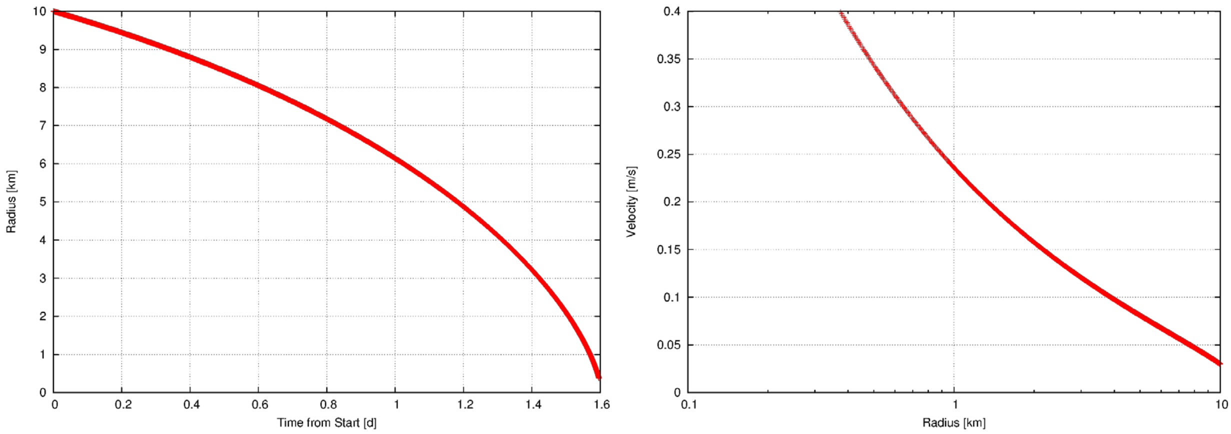


Figure 5-15: Approach to Primary from 10 km Observation Station

5.5.2 Targeting from 35 km to the Secondary Body

Targeting from the 35 km observation station requires a larger initiation manoeuvre (around 5.4 cm/s rather than 3 cm/s) and the transfer duration is longer with more than 6 days. The arrival velocity increases only by a very small amount, so from this perspective it does not matter whether one departs from the 10 km or the 35 km station. The longer transfer duration simplifies operations because there is more time for trajectory assessment and corrective action.

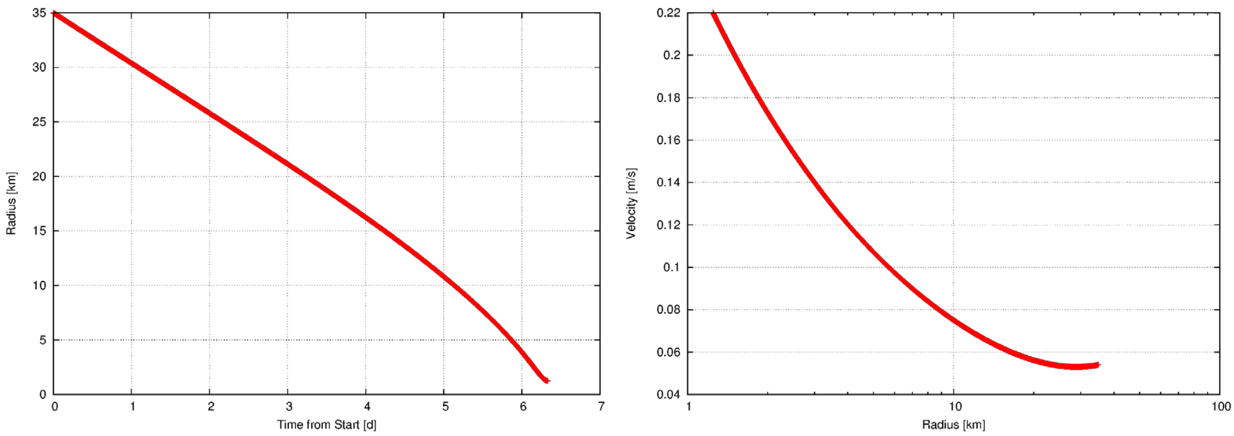


Figure 5-16: Approach to Secondary from 35 km Observation Station

5.5.3 Targeting from 35 km to the Primary Body

When targeting the primary from the 35 km station, again the main difference with respect to starting from the 10 km station is the increased transfer duration, which should be an operational advantage.

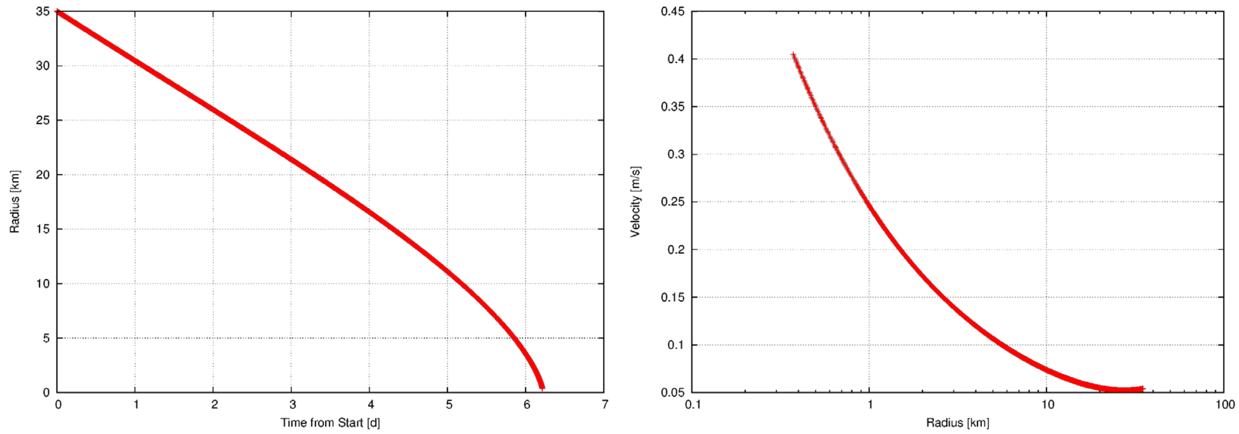


Figure 5-17: Approach to Primary from 35 km Observation Station

5.5.4 Rosetta-Like Lander Deployment

The Rosetta spacecraft deployed the Philae comet nucleus lander on an unguided trajectory from 22.5 km distance, while remaining on a safe trajectory that avoids the nucleus. If applying this strategy here, there will be no braking into the hovering position, no hovering control and no departure manoeuvre. However, the targeting accuracy would decrease an the impact velocity would increase to the levels otherwise required for the braking manoeuvre (4-38 cm/s for the secondary, 40 cm/s for the primary, +/- the uncertainties due to the unknown bulk density). This exacerbates the problem of how to reduce possible rebound velocities to below the escape velocity for the respective target body.

5.5.5 Joint Launch of AIM and DART

The mission profiles and therefore also the launch dates and interplanetary transfers of the AIM and DART sub-missions are very different. Therefore, achieving a joint launch is a non-trivial matter, especially in view of the fixed deadline for DART impact and the limited time available from now.

Launch takes place half a year earlier than the dedicated AIM launch with a Soyuz. AIM will require one Earth swingby and its delta-v budget will be reduced quite significantly. Its mission will increase by half a year.

Conversely, DART will have to perform two close and one far Earth swing bys and its mission duration will increase from 8 months to two and a half years, while undergoing large variations in the Sun (0.8-1.2 AU) and Earth (up to 0.7 AU) distance during the transfer. While the DART transfer would still be DSM-free, the longer transfer and swing-bys would raise the navigation budget.

In short, while the requirements for the AIM spacecraft would not become much more severe (except for 6 months less development time), DART would likely become a completely different mission from the one currently envisaged, and additionally it would be launched almost two years earlier.

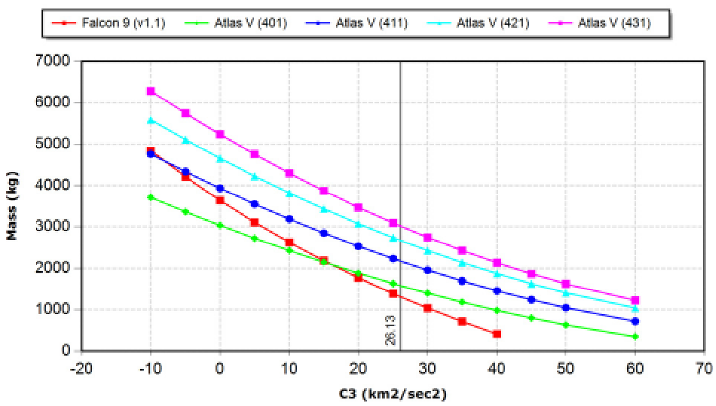
Note that this analysis looks only at one launch date and does not take into account an entire launch period, which would lead to added constraints.

	AIM	DART
Launch	2020/04/27	2020/04/27
Escape velocity [km/s]	5.11	5.11
Escape declination [deg]	-62.5	-62.5
ESB1	2020/10/29	2021/04/28
ESB1 altitude [km]	1539	300
DSM [m/s]	4	-
ESB2	-	2022/04/28
ESB2 altitude [km]	-	2447
Asteroid arrival	2022/06/11	2022/03/11
Arrival velocity [m/s]	962.3	8793
Sol. aspect ang. [deg]	66.3	100.1
Earth distance [AU]	0.796	0.116
Impact angle (sol. I) [deg]	-	40.2
Impact angle (sol. I) [deg]	-	28.4
Total delta-V [m/s]	**967**	**0**

- Alternative mission scenario added for information only
- Joint launch of AIM and DART on large launch vehicle in 2020
- Earth swingbys must be used to separate the trajectories followed by the two spacecraft
- DART transfer still remains DSM-free but navigation budget and other constraints are amplified
- AIM Delta-v budget improves
- *High declination escape is a problem for many launch vehicles!*

Table 5-3: Summary Description of Jointly Launched AIM/DART Missions

A preliminary assessment of the performance of various large US launchers with the C3 required for the joint launch comes up with the results shown in Figure 5-18. It should be pointed out that the required escape declination of -62.5 deg requires a different ascent profile from that usually taken for launches from KSC, where the achieved inclination of the intermediate LEO is normally 28.5 deg. A higher inclination need not be a show-stopper, but it would subtract some performance capability from the values listed below.



- Launch from KSC could offer the following performance figures:
- Falcon 9: 1310 kg
- Atlas V (401): 1580 kg
- Atlas V (411): 2165 kg
- Atlas V (421): 2665 kg
- Atlas V (431): 3016 kg

Figure 5-18: Assessment of Performance for US LV

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6 SYSTEMS

6.1 Requirements and Design Drivers

6.1.1 Requirements

Mission requirements	
Req. ID	**STATEMENT**
MIS-010	The mission shall rendezvous with the 65803 Didymos (1996 GT) binary asteroid system before the DART impact, no later than 22/5/2022 allowing at least 2 months of prior observations.
MIS-020	The launch date shall be no later than 17/10/2020.
MIS-030	The Mission shall be launched with an Arianespace launcher.
MIS-040	The mission cost target is 200M€
MIS-050	The mission shall be able to determine, via remote sensing, the 65803 Didymos (1996 GT) binary asteroid's rotation and orbit dynamics as well as its physical properties, in the following order of priority: • Secondary asteroid mass, size, shape and density • Dynamical state of the secondary asteroid rotation • Shallow sub-surface structure , surface roughness and topology of the secondary asteroid. • Deep subsurface, internal structure, chemical composition and gravity of the secondary asteroid.
MIS-060	The mission shall be able to deploy a microlander (MASCOT-2) on the surface of the secondary and support its operation.
MIS-070	The mission should be able to obtain imagery of the binary asteroid system with a resolution of 1 m.
MIS-08	Optical communications shall be demonstrated.

Table: 6-1: Mission requirements

Payload requirements	
Req. ID	**STATEMENT**
SYS-PL-010	The payload shall be able to obtain imagery of the binary asteroid system with a resolution better than 1 m.
SYS-PL-020	The spacecraft shall enable surface asset (asteroid microlander (MASCOT-2) landing.
SYS-PL-030	The spacecraft shall accommodate the following payload: • Visual Imaging System (VIS) • Thermal Imager (TIRI) • Monostatic High Frequency Radar (HFR)

	• Bistatic Low Frequency Radar (LFR) • MASCOT-2 • Optical Communications Terminal • CubeSat Opportunity Payloads (COPINS)
SYS-PL-040	The payload mass allocation for the asteroid research and technology payloads shall be 74.4 kg including margins.
SYS-PL-050	The VIS camera (also used for navigation) shall be used as a scientific instrument and the spacecraft shall accommodate the instrument such that observations of the secondary asteroid can be made.

Table 6-2: Payload requirements

System requirements	
Req. ID	**STATEMENT**
SYS-010	The spacecraft shall be integrated with a Soyuz 2-1b Fregat launcher to be launched from Kourou.
SYS-020	A mass of 110 kg shall be assumed for the standard Soyuz - Fregat 2-1b launch adapter.
SYS-030	The system shall be designed according to the standard CDF margin philosophy: • For equipment, the following mass margins shall be used: o 5% for off the shelf items o 10% for off the shelf items requiring minor modification o 20% for new developments or items requiring significant modification
SYS-040	A 20% maturity margin shall be added on top of the dry mass computed as mentioned above.
SYS-005	A 2% propellant margin shall be added on top of the propellant mass to account for residuals, with the nominal propellant mass being based on the spacecraft dry mass with margins.
SYS-060	The spacecraft shall be compatible with a mission lifetime of 2.5 years from launch to end of life.
SYS-070	The spacecraft should be designed using European components.
SYS-080	Only technologies at a minimum of TRL 5 by the end of phase B1 shall be used.
SYS-090	The spacecraft total launched wet mass shall not exceed 850 kg including all maturity margins
SYS-PRO-010	The spacecraft shall be capable of 100 m/s for the duration of the near-asteroid operations.
SYS-AOC-010	The spacecraft shall be 3-axis stabilised

SYS-AOC-020	The spacecraft shall provide an Absolute Pointing Knowledge (APK) of 50.00E-6 rad.
SYS-AOC-030	The spacecraft shall provide an Absolute Pointing Error (APE) of 0.34E-5 rad.
SYS-AOC-040	The spacecraft shall provide an Relative Pointing Error (RPE) of 3.00E-6 rad over 1 s.
SYS-TTC-040	The spacecraft shall receive telecommands and transmit housekeeping telemetry at all times in the mission.
SYS-TTC-050	Science data return shall support a total mission science data volume of 25 Gbits over 6 months.
SYS-TTC-060	The spacecraft shall be able to communicate simultaneously with any deployed technology packages.

Table 6-3: System requirements

Operational mission requirements	
Req. ID	**STATEMENT**
MIS-OP-010	A 35 km co-orbiting position shall be assumed with respect to the binary asteroid system for early characterisation observations.
MIS-OP-020	The spacecraft should aim to observe the DART impact from a position 90 degrees to the normal of the local horizontal of the impact site with a slant range 100 km.
MIS-OP-030	The spacecraft shall be able to transfer to different co-flying points both and maintain that position for an indefinite period within the duration of local asteroid operations.

Table 6-4: Operational mission requirements

6.1.2 Design Drivers

The following were identified as design drivers:

- Launch Date – This date must be compatible with the NASA DART mission impact which is around mid October 2022.
- Launcher & upper stage – The mission is restricted to being launched on an Arianespace launcher. The options to be considered are Vega and Soyuz.
- Spacecraft local operations – The spacecraft must be designed to characterise the binary system in preparation for the DART impact. The spacecraft design is driven by its capability to deliver a lander to the surface of the Primary.
- GNC – A robust attitude control which is also at an appropriate cost, should ensure safety. The GNC is also the prime sub-system to achieve the mission requirements: lander delivery, any station keeping and manoeuvres during the mission.

- Cost – The mission design end to end must aim to keep costs as low as possible at equipment and infrastructure level.

6.2 System Assumptions and Trade-Offs

6.2.1 Assumptions

The following was assumed at system level during the AIM-3 CDF study:

- A low cost design philosophy similar to Proba-1 , -2, -V, SAOCOM-CS, SMART-1 and Cheops (Science S-class) rather than a standard Science M- (Marco Polo-R) or L-class (Rosetta) mission
- The design assumes an asteroid rendezvous spacecraft able to carry a primary payload fulfilling the science requirements using the minimum absolutely necessary equipment/capabilities
- Integration of innovative technology/equipment for demonstration purposes should be considered during the design phases.
- The following instruments are considered at the moment of the IFP for the AIM3P CDF study:

Payload	Mass [kg]	Margin [%]	Mass [kg] incl. margin
VIS	2.2	10	2.4
TIRI	3.3	10	3.6
HFR	1.4	20	1.7
LFR (s/c)	1.1	10	1.2
OPTEL	32.7	20	39.3
MASCOT-2	11.8	20	13.0
COPINS	12.0	10	13.2
Total	64.5		74.4

Table 1-5: Payload breakdown for AIM3P CDF Study

6.2.2 System Trade – Offs

The following system level trade – offs have been conducted:

- Navigation approach
- Solar Array Drive Mechanisms (SADMs) or High Gain Antenna Pointing Mechanism.

6.2.2.1 Navigation Approach

With cost being a significant driver, options for navigation approach relevant to operations at the asteroid were investigated.

1. Delta-DOR (Delta Differential One-way Ranging): Ground based system of two stations using pulsars as references to measure the one-way range to the spacecraft from two points and triangulate the spacecraft position. Typically the spacecraft position can be estimated in a number of hours after a manoeuvre or before a manoeuvre to determine a position. The pulsar references are needed for removing systematic propagation errors (ionosphere). The assumption also here

is that there is limited autonomy on-board for medium to long distances. This option represents the most accurate non-relative navigation.

2. Ranging: This is different from Delta-DOR in that the satellite is simply ranged from a single ground station yielding range and range-rate. Plane-of-sky observations are used to track and measure the spacecraft in the other directions other than along the boresight of the ranging signal. Slightly more on-board autonomy is assumed as spacecraft updates are less frequent and of lower accuracy as compared to the Delta-DOR option.
3. Full Spacecraft Autonomy: This is an option where the spacecraft is not reliant on the ground for positional updates. In principle it is an option where some updates from ground are provided a-priori, but the main navigation solution is computed using on-board hardware. The results of the trade-off are shown in Table 6-6.

Criteria	Delta-DOR	**Ranging only**	Full S/C Autonomy
Accuracy	+ Potentially the best achievable absolute position	**0** **Latency of data worse the DDOR and less accurate**	- Dependant a-priori values, dependant on local reference points
Cost	- Expensive ground segment	**0** **Ground segment required, but maybe only one station**	0 Lower cost, but extra FDIR software and equipment needed
FDIR Complexity	+ On-board autonomy has lower requirements as ground is in the loop to the best current a-priori position accuracy	**0** **Increase in on-board model fidelity and autonomy to compensate for higher data latency**	- Very complex FDIR, hazard avoidance dependant on local points of reference.
Risk	+ Ground able to provide support for spacecraft position. Good spacecraft state knowledge.	**0** **Lesser accurate spacecraft state knowledge than DDOR, longer observable integration times required.**	- Lack of regular a-priori information
S/C Configuration	- Requires ground contact so there will be pointing req.	**-** **Requires ground contact so there will be pointing req.**	+ Pointing requirements less, and sun angle main constraint. Ground not in the loop so infrequent earth contacts needed
S/C Resources	+ Only TTC needs to be enabled with DDOR tones	**+** **No specific needs other than ranging tone**	0 Extra processing, potential additional equipment

Table 6-6: Trade-off of navigation options advantages and disadvantages

The conclusion of the trade-off is that ground intervention is normally necessary to predict the location of the spacecraft. The quicker this is done, the more accurate it will be as the spacecraft moves as the position solution is obtained. The risk with using ranging only with respect to using Delta-DOR is that the solution is obtained over a longer period than Delta-DOR. This poses a risk of an inaccurate position solution.

The criteria in this trade-off which carries the most weight is cost, a low cost mission is required by the stakeholder. On-board autonomy is needed but it shall be minimal so to reduce cost. Use of Delta-DOR potentially has a higher cost due to using two ground stations. However more time and computation is needed for the ranging only option. A balance between ranging and Delta-DOR could be reached.

The selected baseline is ranging. Delta-DOR could be used depending on available budget (<u>and is desired at system level</u>).

The spacecraft shall have autonomy during near-asteroid operations and should <u>tolerate a positional error in keeping with spacecraft position determined by ranging from the ground</u>.

6.2.2.2 Solar Array Drive Mechanisms and/or High Gain Antenna Pointing Mechanism

The following mechanism trade deals with a choice between operational flexibility and cost. Embarking a high gain antenna (HGA) pointing mechanism and/or a solar array drive mechanism (SADM) has a cost penalty, although it allows a potentially more optimised approach to power generation and telemetry.

The following show a number of boundary geometric conditions (1 to 4), which will be used to trade-off a number of mechanism complement options (A to D).

Mission geometry scenarios:

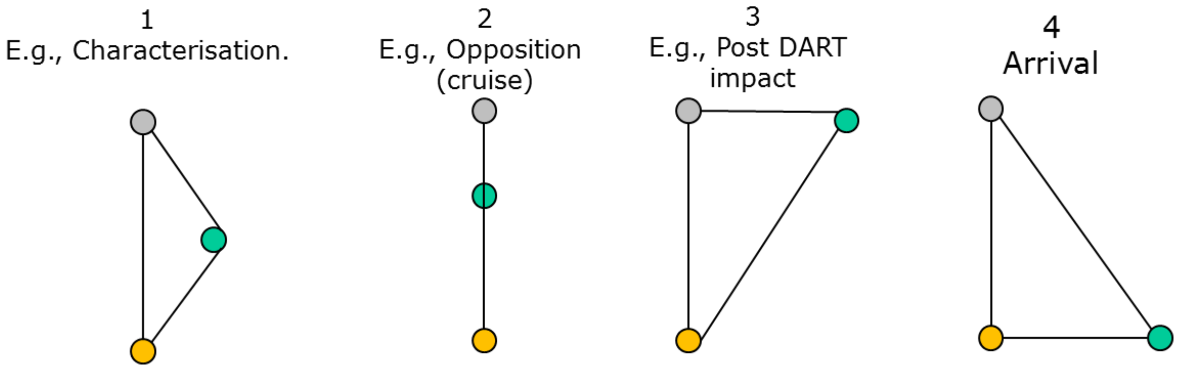
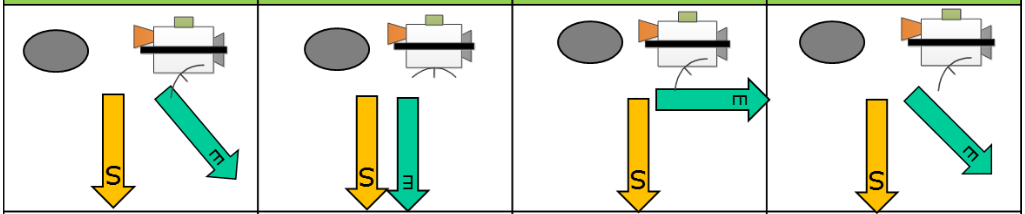


Figure 6-1: Mission geometry scenarios definition for HGA/SADM trade-off

Spacecraft mechanisms complement options:

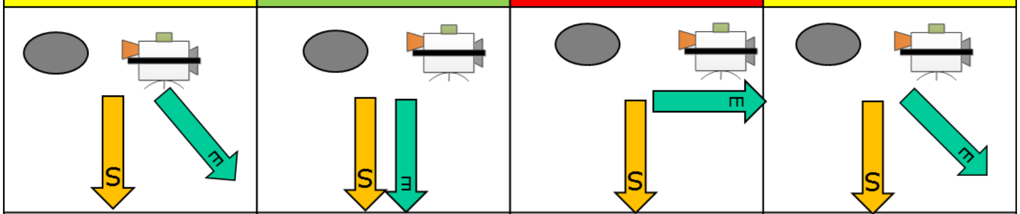
- Case A: SADM and HGA pointing
- Case B: SADM only
- Case C: HGA pointing only
- Case D: No pointing of solar array or HGA using mechanisms

Table 6-7, Table 6-8, Table 6-9 and Table 6-10 shows the impact of each scenario on the given mechanism complement case.



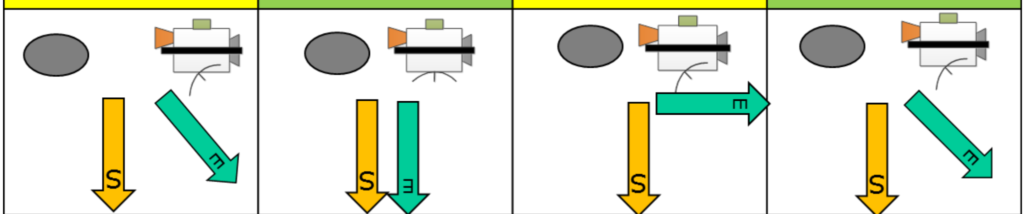
1	2	3	4
- HGA pointing possible - Power flexibility for manoeuvres	- Power flexibility for manoeuvres - Limited benefit to have extra mechanisms	- HGA pointing possible - Power flexibility for manoeuvres	- HGA pointing nearly possible - Power flexibility for manoeuvres - Likely HGA cannot turn completely

Table 6-7: <u>Case A</u> - SADM and HGA pointing against each geometry scenario



1	2	3	4
- Manoeuvre required to increase link - Power flexibility for manoeuvres	- Power and communications possible at max performance - Power flexibility for manoeuvres	- Manoeuvre required to increase link - Power flexibility for manoeuvres	- Manoeuvre required to increase link - Power flexibility for manoeuvres

Table 6-8: <u>Case B</u> - SADM only against each mission geometry scenario



1	2	3	4
- HGA pointing possible - Consequences for manoeuvres in terms of power	- Power and communications possible at max performance - Consequences for manoeuvres in terms of power	- HGA pointing possible - Consequences for manoeuvres	- HGA pointing nearly possible - Likely HGA cannot turn completely - Consequences for manoeuvres

Table 6-9: Case C - HGA pointing only against each mission geometry scenario

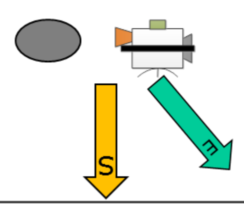
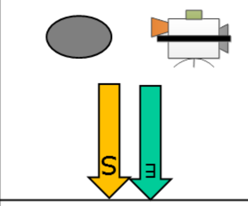
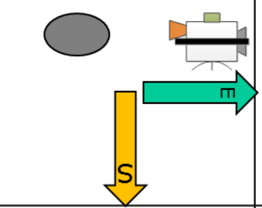
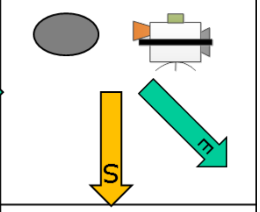
1	2	3	4
			
- Reduced SA power when communicating - Consequences for manoeuvres - Manoeuvre required to increase link	- Power and communications possible at max performance - Consequences for manoeuvres	- Reduced SA power when communicating - Consequences for manoeuvres - Manoeuvre required to increase link	- Loss of SA power when communicating - Consequences for manoeuvres

Table 6-10: Case D – no pointing mechanisms against each mission geometry scenario

There are other trade criteria to be considered in addition to relating each scenario to a given configuration. These are captured in Table 6-11 to form the final conclusion.

Criteria	A (SADM+HGA Pt)	B (SADM Only)	C (HGA Pt Only)	D (None)
Scenario 1	+	0	0	0
Scenario 2	+	+	+	+
Scenario 3	+	-	0	-
Scenario 4	+	0	+	0
Cost (weighted x2)	- -	0	-	+
Risk	0	+	0	0

Table 6-11: HGA pointing mechanism/SADM trade conclusion

As a result of the trade the following was concluded:

- For a fixed HGA and solar array the spacecraft cannot be optimised for every part of the mission. Compromises must be made, with the asteroid characterisation phase the driving case.
- Potential benefit of HGA pointing mechanism
  - Pro: Operational flexibility
  - Con: *Very* high extra cost of HGA mechanism

- At larger distances (scenario 2), the solar array requirements mainly driven by S/C power demand rather than sun aspect angle (see Power section).
- Lack of SADM could require a larger battery and/or larger solar array area (mass impact assumed low, see power section) during manoeuvres
- Addition of SADM or HGA pointing added operational flexibility and sub-system mass reduction could outweigh cost penalty for inclusion. However at a price, particularly for the HGA
- **Configuration D (no HGA pointing and fixed solar arrays) selected as baseline**
    - (Configuration B is retained as a backup option depending on cost and more detailed analysis of spacecraft options).

Therefore the only retained mechanism is the Hold Down and Release Mechanism for solar array deployment, considered part of the Solar Array.

The selection of Configuration D was assessed against the current mission constraints. The most demanding mission phases for the spacecraft are at 2.2 AU from the Sun and 3.2 AU from Earth. Specially, 2.2 AU for Solar Arrays and 3.2 AU for communications. Fixed configuration results in oversizing during local asteroid operations to accommodate the worst case at the largest distances.

The configuration design requirements of the solar arrays were then analysed with respect to a mission simulation with the following assumed design points:

- Most distant position from Sun is 2.2 AU
- Constant Earth pointing for comms
- Spacecraft mode Active Drift Transmit of 251 W average.

The result was that the maximum Solar Aspect Angle is expected to be 20 degrees with these constraints. Therefore the solar array is sized assuming a maximum Solar Aspect Angle of 20 degrees for a assumed a 25 deg HGA off-set of boresight.

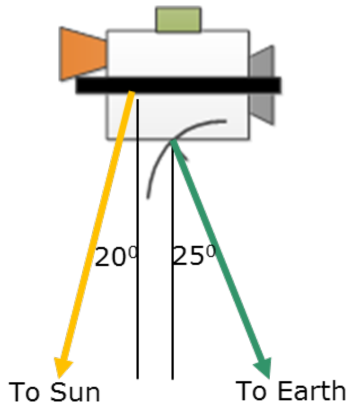


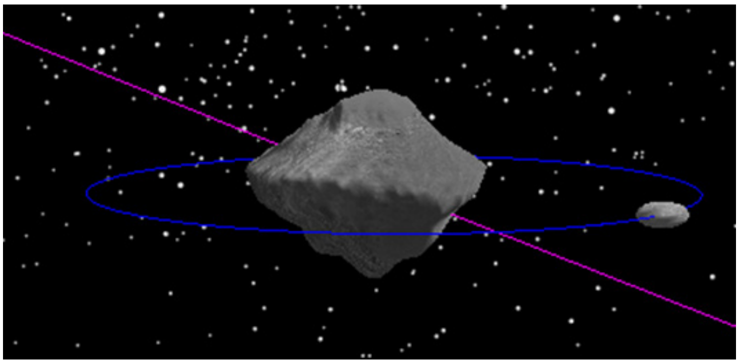
Figure 6-2: Assumed S/C Power / Communications configuration

6.3 Mission Architecture

6.3.1 Target Asteroid System - Didymos

The characteristics of the asteroid system are shown in Figure 6-3.

The system is binary with the secondary foreseen for the DART impact. The asteroid system must be mapped and characterised in preparation for DARTs arrival.



Didymos	Spectral type Xk
Primary rotation	2.26 hr
Binary orbit period	11.91 hr
Binary orbit semi-major axis	1.05 km
Primary-Secondary Separation	1.10 km
Primary diameter	800 m
Secondary diameter	150 m
Magnitude H	18

Heliocentric Orbit	
Eccentricity	0.384
Inclination	3.41°
Semi-major axis	1.645 AU
Heliocentric period	2.11 yr
Aphelion	2.28 AU
Perihelion	1.01 AU
Orbit Period	770 days

Figure 6-3: The assumed characteristics and ephemeris of the Didymos Binary Asteroid System

6.3.2 Mission Constraints

With reference to the mission requirements and Figure 6-4 the AIM mission will have the following geometrical constraints when operating in the Didymos system:

1. Arrival (MIS-010) - Two months before impact is the required arrival of the AIM spacecraft. The Sun-Spacecraft-Earth angle will be a maximum 45º. Therefore some compromise between communication and solar generation is needed. However the spacecraft configuration should not be sized for this case, but rather the asteroid characterisation case.
2. Asteroid characterisation (SYS-010, MIS-OP-010) – During science observation the Sun-Spacecraft-Earth angle will begin <45º and move towards 0º. This means there is a significant dynamic between the sun position and the earth position, while also maintaining an observation

position for the payload. The spacecraft configuration must be able handle the large delta in angles, and in doing so, will also allow the case of arrival to be handled. This part of the mission is considered the sizing case for the asteroid operations.

3. DART Impact (MIS-OP-020) – The mission geometry at this point is likely to be < 45° so will be handled by the design case (asteroid characterisation). The main constraint could be the final position of the spacecraft to observe the DART impact with respect to the asteroid, and could be incompatible with design case configuration.
4. Mission post-DART observations (SYS-010, MIS-OP-010) – This is investigations of the consequence of the DART impact. The spacecraft has requirement to operate at this time that are the same as for the pre-impact phase.
5. Extended mission post-DART observations (requirements N/A) – This is an option for an extended investigations of the consequence of the DART impact. The spacecraft has no requirement to operate at this time, and as such this should not influence the spacecraft design.

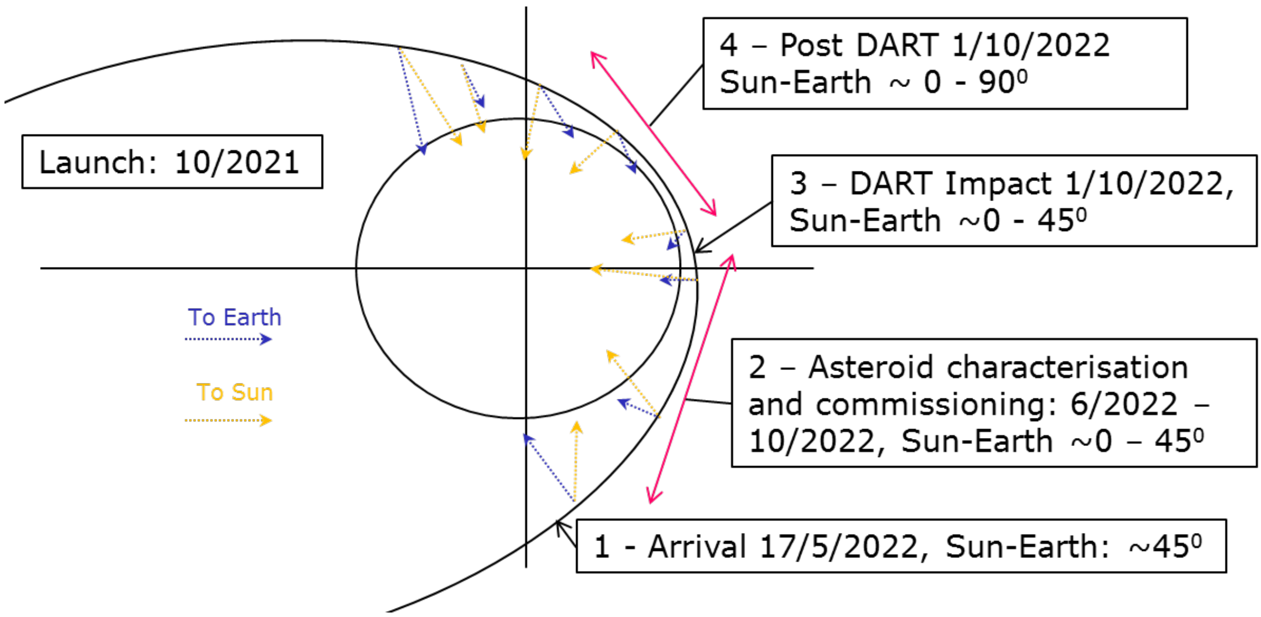


Figure 6-4: AIM mission geometrical constraints for each mission phase in proximity to the asteroid

In summary the spacecraft is required to be designed for the phase at which the mission requirements will need to be met. This is the asteroid characterisation phase. It also represents a challenging phase over a number of months with varying Sun and Earth vectors. More detail and specific angles can be found in the mission analysis chapter, used for sizing the communications and power subsystems.

6.3.3 Near Asteroid Operations

For operations in the Didymos system there are a number of options for how to approach this (see Figure 6-5).

A. Orbit Around the Primary – This option requires capture around the asteroid. The advantage is selecting an orbit which allows the primary Didymos asteroid to rotate underneath the sub satellite point. The disadvantage is the influence for the secondary gravitationally and its proximity spatially. Also this position is less flexible for lander deployment and manoeuvres.
B. Co-flying – This involves what is effectively a rendezvous with the Didymos system and allows the opportunity for fly around. Also, positions for observation can be selected to optimise surface shadowing. Manoeuvres for lander deployment have a fixed hold-point from which to begin.
C. Baricentric Orbit – This option is similar in constraints to option A with the added disadvantage of being out of sync with the rotation of the primary.

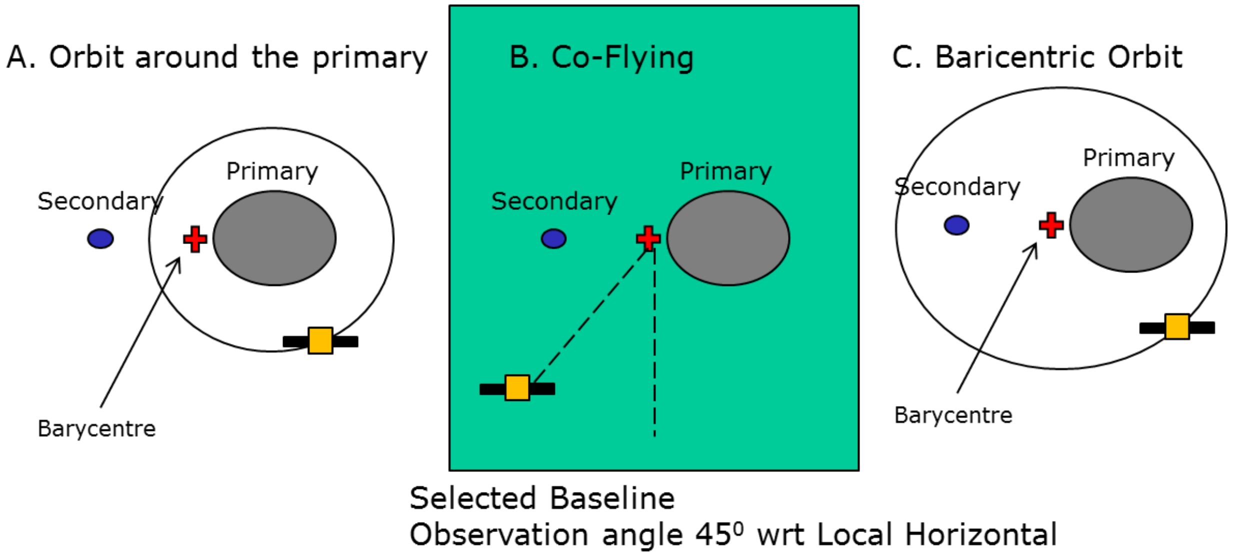


Figure 6-5: Options for local asteroid operations

Option B was selected for its flexibility for positioning the satellite as needed, since the ΔV cost of station keeping was found to be low.

A number of hold-points were also defined for the co-flying case (Figure 6-6).

- P1 – Safe position pre-defined in case of the spacecraft entering safe-mode. This a position which allows a safe sun-pointing un-controlled drift.
- P2 – Initial entry gate for the manoeuvre to deploy the lander.
- P3 – The position from which to observe the DART impact at a safe distance. The requirement is 100 km at 90º to the local horizontal normal (MIS-OP-020)
- P4 – A position defined by the requirements of viewing the whole Didymos system at a resolution of 1m (SYS-PL-030). See Section (6.5.4).

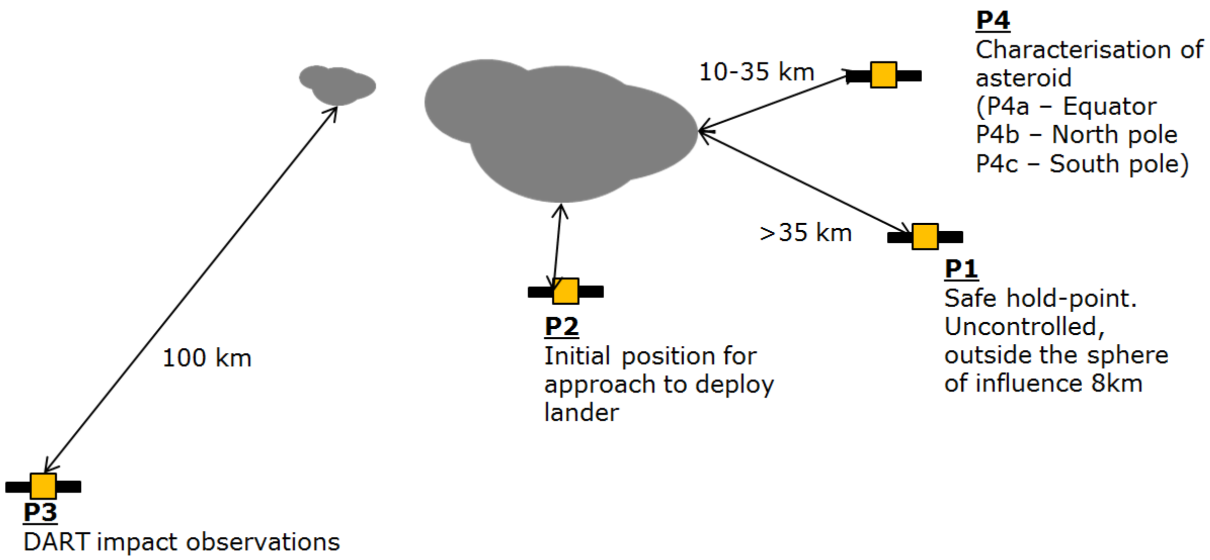


Figure 6-6: Definition of spacecraft hold-points for near asteroid operations

6.4 Potential Launchers

A number of launchers were considered both as primary and back-up. First an overview of launchers is shown, and their compatibility with the AIM mission and spacecraft is assessed briefly. Following this, in more detail, the first design iteration which considers a VEGA launch is analysed.

1.1.1 Summary of Launcher Options

The early assumptions for the AIM spacecraft, to compare to the launch vehicles in Table 6-12 are as follows:

- Spacecraft dimensions approximately 2.1 m x 2.2 m
- Spacecraft mass approximately 825 kg.
- Launch date end of 2020.

LV	AIM Escape Mass (27km2/s2)	Performance Assumptions	Programmatic Comments	Fairing (m)	Comments
Soyuz (CSG)	850 kg	• Fregat to LEO 200 km, e=0, i=70 deg. • Fregat escape	• Available • Ascent safety TBC	•Diam: 3.7 •Length: 2.8	Considered good
Vega (CSG)	< 150 kg	• STAR37FM • Spin-up with AVUM • SSO, 400 km	• Available • Ascent safety TBC	Diam: 2.4 Length: 3.1	Performance is too low
PSLV (IND)	<170 kg	• STAR37FM • SSO, 500 km	• Available	?	Performance likely to be too low
Proton (RUS)	~3300 kg	• Direct	• Likely to be underutilized	Diam: 4.3 Length: 7.2	Backup?
Rockot (RUS)	< 150 kg	• STAR37 • SSO, 400 km	• Doubts for after 2020	Diam: 2.5 Length: 3.7 m	Performance too low

LV	AIM Escape Mass (27km2/s2)	Performance Assumptions	Programmatic Comments	Fairing (m)	Comments
Ariane 5 (ESA)	<<4100 kg	• Direct • V-inf 3.5 • Dec 3.8 deg	Under used	Diam: 5.2 Length: 6.6	Too Underused
Falcon 9 (US)	< 1000 kg	• Direct	Large margin on performance. US Launcher	Diam: 2.5 Length: 1.5	Considered okay. Back-up?
Dnepr (RUS)	<230 kg	• STAR48B • Spin-up with AVUM • SSO, 400 km	Availability in question	Diam: 2.5 Length: 1.5	Fairing too small. Performance too low.

Table 6-12: Launcher options for AIM

The spacecraft seems compatible with a Soyuz, which makes the most optimum use of launch performance. Ariane 5 is likely to be under used. Vega the remaining ESA launcher is analysed in the next section, but is not considered feasible to launch AIM.

If the restriction of European launchers is lifted, Falcon-9 potentially is an option. The PSLV launcher does not have enough performance in terms of mass.

The analysis is a first iteration and further interface with the appropriate launcher authority will be needed to confirm the performances needed for AIM.

1.1.2 VEGA Design Case

The design process during the CDF study started with assuming VEGA plus an upper stage to perform the Earth escape manoeuvre. The VEGA performance is presented in Table 6-13.

PAYLOAD FAIRING	
Diameter:	2.600 m
Length:	7.880 m
Mass:	540 kg
Structure:	Two halves - Sandwich panels CFRP sheets and aluminum honeycomb core
Separation:	Vertical separations by means of leak-proof pyrotechnical expanding tubes and horizontal separation by a clamp-band

PAYLOAD ADAPTERS	
PLA 937 VG	
Height:	1461 mm
Mass:	77 kg
PLA 1194 VG	
Height:	1071.5 mm
Mass:	78 kg (TBC)

AVUM UPPER STAGE	
Size:	2.18-m diameter × 2.04-m height
Dry mass:	688 kg
Propellant:	381 kg/196 kg of NTO/UDMH
Subsystems:	
Structure:	Aluminium cylindrical case with 4 titanium propellant tanks and supporting frame
Propulsion:	MEA (evolution of RD-869) – 1 chamber
- Thrust	2.45 kN – Vacuum
- Isp	314.6 s – Vacuum
- Feed system	Regulated pressure-fed 87 l (3.9 kg) GHe tank MEOP 328 barA
- Burn time/restart	Up to 612.5 s / up to 5 controlled or depletion burns
RACS:	Six 240 N hydrazine thrusters N_2H_4 ; 39 l (38.6 kg) N_2H_4 tank MEOP 26 barA
Avionics:	Inertial 3-axis platform, on-board computer, TM & RF systems, Power
Attitude control:	
- Pitch, yaw	*Main engine ±10 deg gimbaled nozzle → boosted phases* *Six RACS thrusters → ballistic phases*
- Roll	*Roll rate and attitude controlled by four of the six RACS thrusters*

Table 6-13: VEGA Performance

Among different propulsion stage options, Lisa Pathfinder CPM, STAR 48B, STAR 37FM (STAR 31 no longer in production) were considered. Electric propulsion was not considered, mainly due to the fact that this would result in a significantly longer mission that would make it impossible for the spacecraft to reach the asteroid in time.

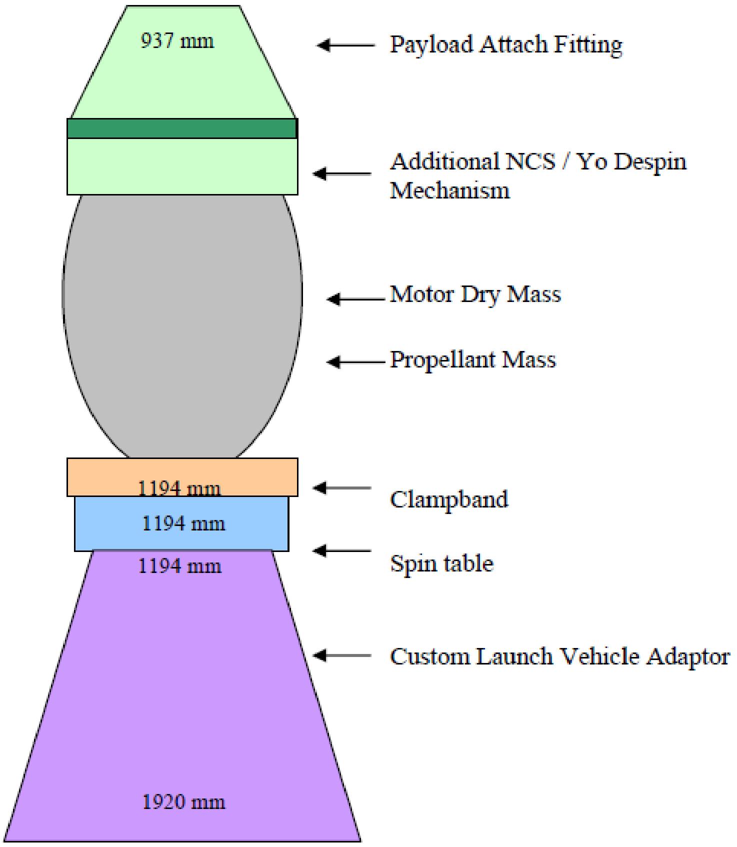


Figure 6-7: Upper – Stage Integration with VEGA

According to the latest VEGA User Manual available when the AIM-3 CDF study was conducted, the launcher can lift 1485 kg to 400 km SSO (version released in April 2014). This update leads to ruling out the use of STAR 48B motor with a minimal fuel load of 1608kg for the escape strategy. The Lisa Pathfinder Propulsion Module was also not considered a feasible solution due to added complexity for the mission and cost. From this, the option which was considered was using the STAR 37FM motor; nevertheless, the integration with VEGA of an upper stage (e.g. control, AVUM counter-spin capabilities) is not trivial.

A summary of the additional mass that has to be considered when integrating a STAR 37FM with VEGA is given in Table 6-14. To be noted that due to limited information regarding spin table, nutation control system and yo-yo despin mechanisms, STAR 48 was used as reference for these.

Propulsion module	**179.20 kg**	**0.00**	0.00
STAR 37 FM dry mass	86.20 kg		
STAR 48 spin table	36.00 kg		
Separation clampband	7.00 kg		
Nutation Control System + yo-yo despin	5.00 kg		
Payload Attach Fitting	45.00 kg		

Table 6-14: STAR 37FM Additional Integration Mass for VEGA

The mass budget for the VEGA design case is summarised in Table 6-15.

AIM3 Spacecraft				
	Without Margin	Margin		Total
Dry mass contributions		%	kg	kg
Structure	57.20 kg	0.00	0.00	57.20
Thermal Control	5.70 kg	0.00	0.00	5.70
Communications	25.70 kg	0.00	0.00	25.70
Data Handling	11.50 kg	0.00	0.00	11.50
GNC	28.60 kg	0.00	0.00	28.60
Propulsion	26.50 kg	0.00	0.00	26.50
Power	35.00 kg	0.00	0.00	35.00
Harness	10.00 kg	0.00	0.00	10.00
Instruments	15.70 kg	0.00	0.00	15.70
Total Dry(excl.adapter)	215.90			215.90
System margin (excl.adapter)		20.00	%	43.20
Total Dry with margin (excl.adapter)				259.10
Other contributions				
Propulsion module	179.20 kg	0.00	0.00	179.20
Wet mass contributions				
Propellant	70.00 kg	N.A.	N.A.	70.00
Propellant STAR 37FM	1066.00			1066.00
Adapter mass (including sep. mech.), kg	0.00 kg	0.00	0.00	0.00
Total wet mass (excl.adapter)				1574.30
Launch mass (including adapter)				1574.30

Table 6-15: Spacecraft Mass Budget for VEGA Design Case

Considering the performance of the launcher stated above (1485 kg to 400 km SSO), the mass budget overview at the time of the design did not lead to a feasible solution (the 215.9kg dry mass is around 75kg higher). The design could be reiterated, but for the purpose of the AIM – 3 CDF study and due to limited time frame, the approach was to consider Soyuz 2.1b/ Fregat MT as the option to be investigated further during the study.

6.5 System Baseline Design

6.5.1 Overview

The main characteristics of the baseline design case are summarised in Table 6-16:

System Main Characteristics	
Launcher	Soyuz 2.1b/Fregat MT
Launch date	2020
Asteroid arrival date	05/2022
Time to DART impact	5 months
Time after DART impact	2 months
Total mission duration	~1.5 years

Table 6-16: Mission Overview

6.5.2 Mission Architecture

Assuming a Soyuz 2.1b/Fregat MT launch scenario from Kourou, the following baseline mission architecture was selected:

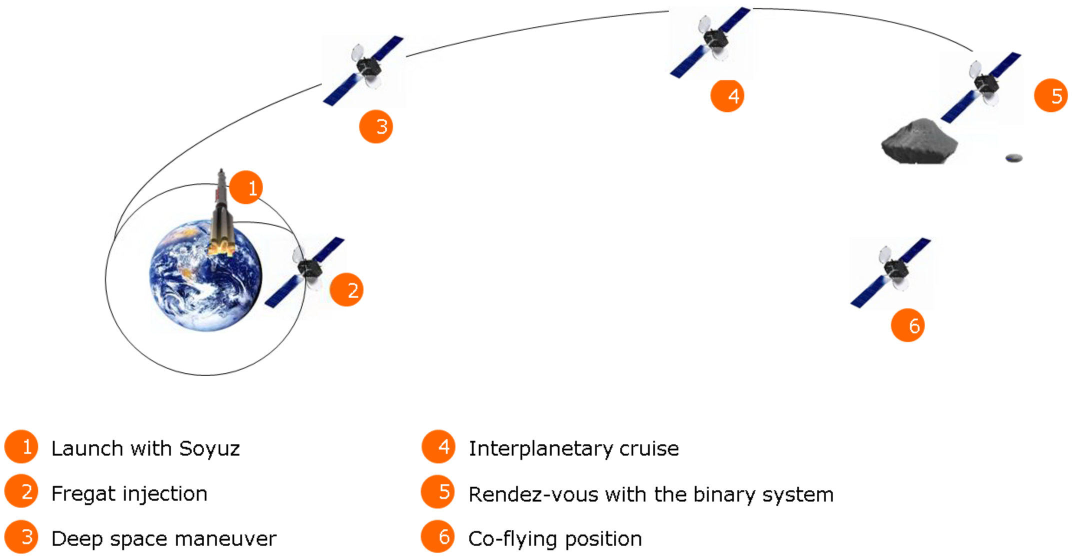


Figure 6-8: Mission architecture

6.5.3 Mission Phases and Modes

The spacecraft modes are defined as:

- Launch mode (LAU) – Lift-off to separation. All sub-systems OFF, battery is fully charged.
- Safe Mode (SAFE) – In case of a detected fault the spacecraft shall place all instruments into stand-by and acquire a sun-pointing attitude.
- Active Drift Receive (ACT-RX) – Nominal platform operations. Used during the cruise phase primarily. The payload is OFF and the RF communications transmitter is OFF.
- Active Drift Transmit (ACT-TX/RX) - Nominal platform operations. Used during the cruise phase primarily. The payload is OFF and the RF communications transmitter is ON.
- Manœuvre (MAN) – For DSM, Asteroid Rendezvous, Attitude Maintenance. Nominal platform operations, propulsion system 100% powered and the payload is OFF with the exception of the VIS imaging system
- Collision Avoidance Mode (CAM) - propulsion system 100% powered and the payload is OFF with the exception of the VIS camera. Placing the spacecraft in a safe position where an un-controlled drift does not result in a collision with the asteroids. Transition from CAM to Safe mode only.
- Measurement Mode (MEAS-N-RX) – Nominal platform operations, Payload is ON with the exception of the OPTEL experiment. RF communications transmitter is OFF.

- Measurement Mode (MEAS-N-RX/TX) – Nominal platform operations, Payload is ON with the exception of the OPTEL experiment. RF communications transmitter is ON.
- OPTEL Experiment (MEAS-O-RX/TX) – Nominal platform operations, Payload is ON and OPTEL experiment is ON. RF communications transmitter is ON.
- Lander Deployment (LAN-DEPL) - Nominal platform operations, Payload is ON and OPTEL experiment is OFF. RF communications transmitter is ON. Lander is being charged and a different GNC mode is active.

The allowed transitions of the spacecraft modes are shown in Figure 6-9.

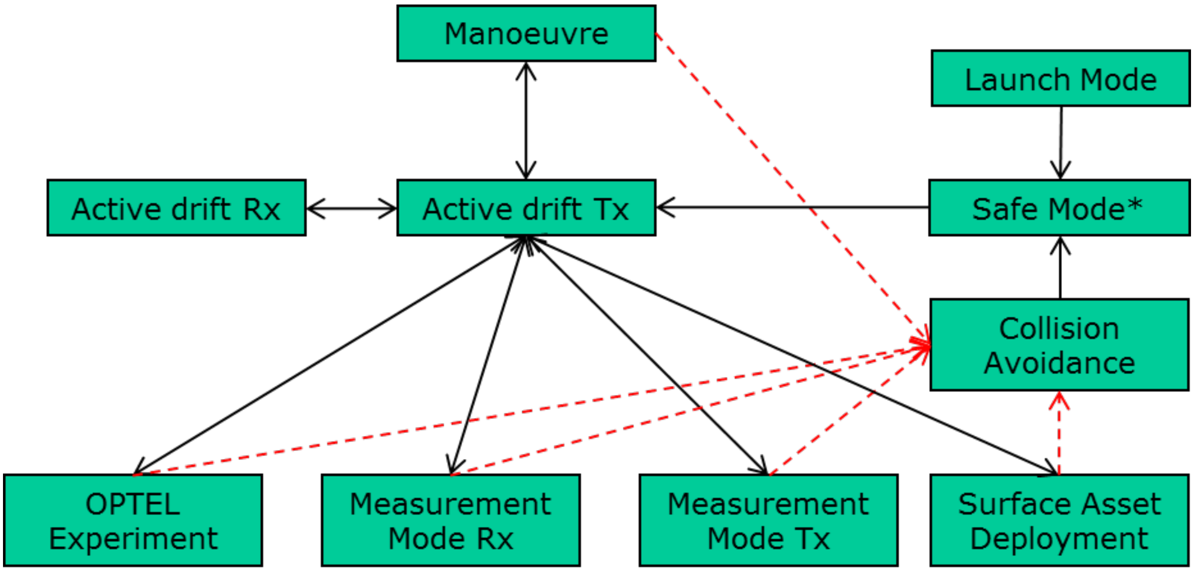


Figure 6-9: Spacecraft Mode Transitions

Note: *The SC can enter safe mode from any other mode. ** The SC can enter CAM from those modes shown, and the only transition to safe is allowed on completion of the CAM.

The mission phases and the spacecraft operative modes associated to each phase are described in Table 6-17.

Mission Phase	Day	Duration		S/C Mode
LEOP				
Launch, into intermediate circular orbit ~ 200 km, 70° incl.	1	< 10 min.		Launch
Parking orbit drift	1	< 1.5 hour		Launch
Fregat burn, escape	1	15 min.		Launch
De-tumble and slew to Earth&Sun-pointing attitude	1	TBD		De-tumble
Deployment	1	< 1 hr		Deployment
Transfer				
Drift, active 1	2	~ 2 months	~ 2 months	Transfer Active
Deep Space Manoeuvre 1		< 100 min.		Orbital Correction
Drift, active 2		< 1 week	Total since DSM 1 ~ 17 months	Transfer Active
Deep Space Manoeuvre 2 (if needed)		< 10 min.		Orbital Correction
Drift, active 3	pre-OI	~ 17 months		Transfer Active
Orbit Insertion (match asteroids orbit)		< 3h total, but broken down in multiple burns		Orbital Correction
Nominal				
Measurement Operations - Asteroids Shape Modelling (NAC) + Commissioning Instruments		1 month	5 to 5.5 months	Measurement
Approach to Station 1 (35 km from Primary Asteroid)		TBD		Asteroid Transfer (+ Orbit Corrections)
Measurement Operations from Station 1 - Asteroids Observation		1 month		Measurement
Approach to Station 2 (TBD m from Primary Asteroid)		TBD		Asteroid Transfer (+ Orbit Corrections)
Measurement Operations - Lander deployment & Proximity		1 month		Measurement
Retreat to Station 3 (100 km from Primary Asteroid)		TBD		Asteroid Transfer (+ Orbit Corrections)
Measurement Operations - DART impact & post-impact observations		2 months	2 months	Measurement
Orbit corrections		< 10 min.		Asteroid Orbital Correction
RWL offloading		< 10 min.		RWL Unloading

Table 6-17: Mission Phases and Spacecraft Operative Modes

6.5.4 Manoeuvres Overview

In summary, the main manoeuvres in the vicinity of the asteroid concern the following:

1. Transition from 35 km arrival station point to 10 km station point.
2. COPINS deployment - "flyby" to within 1 km of secondary, then return to 10 km station point (TBC).
3. MASCOT-2 deployment - hover within 1 km of secondary, then return to 10 km station point (TBC).
4. Detailed characterisation, move from equatorial position (at 10 km) to 45 degrees north.
5. Detailed characterisation, move from 45 degrees north position (at 10 km) to 45 degrees south.
6. Retreat to 100 km distance safe point.
7. Return to 10 km distance point.

The first five manoeuvres each require about 1 m/s delta-V. Assuming that manoeuvres 6 and 7 will need to be performed in about the same time as manoeuvre 1 (while the distance to cover is four times higher), each of these will require about 4 m/s. The total delta-V involved for the above list of forseen manoeuvres is thus 13 m/s.

In addition about 1 m/s per month is required for station keeping in the vicinity of the asteroid; for 9 months this adds up to 9 m/s. Reaction wheels off-loading will require around 10 m/s over the entire mission (including Earth-asteroid transfer).

In total about 32 m/s will thus be required in addition to the delta-V for the transfer manoeuvres. Typically a 100% margin is taken for this type of delta-V estimates, leading to a rounded total of about 70 m/s. From the start of the AIM-3 study a total of 100 m/s was taken into account in the Propellant consumption estimate, which is thus sufficient by far. Even though this means a total margin of over 200%, the 100 m/s requirement was maintained because:

- the additional 30 m/s may be needed for additional manoeuvres that have not yet been foreseen;
- the 30 m/s extra margin represents less than 10 kg of propellant, and removal of this will not lead to the possibility to incorporate smaller propellant tanks.

6.5.5 Budgets

Table 6-18 summarises the mass budget for all the subsystems part of the AIM-3P spacecraft design during the study:

AIM3P Spacecraft					
	Target Spacecraft Mass at Launch			850.00	kg
			Below Mass Target by:	27.96	kg
	Without Margin	Margin		Total	% of Total
Dry mass contributions		%	kg	kg	
Structure	63.27 kg	18.89	11.95	75.22	21.49
Thermal Control	7.60 kg	9.34	0.71	8.31	2.37
Communications	14.40 kg	12.08	1.74	16.14	4.61
Data Handling	17.30 kg	8.55	1.48	18.78	5.37
GNC	26.26 kg	5.16	1.36	27.62	7.89
Propulsion	65.62 kg	4.91	3.22	68.84	19.67
Power	38.60 kg	5.52	2.13	40.73	11.64
Harness	20.00 kg	0.00	0.00	20.00	5.71
Instruments	63.55 kg	17.07	10.85	74.40	21.26
Total Dry(excl.adapter)	316.60			350.03	kg
System margin (excl.adapter)		20.00	%	70.01	kg
Total Dry with margin (excl.adapter)				420.04	kg
Other contributions					
Wet mass contributions					
Propellant	292.00 kg	N.A.	N.A.	292.00	41.01
Adapter mass (including sep. mech.), kg	110.00 kg	0.00	0.00	110.00	0.13
Total wet mass (excl.adapter)				712.04	kg
Launch mass (including adapter)				822.04	kg

Table 6-18: Spacecraft Mass Budget

6.5.6 Equipment Lists

The following summarises the equipment lists defined during the CDF study:

Element 1 - AIM3P Spacecraft						
FUNCTIONAL SUBSYSTEM	nr	Mass (kg) per unit	Total Mass (kg)	Margin (%)	Margin (kg)	Mass (kg) with Margin
Structure			**63.27**	**18.89**	**11.95**	**75.22**
IF_Adapter	1	4.13	4.13	20.00	0.83	4.96
IF_Adapter_ring	1	7.00	7.00	10.00	0.70	7.70
Bottom_Panel	1	4.88	4.88	20.00	0.98	5.86
Mid-Plate	1	5.54	5.54	20.00	1.11	6.65
Top-Panel	1	5.27	5.27	20.00	1.05	6.33
PY-panel	1	4.64	4.64	20.00	0.93	5.56
PX-panel	1	4.64	4.64	20.00	0.93	5.56
MX-Panel	1	4.64	4.64	20.00	0.93	5.56
MY-Panel	1	4.64	4.64	20.00	0.93	5.56
LWR-Shear-Panel	4	0.51	2.06	20.00	0.41	2.47
Upper_Shear_Panel	1	2.64	2.64	20.00	0.53	3.17
External-Tank_Support	8	0.40	3.20	20.00	0.64	3.84
Press-tank_Support	1	3.00	3.00	20.00	0.60	3.60
Miscellaneous	1	7.00	7.00	20.00	1.40	8.40
Thermal Control			**7.60**	**9.34**	**0.71**	**8.31**
MLI	1	4.10	4.10	10.00	0.41	4.51
Radiator	1	0.40	0.40	5.00	0.02	0.42
Black paint	1	2.50	2.50	10.00	0.25	2.75
Heater lines	1	0.10	0.10	5.00	0.01	0.11
Miscellaneous	1	0.50	0.50	5.00	0.03	0.53
Communications			**14.40**	**12.08**	**1.74**	**16.14**
miniDST X-Band	2	1.50	3.00	20.00	0.60	3.60
HPA - TWTA	2	1.50	3.00	10.00	0.30	3.30
0.00	0	0.00	0.00	0.00	0.00	0.00
RFDN	1	4.00	4.00	5.00	0.20	4.20
LGA	2	0.80	1.60	5.00	0.08	1.68
HGA - metasurface	1	1.50	1.50	20.00	0.30	1.80
ISL electroc box	1	1.20	1.20	20.00	0.24	1.44
ISL antenna	2	0.05	0.10	20.00	0.02	0.12
Data Handling			**17.30**	**8.55**	**1.48**	**18.78**
CDMU	1	5.00	5.00	5.00	0.25	5.25
RTU	1	12.30	12.30	10.00	1.23	13.53
GNC			**26.26**	**5.16**	**1.36**	**27.62**
Star Tracker OH (Hydra)	2	1.25	2.50	5.00	0.13	2.63
Star Tracker EU (Hydra)	2	1.75	3.50	5.00	0.18	3.68
Sun Sensor	4	0.22	0.86	10.00	0.09	0.95
IMU	2	0.00	0.00	20.00	0.00	0.00
Reaction Wheels (Teldix 12-75/60)	4	4.85	19.40	5.00	0.97	20.37
Propulsion			**65.62**	**4.91**	**3.22**	**68.84**
Pressurant tank	1	8.10	8.10	5.00	0.41	8.51
Pressurant	1	1.50	1.50	5.00	0.08	1.58
MMH tank	2	6.40	12.80	5.00	0.64	13.44
MON tank	2	6.40	12.80	5.00	0.64	13.44
Propellant residuals	1	5.72	5.72	0.00	0.00	5.72
Thrustes (Low)	24	0.65	15.60	5.00	0.78	16.38
Gas pyro valves	6	0.15	0.90	5.00	0.05	0.95
Pressurant filter	1	0.08	0.08	5.00	0.00	0.08
Liquid pyro valves	10	0.30	2.95	5.00	0.15	3.10
LP transducer	4	0.22	0.89	5.00	0.04	0.94
HP transducer	1	0.22	0.22	5.00	0.01	0.23
Pressure regulator	1	1.20	1.20	5.00	0.06	1.26
Non return valves	4	0.08	0.32	5.00	0.02	0.34
Propellant filters	4	0.15	0.62	5.00	0.03	0.65
FDVV	7	0.05	0.32	5.00	0.02	0.34
Pipework	1	1.00	1.00	20.00	0.20	1.20
Orifices	2	0.05	0.10	5.00	0.01	0.11
Brackets / fasteners	1	0.50	0.50	20.00	0.10	0.60
Power			**38.60**	**5.52**	**2.13**	**40.73**
Solar Array wing	2	9.80	19.60	5.00	0.98	20.58
Battery	1	4.00	4.00	10.00	0.40	4.40
PCDU	1	15.00	15.00	5.00	0.75	15.75

Instruments			**63.55**	**17.07**	**10.85**	**74.40**
VIS Camera	*1*		*2.20*	*10.00*	*0.22*	*2.42*
Thermal Infrared Imager	*1*		*3.30*	*10.00*	*0.33*	*3.63*
Monostatic High Frequency Radar	*1*		*1.40*	*20.00*	*0.28*	*1.68*
Bistatic LowFrequency Radar	*1*		*1.10*	*10.00*	*0.11*	*1.21*
OPTEL (Optical terminal)	*1*		*32.75*	*20.00*	*6.55*	*39.30*
MASCOT-2	*1*		*10.80*	*20.00*	*2.16*	*12.96*
Deployable payloads (incl. deployer)	*1*		*12.00*	*10.00*	*1.20*	*13.20*
Propellant						**292.00**

Table 6-19: AIM-3 Equipment List

7 CONFIGURATION

7.1 Requirements and Design Drivers

The following list of requirements apply to the configuration discipline for the AIM-3 mission objectives.

SubSystem requirements		
Req. ID	**STATEMENT**	**Parent ID**
CFG-010	The configuration shall fit in the selected launcher fairing	MIS-020
CFG-020	The interface to the launcher shall be based on the available existing standard interfaces	MIS-020
CFG-030	The configuration shall accommodate all spacecraft equipment and instruments to comply with mission objectives, as well as power and communication requirements	SYS-020, SYS-PL-010, SYS-PL-020
CFG-040	The configuration shall provide an unobstructed field of view for the camera's, sensors and the antenna's, and unobstructed deployment for the mechanisms	SYS-020, SYS-PL-010, SYS-PL-020
CFG-050	The configuration shall provide access for the Assembly, Integration and Verification [AIV] of the spacecraft, including servicing of components during ground operations	N/A
CFG-060	The configuration shall provide an unobstructed position for the thruster sets to fulfil the mission objectives	N/A

7.2 Assumptions and Trade-Offs

In order to achieve some of the study objectives, e.g. low cost, the configuration trade-off has assessed the availability and suitability of COTS small spacecraft platforms. The COTS spacecraft were found to be incompatible with the requirements of the mission. The specific requirements for the propulsion sub-system together with the GNC and orbital requirements of the mission resulted in a dedicated design. Conceptual layout and design are common to existing spacecraft, and as such considered conventional design which do not require specific developments. Interface adapters with release devices were found available compatible with the requirements.

7.3 Baseline Design

Figure 7-1 shows the AIM-3 configuration in stowed position inside the Soyuz Launcher fairing. In addition the stowed configuration is shown attached to the Fregat Upper Stage, as well as an additional view from the back.

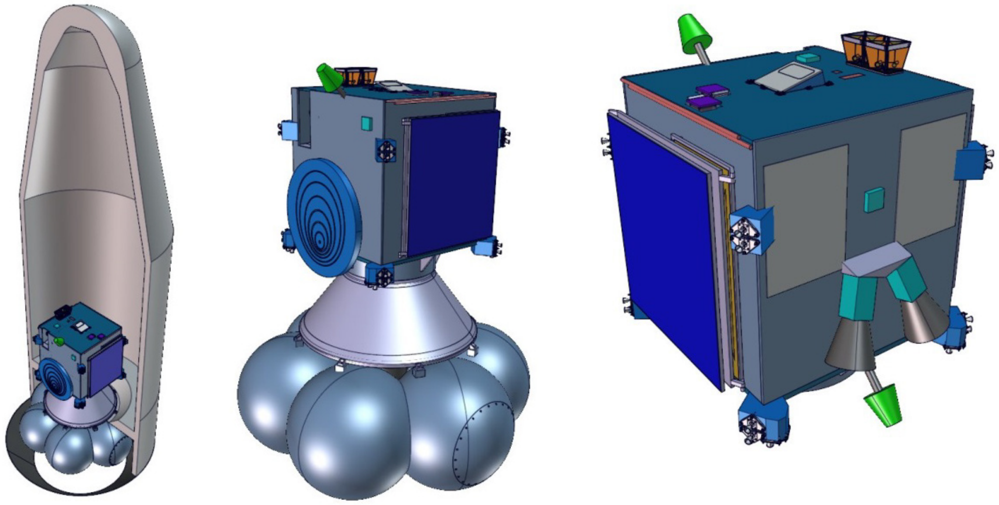


Figure 7-1: The AIM-3P spacecraft stowed inside the Soyuz Launcher Fairing

Two different views of the deployed spacecraft are shown in Figure 7-2. The solar arrays do not require a Solar Array Drive Mechanism [SADM], only a straight forward deployment. The Solar Array and the S/C side with the [High Gain Antenna] HGA are oriented towards Earth/Sun during the mission, as has been identified and agreed by relevant disciplines. The Optical Communication system is inside the S/C and has an opening in the HGA panel to have a clear view to Earth.

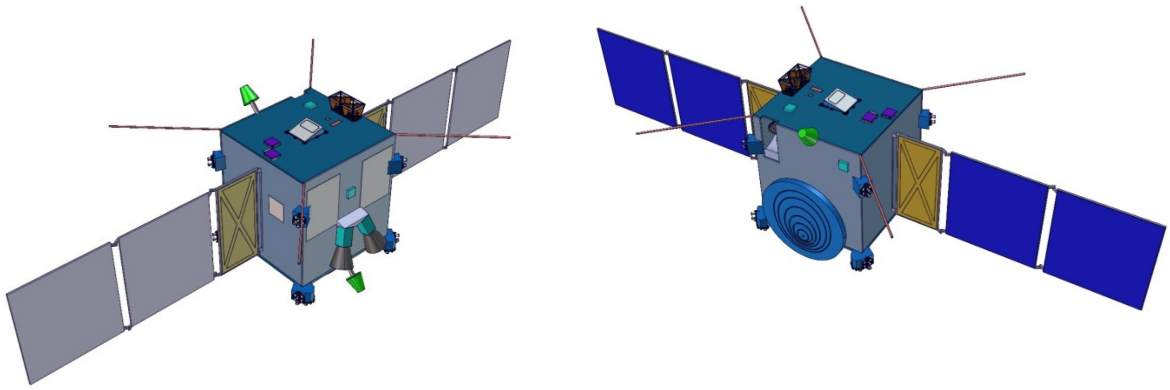


Figure 7-2: AIM-3P spacecraft in deployed configuration

Figure 7-3 shows the opening in the S/C structure for the Laser Communication instrument, including an indicative Field of View [FoV]. This FoV will have to be further designed when the overall mission and communication objectives are detailed. Also shown in Figure 7-2 and Figure 7-3 are the thruster pods with thrusters. The current configuration and orientation have been taken from the ROSETTA S/C. IN the next phases of development these will have to be detailed based on the Mission, AOCS and Propulsion requirements.

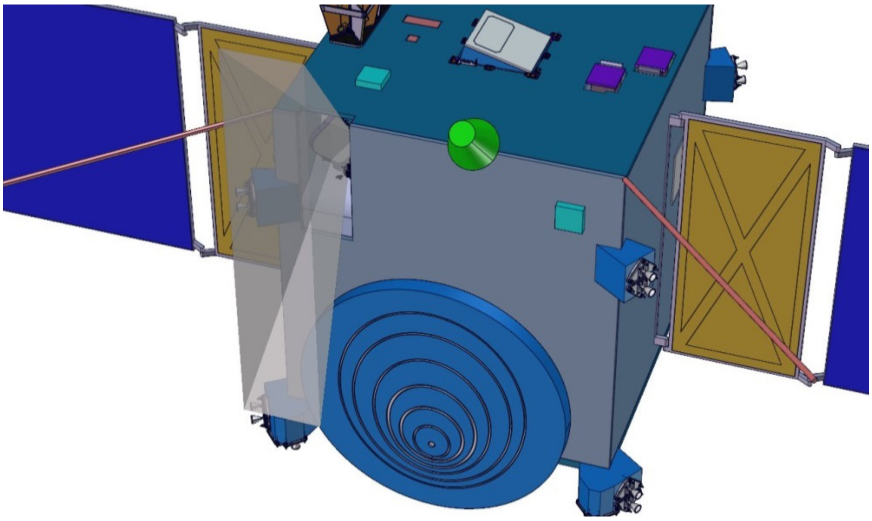


Figure 7-3: Laser Communication indicative Field of View

Based on the requirements on the spacecraft for Sun-Earth-Target orientation, the study team has identified and selected the top panel for the instruments of this mission. This panel shall provide an unobstructed view towards the asteroid. Integrated on this panel are the Visual Imaging System [VIS], the Thermal InfraRed Imager [TIRI], the MASCOT-2 lander which will be deposited on the asteroid, the radar instruments and the optional CubeSat dispensers. For the purposes of this study, two CubeSat dispensers have been included each with a 3U (3 unit CubeSat) capability. One of these dispensers has been loaded with a 3U CubeSat, while the other has a 1U and a 2U payload. The actual complement of CubeSats will depend on further study and mission definition. Figure 7-4: shows the direction in which the MASCOT-2 payload will be released. The details of this payload are described in the Instruments chapter. The details of the design are not part of this assessment. The model provided by the MASCOT-2 team contained sufficient detail for an integration of this payload into the configuration for the AIM-3P study.

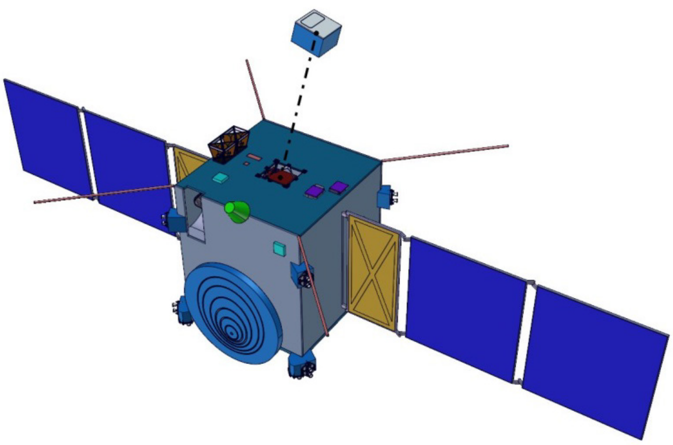


Figure 7-4: Release of the MASCOT-2 payload towards the Asteroid

Figure 7-5 and Figure 7-6 show the possible deployment of the 3U and the 2U + 1U CubeSat deployments.

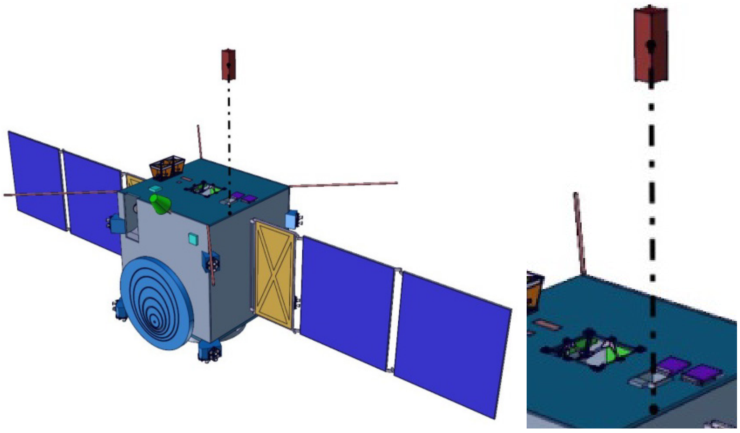


Figure 7-5: A 3U CubeSat deployment

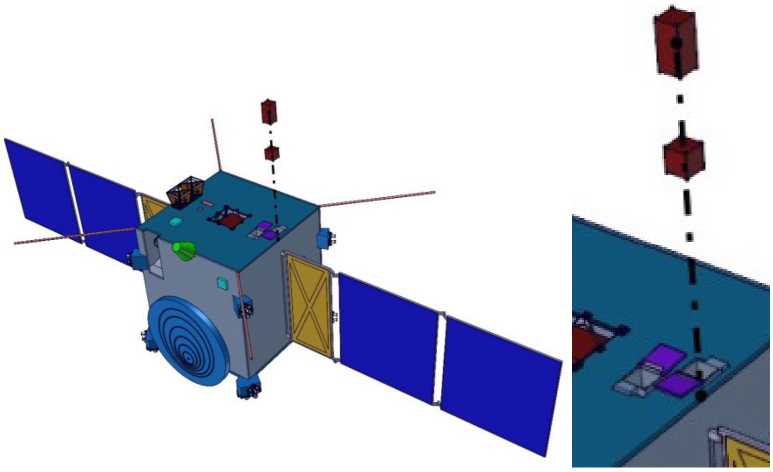


Figure 7-6: A 2U + 1U CubeSat deployment

Figure 7-7: identifies different equipment and instruments integrated into the design of the AIM-3P spacecraft.

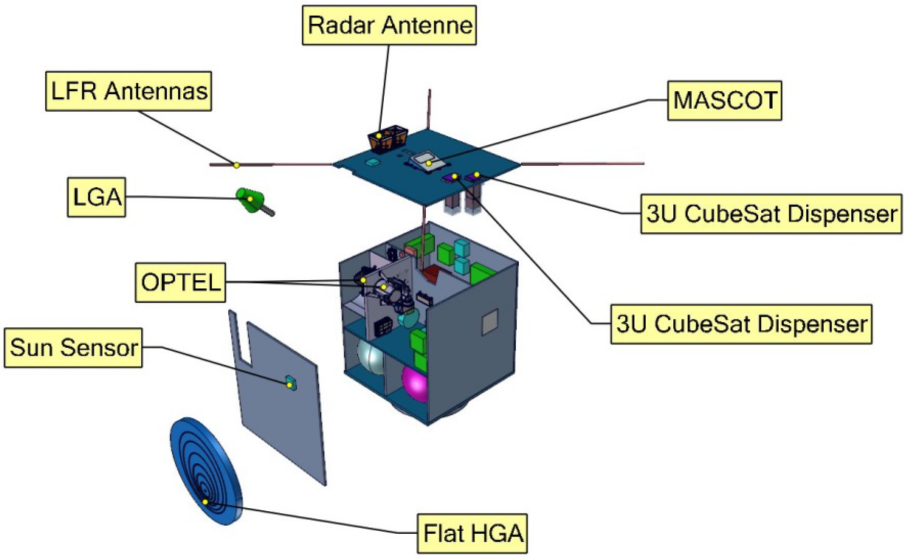


Figure 7-7: AIM-3P spacecraft external sub-systems, equipment and instruments

Figure 7-8: shows additional preliminary layout of equipment and instruments of the AIM-3P spacecraft.

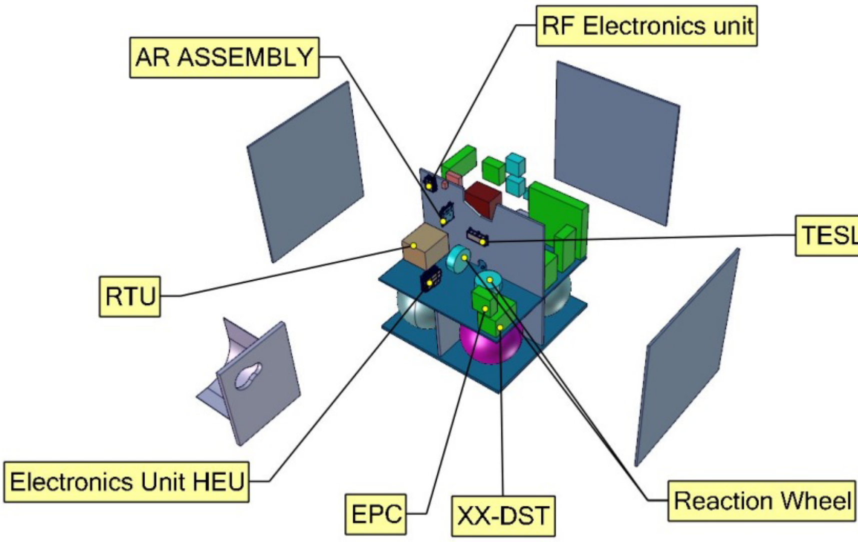


Figure 7-8: AIM-3P spacecraft internal sub-systems, equipment and instruments 1

Figure 7-9: shows further configuration of the spacecraft preliminary internal layout.

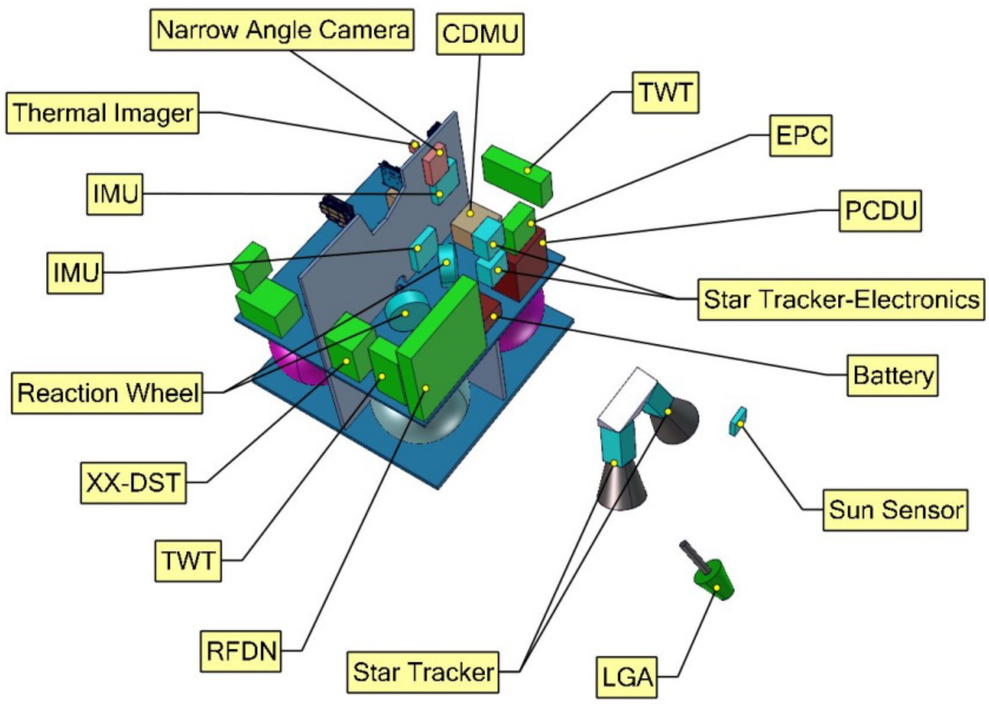


Figure 7-9: AIM-3P spacecraft internal sub-systems, equipment and instruments 2

Figure 7-10 shows the layout of the propulsion tanks, inside the S/C body.

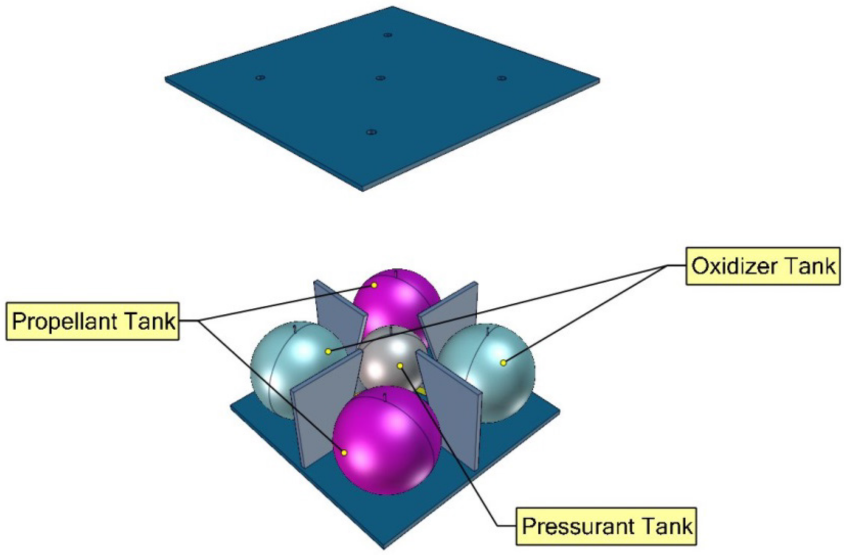


Figure 7-10 Propulsion Tank layout

The layout of all the internal equipment and instruments has been balanced within the available information. This may still change based on thermal, electrical, EMC and mass balance requirements.

Figure 7-11: provides a view of the structure and structural elements of the AIM-3 spacecraft. More details for the structures can be found in the Structures chapter.

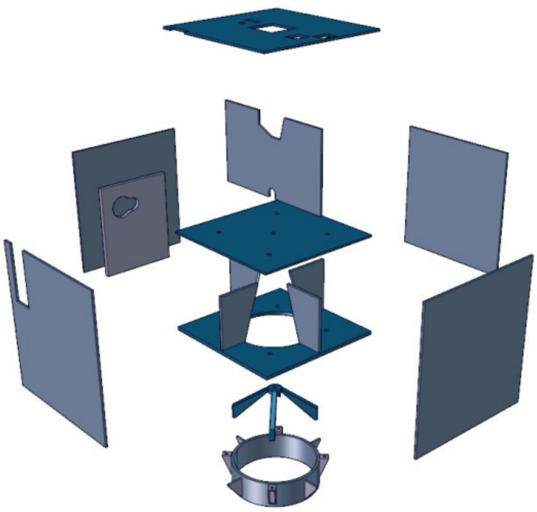


Figure 7-11: Structures layout of the AIM-3P spacecraft

7.4 Overall Dimensions

Figure 7-12: shows the main dimensions of the AIM-3P spacecraft in stowed configuration.

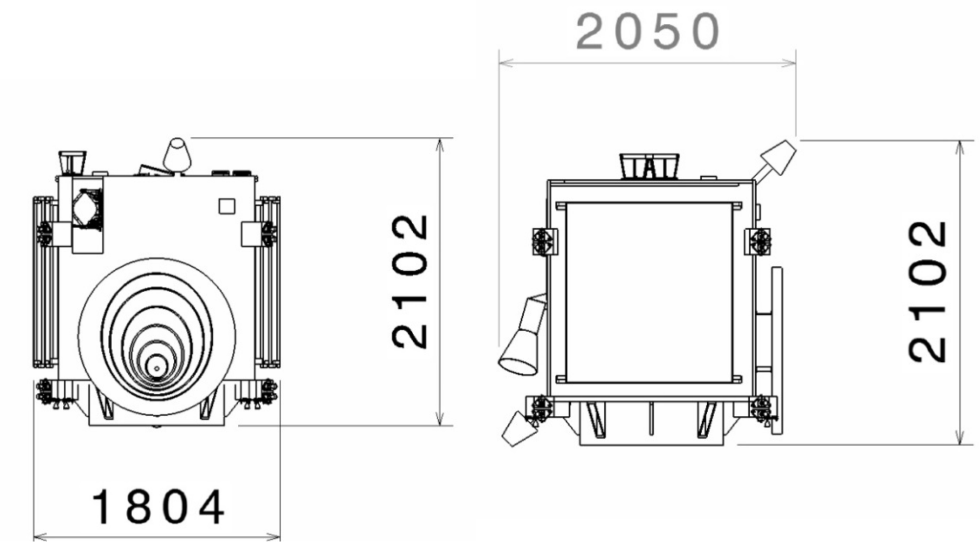


Figure 7-12: Outer dimensions for stowed AIM-3 spacecraft

Figure 7-13: shows the main dimensions of the AIM-3 spacecraft in deployed configuration.

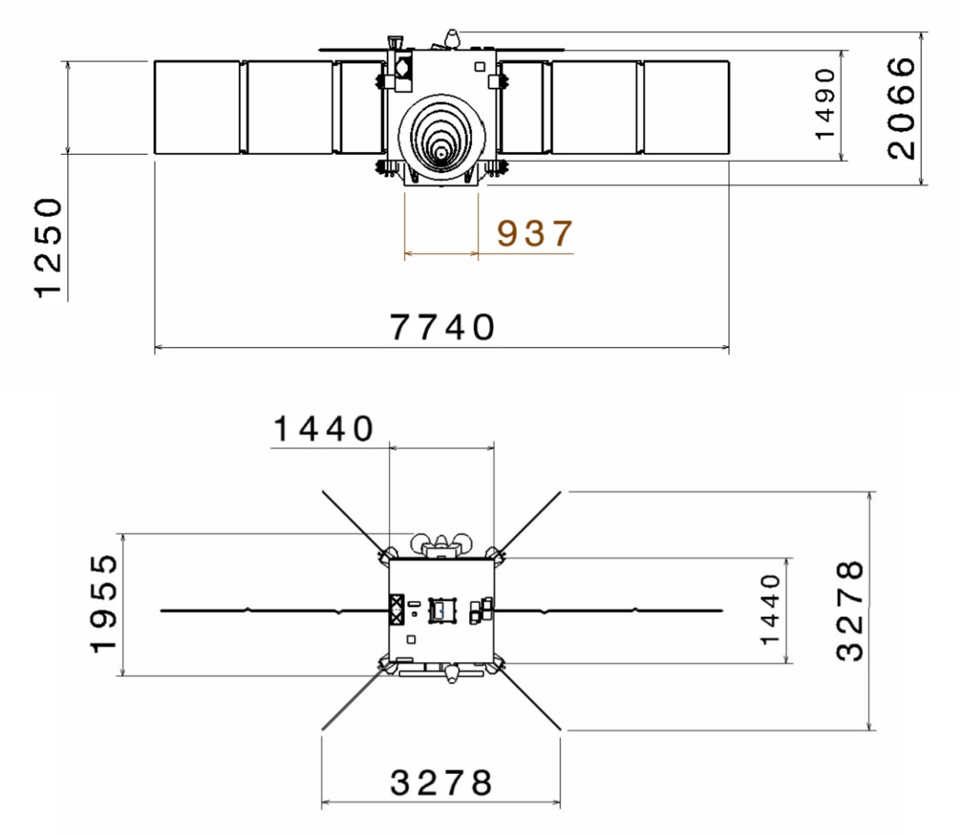


Figure 7-13: Outer dimensions for the AIM-3 deployed spacecraft

8 STRUCTURES

8.1 Requirements and Design Drivers

The structure shall fulfil the following general requirements:

SubSystem requirements		
Req. ID	**STATEMENT**	**Parent ID**
STR-010	Aim for simple load paths	
STR-020	Withstand the design limit loads without failing or exhibiting permanent deformations that can endanger the mission objectives	
STR-030	Ensures sufficient stiffness to decouple spacecraft modes from those of the launch vehicle	
STR-040	Provide support and containment for spacecraft units, equipment and subsystems	

8.2 Assumptions and Trade-Offs

A Soyuz launch from Kourou is the baseline for AIM-3P, the following Soyuz requirement applies:

- 1st Fundamental Lateral Frequency ≥ 15 Hz
- 1st Longitudinal Frequency ≥ 35 Hz

This requirement is assessed by FEM analysis within this study.

8.3 Baseline Design

The Structure for the 3-axis stabilised spacecraft requires a dedicated design, but its layout is a conventional design. A standard adapter ring is foreseen that is compatible with the existing Soyuz Ø1194-SF payload adapter (mass of 110 kg, as incorporated in the AIM-3P launch mass budget).

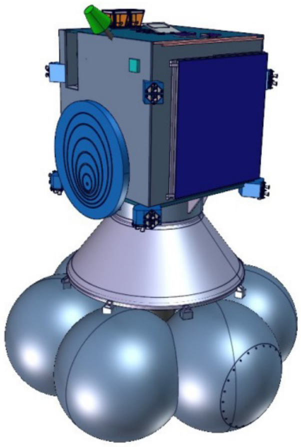


Figure 8-1: Spacecraft with payload adapter above Fregat

For AIM-3P, being the follow-up study of AIM3, no significant modifications for the S/C structure were foreseen at study kick-off unless justified by:

- Mission requirements
- Accommodation of instruments

One outcome of the AIM-3P study is that for the S/C structure no significant modifications of the structural concept are required, except moving up of the mid plate wrt AIM3 to give more room to the tanks which have grown in length.

The rational for the general structural layout is:

1. To create room for instrument placement flat panels are preferred
2. As the interface to the PLA is circular, a transition to a box structure is required
3. Load transfer from circular lower section to box-type top section is via shear panels
4. To accommodate the pressurant tank in addition to the four propellant tanks, which are rather big, placement in the centre on a lower support structure is opted for; this requires a cut-out in the lower shear panels which ideally would be of cross-type, uninterrupted
5. As the load path continues to the outer panels, the upper part of the box consists of one shear panel only; this help to create more room and options to place instruments
6. The propellant tanks need some extra support connection to the cylindrical lower section of the structure
7. In order to comply with stiffness requirements from Soyuz also the shear panels need extra support to the cylinder; the same will probably be required for compatibility with other launchers.

The last point, namely stiffening the upper box section wrt the lower cylindrical section, is an outcome of the FEM analysis; this point is explained in more detail below.

The baseline design is illustrated in the following pictures:

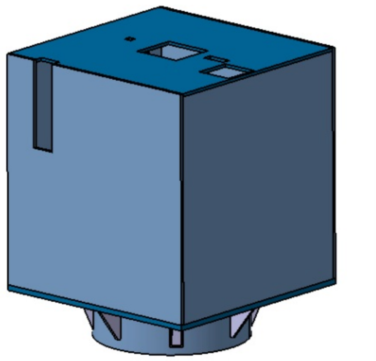


Figure 8-2: Spacecraft overall structure

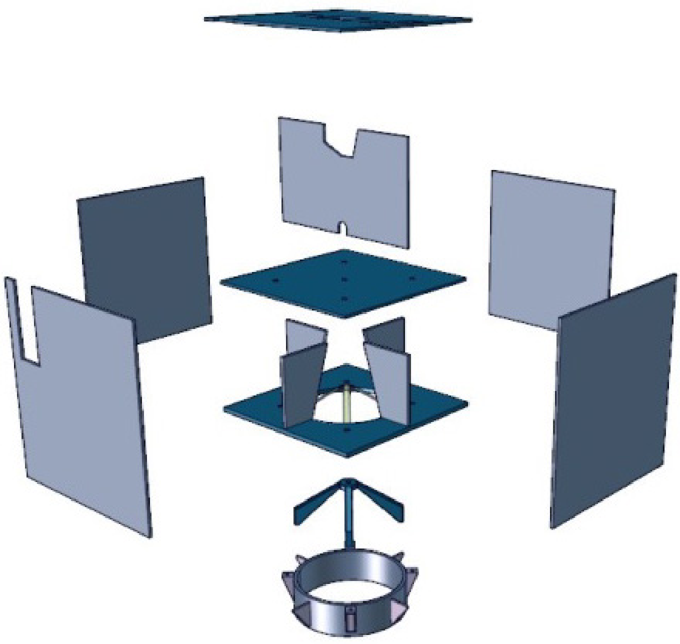


Figure 8-3 : Spacecraft structure exploded view

Regarding the material choice of the structure, metallic structures are preferred at this stage, as thermal control of the units is considered to be achieved by conductive cooling, refer to the Thermal section in this document. As such, sandwich panels with aluminium face sheets and core are chosen as they provide excellent mechanical performance at minimal weight. The core takes the shear loads and creates a distance between the face sheets which take the in-plane stresses and allow units to be attached.

8.4 List of Components

Figure 8-4 illustrates the different structural components of AIM-3P; it consists of three horizontal and four outer vertical panels.

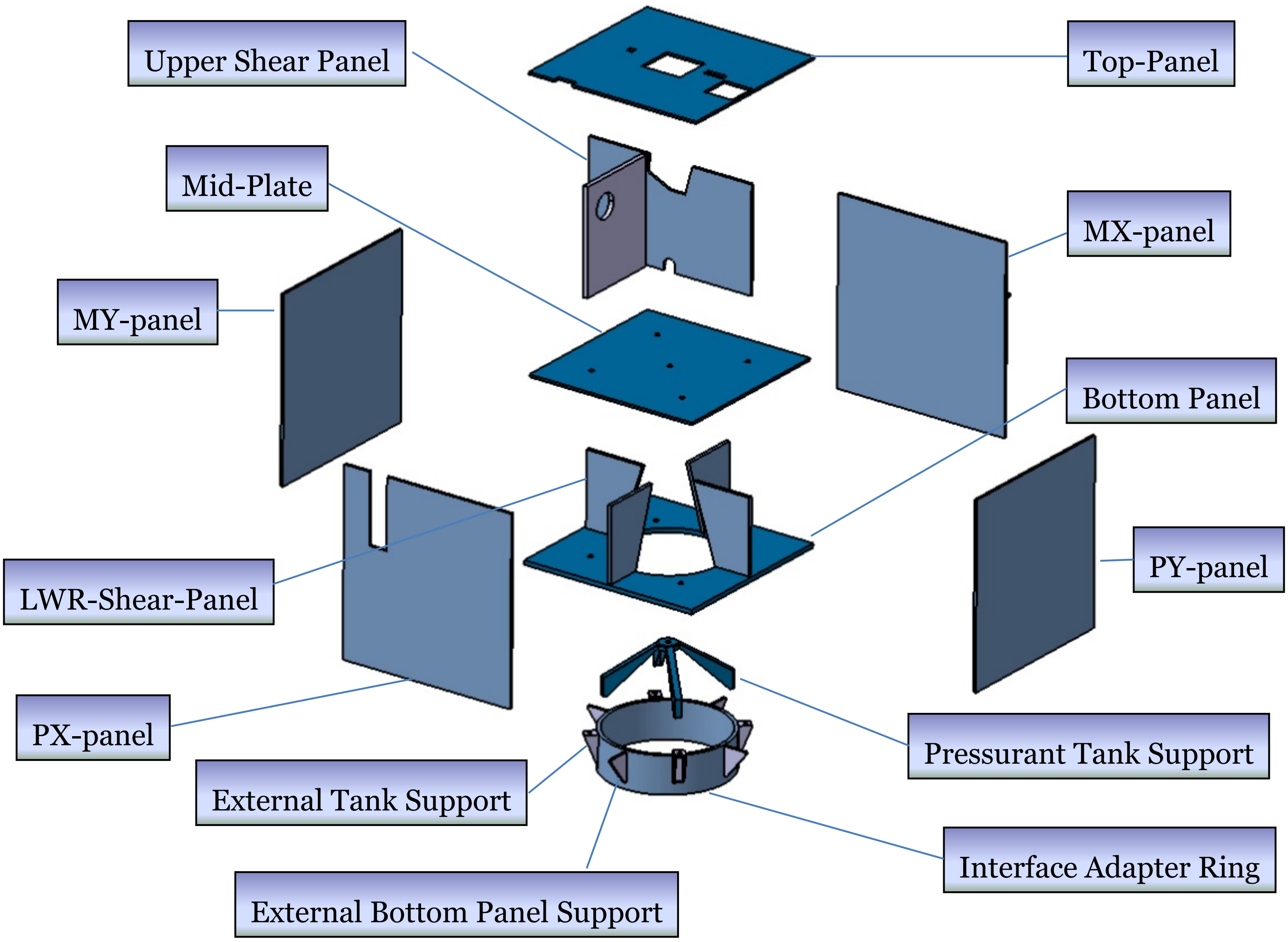


Figure 8-4 : Spacecraft structural items

The standard adapter foreseen is compatible with the existing Soyuz Ø1194-SF payload adapter with a mass of 110 kg.

More details on the chosen configuration may be found in the Configuration chapter.

The different items are made of solid aluminium or aluminium honeycomb.

Item	Material	
Interface Adapter	Al - 3.4364.T7351 (0.5 mm)	5/32-5056-.0007 (30 mm)
Interface Adapter Ring	Aluminium	
Bottom Panel	Al - 3.4364.T7351 (0.5 mm)	5/32-5056-.0007 (20 mm)
Mid-Plate	Al - 3.4364.T7351 (0.3 mm)	5/32-5056-.0007 (20 mm)
Top-Panel	Al - 3.4364.T7351 (0.3 mm)	5/32-5056-.0007 (20 mm)
PY-/MY-panel	Al - 3.4364.T7351 (0.2 mm)	5/32-5056-.0007 (20 mm)
PX-/MX-panel	Al - 3.4364.T7351 (0.2 mm)	5/32-5056-.0007 (20 mm)
LWR-Shear-Panel	Al - 3.4364.T7351 (0.2 mm)	5/32-5056-.0007 (20 mm)
Upper Shear Panel	Al - 3.4364.T7351 (0.2 mm)	5/32-5056-.0007 (20 mm)
External Tank Support	Aluminium	
Pressurant Tank Support	Aluminium	
External Bottom Panel Support	Aluminium	

Table 8-1: Satellite structural items and materials

8.5 Mass Budget

The following table lists

- Dimensions in terms of item area and thickness
- Materials type (solid or sandwich with thicknesses)
- Properties (density)
- Item mass and total mass.

It should be noted that the External Bottom Panel Support is listed together with External Tank Support.

ELEMENT 1 -		Project 0.1 MATERIAL PROPERTIES								
		Sandwich plate							Solid Plate (give only core depth)	
Create a New Design Sheet / Add a new unit / Item	Nr.	sheet material	skin thickness mm	skin density [kg/m3]	sandwich: core thickness solid: plate thickness beam: cross section [mm] or [mm²]	Core material	Core density [kg/m³]	Glue density [kg/m²]	Material	density [kg/m³]
EXAMPLE	2	CFRP [0,60,-60]s	0.75	1654	30	5/32-5056-.0007	41.652	0.35	STEEL	7950
IF_Adapter	1	Al - 3.4364.T7351(0.5 mm)	0.5	2800	30	5/32-5056-.0007	41.652	0.35		
IF_Adapter_ring	1				20				ALUMINUM	2770
Bottom_Panel	1	Al - 3.4364.T7351(0.5 mm)	0.5	2800	20	3/16-5056-.0007	32.04	0.35		
Mid-Plate	1	Al - 3.4364.T7351(0.3 mm)	0.3	2800	20	3/16-5056-.0007	32.04	0.35		
Top-Panel	1	Al - 3.4364.T7351(0.3 mm)	0.3	2800	20	3/16-5056-.0007	32.04	0.35		
PY-panel	1	Al - 3.4364.T7351(0.2 mm)	0.2	2800	20	3/16-5056-.0007	32.04	0.35		
PX-panel	1	Al - 3.4364.T7351(0.2 mm)	0.2	2800	20	3/16-5056-.0007	32.04	0.35		
MX-Panel	1	Al - 3.4364.T7351(0.2 mm)	0.2	2800	20	3/16-5056-.0007	32.04	0.35		
MY-Panel	1	Al - 3.4364.T7351(0.2 mm)	0.2	2800	20	3/16-5056-.0007	32.04	0.35		
LWR-Shear-Panel	4	Al - 3.4364.T7351(0.2 mm)	0.2	2800	20	3/16-5056-.0007	32.04	0.35		
Upper_Shear_Panel	1	Al - 3.4364.T7351(0.2 mm)	0.2	2800	20	3/16-5056-.0007	32.04	0.35		
External-Tank_Support	8				10				ALUMINUM	2770
Press-tank_Support	1				10				ALUMINUM	2770
Miscellaneous	1				10				ALUMINUM	2770
14										

ELEMENT 1 -			DIMENSION							
Create a New Design Sheet / Add a new unit / Item	Nr.	unit density Panel: area density beam: length density [kg/m²] or [kg/m]	Dim1 [m]	Dim2 [m]	Dim3 [m]	Dim4 [m]	area [m2]	Item mass [kg]	M_struct [kg]	Material
EXAMPLE	2	4.081			1	1	1.00	4.081	6.37	STEEL
IF_Adapter	1	4.400			1.00	0.9	0.90	3.960	4.13	sandwich
IF_Adapter_ring	1	0.000			1.00	0.00	0.00	7.000	7.00	ALUMINUM
Bottom_Panel	1	3.791			1.135	1.135	1.29	4.883	4.88	sandwich
Mid-Plate	1	2.671			1.44	1.44	2.07	5.538	5.54	sandwich
Top-Panel	1	2.671			1.405	1.405	1.97	5.272	5.27	sandwich
PY-panel	1	2.111			1.44	1.525	2.20	4.635	4.64	sandwich
PX-panel	1	2.111			1.44	1.525	2.20	4.635	4.64	sandwich
MX-Panel	1	2.111			1.44	1.525	2.20	4.635	4.64	sandwich
MY-Panel	1	2.111			1.44	1.525	2.20	4.635	4.64	sandwich
LWR-Shear-Panel	4	2.111			0.5	0.487	0.24	0.514	0.51	sandwich
Upper_Shear_Panel	1	2.111			1.44	0.87	1.25	2.644	2.64	sandwich
External-Tank_Support	8	0.000			1	0.4	0.40	0.400	0.40	ALUMINUM
Press-tank_Support	1	0.000					0.00	3.000	3.00	ALUMINUM
Miscellaneous	1	0.000					0.00	7.000	7.00	ALUMINUM
14									63.27	

Table 8-2: Satellite structural mass breakdown

Based on this table, the calculated mass is 63.27 kg without margins and 75.92 kg with 20% margins.

8.6 FEM Model for Soyuz Compatibility

FEM description:

- Launch configuration
- Overall mass 730 kg
- Tanks & support, propellant & solar arrays (mass) modelled
- Structural mass distribution as of study model (except "Miscellaneous")
- Instruments, etc. masses are "smeared" onto panels.

The aim of the FEM is to roughly represent the global modes of the S/C. Local modes are not representative. The actual position of the instruments is not considered; instead the mass of all satellite parts except the components in chapter 8.4 and the tanks is smeared over PX-/MX-/PY-/MY-/Bottom-/Top-Panels and Mid-Plate. The solar array mass is allocated to the PY-/MY-Panels and evenly distributed.

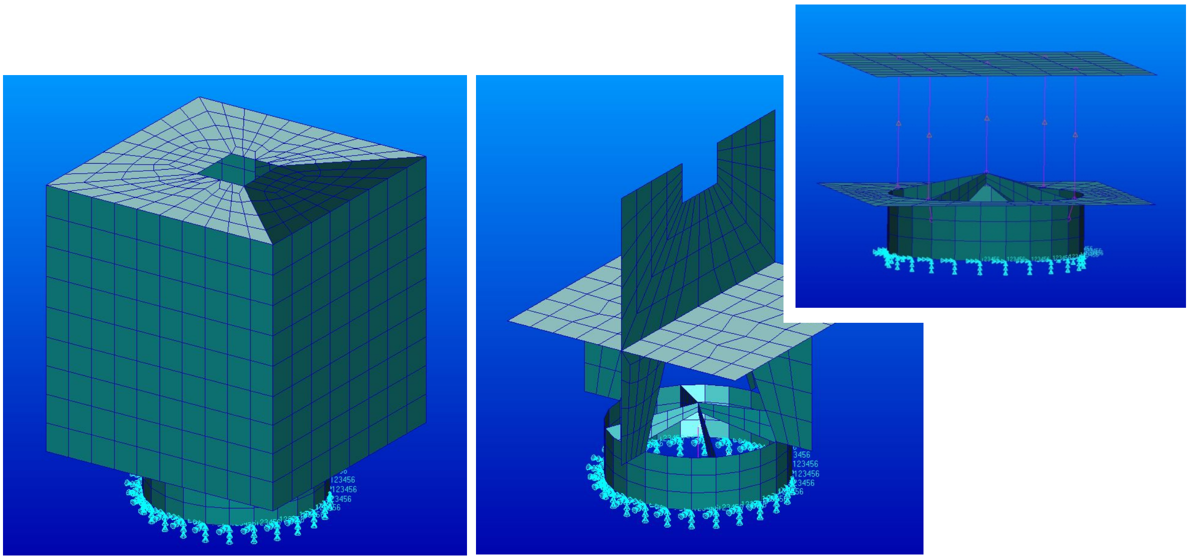


Figure 8-5: Spacecraft FEM

In Figure 8-5 mesh, boundary conditions and simplified tanks' models are shown.

The tanks are modelled as rigid elements with the tank plus propellant mass concentrated half way between lower and upper attachment points to the panels; these positions are very close to the CoGs of each tank. The lower tank flange is fixed to the Bottom Panel with all translational degrees of freedom, whereas the upper tank flange can move freely in longitudinal direction.

8.7 Normal Modes Analysis

As a result of the analysis the structural design and in particular the mass budget for the structure will be reassessed.

8.7.1 Initial Results

The first analysis is run without External Bottom Panel Support; this is identical to the structural design of AIM3 with higher Mid-plate.

The stiffness is marginally too low as for the lateral modes the frequencies of 14.6 Hz & 14.7 Hz do not meet the Soyuz requirement. The longitudinal stiffness requirement is met.

- 1st Fundamental Lateral Frequencies 14.6 Hz & 14.7 Hz < 15 Hz
- 1st Longitudinal Frequency 48.6 Hz ≥ 35 Hz

Increasing the face sheet thickness of the Lower Panel from 0.5 mm to 0.7 mm does only slightly increase the lateral frequencies; but the requirement is not met. A significant relative displacement between Interface Adapter and Bottom Panel can be noticed.

8.7.2 Final Results

In order to stiffen up more the Interface Adapter to Bottom Panel connection which shows high relative motion, four External Bottom Panel Supports are added between the External Tank Supports such that there is a support every 45 degrees.

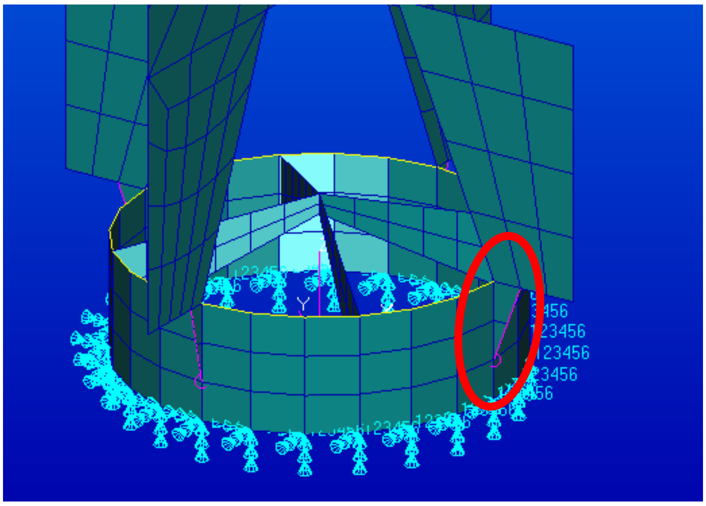


Figure 8-6 : Spacecraft External Bottom Panel Supports

As a result the lateral frequencies increase by about 3 Hz and the Soyuz requirements are met:

- 1st Fundamental Lateral Frequencies 17.5 & 17.5 Hz ≥ 15 Hz

The longitudinal stiffness further increases also:

- 1st Longitudinal Frequency 60.2 Hz ≥ 35 Hz

The deformed shapes of the first global bending and longitudinal modes are shown below (PX-Panel and MY-Panel removed for clarity).

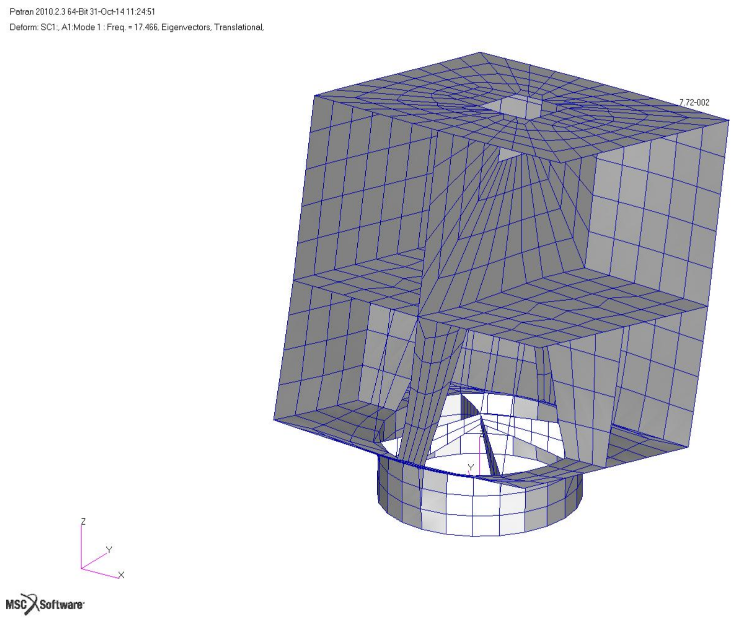


Figure 8-7 : Spacecraft global lateral mode in yz-plane

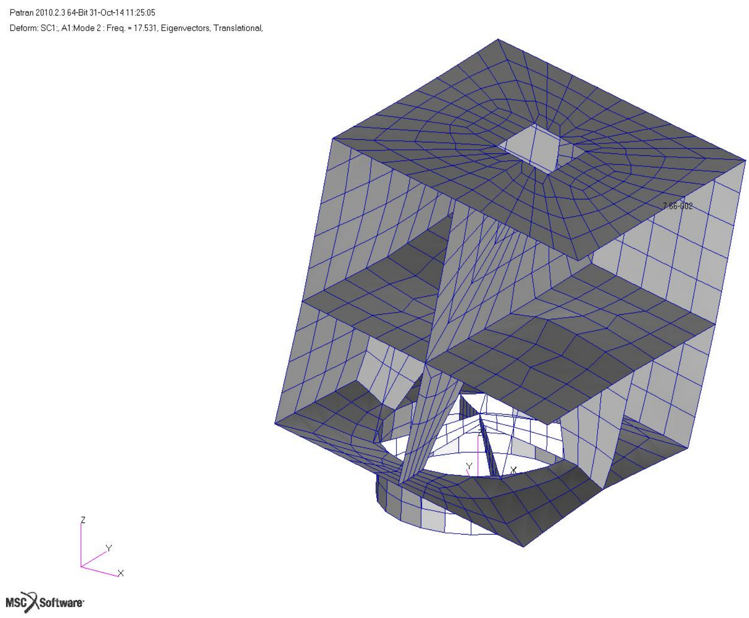


Figure 8-8 : Spacecraft global lateral mode in yz-plane

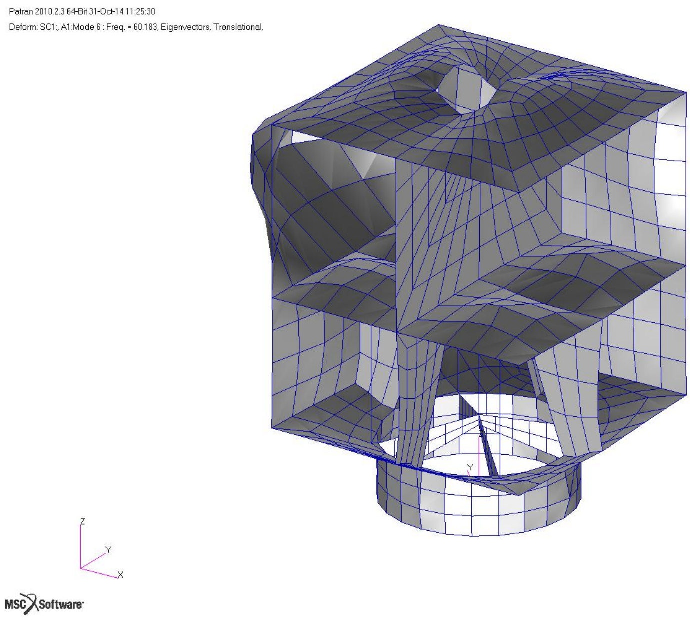


Figure 8-9 : Spacecraft global longitudinal mode

The reinforcement brings the structural mass without margin up by 1.6 kg to 63.27 kg as shown in Table 8-2.

8.8 Technology Requirements

For this mission, no new technology developments wrt to structural components are required.

9 PROPULSION

9.1 Requirements and Design Drivers

The requirements relating to the chemical propulsion systems are to provide, for the various mission phases (deep space manoeuvre, rendezvous with the binary system, Co-flying position).

- The required velocity increments established during the study by Mission Analysis and GNC:
  - Direct insertion into interplanetary trajectory by Soyuz/Fregat launcher. No velocity increment during departure phases
  - 1250 m/s for orbital manoeuvres (DSM and rendezvous with the asteroid)
  - 100 m/s velocity increment as AOCS budget as a conservative assumption provided by mission analysis. This is also conservative, when including the 100 % AOCS margin. Therefore, no other margin was applied to the AOCS budget
- The required thrust levels for main manoeuvres
- The required attitude control capabilities (RCS thrusters)
- Provide force-free and torque-free translation and rotation of the spacecraft respectively.

The inputs with respect to the spacecraft mass were taken from the first iteration, but the assumptions are correct and in line with the final dry mass allowing additional margin:

- 340 kg of spacecraft dry mass, excluding the propulsion subsystem (PSS) hardware, including payload.

Design drivers are the performance characteristics of the various available thrusters and the trade-off concerning system complexity versus performance. Given the large Δv to be provided, high performing thruster models are preferable to reduce the subsystem mass. The low-cost philosophy of the overall spacecraft design might require a more simple, less performing, and therefore heavier PSS. Several PSSs were considered and a trade-off was performed.

The number of RCS thrusters is derived from the AOCS requirements (force-free / torque free movement). A constraint due to the relatively small spacecraft mass is introduced from the minimum impulse bit requirement (communicated to be smaller than 50 mNs).

Furthermore the following was applicable:

- Preferably European COTS component selection
- Single fault tolerant system design.

9.2 Assumptions and Trade-Offs

9.2.1 Thrust Level

During the study it was discussed to perform the main velocity increments with an additional set of small thrusters (Rosetta like, with four times 10 N or four times 22 N) or with a larger apogee engine (400 N) in order to reduce the burn time and limit the thrusters' peak power consumption over the burn period. With a 400 N engine, the main burn time would be 10 times shorter than the one using 4 x 10 N thrusters. Assuming a spacecraft wet mass of about 600 kg and 40 N thrust the firing time would be 5.2 h if the complete velocity increment was provided at once (conservative time estimate). Using the larger engine this could be reduced to 0.52 h.

However, it was found that the power consumption of the smaller thrusters can be readily supplied by the power subsystem, whilst adding the 400 N engine would add substantial dry mass to the PSS. Also, cost and system complexity would be largely increased. From the Rosetta mission it is known that the Airbus Defence and Space ACS thrusters (10 N class) have been qualified for a burn time of 10 h and therefore the complete velocity increment could be provided in one burn. In fact, for the AIM-3 study the "to be provided" velocity increment is also split up into the DSM and rendezvous manoeuvres ranging from 479 m/s down to 12.5 m/s with the above mentioned total of 1250 m/s.

9.2.2 Thruster Options Investigated in this Study

Thruster	Thrust [N]	I_{sp} [s]	Propellant
ADS S 10-26	10 N	291	MMH/MON-3
ADS S22-2	22 N	290	MMH/MON-3
ADS S400-12	400 N	318	MMH/MON-3
ADS 1 N	1 N	220	Hydrazine
MOOG MONARC-1	4.5 N	226	Hydrazine
ADS CHT-20	20 N	218	Hydrazine
MOOG DST-11	22 N	307	Hydrazine/MON-3

Table 9-1: Thrusters options investigated during the AIM-3 study

The thrusters used in this study are summarised with their key parameters in Table 9-1. When using a 12 + 12 thruster configuration the thrust vector is in line with the spacecrafts main axis. When using a 8 + 8 configuration the thrusters are canted by 45 degrees, the loss in efficiency was accounted for by adding an additional Δv budget to the AOCS budget as a function of the canting angle.

9.2.3 PSS Trade-Off

At the beginning of the study a bipropellant propulsion (BP) subsystem (MMH/MON-3) was used as a baseline (see table below, first row). It contained 24 (12 + 12 redundant) thrusters with 22 N thrust each. During the study several other propulsion subsystems were studied, including monopropellant (MP) options. They are all summarised in Table 9-2, the "Mass penalty wrt baseline [kg]" has been calculated using the PSS mass of the first studied, then "baseline" PSS. The assumed spacecraft dry mass, including the payload and excluding the PSS dry mass was different than the one given in the final requirements, since the trade-off took place early during the study. The baseline design (see next section) is of course based on the final requirements.

PSS	*Dry Mass [kg]*	*Propellant [kg]*	*Total [kg]*	*Mass penalty wrt baseline [kg]*	*Power (operation long burn) [W]*	*Comments / Complexity*
BP 24x22N	48.7	179	228	**0**	64 (4x16 W)	Good compromise between complexity and performance
BP 24x10N	48.9	186	235	**7**	32 (4x8 W)	Impulse bit smaller than 22 N thrusters, used on Rosetta, great heritage
BP 24x22N + 1x400N	56+x?	164	230?	**2?**	35 + RCS	Most complex, costliest, large AE dimensions
MP 24x20 N	47.3	252	299	**71**	64 (4x16 W)	Simplest configuration, cheapest
MP 16x1N + 8x20N	46.3	253	299	**71**	64 (4x16 W)	Can be used if minimum impulse bit too large, still cheap
DM 16x1N + 8x22N	50.7	171	208	**-20**	80	Lowest PSS mass due high Isp thrusters

Table 9-2: Trade-off of propulsion subsystem based on *preliminary values* for spacecraft dry mass

As a second option, a bipropellant PSS using the same thrusters as used on Rosetta was studied. Also the minimum impulse bit matched the requirements better and also the power consumption in continuous burn phases is low. This option (row 2, green background in the table) was finally baselined for the AIM-3 spacecraft due to its superior performance compared to the hydrazine PSS.

The third option displayed in the table is the PSS using the larger 400 N apogee engine to provide more thrust in the main burn phases was quickly ruled out due to the system complexity, dry mass (the engine frame and thermal shield was assumed to have a mass of 10 kg; this is represented by the "x" in the table), and the higher costs.

After this, two monopropellant subsystems (hydrazine) were studied, the first one using 12+12 20 N thrusters, the second one using 8+8 1 N thrusters for the RCS/AOCS and 4+4 20 N thrusters for the main burn. One can clearly see that even though the dry mass is smaller, the total PSS wet mass is substantially larger due to the significantly smaller specific impulse of the hydrazine thrusters.

As a backup option the dual mode PSS is given in the last row of the table. A dual mode PSS combines the advantage of the high specific impulse of a bipropellant thruster (hydrazine / MON-3, 4+4 times 22 N) with the simplicity of the monopropellant thrusters for AOCS (hydrazine, 8+8 times 1N). It has to be said that for this configuration it still needs to be checked if the small thrust level provides sufficient control authority of the spacecraft. Therefore, for the backup option described in 9.5.1 was equipped with the MOOG 4.5 N ACS thrusters as shown in Table 9-1.

9.3 Baseline Design

PSS	Dry Mass [kg]	Propellant [kg]	Total [kg]	Power (operation long burn) [W]
BP 24x10N	68.8	286	354.8	32 (4x8 W)

Table 9-3: Bi-Propellant Baseline design

9.3.1 Propellant Budgets

From the ΔV and AOCS inputs, the following propellant budgets have been determined, using the margin policy as defined in 9.3.3:

- 86 kg of propellant

9.3.2 Dry Mass

The dry mass of the propulsion subsystem is the sum of all components in the subsystem design including propellant residuals and pressurant.

- 68.8 kg dry mass

9.3.3 Margin Policy

The following margins have been considered:

- 2% on propellant residuals considered for sizing of the tanks and included in the PPS's dry mass budget
- 2% on main burn ΔV budget for steering losses (not applied to AOCS budget)
- 5% on main burn ΔV budget and steering losses for reserves
- The component margins have been set according to the system engineering guidelines and are given in Table 9-4
- The standard margin philosophy with respect to the AOCS budget is adding 100 % to the requirement. As mentioned in section 9.1 no additional AOCS margin was applied, since this was already included in the requirement and the value of 100 m/s is still considered to be conservative.

9.3.4 Architecture

The general layout of all chemical spacecraft propulsion systems is in principle the same, but important differences can be found in the:

- Size, shapes and number of tanks for propellant and pressurant
- Number of thrusters and their thrust level and specific impulse
- Failure tolerance, priming strategy, and isolation strategy and their impact on the components and logical design of the subsystem.

Note that in these architectures, multiple oxidizer tanks are connected to each other and that multiple fuel tanks are connected to each other. The exact draining of propellant components and depletion of one tank before the other has not been assessed.

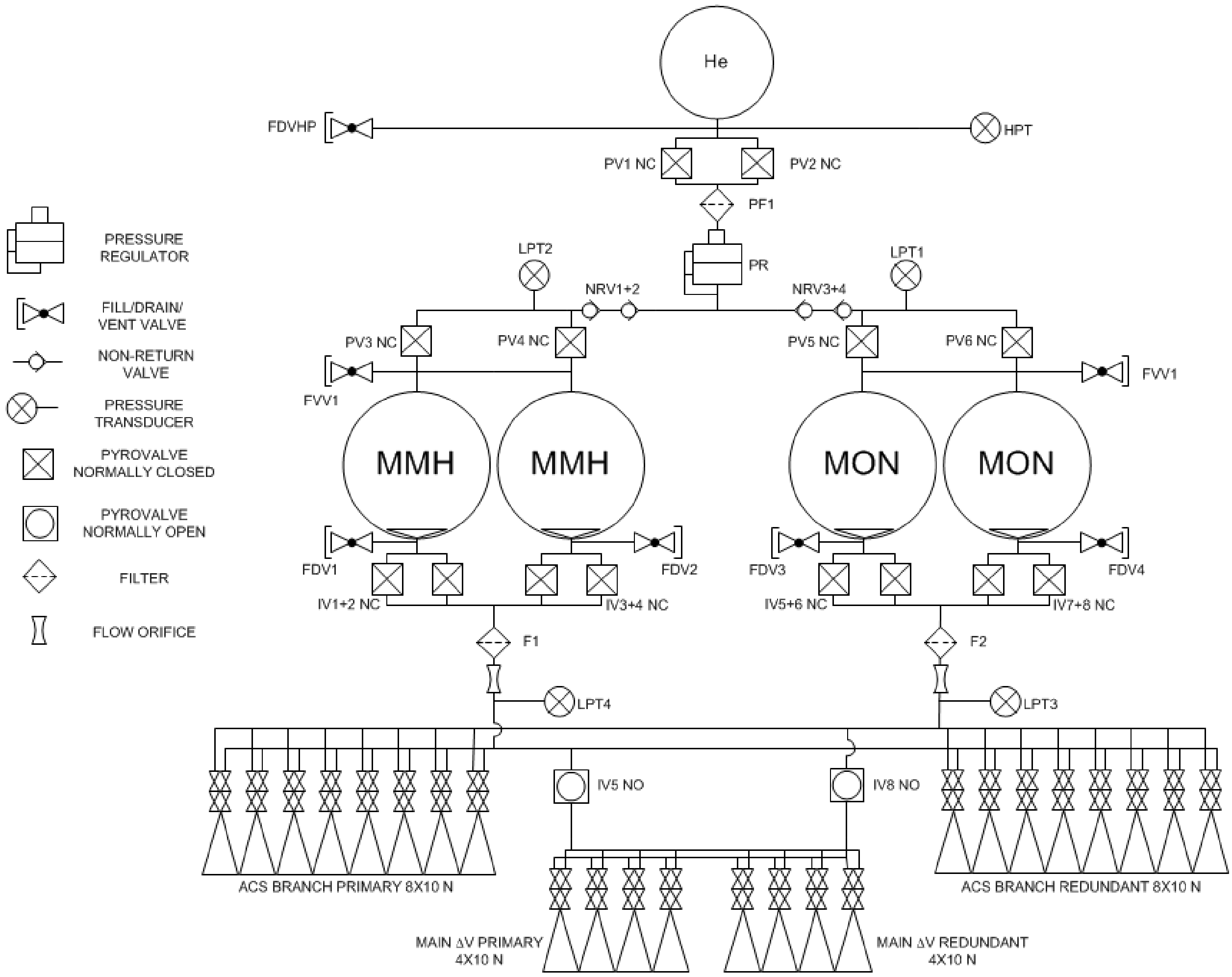


Figure 9-1: Baseline bi-propellant design

In Figure 9-1 the schematic of the baselined bi-propellant PSS is shown. On the top of the propellant tanks the pressure control assembly is located. The pressurant chosen is helium which is contained in the pressure vessel. The parallel redundant pyrovalves are fired for the first pressurisation of the system just before the DSM approximately 2 month after launch. The pressure regulator reduces the pressure from the pressurant tank to under the MEOP of the propellant tanks.

The propellant tanks are also isolated further downstream prior to first pressurisation by another series of redundant pyrovalves. FDVVs are located on top and on the bottom of the tanks for filling, venting, and draining purposes. Non return valves are installed in between the pressure regulator and the propellant tanks to avoid any backflow of the propellant into the pressure control assembly.

Normally closed pyrovalves (parallel to be 1 FT) separate the propellant from the thruster valves. A filter catches possible debris which is caused by the firing of the pyrovalves. Orifices limit the fuel flow and therefore avoid waterhammer effects in the piping during priming. Two branches of 8 ACS thruster are installed as well as two branches of 4 main burn thrusters. The main burn thrusters can be isolated from the rest of the PSS by two normally open pyrovalves after arrival at the asteroid. Each thrusters is equipped with a latch valve and a flow control valve which give together with the tanks' downstream pyrovalves 3 barriers against propellant leakage at the launch pad. Pressure transducers are installed to measure the high helium pressure, the reduced MEOP and the thruster feed pressure (both low pressure).

9.3.5 Isolation of Equipment

The first pressurisation of the system takes place before just before the first main burn for the DSM is planned, which is supposed to take place about two month after launch. After that, the PSS stays typically 6 month inactive until the final arrival manoeuvre at Didymos (see mission analysis chapter) in planned. Sometimes the pressure controller needs to be isolated, depending on its internal leakage rate, the ullage in the propellant tanks, and the duration of the mission. The problem might be that due to the leakage through the pressure controller the pressure in the tanks exceeds the qualified MEOP. Therefore, an analysis has been carried out to define whether isolation equipment is needed in the pressure control assembly to isolate the pressure regulator after the DSM. The assumption for each step are conservative.

Internal leakage rate of a typical pressure controller:

$$1x10^{-4}scc/s \ = \ 1x10^{-8}scm/s$$

This translates into a leakage rate of helium in kg/s:

$$1.784710^{-9}kg/s$$

And a leaked mass after 180 days:

$$m_{leak} = 0.0277\ kg$$

The lowest DSM size is reported as 171 m/s by mission analysis which corresponds to 27 kg of propellant. The ullage volume of the tanks based on the current filling ratio is:

$$V_{ullage} = 0.042m^3$$

After the DSM the ullage volume is increased corresponding to the propellant used:

$$V_{ullage,DSM} = 0.088m^3$$

This volume is now filled by the leaked helium and together with the leaked helium mass it corresponds to a pressure increase due to leakage of:

$$\Delta p = 2.113\ bar$$

The MEOP of the tanks which are currently in the baseline option is 24.6 bar. Considering the expected pressure increase after the DSM of Δp=2.113 bar, a tank pressure of 22.49 bar shall not be exceeded. The maximum thruster inlet pressure for the baseline hardware is 22 bar, for the nominal thrust at the nominal mixture ratio this value is at approximately 17 bar. The pressure drop of the filter, orifice and piping downstream of the tank can be safely assumed to be less than 1 bar. Therefore, the lowest to be expected pressure at the thruster inlet is 21.49 bar at the end of the DSM, which is fully within the operational box of the thrusters delivering the needed thrust and nominal I_{sp}. Therefore, an isolation of the pressure regulator is not necessary.

9.3.6 Mass-flow Rate Compatibility

During the main burns of the four thrusters for DSM and asteroid arrival manoeuvre the tanks need to be pressurised to ensure minimum required inlet pressure at the thruster face for highest performance. Therefore, the pressure regulator needs to be able to allow for sufficient helium mass-flow rate into the tanks.

The maximum by the manufacturer stated thruster mass-flow rate is 4.2 g/s. In total, assuming four thrusters active during main burns, 0.0168 kg/s are drained from the tanks. This corresponds to a total volumetric flow rate of the propellant of:

$$\dot{V}_{prop,tot} = 1.484x10^{-5}m^3/s$$

The maximum mass-flow rate for the pressure regulator is 0.6 g/s. At the reduced maximum operating pressure which was calculated in the section above, this translates into a gas volumetric flow rate of:

$$\dot{V}_{pressurant,max} = 1.83x10^{-4}m^3/s$$

Which is an order of magnitude higher than what is depleted by firing the four main thrusters. Therefore, the selection of the pressure regulator is justified.

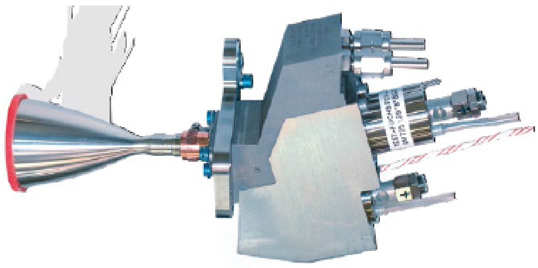
9.4 List of Equipment

The list of equipment for the baseline configuration is given in Table 9-4 with the components masses, the quantity of each component, the mass per component and the subsystem margins applied. The helium mass (pressurant) and propellant residuals are also accounted for in this table.

Unit Name	Quantity	Mass per quantity excl.	Maturity Level	Margin	Total Mass incl. margin
Pressurant tank	1	8.10	Fully developed	5	8.5
Pressurant	1	1.50	Fully developed	5	1.6
MMH tank	2	6.40	Fully developed	5	13.4
MON tank	2	6.40	Fully developed	5	13.4
Propellant residuals	1	5.72	Fully developed	0	5.7
Thrustes (Low)	24	0.65	Fully developed	5	16.4
Gas pyro valves	6	0.15	Fully developed	5	0.9
Pressurant filters	1	0.08	Fully developed	5	0.1
Liquid pyro valves	10	0.30	Fully developed	5	3.1
LP transducer	4	0.22	Fully developed	5	0.9
HP transducer	1	0.22	Fully developed	5	0.2
Pressure regulator	1	1.20	Fully developed	5	1.3
Non return valves	4	0.08	Fully developed	5	0.3
Propellant filters	4	0.15	Fully developed	5	0.6
FDVV	7	0.05	Fully developed	5	0.3
Pipework	1	1.00	To be developed	20	1.2
Orifices	2	0.05	Fully developed	5	0.1
Brackets / fasteners	1	0.50	To be developed	20	0.6
Click on button below to insert new unit					
	18	**65.6**		**4.7**	**68.8**

Table 9-4: Bi-Propellant Baseline design components list

9.4.1 Thruster



Property	Value
Oxidiser	N2O4, MON-1, MON-3
Fuel	MMH
Nom. Thrust (Range min/max)	10 (6 - 12.5) N
Nom. Specific impulse (Range min/max)	291 (270 - 298) s
Nom. Mixture Ratio (Range min/max)	1.65 (1.20 - 2.10)
Nom. Inlet Pressure (Range min/max)	? (10 - 23) bar
Nom. Mass flow (Range min/max)	3.5 (2.3 - 4.3) g/s
Nozzle Area Ratio	150
Mass (Thruster with Valve)	0.650 kg

Figure 9-2: Airbus Defence and Space S 10-26 thruster

As a baseline thruster, the Airbus Defence and Space S 10-26 thruster has been used. This thruster has a great heritage and was for example also used for the Rosetta mission where it was also tested for long duration burns. An image and the major technical specifications are given in Figure 9-2. This component is completely ITAR free (European COTS component requirement) and is equipped with dual valves which consume 8 W when opened and 16 W when opening (peak).

9.4.2 Propellant Tank

Based on the final iteration with the latest system mass obtained during the AIM-3P study 4 identical propellant tanks were chosen (2 fuel tanks, 2 oxidizer tanks). Due to the increased spacecraft mass with respect to the AIM-3 study, different tanks had to be selected to accommodate for the additional propellant. The best compromise was using an Airbus Defence and Space tank which has been qualified for the use with hydrazine. For the application within AIM-3P this tank would have to be delta-qualified for MON and MMH, which from the material compatibility standpoint should not be an issue. The different densities of the propellant compared to hydrazine will probably require a mechanical delta qualification as well as a propellant management device delta qualification. The fact that this component is European was considered to be more important than choosing a US tank from ATK, which had also to be reactivated.

Therefore, it was decided to baseline the Airbus Defence and Space tank OTS 31/0 with a dry mass of 6.4 kg each, a usable volume of 78 litre each and a diameter of 0.586 m. The spherical tank is equipped with polar mounted fixations and is made from light weight titanium Ti6Al4V. The MEOP of the tank is 2.76 MPa and the burst pressure is 5.52 MPa. Given the final propellant mass of the AIM-3P spacecraft the filling ratio of the tank results to 93%.

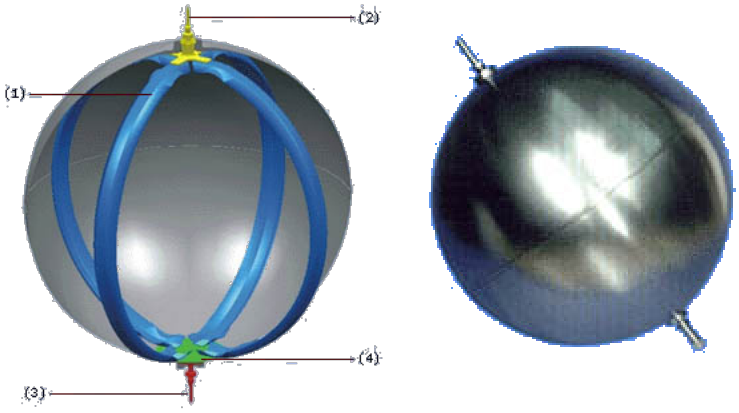


Figure 9-3: Airbus Defence and Space OST 31/0 propellant tank

9.4.3 Pressurant Tank

As a pressurant tank, the MTA35 with a volume of 35.5 l manufactured by MT Aerospace in Germany has been chosen. Its mass is 8.1 kg, the diameter 0.43 m, the MEOP 27.5 MPa and its burst pressure 55 MPa. Its liner is made from steel and it is wrapped with a fibre / epoxy layer. The filling ratio of the tank is currently at 99.7 %. Any further modification such as slightly larger tanks will lead to the selection of also a larger pressurant tank. The next larger one would be a Composites Aquitaine tank with 51 l volume and a mass of 11.2 kg (i.e. a mass penalty of 3.1 kg with respect to baseline).

9.5 Options

9.5.1 Dual-Mode PSS Option

As a backup design option, the dual mode option (Table 9-5) is proposed. This option can provide better performance because a thruster with a very high I_{sp}=307 s can be used (MOOG DST-11). This thruster uses hydrazine as fuel and MON-3 as oxidizer and is planned in a 4+4 redundant configuration. Therefore one can use after the completion of the main burns and main thruster isolation only the hydrazine thrusters for the AOCS. In this case the 4.5 N thrusters of MOOG have been proposed (see Table 9-1 and Table 9-6). The dry mass of this PSS is lower (61.5 kg) to the one of the baseline (68.8 kg) and the propellant mass is with 289 kg comparable (see Table 9-5). A disadvantage of this configuration is that the power consumption of the thrusters is higher than for the baseline. No schematic was drawn for this configuration but it is related to the one of the baseline in Figure 9-1. The components list of the dual-mode option is given in Table 9-6.

PSS	*Dry Mass [kg]*	*Propellant [kg]*	*Total [kg]*	*Mass penalty wrt baseline [kg]*	*Power (operation long burn) [W]*
DM 16x4.5N + 8x22N	61.5	289	350.5	**-4.3**	80 (4x20)

Table 9-5: Dual-Mode backup design option

Unit Name	Quantity	Mass per quantity excl.	Maturity Level	Margin	Total Mass incl. margin
Pressurant Tank	1	8.10	Fully developed	5	8.5
Pressurant	1	1.18	Fully developed	5	1.2
Hydrazine tank	2	6.40	Fully developed	5	13.4
MON tank	1	9.07	Fully developed	5	9.5
Propellant residuals	1	5.76	Fully developed	0	5.8
Thrustes (Low)	16	0.49	Fully developed	5	8.2
Latch valves	2	0.60	Fully developed	5	1.3
Thrusters (22N)	8	0.77	Fully developed	5	6.5
Liquid pyro valves	10	0.30	Fully developed	5	3.1
LP transducer	4	0.22	Fully developed	5	0.9
Propellant filters	2	0.15	Fully developed	5	0.3
FDVV	5	0.05	Fully developed	5	0.3
Pipework	1	1.00	To be developed	20	1.2
Orifices	2	0.05	Fully developed	5	0.1
Brackets / fasteners	1	0.50	To be developed	20	0.6
Gas pyro valves	3	0.15	Fully developed	5	0.5
Pressurant filters	1	0.08	Fully developed	5	0.1
Click on button below to insert new unit					
	17	**58.6**		**4.7**	**61.5**

Table 9-6: Dual-Mode backup design option

9.5.2 Lower Redundancy Mass Savings

As per requirement of the mission, the PSS should be designed with as little dry mass as possible. As per request of the customer, it was assessed how much dry mass saving is possible if the failure tolerance is reduced from 1FT to zero FT. The estimation has been done based on the baseline PSS.

- Zero FT only in valves; no thruster isolation after main burns: 2.25 kg dry mass savings
- Zero FT additionally in the thruster (not recommended since one thruster failure would violate the force free / torque free movement requirement): 10.05 kg dry mass savings.

10 GNC

10.1 Requirements and Design Drivers

The system requirements translate into a set of GNC subsystem requirements, notably:

Req. ID	STATEMENT
GNC-010	The GNC shall use the VIS payload camera as sensor.
GNC-020	The guidance shall be able to load a trajectory to approach the secondary body such as to deploy a micro lander (MASCOT-2) and to return to an observation position.
GNC-030	The guidance shall be able to load a TBD trajectory such as to deploy the COPINS and to return to an observation position.
GNC-040	The FDIR system shall avoid collisions with both asteroid bodies during an approach to drop guest payloads.
GNC-050	The AOCS shall guarantee S/C pointing performance at OPTEL's interface of: • Absolute Pointing Error: 3.36E-04 radians (70.0 arc sec) • Absolute Pointing Knowledge: 5.00E-05radians (10.3 arc sec) • Relative Pointing Error over 1s: 3.00E-06 radians (0.62 arc sec)
GNC-060	The AOCS shall create an attitude vector update with a frequency of 10 Hz, and perform control with a frequency of 1 Hz.
GNC-070	The AOCS shall guarantee S/C pointing performance at the payload's interface of 0.53 degree.
SYS-080	Only technologies at a minimum of TRL 5 by the end of phase B1 shall be used.
SYS-090	The spacecraft total launched wet mass shall not exceed 850 kg (TBC) including all maturity margins and launch adapter.
SYS-AOC-010	The spacecraft shall be 3-axis stabilised.
SYS-PL-030	The payload shall be able to obtain imagery of the secondary asteroid with a resolution better than 1 m.

Table 10-1: GNC subsystem's requirements

10.2 Assumptions and Trade-Offs

10.2.1 Assumptions

The following assumptions were taken in the design and trade-off of the GNC subsystem:

- The spacecraft's wet mass lies between 800 kg and 850 kg
- The solar panels are fixed and their normal orientation is perpendicular to the orbit plane
- The antenna is a fixed antenna
- There is no on-board autonomy for navigation, the ground operations team determines the state vector of the spacecraft from ground. The turn-around time is considered to be 8 hours (Marco Polo-R, Rosetta heritage).

10.2.2 Trade-Offs

Several trade-off must be considered in the design of the GNC subsystem:

- The sensor used for the detection of the asteroid during the approach to the system
- The choice of the attitude sensors and of the actuators due to payload, characterisation and manoeuvre requirements
- The guidance strategy for release of secondary payloads.

10.2.2.1 Approach and detection

Upon arrival at Didymos the asteroid system must be detected in advance to perform several manoeuvres in order to insert the spacecraft in a neighbouring orbit. This can either be done with the star tracker by detecting Didymos against a known background star catalogue, or if the camera has a better resolution than the star tracker, to perform the same operation using the camera as the sensor. The choice will depend on how many days/weeks prior to insertion the asteroid must be inserted. For this it will be necessary to put the sensor in such a position that it is possible to search for the system with the sensor (i.e. camera/star tracker) and observe Didymos while the solar panels are still illuminated by the Sun, and the antenna is pointing towards Earth. Given that the navigation camera will have better characteristics than the star trackers, it is proposed to use the navigation camera with a star tracker as a back up.

10.2.2.2 Navigation and guidance strategies

The baseline characterisation orbit for AIM3-P is a heliocentric (co-flying) orbit similar to that of Didymos, but with a slightly smaller semi-major axis such that the spacecraft is slightly closer to the Sun to observe the illuminated side of Didymos. This orbit will be perturbed by the solar radiation pressure and the gravitational force of Didymos. Orbit determination can be done through the Deep Space Network at arrival, obtaining an accuracy in position better than 100 meters. Once inserted in orbit, the navigation camera can image the asteroid against a star background to characterise the precise size and shape of the asteroid (sizing is performed by comparison of angles with known stars). By determining the size of the asteroid it is also possible to determine the distance to the asteroid, and therefore perform relative navigation with respect to it. This is done by sending the images down to the ground control that will post-process them, determine the relative state vector and upload it to the spacecraft with the next transmission. The precision is then expected to improve up to 10 meters accuracy. There are restrictions in the amount of orbital manoeuvre corrections possible from the ground segment (approximately 2-3 every week). This is guaranteed in a heliocentric orbit and is compatible with the time that is needed for state vector determination.

In order to obtain better navigation accuracy for the ground segment and asteroid geometric reconstruction it is desirable to take images with an illumination angle on the surface between 30° and 60°. The chosen illumination is 45 degrees, therefore placing the spacecraft at this angle either behind or in front of the binary. The heliocentric (co-flying) orbit is thus chosen to be able to observe the primary under an angle of 45°. This combined with the chosen distance of 35 km and the relative position can be approximated in Figure 10-1.

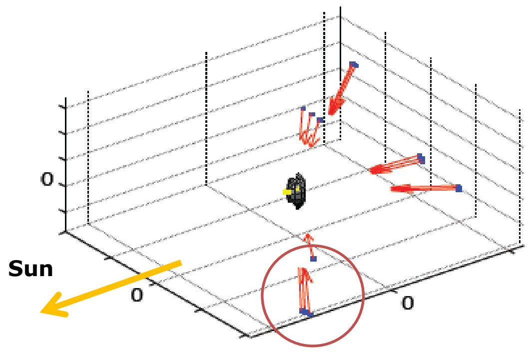


Figure 10-1: Observation direction of spacecraft toward asteroid from 35 km distance and 45 degrees relative angle (circled in red)

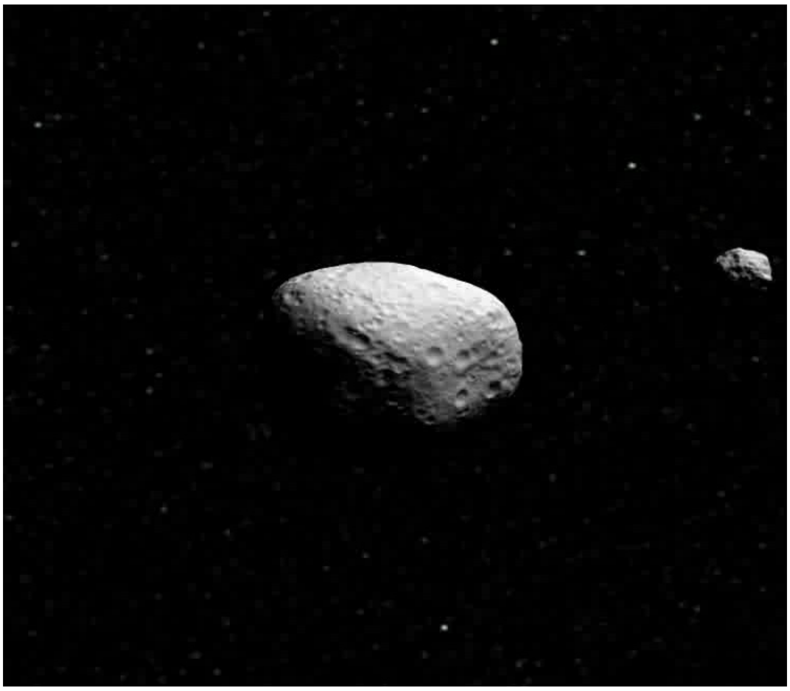


Figure 10-2: Simulated imaging of binary system by camera from 35 km distance and 45 degrees relative angle

A distinct disadvantage in this observation position is the line of sight on the high and lower latitudes of the asteroid. The resolution in these parts will be lower than in the equatorial bands of the asteroid. However this effect may be mitigated by the angular tilt of the rotation motion of the poles: the larger this angle is, the more one of the two polar regions will come closer to the orbital plane and the better they will be imaged. However this also means that the other polar region will be permanently in shadow and therefore (partly) unobservable. Alternatively, it would be possible to perform a force motion to bring the spacecraft out of its orbital plane and temporarily place it in a non-Keplerian orbit to have a better view of the (illuminated) pole.

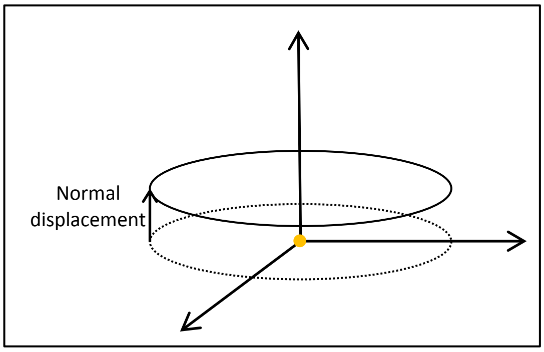


Figure 10-3: Non-Keplerian orbit

10.2.2.3 Actuator system

The actuator system is used for these types of manoeuvres:

- Repointing manoeuvres
- Orbital manoeuvres
- Station keeping manoeuvres.
- OPTEL communications manoeuvres

Due to the mission's duration an offset arises between solar panel-Sun vector and antenna-Earth vector, and can increase to be up to 80 degrees (see mission analysis section). Therefore, if the Earth communication's antenna is fixed, it may be necessary to repoint the spacecraft to send data to Earth (see communications' section). To perform this attitude change use can be made of the reaction control wheels or, alternatively, the thrusters. Using the reaction control wheels would take around 15 minutes to turn the spacecraft, in such a rate that there is no passivation necessary. Alternatively the thrusters can be used, but tranquilisation of solar panels, antenna and propellant would be needed afterwards, which would double the time of the manoeuvre. It is therefore preferable to use the reaction control wheels for repointing manoeuvres.

During the phase of OPTEL experiment (Mode 9) the reaction control wheels are used for the stabilisation of the spacecraft. The relative performance error requirements of the OPTEL experiment are shown in Figure 1-4. They are derived from previous Earth observation missions using reaction wheels and passive dampers as used on BepiColombo (thus the RPE also covers harmonic excitations due to wheels). The chosen wheels from Rockwell Collins (Teldix) have a TRL 9 and have flown in other interplanetary missions requiring accuracy.

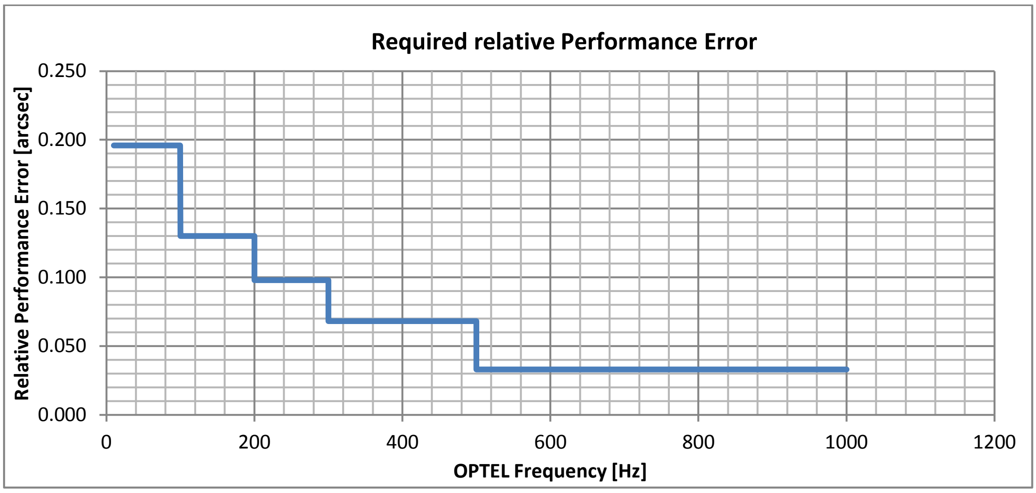


Figure 10-4: Relative performance error requirements for OPTEL

Station keeping for orbit maintenance is done 2-3 times per week to reposition the spacecraft in its characterisation point. The 10 N thrusters of the propulsion system can provide the necessary thrust for these manoeuvres, since the magnitude is of 2-3 cm/s. Orbital manoeuvres are in the order of a few centimetres per second. These manoeuvres can be performed with the 10 N thrusters as well.

10.2.2.4MASCOT-2 deployment

In AIM-3P, the MASCOT-2 lander is to be deployed on the secondary body, with the requirement that the impact velocity on the surface be half the escape velocity. As can be seen in Table 10-2, the escape velocity is very small. Initially a hyperbolic fly-by was chosen to ensure that if at any point control of the spacecraft is lost by ground or an anomaly is detected, the spacecraft should not be on a collision course with either the primary or the secondary bodies. However, if the spacecraft is to perform such a fly-by to release MASCOT-2, the velocities will be in the order of a few meters per second, exceeding this maximum impact velocity by an order of magnitude. Two solutions can be considered:

- The spring release mechanism will store energy to be released upon MASCOT-2 ejection such as to impart a Delta-V in the chosen direction. This velocity will counteract the relative velocity of the spacecraft w.r.t. the secondary body such as to obtain a relative velocity of a few centimetres per second. A drawback of this method is that the energy release of the springs, which is the same concept as CubeSat releases, can only be planned accurately up to a few cm/s. Also, it may not be possible to correct enough velocity with this system, depending on the velocity of the fly-by.
- The second solution is to temporarily insert the spacecraft in the secondary's orbit or into a proximity orbit that is stable long enough to deploy MASCOT (1 or 2 orbits). The spacecraft is inserted in front or behind the secondary body in the same orbit such as to maintain a relative position around the asteroid for a limited amount of time. When the illumination conditions are correct for the deployment, MASCOT is released by the spring-release mechanism towards the surface of the secondary. In this orbit there are two attitude configurations

possible: inertial and co-rotating. The inertial attitude is equivalent to that in the characterisation point (solar arrays towards the Sun) has the advantage that communications and power remain unaffected. It however creates three challenges: to only be able to partially use the navigation camera to determine the relative position with respect to the two bodies (only several minutes per revolution), to obtain a more complex spacecraft configuration, to need good timing for the MASCOT-2 release. The co-rotating attitude would maintain one side of the spacecraft pointing to the secondary body, limiting the communication's time and the capacity to recharge the batteries. Additionally, it would entail a relatively high attitude rotation rate of the spacecraft, possibly affecting the stability due to propellant sloshing.

• Parameter	Primary asteroid	Secondary asteroid	Unit
Radius	400	75	m
Volume	268082573.1	1767145.9	m^3
Density	1.7	1.7	g/cm^3
Density	1700	1700	kg/m^3
Mass	455740374281	3004147975	kg
GM	30.42	0.200492029	m^3/s^2
Escape velocity at the surface	0.390	0.0731	m/s
Half escape velocity at the surface	0.195	0.0366	m/s
Sphere of influence	8610	148	m

Table 10-2 : Assmed asteroid properties

Navigation and guidance directed from ground can be performed in the same manner as in the Rosetta mission, through commands sent from ground to the spacecraft. The sequence of approach to the asteroid is pre-programmed and commands are executed by time tags.

Alternatively, a third solution may be envisaged to drop the payload(s): a close approach to the secondary body where the spacecraft will approach the surface, perform a hover, release the payload and perform an escape manoeuvre. This is similar to the Marco Polo-R touch-and-go scenario and requires relative navigation and on-board autonomy due to the delay in communication, making it impossible to give real-time commands. This will require the same hardware configuration and GNC design as the Marco Polo-R mission, which has reached TRL5.

10.3 Baseline Design

10.3.1 Mission Phases

10.3.1.1 Arrival

This sensor is shared with the payload optical instrument. Therefore, in order to characterise the system (technology demonstration) and to calculate the spacecraft's position w.r.t the system (navigation), it must be able to:

- Image the full binary system in one image during observation
- With a resolution of at least 1 meter.

It is also important to note that thanks to post-processing – which requires a specific adaptation to the project – the accuracy can be increased up to twice the resolution.

The design of the camera is based on the AMIE camera developed for SMART-1. From the first iteration of the AIM-3 mission a camera optics and sensor were assumed to be modified to comprise of a 2048 x 2048 pixel sensor. This achieves a resolution close to the requirement (1.6 m/px) at 35km distance, where the whole system can be viewed in one window. GNC rad-hard sensors are sized up to 1024x1024. However, a greater size can be achieved by stacking sensor arrays, accepting development risks due to the project timeframe. The field of view has been modified to be 10.3 degrees.

Parameter	Value	Unit
Diameter (optics)	20 (TBC)	Cm
Focal lengths (optics)	154.9 (TBC)	mm
Quantum efficiency (CCD)	0.1	-
Wavelength efficiency (CCD)	0.7	-
Dark current (CCD)	0.02	electron/s
Readout noise	11.3	Electron
Pixel size	14	µm

Table 10-3 : Selected camera

Camera		Resolution requirement	Characterisation		
Pixel array	FoV	Total plane size per direction	Distance	Resolution	Inside Sol?
[px]	[deg]	[m]	[km]	[m/px]	[-]
2048	10.3	6308	35	1.611	FALSE

Table 10-4: Camera sensor performance

When arriving at the binary system, detection of the primary can be achieved from long distances in the same fashion as performed for Rosetta: the camera points to the area where the system is expected to be, and a search against the star catalogue is performed to find the asteroid. Once the asteroid is found, the navigation camera follows it in its line of sight.

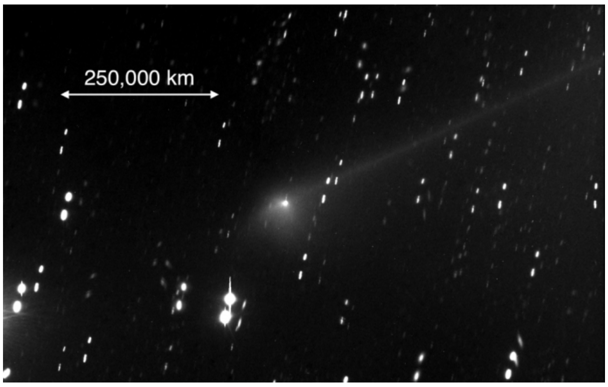


Figure 10-5: Detection and tracking of the Rosetta comet against the star background [Source: ESA]

Depending on the albedo, the asteroid will be detected earlier or later. Usually, it is possible to detect objects when their observed magnitude is less than 12. At this point, if the signal-to-noise ratio is above 4, and the object can be detected in the detector. Depending on the albedo of the asteroid the detection distance may be different, but in all cases the detection distance is close to 2000000 km or larger, allowing several weeks before insertion into the co-flying position. During approach, the asteroid becomes larger than 1 pixel at a distance of 8850 km. From that point on, the brightness received by the detector increases significantly and it will be necessary to adjust the light signal to avoid saturation of the detector. This can be done in the same fashion as Rosetta: in front of the optical lens a circle with several filters is placed. At predetermined distances the filter is (mechanically) rotated to allow for correct contrast.

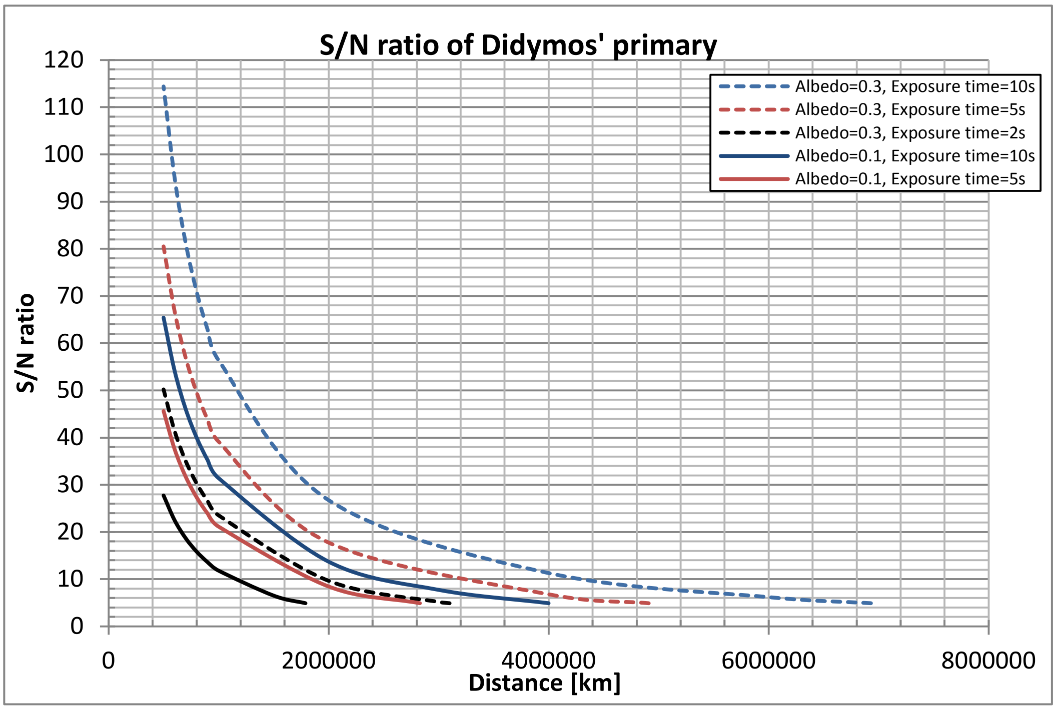


Figure 10-6: S/N ratio of Didymos' primary

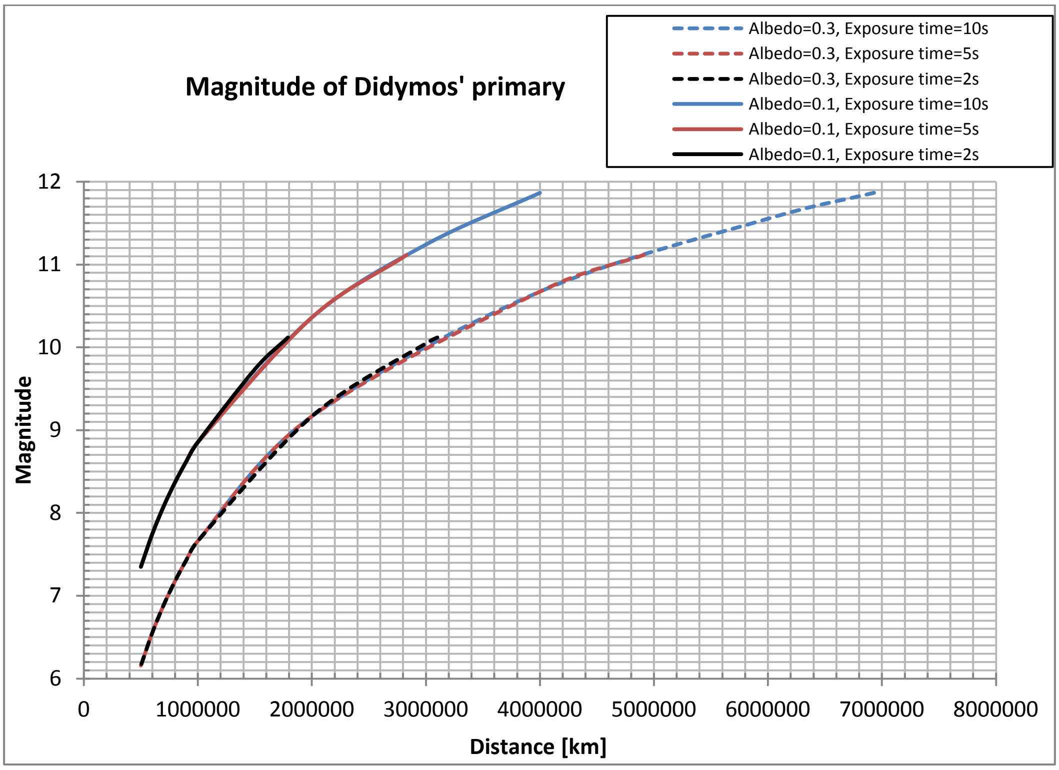


Figure 10-7: Magnitude of Didymos' primary

10.3.1.2 Characterisation phase

During this phase the binary system is observed by the payloads: the VIS (shared with GNC) and the Thermal IR Imager. It is therefore important to have a high pointing accuracy and good stability during this phase. The characterisation phase lasts for several months (approx. 3 months). The heliocentric orbit is chosen to be able to observe the primary under an angle of 45°. This combined with the chosen distance between 10 - 35 km and the relative position can be approximated in Figure 10-1.

The navigation will be controlled by the ground system (as per system request) and will thus require remote measurements for position determination. The shared NAC camera, and an IMU on board will provide these measurements for orbit determination, together with ground segment measurements (either ΔDOR or ranging).

10.3.1.3 Impact observation phase

AIM should observe the results of the DART impactor from 100 km distance. The GNC/AOCS strategy at this point is the same as in the characterisation phase.

10.3.2 FDIR

The FDIR of the spacecraft relies on an autonomous navigation determination and automatic triggering of the FDIR mode.

During the characterisation phase FDIR should be triggered either after control failure (e.g. failure of a reaction wheel thus temporarily losing control of the attitude), or a too large state vector error, e.g. coming out of the observation point's limits or getting too close to the asteroid. If the asteroid exits the field of view of the camera (either by rotation or translation) or occupies too large a part of it (i.e. the spacecraft is too close) during observation, the FDIR should be triggered to retreat from the binary to a farther, safer position.

If during the OPTEL experiment the attitude rate becomes too high, the FDIR should trigger the Safe Mode to analyse the loss of control.

A fly-by for release of payloads will be time-tagged, and therefore FDIR should rely on the comparison of the computer navigation on-board versus the guidance uploaded for the execution of the manoeuvres. In case of off-nominal conditions a collision avoidance manoeuvre is activated.

10.3.2.1 Safe hold point

Should an anomaly be detected during one of the other mission phases, it is important to retreat to an orbit that is far away enough from the asteroids such that if the spacecraft is left unattended for several days (e.g. problem in the communications system, turning off / resetting anomalous equipment, etc.) there is no risk of collision with either the primary or the secondary.

Given that the observation point satisfies these conditions, it is chosen as the point safe hold point to go to in case FDIR is triggered.

10.3.3 GNC Modes

The following GNC modes will be used in the mission. Included are also the sensors and actuators used in each mode.

Number	Mode name	Active AOCS/GNC hardware	Description
1	Launch Mode	All standby	All systems off.
2	Safe Mode	Sun sensor, IMU, Reaction wheels	In this mode GNC/AOCS orients the solar arrays to point towards the Sun, and tries to reacquire a normal pointing mode to allow for Earth communications and go back to operational modes.
3	Active Drift Receive	Star Tracker, IMU, Reaction wheels	Nominal attitude control with antenna pointing to ground, spacecraft drifting.
4	Active Drift transmit	Star Tracker, IMU, Reaction wheels	Nominal attitude control with antenna pointing to ground, spacecraft drifting.
5	Manoeuvre	Star Tracker, IMU, Camera, Reaction wheels	DSM, Orbit insertion, attitude corrections
6	Collision Avoidance	Star Tracker, IMU, Camera, Reaction wheels	GNC performs an orbit manoeuvre as commanded by FDIR to a pre-defined safe hold-point
7	Measurement mode RX	Star Tracker, IMU, Camera, Reaction wheels	Nominal attitude control with antenna pointing to ground and camera pointing to asteroid, spacecraft drifting.

Number	Mode name	Active AOCS/GNC hardware	Description
8	Measurement Mode Tx	Star Tracker, IMU, Camera, Reaction wheels	Nominal attitude control with antenna pointing to ground and camera pointing to asteroid, spacecraft drifting.
9	OPTEL experiment	Star Tracker, IMU, Camera, Reaction wheels	High accuracy attitude control at high rate, OPTEL experiment pointing to ground, no spacecraft manoeuvres allowed, AOCS mode restricts attitude correction sizing.

Table 10-5: GNC/AOCS modes

10.4 List of Equipment

10.4.1 Sensors

For this mission three sensors are needed:

- Navigation camera, shared with the payload. This camera must have the specifications as determined by the trade-off (see Section 10.2.2.1)
- European IMU, with a redundant unit. This unit is being developed in Europe and will be ready by Phase B/C of this missionTRL 4 is expected towards the end of 2015.
- Star tracker, with redundancy in the optical heads and the electronic unit. This is the main attitude measurement system. The HYDRA star tracker offers high-pointing accuracy and fail-safe redundancy.

Two actuators are needed:

- Reaction control wheels, with a 3 +1 (redundancy) configuration. For a light spacecraft the Teldix 12-75/60 provide enough torque
- Thrusters, in a 8+8 (redundancy) configuration, the 10 N thrusters provide enough thrust for this mission, due to the low thrusts needed in the asteroid environment and the mid-course orbital manoeuvres (selection of hardware is made in Section “propulsion”).

Table 10-6 shows the chosen equipment (NAC not included since it is officially a payload).

Element 1	-			MASS [kg]			
Unit	Unit Name Click on button above to insert new unit	Part of custom subsystem	Quantity	Mass per quantity excl. margin	Maturity Level	Margin	Total Mass incl. margin
1	Star Tracker OH (Hydra)		2	1.250	Fully developed	5	2.6
2	Star Tracker EU (Hydra)		2	1.750	Fully developed	5	3.7
3	Sun Sensor		4	0.215	To be modified	10	0.9
4	IMU		2		To be developed	20	0.0
5	Reaction Wheels (Teldix 12-75/60)		4	4.850	Fully developed	5	20.4
-	Click on button below to insert new unit						
	SUBSYSTEM TOTAL		5	26.3		5.2	27.6

Table 10-6 : AOCS/GNC equipment mass budget

10.5 Options

10.5.1 European IMU

Currently only US solutions are available on the market for IMUs. However a European solution is being developed: “European IMU”. A breadboard should be available by the end of 2014 and the current planning has it ready for missions launching after 2022. This European IMU is developed to have as a minimum the same specification as MIMU, Honeywell’s American IMU. This European IMU is the reference IMU for this mission. However, should delays arise in the development, Honeywell’s MIMU has the same characteristics and could be chosen as a backup.

10.5.2 Altimeter and Image-Processing FPGA

An altimeter can be an important safeguard when approaching the primary for the release of MASCOT-2, since it can be used in real-time, in contrast to ground commands that arrive with minutes of delay, to detect whether the spacecraft is getting too close to the asteroid and therefore avoid a fatal collision with it, acting as a safeguard.

HFR and the Optel-D could also provide altimetry information.

10.6 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
IMU	European IMU	Astrium, TRL 3	N/A	Breadboard ready in 2014 (TRL4-5)

11 POWER

11.1 Requirements and Design Drivers

The Power Subsystem (PSS) shall provide electrical power to the Spacecraft (S/C) throughout the mission phases.

Among these phases, a few can be distinguished as design driving phases: Launch and Early Operations Phase (LEOP) or Safe mode might be the sizing case for the battery, while Active Drift Transmit might be the sizing case of the Solar Arrays (SA).

During the cruising/transfer phase:

- The Solar Array should generate at the furthest distance from the Sun and under the Worst Case (WC) Solar Aspect Angle (SAA) enough power to guarantee the communication with Earth.
- The solar array should be sun pointing. Exception might be for a short duration the case of the correction manoeuvres, the asteroid orbit insertion manoeuvre, or any failure or contingency case that might result in higher than predicted SAA.
- The battery should provide the required energy at LEOP and Safe mode. Additionally, it should be able to guarantee a minimum duration of telecommunication with Earth under contingency cases at the furthest distance from the Sun in case the SA cannot deliver the required power.
- The battery should be cycled and charged according to a plan considering also the Propulsion and GNC Subsystems to optimise the trajectory or altitude control.

During operations close to asteroid:

- For science operations the S/C should be pointing to the asteroid, and for the given WC SAA the SA should be able to generate sufficient power to cover the power demand of the instruments, telecommunications or the Optical Terminal (OPTEL).

The PSS should be Single Point Failure Free (SPFF), but it can be allowed to lose part of the power capability after a failure (i.e SA or battery string lost), though maintaining a 20% system margin on the power budget. The power budget should take into account the following points:

- S/C–Sun distance
- Sun and eclipse durations (if applicable)
- Solar aspect angle
- Pointing accuracy
- Environmental temperature and degradation effects
- Reliability and safety aspects
- Any one failure in the system not counting solar array string and battery cell failure
- Failure detection, isolation and recovery scenarios.

At this point it is important to define as the maximum power the SA can generate the one generated with 0 degrees SAA at 1AU distance from the Sun.

11.2 Assumptions and Trade-Offs

11.2.1 Facts from Simulations

The following plot from simulations shows the distance of the S/C in respect to the Sun, Earth and the asteroid.

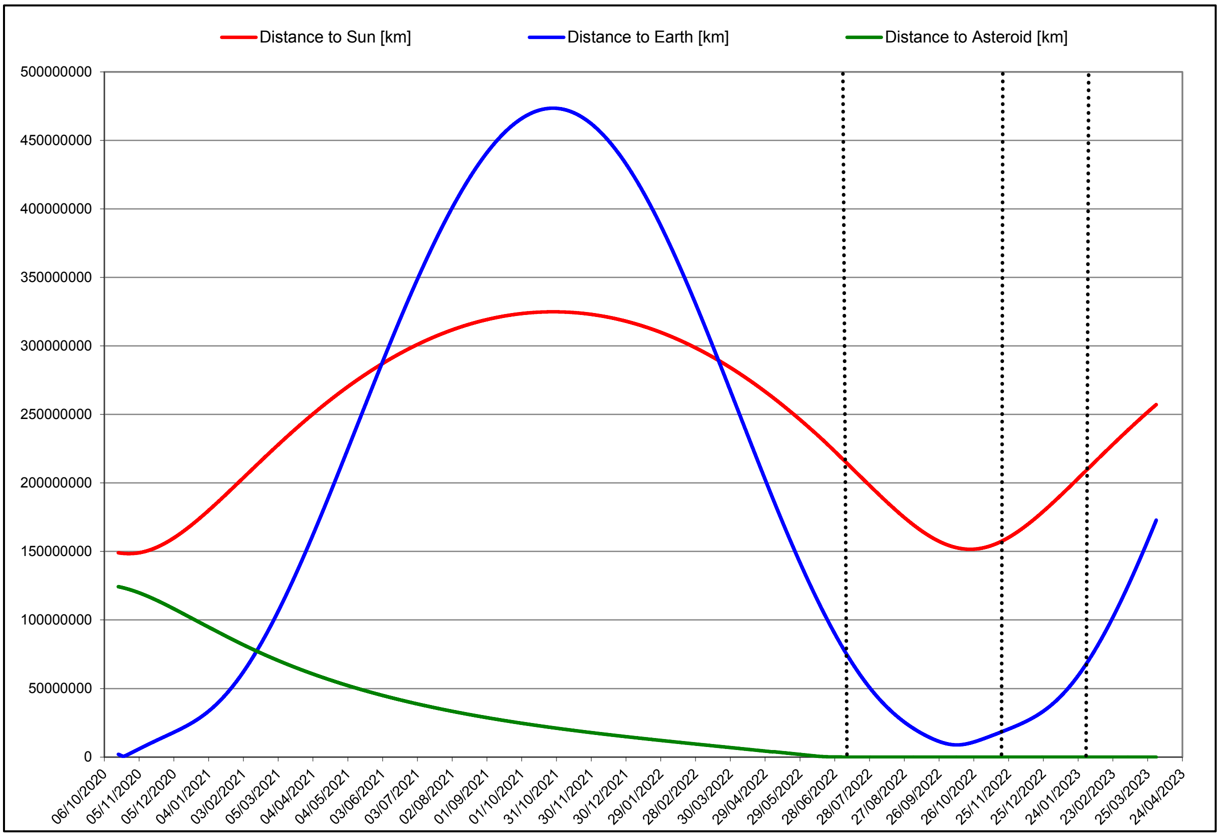


Figure 11-1: S/C – Sun, Earth, Asteroid distances

One can see that the maximum distance from the Sun is about 2.2AU, which results in a reduction of the generated power down to 20% of the one generated at 1AU. This means that the SA would be significantly oversized to cover the power demand at that distance.

Another observation from Figure 11-1 is that when the S/C is at the maximum distance from the Sun, it is at the same time at the maximum distance from Earth. This means that if communication with the Ground Stations is needed at that moment a relative big power demand from the TT/C subsystem will be required.

But the sizing of the SA depends as well from the SAA, which would have variable values through the mission phases. According to the design drivers RD[1] for the simulation of the AIM-3P mission, in order to define the WC SAA the configuration of the S/C should be done in a way to cover the following aspects:

- To achieve optimal illumination conditions on the asteroids' surfaces and instruments resolution for remote sensing during the nominal operations phase, the spacecraft is assumed to be flying in formation with the asteroid system such that the spacecraft lies between 10km and 100km away from the asteroid at a 45° phase angle position ahead of the asteroid on its orbit plane. Then, in order to allow pointing the instruments towards the asteroid while being power optimised (+X towards the Sun), the instruments must be placed on any spacecraft panel but +X (+Z has been chosen) with their boresights pointing towards a direction 45° away from -X (-X to +Z quadrant has been chosen).

- The high gain antenna has been placed on the +X panel with a fixed pointing offset of 25° with respect to +X on the +X to +Z quadrant (to keep, as much as possible, pointing manoeuvres around the minimum inertia axis).
- The fixed solar panels have been placed such that their normal direction is +X in the spacecraft body-fixed frame, and deployed along the ±Y axis (minimum inertia around these axes) following a conventional configuration.

According to the assumptions above, the configuration of the S/C would be as shown below:

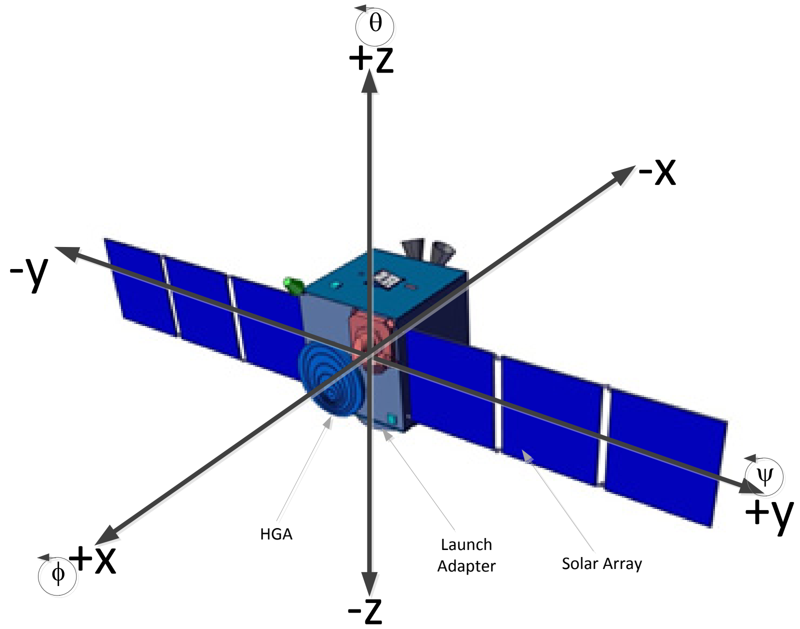


Figure 11-2: S/C configuration

The resulted SAA evolution is shown in the plot below, for the two different cases of interest: Earth communications (blue line) or Asteroid tracking (red line). The bold black lines indicate the part of the mission that the S/C is at a distance of about 2 to 2.2AU from the Sun, while the dashed lines indicate the arrival at the asteroid, end of mission and end of extended mission.

The two smaller black lines indicate the part of the mission that the Earth is behind the Sun, thus making the communications impossible to occur. In this case, the S/C is in receiving mode, not transmitting, thus consuming less power.

One important point that is worth mentioning is that the sizing case for the SA is at that distance of 2.2AU from the Sun, with the WC SAA of 20 degrees. A further optimisation might be possible if the proper strategy would be followed to allow manoeuvres for

Earth pointing (for communications) and then power optimised (for recharging the batteries). But this should result after a trade-off between all the affected subsystems (GNC, TT/C, PPS). This trade off should include assumptions and calculations of the required operations for the manoeuvre, duration of the manoeuvre, propellant used, etc. An example of such a manoeuvre is the period between the two small black lines that the communications are not possible, when the S/C could perform a manoeuvre to be power optimised and not Earth pointing.

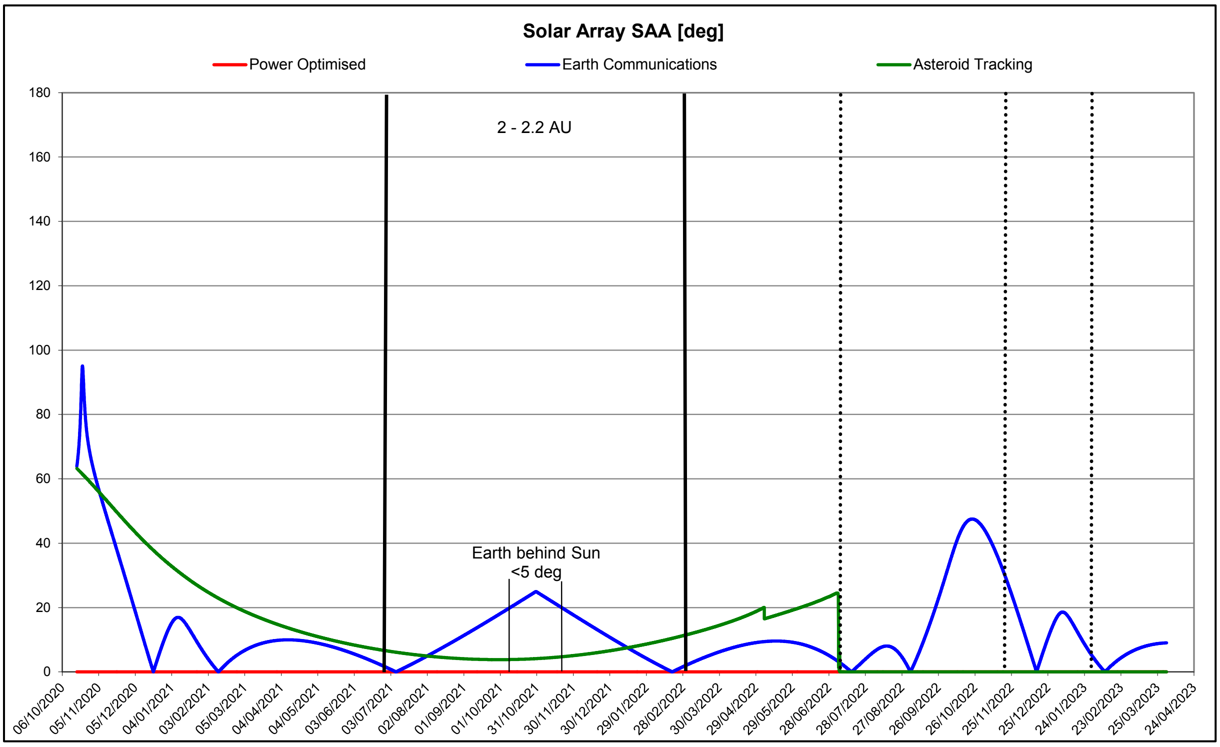


Figure 11-3: SAA Evolution for Constant Earth Pointing (Blue, 25 deg HGA Boresight measured from +X) and Asteroid Pointing (Green)

11.2.2 Solar Array

The trade - off on the Solar Arrays is limited to the configuration regarding how they will be placed on the S/C. There are two options: one is to have them parallel to one axis of the S/C and the S/C trajectory is done with 45° angle, and the other one is to cant the SA wings with 45° angle in respect to one axis of the S/C. For AIM3 the configuration of Figure 11-2 is chosen, because it is more suitable for the accommodation of the equipment external to the S/C.

11.2.3 Battery

An initial trade- off related to the battery is about the selection of the cells. There are two approaches common for European industry:

- Small cells approach: Individual cell overcurrent/overcharge protections, strings in parallel without interconnections, cells matched & battery charge management done at battery level, one additional spare string for tolerance to the failure of any cell.

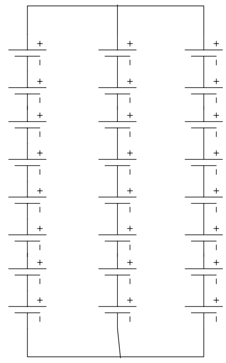


Figure 11-4: Small cells approach

- Large cells approach: Cells in parallel interconnected, individual charge management/balancing capability for each group of cells in parallel, by-pass protection for each group of parallel cells, one additional group of cells in series for tolerance to the failure of any cell.

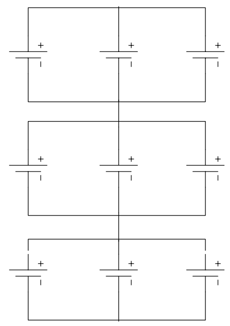


Figure 11-5: Large cells approach

For both choices there are advantages and disadvantages, but the overall conclusion is that they both fit well to the needs of this mission, thus the final choice will be mainly cost driven.

A second trade-off is about the number of cells in series. A light battery can be built with 7 cells in series, leading to a bus voltage range of 17.5V < Vbus < 29.4V, or a heavier battery, but more compatible to the typical 28V bus, with 8 cells in series with a bus voltage range of 20V < Vbus < 33.6V.

Attention should be paid also on the operating temperature of the battery that might be low, thus affecting the qualification, but on the other hand the low number of cycles that the battery would undertake due to the short mission lifetime and the low Total Ionising Dose (TID) can allow high Depth of Discharge (DoD) and small End of Life (EoL) degradation.

11.2.4 Power Budget and Energy Balance

Since during the cruising phase the maximum distance from the Sun is 2.2AU, and the orbit insertion phase will be performed at 1.6AU from the Sun, it is essential to carefully calculate all the consumptions from the subsystems of the S/C. From the moment that the power budget is built, a trade-off can be performed to examine if this power will be delivered only from the SA as usual, or if a reduction of the SA surface can be performed, by oversizing the battery and accept its contribution for limited durations.

11.2.5 PCDU

Two factors that are strongly bonded with the PSS size are the SAA and eclipse durations, and the maximum distance from the Sun. The SAA range that is considered for the AIM3P study is very wide as shown in Figure 11-3, and the maximum distance from the Sun is 2.2AU. This fact gives an advantage on the usage of a MPPT Solar Array Regulator (SAR) instead of the typical Sequential Switching Shunt Regulator (S^3R).

Regarding having a 28V regulated bus, or unregulated battery bus, the unregulated one has the advantage of being simpler, having lower cost and mass due to the absence of Battery Charge and Discharge Regulators (BCRs and BDRs).

11.3 Baseline Design

The power budget as extracted from the workbook of this study can be seen in the table below:

Table 1-2 also shows the power budget showing the phases, distance from the Sun, the WC SAA and the consumptions of the sub systems.

Eclipse Mode :				Thermal	Comms	Propulsion	DHS	GNC	VIS Camera	Thermal Infrared Imager	Monostatic High Frequency Radar	Bistatic Low Frequency Radar (Orbiter)	Bistatic Low Frequency Radar (Lander)	OPTEL (Optical terminal)	MASCOT-2	Cubepod (Pod + 3 cubesats)	Opportunity payloads		Harness (excl. PSS)	TOTAL CONSUMPTION
Initialisation Mode				linked	linked	linked	linked	linked	linked	linked	linked	linked	linked	linked	linked	linked	linked	linked		
			Ppeak	0 W	71 W	#NAME?	47 W	90 W	9 W	14 W	16 W	10 W	4 W	129 W	1 W	1 W	30 W	0 W	#NAME?	#NAME?
Launch Mode	Solar Flux	0 W/m²	Pon	0 W	54 W	17 W	20 W	80 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	3 W	175 W
	(0 if Solar Array not used)		Pstdby	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W
	Remaining Battery Capacity		Duty Cycle	0 %	11 %	6 %	100 %	10 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %		#VALUE!
		100%	Paverage	0 W	6 W	1 W	20 W	8 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	1 W	36 W
Tref	Erreur!	Eclipse Mode NOT Included	Total Wh	#VALUE!	#VALUE!	#VALUE!	#VALUE!	#VALUE!	#VALUE!	#VALUE!	#VALUE!	#VALUE!	#VALUE!	#VALUE!	#VALUE!	#VALUE!	#VALUE!	#VALUE!	#VALUE!	#VALUE!
Safe Mode	Solar Flux	1366 W/m²	Pon	86 W	54 W	17 W	47 W	80 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	6 W	290 W
	(0 if Solar Array not used)		Pstdby	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W
	Remaining Battery Capacity		Duty Cycle	100 %	96 %	6 %	100 %	38 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %		#VALUE!
		80%	Paverage	86 W	52 W	1 W	47 W	30 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	4 W	220 W
Active Drift Receive	Solar Flux	282 W/m²	Pon	72 W	54 W	17 W	47 W	80 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	5 W	276 W
	(0 if Solar Array not used)		Pstdby	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W
	Remaining Battery Capacity		Duty Cycle	100 %	11 %	6 %	100 %	100 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %		#VALUE!
		100%	Paverage	72 W	6 W	1 W	47 W	80 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	4 W	211 W
Active Drift Transmit	Solar Flux	282 W/m²	Pon	64 W	54 W	17 W	47 W	80 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	5 W	268 W
	(0 if Solar Array not used)		Pstdby	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W
	Remaining Battery Capacity		Duty Cycle	100 %	100 %	6 %	100 %	100 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %		#VALUE!
		100%	Paverage	64 W	54 W	1 W	47 W	80 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	5 W	251 W
Manoeuvre	Solar Flux	607 W/m²	Pon	52 W	54 W	39 W	47 W	80 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	5 W	278 W
	(0 if Solar Array not used)		Pstdby	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W
	Remaining Battery Capacity		Duty Cycle	100 %	57 %	100 %	100 %	100 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %		#VALUE!
		100%	Paverage	52 W	31 W	39 W	47 W	80 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	5 W	254 W
Collision Avoidance	Solar Flux	607 W/m²	Pon	52 W	54 W	17 W	47 W	20 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	4 W	194 W
	(0 if Solar Array not used)		Pstdby	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W
	Remaining Battery Capacity		Duty Cycle	100 %	57 %	11 %	100 %	100 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %		#VALUE!
		100%	Paverage	52 W	31 W	2 W	47 W	20 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	3 W	155 W
Measurement Mode Rx	Solar Flux	607 W/m²	Pon	46 W	54 W	17 W	47 W	80 W	9 W	14 W	16 W	10 W	4 W	0 W	1 W	1 W	20 W	0 W	6 W	326 W
	(0 if Solar Array not used)		Pstdby	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W
	Remaining Battery Capacity		Duty Cycle	100 %	11 %	11 %	100 %	100 %	100 %	100 %	100 %	100 %	100 %	0 %	100 %	100 %	100 %	0 %		#VALUE!
		100%	Paverage	46 W	6 W	2 W	47 W	80 W	9 W	14 W	16 W	10 W	4 W	0 W	1 W	1 W	20 W	0 W	5 W	261 W
Measurement Mode Tx	Solar Flux	607 W/m²	Pon	45 W	54 W	17 W	47 W	80 W	9 W	14 W	16 W	10 W	4 W	0 W	1 W	1 W	20 W	0 W	6 W	324 W
	(0 if Solar Array not used)		Pstdby	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W
	Remaining Battery Capacity		Duty Cycle	100 %	50 %	11 %	100 %	100 %	100 %	100 %	100 %	100 %	100 %	0 %	100 %	100 %	100 %	0 %		#VALUE!
		100%	Paverage	45 W	27 W	2 W	47 W	80 W	9 W	14 W	16 W	10 W	4 W	0 W	1 W	1 W	20 W	0 W	6 W	281 W
OPTEL Experiment	Solar Flux	607 W/m²	Pon	44 W	54 W	17 W	47 W	80 W	0 W	0 W	0 W	0 W	0 W	129 W	0 W	0 W	0 W	0 W	7 W	379 W
	(0 if Solar Array not used)		Pstdby	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W
	Remaining Battery Capacity		Duty Cycle	100 %	50 %	11 %	100 %	100 %	0 %	0 %	0 %	0 %	0 %	100 %	0 %	0 %	0 %	0 %		#VALUE!
		100%	Paverage	44 W	27 W	2 W	47 W	80 W	0 W	0 W	0 W	0 W	0 W	129 W	0 W	0 W	0 W	0 W	7 W	336 W
Lander Deployment	Solar Flux	607 W/m²	Pon	52 W	34 W	0 W	47 W	80 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	4 W	218 W
	(0 if Solar Array not used)		Pstdby	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W
	Remaining Battery Capacity		Duty Cycle	100 %	100 %	0 %	100 %	100 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %	0 %		#VALUE!
		100%	Paverage	52 W	34 W	0 W	47 W	80 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	4 W	218 W

Table 11-1: Power budget from workbook

	1	2	3	4	5	6	7	8	9	10
Phase	**launch**	**safe**	**Actrive Drift Receive**	**Actrive Drift Transmit**	**manoeu vre**	**collision avoidan ce**	**Measure ment Rx**	**Measure ment Tx**	**OPTEL**	**lander deploymen t**
Distance to the Sun	2.2 AU	2.2 AU	2.2 AU	2.2 AU	1.6 AU	1.6 AU	1.6 AU	1.6 AU	1.5 AU	1.5 AU
max SAA (degrees)	n/a		20	20	49	49	49	49	49	49

Table 11-2: Solar distance and SAA vs mode summary

The calculated surface for the SA is 5.6m² with 16 cells in series and 98 strings in parallel. The sizing case is the transfer phase, when the S/C is 2.2AU from the Sun and the TWTA is used for communication with Ground Station. At that distance and with 20 degrees WC SAA the SA can generate around 330W, while the power demand is 327W (including transmitting) or 274W (excluding transmitting) as it is shown in the power budget.

The MPPT will act when the SA power is low and is not enough to fully charge the battery and provide the required power for the platform and payloads. The rest of the time the PCDU will be under conductance control and the operating point of the SA will move in a way to provide the correct amount of power needed.

Figure 11-6 shows the I-V characteristic of a solar cell (red color) and the power generated (blue color). It also shows the Maximum Power Point and the operating point of the solar cell at 1AU.

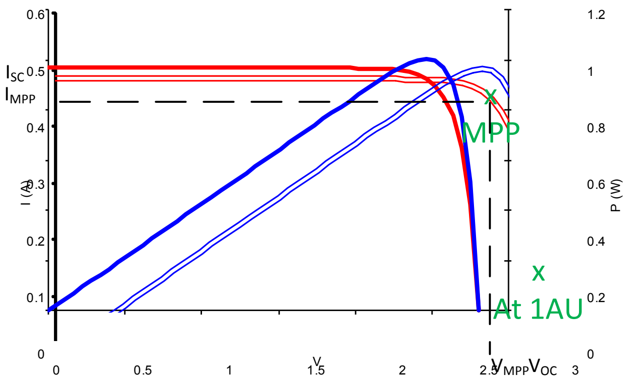


Figure 11-6: SA I-V curve

According to the arrangement of the cells in series (16) and strings in parallel (98), the short circuit current (I_{SC}), and the open circuit voltage (V_{OC}) the Maximum Power Point current and voltage (I_{MPP}, V_{MPP}) would have approximately the following values:

I_{MPP} =98*0.43=42.14A

V_{MPP} = 16*2.5=40V

MPP= 40*42.14=1685.6W → this is the maximum power the SA can generate at 1AU distance from the Sun. But since at that period of the mission the consumption on the bus would not be that high, the MPPT will not act, but the conductance control of the SAR will be in charge, moving the operating point of the SA to deliver lower current. Maybe the V_{MPP} would higher than at 1AU due to the lower temperature, but in any case the product I_{MPP}*V_{MPP} would be around 330W resulting in approximately:

I_{MPP} = 7.5A

V_{MPP} =44V

The battery would be a 8s10p arrangement with ABSL 18650HC cells resulting in an unregulated bus with 20V < Vbus < 33.6V delivering 15Ah. This means that the available energy would be:

80% DoD * 15Ah * 28V = 336Wh

Assuming 287W from the power budget for the safe mode, then:

336Wh/287W = **70 minutes** autonomy for the S/C to come back from safe mode and non-Sun pointing to nominal operation and Sun pointing.

The PCDU will have SAR with MPPT able to deliver up to 500W for an unregulated battery bus with 20V < Vbus < 33.6V. It is worth to mention that since the SA would be Sun pointing during the cruising phase, and as well during the science operations close to the asteroid, the bus voltage would be almost constant at the level of 28V for the vast majority of the mission.

The transmission strategy should be adapted if needed in order to fit with the capabilities of the PCDU in discharging and charging cycles of the battery. It is important to identify that the consumption during the non-transmitting period is as critical as the one during transmitting, because the lower is that consumption, the faster will be the recharge of the battery, thus the lower will be the Depth of Discharge (DoD), thus the longer time will be allowed the transmission.

11.4 List of Equipment

11.4.1 Solar Array

The SA main features are:

- 2 SA wings with solar cells 30% 3J AsGa
- 2 fixed wings of 2.8m² each
- SA mass: 3.5 kg/m² → 19.6kg
- 16 cells in series with 98 strings in parallel
- Worst Case power generation EoL considering one string lost: 1625 Watt at 1AU 0° SAA, 330W at 2.2AU, 20° SAA

11.4.2 Battery

The battery main features are:

- 1 battery with 10 strings and 8 cells in series per string (8s10p)
- 15Ah capacity
- End of Charge voltage: 33.6V

- End of Discharge voltage: 20V
- Mass: 4 kg
- Dimensions: 220mm * 175mm * 115mm

11.4.3 PCDU

The PCDU main features are:

- 28V unregulated bus (20V < Vbus < 33.6V)
- MPPT modules
- Distribution by LCLs
- Mass: 15 kg.
- Dimensions: 350mm * 300mm * 200mm

AIM3P Spacecraft		MASS [kg]			
Unit Name Click on button above to insert new unit	Quantity	Mass per quantity excl. margin	Maturity Level	Margin	Total Mass incl. margin
Solar Array wing	2	9.8	Fully developed	5	20.6
Battery	1	4.0	To be modified	10	4.4
PCDU	1	15.0	Fully developed	5	15.8
Click on button below to insert new unit		0.0	To be developed	20	0.0
SUBSYSTEM TOTAL	**3**	**38.6**		**5.5**	**40.7**

Table 11-3: Mass budget with margins

12 TELECOMMUNICATIONS

12.1 Requirements and Design Drivers

The main requirement for the TT&C subsystem is to provide reliable data links between the S/C and the ground control centre with the optimum frequency, duration and data-rate that meets the overall mission requirements. The driving requirements for the TT&C subsystem are summarised as follows:

SubSystem requirements		
Req. ID	**STATEMENT**	**Parent ID**
COM-010	To support S/C Management during the entire mission: as a minimum1.5Kbps of TM shall be achievable	
COM-020	To provide high rate scientific data TX to Earth to ensure following data to be downloded: ECP 1258.76 Mbit 1st DCP 11261.56 Mbit 2nd DCP 2523.13 Mbit 3rd DCP 3784.69 Mbit Lander phase + reserve: 2700Mbit TBD volume for DART impact monitoring	
COM-030	To support radio-tracking, with DDOR (TBC)	
COM-040	To be coordinated with DART for Spectrum management (both on X-Band) and for Ground cross-support if needed	
COM-050	To embark an Optical Space Communcation Terminal as P/L: • The whole mission must be feasible with the RF system only • Advantage of capacity provided by Optical Communications can be exploited if and when available (e.g. weather conditions, OGS congestion, geometry, stability, troubleshooting…) • Parallel RF communications shall be possible with Optical Comms, at least as part of the validation campaign. Not strictly needed as an operational req.	
COM-060	To provide data relay capability for lander (Mascot2)/secondary satellites (COPINS CubeSat(s)): • Volume transmitted by deployed satellites 1.5 – 20 Gbit (TBC) all over the mission, instantaneous rate towards AIM S/C not more than 1Mbps (TBC) • Functionality: – Multiple nodes (Mascot2+CubeSat(s), AIM as master node – (Coarse) range/range-rate estimation (TBC) – Robustness against unreliable slave notes	

Main design drivers are summarised as follows:

- This is a Deep Space mission (>2Mkm from Earth surface)
  - S-Band not supported for new ESA Deep Space missions
  - Specific X-Band frequency slot assigned therefore Specific product line/development for Deep Space.
- For communication subsystem dimensioning, the logic is as following:
  1) Sizing for delivery of scientific data prior DART impact
  2) Ensure adequate Ground visibility (1.5kbps) of onboard events at asteroid orbit insertion with same design as 1)
  3) Adequate Ground visibility of onboard events at DSM
  4) Ensure that the design is achieving the minimum Ground visibility (1.5kbps) at maximum range with design.
  5) Delivery of DART impact scientific data within 2 (TBC) months
- Simple architecture
- Design to cost & schedule
- Mass & power figures shall be kept as low as possible, without being a first priority.

COM-050 is not considered in this chapter as the optical terminal is a completely independent S/S from HW and functional point of view and traded in its dedicated section and at system level.

12.2 Assumptions

The main assumptions identified are as follows:

- At Maximum Earth-S/C distance (3.2AU), the S/C is in a “quiescent” mode (e.g. no thrusters firing, no P/L activity): a transmission rate of 1.5kbps is sufficient for minimal spacecraft management
- S/C arrival in asteroid vicinity @ 1AU from Earth
- DSM @ 0.25AU from Earth
- No need for (near) real-time two-way communications with Earth during DART impact phase and asteroid tracking (imaging)
- The S/C is able to point to the Earth for communications via HGA with accuracy better than 0.5°
- For ISL, no specific requirements for shock at landing of MASCOT2 are tackled. Standard electronics are assumed
- For ISL, antenna covered by asteroid soil and/or magnetic-dust effects are not considered in link budget computation nor in carrier frequency selection.

12.3 Trade-Offs

The main trade-offs identified are as follows:

12.3.1 Bandwidth Trade-Offs and ConOps

As AIM is a Deep Space mission (Category B) the use of S-Band is not in line with the current ESA plan and at least X-Band shall be considered (8400-8450 MHz downlink, 7145-7190 MHz uplink).

With the assumed mission profile and a preliminary design, the link can be closed as follows.

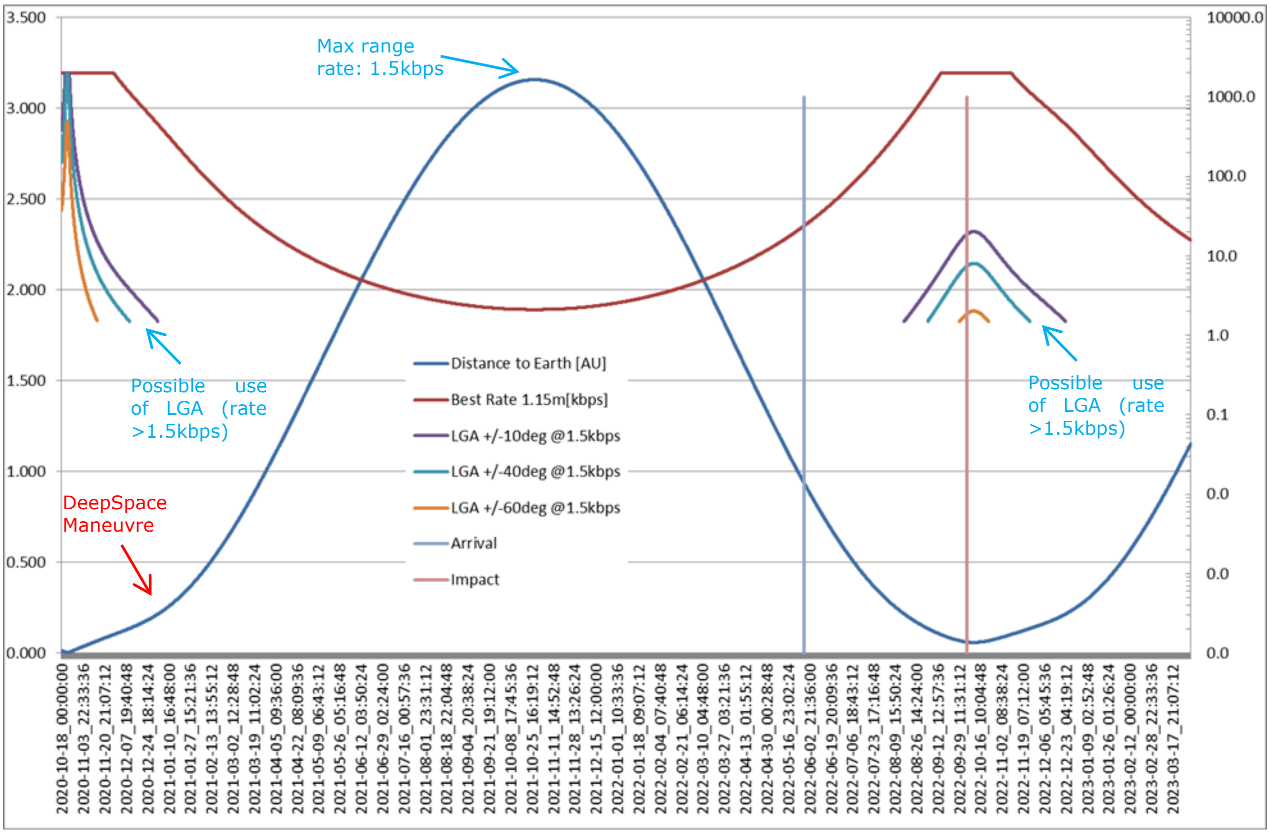


Figure 12-1: Rates over the whole mission profile

In Figure 12-1 is shown the achievable rate via the HGA over the mission (S/C-Earth distance is shown as well for reference). Also shown is the rate achievable with an LGA for different gain w.r.t. the boresight only when above the minimum threshold of 1.5kbps.

Rates are limited to 2Mbps to keep the heritage of existing HW applicable (symbol rate <10Mbps).

From the achievable rates, the bandwidth available on X-Band allocation is deemed sufficient.

Having no radioscience requirements calling for a triple frequency link, there is no need for a higher frequency than X-Band, which is therefore assumed as baseline.

12.3.2 Antenna Trade-Offs

Different aspects of onboard antennas have to be traded. Due to the S/C-Earth range there is the need for a HGA and hence the following aspects need to be evaluated:

1. Antenna network topology: LGA + MGA + HGA vs LGA + HGA
2. HGA steerability
3. HGA diameter (Gain) vs RF power (DC consumption)
4. HGA technology

12.3.2.1 Antenna network topology

Regarding the first point, the main difference between the two options (LGA + MGA + HGA vs LGA + HGA) is the management of safe mode with and without a medium gain antenna: an MGA can guarantee some extra gain w.r.t. an LGA over a wider cone with respect to a HGA, and w.r.t. a HGA is therefore less critical on pointing accuracy/constraints.

At maximum range of 3.2 AU, due to the limited RF power foreseen by the preliminary design, the MGA is not providing enough gain to close the link at the requested rate (COM-010). At short range, the LGA can provide some gain thanks to a coarse pointing towards the boresight direction, therefore the MGA is not strictly needed.

Due to the limited benefit of an MGA against the increase in S/S complexity and mass, an LGA + HGA only solution is selected as baseline.

12.3.2.2 HGA steerability

Concerning the second point, as shown in Figure 12-2 and Figure 12-3, over the whole mission, a very wide range of SSE will be experienced (ranging from 0° to 75°): for a S/C with a fixed directive antenna and also fixed solar panels this leads to some conflicts for parallel communications and power generation.

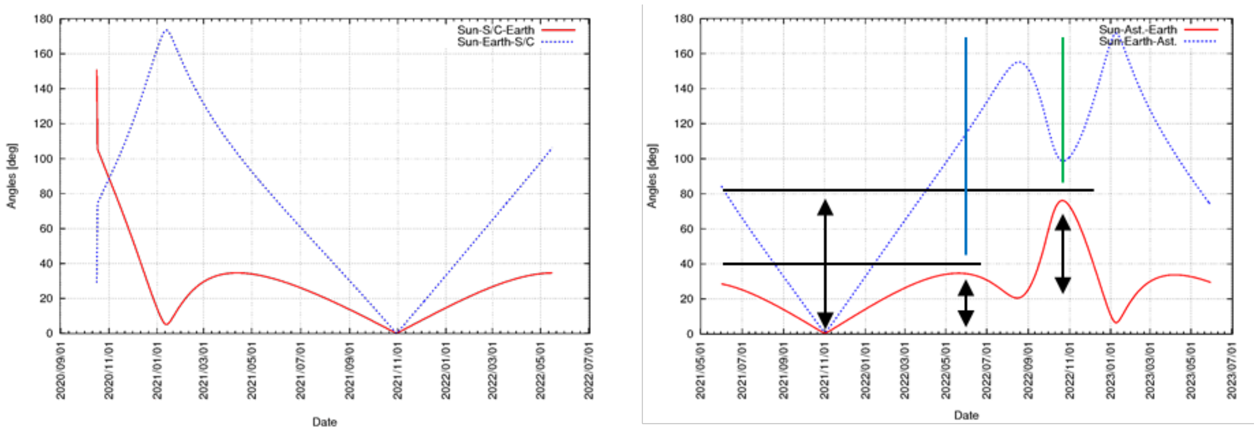


Figure 12-2: SSE & SES angles over the mission

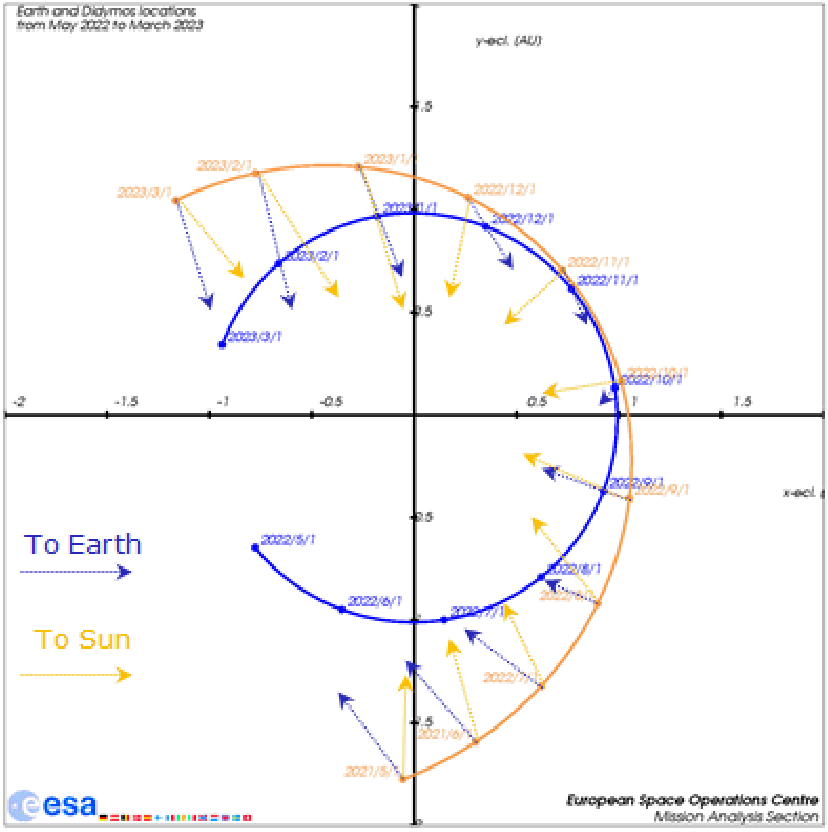


Figure 12-3: Heliocentric visualization of SSE & SES angles

The criticality is in the variation of this angle, more than its magnitude: this is in particular the case during proximity operations (period in between the blue and the green vertical lines on the right of Figure 12-2). On the contrary, a constant SSE value could have been compensated with a tilt between a directive antenna and the S/C body.

Thanks to a tilt between the HGA and the Solar panel of 25° and allowing the upside-down rotation of the satellite (effectively leading to a 25° tilt in the other direction) it is possible to reduce the maximum Sun off-pointing while pointing toward Earth for Comms and keep decent power generation (max Sun-Solar Array angle <49°). This is marginal with the present mission design and is only coupled with power generation-COM needs, not any other specific pointing requirements like detailed observations or Lander deployment (See Power chapter for details).

Thanks to the identified strategy, a steerable HGA is not strictly needed. The selected baseline is a fixed HGA with a tilt angle of 25° with respect to the satellite body (and thus Solar Array).

12.3.2.3 HGA diameter (Gain) vs RF power (DC consumption)

For the 3rd trade-off on the antenna solution, the required EIRP can be decomposed between Antenna gain and RF radiated power as shown in Figure 12-4 for the estimated EIRP needed, with minimum pointing error (0.5deg BPE) at input of the antenna.

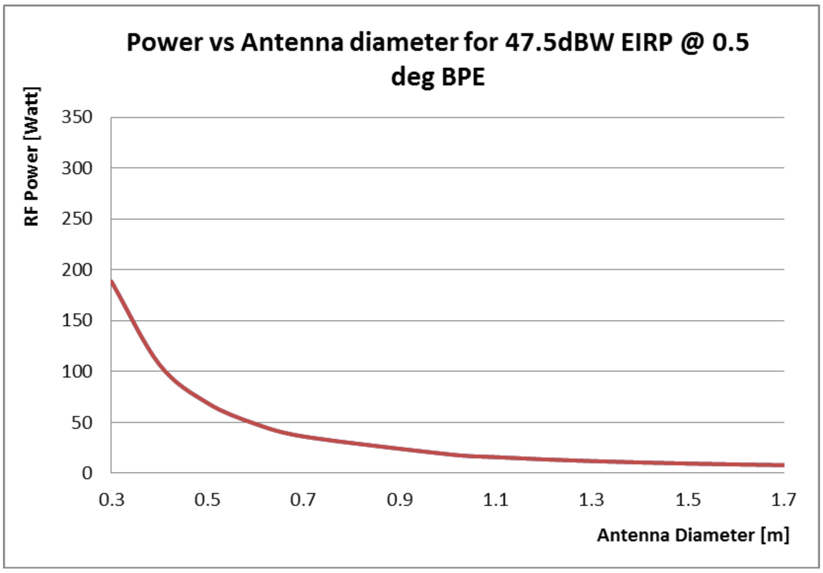


Figure 12-4: RF power vs Antenna diameter, 0.5° BPE

Not considering onboard loss, using a state-of-the-art SSPA (today assumed to be in the range of 15-17W, ongoing/planned development more than 20W), an antenna with a 1m diameter is needed, that increases to 1.6m when considering the estimated 4dB of onboard loss. If a small TWTA with 35W is considered, a 0.7m antenna can be used, to be increased to 1.15m to account for onboard losses between the HPA and the HGA.

Due to the criticality of relying on ongoing development that appears to fall on to the critical path of the AIM-3 schedule, a viable design solution based on today's available products is preferred: a TWTA 35W output and 1.15m HGA is selected as baseline.

Twice the RF power can be assumed possible with TWTA technology if needed, at the price of increased DC power consumption: this can boost data transmission, or reduce the antenna diameter, or increase the robustness over off-pointing, or any combination of these.

12.3.2.4 HGA technology

Classical antenna reflector technology can be used to achieve the requested gain of about 37.5dBi at 0.5deg from boresight.

As an alternative, an advanced but well established solution can be adopted to achieve a lightweight, low profile and at the same time quite easily shaped pattern. Pattern shaping can be of interest to widen the useful antenna cone, making communications more robust over pointing errors or less demanding w.r.t. pointing towards Earth (making easier the parallel operation of P/Ls and Power Generation). Moreover, shaping can be useful to introduce a beam directivity decoupled from the physical accommodation of the antenna, simplifying the accommodation and structural support of the antenna.

The use of Conjugate-matched metasurface antenna is considered and adopted as baseline solution, with TRL 4 already achieved end of 2014 (in Figure 12-5 a picture of an existing EM of a directive X-band antenna is shown). A classic reflector is assumed as backup solution.

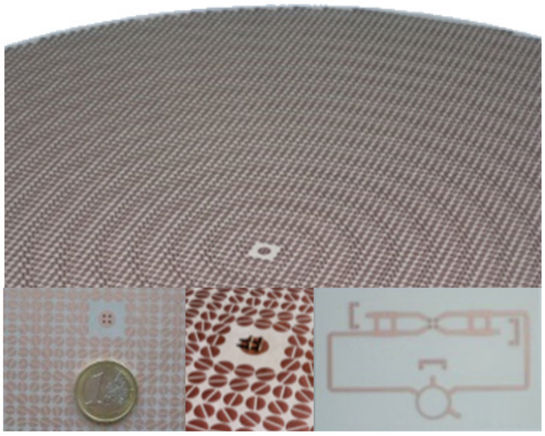


Figure 12-5: Directive metasurface antenna prototypeLander – Earth Data Rely System

12.3.3 Intersatellite link (ISL)

The satellite is required to act as data relay for data collected by the lander (MASCOT2) and other released nanosatellites (COPINS CubeSats) to be forwarded to Earth. Functional and performance requirements are still to be consolidated; therefore the selection has to be reiterated once frozen.

From COM-060, data generated from MASCOT-2 and CubeSats need to be transmitted to the AIM main S/C for retransmission to Earth at a rate of 1Mbps (TBC). Also the inverse communication path (from Earth to MASCOT-2/ CubeSats via AIM S/C) is foreseen to deliver telecommand: this capability need to be further investigated and consolidated. Local communications (MASCOT-2/ CubeSats ←→ AIM S/C) are also possible, if needed, from communication point of view.

It is still to be consolidated if data has to be transmitted during the descent of the lander and if radio-localisation (on the surface only or also over the descent phase) is a requirement. Also for the CubeSats it needs to be determined if any (coarse) radio-localisation function is needed to perform flight dynamics task including collision avoidance (TBC).

Identified alternatives are:

- Proba-3 (Gamalink):
  - Best estimation at present is 1.5kg (including 20% margin) redundant and 0,3kg non-redundant w/o casing (for lander)
  - Preliminary Link Budget can be closed for 1Mbps (4MHz BW), 100km range and 1W RF and hemispherical antennas on both side. TBC
  - No support of Lander localisation (radiotracking)
- FLYCON:
  - Proof of concept for a dual Communication and Navigation system for formation flying based on commercial wireless standard (WiMax, WiFi…). TRL 4 by Semester 2 of 2015
- CubeSat-like:

- Some development in the frame of QB50 or modification of existing Space-Earth links.

A NASA ELECTRA based solution is not considered, even though it is expected to be well performing and covering all the functional requirements. This is due to the large allocations (mass, power, volume) expected, the need for development on the lander side (no ELECTRA terminal exists that fits the allocated resources on the lander) and programmatic aspects (to be procured via NASA).

A Proba-3 derived solution is the assumed baseline due to its capability of providing the link capacity as defined at present. This selection needs to be reiterated due to the early stage of such development and missing frozen set of requirements for the AIM mission.

Hereafter a preliminary list of capabilities provided by the selected baseline:

- Name: GamaLink
- Supplier: TekEver (PT)
- Expected data rate: at least 1Mbps (as HW capability) two-ways, actual rated based on link budgets
- RF TX power: up to 1W (TBC)
- Band: S-Band. This spectrum portion is able to ensure sufficient BW for the required bitrate and is the same as the unit under development. Exact frequency band selection to be performed considering existing products and ITU regulations applicable to a DS mission
- Directivity: omnidirectional on all nodes (including AIM -S/C) - TBC
- Redundancy: redundant on AIM S/C, single string on the COPINS CubeSats, TBD on MASCOT2
- Configuration: Form factor suitable for CubeSat on the COPINS CubeSats
- Mass: <1.5kg on AIM-S/C (fully redundant and shielded), <200g (non redundant, unshielded) for released elements (MASCOT2/COPINS CubeSats)
- Power: 1W RX (single receiver), 4W TX+RX(1W RF, single receiver). All TBC
- Other features:
    - Networking capability (multiple nodes communications, based on CDMA)
    - Coarse range estimation (1-10 m TBC, based on RF base band processing/Kaman filtering TBC) .

12.4 Baseline Design

The selected configuration foresees a quite simple architecture and is composed as follows:

- 2 X-Band miniaturized transponder with DS X-Band extension
- 2 Travelling Wave Tube Amplifiers 35W RF
- 1 Radio Frequency Distribution Network
- 2 Low Gain Antennas
- 1 fixed HGA Matched conjugate metasurface

- 1 Proba-3 derived internally redundant ISL system

Figure 12-6 shows the TT&C related communications subsystem architecture: the RF distribution network is to be considered as preliminary and needs to be refined via consolidated mission design.

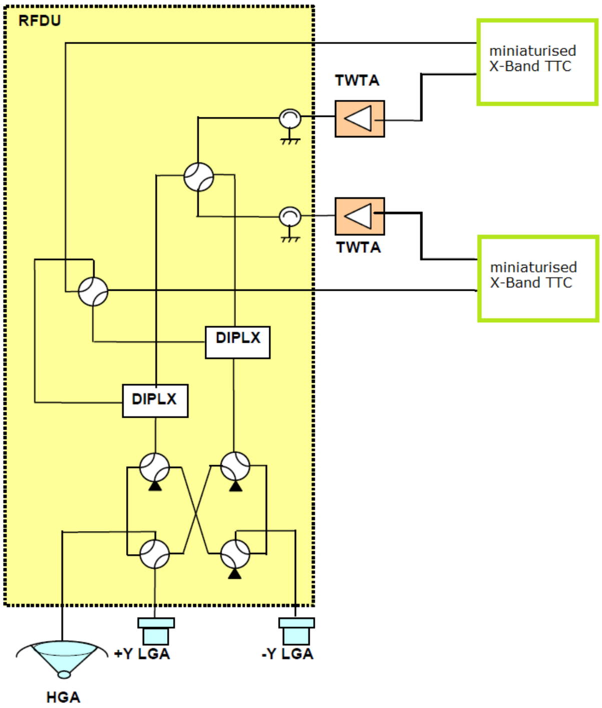


Figure 12-6: Communications subsystem baseline architecture

The ITU Frequency allocation request has to be coordinated with the DART mission to prevent any possible cross-interference issue due to the relative small angular separation of the two spacecraft as seen from Earth at time of asteroid vicinity or any cross-track among the two satellites once both are in proximity of the asteroid.

To best meet the allocations and requirements (including PA) of a small-class mission, a miniaturised transponder, at present under development, is assumed as baseline. Such unit is the miniaturised TTC (MSBT) from Thales Spain. Key features are as follows:

- Combined TX & RX Digital sections → mass/power saving
- Higher level of integration (system on a chip for frequency synthesizer at other blocks) → mass/power saving, faster unit tuning
- Modular (core module S-Band with X-Band extensions)
- Scalable (SSPA output tunable on mission needs): max rate ≥2Mbps
- PWR: 6W in RX mode (HOT redundant), 8W in TX mode (+ HPA)
- Mass: 1kg (non redundant)
- Volume:
    - Core module (S-band): 200x150*200 mm^3

- X-Band ext.: 200x50x200 mm3
- Lifetime: 5 Year
- Ranging: DDOR
- High rate TM on GMSK
- Rad: 40Krad (TBC)
- TRL 4 by late 2014

In Figure 12-7 a block diagram description of the unit where the higher level of integration among TX and RX digital modules is clear: here it is shown the S-band core version, an Up-Down X-Band conversion stage to be add, as preliminarily shown in Figure 12-8.

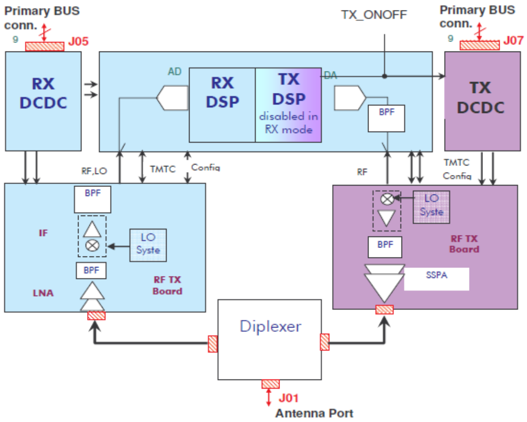


Figure 12-7: Miniaturised SBT from TAS-E

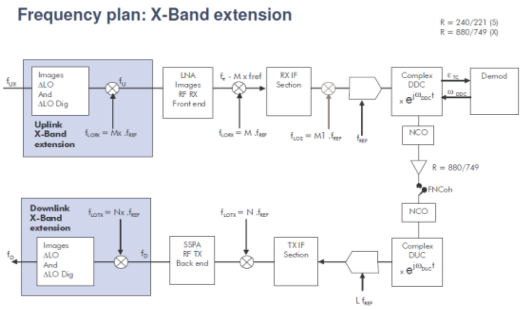


Figure 12-8: Preliminary design for the extended to X-band frequency plan

As backup solution, the standard Deep Space Transponder (DST) from TAS-I is assumed with mass, power and cost penalties but with also pros like higher level of QA.

They both comply with RD[13], RD[14], RD[15].

The selected baseline is able to deliver data volumes as specified in the P/L requirements with transmission times as reported in Table 12-1: of particular interest is the "Total per phase" line; there the data are assumed generated linearly over the phase and then linearly transmitted on the DTE link. For reference, a different scheme is reported where all data is stored onboard and transmitted at the end of each phase: this is seen as impractical for many reasons but shows that some phased postponed transmission can reduce transmission time therefore the energy budget.

S/C Arrival at Asteroid	May 29, 2022																												
Stage	1		2						3				4		5				6		7		8						9
Starting	May 15		Jun 1						Jul 16				Aug 16		Sep 1				Oct 1		Oct 6		Oct 16						Nov 16
VIS (payload use)	Approach phase (instrument checkup)																												
NIR																													
HFR																													
OPTEL																													
Landers													Rehersal	Deployment															
LFR																													
X-Band Data transmission																													
				Early characterisation phase (ECP) (@35 km)						DCP 1.period (@10 km)			LDP (including 2 flybys)			DCP 2.period (@10 km)			Start Retreat to 100 km distance		IMPACT of DART	Move to Observation point			DCP 3.period (@10 km)				
Active for primary objectives																													
Optional																													
								ECP must be completed by this date!																					
				Linear Interpolation						Linear Interpolation																			
Achievable rates [kbps] -->			20	42	64	86	108	130	150	250	350	450	500	500	500	500	500	500	500	500	500	500	500	500	500	500	500	500	500
Per phase volume generation [Mbit]								1258.76				11261.56		2700				2523.13										3784.69	
time to complete transmission (Postponed) [h]								2.69				6.95		1.50				1.40										2.10	
Linearized volume generation [Mbit]			209.79	209.79	209.79	209.79	209.79	209.79	2815.39	2815.39	2815.39	2815.39	1350.00	1350.00	630.78	630.78	630.78	630.78					630.7821	630.7821	630.7821	630.7821	630.7821	630.7821	
Distributed transmission taime [h]			2.91	1.39	0.91	0.68	0.54	0.45	5.21	3.13	2.23	1.74	0.75	0.75	0.35	0.35	0.35	0.35					0.35	0.35	0.35	0.35	0.35	0.35	
Total per phase [h]								**6.88**				**12.31**		**1.50**				**1.40**										**2.10**	

Table 12-1: Transmission time over mission phases

Link Budgets

Due to the complexity of the mission, many link budgets were computed in different conditions. For reference most relevant cases are presented.

The results on downlink and uplink computations can be found in Table 12-1 and Table 12-2. ECSS (RD[13]) defines the link budgets guidelines.

The computations do not consider simultaneous RNG: dedicated ranging measurement slots are assumed.

The general conditions for Link Budget computations are:

- 35 m G/S antenna (49.9dB/K G/T, 107dBW EIRP)
- Min G/S Elevation: 10°
- Onboard estimated loss: 4dB
- Onboard Tx power = 35W (@ TWTA output)

	Downlink			
Frequency	8425 MHz			
Scenario	Max distance	Asteroid Orbit insertion	Time of DART impact	Reduced visibility @ Asteroid orbit insertion
Slant Range	3.3AU	1 AU	0.1 AU	0.13AU
Antenna	HGA, 0.5Deg BPE	HGA, 0.5° BPE	HGA, 0.5° BPE	LGA, 40° from Boresight
Spacecraft Antenna gain	36dBi	36dBi	36 dBi	4 dBi
TM Coding	Turbo ¼	Turbo ¼	Turbo ¼	Turbo ¼
Information rate	1.5 kbps	25 kbps	>2 Mbps	1.5 kbps
Carrier Recovery margin	8 dB	11 dB	>3 dB	6.7 dB
Link margin	4.8 dB	3.1 dB	>3.1 dB	3.5 dB

Table 12-2: Downlink budgets

12.5 List of Equipment

The list of Communications equipment is given in Table 12-3.

Element 1	AIM3 Spacecraft			MASS [kg]			
Unit	Unit Name Click on button above to insert new unit	Part of custom subsystem	Quantity	Mass per quantity excl. margin	Maturity Level	Margin	Total Mass incl. margin
1	miniDST X-Band		2.00	1.50	To be developed	20	3.6
2	HPA - TWTA		2.00	1.50	To be developed	10	3.3
3							
4	RFDN		1.00	4.00	To be modified	5	4.2
5	LGA		2.00	0.80	To be modified	5	1.7
6	HGA - metasurface		1.00	1.50	Fully developed	20	1.8
7	ISL electroc box		1.00	1.20	To be developed	20	1.4
8	ISL antenna		2.00	0.05	To be developed	20	0.1
-							
SUBSYSTEM TOTAL			7	14.4		12.1	16.1

Table 12-3: Communication System equipment list

12.6 Options

As an option, a Data Relay Package in support of DART DTE Link can be added on the AIM spacecraft.

Due to the criticality of a live, not repeatable, transmission from the NASA DART S/C at time of impact, the possibility for AIM to act as a backup receiver is seen as of interest.

Different concepts are identified for possible further evaluation:

1. To modify the AIM transponder to be able to receive on DART TX frequency *(major bandwidth requirement modification for AIM RX & temporary loss of ground control)*
2. To have a dedicated RX module on AIM for the DART standard direct to Earth link *(mass penalty from AIM). This can be done in a flexible/reconfigurable fashion to meet future missions needs (e.g. Probe EDL sniffing, other lander receptions, radioscience…). The already foreseen ISL unit on AIM may be modified and extended to receive on X-Band to perform such task with optimisation of mass and resources.*
3. To have a dedicated ISL between the two S/Cs *(this implies modification of the DART spacecraft but on AIM the already existing ISL system may be reused).*

12.7 Technology Requirements

A few developments are foreseen for this mission, with always backup solutions that, with some drawbacks, can still make the mission feasible within the schedule.

No pre-developments are foreseen for the Communication S/S as the situation is as follows:

- Miniaturised Transponder:
  - TRP ongoing ((E)BB of all S-Band RF and IF core module, Digital module emulated) with TAS-E:
    - (E)BB Testing ongoing

   - End of activity (TRL 4) by end of 2014
  - follow-on activities:
    - GSTP and flight opportunity on Proba-3
    - Deep-Space scenario, as needed for AIM mission, is not yet supported by follow-ons
- Metasurface HGA:
  - BB/EM of comparable design already achieved
  - follow-on activities well identified in a detailed roadmap with completion expected by 2017
- ISL:
  - Development of reference unit (Proba-3 ISL) started July 2014 (Tekever, PT) with delivery of FMs foreseen for Tri-1 2016.
  - To be evaluated in detail the AIM-specific requirements (functional, performance and environmental) and if any, modifications with respect to such development.

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13 DATA HANDLING

13.1 Requirements and Design Drivers

13.1.1 Requirements

The following DHS requirements have been identified:

- The DHS shall process the uplink signal received by the TT&C system, validate the commands and distribute them to the appropriate spacecraft subsystem for execution
- The DHS shall provide the capability of direct commanding essential satellite equipment without software intervention by means of High Priority Commands
- The DHS shall autonomously collect, prepare and store platform housekeeping data for transmission to ground
- The DHS shall autonomously collect all scientific and housekeeping data from the Payload and store it prior to download to ground. The total data volume generated by the payloads during the mission is 20.26 Gbit
- The DHS shall provide enough processing resources for:
  - Monitoring and management of satellite high level functions: attitude control, power conditioning, propulsion, thermal control and payload services
  - Support high-level autonomy functions (FDIR, Reconfiguration, Mode Transition)
  - Allow to be programmed using RTOS, high level languages and productivity tools (e.g. auto-coding) to allow faster ASW development
- The DHS shall be autonomous, in line with the mission objectives. It shall therefore include an autonomous start-up and initialisation capability upon availability of spacecraft power or upon DHS reset and autonomous reconfiguration of failed DHS parts
- The DHS shall provide adequate I/O as well as data bus connection for interfacing with all on board equipment and instruments.
- The maximum payloads data rate to the DHS are the following:

Payload	**Max data rate (kbps)**
VIS Camera	251
Thermal Imager	0.06
High Frequency Radar	80
Low Frequency Radar	20
OPTEL-D	5
Deployed MASCOT2/CubeSat	< 1Mbps

- The DHS shall support time-tagged telecommands storage and execution with defined accuracy

- The DHS shall include the satellite master clock and shall perform time distribution for all spacecraft units and payload units
- The DHS shall support FDIR both for its own proper functioning and for the whole platform system.

13.1.2 Design Drivers

In order to accomplish the low cost/mass and tight schedule targets whilst fulfilling the technical specifications, the less critical approach is to adapt a standard core computer already qualified in numerous projects.

In order to minimise risks, the following key points should be followed in the selection of existing DHS's:

- The DHS shall offer a high level of configurability for mission dependant functionalities, i.e. number of direct TC's, number of direct I/O's, number of input alarms, reconfiguration sequences, data storage size. This minimises the changes on the major hardware DHS modules in order to fulfil the AIM3 specifications
- The selected DHS's shall be available in a range of quality levels and performance. In order to limit cost, design shall focus on performance and functionalities more than reliability/availability
- The DHS shall have long heritage in design, development, assembly, integration and verification of hardware and software
- The DHS shall provide standard serial busses (SpaceWire, MIL-STD-1553, CAN) and standard discrete interfaces according to ECSS-E-ST-50-14C. A design based on consolidated standards favours the reuse of other spacecraft equipment interfaced with DHS
- The DHS shall already provide qualified SW building blocks, in particular a BSP for a RTOS, drivers for I/Os, HW/SW interface layer
- The DHS shall have an optimised mass budget through the grouping of individual functions into single modules
- The DHS shall have a modular box architecture enabling optional integration of additional modules/functions into the same mechanical housing.

13.2 Assumptions and Trade-Offs

13.2.1 Assumptions

- No critical requirement involving new or specific technologies has been identified for the DHS. In case the autonomous navigation would imply extended computational needs, a dedicated processing module is assumed as part of the camera sensor in the GNC system. A possible example of such additional processing module is described in RD[22].
- The payloads data rate is compatible with widely used data link solutions like MIL-STD-1553, CAN, SpaceWire.
- It is assumed that payload instruments prepare both the science data and payload housekeeping data for download to ground. The DHS is not required to perform any processing of the payload data with the exclusion of packetisation function

and storage before ground sending. The performance of the DHS processing function is therefore assumed to be compatible with SPARC LEON2/3 CPUs.

- During critical mission phases (e.g. lander deployment, close proximity operations) the autonomous reconfiguration time in case of DHS failure is assumed to be compatible with cold redundancy of the processor module executing the application software. Typically, in case the DHS nominal processing function fails, the redundant one can take over within 5 to 15 seconds, depending on whether a self-test is performed or not.

13.2.2 Trade-Off

In order to achieve the low cost objectives, the DHS trade-off has assessed the suitability of available COTS based equipment and equipment not fully qualified according to ESA standards.

Three different DHS platforms have been considered:

- The CDMU and RTU designed by Astrium as part of the standard AS250 spacecraft platform and used in the SEOSAT and CHEOPS missions
- The ADPMS spacecraft controller developed by Verhaert Space for Proba 2
- The OBDH designed by Surrey as part of the SSTL-150 spacecraft platform and used in several commercial LEO missions.

13.2.3 Astrium AS250 CDMU and RTU

The AS250 CDMU and RTU units have been designed with the aim of providing an avionic product suitable for a wide range of missions. They offer the same performances for different class quality programs, with adapted EEE selection and manufacturing quality level.

The CDMU is formed by two identical boards in hot/cold redundancy with interlink between CPUs.

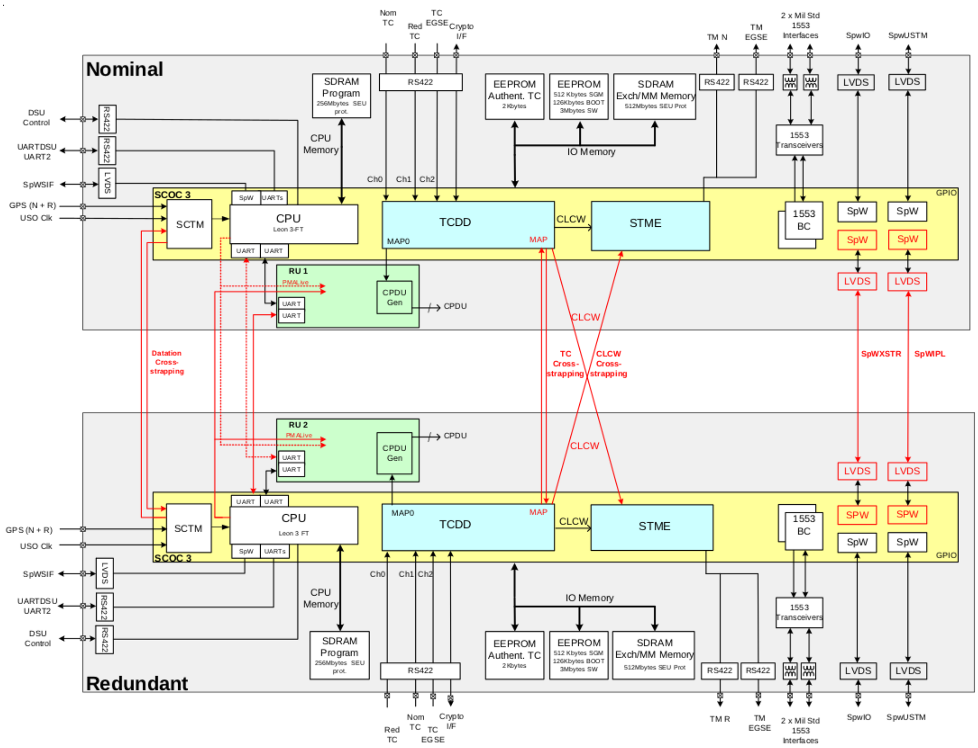

Figure 13-1: AS250 CDMU architecture

The core of the unit is a complex System on a Chip ASIC (called SCOC3) gathering on a single die all the main digital functions of the CDMU. Its small size allows it to integrate the whole computer on one board. At architectural level, it integrates in a single chip a LEON3 processor with caches, Memory Management Unit and Floating Point Unit. The Telemetry/Telecommand logic and all the standard peripherals controllers (SpaceWire, CAN bus, Mil-1553, UART) are also integrated along with the processor in the same die. The performances are up to 68 MIPS@80MHz. The reconfiguration function and up to 16 Gbit Mass Memory are hosted in the same board. The SCOC3 ASIC would allow to interface up to 20Gbit of SDRAM.

Cross-strap links are implemented to connect the processor of each board to the TC Decoders, the Reconfigurations Units, the Safeguard Memories and the Mass Memories of both boards.

The available interfaces include 7 SpaceWire links (4 are allocated internally to TM/TC channels and for cross-strapping, the others are available for the platform usage) and 2 CAN or Mil-Std-1553 busses as spacecraft bus.

Figure 13-2: AS250 OSCAR Computer

The RTU is a fully redundant unit based on four main constituents:

- Two CDMU interfaces modules nominally used in cold redundancy. Each module interfaces a redundant Mil-Std-1553 bus and is in charge of the control of the standard interface modules, of the AOCS interface modules and of the propulsion interface modules.
- Two standard interface modules nominally used in hot redundancy. Both standard interface modules are in charge of the telemetry acquisition signals, of the control of UART links, of the distribution of synchronisation signals and of the generation of the high power commands and low level commands.
- Two AOCS modules nominally used in cold redundancy and in charge of controlling and monitoring all AOCS sensors/actuators. It is also in charge of powering of magnetometers and of pressure transducer, of the control of reaction wheels and of magnetorquers.
- Two propulsion modules nominally used in cold redundancy and in charge of the control of latching valves, flow control valves and cat bed heaters.

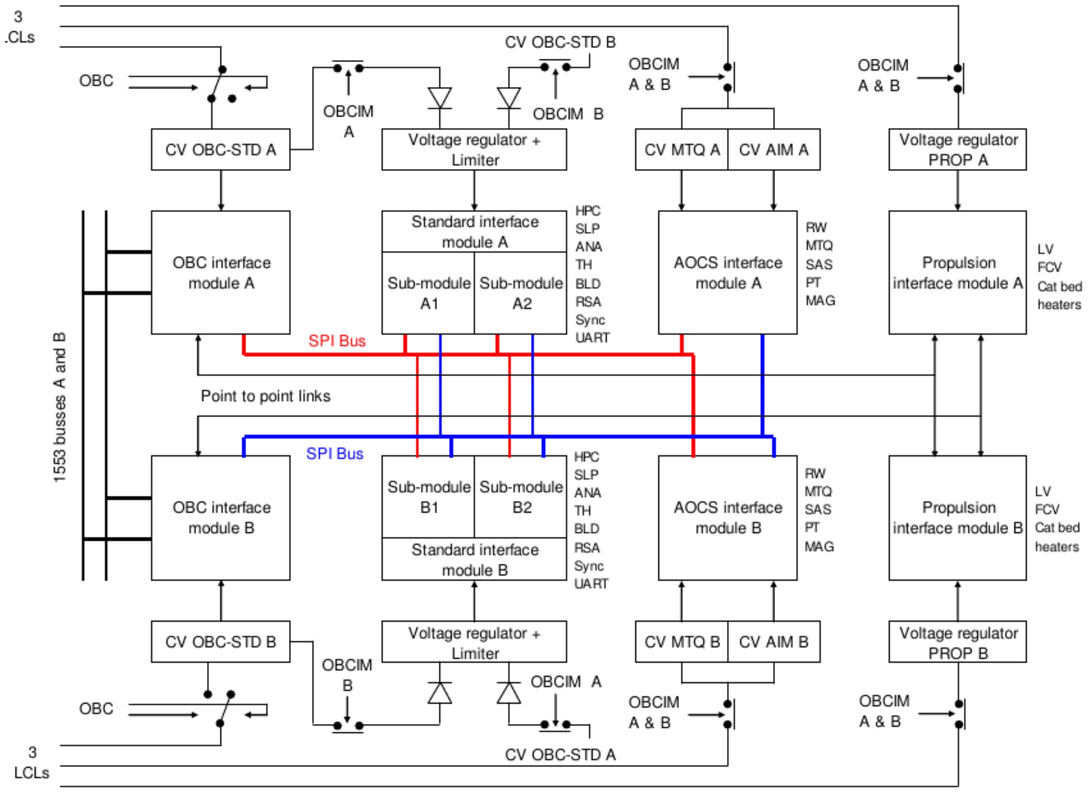

Figure 13-3: AS250 RTU functional overview

In this study the Astrium AS250 CDMU and RTU has been selected as baseline. It provides a high number of standard I/O interfaces and would probably need few HW design changes limited to the RTU to comply with the AIM3 needs. In order to further reduce the costs, the parts associated to the I/O interfaces that are not necessary in AIM3 could be not procured or mounted.

A possible limitation is the mass memory that is limited by design to 16 Gbit. In case additional data storage would be needed the selection should be reconsidered.

Figure 13-4: AS250 RTU

13.2.4 ADPMS

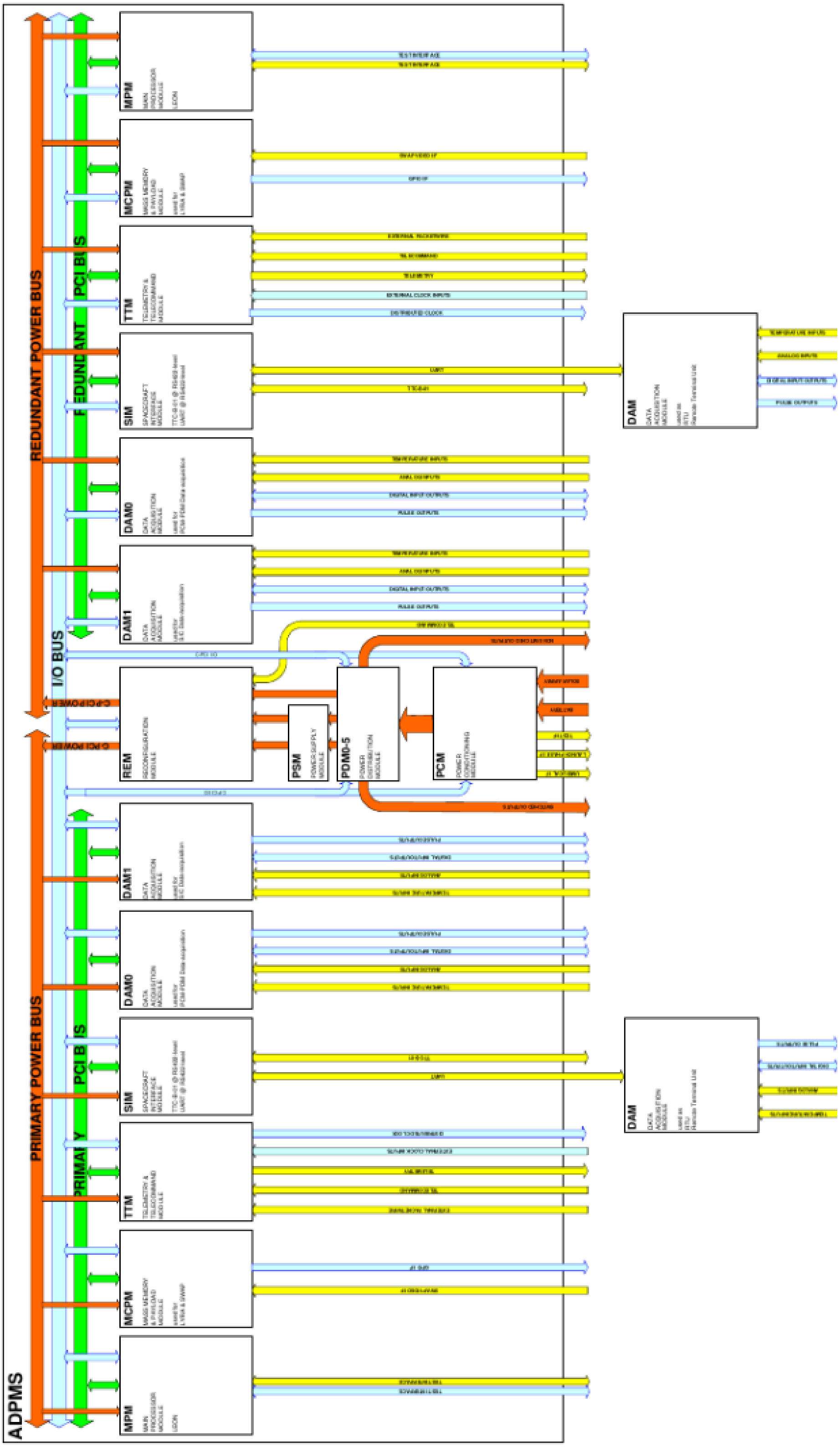

Figure 13-5: ADPMS functional overview

ADPMS (Advanced Data & Power Management System) is a control unit based on LEON2-FT processor and designed by Verhaert Space for Proba 2 and in general for small satellites with high performances in terms of autonomy, on-board processing capability and payload management. The main objective was to design an easy expandable and highly integrated control unit in order to provide more resources in terms of mass and power to payloads.

The ADPMS architecture defines a two-lane system with a switch-over redundancy scheme. The nominal lane is fully powered and controls the platform, while the redundant lane is fully powered off in order to decrease the failure rate and overall power consumption. The ADPMS has a different TC module redundancy concept in order to eliminate the need for a separate power-switch. To compensate for the fact that TC decoders are cold-redundant, a reconfiguration and emergency command module has been implemented. Differently from most of the available DHS systems, the ADPMS incorporates also a power system consisting of power conditioning and power distribution modules.

All modules have 3U Compact PCI format and communicate to each other via a high speed PCI backplane. The re-use of this industrial standard allows the system to expand with reduced design complexity and test effort.

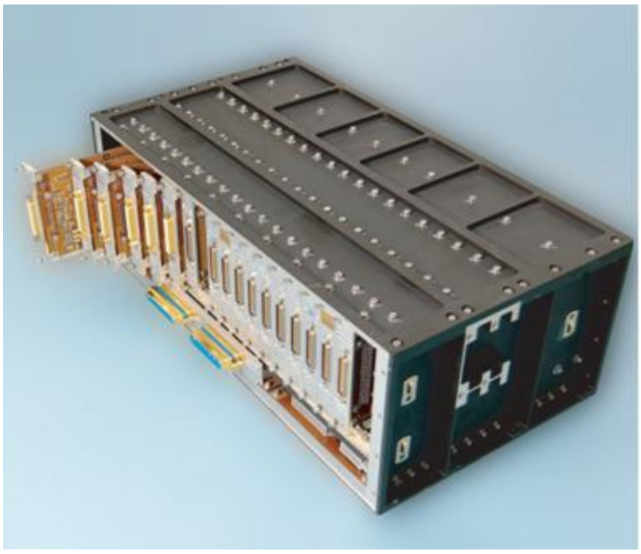

Figure 13-6: ADPMS

The spacecraft communication module is based mainly on a large number of UART channels and therefore would require design changes in case serial communication busses (Mil-Std-1553 or CAN) are requested. SpaceWire links shall also be added to the design in order to interface with the camera and other possible high speed equipment.

Mass memory is limited to 4 Gbit, but the design allows for an easy extension.

Despite the high level of integration, low mass and power (13 kg, 17 W) and autonomous operations the ADPMS has not been considered as baseline due to the design changes needed for the communication and for the mass memory modules.

13.2.5 SSTL OBDH

SSTL is highly experienced in developing low cost COTS based data handling modules designed mainly for LEO applications. The OBDH subsystem proposed in several missions is based on a redundant architecture with a CAN bus local network connecting a number a standard modules: The On Board Computer, the High Speed Data Recorder, and the Payload Interface Unit (PIU)/Remote Terminal Unit (RTU).

The On Board Computer module is the OBC386 computer board or the recently developed OBC750 based on the powerful IBM PPC750FL CPU and offering also MIL-ST-1553 as spacecraft bus.

Figure 13-7: SSTL OBC750, High Speed Recorder and PIU/RTU

The High Speed Recorder can store up to 16 Gbytes and support high speed acquisition via SpaceWire links.

A range of PIU/RTUs has been developed providing TM/TC interface between platform and the payload.

Many functionalities that in most of the available DHS systems are normally implemented in HW, i.e. Telemetry, Telecommand and Reconfiguration functionalities, are implemented in software by the onboard computer or by microcontroller modules distributed throughout the system.

SSTL OBDH is not considered at the this stage of the AIM3 project a suitable solution for the following reasons:

- SSTL has developed its own engineering and process approach that differs from the ECSS standards. The understanding of the discrepancies with the ECSS standards and their mitigation could be a long and complex task.
- The limited autonomy level provided by the existing SSTL OBDH platforms is probably not appropriate for the operational needs of AIM3 since it would require frequent ground-in-the-loop commanding cycles.
- A clear reliability analysis could be difficult to perform due to the extensive use of COTS components and the missing heritage of deep space missions to prove a certain reliability level.

13.3 Baseline Design

The proposed baseline DHS architecture is quite standard and is based on the fully redundant CDMU-RTU pair re-used from the SEOSAT mission.

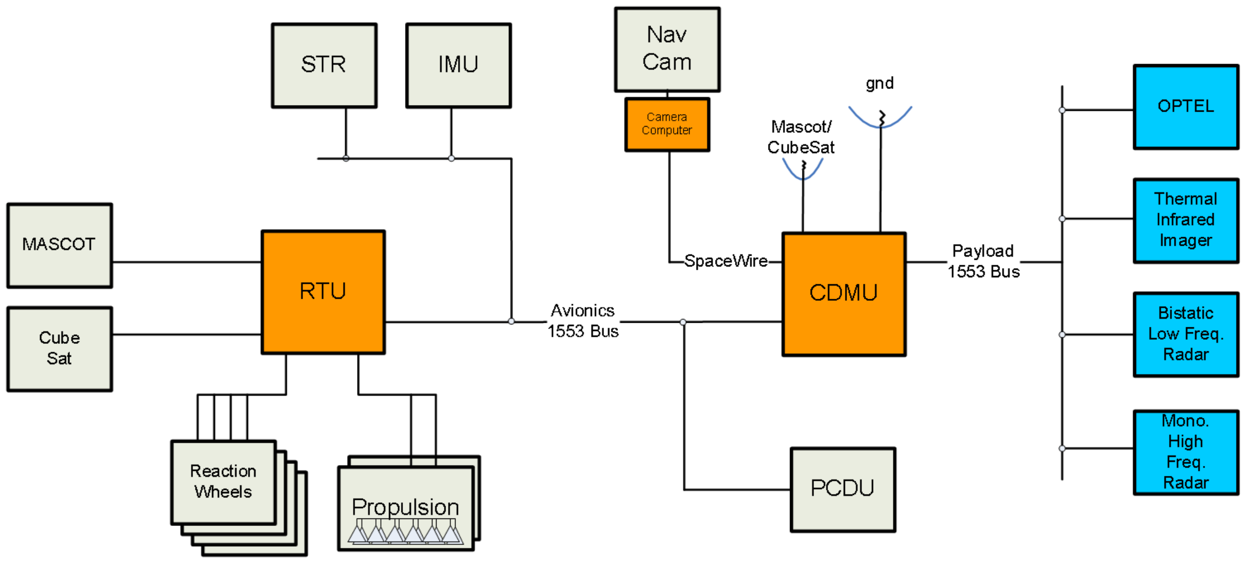

Figure 13-8: AIM3 DHS architecture

The CDMU operates as controller on a set of two redundant 1553 busses, one dedicated to the platform including the power management unit, the other dedicated to the payload. Analog and discrete interfaces are handled through the RTU. Data from the VIS camera are transferred to the CDMU via dedicated SpaceWire link and elaborated by the GNC application. MASCOT2 lander and CubeSat are connected to the RTU before deployment. After deployment all commands (from ground or pre-planned on board) not addressing the AIM satellite are routed to the ISL unit connected to the CDMU and dispatched to MASCOT2 or CubeSat.

13.4 List of Equipment

The table hereafter shows the chosen equipment.

Element 1	-			MASS [kg]			
Unit	Unit Name Click on button above to insert new unit	Part of custom subsystem	Quantity	Mass per quantity excl. margin	Maturity Level	Margin	Total Mass incl. margin
1	CDMU		1	5.0	Fully developed	5	5.3
2	RTU		1	12.3	To be modified	10	13.5
-	Click on button below to insert new unit						
	SUBSYSTEM TOTAL		2	17.3		8.6	18.8

Element 1	-				-
Unit	Unit Name Click on button above to insert new unit	Part of custom subsystem	Quantity	Ppeak	Pon
1	CDMU		1		21.0
2	RTU		1		26.0
-	Click on button below to insert new unit				
	SUBSYSTEM TOTAL		2	0.0	47.0

Table 13-1: AIM3 DHS mass and power

13.5 Options

In case lower mass and power would be needed, the AS250 RTU could be replaced by the AFIO (Advanced Flexible I/O) unit developed by RUAG as generic and easily adaptable RTU. Models of AFIO has been recently developed and qualified at EQM level. Two module types are present in this unit: the AOCS module for sensors, reaction wheels, propulsion interface and the Standard I/O module for thermistors, Bilevel, Analog, relay status etc.

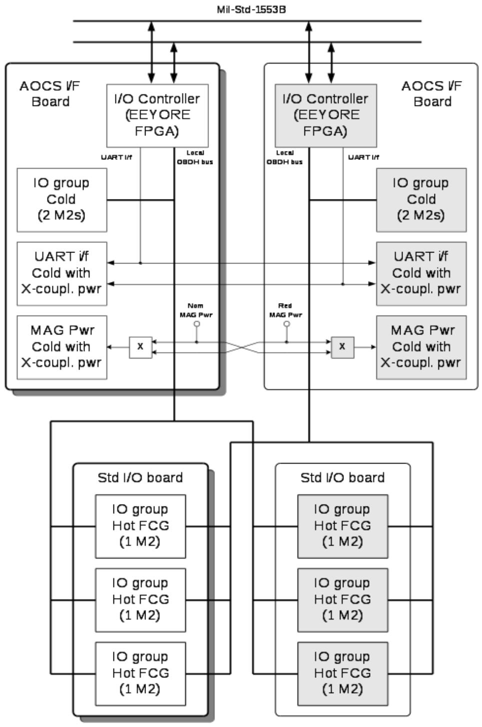

Figure 13-9: RTU architecture based on AFIO design

The RTU architecture is based on multiple instantiations of a mixed signal ASIC (called M2) that directly interfaces sensors and actuators. The M2 ASICs are internally connected by an OBDH bus. An I/O Controller hosted in the AOCS board is in charge of the communication with the CDMU and controls the internal OBDH bus. The modular architecture allows to easily increase the number of interfaces by adding additional I/O standard boards to the internal bus. The link between the RTU and the CDMU (currently Mil-std-1553 bus) can also be replaced by CAN or SpaceWire with limited design changes in the I/O controller.

13.6 Technology Requirements

The following technologies are required to this domain:

Equipment and Text Reference	Technology	Suppliers and TRL Level	Technology from Non-Space Sectors	Additional Information
CDMU	European	Astrium, TRL 8	N/A	Full reuse from SEOSAT mission
RTU	European	Astrium, TRL 7	N/A	Reuse from SEOSAT.

Equipment and Text Reference	Technology	Suppliers and TRL Level	Technology from Non-Space Sectors	Additional Information
				Limited changes could be needed. Parts associated to the I/O interfaces that are not necessary in AIM3 could be not procured or mounted.

14 THERMAL

14.1 Requirements and Design Drivers

14.1.1 Basic Requirements for the Thermal Control Subsystem (TCS)

The TCS shall provide the thermal environment (temperature ranges, gradients, and stability) required to ensure full performance during each mission phase and mode (active drift receive/transmit, orbit insertion, measurement mode with/without transmit, orbit corrections) as well as for the complete duration of the mission.

The thermal control design shall be compatible with the in-orbit environment expected for all nominal variations of the solar aspect angle, Earth albedo and infrared radiation (LEO operation), Asteroid albedo and infrared radiation (during fly-by's/measurements). Worst-case hot and cold conditions have been preliminarily identified and analysed.

14.1.2 Thermal Stability Requirements

Demanding pointing requirements have consequences on thermal control in terms of requirements on absolute temperature, and on temperature variations as a function of time and space.

The TCS shall guarantee the thermal stability between the LOS of the navigation camera and the LOS of the star tracker to ensure navigation requirements are fulfilled. It is necessary to minimise the impact of the variation of the bias between both LOS, due to thermo-elastic distortions.

The thermal control shall guarantee that the contributions of thermally induced effects to the observation error budget remain below the allocations.

14.1.3 Design Drivers

- Low cost mission with incentives to use high TRL equipment
- Deep space mission, as far as 2.2 AU from the Sun during the interplanetary transfer, limiting the power generation and heater power
- No hibernation during the interplanetary transfer as no benefit for Operations (cost) is foreseen
- The spacecraft's attitude is <u>not</u> driven by the TCS because it is already heavily constrained by the Communications and Power systems, dismissing the possibility of using Sun illumination on the radiators to compensate for internal thermal dissipation variations.

14.2 Assumptions

Typical Platform (PF) equipment operational temperature range between -10 °C and +40 °C has been used for the preliminary sizing of the whole PF TCS, while the PL temperature range are summarised in the "Payload Thermal Control" section. This will of course need to be consolidated in future design phases since some equipment generally have more stringent (e.g. battery) or more relaxed (e.g. electronics) requirements.

The attitude of the spacecraft considered so far provides one surface oriented away from the Sun (the surface opposite that which carries the HGA) during all mission phases except some short manoeuvres. This surface is therefore chosen as the preferred radiators' location because it has the lowest sink temperature (parameter that characterises how much heat can be extracted from a surface) and will from now on be referred to as the spacecraft's "anti-Sun face".

The values for the internal thermal dissipation of the spacecraft during the different phases are extracted from the power budget. Concerning the TWTA power demand, roughly ¼th of it is converted to RF power and thus is not accounted as thermal dissipation.

14.3 Uncertainties and Margins

Throughout this study, the uncertainties associated with computed temperatures, justified by the project maturity, are the following:

- + 10 °C in the Hot Case
- - 10 °C in the Cold Case.

Besides the classical mass margin with respect to the product maturity, the following margins have been taken for the TCS:

- + 10 % on calculated radiator surface with justification radiators trimming
- + 100 % heaters mass to cover variations of external environment and the fact that no modelling was performed.

14.4 Baseline Design

14.4.1 General Concepts

The AIM-3 spacecraft thermal control system is designed to maintain all the units within their required temperature range during all the phases of the mission while minimising heater power and mass.

The thermal control will rely on common and robust concepts, using well-proven techniques and hardware, such as multi-layer insulation (MLI) and radiator foils covering the spacecraft external surfaces, black paint coating the internal surfaces, temperature sensors and software controlled heaters.

As some Sun illumination is expected on the radiators during manoeuvres, a surface finish with high IR emittance and low solar absorptance is chosen for the radiators, such as optical solar reflectors (OSRs):

$$\begin{cases} \text{IR emittance}, \varepsilon = 0.80 \\ \text{Solar absorptance}, \alpha = 0.12 \end{cases}$$

Table 14-1: Typical OSR optical properties

The spacecraft is composed of three cavities: the bottom one accommodating the propulsion module, and front and back ones gathering the different platform units and payloads.

While this configuration makes accommodation simple, it poses the problem that the front cavity has got no common wall with the anti-Sun face for heat extraction. The thermal dissipation of the front cavity therefore has to be extracted by radiators located on the sides of the spacecraft dedicated to the solar arrays (SAs) instead of the anti-Sun face. However, these sides have higher sink temperatures because of the IR flux from the SAs to the spacecraft, and hence require larger radiators than on the anti-Sun face. Another solution would be the use of heat transport devices but these come with a mass and complexity penalty (not foreseen in the baseline current design).

Thermal constraints must therefore be taken into account early in the units accommodation in order to select the most favourable ones for the front and back cavities. In the current design, the elements with the highest dissipations and duty cycles have been accommodated in the back cavity (the one associated to the anti-Sun side).

14.4.2 Platform Thermal Design

The units will preferably be directly mounted on the platform walls acting as radiators. A classical thermal control based on the use of interface filler, internal black paint, OSRs and MLI is foreseen. Objectives are to get the better conductive coupling to the radiators and homogenisation of the internal spacecraft temperature by improving the radiative couplings inside the cavities.

Finally, electrical heaters will prevent excessive cooling of units, structures and propellant during the cold phases (typically during the interplanetary transfer) and eclipses (if any). They will be actively controlled through thermistor temperature acquisition.

Most of the units are conductively cooled: dissipation is transferred from the baseplate or the foot/bracket of the unit to the radiator via the supporting honeycomb panel. The conductive couplings can be improved by means of thermal interface fillers and aluminium doublers. Aluminium doublers will also be used to increase inertia under units having a transient dissipation peak such as TWTA.

All the internal units and the sidewalls are painted black in order to homogenise the temperature of the cavities.

No critical issues were found with respect to the units' temperature requirements, but special care shall generally be taken for:

- The battery thermal control, as the operating temperature range is generally tighter than those of the others units, even though not stringent
- The reaction wheels thermal design, as their geometrical accommodation constraints prevents from mounting them directly on a radiator panel.

14.4.3 Propulsion System Thermal Control

The propulsion internal equipment is collectively controlled, taking advantage of their modular layout. The thermal collective control concept has the advantage to better homogenise the temperature of the lines and to be more flexible in terms of design adjustments and integration. It will be insulated from the platform cavities by classical MLI. External parts of the propulsion system (main engine thruster, attitude control thrusters, etc.) will nevertheless require an individual heating line to maintain them in their temperature range.

14.4.4 Appendages Thermal Control

All the spacecraft external appendages (antennas, attitude sensor and actuators) will be as much as possible decoupled from the spacecraft, and have their individual thermal control.

14.4.5 Star Tracker Thermal Control

The Star Trackers thermal design will be optimised in order to reduce as much as possible temperature fluctuations of their interface along the orbit. They can be thermally decoupled from the platform thanks to a CFRP bracket (with a small conductivity) and fitted with a radiator plate mounted between the Star Trackers' foot and the bracket. This design allows to limit temperature gradients into the Payload Interface Panel and to provide the requested thermal environment to the star trackers optical head.

14.4.6 Payloads Thermal Control

The following table summarises the estimated temperature range for the PL instruments:

Payload	**Operational temperature**	**Thermal Stabilization range**	**Non-Op temperature**
	[C°]	[C°]	[C°]
VIS Camera	-30, +40;	TBD	-40, +60;
Thermal Infrared Imager	-30, +30;	**0,01**	-30, +45;
Monostatic High Frequency Radar	-40, +50;	TBD	-45, +50;
Bistatic Low Frequency Radar (Orbiter)	-40, +50;	TBD	-45, +50;
Bistatic Low Frequency Radar (Lander)	-40, +50;	TBD	-45, +50;
OPTEL (Optical terminal)	-15, +60;	TBD	-40, +60;
OPTEL Laser terminal	>= 0	TBD	TBD
MASCOT-2	-30, +30;	TBD	-35, +50;
COPINS (including deployer)	-30, +30;	TBD	-35, +50;

Table 14-2: AIM-3P PL equipment temperature range (qual. TBC)

It is ASSUMED that such temperature ranges are qualification limits.

The following figure shows the PL Instruments internal accommodation:

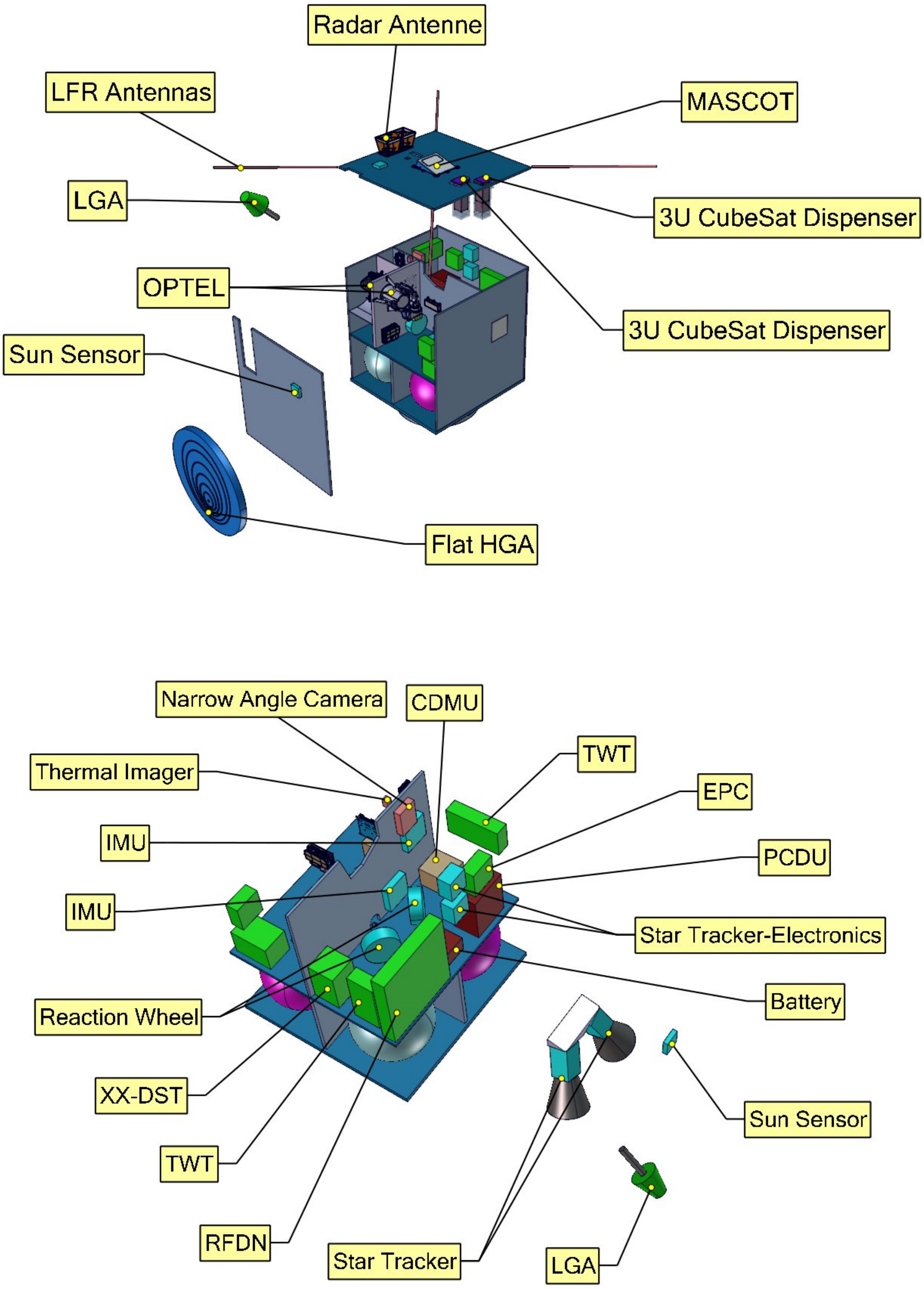

Figure 14-1: AIM-3P PL internal accommodation

The instruments are assumed to be mounted upper bay of the satellite. The most dissipative units shall benefit from the "shadow-side" excellent view of cold space for almost the entire mission, including the encounter with the asteroid.

The other PL instruments shall share the upper shear panel in order to minimise the heater power request and have a more stable temperature variation during operation.

In particular, due to the thermal stability requirement of 0.01degC, the Thermal Infrared Imager shall be thermally "independent" from the rest of the S/C. this is guaranteed with the use of MLI and thermal washer. In addition, beside the heater request for the NON-operative phase, it shall be equipped with "compensation" heater in order to further stabilise the temperature fluctuation.

The current configuration shows a Reaction Wheel (RW) facing the PL bay. This could induce a temperature fluctuation on the PL as the RW is usually designed to radiate its own power dissipation. In order to reduce such temperature fluctuations the PL shall be thermally decoupled from the RW by means internal MLI blanket and thermal washer.

It is assumed that units such as the OPTEL-D Optical Terminal that have "high" minimum temperature limit shall be placed on the shear panel in order to be further isolated from the external environment.

Currently it is not foreseen to use any Heat Pipe (HP) to link thermally internal unit to the external panel, as it has been assumed the high dissipation units are flat-mounted on the "shadow-side".

But, in case of configuration constraint of some PL unit, heat pipes can connect internal units with a dedicated radiator (on the upper side of the S/C). The mass penalty will be minimal as:

- A portion of the external panel will be uncovered by the MLI
- A minimum quantity of OSR tiles will be glued
- 2 short (<500mm) HP will be added to the TCS mass budget(<1kg).

Instruments are generally covered with multi-layer insulation (MLI), but some parts may be left uncovered to aid in cooling of the structure and CCD if needed. All of the critical optical components are generally fitted with heaters so that during the first few days after the launch their temperatures can be maintained above those of the rest of the structure. These heaters are intended to reduce the risk of contamination in the critical time period after launch and/or to bake out the moisture from critical structures (typically graphite fibre/ epoxy structure may require this).

14.4.7 Preliminary Sizing

At this early stage of the project, a preliminary sizing was performed, the goals of which are to estimate the required radiator area of the spacecraft and the average heater power during the different mission phases. This is based on seeking the worst hot and cold cases that the spacecraft will experience throughout its complete lifetime and which size the TCS. The identified worst hot case is during the orbit correction at the asteroid, while the worst cold case is the active drift during the interplanetary transfer when only receiving communications and not transmitting. The characteristics of both cases are tabulated here below along with the resulting radiator area and heater power.

Worst Hot Case Orbit corrections (at asteroid)	Worst Cold Case Active drift, Receive
• Sun distance = 1.6 AU • Internal power consumption = 483 W (including RF power and heaters)	• Sun distance = 2.2 AU • Internal power consumption = 274 W (including heaters)
• Radiator area = **0.9 m²** (with margin)	• Heater power = 48 W (no margin considered)
Worst Hot Case measurement + Tx	
• Sun distance = 1.6 AU • Internal power consumption = 373 W (including RF power and heaters)	

Table 14-3: TCS worst sizing cases yielding the radiator area and maximum average heater power

The average heater powers required to prevent excessive cooling during the different phases are also calculated and tabulated here below.

	Active drift, Receive	Active drift, Transmit	Maneuver, Coll. Avoid. Lander deployment	Msrmt mode, non-transmit	Msrmt mode, with transmit	OPTEL
PF	20 W	12 W	0-12 W	20 W	19 W	18.4 W
PL	53 W	53 W	53 W	26.4 W	26.4 W	26.4 W

Table 14-4: Average heater power throughout the different mission phases

The resulting TCS configuration is shown here below. It consists of 0.1 m² OSRs on the spacecraft SAs sides acting as radiators for the front cavity and 0.73 m² of OSRs on the anti-Sun side acting as radiators for the back cavity. The remaining external surface is covered with MLI.

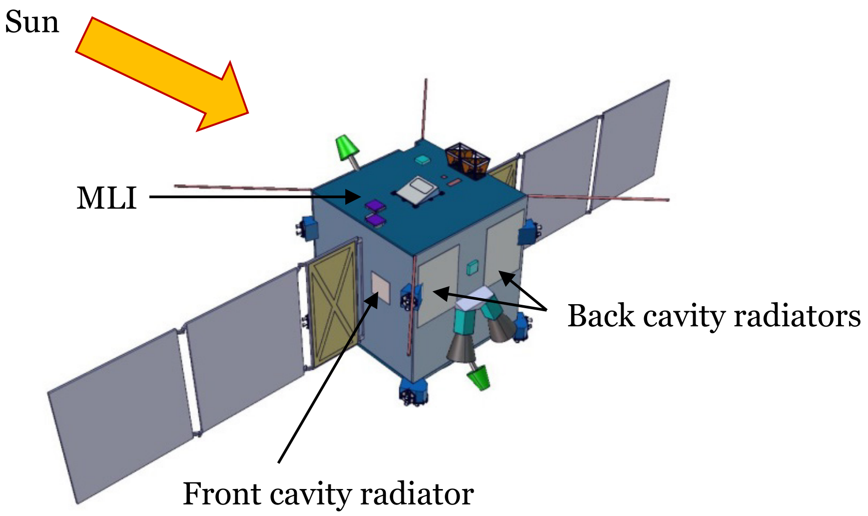

Figure 14-2: TCS external configuration

14.5 List of Equipment

Here is a detailed description of the TCS's equipment and mass breakdown.

MLI

- Area = 10.14 m^2
- Density = 0.4 kg/m^2
- Mass = 4.1 kg

Radiator

- Area = 0.9 m^2
- Density = 0.5 kg/m^2
- Mass = 0.4 kg

Black paint

- Area = 16.90 m^2
- Density = 0.15 kg/m^2
- Mass = 2.5 kg

Heater lines

- Installed power (PF) = 54 W
- Installed power (PL) = 53 W
- Heater power density = 0.1/cm^2
- Heater area = 0.192 m^2
- Heater density = 0.15 kg/m^2
- Mass < 0.1 kg

Miscellaneous

- Thermistors, thermal washers, thermal fillers, etc.
- Mass = 0.5 kg

Component	**Mass [kg]**	**Maturity margin**	**Margin mass [kg]**	**Mass with margin [kg]**
MLI	4.1	10 %	0.4	4.5
Radiator	0.5	5 %	0.0	0.5
Black paint	2.5	10 %	0.3	2.8

Heater lines	0.1	5 %	0.0	0.1
Miscellaneous	0.5	5 %	0.0	0.5
Total	7.7	9 %	0.7	8.4

Table 14-5: TCS baseline design mass budget

14.6 Future Work

Next steps for the TCS design include:

- Consolidation of the units accommodation with respect to their dissipations and duty cycles
- Analysis of payload thermal requirements, particularly any potential Infrared imagers.
- Thermal modelling of the spacecraft
- Detailed thermal analysis of all cases and phases
- Detailing of the TCS at unit level (e.g. batteries, tanks and reaction wheels)
- Planning of the heater lines redundancy scheme
- Drawing a test plan.

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15 ONBOARD AUTONOMY

The autonomy needs are dictated by the time criticality of the required operational decisions to be taken, where ground-in-the-loop commanding cycle is not feasible; by technical or operational constraints preventing or hindering the direct ground commanding (e.g. spacecraft visibility, ground stations availability, bandwidth limitations); by the operations costs reduction needs (trade-off analysis is required).

15.1 Requirements and Design Drivers

The autonomy drivers identified for the AIM-3 mission are presented in the following sub-sections.

15.1.1 Operations Concept

From the operations costs perspective, AIM-3 is envisaged as a low-cost mission, which may warrant transfer of some operational responsibilities from ground to the on-board autonomy functions.

From the operational constraints perspective, the mission architecture may require on-board activities, which are nominally performed in the ground-in-the-loop context. This would include, but not be limited to, the orbital and attitude configurations, ground station visibility and availability, required response time.

15.1.2 Trajectory Planning and GNC

The required mission execution manoeuvres, especially in the proximity of the asteroid bodies, may require autonomous functionalities, due to the timeliness constraints.

The station keeping functions may reduce the operations costs, if performed autonomously.

15.1.3 Hazard Avoidance

Autonomous hazard avoidance functions may be required for operations in the proximity of the asteroid bodies to ensure the spacecraft safety and mission success, e.g. within the sphere of influence, activities related to the Lander deployment.

15.1.4 Lander Deployment

Specific autonomous functions requirements to perform the Lander deployment, induced by the Lander (e.g. landing zone targeting) are to be specified. Requirements for the spacecraft manoeuvring and safety, if not achievable by the pre-planned ground operations, will have to be specified as autonomous functions.

15.1.5 FDIR Goals

Specific FDIR (RAMS) requirements, especially for the critical mission phases (e.g. operations inside the sphere of influence, close proximity operations, Lander deployment), will affect the autonomy level required.

15.2 Assumptions and Trade-Offs

15.2.1 Operations Concept

Based on the currently elaborated Operations concept, the envisaged man-power and facilities usage are considered to be nominal. Accordingly, no need for or benefit from the transfer of functionalities from ground operations to the on-board autonomous functions have been identified.

As sufficient ground stations visibility can be achieved to support the envisaged operations, no need for the autonomous functionalities has been identified from this perspective.

15.2.2 Trajectory Planning and GNC

In accordance with the current operations concept, all the phases of the mission execution can be supported through the in-advance ground planning and time-based execution of the mission timeline. In this regard, no specific autonomy needs have been identified.

Following the current operations analysis, station keeping requirements can be achieved through tracking and approx. 2 planned manoeuvres per week. No autonomy requirements have been identified in this regard.

15.2.3 Hazard Avoidance

Based on the currently elaborated operations concept, hazard avoidance is envisaged to be achieved through the ground analysis of the operational environment and planned accordingly spacecraft trajectories and manoeuvres. No autonomous hazard avoidance is currently required.

15.2.4 Lander Deployment

Currently available initial information on the envisaged Lander deployment strategy suggests an approach based on the ground-planned spacecraft trajectory and attitude, and activation of the Lander release mechanism. No specific autonomy needs are currently identified in this regard.

15.2.5 FDIR Goals

A critical scenario identified is a potential failure while the spacecraft is within the sphere of influence and during the proximity operations (i.e. Lander deployment). Based on the current operations concept, the spacecraft, during this scenario, will be operating while on the escape trajectory from the asteroid body. This would mitigate a collision feared event for any non-thrusted failure mode. For these cases, no specific autonomy or FDIR requirements have been identified.

For the potential thrusted failure modes further analysis will be required.

A change to an assumption of the escape trajectory will require re-evaluation of the FDIR goals, as well as the hazard avoidance autonomy requirements.

15.3 Baseline Design

The elaborated mission profile can be considered as "nominal" deep space mission. Accordingly, this implies the following conclusions:

- Real-time control from ground not feasible due to propagation delay
- Ground-based mission planning is required (e.g. spacecraft navigation)
- Operations through the time-based mission plan execution on-board the spacecraft.

Hazard avoidance functionalities are currently considered to be part of the mission planning.

Lander deployment functionalities are to be pre-planned and are currently considered to be part of the mission planning.

FDIR functionalities are considered to be nominal, under the current assumption the escape trajectories and non-thrusted failure modes. For this, mission planning is to ensure the safe mode reachability and adequacy at any point of the mission execution.

Based on the above, the currently recommended autonomy level is E2 (ECSS-E-ST-70-11C).

15.4 Options

15.4.1 Potential Drivers for Autonomous Navigation

Should additional drivers for inclusion of the autonomous navigation be identified, it would imply the need for the position determination functionalities, which would lead to the requirements on additional sensors.

This would also lead to the extended computational needs on-board for image processing (FPGA, co-processor), localization, and GNC functionalities.

15.4.2 Further Hazard Avoidance Needs

Mission/System hazard analysis will need to be performed. This should include hazards identification, as well as classification into time-critical/non-time-critical hazards. Based on this, introduction of the specific hazard avoidance functions may imply the need for additional sensors (e.g. an altimeter). This may imply extended computational needs on-board, adaptations to the avionics architecture. The major factors in this regard will be:

- Sense-to-commanding timing needs
- Spacecraft reactivity and performances needs and their feasibility.

15.4.3 Potential Lander Deployment Constraints

Further elaboration of the Lander deployment concept may lead to the need of the landing zone targeting. Currently it is envisaged to be achieved through the pre-planned ground operations. Should any need for autonomous functions be identified in this regard, system impacts comparable to the autonomous navigation can be expected. It may also imply the need for the autonomous hazard avoidance functions.

15.4.4 Technology Demonstration Objectives

Currently no technology demonstration objectives, related to Autonomy, have been specified.

As inclusion of an optical terminal is being considered for the communication technology demonstration purposes, its potential use for the autonomy functions, i.e. doubling as a laser altimeter, might be investigated as a technology demonstration objective.

16 SIMULATION AND VISUALISATION

16.1 Introduction

In the scope of the AIM-3P study, different analyses have been performed but also some 3D visualisation support has been provided.

The objective of the analyses was to characterise some design driving factors and their impact in the mission and system design trade-offs, in particular the relation between the spacecraft trajectory (relative to Sun, Earth and the asteroids system), the pointing strategies and the configuration, as well as close-proximity operations dictated by the payload operations.

The 3D visualisation support aimed to further help the study team understanding the mission geometry and the implications on the close-proximity operations from the payloads operations point of view.

For this purpose a Project Test Bed (PTB) simulator has been setup.

16.1.1 Description of the Simulator

The PTB is a simulator based in EuroSim. The Position Environment Model (PEM) has been used for the computation of the Solar System bodies' orbits and rotations, while Mission Analysis ephemeris has been used for the spacecraft trajectory. Furthermore PC-IGS has been used as the 3D visualisation framework.

The EuroSim simulation environment provides the user-interface to control the simulator and generate reports with the numerical results of the simulation, as well as the connection to the 3D visualisation.

16.2 Requirements and Design Drivers

The design drivers analysed in this section cascade down directly from the mission schedule assumed upfront, and the chosen interplanetary cruise and nominal operation trajectories to satisfy payload operations constraints.

In the following sections the spacecraft distances to Sun and Earth, the relative positions spacecraft, Sun, Earth and asteroid, and the illumination conditions on the asteroids are presented through the mission timeline.

Furthermore the relevant facts brought forth by those factors are highlighted.

16.2.1 Distances to Sun and Earth

Distances to Sun and Earth widely change through the mission, as can be seen in Figure 16-1 (where vertical dotted lines represent arrival to asteroid system, nominal end of mission and foreseen extended end of mission), but more important is to highlight the following facts:

- During the nominal operations phase, favouring the design, distance to Earth is at its minimum (below 0.5AU) and so is distance to Sun (below 1.5AU).
- Nevertheless these figures degrade towards the end of the mission, which is relevant considering any delay or optional mission extension.

- During the interplanetary cruise phase, the spacecraft reaches the furthest distances to Earth (up to 3.2AU) at the same time as the furthest distances to Sun (up to 2.2AU).

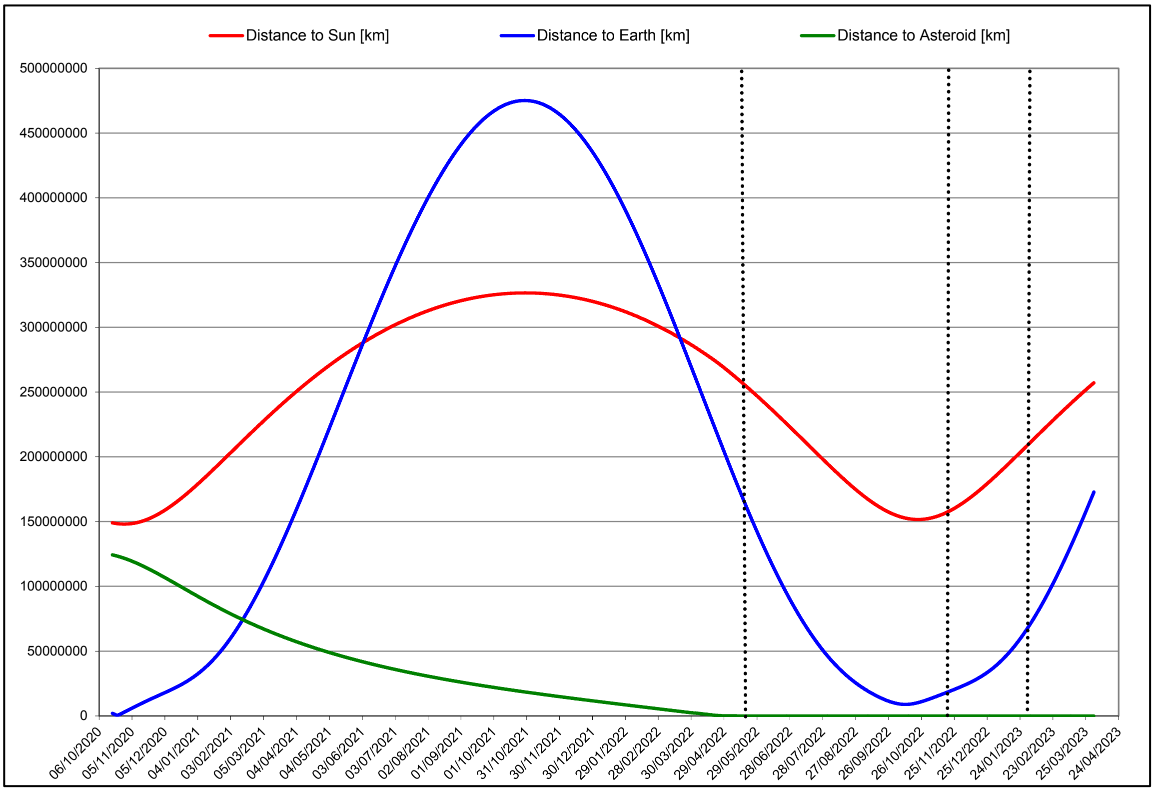

Figure 16-1: Distances from Spacecraft

This extreme range of operation conditions, depending on power and communication needs at each phase, is likely to shift the driving case from one phase to another.

16.2.2 Earth Solar Elongation and Phase Angle

Similar to the spacecraft distances to Sun and Earth, also their positions relative to the spacecraft widely change through the mission.

This can be seen in Figure 16-2 (where vertical dotted lines represent arrival to asteroid system, nominal end of mission and foreseen extended end of mission) by means of the Earth solar elongation (Sun-S/C-Earth angle, thick blue plot) and the Earth phase angle (Sun-Earth-S/C angle, thin blue plot).

In particular the following facts are relevant to highlight:

- During the interplanetary cruise phase, Sun and Earth tend to be aligned towards the period when the spacecraft is at its furthest distance from both (Earth solar elongation near to zero).
- During this period of alignment, Earth is behind the Sun as seen from the spacecraft (Earth phase angle near to zero) which prevents following the spacecraft or any possible communication with it.

- During the nominal operations phase, the angular distance between Sun and Earth as seen from the spacecraft changes from 20° up to 70°.

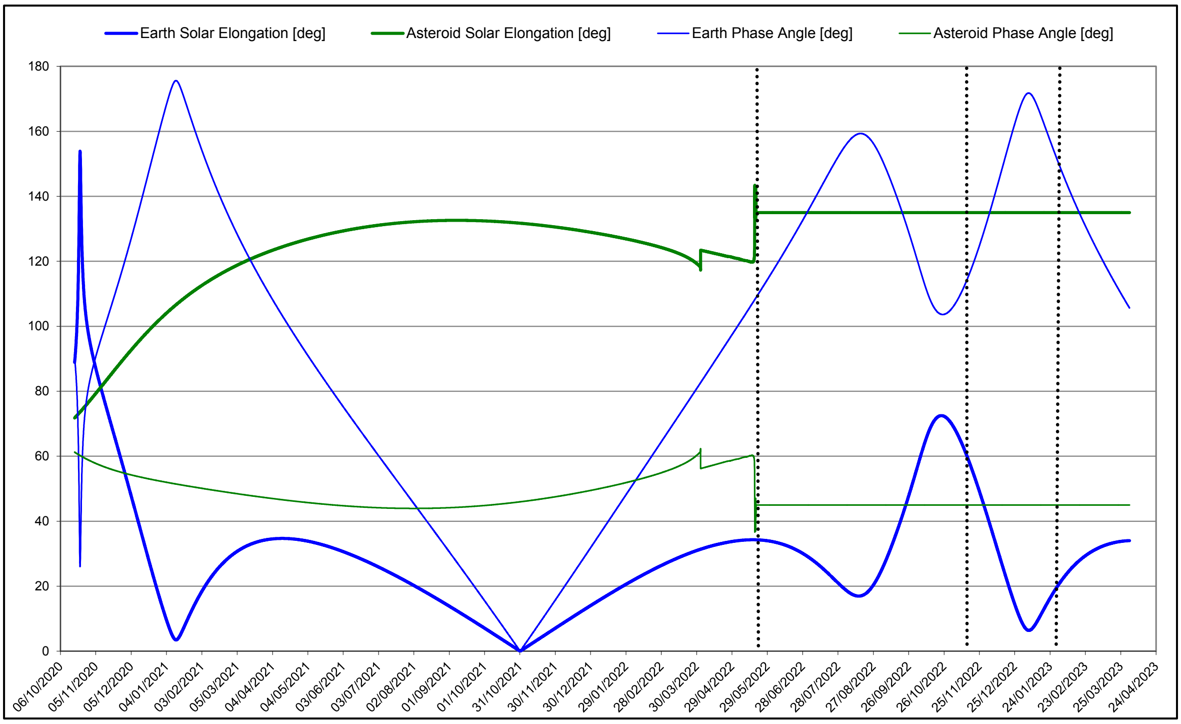

Figure 16-2: Solar Elongations and Phase Angles

In order to cope with these differences, depending on power and communication needs at each phase, either a flexible spacecraft configuration or a pointing strategy is required. The chosen approach is likely to shift the driving case from one phase to another.

16.2.3 Illumination Conditions at Asteroid System

Multiple factors, most of which are still uncertain, affect the illumination conditions on the surface of the primary and secondary asteroids so, in order to perform an initial analysis, the following assumptions have been taken:

- Primary asteroid rotation axis orientation and period are known
- Secondary asteroid orbit radius and period are known
- Secondary asteroid orbit plane in the equatorial plane of the primary asteroid
- Secondary asteroid rotation axis orientation same as the primary
- Secondary asteroid rotation period same as its orbit period.

This implies that illumination conditions on the secondary asteroid can be assumed to be the same as for the primary just adding the effect of any possible eclipse caused by the primary on the secondary.

The first factor presented is the eclipse conditions on the secondary asteroid. And the second factor is the latitude zones for permanent illumination/darkness and those in between with a cyclic day/night variation for each rotation period.

Because of the assumptions given above, both factors depend only on the orientation of the asteroid system with respect to the Sun which can be seen in Figure 16-3 for one of the pole solutions (where vertical dotted lines represent arrival to asteroid system, nominal end of mission and foreseen extended end of mission).

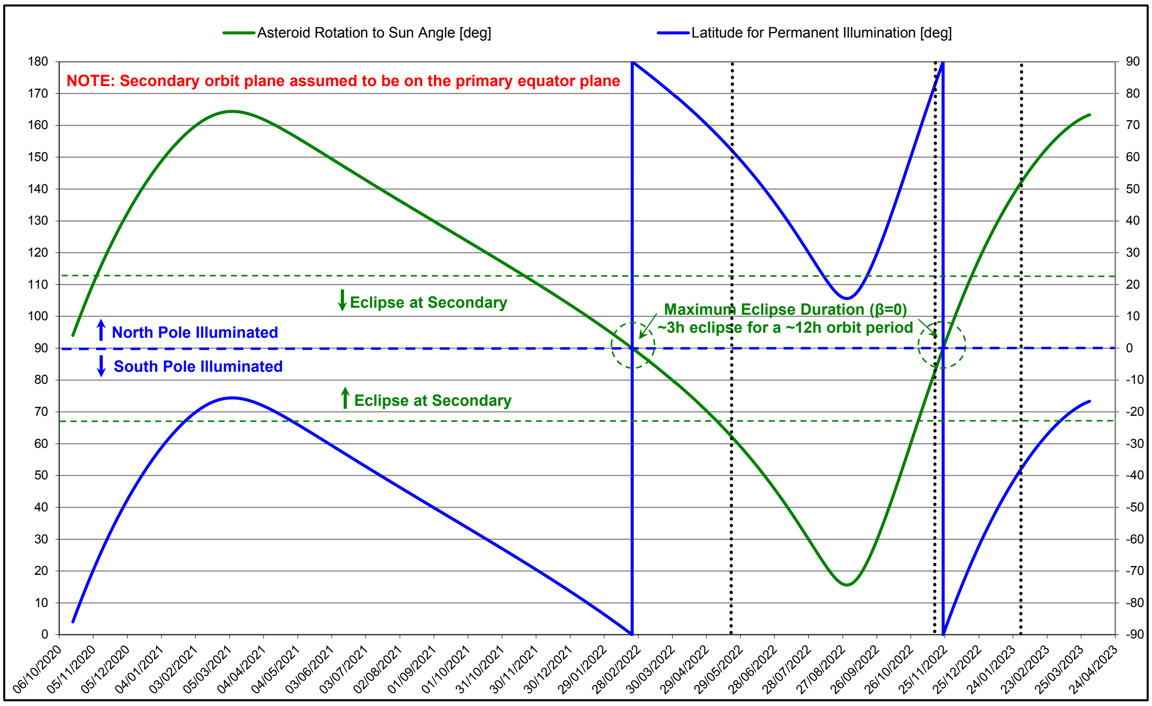

Figure 16-3: Illumination Conditions on the Asteroids' Surface

Due to the radius of the secondary orbit, there is a band (represented by the green dashed lines) for the asteroid rotation axis to Sun direction angle (green plot, left scale) that causes the secondary to cross the shadow cast by the primary during part of its orbit.

In particular, the secondary would be affected by eclipses only towards the end of the nominal operations phase becoming maximum (around one fourth of the orbit period) at the nominal end of mission (when the angle becomes 90°, zero orbit beta angle). And only towards the end of the potential extended mission it gets again out of the eclipse zone.

Regarding the illumination/darkness cycle on the surface of the asteroids, the latitude for permanent illumination (blue plot, right scale) has been derived from the rotation axis to Sun direction angle, in order to ease the interpretation:

- For positive values, any latitude above the plot latitude value up to the North Pole are permanently illuminated.
- For negative values, any latitude below the plot latitude value down to the South Pole are permanently illuminated.
- In both cases the opposite latitudes become of course permanently in darkness.
- Any latitude in between opposite latitude values has a day/night cycle variation for each rotation period.

This cycle is represented in Figure 16-4 (where vertical dotted lines represent arrival to asteroid system, nominal end of mission and foreseen extended end of mission) for some representative latitudes. In particular, the percentage of the rotation period that the surface is in darkness (night) for each given latitude is provided.

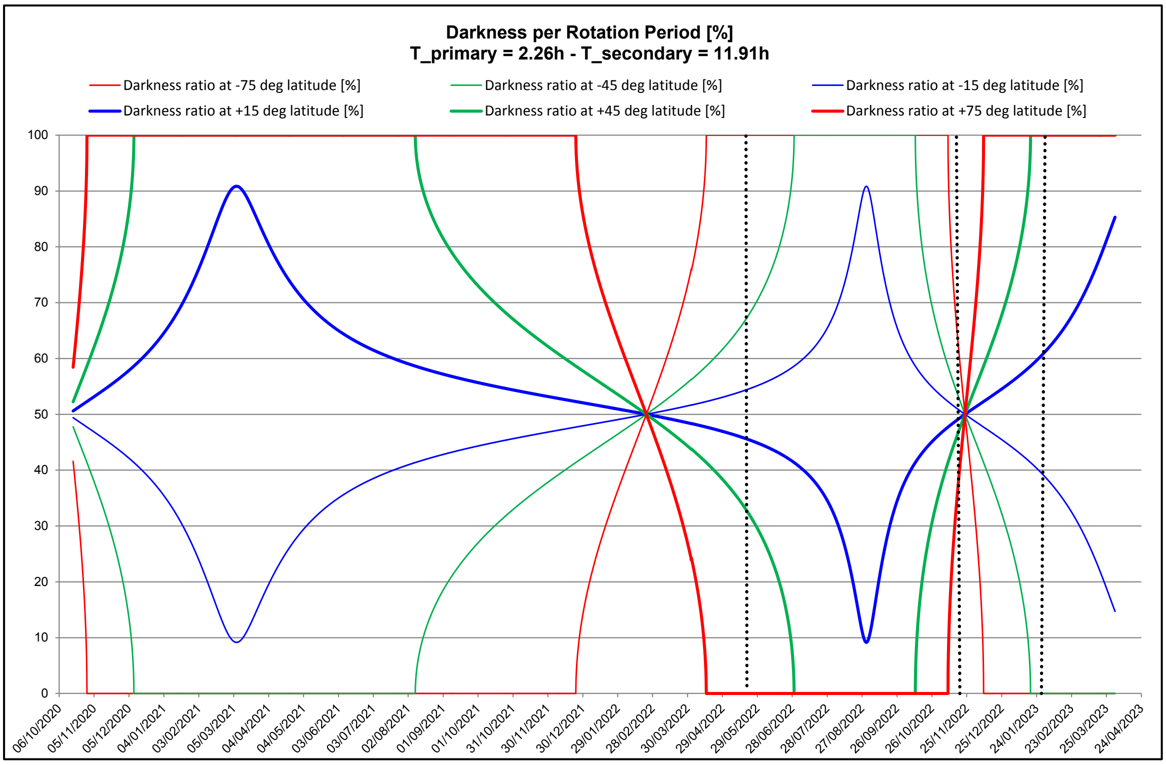

Figure 16-4: Illumination/Darkness Cycle at Asteroids' Surface

Both figures show that the only latitudes that have such day/night cycle for the whole duration of the nominal operations phase are those between ±15°.

Finally, as mentioned before, these two effects get combined for the secondary asteroid; meaning that some longitude bands (irrespectively of the latitude) could have an additional period of darkness due to the eclipse which can be used to artificially extend the duration of these periods if required.

These factors are likely to drive, due to remote sensing needs, the required position of the spacecraft relative to the Asteroid and Sun and, due to power and thermal needs, the landing latitude for the lander.

16.3 Design Trade-Offs

The baseline configuration design aimed for a solution without mechanisms, so fixed solar arrays, high gain antenna and payloads when possible, to be challenged against all the driving factors presented in the previous chapter.

16.3.1 The Solar Panels

The fixed solar panels have been placed such that their normal direction is +X in the spacecraft body-fixed frame, and deployed along the ±Y axis (minimum inertia around these axes) following a conventional configuration.

16.3.2 The Remote Sensing Instruments

To achieve optimal illumination conditions on the asteroids' surfaces and instruments resolution for remote sensing with the VIS during the nominal operations phase, the spacecraft is assumed to be flying in formation with the asteroid system such that the spacecraft lies between 10km and 100km away from the asteroid at a 45° phase angle position ahead of the asteroid on its orbit plane.

Then, in order to allow pointing the instrument towards the asteroid while being power optimised (+X towards the Sun), the instrument must be placed on any spacecraft panel but +X (+Z has been chosen) with their boresights pointing towards a direction 45° away from -X (-X to +Z quadrant has been chosen).

The partially defined configuration so far provides already some results as can be seen in Figure 16-5 (where vertical dotted lines represent arrival to asteroid system, nominal end of mission and foreseen extended end of mission).

Comparing the Power Optimised (+X towards the Sun, instrument align with asteroid direction) and Asteroid Tracking (instrument towards the asteroid, +X aligned with the Sun direction) pointing modes in terms of power generation (Solar Aspect Angle).

- During the interplanetary cruise phase, observing the asteroid requires operating with solar aspect angles below 25° towards the asteroid arrival, but are much bigger (up to 65°) at the beginning of the cruise phase. This is not an issue for the instruments that will not be observing in this phase due to the very large distance.
- During the nominal operations phase, by design, both pointing modes result in the same attitude and provide optimal power optimisation (zero solar aspect angle).

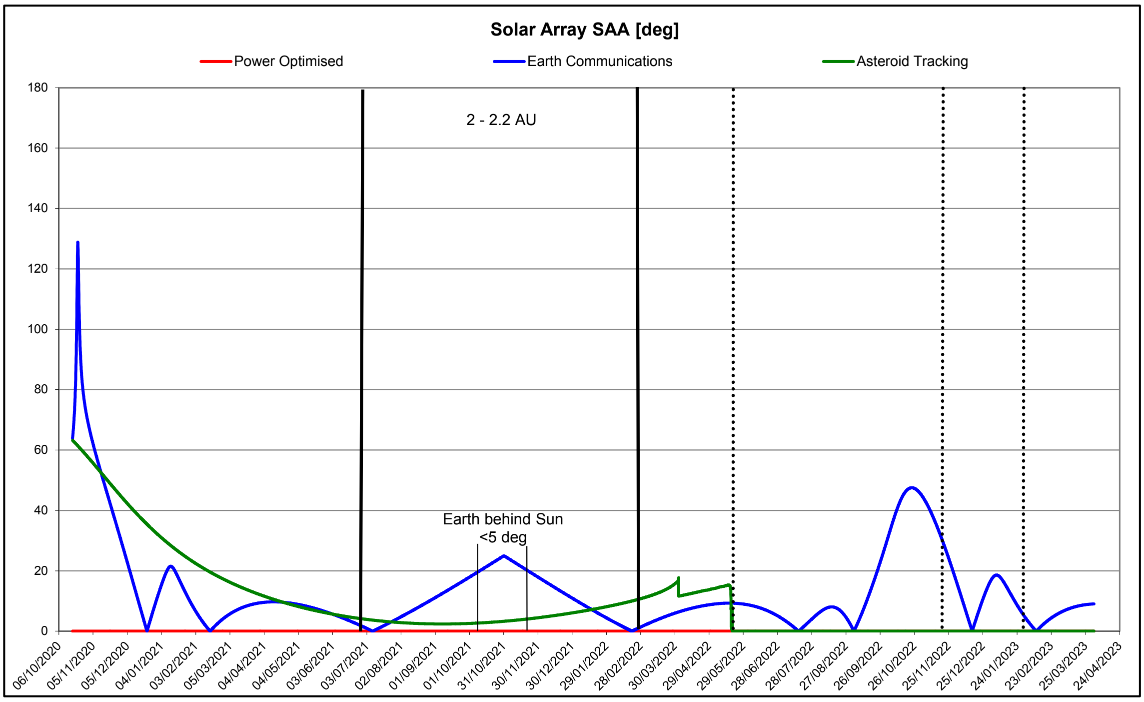

Figure 16-5: Solar Aspect Angels per Pointing Mode

16.3.3 The High Gain Antenna

To maximise power generation while pointing the high gain antenna to Earth, a fixed pointing offset with respect to the solar panels' normal can be chosen. Having another look at Figure 16-2, the angular distance between Sun and Earth (Earth solar elongation) covers the range 0° to 70° having the maximum range variation during the nominal operations phase.

Nevertheless a first estimation of the power needs for communications moves the driving case to the interplanetary cruise phase when the spacecraft is at its furthest distance from Sun and Earth. In particular, a solar aspect angle of maximum 20° is required to ensure continuous communication with Earth.

Additionally there is an exclusion zone of 5° around the Sun, when the Earth is behind the Sun (see Figure 16-5), preventing any communication. So this period can be excluded from the solar aspect angle constraint, due to the lower power needs if just in receiving mode or because, alternatively, power optimised attitude could be used.

Based on previous constraints, the high gain antenna has been placed on the +X panel with a fixed pointing offset of 25° with respect to +X on the +X to +Z quadrant (to keep, as much as possible, pointing manoeuvres around the minimum inertia axis).

Comparing now the Earth Communications (antenna towards the Earth, +X aligned with the Sun direction) pointing mode to the other pointing modes in terms of power generation (Solar Aspect Angle) in Figure 16-5.

- During the interplanetary cruise phase, the solar aspect angle remains below 20° except for the exclusion period when it is not applicable.
- During the nominal operations phase, the solar aspect angle can go up to ~45° but still well below the limits for this phase due to the closer distance to Sun and general lower power consumption needs.

This last aspect can be better understood in Figure 16-6 (where vertical dotted lines represent arrival to asteroid system, nominal end of mission and foreseen extended end of mission).

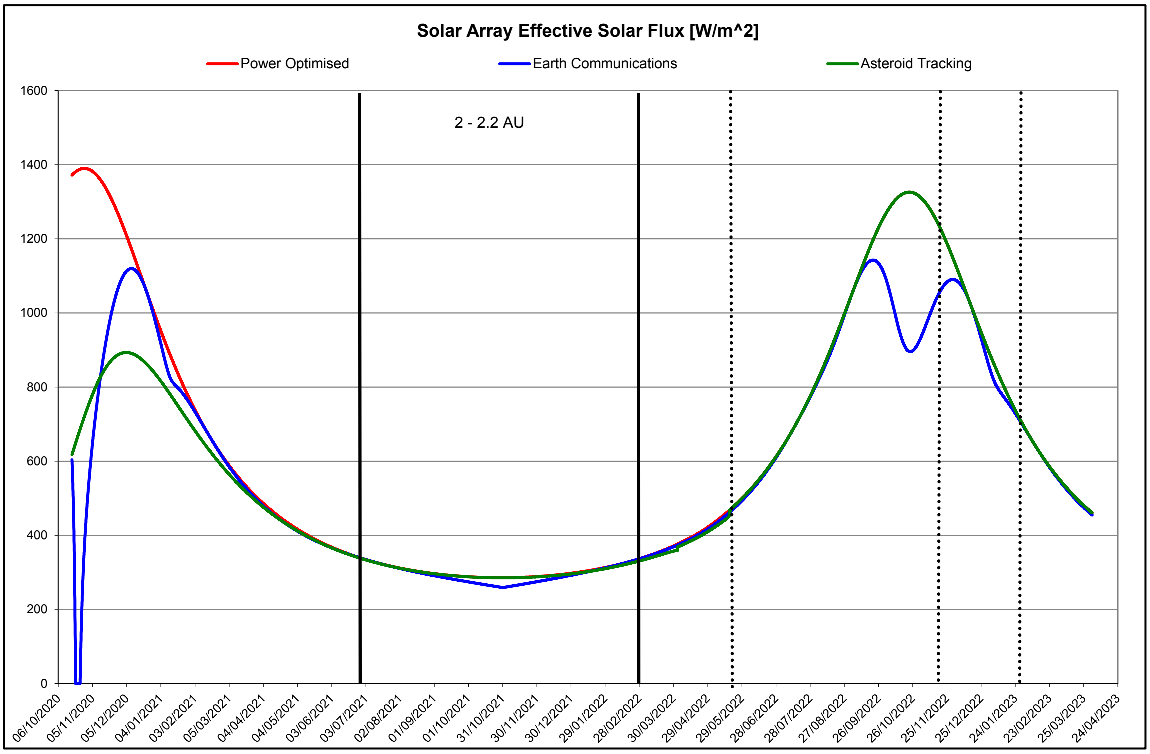

Figure 16-6: Solar Array Effective Solar Flux

In this figure all effects, solar array orientation for each pointing mode and distance to Sun, are combined to provide the actual effective solar flux available per square meter through the mission.

In particular how the power generation has been optimised for any pointing mode during the period the spacecraft is at its furthest distance from the Sun and Earth (between bold black lines) but minimising the impact while pointing to Earth for communications during the nominal operations phase when the spacecraft proximity to the Sun compensates the bigger solar aspect angles.

16.3.4 Final Remarks

Regarding the position of the spacecraft relative to the asteroid system, any other distance or position at 45° phase angle (any on the surface of a 45° semi-angle cone around the asteroid to Sun direction) can be chosen without any impact on the configuration.

This flexibility can be used to observe the asteroids under different illumination conditions for the VIS, resolution or to improve visibility of the poles for VIS; TIRI, HFR, and to operate the bistatic LFR with different aspect angles with respect to the local vertical of the landing site, through the asteroid..

It is important to stress that the asteroids' poles are not necessarily oriented perpendicular to the asteroid system orbital plane, and in that case moving the spacecraft up and down off the orbital plane might not provide better visibility of the poles.

Furthermore, it is important to realise whether improving the visibility of the poles might require bringing the spacecraft to positions relative to the asteroids out of the 45° phase angle cone.

This would imply, for the described configuration and spacecraft operation concept, that some margin on the maximum solar aspect angle allowed during the nominal operation phase shall be maintained, even if the margin is not anymore driven by communication needs.

Regarding the benefit of using a steerable antenna to avoid the need for a pointing strategy trading-off science and communication slots, it is important to highlight that the pointing range to be covered during the nominal operations phase is -60° to 0° azimuth (in the XY plane measured from +X positive towards +Y) and -10° to +60° elevation (out of the XY plane positive towards +Z).

Finally regarding the illumination conditions of the asteroids' surfaces, it is important not to overlook that for for the pole solution that is closer to the ecliptic, latitudes below -15° will be dark for the whole nominal operations phase becoming illuminated only early after the nominal end of mission and for the rest of the extended mission.

16.4 3D Visualisation

The visualisation needs covered during this study were mostly related to understanding the impact of the alternative spacecraft trajectories relative to the asteroids system on payload operations and spacecraft configuration.

In particular, illustrating for each configuration trade-off and pointing strategy the relative positions of Sun, Earth and asteroids with respect to the spacecraft, and the asteroids' observable areas and illumination conditions from the spacecraft position.

For this purpose, three views have been defined:

1. **Central body view**. Asteroid system centred view in an asteroid orbit fixed frame (+X is the Sun to primary asteroid direction, +Z is orthogonal to the asteroid orbit plane, and +Y orthogonal to the previous ones and towards the primary asteroid velocity).
2. **Spacecraft view**. Spacecraft centred view in the spacecraft body fixed frame.
3. **Instrument view**. Simulation of the scenes observed by remote sensing instrument's depending on their Field of View (FOV) sizes.

These views can be all seen in Figure 16-7 for the spacecraft nominal operations position 35km away from the asteroid system at 45° phase angle (angle between asteroid to Sun and asteroid to Spacecraft directions) and ahead of the asteroid on its orbital plane.

The red, green and dark-blue arrows represent respectively +X, +Y and +Z body frame axes for the spacecraft and the primary asteroid, while the yellow, light-blue and grey ones represent respectively the directions towards the Sun, Earth and primary asteroid. Note that the spacecraft is represented out-of-scale to help the visualisation.

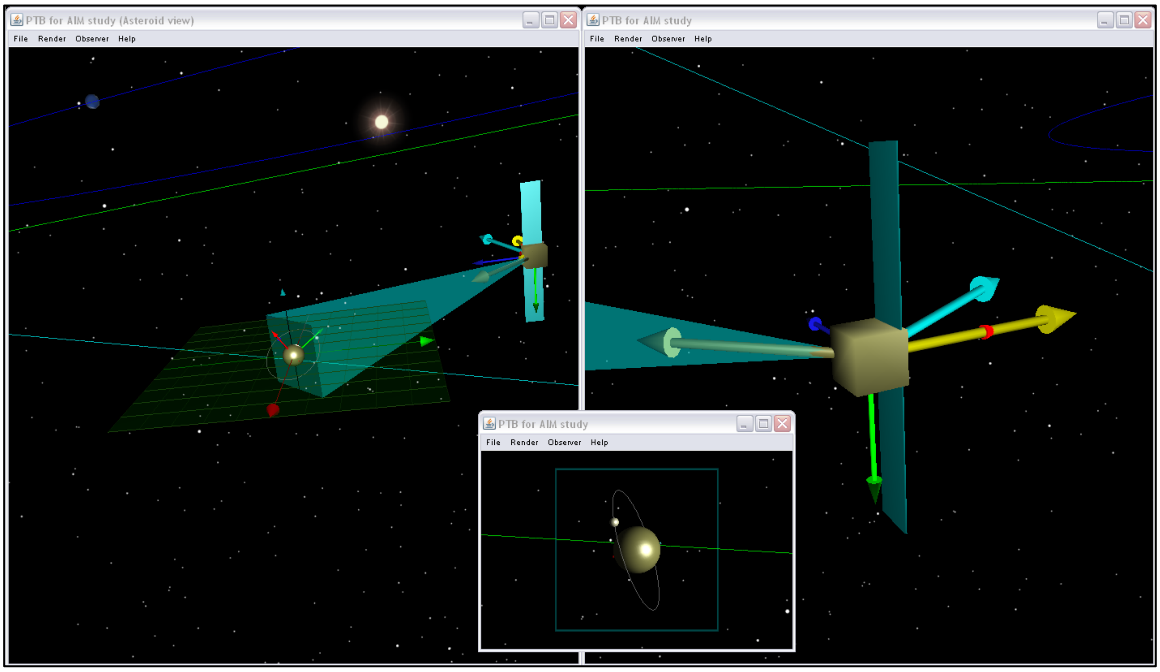

Figure 16-7: Central Body, Spacecraft and Instrument Views

During the nominal operations, as highlighted in section 16.3.2, the asteroid system can be observed while keeping power optimised attitude (spacecraft red and yellow arrows aligned). Nevertheless due to the asteroid system orientation, visibility and illumination conditions only allow to access the northern hemisphere of the asteroids.

In order to improve visibility of the poles or observe the asteroids under different illumination conditions, the spacecraft must manoeuvre to other positions relative to the asteroid system.

As an example, the spacecraft is represented in Figure 16-8 at the same distance but 30° phase angle and out of the asteroid orbital plane. From this position the North Pole is directly visible (blue dot) but, in order to maintain the asteroid system in the FOVs, it is not possible to keep the power optimised attitude (spacecraft red and yellow arrows not aligned anymore).

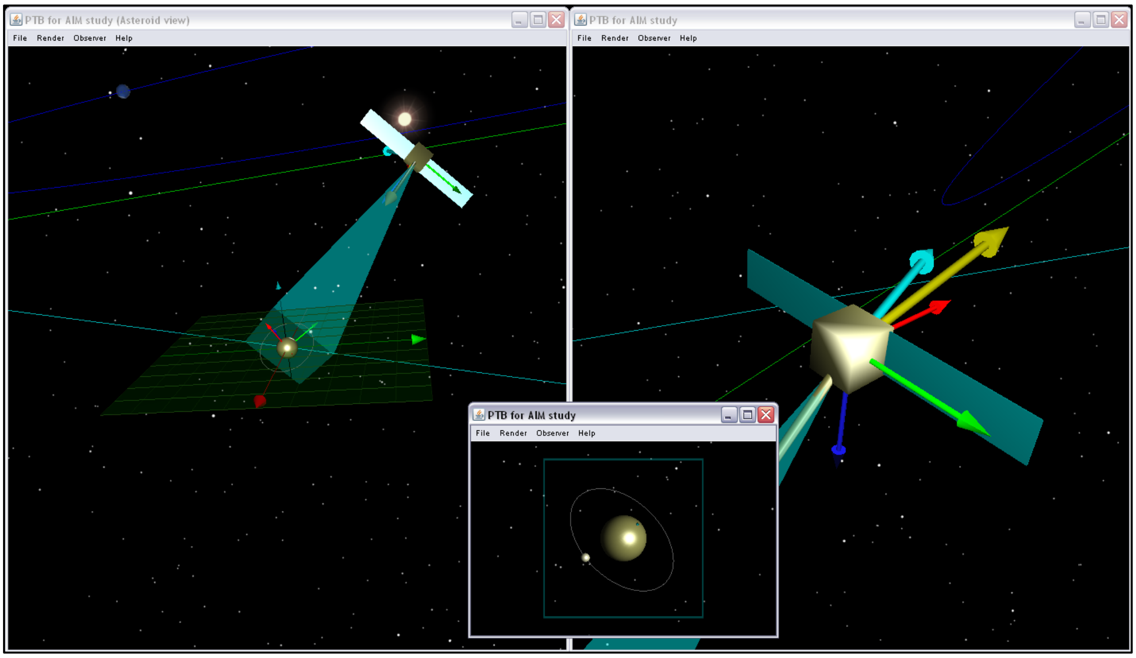

Figure 16-8: Spacecraft Position to Observe the North Pole

As a final example, the spacecraft is represented in Figure 16-9 at the nominal operations position but during communications pointing. The high gain antenna is pointed to Earth (spacecraft red beam and light-blue arrows aligned) but then power optimisation cannot be reached (spacecraft red and yellow arrows not aligned) nor can the asteroid system be observed (asteroids out of the FOV).

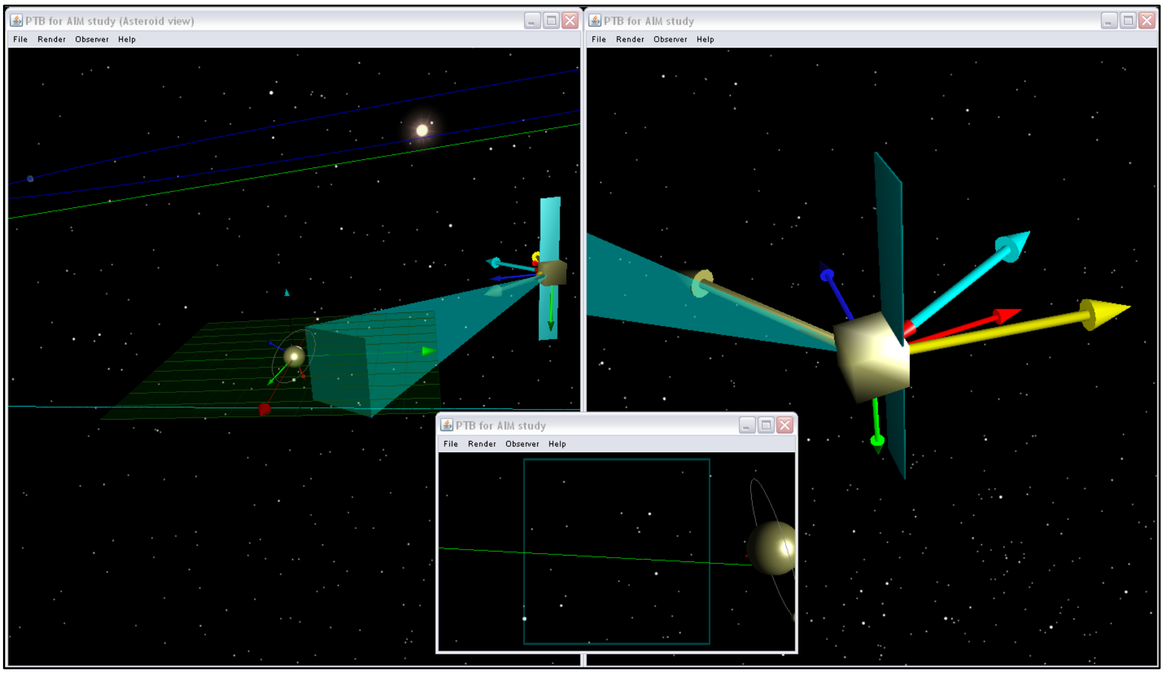

Figure 16-9: Spacecraft Attitude during Communication Periods

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17 GROUND SEGMENT AND OPERATIONS

17.1 Requirements and Design Drivers

Following launch, the AIM spacecraft will be directly injected into its deep space transfer orbit by the final stage of the launcher composite before separation. A launcher injection correction (LIC) manoeuvre will be required by launch plus seven days at the latest (i.e. at the boundary of the Earth's sphere of influence). The fixed (non-steerable) solar arrays will be automatically deployed following separation and attitude acquisition and no other appendage deployments are required for platform health or operations. There is a fixed HGA and LGAs for X-band communications.

It must be noted that Operations strongly recommends against having a fixed HGA as well as fixed solar arrays for an asteroid approach mission. This design has a significant impact on the operability of the spacecraft and must be reconsidered in the next phase.

The only deep space manoeuvre (DSM) is required at launch plus 2 months with no further orbit transfer manoeuvre (e.g. planetary swing-by) required before asteroid rendez-vous at DSM plus 17 months (launch plus 19). There will be a superior solar conjunction at launch plus 13 months / arrival minus 6 months where the Sun-S/C-Earth angle will be less than 3° (cut-off limit for routine commanding) for less than 1 week. AIM is projected to arrive at impact minus approximately 5 months.

AIM will co-fly with the Didymos binary system, initially at 35km separation and then at 10km. At no point is AIM required to orbit the primary element or the system as a whole. The asteroid approach and co-flying orbit insertion sequence has been designed by Missions Analysis according to the methodology used for Rosetta. It will be a series of 5 burns executed 1 week apart and reducing in magnitude over the sequence. The final arrival burn has been planned for one week in advance of the required arrival date to allow for post-insertion orbit corrections.

The six weeks of the Early Characterisation Phase will be performed at 35km separation at which one station keeping manoeuvre every five days will be required to remain within the deadband for science operations. The scientific results from this phase are required to be delivered to NASA by DART impact minus 11 weeks TBC. Following this, AIM will approach to 10km for the Detailed Characterisation and Lander Deployment Phases at which one station keeping manoeuvre every 2-3 days will be required. The current consideration of the Customer is that the MASCOT lander can be deployed from the 10km orbit such that no close fly-by of the system is required.

For a two week period around the asteroid impact by DART, AIM will withdraw to an observation distance of 100km although there is no requirement to observe the impact nor is there a requirement to relay data from the impactor. Following this phase, AIM will return to the 10km orbit and continue science operations (re-characterisation to determine the effect of the impact) until mid-November 2022. There is an option for an extension until the end of January 2023.

17.2 Assumptions and Trade-Offs

17.2.1 AIM as a Small Science Mission

As emphasised by the Customer, the mission has a clearly defined and restricted overall budget, therefore, the Small Science Mission operations concept shall be applied. There are a number of elements to this concept and their applicability to the AIM mission is discussed hereafter.

The systematic, extensive and rigorous use of standards, also in S/W interfaces to ground, enables the extensive re-use of existing ESOC and ESTRACK infrastructure (all of which is ECSS compliant) to the extent of only having to configure stock deliveries or installations when the mission is fully compliant. This will be assumed for AIM.

Heritage in the space segment (enabled by a series of spacecraft reusing the same bus, for example), where background knowledge is already existing in the MOC, allows the reuse/adaptation of all or a subset of the procedures and operational databases, requires less review activity and less training, and, under favourable conditions, allows the sharing of manpower/systems with similar ongoing missions. There appears to be little heritage in the current AIM spacecraft, although it is assumed that other characteristics of the mission will allow some manpower sharing with other missions (see below).

The commonality/synergy of AIV and operations tools enables the paradigm of “one-off development, multiple applications” (e.g. SVF can be used as a simulator and EGSE doubles as a control system), however, this has to be carefully balanced with the points above covering the reuse of MOC infrastructure and training needed. A possible compromise is to have the MCS and Mission Planning System procured by the MOC whilst the simulator is delivered by Industry. This approach was followed previously by SMART-1, for example, for which project-wide cost savings were made by having the simulator development done by the Prime contractor for the satellite (part of the development of five systems in total, including the EGSE). On the one hand, this was a cost which was not borne by Operations as is usually the case but, unfortunately, the simulator was not a priority for the Prime until they needed it which meant that it was delivered late to ESOC and with numerous interface bugs that needed debugging by the FCT. This was compounded by its hardware-plus-software architecture which resulted in no second simulator being available for dedicated MCS testing. On the plus side, there was closer synchronisation with later developments and an increased confidence in the testing results (of OBSW patches, for example). Given the ongoing developments in the Ground Systems Test and Validation infrastructure (GSTVi) at ESOC and the shortened implementation phase (which may well result in the need of OBSW updates during the cruise) it is assumed that it is a viable option for AIM to allow the spacecraft Prime to develop the simulator in parallel with the EGSE (and take advantage of the later close synchronisation and increased fidelity of the simulator) although the extent of its benefit to the project costs (if there are any) can only be TBD at this stage.

Similarly, further SMART-1 ground segment savings were realised by using the same database at ESOC and Industry to remove the need for DB conversions and the risk of inconsistencies. This saved a lot of effort (especially as there was no dedicated Analyst in the FCT) but it did result in a lot of DB releases in the run-up to launch.

Manpower cost is directly proportional to the duration of the mission (its preparation and execution), so a short timescale is advantageous. The AIM project itself is limited to a short 3.5 years preparation and 2 years execution.

Best effort commitments on operational requirements need to be acceptable to the mission as shown, for example, in reduced availability and data return requirements. This enables the provision of a lower level of service which, in turn, can allow such characteristics as fewer engineers, limited contingency preparations, office-hours only support or less redundancy in the ground systems. The AIM mission profile appears to be compatible with this: arrival at the asteroid before DART, remain in a co-flying orbit, provide a set of observation data by a given deadline, continue to characterise the binary system before and after the impact but with no requirement to observe the impact, and all with an apparently generous margin in the science timeline.

A small mission should have no real-time operations in flight, only preplanned operations. At the same time, there should be minimal mission planning required. AIM appears to have plenty of margin in its science timeline and no significant mutual operations constraints (excluding those currently imposed by the fixed appendages).

There should be no time critical operations, especially during the LEOP, as this typically requires significant effort in the preparation phase. The criticality of the AIM solar array deployment is TBD.

The spacecraft should be simple to operate and the routine operations should, ideally, be repetitive. Even with the burden of its fixed solar arrays AIM has no apparent resource limitations, although the additional burden of the fixed HGA raises significant questions with regards to its operability. Assuming this can be addressed, simple platform operations are assumed. Likewise, simple instruments operations are assumed as well as no operational responsibility for anomaly investigations on the payload. The routine deep space operations will be repetitive and even the asteroid operations appear to follow a cyclic nature.

Limited and simple operational interfaces need to be designed into the small mission's concept, especially with regards to mission planning (which must be controlled by the operations entity in order to reduce interfaces) and science data delivery. In addition, if interface or line costs can be reduced, science data should not be transferred over any operational network after reception at the ground station but rather left on a server at the ground station for later recovery by the customer. This last element is not really applicable to AIM as the science data is also the navigation data (VIS data sets) and cannot be entrusted to non-operational networks. But it is assumed that the AIM science operations will be fairly static (well defined in advance and not likely to change much) such that general requests can be made by the science community but then implemented according to rules owned by operations.

Managerial interfaces with 3rd parties also need to be kept to a minimum and kept as informal as possible.

Small missions can greatly benefit from the use of technical advances to reduce operational workload (eg. data transfer protocol, file management for science data, use of GPS for reduction of Flight Dynamics support etc.) with the proviso that entirely new developments in the space or ground segments increase cost and risk for the mission that introduces them. Simple processes on-board are simple to validate and simple to automate around on-ground. At the same time, a valuable tool for operations simplification and automation is a reliable and flexible On Board Control Procedure (OBCP) function. This will be assumed for AIM.

Small missions should limit the level of service required from Flight Dynamics. The navigation requirements of the cruise phase and asteroid characterisation phases of AIM are comparatively relaxed but, nevertheless, AIM is still a deep space, asteroid rendez-vous mission.

There should be high spacecraft autonomy (target < 1 month for interplanetary spacecraft) and survivability (such that there is an indefinitely stable, i.e. 'routinely' useable, safe mode which Operations does not have to protect against). This will be assumed for AIM.

Excluding the needs for navigation, there should be a reduced duration and frequency of ground contacts (e.g. achieved through high efficiency due to high data rates, HK compression, etc.). Eight hours per day at the asteroid will be assumed for the CaC.

There must be a light documentation requirement (reporting and other quality tracking activities largely reduced) otherwise the small team of engineers will be expected to do the work of a large team of engineers.

A small mission implies a small team plus additional support from ESOC multi-mission staff and resources to cover workload peaks (SVT and LEOP). In the extreme, if mission complexity is low, mission operations can be taken over entirely by an existing mission team as an additional task. AIM will assume the sharing of engineers with other missions. The option of a complete take-over by an existing mission can be investigated during Phase A/B1.

17.2.2 Deep Space and Asteroid Navigation

There have been questions about the requirements for navigation during the cruise and asteroid phases in the drive to reduce costs by reducing ground station time.

Doppler, which is the fundamental component of navigation, can be had for free when receiving TM via a coherent link such that, together with ranging, a single 8-10hrs pass every month during a coasting cruise is sufficient for deep space navigation. This has to be traded-off, however, against the risk of a spacecraft safe mode during which the sun pointing attitude is maintained via thrusters. If this were to happen shortly after the pass, that would leave the spacecraft thrusting for an entire month with the possibility of it changing the orbit so much that even the spiralling search pattern of the ground station cannot find it again. The accepted compromise is one pass every week and will be used as the baseline for this mission during the Deep Space Cruise Phase.

As a general rule, RARR (Range And Range Rate (Doppler)) data is sufficient in tracking campaigns for DSMs and Earth swing-bys if regular coverage is possible in advance, whilst the addition of Δ-DOR data is strongly recommended for tracking campaigns around all other major events such as Planetary swing-by and orbit insertion. Before the decision on Δ-DOR can be taken, however, the mission trajectories need to be studied in more detail. Additional factors such as the size of (and, hence, fuel used by) the dispersion correction manoeuvre that follows a main event (e.g. manoeuvre or swing-by) that can be more accurately calculated when Δ-DOR data is included in the orbit determination, or simply the ground-based accuracy required for the start of a rendez-vous campaign given the sensitivity of the navigation camera, need to be considered. This report will make assumptions based on these arguments.

For the AIM asteroid approach and orbit insertion, Mission Analysis took the Rosetta comet approach as an example. This was a sequence of 8 manoeuvres over 2.5 months for which Δ-DOR was seen as an enhancement rather than the baseline (although, in reality, these operations would not have taken place without the inclusion of Δ-DOR data). More important to the navigation is the imaging of the body from the spacecraft. For AIM, this will first be done by the star tracker and then by the VIS instrument acting as the navcam. TBD if AIM will be able to see the asteroid before commencing the arrival sequence.

Based on this, and the orbit insertion of Venus Express, and the planned orbit insertion of ExoMars, the following tracking campaign for the AIM rendez-vous and orbit insertion sequence has been defined purely for the purposes of costing, it is not an attempt to define the real campaign. Such a campaign results in an additional 64 hours of ground station time and no other discernable costs. As directed by the Customer, the asteroid has been well enough observed from the ground, so operations to progressively approach the asteroid and measure its gravitational constant are not required. A test manoeuvre two weeks prior to the first insertion burn (as done with Rosetta) will be assumed in the mission timeline.

Start	**End**	**Manoeuvre**	**RARR**	**Δ-DOR**
T-42days	T-28days	#1	4ppw	1ppw
T-28days	T-21days	#2	4ppw	1ppw
T-21days	T-14days	#3	7ppw	2ppw
T-14days	T-7days	#4	7ppw	4ppw
T-7days	T+0	#5	7ppw	4ppw
T+0	T+7days	Insertion corrections as required	7ppw	4ppw
T+7days	…	…	7ppw	-

Table 17-1: Asteroid approach and insertion example

It is assumed that the tracking campaign for the DSM at launch plus two months can be done with RARR data alone, especially because it will be done at (used as the target for) the end of the Commissioning Phase during which regular and extensive tracking data will have been collected.

This section has focused on identifying the minimum amount of ground station coverage possible to satisfy the navigation requirements, but this does have to be tempered somewhat by the requirements of operations. In particular, for an asteroid rendez-vous mission, the operators need time at the beginning of the Operation's Phase E to build up flight experience and confidence with the spacecraft. This involves extended coverage (with respect to the small amount of coverage previously identified for a deep space cruise) during which time as much data as possible can be collected on the newly launched spacecraft, and any anomalies or general characteristics can be given time to show themselves and be analysed. For this, an average of 8 hours coverage per day, 7 days per week will be assumed throughout the Commissioning Phase, plus an additional one month of Initial Cruise Phase (ICP) with 8 hours per day, 5 days per week. This is, nevertheless, much less than for a full-scale mission.

17.2.3 Deep Space Hibernation

There is no benefit from an Operations point of view to have a Hibernation Phase, quite the opposite. It was necessary for Rosetta because there would not have been enough power for the nominal operations of the spacecraft at its maximum separation from the Sun but there is no such constraint for the AIM mission so a Deep Space Hibernation Phase should not be considered further.

17.3 Baseline Design

Operations Preparation will begin at Launch - 3.5 years (not including Mission Analysis) with the SOM and three SOEs initially spending only 50% of their time on the project. Synchronising with the current Programmatics schedule, the GSRQR must occur early-Q4 2017 and the GSDR a year later, leaving just 2 years for the Phase D (Production and Validation) activities. One of the first activities will be a heavy pruning of the document tree and a rationalisation of the ground segment requirements.

The design of a small mission FCT is highly speculative at this stage. The intention here is to create a model for costing which represents sub-teams in the FCT with their own internal backup and variable manning according to the mission phase. In practice, whole engineers will not be added to, or removed from the mission but rather have part of their time assigned from other missions either in development or routine ops. And even this is an assumption as it depends heavily upon the status of the other missions to be shared with at the time. Further analysis is required.

There will be one SOM and the FCT will be built up to the following composition of SOEs to run from L – 1.5 years until the end of mission:

- Two full-time for AOCS and GNC
- Two full-time for Power, Thermal, Comms and Propulsion
- Two full-time for Data Handling, Payload, Mission Planning and Analyst duties (e.g. MIB management)
- One SPACON contributed to the pool of Interplanetary mission SPACONs.

Commissioning is planned to run over the first two months to allow office-hours-only working and, thus, avoid an excessive workload on the operations teams. The daily interaction with the spacecraft over this phase, and the Initial Cruise Phase, also allows time for the spacecraft to “show its character” before the long deep space cruise with little ground contact.

For the period of two months up to, and including, LEOP there will be an increase in the size of the operations teams to allow for the training and execution of a 2x12 hours on-console support pattern for LEOP. This will be achieved through the temporary support of engineers from other missions who will be given high-level training and validated for Team-B activities.

Following the completion of LEOP there will be no further use of critical operations infrastructure at ESOC. Throughout the cruise phases and the approach phase, the different Deep Space Antennas will each be used in turn (as best as possible) to maintain their mission readiness and thus avoid the need of dedicated tests in preparation for the rendez-vous.

There will be a strict adherence to ECSS standards and a maximum (exclusive) use of the generic ESOC Mission Operations Infrastructure and EGOS Systems, with the simulator delivered by the Prime. Beyond prime and backup chains there will be no further redundancy in the Data Systems. There will be a focus on the development of Routine Operations and their automation (such as TM checking and remote engineer notification) with limited contingency operations preparations. If possible, this will be compensated for by a guaranteed block of Industry support across the mission lifetime.

Phase	Duration	Comments
LEOP	3 days	~24h coverage with 3 DSAs
Commissioning + DSM	2 months	10hrs station coverage on average per day but only 8x5 Commissioning operations
Initial Cruise Phase	1 month	8hrs/day 5days/wk station coverage Time to build experience/confidence with the spacecraft
Deep Space Cruise Phase	11 months	Asteroid/arrival ops. preparations continue One 8hrs pass / week
Asteroid Approach Phase	4 months	Small Sims campaign 8hrs/day 4days/wk station coverage First navcam images at some point with orbit refinement Instrument commissioning well before first arrival manoeuvre Test burn two weeks before the first burn
Asteroid Arrival & Initial Ops Phase	2 months	Dedicated navigation campaign (with DDOR, Table 17-1) around the arrival sequence Daily coverage
Asteroid Ops	5.5 months	8hrs Daily coverage Routine Ops Offline mission
Disposal		Graveyard orbit and passivation? Leave it where it is?

Table 17-2: Flight Operations Timeline

For the Routine Phase of asteroid operations there will be a well-defined sequence of operations to follow and no need for exceptional working practices, i.e. office-hours only support, no short-term reactions to scientific requests.

The CubeSats will be treated as blackbox payloads and their operations limited to switch-on, switch-off, release and data relay. MASCOT-2 operations will be similar although more care and attention will naturally be given to its release (from the routine operations 10km separation orbit) for landing on the secondary body.

The timeline of the sequence of orbit manoeuvres required for the one week of High Frequency Radar observations is compliant with the required frequency of station keeping manoeuvres at 10 km asteroid separation such that no, or limited, additional effort from the FCT is expected, i.e. overall, there will be a consistent level of operations intensity throughout the asteroid phase. This is, of course, assuming a strict adherence to the directive that the DART impact need not be observed.

18 PROGRAMMATICS/AIV

18.1 Requirements and Design Driver

The main requirements and design drivers for the Asteroid Impact Mission from a programmatics point of view are:

- Phase A/B1 start 23/02/2015
- Launch window 17/10/2020 – 06/11/2020
- Low cost
- Equipment and applied technologies shall have reached at least TRL 6 at the start of the mission implementation phase (B2/C/D).

18.2 Assumptions and Trade-Offs

- Launch mass incl. adapter ~ 817 kg
- Adapter incl. separation system ~ 110 kg

An extended payload with a mass of about 74.4 kg (incl. margin) shall be accommodated:

- Lander MASCOT-2: (292x278x197 mm) 13.0 kg
- COPINS: 13.2 kg
- VIS Camera: 2.4 kg
- Thermal Infrared Imager: 3.63 kg
- Radar components (1.7 + 1.2 + 2.0 = 4.9 kg)
- OPTEL (optical terminal): 39.3 kg

The optical terminal is a new development which has been assessed in much detail and the development plan foresees completion of EQM testing end of first quarter 2018.

Most other equipment and components used will be “off-the-shelf” with a TRL of 8 or higher. Only the following equipments have a TRL below or equal to 4:

- Deep Space Transponder
- High Gain Antenna
- High Power Amplifier

These equipments should achieve TRL 6 before the start of Phase B2, but, in case this cannot be achieved, high-TRL back-up solutions are already identified.

The structure of the spacecraft will be new, but it will be a conventional cube consisting of honeycomb structural panels. A lander based on MASCOT from DLR, including its own separation system, will be mounted on the surface of the spacecraft. The lander is considered like a COTS equipment and therefore not discussed in detail in this report. Only the need date will be highlighted in the schedule.

The AIM mission is a European project with only European companies and agencies involved. The spacecraft will be launched from Kourou with a Soyuz.

18.3 Model Philosophy

A protoflight approach is proposed. This is considered feasible because a simple design from thermal and structural point of view is anticipated. The structure or its components shall be acceptance-tested before being assembled into the PFM.

An Avionics Model (AVM) shall be built for functional testing of the avionics architecture, software and test procedure debugging, troubleshooting in support of PFM and as reference ground platform in support of flight operations. For the AVM, all electronic components important for the GNC and AOCS subsystem must be available including the camera of the payload which is used by the GNC.

18.4 Schedule

The schedule respects the planned start of Phase A/B1 (23/02/2015) and is arranged to result in a launch at the start of the launch window (17/10/2020) calculated by mission analysis.

Assuming typical phase durations, based on statistical evaluation of ESA satellite missions, the Phase A should have been started end of 2013.

However for small satellites like this, a reduced schedule is considered necessary as shown in Figure 18-1:

- Phase A duration is reduced from 12 to 9 month and Phase B1 duration from 9 to 6 due to the already relatively detailed concept.
- A short contract preparation of only 4.5 months is allocated (instead of 6 months). This is very tight but could be achieved if activities in Phase A/B1 are arranged such that the proposal for Phase B2/C/D is delivered for SRR or shortly after it. It helps that satellite and payload are rather simple and that only European companies and agencies are involved.
- The typical Phase B2 duration of 12 months is retained.
- The Phase C/D is short with 32 months instead of 36 months. This was justified by the rather simple design, the small size of the spacecraft, the assumption that the lander is nearly off-the-shelf and the fact that no complex instruments will be used.
- The extended payload suite of about 74 kg, compared to 15 kg in the VEGA launch scenario makes this the more challenging and imposes to plan already early four double shift integration activities.
- The contingency has been cut from 6 to 3 months to allow a start of Phase A beginning of 2015, retaining the desired launch date.
- However because the start of Phase A/B1 is now only 7 weeks later than in the original plan it is necessary that either phase B2 activities are advanced already during the negotiation period or Phase B2 is shortened.

The development of the Deep Space Transponder, for which TRL 4 is expected to be reached end 2014, and the High Power Amplifier, for which only TRL 4-5 is expected to be reached by beginning of Phase B2 remain critical. For the transponder it means that the development needs to be tightly followed and if possible accelerated with project funding. Otherwise it would be typically ready only mid 2020, which is too late for integration. The High Power Amplifier development is considerably behind schedule and is expected to reach TRL 4-5 only in about 2 years. Therefore, the project must by then fund an accelerated development and prepare for the possible use of the backup solution. The High Gain Antenna is not critical because it reached TRL 4 end 2013 and a protoflight model is expected to be ready in 2016. Additionally the European IMU expect TRL 4 at the end of 2015.

The development plan for the optical terminal, also shown in the schedule below, is compatible with the overall schedule, but with the restriction that TRL 6 is only reached in early phase C. This imposes that a waiver for the TRL requirement is approved during negotiation of the industrial phase (B2/C/D) and that a very well controlled development is demonstrated.

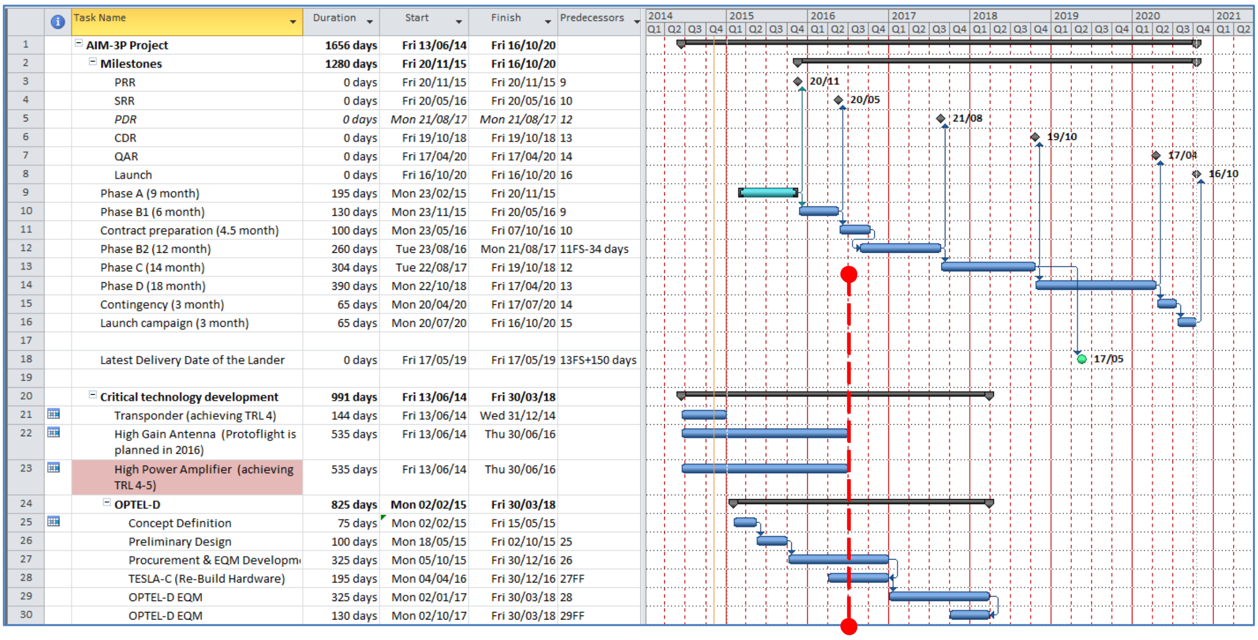
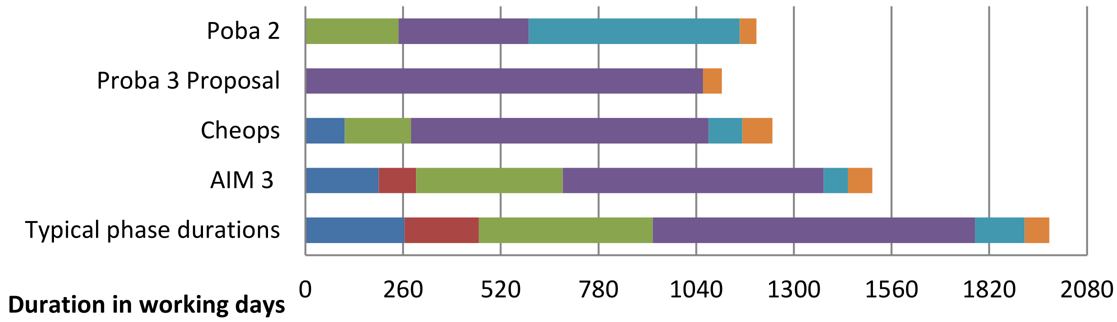


Figure 18-1: AIM3 schedule

Figure 18-2 compares the proposed phase durations of AIM3P with other projects. The durations of phases A and B for Cheops are shorter, but that project is based on an existing platform with only one new instrument. Also the Phase B and Phase C/D of Proba 2 were shorter than anticipated for AIM3P, but the Proba 2 platform was a largely recurring design from Proba 1. The large duration of the contingency shown for the Proba 2 bar is not the originally planned contingency but the difference between the expected and the actual launch date. The delay was caused by the payload. Proba 3 is an example of a new development, and for the phase C/D 50 % more time than for AIM3P is allocated. For the Phase A of Proba 2 and the Phases A and B of Proba 3 no information was available for this comparison.

This confirms that the schedule of AIM3P is very tight. To achieve the project in this short timeline it is necessary that any technology issues are addressed without delay.

Comparing Schedule of AIM-III with other Missions



	Typical phase durations	AIM 3	Cheops	Proba 3 Proposal	Poba 2
Phase A (working days)	264	195	105	0	0
Contract time (working days)	198	100	0	0	0
Phase B (working days)	462	390	176	0	248
Phase C/D (working days)	858	694	792	1058	347
contingency (working days)	132	65	90	0	561
Launch Preperation (working days)	66	65	80	50	44

Note: 260 working days are about 1 year elapsed time.

Figure 18-2: Comparison of AIM3 phase durations with other projects

18.5 Conclusions

The required launch date seems to be achievable, but only with a very tight schedule. This schedule allows no space for critical technical issues and long contract negotiations.

Only one contract negation phase can be permitted, i.e. a Phase A/B1 (as foreseen) and a Phase B2/C/D. No break between B2 and C/D can be accommodated.

In addition it imposes advanced Phase B2 activities already in the negotiation phase or a reduced Phase B.

A protoflight approach is proposed, anticipating a rather simple structural and thermal design.

Critical technology developments are identified, for which however backup solutions exist. In particular for the high power amplifier it must be decided before the start of the implementation phase whether it can be baseline without endangering the launch date, or if the backup solution shall be anticipated.

For the optical terminal a special permission is needed that TRL 6 will be only achieved early in Phase C. This requires particularly that a very well controlled development is demonstrated.

19 TECHNICAL RISK ASSESSMENT

19.1 The Technical Risk Assessment Process

Risk management is an organized, systematic decision making process that efficiently identifies, analyses, plans, tracks, controls, communicates, and documents risk to increase the likelihood of achieving the project goals. The procedure comprises four fundamental steps Figure 19-1:

- Step 1: Definition of the risk management policy which includes the mission success criteria and the severity & likelihood categorisations
- Step 2: Identification and assessment of risks in terms of likelihood and severity
- Step 3: Decision and action (Risk acceptance or implementation of mitigating actions)
- Step 4: Communication and documentation.

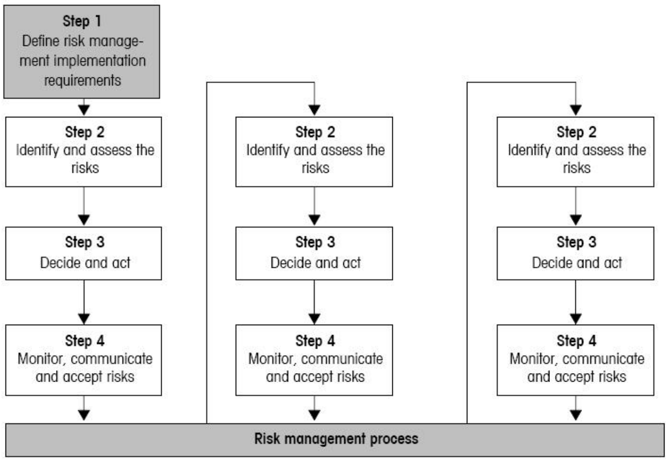


Figure 19-1: Risk Management Process

At this point it is necessary to define a clear risk management policy for AIM-3

19.2 AIM-3 Risk Management Policy

The CDF risk management policy for the AIM-3 study aims at handling risks which may cause serious programmatic/schedule, cost, safety, mission objectives, impact on the project.

The following actions have been carried out as a basis for the implementation of the risk management process:

- Identification of the AIM-3P project success criteria
- Establishment of a scoring scheme for the severity of consequences of undesired events affecting mission success criteria (1-5)
- Definition of likelihood or probability of occurrence levels (A-E)
- Establishment of a risk index scheme to denote the magnitudes of the risks of the various risks scenarios

- Establishment of criteria to determine the mitigating actions to be taken on risks
- Establishment of a method for the ranking and comparison of the risks.

19.2.1 AIM3P Mission Success Criteria

The AIM-3 project/mission is considered successful when the, programmatic/schedule, cost, safety, mission objectives are met. These are collected in Table 19-1:

Domain	Success Criteria
Programmatic/ schedule	P1. AIM3 mission Launch Opportunity date is late 2020 – arriving 2 months prior to DART impact P2. TRL 5 Expected at phase B1 P3. AIM 3 launcher will be Soyuz; use of a Vega combined with a STAR-family upper stage is a goal.
Mission Objectives	M1. AIM3 will perform measurements of the Didymos i. orbital state ii. rotational state iii. size, mass iv. surface and shallow subsurface properties v. density, geology M2. AIM3 shall perform measurement of the orbital state change after the impact of DART M3. AIM3 shall monitor the impact results and the ejecta after the impact event. M4. AIM3 shall characterize deep interior and gravity of Didymos
Technology Objectives	T1. Demonstrate high speed downlink optical communication (Qualify optical communication for future missions). T2. Landing on Asteroid with a main lander (MASCOT-2) and perform radar observation with a contained cost. T3. Enhanced Challenge on CubeSat University Satellite: a. Deploy 3 to 4 CubeSats around an asteroid and first time beyond Earth Orbit. b. High Image Return for the Agency and increase interest of general public on this scientific mission.
Safety / Technical	S1. The integrity of AIM3P shall be guaranteed during the observation of the DART impact on Didymos secondary asteroid.
Cost	C1. Cost of AIM3P shall not exceed the pre-determined budget → please refer to Cost Section in this Document.

Table 19-1: AIM3P Mission Success Criteria

19.2.2 AIM3P Mission Severity Categorisation

The risk scenarios are classified according to their domains of impact. The consequential severity level of the risks scenarios is defined according to the worst case potential effect with respect to cost, schedule, safety and technical performance.

Identified risks that may jeopardise and/or compromise the AIM-3 mission will be ranked in terms of likelihood of occurrence and severity of consequence.

The scoring scheme for the AIM-3P study with respect to the severity of consequence on a scale of 1 to 5 is established in Table 19-2:

Severity	Programmatics/Schedule	Mission Objectives	Cost
Catastrophic 5	Launch opportunity Lost: AIM3P project cancelled Satellite: Programmatic issues cause the cancellation of the project **AIM3 Mission/Project not Approved** **CAUSE**: Scientific and Technology objectives considered insufficient or not attractive. **RISK prevention/Control**: To define a clear set of: <u>SCIENTIFIC OBJECTIVES:</u> S1-Binary Characterisation S2-Direct measure the orbital state change after the impact S3- Impact observation S4-Observation of Impact Crater and change of orbital state. <u>TECHNOLOGY OBJECTIVES:</u> T1- Demonstrate high speed optical downlink T2- Demonstrate Asteroid Landing (main lander is MASCOT-2) and t2a. survive 3 months t2b. able to communicate with orbiter T3- Enhanced Challenge on CubeSat (University) Satellite: t3a. Deploy 3 to 4 CubeSats around an asteroid (first time beyond Earth Orbit). t3b. Higher Image return and Increase interest of general public on this scientific mission (can also allow proposing a better geografic return approach).	Loss of Complete Spacecraft before performing mission (no mission objective achieved)	Cost increase result in project cancellation
Critical 4	♦ **2020 Launch opportunity at risk** (for AIM3 mission Expected Launch date is November 2020) - Launcher readiness at risk for the expected launch date. - AIM3 Satellite: TRL value too low for some items cause critical impacts on costs due to the need to recover and meet the schedule. *Note* if the 2020 launch date is missed, AIM3 may fly to Didymos on late 2022 and arrive 21 months later (in this case as DART launch is on 28/07/2021 and impact on 06/10/2022 there will be no asteroid in flight characterization before the impact and no DART impact monitoring-> No AIDA mission)* ♦ **No possibility to perform AIDA mission** - DART mission cancelled	C1) Impossibility to fulfil Binary characterization : - orbital state; - rotational state; - size, mass; - density, geology C2) AIM3 does not achieve measurements of the orbital state change after the DART impact.	Critical increase in estimated cost Cost margin lost (above cost estimated margin)

Major 3	Launch opportunity at risk -Launcher readiness potential delay for the expected launch date -Satellite: TRL value too low for some items, causing impacts on costs due to the need to recover and meet the schedule	M1) Impossibility to observe impact crater and change of the orbital state because AIM3P is destroyed by debris ejected by DART impact. M2) Monitor impact results after impact event. M3) Micro Lander (MASCOT-2) fails to land on Didymos M4) Failure of high speed downlink communication	Major increase in estimated cost Cost margin heavily impacted
Significant 2	Loss of time margins.	S1) Micro Lander (MASCOT-2) fails to operate on Didymos S2) Characterisation of deep interior and gravity not successful.	Significant increase in estimated cost
Minimum 1	No/ minimal consequences	No/ minimal consequences. CubeSats fail to deploy or to operate	No/ minimal consequences.

Table 19-2: Severity Categorization

The likelihood of occurrence is normalised on a scale of A to E in Table 19-3:

Score	Likelihood	Definition
E	Maximum	Certain to occur, will occur once or more times per project.
D	High	Will occur **frequently**, about 1 in 10 projects
C	Medium	Will occur **sometimes**, about 1 in 100 projects
B	Low	Will occur **seldom**, about 1 in 1000 projects
A	Minimum	Will **almost never** occur, 1 in 10000 projects

Table 19-3: Likelihood Categorisation

19.2.3 Risk Index

A Risk Index is given as a combination of the likelihood of occurrence and the severity of consequences for a given risk item.

For the AIM-3 mission, risk ratings of low (green), medium (yellow), and high (red) will be assigned based on the criteria of the Risk Index Scheme. The level of criticality for a risk item is denoted by the analysis of the risk index. Following the scheme described in the previous section the highest possible Risk Index will therefore be 5E, and the lowest possible Index, 1A. The risk Risk Index Chart is provided in Table 19-4.

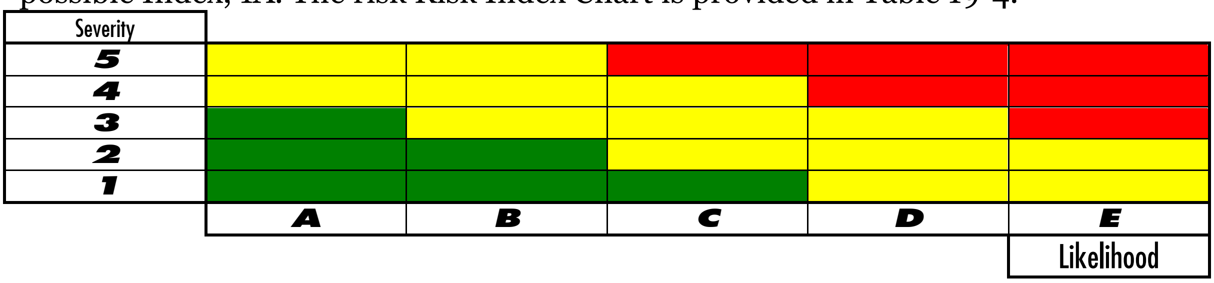


Table 19-4: Top Risk Index Chart

19.3 AIM3P Risks Identification

The main areas identified for risk, being programmatic/schedule, cost, safety, and mission & technical objectives, have been analysed with a top-down approach by developing a set of risk trees and identifying the potential causes of their occurrence.

19.3.1 Technical Risks

No technical risks have been identified as show-stoppers.

During the study, the following main areas affecting the technical risk have been identified:

- TRL value too low: a TRL value of 4 has been identified for the Deep Space Transponder, High Gain Antenna, the High Power Amplifier, the European IMU and the IPSRU of the Optical communication. For these critical technology development, high-TRL back-up solutions have however been identified
- Failures occurring during the mission
- Satellite collision with debris, (this has been categorised as a safety risk).

The controls and mitigations for making them acceptable are fully in line with the state of the art and with the design capability and resources of the AIM3 project.

19.3.2 Mission Objectives

The main risks that will affect the achievement of mission objectives will be generated by failures of the spacecraft or of the payloads, or by incorrect operations (the loss of AIM3 due to debris impact has been analysed among the technical risks under the safety).

The difficulty of this kind of asteroid mission is related to mission engineering, GNC engineering, mission operations and for the landing of the MASCOT-2 on the secondary.

These risks can be mitigated applying the approach used by ESA projects to design and operations, and no technology issues have been identified.

19.3.3 Programmatic/Schedule

The major risk contributors for this AIM3P study have been identified in the programmatic area.

Two main severity categories have been identified:

Catastrophic-> Project cancellation

Critical –> AIM3P readiness and launch opportunity at risk

A solution to mitigate since the beginning the risk on this programmatic area (avoiding high risks of project cancellation and to mitigate potential effects of delays) can be to define a sound approach in the definition of the mission goals and mission success criteria.

The launcher risk has been analysed and solved with the choice of Soyuz, allowing enough margins for the required mass.

19.4 Risk Log

The analysis of the risks for Programmatics/Schedule, Mission Objectives and Cost has led to the identification of the causes and mitigations reported in the following Table 19-5.

Risk Type	Rank	Risk	Cause	Mitigating action	Final Rank	Remarks
Program matic/ Schedule	5D	Launch opportunity lost; AIM3P Project cancelled.	Programmatic Issues / lack of funding change of priorities cause loss of interest in AIM3P mission	To define clear objectives and well explain the importance of the AIM3 mission for European future space programs: RISK prevention/Control: To define a clear set of: SCIENTIFIC OBJECTIVES: S1-Binary Characterisation S2-Direct measure the orbital state change after the impact S3- Impact observation S4-Observation of Impact Crater and change of orbital state. TECHNOLOGY OBJECTIVES: T1- Demonstrate high speed optical downlink T2- Demonstrate Asteroid landing (main lander is MASCOT-2) and its capability to survive 3 months and to communicate with the Orbiter T3- Enhanced Challenge on CubeSat (University) Satellite: Deploy 3 to 4 CubeSats around an asteroid (first time beyond Earth Orbit) Using CubeSats for this mission can have a Higher Image return and Increase of interest of general public on this scientific mission (can also allow proposing a better geographic return approach)	5C	It is considered very important to define high profile Scientific mission objectives and Technology Objectives as per Table 16-1 and Table 16-2
	4D	2020 Launch opportunity at risk	DART Mission Cancelled cause the cancellation of AIM3P mission	For Programmatic risk decrease, it is recommended to develop an approach showing that AIDA mission is an added value for AIM3P but not the sole value for it, and To demonstrate that the AIM3P mission has important scientific return independently from AIDA.	4C/B	It is considered very important to define Mission Success Criteria for AIM3P that make it independent from AIDA mission and To increase the level of independency of AIM3P in order to guarantee and show that in case of DART mission cancellation AIM3P has enough technological and scientific return values that make its mission worth to be performed.

Risk Type	Rank	Risk	Cause	Mitigating action	Final Rank	Remarks
	4D		AIM3P delays cause the satellite not to be ready for the 2020 The following equipment have TRL<4; they should achieve TRL 6 before the start of Phase B2: -Deep space transponder -High Gain Antenna High power amplifier - TTC unit - European IMU -In the Optical High Speed Downlink: The LASER UNIT the EDFA booster amplifier incl. PCU hsaTRL=4 TheELECTRONIC UNITS the IPSRU Electronics, BB has TRL=4	To identify early the project critical issues: . *(X/X-DST has been replaced by TTNC Deep Space Transponder that could be however used as back-up solution ; provided that issues on TTNC are identified early in the project to avoid that the change impacts the schedule)* *For the HGA, the selected baseline is Metasurface (TRL 4 end 2013); in case of need a back-up solution exist to implement a Standard Parabolic Reflector.* *it is assumed to increase in line with project needs due to technology program already implemented.* *Miniaturized TTC (TRL 4 by end 2014) → Selected Baseline* *Standard DST→ Backup (existing as of today)* *This unit is being developed in Europe and will be ready by Phase B/C of this mission. TRL 4 is expected towards the end of 2015. Backup is an ITAR IMU item from Honeywell* *1) IT IS RECOMMENDED TO:* *− Start pre-development design and pre-development tests campaign . for each of those equipment to start as early as possible the relevant contracts for pre-development.* *2) AS AN ALTERNATIVE Back-up solutions have been identified: ATA MSTAR/MIRU for MIT LL -NASA Goddard LLCD Program ATA products have reached high TRL (TRL=7)* If the 2020 launch date is missed, AIM3 may fly to Didymos on late 2022 and arrive 21 months later performing anyway its mission.	4C/B	-; *presently at TRL 4,* In the Optical Head Unit the IPSRU has TRL=4 -In *RISK MITIGATIONS: NOTE: The IPSRU is not essential for the optical communications demonstration, but preferred to be included because it would be more in line with what a standard interplanetary laser comm terminal would include.* *Therefore if the IPSRU's TRL will result too low, as a mitigating action there is the possibility to discard it. The data rate will be reduced by a factor of 2, however it will still be sufficient for a meaningful demonstration of the downlink demonstration.* In this case, if the DART launch is on 28/07/2021 and impact on 06/10/2022, there will be no in-flight asteroid characterization before the impact and no DART impact monitoring -> No AIDA mission)

Risk Type	Rank	Risk	Cause	Mitigating action	Final Rank	Remarks
				It is highly recommended to demonstrate that the AIM3P mission has important scientific return independently from AIDA.		
Mission Objectives	5D	No Mission Objectives Achieved: Loss of complete spacecraft before performing the mission objective.	• AIM3 fatal failure/incorrect operations. • AIM3P fails to reach Didymos due to failure of one or more main subsystems: • DHS • Power • Propulsion • Communication • AOCS	- To develop and implement a sound reliability and safety approach that fits with the Mission Objectives and Operational and Environmental conditions. - To perform Mission /System Hazard Analysis and to identify time-critical conditions planning to ensure Safe Mode reachability and adequacy at any point of mission execution). - Ensure sufficient Ground Station visibility. - Develop Planned trajectories and manoeuvres. - To identify and analyse mission execution manoeuvres. - Ensure well defined FDIR goals. - To develop Project reliability approach to perform and implement reliable design and allocate optimised redundancies. - Fully redundant AOCS system.	5C	To define appropriate Mission Success criteria and associate well defined reliability and safety requirements In the frame of the AIM3P study the reliability to be implemented in the design has been evaluated and a preliminary redundancy approach has been proposed for reaching a design consistent with the needs for AIM3 mission success.
Mission Objectives	4D	-Impossibility to fulfil Binary characterization ; - AIM3P does not achieve measurements of the orbital state change after the impact	• Impossibility to fulfil Binary characterization : -orbital state - size, mass - surface and shallow subsurface properties - density, geology Or • No orbital state change measurements after the impact.	- To ensure adequate resources and effort on PL development, qualification and testing - To define clear requirements for the achievement of the scientific results and of the mission success criteria. AIM3P has 3 P/L's: - Visual Imaging System, if fails: *-> no drop of Lander* *-> no orbital state characterization* *-> no impact crater observation* *-> no determination of asteroid size, mass, gravity*	4C	For Characterization: *Position in heliocentric orbit, with station keeping at safe distance, at 45 degree angle for viewing.* *Minimum resolution 1 m/pixel for science. Actual capture can have less resolution, post-processing can refine the resolution up to 2x.* *Camera specifications: Fov=5.3 degrees.* *Camera must view the whole system, i.e. 3.3 km plane diameter.* *Choosing a 2048x2048 detector, it is possible to see the whole system with 1.6 m/px accuracy from 35.65 km distance.* *Observation angle for 3D asteroid reconstruction possible*

Risk Type	Rank	Risk	Cause	Mitigating action	Final Rank	Remarks
				- Thermal IR Imager, if fails: *Partial loss of surface characterisation and full loss of shallow subsurface characterization results in degraded "science" return and possibly insufficient determination of the surface properties.*		*between 20 and 70 degrees, optimal between 30 and 60 degrees due to shadows observation on features.* *Ground segment: orbit determination optimal around 45 degrees.* *45 degree angle chosen.* *Asteroid rotation not perpendicular to orbital plane: one of the two poles will not be observed well, but IR payload may help in characterisation.*
Safety	**3D**	- Debris generated from DART impact with Didymos cause the loss of AIM3 and the impossibility to observe impact crater and asteroid change of orbital state	AIM3P not at safe distance from Didymos when DART hits it and is damaged by the debris generated by the impact.	Risk mitigation criteria are applied: - AIM3P Safe Distance from Didymos when the impact occurs (100 km distance when DART impacts on Didymos). - Trajectory Planning and mission execution manoeuvres considering escape trajectory.	**3B**	.
Technology Objective	**3D**	MASCOT -2 fails to land/operate on Didymos	MASCOT02 failure MASCOT-2 wrong separation MASCOT -2 misses the secondary MASCOT-2 bounces off the secondary MASCOT-2 is exposed to excessive solar irradiation Landing impact caused by a rock on the landing surface Dust of landing site surface cause MASCOT-2 to be partially/totally covered by powder Too soft landing site surface cause MASCOT-2 to partially sink and being Partially buried	To ensure adequate resources and effort on MASCOT -2 PL development, qualification and testing. To foresee analyses and tests for MASCOT safe landing.	**3C**	*Deployment of MASCOT to primary asteroid:* *Maximum permitted velocity at surface = half escape velocity (40cm/s) = 20 cm/s.* *Deployment of MASCOT to secondary asteroid.* *Maximum* permitted *velocity at surface = half escape velocity (14.6cm/s) = 7 cm/s.* *It is recommended to investigate the possibility to use one or more CubeSats to land on the secondary and to develop the Landing as a more articulated operation using the CubeSats for exploring the landing sequence and the landing site before starting the MASCOT-2operations (this will generate an interesting source of data for the success of the MASCOT-2 mission).*
Technology Objective	**3D**	Loss of OPTICAL high speed downlink demonstration	No or Poor demonstration	To define OPT operational requirement that fit with the tecnology demonstration goals v.r.t. mission objectives and OPT SS	**3C**	

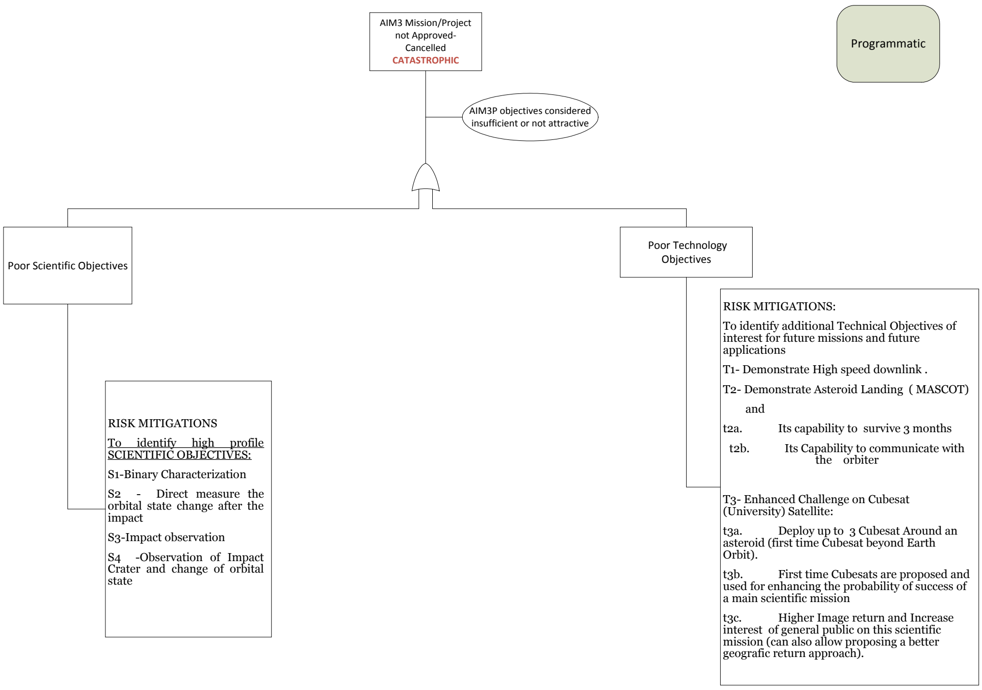
Risk Type	Rank	Risk	Cause	Mitigating action	Final Rank	Remarks
			Optel-D P/L fails and do not perform its experiment	performances/resources available for AIM 3P mission. To apply a smart redundancy approach (identify the critical items and implement redundancy on them) To develop/implement a reliability approach that fits with the mission constraints and environment		
Cost	**5B/A**	Cost increase results in project cancellation	No high risk conditions have been identified for this event during the study.	.		Likelihood for this event is considered very low: B or even A
	3D	Critical increase in estimated cost	This risk has been evaluated and found to be with a medium likelihood	Cost Drivers: - Complex GNC Manoeuvres - Development of GNC algorithms - Development of GNC SW - Difficult Ground Control - A certain level of intelligence may be required on-board of the spacecraft - Deep Space Environment	**3C**	It is recommended to have an experienced prime contractor in order to cope with the complexity of the AIM3 mission and to put high attention on cost products and on TRL instead of mass reduction. For the Cost, the evaluation and discussion of the risks is covered in the dedicated cost section in this document, and the separate cost report.
	3E	Major increase in estimated cost	This risk is considered a bit more likely than the previous one.	Same risks mitigations of the above one.	**3C**	

Table 19-5: Top Risk Log

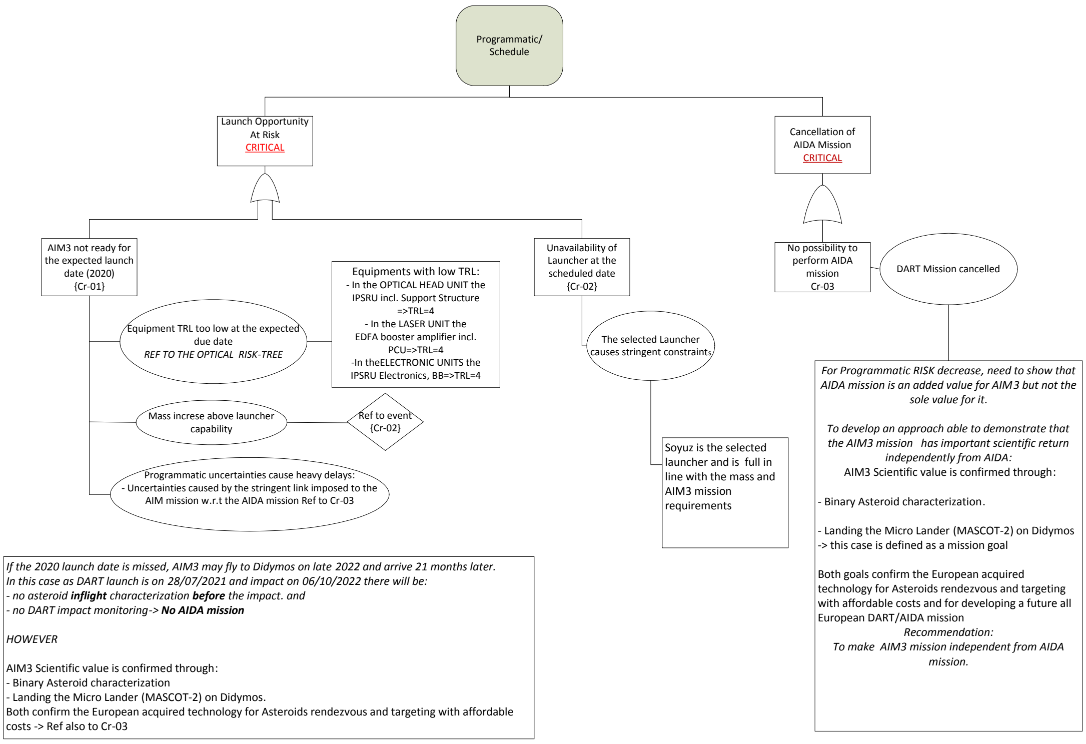
19.5 Risk Trees

Dedicated Risk Trees have been developed with a top-down approach that, starting from the undesirable event under evaluation (RISK), has allowed to identify the causes that can generate it and the relevant mitigations that can reduce its likelihood.

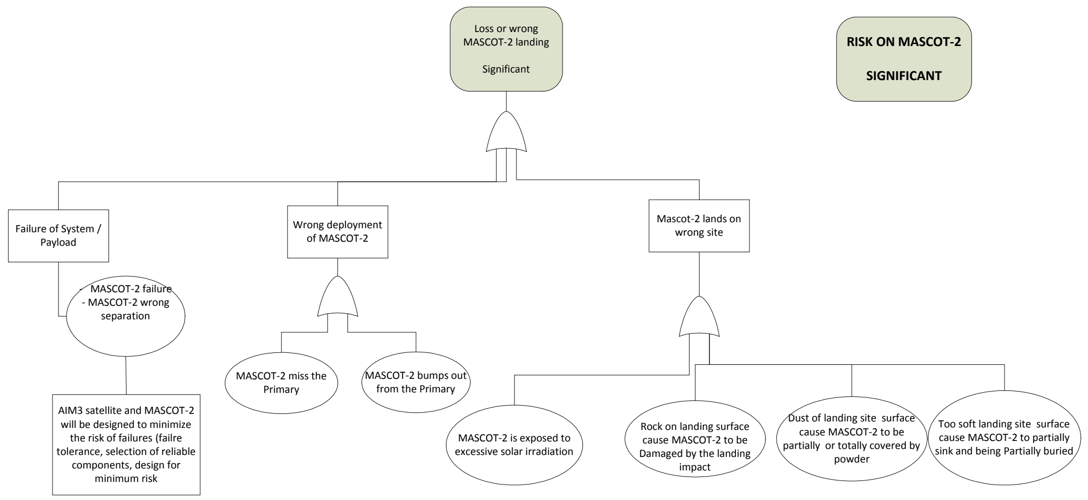
19.5.1 Programmatic Risk Tree – CATASTROPHIC



19.5.2 Programmatic Risk Tree - CRITICAL



19.5.3 Technology Objective Risk Tree – MASCOT-2



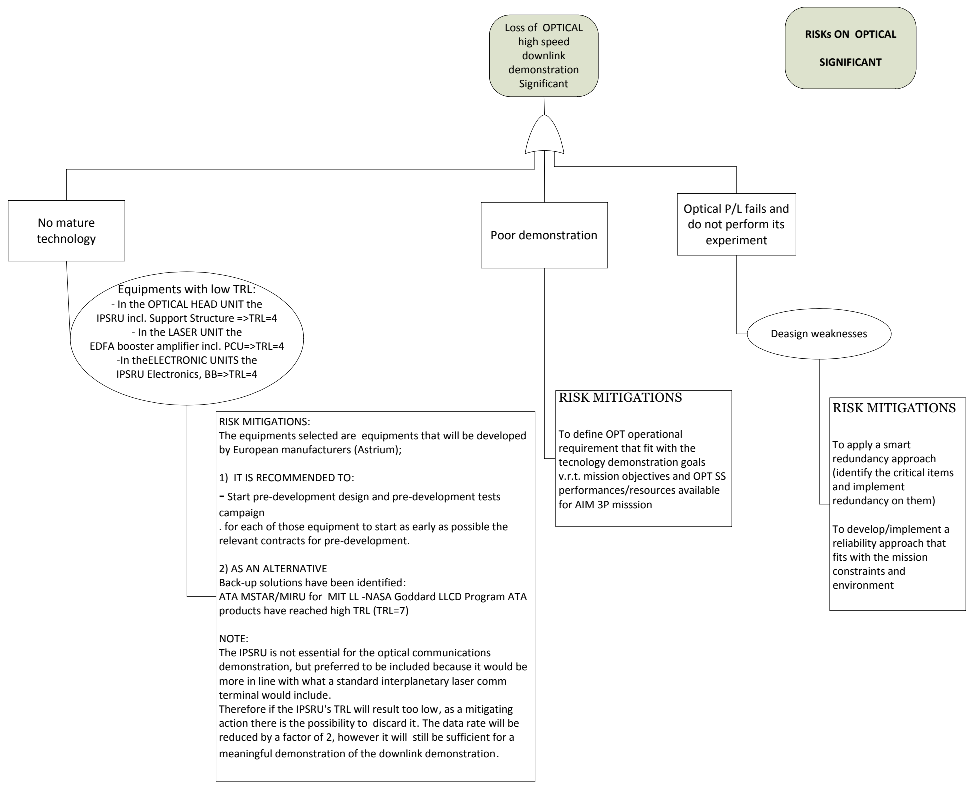
-4 CubeSats are foreseen; The Cubesat deployment device will deploy all at the same time around the secondary asteroid.

The 4 CubeSats will be deployed around the secondary.

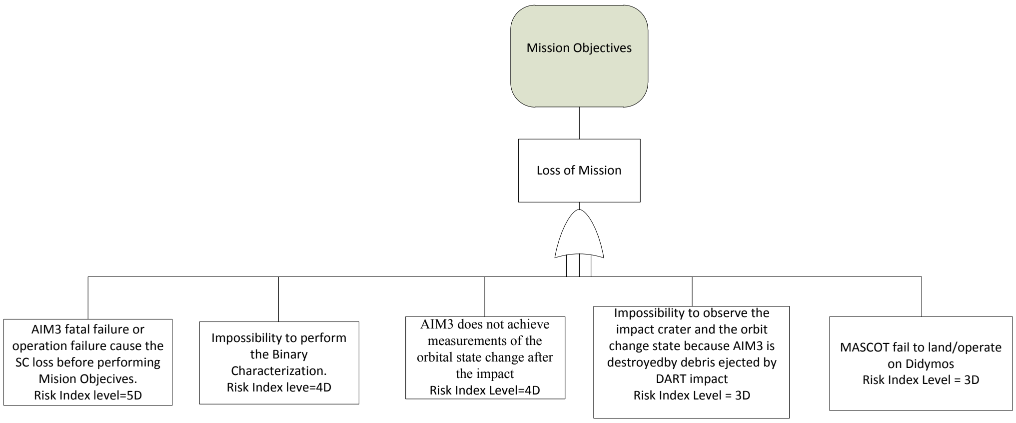
The MASCOT-2 will be deployed on the secondary

Note:
Two independent dispensers are foreseen for 4 CubeSats.
It is recommended to investigate the possibility to use one or more CubeSats to land on the secondary and to develop the Land ing as a more articulated operation using the CubeSats for exploring the landing sequence and the landing site before starting the MASCOT -2operations (this will generate an interesting source of data for the success of the MASCOT -2 mission).

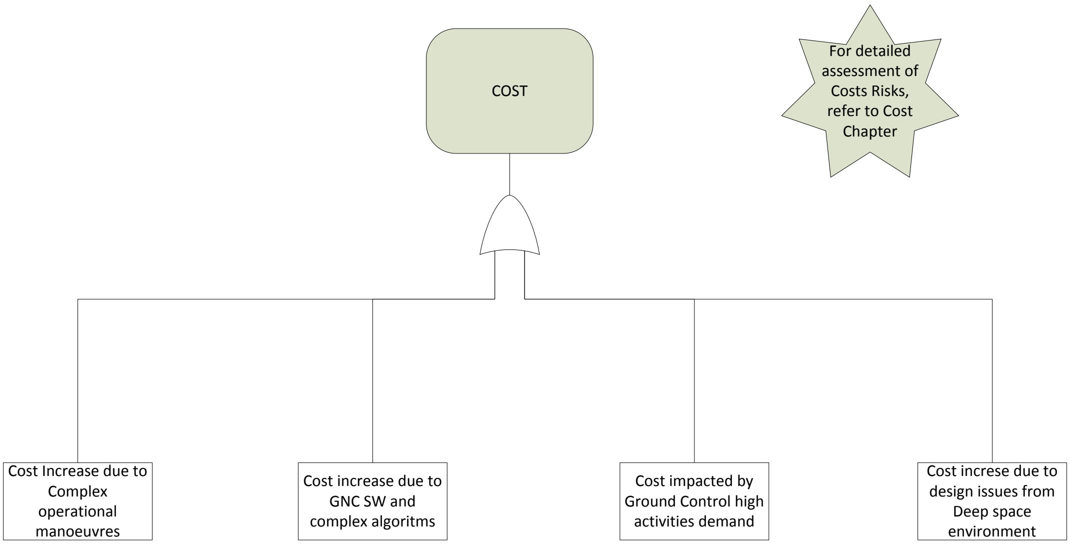
19.5.4 Technology Objective Risk Tree – Optical High Speed Downlink



19.5.5 Mission Objectives Risk Tree



19.5.6 Cost Risk Tree



19.5.7 RISK Log Results

No technical risks have been identified as show-stoppers. For all the technical risks identified during the study (too low TRL, satellite collision with debris, failures occurring during the mission) the controls and mitigations for making them acceptable are fully in line with the state of the art and with the design capability and resources of the AIM3P project.

The launcher risk has been analysed and solved with the preferred choice of Soyuz.

For cost, the discussion is left to the cost discipline, however the cost target and budget should be carefully evaluated/negotiated at the beginning of the project and then monitored during its development.

The major RISK contributors for this AIM3P study have been identified in the programmatic area.

Two main severity categories have been identified:

Catastrophic -> Project cancellation

Critical -> Risk of having AIM3P not ready for the expected launch date

For programmatic risk decrease, it is recommended to show that the AIDA mission is an added value for AIM3 but not its sole reason, and to develop an approach able to demonstrate that the AIM3P mission has important scientific return independently from the AIDA mission.

19.6 Conclusions

It is recommended to make AIM3P mission goals totally, or at least more, independent from the AIDA mission and create dedicated Mission Success Criteria for AIM3P like for example:

- To focus on the objective to perform the Binary Asteroid characterisation by the AIM3P mission
- Landing the Micro Lander (MASCOT) on Didymos
- Perform OPTICAL high speed downlink demonstration
- To highlight that these goals confirm the European acquired technology for Asteroids rendezvous and targeting with affordable costs and for developing a possible future all-European DART/AIDA mission.
- Enhanced Challenge on CubeSat University Satellite; Deployment of 3 to 4 CubeSats around an asteroid and for the first time in a mission beyond Earth can constitute a High Image Return for the Agency and Increase interest of general public on this scientific mission.

There are six examples of equipment on the critical path to mission success. These are the TTC transponder, SSPAs, HGA, European IMU and IPSRU of the Optical System.

However for all of them there is reasonable confidence that the required TRL level can be reached or that a suitable back-up solution is available.

20 COST

This chapter presents the cost estimate of the CDF study AIM-3P, which is based on AIM-3 but with increased amount of payload. A target of 200M€ was given. A main requirement was to design AIM-3 as a low cost mission.

20.1 Cost Trades

20.1.1 Analysis of the Cost Target

	SOYUZ
Target	200 M€
Launch Price	65 M€
Remaining	**135 M€**
Target includes ESA own costs and mission operations	

Table 20-1: Calculation of remaining budget

Also Vega as launcher was investigated in AIM-3. Its launch price at 35M€ was less than Soyuz, but the P/L performance was not sufficient.

The target is understood to include all costs accrued at ESA project level; it includes ESA internal costs and mission operations costs too. Therefore the costs for spacecraft development need to be significantly below the remaining budgets as calculated in Table 20-1.

Rosetta	447 k€/kg	
Clementine	350 k€/kg	NASA asteroid probe
Bepi Colombo	342 k€/kg	proposal, actual are considerably higher
Gaia	333 k€/kg	
AIM CDF Study	330 k€/kg	estimate
AIM-3 CDF Study	288 k€/kg	estimate
AIM-3P CDF Study	tbd	estimate
Proba-V	136 k€/kg	
Proba-2	130 k€/kg	

Table 20-2: Specific development costs of historical spacecraft

Historical data of spacecraft developments are listed in Table 20-2. However, proper reference missions for AIM-3P don't exist. The specific costs stem from industrial development prices of spacecrafts, excluding payloads. The present estimate for AIM-3P is listed for benchmark. All interplanetary missions are above the target and estimate, while the low cost missions of Proba are below by factor 2.

The historical costs of Proba demonstrate that low cost missions are feasible. A further analysis of low cost missions has been made in chapter 20.1.3.

20.1.2 Cost Drivers

With respect to a low cost mission the spacecraft design has been kept simple. No extreme cost drivers were detected at the hardware level. Hardware components could be selected at high TRL as far as possible, except some new products of the communication subsystem. The difficulty of this kind of asteroid missions is related to mission engineering, GNC engineering and mission operations.

Cost driving elements are:

- Complex GNC manoeuvres
- Mission control from ground versus autonomous spacecraft
- Deep space environment.

Cost saving elements are:

- Sufficient mass margins: Costs can be saved by selection of low cost components at high TRL, instead of expensive light-weight components.
  The AIM-3 design consisted of a limited number of payloads, and thus provided a high mass margin for the selected Soyuz launcher. With the increased number of payloads in AIM3P the mass margin is reduced. Care must be taken to ensure that the once robust design will not turn into the need for a light-weight design.
- The mission goals are clearly defined. In order to keep the low cost approach, the requirements should be kept at a minimum. Also in later project phases identification of additional mission goals should be avoided.

20.1.3 Analysis of Low Cost Missions

In the frame of this CDF study the cost differences between AIM-3 and Proba-2 have been analysed. The purpose was to understand the low cost approach of Proba-2 and its applicability for AIM-3.

The estimated specific cost (€/kg) of the total spacecraft development of AIM-3 is about factor two higher than for Proba-2. At hardware and software level the AIM-3 estimate was found relatively close to Proba-2, also because of similar assumptions for the model philosophy. The major cost difference was detected at system level.

The hourly rates of the prime contractor of Proba-2 (about 80€/h) are about half of the AIM-3 assumption (150€/h). A backwards calculation of the team sizes of the primes, flat loaded over the development schedules, results in:

- 3 to 4 persons for Proba-2,
- About 30 persons for AIM-3.

Even for a small satellite with a simple orbit (LEO, SSO) the resulting team size for Proba-2 seems to be too low. It has been discussed, that ESA engineering was supporting the prime contractor’s team.

The resulting team size for AIM-3 is below the statistical averages of ESA satellite projects. Due to the nature of the asteroid mission, which is a first of its kind, a further reduction could add development risk.

Clementine: An example of a low cost planetary mission was the NASA project Clementine, launched in 1994. The mission was planned to a lunar orbit with lunar mapping and then further to an asteroid flyby. The mission duration was less than 1 year. After departure from moon the spacecraft was lost due to a software failure.

The spacecraft dry mass was 232 kg. It was launched on Titan IIG from Vandenberg. The development schedule (feasibility study till launch) was 25 month.

The actual costs are originally reported in M$ in E.C. 1995. A conversion has made to M€ 2014 by use of a factor 1.56:

	EC 1995	EC 2014
Spacecraft Bus	52.0 M$	81.1 M€
Payload	4.8 M$	7.5 M€
Launch	21.3 M$	33.2 M€
Ground Segment	1.5 M$	2.3 M€
Operations (1 year)	5.3 M$	8.3 M€
Total	**85.0 M$**	**132.6 M€**

Table 20-3: Actual Costs of Clementine

The specific cost of the spacecraft development was 350k€/kg. Although this is in the range of European interplanetary missions (see Table 20-2), Clementine is referred to be an example of a low cost mission (RD[29]).

20.1.4 SADM versus Fixed Panels

At the time of the trade-off, the following two solar array options were evaluated:

1. SA area = 8m², fixed panels, batteries = 4kg, PCDU = 15kg
2. SA area = 5.6m², SADM, batteries = 3.5kg, PCDU = 13kg

Although SADM has been assumed off-the-shelf, the option 1 with fixed solar arrays is cheaper. However, the difference in costs is low. Later in the study the non-SADM SA area diminished from 8 to 5. 6m²; the trade-off was not revisited, as the relative cost comparison remains valid.

20.2 Class of Estimate

The cost estimate has been performed within the CDF environment by ESA/ESTEC Cost Engineering (TEC-SYC). The type of cost estimate is Class 4 (as described in the ESA Cost Engineering Chart of Services).

The accuracy of the complete estimate is expected to be within -20%/+20%.

20.3 Cost Estimate Methodology

The following methods have been used:

- Reference to similar equipment/subsystem/system level costs, taking into account the amount of new development required
- Expert judgements
- Equipment cost models
  - ESA Standard CERs
- System-level cost relationships (for the Prime contractor and subcontractor activities), based on observed relationships for relevant references
- Project Office cost model POCoMo
- Risk model Opera

The cost estimation has been performed at subsystem and equipment level.

20.4 Scope of Estimate

The estimate includes:

- Industrial Development Phases B, C/D and E1
- Ground Segment Preparation
- Phase E, Mission Operations
- Launch (Soyuz)
- ESA internal cost
- Cost Risk Margins

The scope excludes:

- Payload development and procurement
- Scientific operations
- Pre-development of technologies (if needed).

20.5 Main Assumptions

- A low cost approach has been considered as far as possible
- The price of the Soyuz launch is guessed at 65M€, which is at the lower end of the range. It may cost more, even above 70M€
- The mission operations costs are preliminary guessed. This needs to be updated from an estimate prepared by ESOC
- The Data Handling Subsystem consists of a CDMU and a RTU, both assumed to be existing components
- All technical details relevant for the cost estimate are derived from the CDF experts, the CDF workbooks and presentations
- No need for Planetary Protection

- The ESA internal costs are guessed to be 15% on top of the Expected Industrial Price. This is rather optimistic compared to historical data, but in line with the 85/15 rule.

Industrial Organization: Phase A and B1 is assumed to be executed by two contractors in parallel.

A medium to large prime contractor is assumed for this type of mission, because at system engineering level experience is needed especially in the field of GNC. In order to implement a low cost project, the direct procurement of equipment is assumed. Suppliers and vendors shall be from normal ESA industrial sources. No subsystem responsible contractors are needed. In case of additional subsystem level contractors significant costs need to be added.

Probably a subsystem level contractor is needed for the propulsion subsystem, if the Prime does not have that in-house capability.

Spacecraft System Level Activities: The spacecraft system level activities (prime contractor tasks) are an important cost driver of this project. It has been estimated by two tools, a system level CER and a PO cost model.

In order to follow the low cost approach and due to the equipment procurement approach, the present estimate assumes lower complexities for Mgmt, PA and AIT than for average missions of the same size. Normal average complexities are assumed for Engineering and GSE due to the nature of the asteroid mission.

As a result of the costs, the following team sizes could be calculated flat loaded over the schedule:

- Phase B1 13.0 persons (2x)
- Phase B2 19.5 persons
- Phase C/D 32.0 persons
  - 5.0 Mgmt
  - 2.0 PA
  - 25.0 Engineering

Compared to AIM-3 the phase C/D team size increased by about 2.5 persons due to the increased number of payload.

Schedule

- Phase B1 6 month
- Phase B2 12 month
- Phase C/D 32 month
- Phase E 27 month
  - Transit 20 month
  - Asteroid Operations 7 month

Model Philosophy: At system level the following models will be integrated:

- Proto-flight Model

- Avionics Test Bench

At subsystem level QMs will be built and tested for products newly developed. Some EMs will be manufactured for the Avionics Test Bench. Also a Spacecraft Simulator will comprise part of the Avionics Test Bench. A SM of the structure is assumed to be manufactured and tested.

The number of equipment products to be manufactured and tested can be obtained from Figure 20-1.

TRLs: At start of development phase C/D all TRLs are assumed to be at level 5 as minimum. The Transponder, HPA and HGA are presently at TRL 4, but assumed to increase due to technology programs already implemented.

The TRL assumptions can be obtained from Figure 20-1.

	Technology Readiness Level									Hardware Models				
	1	2	3	4	5	6	7	8	9	STM	EM	QM	PFM	FM1
GNC														
Star Tracker (2)	■	■	■	■	■	■	■	■			1		1	1
Sun Sensor (4)	■	■	■	■	■	■	■						1	3
IMU (2)	■	■	■	■	■	■					1		1	1
Nav Cam	■	■	■	■	■	■	■	■			1		1	
Reaction Wheels (4)	■	■	■	■	■	■	■	■			1			4
Electrical Power														
Solar Array (2)	■	■	■	■	■	■	■						1	
Battery (1)	■	■	■	■	■	■	■	■					1	
PCDU (1)	■	■	■	■	■	■					1		1	
Harness														
Harness	■	■	■	■	■	■							1	
TT&C														
Mini DST X-band Transponder (2)	■	■	■	■	▨							1		2
HPA - SSPA (2)	■	■	■	■	▨							1		2
HGA (1)	■	■	■	■	▨	▨						1		1
LGA (2)	■	■	■	■	■	■	■				1		1	1
ISL (1)	■	■	■	■	■	■						1		1
RF Distribution Network (1)	■	■	■	■	■	■	■				1		1	
Data Handling														
CDMU (1)	■	■	■	■	■	■	■	■			1		1	
RTU(1)	■	■	■	■	■	■	■	■			1		1	
Structure														
Primary and Secondary Structure	■	■	■	■	■	■				1			1	
Thermal Control														
Thermal Hardware	■	■	■	■	■	■	■						1	
Propulsion														
Thrusters 10 N (24)	■	■	■	■	■	■	■	■			1			24
MMH Tank (2)	■	■	■	■	■	■	■	■					1	1
MON Tank (2)	■	■	■	■	■	■	■	■					1	1
Pressurant Tank	■	■	■	■	■	■	■	■					1	
Propellant Feed System	■	■	■	■	■	■	■						1	

Figure 20-1: HW and SW Equipment

20.6 Cost Risk/Opportunity

The methodology used to calculate cost-risk contingencies is described in RD[27]. A Monte-Carlo simulation has been applied using the risk model Opera.

The basic estimate derives as Point Estimate. The Point Estimate excludes any risk and management margins, which are normally inherent parts of industrial FFP proposals. It includes 8% nominal profit only. The risk items are segregated to:

- CMA (Cost Modelling Accuracy)
- DMM (Design Maturity Margin)
- POE (Project Own Events)
- EPE (External to Project Events).

The Point Estimate plus all contingencies shared to Industry result in the Expected Industrial Price.

The contingencies shared to ESA are used to cover underestimated costs during negotiation and typical class A changes during development.

The contingency margins estimated for AIM-3 are presented in Table 20-4:

%	Industry	ESA	**Total**
CMA	2.4	7.3	**9.7**
DMM	14.4	n/a	**14.4**
POE	2.5	9.1	**11.6**
EPE	2.0	tbd	**2.0**
Total	**21.3**	**16.4**	**37.7**

Table 20-4: Contingency Margins

All margins are estimated for a 70% confidence level.

The risk register includes the risk of under estimated complexity of the mission as well as an opportunity for further reduction of the system tasks.

20.7 Cost Estimate

All costs are expressed in k€ at EC 2014.

CDF AIM-3	
	figures in k€ EC mid-2014
AIM-3 Spacecraft	
Payload	
LEOP incl Launch Preparations	
INDUSTRIAL COST - POINT ESTIMATE (excl. risk provisions)	
2.4% Cost Modelling Accuracy (Industry share)	
14.4% Design Maturity Margin	
INDUSTRIAL COST - CORE ESTIMATE	
2.5% Project Owned Events (Industry share)	
2.0% External to Project Events (Transferred to Industry)	
EXPECTED INDUSTRIAL PRICE	
Soyuz Launcher including Payload Adapter	
Mission Operations (Provision by ESOC)	
Science Operations (Provision by ESAC)	
TOTAL Phase B, C/D, E1 + Launch + Operations	

ESA INTERNAL	
15.0% ESA Internal Cost (to be confirmed by Project Controller)	
16.4% ESA Project Level Cost Risk Coverage	
7.3% Cost Modelling Accuracy (ESA share)	
9.1% Project Owned Events (ESA share)	
TOTAL ESA PROGRAMME COST	
External to Projects Events (ESA share)	t.b.d.

Figure 20-2: Total Cost Breakdown

CDF			
AIM-3			
Date:	20.06.2014	Cost Estimator	GR
Model Philosophy	ATB + PFM		
Currency	k€	Ec. Cond.	2014
cost Item		Item Estimate	Totals
AIM-3 Point Estimate			
PHASE B			
Phase B1 x 2			
Phase B2			
PHASE C/D			
GNC			
	Star Tracker (2)		
	Sun Sensor (4)		
	IMU (2)		
	Reaction Wheels (4)		
Electrical Power			
	Solar Array (2)		
	Battery (1)		
	PCDU (1)		
Harness			
	Harness		
TT&C			
	Mini DST X-band Transponder (2)		
	HPA - SSPA (2)		
	HGA (1)		
	LGA (2)		
	ISL (1)		
	RF Distribution Network (1)		
Data Handling			
	CDMU (1)		
	RTU(1)		
Structure			
	Primary and Secondary Structure		
Thermal Control			
	Thermal Hardware		
Propulsion			
	Thrusters 10 N (24)		
	MMH Tank (2)		
	MON Tank (2)		
	Pressurant Tank		
	Propellant Feed System		
Onboard Software			
	DHS SW		
	GNC SW		
	Independent Software Verification		
Total Subsystems			
GSE			
	Mechanical and Electrical GSE		
	Spacecraft Simulator		
Assembly, Integration and Test			
System Level PO			
	Management & Control		
	Product Assurance		
	Engineering		
Total Phase C/D			
Total Platform Phase B & C/D Point Estimate			

Figure 20-3: Point Estimate Cost Breakdown

20.8 Conclusion and Recommendation

The AIM-3P asteroid mission is a first-of-the-kind mission in Europe. The technologies and hardware products are available, but complex GNC algorithms need to be developed. A low cost approach can be implemented for the procurement of hardware equipment. The deletion of mechanisms, a rather short in-orbit lifetime and sufficient mass margins allow the selection of low cost components. Also it is recommended to implement a minimum model philosophy with a PFM approach and an ATB for simulation purposes. The procurement of equipment shall be executed directly by the prime contractor without subsystem responsible contractors. The required qualification level of electronic components needs to be clarified in order to withstand the deep space environment. There is the risk that low cost products available for LEO and GTO missions are not appropriated for deep space.

The possibility of a low cost approach for the system level tasks needs to be carefully investigated. Mgmt, PA and AIT probably can be implemented at low costs due to a simplified procurement process of hardware. But reduction of engineering below the average level of historical projects may add risk to the mission. Also the prime contractor needs to be skilled in developing deep space missions.

The cost target of 200M€ seems to be credible. The costs increased from AIM-3 to AIM-3P due to the higher amount of payloads. For a low cost mission a robust design with sufficient mass margins should be implemented. The increasing amount of payloads for AIM-3P is reducing the margins. Examples of low cost missions are Proba-2 and Clementine. The low level of cost as for Proba missions (LEO, SSO) cannot be reached, but compared to other European interplanetary missions the spacecraft development, expressed in relative cost (€/k), is significantly lower. The interplanetary mission of Clementine of NASA is an early example of a low cost mission. The cost estimated for the AIM-3 spacecraft is on a similar level as for Clementine. The €/kg is even lower. The low cost of Clementine led to a failure of the software and to loss of the mission. Therefore a good balance is recommended between cost reduction and mission risk.

The estimated costs exclude costs of the payload, apart from the VIS Camera also used for GNC, and scientific operations.

21 CONCLUSIONS

Early in the AIM-3 study the initially proposed concept involving a Vega launch with a standard solid propellant kick stage proved unfeasible from a launch mass point of view.

The subsequent study of an AIM-3/3P concept based on a Soyuz launch from Kourou has resulted in an overall robust mission design, and no immediate showstoppers or unmanageable risk issues have been found. The asteroid observation scenario (in terms of instruments suite and performance, observation duration and data return) is deemed satisfactory. The platform is based on mostly high-TRL equipment, except in the case of the S/C transponder, HGA, HPA, and GNC European IMU, which are all currently at TRL 4. For these AIM-3 should be regarded as a demonstration opportunity, in line with the requirement to use innovative technology/equipment with a TRL≥6 at the start of Phase B2 for demonstration purposes. Mature equipment alternatives are however available for these items, with minor impacts on the overall spacecraft and mission design.

The current design incorporates sufficient margins at various levels:

- The normally embedded CDF budget margins at equipment and system level.
- The remaining excess Soyuz payload capability of 27 kg (this in addition to the 50 kg “launchability margin” already deducted from the Soyuz launch capability as advertised in the Soyuz User Manual)
- The significant design flexibility left (in terms of changes in equipment and payload instruments type and number, redundancy etc.).

All this leaves much room for further trades and optimisations.

The CDF AIM-3P design is suitable for fast, near-future project implementation, and it is expected that the mission could be developed in time with respect to the stated launch window constraint, although there would be relatively little margin. The complete mission would most likely cost more than the AIM-3P study 200 M€ target, but AIM-3 would still be a very low-cost asteroid mission w.r.t. the actual Rosetta mission and the Marco Polo-R concept.

It is considered very important to define Mission Success Criteria for AIM3P that make it independent from AIDA and to increase the level of independency of AIM3P in order to guarantee and show that in case of DART mission cancellation AIM3P has enough technological and scientific return value.

The study goal to envision an AIM-3P mission similar in philosophy to those of Proba-1, -2, -V, SAOCOM-CS, SMART-1 and Cheops (Science S-class) rather than a standard Science M- (Marco Polo-R) or L-class mission (Rosetta) was incorporated via:

- The possibility to fulfil most of the key mission objectives with a minimal payload, only including a Narrow Angle Camera, and a small Lander. No altimeter or dedicated Wide Angle Camera were found to be required. However, during the AIM-3P the observation payload was significantly reinforced to make more optimal use of the capabilities of the platform and the Soyuz launcher.
- The relatively small spacecraft design, which in mass and size(as well as complexity) is similar to Mars Express and Venus Express.

- The relatively short mission, including a 1.5 year transfer and only 5.5 months of operation (plus up to 2.5 months extension) at the asteroid.
- The spacecraft in its current iteration has no strict need for Solar Array Drive Mechanisms (SADMs) or a High Gain Antenna Pointing Mechanism, but implementing SADMs would increase mission flexibility with a relatively small impact (in terms of mass, power and cost) on the design. In addition, more detailed analysis of the requirements and constraints for the DART impact crater imaging and lander deployment & communication could potentially lead to the necessity to incorporate SADMs.

Dual use of equipment has been incorporated by the use of the VIS Camera as both Payload Instrument and GNC sensor (the option of using the Star Tracker(s) as an asteroid-finding Wide Angle Camera was not found to be required).

21.1 Satisfaction of Requirements

All requirements have been satisfied so far as depicted in the tables hereunder. Further analyses are however required to increase the level of confidence on the preliminary results established in the course of the CDF study.

21.2 Compliance Matrices

Mission Requirements		
Req. ID	**STATEMENT**	**Compliance**
MIS-011	The mission shall rendezvous with the 65803 Didymos (1996 GT) binary asteroid system before the DART impact, no later than 22/5/2022 allowing at least 2 months of prior observations.	Compliant
MIS-012	The launch date shall be no later than 17/10/2020.	Compliant
MIS-020	The Mission shall be launched with an Arianespace launcher.	Compliant, Soyuz from Kourou
MIS-030	The mission cost target is 200M€	Partially compliant; budget incl. Mission Operations likely to exceed the target by a few tens of M€.
MIS-050	The mission shall be able to determine, via remote sensing, the 65803 Didymos (1996 GT) binary asteroid's rotation and orbit dynamics as well as its physical properties, in the following order of priority: • Secondary asteroid mass, size, shape and density • Dynamical state of the secondary asteroid rotation • Shallow sub-surface structure , surface roughness and topology of the secondary asteroid. • Deep subsurface, internal structure, chemical composition and gravity of the secondary asteroid.	Compliant
MIS-060	The mission shall be able to deploy a micolander (MASCOT-2) on the surface of the primary and support its operation.	Compliant, but further analysis of GNC approach

		needed.
MIS-110	The mission should be able to obtain imagery of the binary asteroid system with a resolution of 1 m.	Compliant
MIS-125	Optical communications shall be demonstrated.	Compliant

Table 21-1: Mission requirements compliance matrix

Payload Requirements		
Req. ID	**STATEMENT**	**Compliance**
SYS-PL-030	The payload shall be able to obtain imagery of the binary asteroid system with a resolution better than 1 m.	Compliant
SYS-PL-040	The spacecraft shall enable surface asset (asteroid microlander (MASCOT-2) landing.	Compliant, but further analysis of GNC approach needed.
SYS-PL-050	The spacecraft shall accommodate the following payload: • Visual Imaging System (VIS) • Thermal Imager (TIRI) • Monostatic High Frequency Radar (HFR) • Bistatic Low Frequency Radar (LFR) • MASCOT-2 • Optical Communications Terminal • CubeSat Opportunity Payloads (COPINS)	Compliant
SYS-PL-060	The payload mass allocation for the asteroid research and technology payloads shall be 74.4 kg including margins.	Compliant
SYS-PL-070	The VIS camera (also used for navigation) shall be used as a scientific instrument and the spacecraft shall accommodate the instrument such that observations of the secondary asteroid can be made.	Compliant

Table 21-2: Payload requirements compliance matrix

System requirements		
Req. ID	**STATEMENT**	**Compliance**
SYS-035	The spacecraft shall be integrated with a Soyuz 2-1b Fregat launcher to be launched from Kourou.	Compliant
SYS-040	A mass of 110 kg shall be assumed for the standard Soyuz - Fregat 2-1b launch adapter.	Compliant
SYS-050	The system 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	Compliant

System requirements		
	○ 20% for new developments or items requiring significant modification •	
SYS-051	A 20% maturity margin shall be added on top of the dry mass computed as mentioned above.	Compliant
SYS-052	A 2% propellant margin shall be added on top of the propellant mass to account for residuals, with the nominal propellant mass being based on the spacecraft dry mass with margins.	Compliant
SYS-060	The spacecraft shall be compatible with a mission lifetime of 2.5 years from launch to end of life.	Compliant
SYS-070	The spacecraft should be designed using European components.	Compliant
SYS-080	Only technologies at a minimum of TRL 5 by the end of phase B1 shall be used.	Compliant
SYS-090	The spacecraft total launched wet mass shall not exceed 850 kg including all maturity margins	Compliant
SYS-PRO-010	The spacecraft shall be capable of 100 m/s for the duration of the near-asteroid operations.	Compliant
SYS-AOC-010	The spacecraft shall be 3-axis stabilised	Compliant
SYS-AOC-020	The spacecraft shall provide an Absolute Pointing Knowledge (APK) of 50.00E-6 rad.	Compliant
SYS-AOC-030	The spacecraft shall provide an Absolute Pointing Error (APE) of 0.35E-5 rad.	Compliant
SYS-AOC-040	The spacecraft shall provide an Relative Pointing Error (RPE) of 3.00E-6 rad over 1 s.	Compliant
SYS-TTC-040	The spacecraft shall receive telecommands and transmit housekeeping telemetry at all times in the mission.	Compliant except for a short time during transfer when the Earth is behind the Sun.
SYS-TTC-050	Science data return shall support a total mission science data volume of 25 Gbits over 6 months.	Compliant
SYS-TTC-060	The spacecraft shall be able to communicate simultaneously with any deployed technology packages.	Compliant

Table 21-3: System requirements compliance matrix

Operational mission requirements

Req. ID	STATEMENT	Compliance
MIS-OP-010	A 10 km co-orbiting position shall be assumed with respect to the binary asteroid system for characterisation observations.	Compliant
MIS-OP-020	The spacecraft should aim to observe the DART impact from a position 90 degrees to the normal of the local horizontal of the impact site with a slant range 100 km.	Compliant
MIS-OP-030	The spacecraft shall be able to transfer to different co-flying points both and maintain that position for an indefinite period within the duration of local asteroid operations.	Compliant

Table 21-4: Operational mission requirements compliance matrix

21.3 Further Study Areas

Based on the current level of details provided through this CDF study, the following activities were identified as important next steps for the AIM-3 mission development.

- More detailed analysis of the lander deployment and landing philosophy, including GNC, landing velocities and related requirements for the lander design is needed. The main issue is the secondary asteroid's low escape velocity of 15 cm/s, leading to max. landing velocity of about 7.5 cm/s, 50% of the escape velocity, as is the baseline for the original MASCOT lander (NB:the escape velocity of the primary asteroid 40 cm/s, with a spin period of only 2 h leading to large centripetal forces but it is on the secondary that MASCOT should land to contribute to AIM asteroid mitigation goals. For comparison, the MASCOT lander for the Hayabusa 2 mission is designed for a 32 to 36 cm/s escape velocity and delivered via a hover (sampling rehearsal); the escape velocity of 67P/Churyumov–Gerasimenko, on which Rosetta's Philae landed, is 50 cm/s
- Further consolidation of the payload instruments requirements, including those on the lander.
- Preparation of a development plan associated with a more refined schedule.
- The currently used Arianespace Soyuz performance is based on analysis by ESOC, with deduction of a 50 kg "launchability margin" to arrive at the 850 kg maximum spacecraft launch mass used in the AIM-3/3P study. This performance needs to be confirmed by Arianespace a.s.a.p., especially now that the mass of the spacecraft plus adapter is reaching the assumed maximum Soyuz performance.

21.4 Options and Ideas

- In terms of GNC, the following options were identified:
  - Miniature navigation sensing suite
  - Use of the Thermal IR imager and HFR as additional GNC sensors
  - Various other guidance and navigation experiments (some may be implemented in the baseline design, others as end-of-mission experiments)
  - Use of the Optical terminal as a laser altimeter and possibly a camera (multi-use of equipment)

- The AIM-3 spacecraft could potentially act as a communications back-up for DART, with the following possibilities:
  - Receive only with a (modified) AIM-3 - Earth communications transponder
  - Receive only with dedicated receiver for DART?
  - A dedicated intersatellite link with DART with both transmit and receive capability

For what the End of Mission concerns, it was shown that due to the rapidly increasing distance to the Earth, the possible data rates in AIM-3 – Earth communications become unworkably low within a couple of months after the DART impact. In the current baseline mission design the AIM-3 spacecraft is planned to be operated till just over one month after the DART impact event, plus a possible extension of 2.5 months, but much longer than that does not seem to be feasible. In the current design the AIM-3 spacecraft would have insufficient propellant left for the significant orbit changes needed to reach other possible targets. This means that during the mission extension the AIM-3 spacecraft may be used for particularly risk GNC experiments, such as hazard avoidance tests, very low-altitude observations of the DART impact crater and/or other areas of the asteroid, and even a landing of the entire AIM-3 spacecraft on one of the two asteroids (as was in fact done at the end of NASA's NEAR asteroid mission).

22 REFERENCES

RD[1] AIM, Asteroid Impact Mission Assessment, CDF Study Report CDF-129(A), April 2012

RD[2] AIDA Rationale Document http://esamultimedia.esa.int/docs/gsp/completed/AIDA_MissionRationale_InterimRelease.pdf

RD[3] Final Report of Asteroid Impact Monitoring Environmental and Instrumentation Requirements study: https://www-n.oca.eu/michel/AIMReport_Final.pdf]

RD[4] Deep Space Instrument Design for Thermal Infrared Imaging with MERTIS, Infrared Remote Sensing and Instrumentation XIX, 2011, I. Walter et al.

RD[5] FANTINA Part II: Payload design and development, Marco Polo Proposal, 2012, A. Herique et al.

RD[6] Optical Deep Space Link for AIM System Architecture, Link Analysis and Preliminary Design Concept, AIM CCN, 2014

RD[7] OPTEL-μ: A Compact System for Optical Downlinks from LEO Satellites, Dreischer et al.

RD[8] ISIS ISIPOD 2U CubeSat Deployer Product Overview

RD[9] AIM document TBD

RD[10] Entry on 65803 Didymos in Johnston's Archive, courtesy Wm Robert Johnston, Link: http://www.johnstonsarchive.net/astro/astmoons/am-65803.html

RD[11] P. Scheirich, P. Pravec: Icarus 200 (2009) 531-547: Modeling of Lightcurves of Binary Asteroids

RD[12] Executive Summary Report of the activity "GNC Technology for NEO Proximity, Landing and Sampling Operations - Phase 2", Executive Summary, J. Gil-Fernandez (GMV), v.1.0, 17/06/2014

RD[13] ECSS-E-50-05C, *Radio Frequency And Modulation Standard*, 06 March 2009

RD[14] ECSS-E-50-01C, Telemetry synchronization and channel coding. 31 July 2008.

RD[15] ECSS-E-50-04C, Telecommand protocols synchronization and channel coding, 31 July 2008

RD[16] PN Ranging System, CCSDS 414.0-G-1, March 2010

RD[17] SCOC3: a space computer on a chip, DATE 2010, F. Keobel, JF. Coldefy

RD[18] Koen Puimège, Jo Bermyn, "The ADPMS Ready for Flight: An Advanced Data & Power Management System for Small Satellites and Missions," Proceedings of the 23nd Annual AIAA/USU Conference on Small Satellites, Logan, UT, USA, Aug. 10-13, 2009, SSC09-V-4

RD[19] OBC750 Flight Computer, http://www.sstl.co.uk/Products/Subsystems/On-Board-Data-Handling/On-Board-Computer/OBC750-LEO-flight-computer

RD[20] High Speed Data Recorder, http://www.sstl.co.uk/Products/Subsystems/On-Board-Data-Handling/Mass-Memory/High-Speed-Data-Recorder--HSDR-

RD[21] Advanced Flexible I/O System, TEC-ED Final Presentation day, June 2011

RD[22] Vision-based navigation for approach and landing on small bodies, Workshop on GNC for Small Body missions, January 2009 ESTEC

RD[23] 'AIM CDF Study Report', ESA, April 2012.

RD[24] 'Spacecraft Thermal Control Handbook, Volume I: Fundamental Technologies', 2nd edition, edited by D. G. Gilmore, 2002.

RD[25] Technology Readiness Levels Handbook for Space Applications, TEC-SHS/5551/MG/ap, Issue 1 Revision 6, Dated September 2008

RD[26] ECSS-M-80C Risk Management

RD[27] TEC-SYC Cost Risk Procedure, TEC-SYC/5/2010/PRO/HJ, February 2010

RD[28] ESA Cost Engineering Charter of Services, Issue 4, TEC-SYC/12/2009/GRE/HJ

RD[29] Reducing Space Mission Cost, Wertz & Larson, Space Technology Library

23 ACRONYMS

Acronym	Definition
ACS	Attitude Control System
ADPMS	Advanced Data & Power Management System
AIDA	Asteroid Impact & Deflection Assessment
AIM	Asteroid Investigation Mission
AIV	Assembly Integration Verification
AMIE	Advanced-Moon micro-Imager Experiment
AOCS	Attitude and Orbit Control System
ASW	Application SoftWare
ATB	Avionics Test Bench
AU	Astronomical Unit
AVM	Avionics Verification Model
AVUM	Attitude Vernier Upper Module (VEGA Upper Stage)
BCDR	Battery Charge/Discharge Regulator
BCR	Battery Charge Regulator
BDR	Battery Discharge Regulator
BP	Bi-Propellant
BPE	Beam Pointing Error
BSP	Board Support Package
CCD	Charge-Coupled Detector
CDMU	Command and Data Management Unit
CER	Cost Estimation Relationship
CFRP	Carbon Fibre Reinforced Polymer
CoG	Centre of Gravity
COMMS	Communication
COPINS	Cubesat Opportunity Payloads
COTS	Commercial-off-the-shelf
DART	Double Asteroid Redirection Test
DCP	Detailed Characterisation Phase
DDOR	Delta Differential one way Ranging

Acronym	Definition
DHL	Data Handling
DM	Dual-Mode
DMM	Design Maturity Margin
DoD	Depth of Discharge
DOR	Differential One Way Ranging
DSA	Deep Space Antenna
DSM	Deep Space Manoeuvre
DSHM	Deep Space Hibernation Mode
DSM	Deep Space Manoeuvre
DSN	Deep Space Network
DTE	Direct to Earth
ECP	Early Characterisation Phase
ECSS	European Cooperation on Space Standardisation
EGSE	Electrical Ground Support Equipment
EIRP	Equivalent Isotropic Radiated Power
EMC	Electro-Magnetic Compatibility
EoL	End of Life
EPC	Electrical Power Conditioner
ESA	European Space Agency
ESOC	European Space Operations Centre
EU	Electronic Unit
FCT	Flight Control Team
FDIR	Failure Detection, Isolation and Recovery
FDS	Flight Dynamics Support
FDVV	Fill, Drain, Vent Valve
FEM	Finite Element Model
FM	Flight Model
FoV	Field of View
FPGA	Field Programmable Gate Array
FPL	Free Path Loss
G/S	Ground Station

Acronym	Definition
GNC	Guidance Navigation and Control
GSDR	Ground Segment Design Review
GSE	Ground Support Equipment
GSP	General Studies Program
GSTVi	Ground System Test and Validation infrastructure
GTO	Geostationary Transfer Orbit
HFR	High Frequency Radar
HGA	High Gain Antenna
HKTM	Housekeeping and Telemetry
HP	High Pressure
HPA	High Power Amplifier
OBC	On-Board Computer
OBDH	On-Board Data Handling
ICP	Initial Cruise Phase
IMU	Inertial Measurement Unit
IOD	In-Orbit Demonstrator
IR	Infrared
ISL	Inter Satellite Link
ITAR	International Traffic in Arms Regulations
LCT	Laser Communication Terminal
LEO	Low Earth Orbit
LEOP	Launch and Early Operations Phase
LFR	Low Frequency Radar
LGA	Low Gain Antenna
LIC	Launcher Injection Correction
LIDAR	Meaning of the acronym
LOS	Line Of Sight
LP	Launch Period
LP	Low Pressure
LPC	Launch Period Close
LPO	Launch Period d Open

Acronym	Definition
LU	Laser Unit
MASCOT	Mobile Asteroid Surface Scout
MCS	Mission Control System
MEOP	Maximum Estimated Operating Pressure
MGA	Medium Gain Antenna
MIB	Minimum Impulse Bit
MIMU	Miniature Inertial Measurement Unit
MLI	Multi-Layer Insulation
MMH	Monomethylehydrazine
MOC	Mission Operations Centre
MON	Dinitrogen Tetroxide
MP	Mono-Propellant
MPPT	Maximum Power Point Tracker
N.P.	North Pole
NAC	Narrow Angle Camera
NEO	Near Earth Object
OBCP	On-Board Control Procedure
OBSW	On-Board Software
OPTEL-D	Optical Link for Communications
OSR	Optical Solar Reflector
PCDU	Power Conditioning and Distribution unit
PFM	Protoflight Model
PL	Payload
PLA	PayLoad Adapter
PMD	Propellant Management Device
PN	Pseudo Noise
POE	Project Owned Event
PSS	Power Sub System
R.A.	Right Ascension
RARR	Range and Range Rate (Doppler)
RCS	Reaction Control System

Acronym	Definition
RF	Radio Frequency
RFDN	Radio Frequency Distribution Network
RHCP	Right Hand Circular Polarization
RNG	Ranging
RTOS	Real Time Operating System
RTU	Remote Terminal Unit
S/C	Spacecraft
S^3R	Sequential Switching Shunt Regulator
SA	Solar Array
SAA	Solar Aspect Angle
SADE	Solar Array Drive Electronics
SADM	Solar Array Drive Mechanism
SAVOIR	Space Avionics Open Interface Architecture Initiative
SMART-1	Small Missions for Advanced Research in Technology-1
SOE	Spacecraft Operations Engineers
SOM	Spacecraft Operations Manager
SPACON	Spacecraft Controller
SPFF	Single Point Failure Free
SSE	Sun – Spacecraft – Earth angle
SSO	Sun-Synchronous Orbit
SSTO	Sun-Stabilised Terminator Orbit
SW	Software
TC	TeleCommand
TCS	Thermal Control System
TIRI	Thermal Infra-Red Imager
TM	Telemetry
TRL	Technology Readiness level
TWTA	Traveling-Wave Tube Amplifier
VIS	Visual Imaging System
WC	Worst Case