Micro/Mini-Satellite Interplanetary Mission

PROBA-IP

Executive Summary

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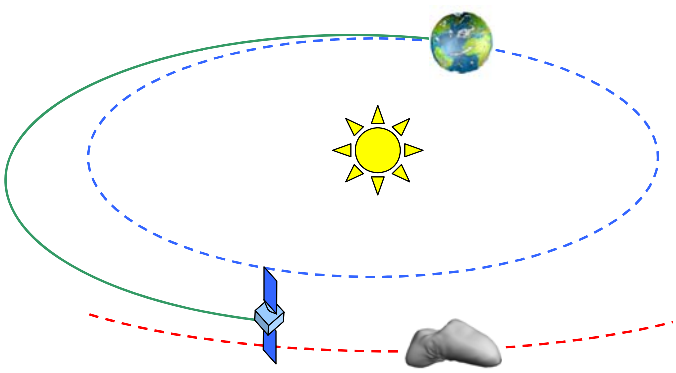

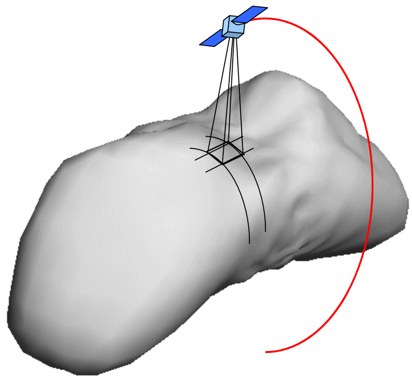

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Acronyms

Acronym	Meaning
ACU	Autonomous Control Unit
AOCS	Attitude and Orbit Control System
AIV	Assembly, Integration and Validation
BOL	Beginning Of Life
CAN	Controller Area Network
COB	Centre of Brightness
COM	Centre of Mass
COTS	Commercial Off-The-Shelf
CPS	Chemical Propulsion System
CPU	Central Processing Unit
DHS	Data Handling System
DSN	Deep Space Network / Navigation
EOL	End Of Life
EPS	Electric Propulsion System
FDIR	Failure Detection, Isolation and Recovery
FOV	Field of view
GNC	Guidance, Navigation and Control
G/S, GS	Ground Segment, Ground Station
HGA	High-Gain Antenna
HPA	High-Power Amplifier
IMU	Inertial Measurement Unit
IP	Image Processing / Interplanetary
LEO	Low Earth Orbit
LEOP	Launch and Early Orbit Operations
LGA	Low-Gain Antenna
LOS	Line of Sight
LPF	Lisa Pathfinder

Acronym	Meaning
LPM	LISA Pathfinder Propulsion Module
NAC	Narrow Angle Camera
NEO	Near Earth Object
PPU	Power Processing Unit
O/B	On board
OBAN	On Board Autonomous Navigation
OBC	On-Board Control
OBCP	On board Control Procedures
OBDH	On-Board Data Handling
OBR	On-Board Resource
OBRM	On-Board Resource Management
PCDU	Power Conditioning and Distribution Unit
PPU	Power Processing Unit
PRM	Propulsion Module
RCS	Reaction Control System
RTE	Radio Tracking Experiment
RTU	Remote Terminal Unit
RW	Reaction Wheels
SA	Solar Array
SADM	Solar Array Driving Mechanism
S/C	Spacecraft
SEP	Solar Electric Propulsion
SEPM	Solar Electric Propulsion Management
STR	Star Tracker
TRL	Technology Readiness Level
TTC,TT&C	Telemetry, Tracking and Command
WAC	Wide Angle Camera

1. INTRODUCTION

1.1. Abstract

ESA's PROBA programme has been based so far on the in-flight validation of new space technologies for Earth-bound missions. Now, ESA is promoting a preliminary study for an interplanetary mission (Proba-InterPlanetary or Proba-IP for short) for close-up reconnaissance of a Near Earth Object (NEO) within the same programme. The main objective of the Proba-IP mission is the in-orbit validation of autonomous onboard guidance, navigation and control technologies for interplanetary cruise and for the targeting of other celestial bodies, primarily using onboard optical systems technologies. Proba-IP will also demonstrate micro/mini-spacecraft's capabilities into the interplanetary missions' domain. Additionally to the previous objectives some scientific objectives are considered as potential experiments in order to determine target's properties as shape, rotation state, gravity field, surface properties and accurate orbit determination by means of a Radio Tracking Experiment (RTE). If resources are available the RTE should allow a precise determination of the asteroid orbit. Present paper provides the proposed mission and system design to cope with the mentioned goals. Current design foresees a three-year mission launched with VEGA in 2015. Escape from Earth is achieved by means of an upper stage. 2.5 years would be devoted to the low-thrust transfer to the asteroid and six months for in-orbit operations. RTE would expand for three months in a photo-gravitational stable orbit. Current estimate of spacecraft wet mass is 385 kg and dry mass of about 300 kg. This study started in mid-2008 and finished at the end of 2009.

1.2. Study Description

The utilization of small and low cost platforms developed and launched in a short timeframe for the in-flight validation of new space technologies for Earth bound missions has been successfully demonstrated within ESA's Proba Programme [1]. As an advance within such context, a Near Earth Object (NEO) mission has been envisaged as Proba-InterPlanetary (Proba-IP). Close-up reconnaissance of NEOs based on relatively inexpensive small satellites would in fact represent a valuable technology validation experience, beneficial to the success of future major scientific and exploration missions. This type of missions could also provide elements for the assessment of how well the threat posed by NEOs to the Earth in the future can be defined and counteracted, and thus pave the way for more complex systems executing specific mitigation strategies as done recently in the Don Quijote feasibility study [2]. This Proba-IP study is then aimed to the preliminary design of this kind of missions using a micro/mini-satellite approach.

The main objective of the Proba-IP mission is the in-orbit validation of autonomous onboard guidance, navigation and control technologies for interplanetary cruise and for the targeting of minor celestial bodies, primarily using onboard optical systems technologies. Such technologies will foster the reduction in operation costs of Solar System's exploration missions and enable characterization of NEOs. Proba-IP will also demonstrate micro/mini-spacecraft's capabilities into the interplanetary missions' domain. Current target for the mission is Apophis.

The study represents a preliminary mission and system study promoted by ESA and performed by a consortium lead by DEIMOS Space (Spain) and including EADS CASA Espacio (Spain), Astrium GmbH Satellites (Germany), SSC (Sweden), SSTL (UK) and Scisys (UK).

1.3. Mission Objectives

The Proba-IP mission will be designed to implement and validate the following technologies and functionalities:

1. Onboard Guidance, Navigation and Control (GNC) technology elements for autonomous spacecraft navigation, guidance and control in interplanetary cruise primarily using onboard optical systems,
2. Autonomous targeting of, and rendezvous with, a NEO,
3. Autonomous achievement and maintenance of a safe close orbit around the target object,
4. Use of solar electric propulsion (SEP) system in as much as possible mission phases after Earth orbit escape,
5. Autonomous onboard resources management (OBRM) and Failure Detections, Isolation and Recovery (FDIR), including autonomous SEP management,
6. Application of novel ground support and operations concepts to support the autonomy level described above,
7. Demonstration of methodology, tools and infrastructure required for the development and implementation of an interplanetary mission including ground validation of the mission's autonomous characteristics.

Additionally to the previous objectives some scientific goals are considered as potential experiments in order to determine target's (Apophis) properties as shape, rotation state, gravity field, surface properties and accurate orbit determination by means of a Radio Tracking Experiment (RTE). If resources are available the RTE should allow a precise determination of the asteroid orbit enough to ascertain whether Apophis will actually represent a threat to Earth.

1.4. Mission Requirements

Main mission requirements attend to the implementation of the abovementioned autonomy experiments. Mainly, the ones related to onboard autonomous GNC (AGNC), which shall extend to as many phases of the mission as possible. In that perspective, AGNC experiments are proposed for the interplanetary phase, the arrival phase, the insertion phase and the in-orbit phase. Such experiments are described in a later section but in summary affect the definition of the spacecraft (through the sensors / actuators and O/B software), the operations and the cost and programmatics of the mission.

Regarding timely aspects, the baseline mission shall depart in 2015 and extend for a maximum of 3 years with up to 2.5 years transfer time and a minimum of 6 months of operations at the asteroid. European Launcher Policy shall be followed and cost of the mission minimized to maximum possible. Cost minimization imposes the use of a small dedicated launcher. Due to the fact that the available small launchers do not provide escape performances, an upper stage to escape from Earth is also required. Solid motors and liquid stages have been both considered for this task.

Spacecraft mass minimization leads to the selection of a target NEO very close to Earth to diminish the impact of the transfer propellant needs. Also, selecting a very close NEO allows reducing the thrust level requirements and thus allows decreasing the size of the power subsystem. This particular aspect has been pushed as much as possible in order to provoke a cascade effect in the reduction of the platform requirements. Regarding the technology aspects, proposed equipments for mission enabling technologies currently under development in Europe shall have an appropriate technology maturity of TRL 5 at end of Phase B (currently, end of 2010).

1.5. Reference Documents

[1] ESA. Proba Missions website, http://www.esa.int/SPECIALS/Proba/index.html

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[5] S. Chesley, "Potential Impact Detection for Near-Earth Asteroids: The Case of 99942 Apophis (2004 MN4)", *Proceedings of IAU Symposium No. 229: Asteroids, Comets, Meteors*, 2005.

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[8] G. Bellei, J.L. Cano, M. Sánchez, "Operational Orbiting Strategies About Minor Bodies", *Presented at the 21st International Space Flight Dynamics Meeting,* Toulouse, France, September 30 - October 2, 2009.

[9] P.S. Jørgensen, J.L. Jørgensen, T. Denver, P.v.d. Braembuche, "The Micro Advanced Stellar Compass for ESA's Proba 2 Mission", *Proceedings of the 5th IAA Symposium on Small Satellites for Earth Observation*, Berlin, Germany, April 4-8, 2005.

2. MISSION ARCHITECTURE

Based on above considerations, the following main building blocks were identified for the Proba-IP mission (Figure 2-1):

- **Low-cost platform**, based on micro/nano-elements and on high integration of systems and subsystems, so as to provide the required on-board resources within constrained mass/volume and access low-cost launch opportunities.
- **Low-cost launcher**, to contribute to the overall mission cost minimization. Currently ESA's VEGA [3] is the baseline, although other possibilities have been analysed as Rockot, Dnepr and PSLV. VEGA selection is based on optimal combination of the application of European Launcher Policy and adequacy to mission requirements.
- **Upper stage**, needed for Earth escape. The current baseline is the Lisa Pathfinder Propulsion Module (LPM), which provides the allowed flexibility to the escape phase and perfectly suited performances. Other options considered included STAR-37 and STAR-48 motors which were found not applicable due to short performances in first case and need to validate the motor for lower propellant loads in the second case.
- **On-board autonomy**, covering: autonomous GNC, on-board resource management, SEP management and FDIR.
- **Novel ground support and operations concepts**, whenever necessary to support the envisaged S/C autonomy level, including nominal ground operations work automation to reduce the operational workload and cost.

Depending on the on-board resources remaining available, a **Radio Tracking Experiment (RTE)** and additional **opportunity payloads** for the imaging of the target asteroid and/or its physical properties' characterization can also be accommodated, to further maximize the mission return.

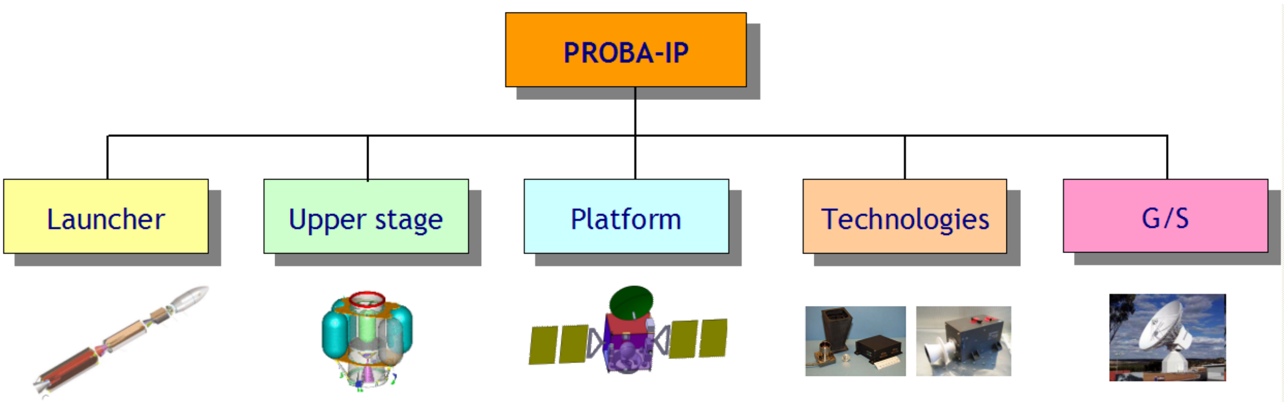

Figure 2-1: Building Blocks of the Proba-IP Mission Architecture

3. MISSION ANALYSIS

First activity at Mission Analysis level was the selection of the target asteroid in the population of NEOs. From an initial set of 5455 bodies extracted from [4], a set of criteria was set up to diminish the number of target objects. A first criteria limiting the asteroid orbit inclination to 5º, the semi-major axis between 0.85 AU and 1.4 AU and the eccentricity to a maximum of 0.37 left the population in only 194 bodies. This criterion fosters reachability with minimum propellant requirements. A second criterion was established in terms of asteroid size through absolute magnitude (and with assumptions on typical asteroid albedos), limiting to asteroids in the range of 200 m-1000 m. The inferior limit in size was imposed to avoid making the in-orbit experimentation much more difficult, as the dynamical environment worsens sensibly for very small asteroids. Last criteria left a population of only 25 asteroids, including Amors, Atens and Apollos in the list.

Low-thrust trajectories to the 25 mentioned minor bodies were analysed in order to check the option which would impact to the minor extent the design of the platform. Assumptions on the launcher and upper stage performances and the low-thrust propulsion system performances were taken to compute the transfers both launching in 2015 and 2017 and then to rank the asteroids in terms of final mass at the asteroid.

After this assessment, best-suited asteroids launching in 2015 were: Apophis, 2002 TC70, 1989 ML and 2001 CC21. Having observed the presence of Apophis in the list with very good arrival performances to the asteroid in terms of mass and the interest of the asteroid from a Planetary Defense point of view [5], it was selected as the baseline for the mission launching in 2015. 1989 UQ was selected as back-up asteroid for launch in 2017 in case any contingency would require delaying the launch window (Apophis was not ranked sufficiently well in 2017 opportunity). As an additional feature, both mission profiles are very similar.

Launcher and upper stage selection impacts to a large extent the mission performances in terms of arrival mass to the asteroid, but also the selection of the low-thrust system. Considered options included the use of propulsion systems providing thrust in the range of 20 mN, value which is small but preliminary found suitable for a transfer of 2.5 years to a close NEO. In this context, following thrusters were considered: PPS-1350, RIT-10, T5 and HEMPT. The first thruster was not further pursued due to the fact that the assumed thrust level implied functioning close to the minimum operational engine limit. Ion thrusters of the RIT-10 / T5 type were found very applicable elements for the mission in terms of performance but were ruled out due to the inherent high complexity they imply in their management. This would make the performance of the SEP management experiment too complex and out of the scope of such autonomy-driven demonstration goal. Thus, the option currently considered is based on the use of the HEMPT [6].
Obtained trajectory profile with the proposed combination of VEGA, LPM and HEMPT is given in Figure 3-1. Earth infinite escape velocity is 1.89 km/s and escape date on 25/01/2015. Transfer time extends for the maximum 2.5 years up to 26/07/2017. Encounter with Apophis is at its aphelion starting then the 6-month in-orbit demonstration campaign. Distances to Sun range from 0.72 AU to 1.10 AU and maximum distance to the Earth is 2.08 AU as observed in Figure 3-2.

Required power devoted for thrusting is represented in Figure 3-3 as well as the total required power for the whole spacecraft. An operational limit is imposed at 1200 W to avoid that the temperature in the solar panels exceeds a threshold. This corresponds to a distance to the Sun of 0.83 AU. At closer distances the income power is controlled by rotating the solar panels.

The escape phase is performed by a staging of the escape manoeuvre in six burns. Between each burn a minimum of two days are left for orbit determination and TM/TC, thus amounting this phase to about 2 weeks duration.

Arrival to the asteroid will be performed along the Sun-asteroid line by properly steering the thrust law, which thus allows having maximum illumination and more uniform environmental conditions at arrival. Last part of the trajectory will include a hovering phase similar to the one performed by Hayabusa [7] in order to approach the asteroid in a controlled manner and initiate from there the in-orbit operations.

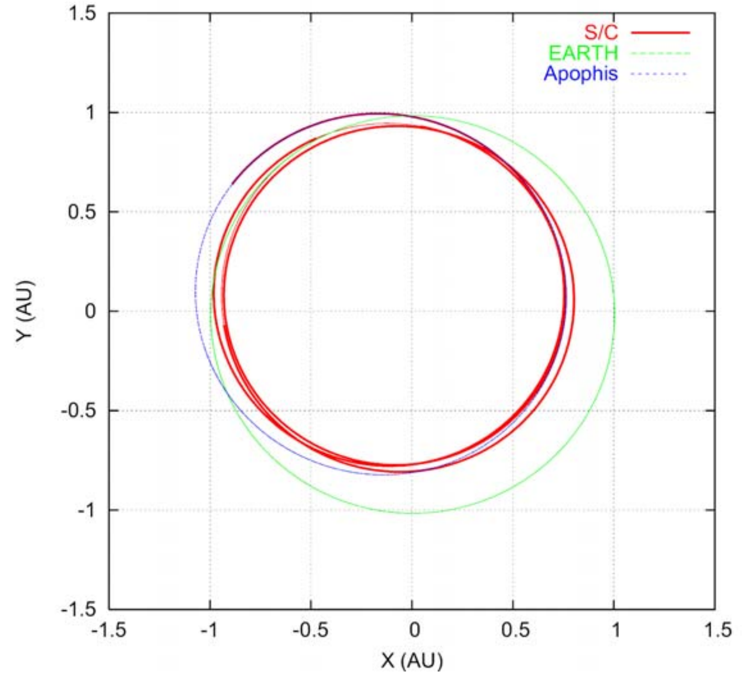

Figure 3-1: Ecliptic trajectory projection to Apophis

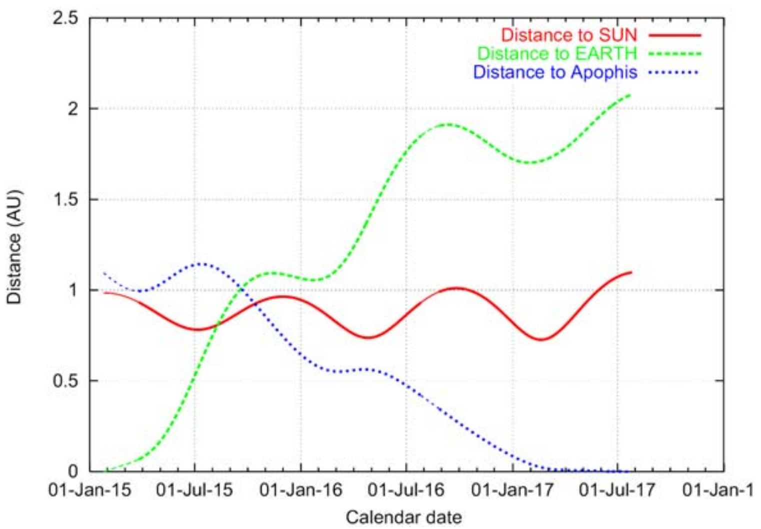

Figure 3-2: Distances to Sun, Earth and asteroid along the transfer to Apophis

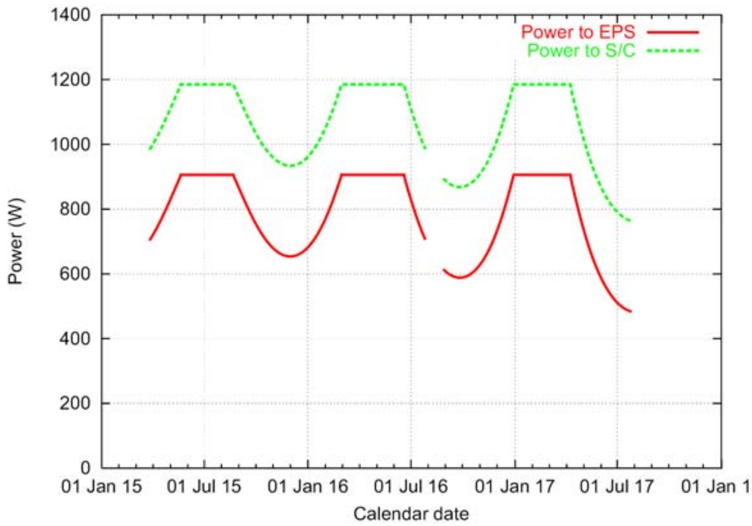

Figure 3-3: Power devoted to EPS and to the rest of the spacecraft along the transfer

Regarding asteroid in-orbit autonomous operations, the results obtained from dedicated stability analysis of uncontrolled polar orbits [8] showed the necessity of some kind of orbital control to ensure the safety of in-orbit operations. Autonomous orbit determination will be performed by means of asteroid feature tracking algorithms and the help of an altimeter. Regarding the control, the performance of a simple altitude control, based on a dead-band scheme with spherical control boxes was evaluated [8] to prevent at least the two most dangerous events of escape and crash onto asteroid surface and maintain a quasi-circular orbit shape. Further manoeuvres were evaluated also in [8] to allow that the orbital plane orientation was kept within established boundaries. Therefore, asteroid in-orbit operations will be carried out by introducing frequent control manoeuvres to avoid that the S/C: a) collides with the asteroid, b) escapes from it, c) enters into asteroid eclipse and d) considerably modifies orbital plane orientation.

Finally, the S/C will be directed to a photo-gravitational stable terminator orbit to perform an RTE during three months. Stability analysis of such orbits is also addressed in [8].

In summary, arrival, hovering, in-orbit and RTE phases will all include dedicated autonomous GNC experimentation.

4. TECHNOLOGY SOLUTIONS

4.1. General Concept

Going from a non-autonomous concept to an autonomous requires developing new technologies that implement the autonomy capability. The rationale then is first to identify what is the autonomy concept pursued and then to address the driving technological needs enabling the implementation of this concept. In the case of Proba-IP, as a technology demonstration mission, the autonomy concept is based on demonstrating technology maturity to be used in future missions. In this context, four main technologies have been selected for deep investigation within the study: GNC, OBRM, FDIR and SEP management. Figure 4-1 shows these four technologies and the relationship with the rest of technological elements of the mission. Following paragraphs are dedicated to present in detail which is the baseline considered for each element.

4.2. Autonomous GNC

The Proba-IP mission shall implement at the maximum possible extent, and validate in flight, autonomous on-board guidance, navigation and control functions. The main GNC mission requirements and design drivers are: a) demonstrating autonomous GNC techniques during the cruise, search, insertion and in-orbit phases; b) allowing precise and safe in-orbit operations and support an accurate characterization of the asteroid target; c) increase of the orbiter on-board autonomy in view of the reduction of operations cost. While the pointing and stability specifications do not go beyond typical interplanetary mission requirements, the GNC challenges of the mission are numerous and lie on the following:

- The high autonomy requirement aiming to minimize ground involvement and show on-board autonomy capabilities
- The faintness of the target with its significant impact (especially for the detection/ far approach phases) on the performance of the navigation sub-system and the design of the image processing algorithms (IP)
- The imprecise knowledge of target characteristics: orbit size, mass and rotational state
- The small and irregular gravity field of the target (size ~100 m to ~300 m, increasing the effects of SRP disturbance and Orbiter/target interaction)
- The possibility to operate at low altitude over the asteroid, which augments the collision risks and the imaging constraints during the in-orbit operations.

Baseline GNC being proposed for the different mission phases, after detailed analyses and trade-offs follows.

Cruise Phase: during this phase, autonomous Vision Based Navigation is implemented based on triangulation performed by the observation of beacons (e.g. asteroids) with known position and sufficient angular separation so that the spacecraft location can be deduced, supported by a navigation filter that accommodates different observation times. As the distance range is far from the main belt of asteroids, the navigation performances are intrinsically poor for average camera performances, and dedicated analysis of the suitability of available cameras was made. The trade-off yields that a good option is the use the μASC of DTU [9], which combines low weight with acceptable navigation performances in the order of few thousand kilometres in position.

Far and Close Approach: during far approach phase, the target becomes closer than any other objects and target observation provides the highest positioning accuracy in the direction perpendicular to the LOS vector. The observability across track is achieved from the LOS camera measurements. However, along track observability can be also derived during far approach by the use of an approach trajectory with varying geometry ("dog-leg") that is different from typical flyby trajectories. In fact, Proba-IP approach is a rendezvous, whose trajectory is curved in the sense that the approach geometry naturally varies, permitting to estimate along and across track navigation errors. By taking into account these navigation errors a complete low-thrust strategy for trajectory corrections has been studied. Same camera as for the cruise phase can be used.

Nearer to the asteroid, the asteroid image itself starts filling more than one pixel: this defines the Close Approach start, which is characterized by the use of centre of brightness (COB) image processing technique to compute LOS.

In-orbit operations: for in-orbit operations the main trade-offs considered were: a) the selection of the in-orbit navigation technology whether based on the use of a WAC or to continue using the cruise camera also nearer the asteroid; b) the selection of a range finder (altimeter/LIDAR) or the use of an IMU (accelerometers) to solve the scale ambiguity of the camera. A technology demonstration scenario was built made of different types of manoeuvres around the asteroid. The analyses supported the suitability of the μASC with a different focusing set-up for in-orbit operations making use of COB navigation (plus altimeter), which is simple and provides relatively good accuracy. At closer distances, a feature-based navigation complemented by the measurements of an altimeter was considered the best option. Altimeter measurements are assumed below 50 km distance to the asteroid.

GNC/ACS Modes: Only operational modes are considered here (see Figure 4-2), commissioning modes are not (stand-by mode and rate reduction mode). The phase when the spacecraft is attached to LPM is not considered here either.

- Safe mode (SM). This has to cover different needs: a) Sun Acquisition and Survival mode: the Sun sensors are used to acquire the Sun and rotate the satellite and the solar arrays in the proper direction. It is necessary to cover the case of STR outages and in case of other major outage, b) to perform a 3-axis control, possibly with relaxed performances (with respect to normal mode). STRs are mandatory for this mode
- The Collision Avoidance Mode (CAM). This mode is required to handle the risk of collision with the asteroid in case of a severe contingency.
- Inertial pointing mode (IPM). This mode implements inertial pointing and slew and unloading capabilities in order to point to different inertial directions: a) Inertial Pointing for Navigation in which the satellite is inertially pointed, used for taking images for Cruise and Approach navigation purposes (beacons imaging) and b) Inertial Pointing Sub-Mode for ground communication and science in which the satellite is inertially pointed. It is used for communicating with ground and RTE
- The Wheel-Off-loading Sub-Mode: Allows the regular off-loading of the wheels using propulsion.
- Slew Sub-mode: it is a supporting mode used to perform attitude manoeuvres.
- The Asteroid-Pointing Mode (APM). It points the navigation/science camera and instruments toward the asteroid and maintains the pointing. This mode can be used both for Approach and In-orbit Navigation and Science

- The Electrical Propulsion Control Mode (EPCM). This mode is used during Cruise and Approach to perform the thrust steering of the electric propulsion
- The Orbit Control Mode (OCM) is used for the In-Orbit Navigation when manoeuvres have to be performed to do orbit control.

It is not expected that the phases of the mission when the spacecraft is completely supervised by ground and so not autonomous (e.g. for engine testing) need specific operational modes. They are managed by opportune switching to IPM whenever the antenna cannot be pointed to Earth due to the orientation of the spacecraft.

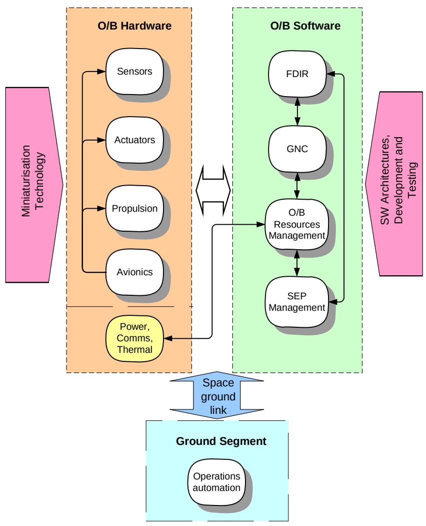

Figure 4-1: Diagram with the relation between the technologies considered in Proba-IP

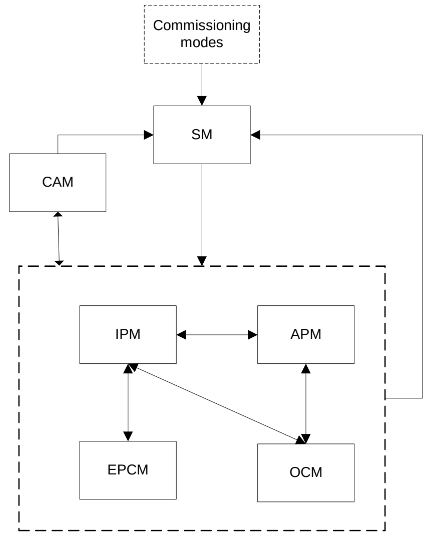

Figure 4-2: Operational modes diagram

4.3. Autonomous SEP Management

The SEP system will be mission relevant for the cruise and targeting phase as it will provide the required ΔV for trajectory corrections. For Proba-IP special requirements for the SEP are derived from the autonomous nature of the mission and the small satellite approach: autonomous nominal operations, autonomous subsystem failure detection, isolation and recovery (FDIR) and minimum system dry mass. In this context, autonomous SEP management is one of the key technologies for autonomous spacecraft operation, in particular related to the system capability to deal with unexpected transient malfunctions.

Most of the missions so far have used the SEP system embedded in their overall AOCS and On-Board Data Handling System. Even long-term missions like SMART-1 relied on direct control of most system parameters by ground command and therefore strongly linked their SEP management logics with the spacecraft control. In order to provide autonomous SEP System operations for the Proba-IP mission, system complexity must be significantly reduced e.g. by reduction of interfaces, nominal operation modes and complex FDIR procedures. Several concepts and architectures for the implementation of an autonomous SEP management system have been analysed and traded-off in the study, based on two main options:

- SEP control by spacecraft recourses (e.g. OBC, or Autonomous Control Unit (ACU))

- SEP inherent control (Intelligent PSCU / PPU (iPPU, iPSCU)

Both options imply an increase in control complexity but are envisaged to allow the SEP system running in optimum conditions considering all failure cases. The preferred solution is to implement the SEP autonomy functions in the OBC software. This allows the procurement of standard OBC and PPU/PSCUs. The Autonomy Control Function (ACF) within the OBC must be designed such that it gathers all TM data from the SEP system, processes it and controls the SEP system with mid and high-level commands (low level control will be done by PPU/PSCU). Spacecraft relevant parameters (operational modes, performances, FDIR history, etc) will be directly provided by the OBC software to the ACF.

4.4. Autonomous OBRM

At a system level, OBRM (or any operations) can conceptually be divided into two parts. The first deals with reacting to and managing the current state by maintaining parameter models of on board resources, monitoring these for conditions, generating software events and allowing activity triggering in reaction to those events. The second part is more computationally complex, requiring the support of software simulated hardware models and dedicated OBC resources. It deals with looking ahead into future, projecting hardware simulation model, evolving resource parameters models accordingly, predicting events and planning. A trade-off involving cost, complexity and maturity was carried-out during the study to select the most suitable OBRM technologies to be implemented on Proba-IP for demonstration purposes. Currently a total of five different OBRM functions are envisioned to handle:

- Software Heater Control (Intelligent Heater Control)
- Contact Prediction (Contact Plan)
- Propellant Monitoring
- Payload Timeline Validation, Control and Repair (TVCR)
- Energy monitoring

4.5. FDIR

The Proba-IP mission requires a high level of autonomy throughout the majority of mission phases. As a result, the development of a relatively simple, robust and testable FDIR concept is required to ensure that the spacecraft can maintain its desired operational state for long periods of time without frequent interventions from ground operators. At the same time it shall be compatible with the testing of new FDIR technology.

For Proba-IP no credible single point failure can lead to the loss of the mission. This results in a generally dual redundant solution. This means all units are duplicated (when practical); either internally or by flying two identical modules. This allows switching between units if an anomaly is detected on the primary one. For subsystems where it is impractical to fully duplicate equipment (e.g. batteries, solar arrays) a graceful redundant solution is adopted. This means that performance of the spacecraft will degrade gradually if failures occur (e.g. loss of a string on the solar array). The technologies selected for the FDIR concept are: CAN FDIR, AOCS FDIR, Telemetry Monitoring, OBC watchdog and Propulsion Module watchdog.

5. PROBA-IP PLATFORM

5.1. Proba Missions

The Proba missions of the European Space Agency are focused on the constant increase in miniaturization and integration of platform systems and subsystems. Regarding the platform definition, this is taken into account adopting the latest technologies and techniques, but keeping in mind the cost constraints of a demonstration mission.

The first step of the cost optimisation is to use a micro or mini-satellite platform, having the necessary onboard resources within a constrained mass and volume, in order to access low-cost launch opportunities. For this reason an analysis of existing mini-platforms applicable to this mission was carried out, performing corresponding trade-offs in terms of cost, performance, availability and flexibility.

The European candidate platforms have been based on the successful SMART-1 and GIOVE-A missions, and the in-progress PRISMA and Proba-3 programs. The SANCHO study developed by the ESA's CDF team has been included in the optimisation in order to complement the rest of non-covered technical aspects of the design of Proba-IP platform.

5.2. Technical References and Goals

The above proposed European platform designs, including their related technologies, programmatic / cost methodologies, and subsystems heritage, are available for using on the Proba-IP mission. The considered Launch Segment is based on VEGA as nominal launching vehicle, and its interfaces and available fairing apply to the concept. As for the Ground Segment the goal is the compromise between the low-cost on-ground operations versus the development of the in-orbit technologic payloads related to autonomy for on-board resources management, satellite FDIR, and management of solar electrical propulsion. Finally the LPM is considered as nominal upper stage, and its interfaces apply to the concept, including the separation system based on an 800-interface ring.

Two goals have been considered at System level:

- From the technical point of view, the design has to be based on a micro/mini platform. The associated budgets are focused on a satellite concept in the range of 400 kg and 600 W EOL @ 1 AU for thrusting purposes.
- From the programmatic point of view, the system engineering, development, verification tools, and facilities have to be based on the Light-Sat approach.

The monitoring of the mass, volume and power budgets is a continuous task aimed to minimize these figures during the overall phases of the Proba-IP study. Another objective of this budget monitoring is to be compliant with the maximum accommodation constraints in order to provide mass and power resources to incorporate opportunity payloads to add value to technology demonstration missions in terms of technical or scientific returns.

5.3. Structure and Mechanisms

The primary spacecraft structure is based on a simple, rectangular-box-shaped body with top and bottom decks. The resulting dimensions are 1,000 mm per 1,400 mm per 1,000 mm, and an artistic view is shown in Figure 5-1.

Most of equipment and units are accommodated on the four lateral panels as function of their power and thermal requirements, whereas the main xenon tank is located in the central axis of the satellite to minimize centre-of-gravity shifts during operations, and enable simple balance with the launch fairing. For the rest:

a) The Electrical Propulsion Subsystem (EPS) equipments are located around the central axis and built around the tank location in order to enhance the final spacecraft integration through the bottom panel. The two propulsion engines are mounted out on and under the lower platform, whereas their electronic units are located close to these plasma engines.

b) The reaction control subsystem (RCS) is based on the use of xenon resources working with an architecture of sixteen resistojests (RTJ units), located by pairs in the eight corners of the structure, with an orientation that allows the necessary attitude and translation manoeuvres.

c) The proposed high gain antenna (HGA) for data transmission is mounted on a lateral platform.

d) The technology devices devoted to mapping and characterizing the NEO physical properties during the final approach part of the mission are mounted on the upper platform.

e) Two symmetric wing arrays are attached near to the satellite centre of mass, and each of them is driven by a two-DoF mechanism: two independent SADM units. Each wing array has two solar array panels (SAP), and each SAP is divided in two different topology sections according to the requirements of the final power users.

f) Finally the mechanical interface with the LPM is based on a specific ring adapter attached to the platform, which consists of a cylindrical 800-interface separation subsystem with one simple, robust and redundant separation-point concept. This separation mechanism provides the force to separate the satellite from the propulsion module, once this module has finished its set of burns for Earth escape.

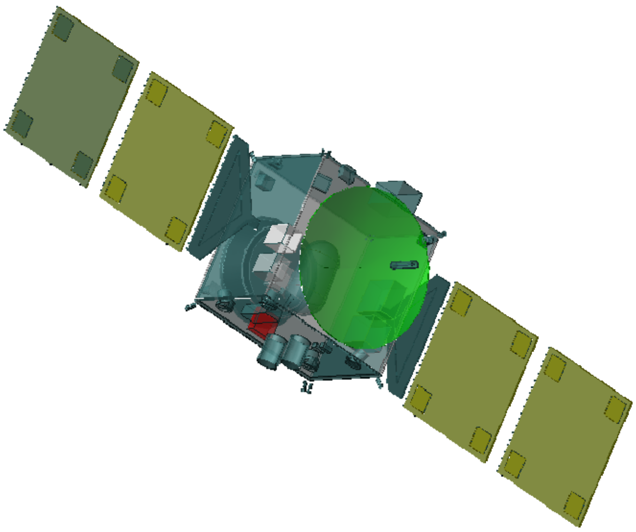

Figure 5-1: Artistic view of the Proba-IP spacecraft

5.4. Data Handling

The data architecture concept is based on a data-transfer using CAN-bus. The core processor is based on already developed units or in-progress development programs. The selected concepts of OBC·750 (SSTL) and LEON·3 (SSC) have been analysed during the study to identify the envelope of their best performances. Only for design purposes, the OBC·750 was chosen as baseline concept to build the data architecture, but the LEON·3 is also a robust feasible concept.

Three identified types of data-interfaces are managed by the processor: CAN, SpaceWire and 1553-B, as it is shown in Figure 5-2.

The on-board computer unit is mounted around a processor redundant board with these elements: TM/TC interface, mass memory, general-purpose input/outputs, and some specific interface boards for data-transfer with the rest of spacecraft.

The intelligence of the satellite is centralized in the on-board computer, which controls the following high-level functions: satellite monitor and control; TM/TC interfaces; data storage, telemetry and compression; autonomous OBRM; GNC, AOC and RCS management; autonomous FDIR; EPS management (control logic, control loop, EP parameters, internal EP FDIR, and housekeeping data control).

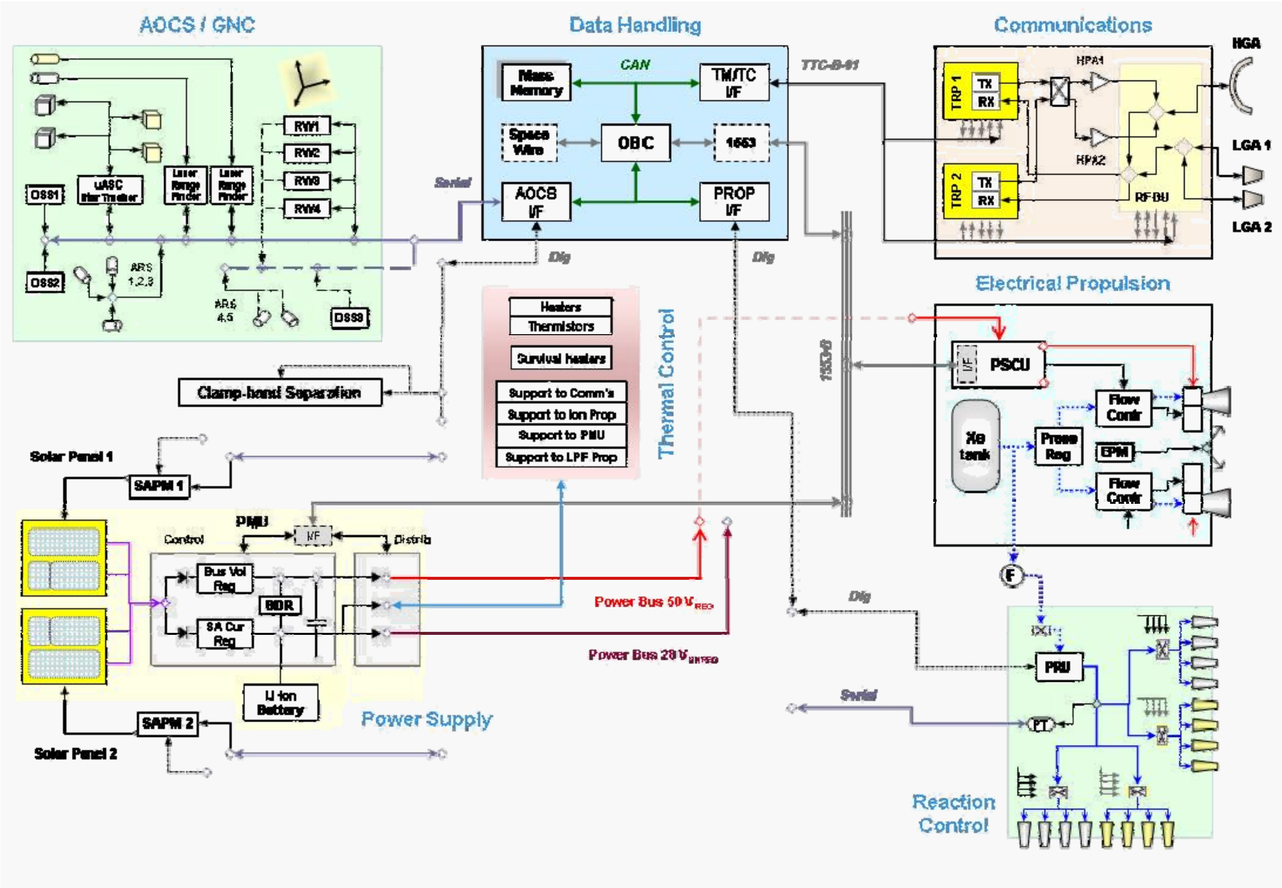

Figure 5-2: Scheme of data architecture

5.5. Power Supply

The Power Supply Subsystem (PSS) elements include the two deployable solar arrays, a battery, and a power control and distribution unit (PCDU). The main classical power functionalities are based on these items: battery charge control, bus power control and distribution, and thermal system operations.

The design of solar panels is based on solar tripe-junction cells, due to their better efficiency and lower mass density. The design of the battery is based on the technology of Li-Ion cells, which are adequate for small satellites where the mass and size are critical parameters, and also offer better energy density.

The final concept for the topology of the solar panels and the battery was analysed during the study focusing on these mission points: the selected strategy during the first initial orbits; the LEOP eclipse times; the estimation of the power and temperature evolution with respect to the Sun distance during the cruise phases; the strategy during cruise phases for managing the orientation of solar panels with respect to the EP consumption needs; and the selected final power budgets of Proba-IP subsystems and equipments.

The power architecture solution will support two buses for efficiency in the treatment of the energy generated by the solar panels: a power bus devoted to the EPS, regulated at 50 V_{DC}, meanwhile the rest of spacecraft equipments are working with a non-regulated bus of 28 V_{DC}. Figure 5-3 provides a scheme of the SA topology for the two power buses and the SA dimensions.

5.6. Thermal Control

The main thermal requirements for the system are based on rejecting to deep space the heating power dissipated by the units through the external satellite surfaces, used as radiators. The drivers for the thermal design emerge from the need to provide thermal supports to the following subsystem and equipments.

a) The identified subsystem with high critical thermal needs are the EPS, when its equipments are off and working, the TTC–Communication subsystem, when there is data transmission from satellite to Earth, and the S/C power unit (PCDU).

b) Other equipments with important thermal needs are the on-board computer (OBC) unit and the reaction wheels (including their electronic units), when they are working in a steady state and with maximum acceleration/deceleration profiles.

c) Also a dedicated thermal support is required for the LPM upper stage during the initial phase when the propulsion motor is in stand-by and working during the burn manoeuvres. After the Earth escape the propulsion motor is not operative and it is separated from the satellite, and therefore the thermal support is not necessary in the following mission phases.

A set of analyses were performed to estimate the size of the external surfaces dedicated to thermal radiators, concluding that the size of the spacecraft must allow the accommodation of these radiators on the panel without solar illumination, which is parallel to the ecliptic plane of the trajectory.

Following a classic thermal concept (based on a software control-loop with heaters, radiators, MLI devices and thermistors) tailored to the Proba-IP spacecraft, the equipment units have to be mounted on the external panels with a good thermal conductance, with dedicated areas as thermal radiators, and with a consumption range of the heaters between 70 and 100 W. This power does not need to be added to the peak power of each phase but it must be added to phases with low power dissipation.

Other thermal concepts based on the design with loop heat pipes were also analysed, and were discarded because the benefits of thermal design for the proposed Proba-IP concept are poor versus the complexity and integration of these types of thermal devices.

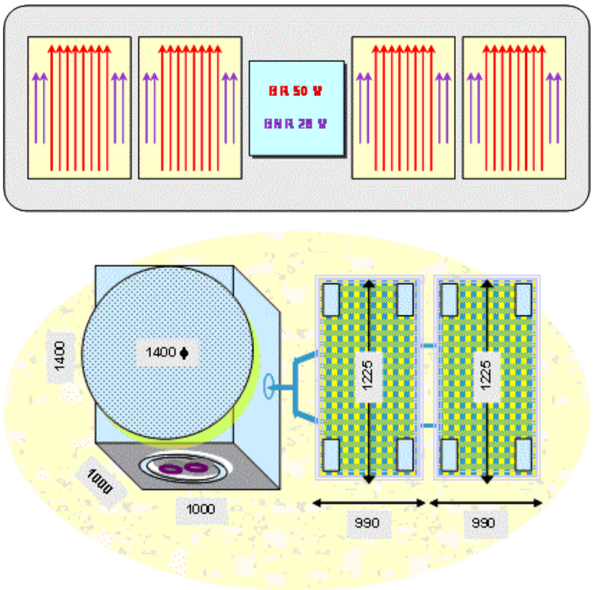

Figure 5-3: Solar array topology and dimensions

5.7. Electrical Propulsion

The EPS topology is based on the use of the HEMP-T 3050 thruster [6], due to its medium resource consumption and its low system complexity.

This topology is also considered adequate to the Proba-IP mission including the reduction of data interfaces, better risk management based on single EPS FDIR procedures, and its performances (adequate specific impulse and thrust range).

The considered EPS topology is based on two electrical plasma engines, one nominal and the other redundant. In this way it is possible to comply with the required thrusting time computed for the mission and have redundancy in case of failure of one unit.

Based on this concept, the EPS comprises two redundant electrical engines, two flow control units (FCU, one for each engine), one power supply and control unit (PSCU, including its data-interface with the satellite data bus), and the xenon tank with its appropriate pipeline and the pressure regulator. The two electrical engines are mounted with a shared pointing mechanism as part of the overall thrust functionality.

Regarding the power supply, the EPS makes use of the regulated 50 V_{DC} bus instead of the unregulated 28 V_{DC} bus for efficiency reasons.

5.8. TTC and Communications

The TTC–Communication subsystem assures in a reliable way the communication link between the spacecraft and the Ground Segment, based on an architecture using only X-band for data transmissions and for the radio science experiment during the final RTE phase of the mission.

The link communication architecture has been sized based on ground facilities with antennas of 15-meter diameter.

The proposed communication TTC implements two different types of antennas: one directional high-gain antenna (HGA for transmission during the cruise and asteroid mission phases of spacecraft housekeeping, main EPS data, and technology payload data); and two hemispherical low-gain antennas (LGA for the LEOP and commissioning phases, and also for emergency support during the rest of operational mission phases).

The layout for the TTC is based on these concepts: minimize the ohmic losses between the amplifier output and the antennas port; minimize the ohmic losses between the antennas port and the receiver input; avoid the use of command-controlled components as much as possible; and use passive components. Different block diagrams and appropriate link budgets have been analysed as a function of the proposed transmission slots and expected size of the generated mission data. The most interesting communication block is composed of these concepts.

a) There are two X-transponders with power amplifiers (working in cold redundancy in transmission, and hot redundancy in reception), where different existing transponders with variable data-bit rates have been analysed for using in deep-space missions.

b) A radio-frequency distribution unit (RFDU with the possibility of choosing either a RF switch, or a hybrid coupler as a function of the final reliability figure), and the appropriate RF harness.

5.9. Attitude control

The AOCS architecture is based on the following concepts.

a) The satellite is three-axis stabilized.

b) The reaction wheels are mounted on a pyramidal configuration with a topology based on four nominal reaction wheels working together (assuming that in case of failure of one reaction wheel, the performances during the operation manoeuvres will be slower and the pointing maintenance and stability figures will be different).

c) Use of an RCS connected to the xenon main propulsion system. The RCS thrusters will be used for small delta-V implementation, wheels desaturation and correction manoeuvres.

d) The star trackers are based on a central processor with four optical heads [9], which are mounted to avoid blinding by the Sun in all mission phases and by the Earth after launch and escape. The sun sensors are used for attitude acquisition (in the initial phases), for sun pointing when the solar mechanisms are working, and for any other safe mode application.

The navigation sensors are necessary to satisfy the GNC requirements of the Proba-IP mission related to autonomous guidance, autonomous navigation, target detection and approach operations. Different schemes, type of units and topologies have been analysed, so as to be compliant with different mission scenarios, and based on different options using WAC and NAC cameras mounted on the upper platform.

Finally the GNC block is established with these concepts.

a) A distributed star tracker configuration based on the micro advanced stellar compass [9], working either as classic star identification in some cruise phases of the trajectory or as innovative camera during the phases of NEO approach.

b) Two laser range finder (LRF) units, to be used in the asteroid phase.

5.10. Baseline of the spacecraft

Several feasible options are possible depending on the final degree of complexity, cost and implementation for the mission. The most attractive option is based on a baseline with final mass and power figures with a wet mass of 385 kg and a maximum total power production of less than 1,200 W. The detailed figures are the following:

a) Dry mass of the spacecraft is 300 kg (including margins) and the propellant mass is 85 kg.

b) The power dedicated to the EPS is 615 W @ 1 AU, and the power of the rest of spacecraft subsystems is 210 W @ 1 AU (assuming the overall electrical efficiencies and considering a maximum solar incident power equal to 2,000 W/m2).

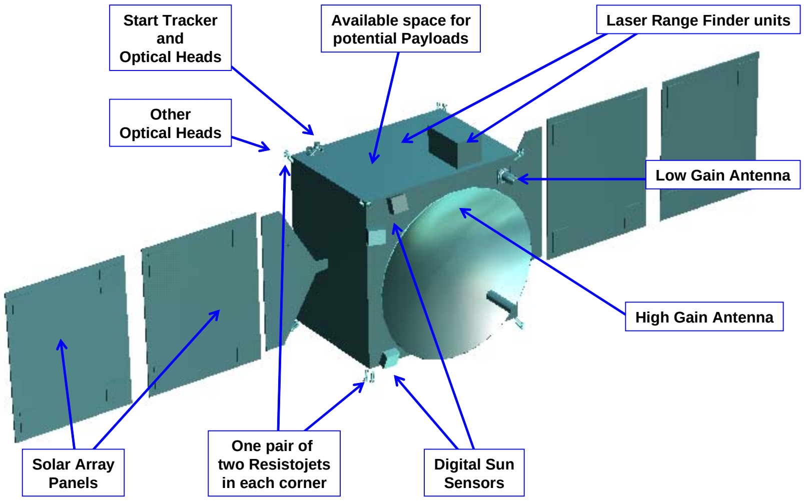

Figure 5-4: Layout for proposed external accommodation

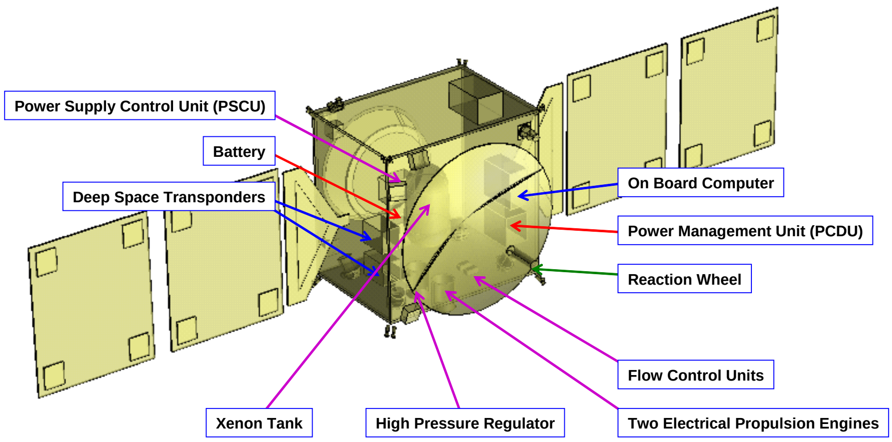

Figure 5-5: Layout for proposed internal accommodation

6. PROBA-IP OPERATIONS

6.1. Mission Timeline

The following mission timeline is proposed for the mission:

1. **LEOP and Commissioning**: which is currently assumed as the phase between launch and escape from Earth. In this phase the apogee raising sequence with the LPM is implemented with nominally 6 burns centred at perigee. It includes all the commissioning activities except the ones applicable to the technology experiments. The OD activities will be carried out on ground with the support of the ESA ESTRACK 15 m antennas and associated facilities. Manoeuvres will be computed on ground and radioed to the S/C for implementation at the selected perigee passes. Expected duration of this phase is about two weeks.
2. **Cruise Phase**: this phase corresponds to the interplanetary low-thrust transfer to the vicinity of the target asteroid. It will end with the start of the relative navigation to arrive to the target body. Various situations can be distinguished during this phase: during the first 60-day coast arc the experiments on autonomy will be activated and validated one by one: FDIR, OBRM, AGNC and SEPM in this proposed order. Thus, the autonomous GNC experiments will be activated and validated and left aside while others are also processed. Thus, during this sub-phase the ground intervention will be required for all processes and in particular for OD activities. No manoeuvre is expected within this phase, as even the launcher correction is expected to be performed by re-optimising the thrust steering law to be implemented in the next sub-phases.

   During the rest of the sub-phases within the cruise the autonomous GNC process will be active. Thus, once every week the EP engine will be stopped (if while thrusting) and the autonomous vision based interplanetary navigation mode will be activated to determine the S/C state in space. Once performed so, the G&C algorithms will recompute the thrust steering law to get to the target in due time. Thus, all the GNC processes will be carried onboard. Ground support will be required for TM/TC purposes and for monitoring of the SEP and GNC functioning.
3. **Far Approach Phase**: this phase starts with the taking of the first navigation images of the target asteroid and ends when the target asteroid occupies more than 1-pixel in the camera frame. The same autonomy concept as the one explained for the thrusting phases during cruise applies to this phase with the exception that the navigation is performed relative to the target body and the G&C algorithms are relative and are activated less frequently than in the previous. This phase lasts approximately 3.5 months.
4. **Close Approach Phase**: this phase starts with the end of the previous phase and ends with the insertion in hovering orbit. In what regards autonomous GNC, this phase is expected to be implemented in the same manner as for the Far Approach. However, monitoring from Earth will be set at a daily frequency to enable quick intervention in case of contingencies not tractable onboard. This phase lasts approximately 1 week.
5. **Hovering Phase**: this phase is meant to allow a steady final approach to the asteroid along a path of maximum illumination prior to start the in-orbit operations. It will start when the rangefinder measurements to the asteroid are collected (typically 50 km from the surface). Differently than for the Close Approach the navigation will be then supported by rangefinder measurements and the G&C will allow having a trajectory along the Sun-asteroid line (maximum illumination conditions). The maneuvers will be implemented by using the RCS. Ground intervention is proposed in a daily

basis for monitoring purposes and contingency operations, if needed. This phase is foreseen to last 1 week.

6. **GNC In-orbit Demonstration Phase**: this phase is actually meant to allow performing the experimentation of in-orbit autonomous GNC experiments. As a derivation of this phase a detailed study of the asteroid can be obtained, so as to obtain accurate information on its shape, rotation state, gravitational field, surface properties, etc. As for the previous, navigation will be performed autonomously with both the support of the vision based system and the rangefinder. Maneuvers will be performed autonomously with the use of the RCS to keep the S/C within the established proposed control spheres and also avoid eclipses. Ground intervention is proposed on a daily basis for monitoring purposes and contingency operations, if needed. This phase is foreseen to last for 1.5 months.
7. **Transfer to RTE Orbit Phase**: no autonomous G&C modes are currently foreseen for this phase, but autonomous navigation would be still possible. It is foreseen to have the help of ground to plan, compute and telecommand the RCS maneuvers that will bring the S/C from the last orbital position to the RTE orbit behind the asteroid. Ground intervention is proposed on a daily basis for these purposes. This transfer shall last a maximum of 5 days.
8. **RTE Phase**: this phase is meant to perform the Radio Tracking Experiment. Same situation occurs in this phase as for the previous in the GNC field. However, it will be probably easier to define autonomous G&C algorithms than for the transfer phase. This is due to the stability of the orbit and the repeatability of patterns each orbit. Ground intervention is proposed on a daily basis for the RTE purposes and during the foreseen 3 months.

Table 6-1 provides a summary of the phases with the associated starting dates and phase durations.

Table 6-1: Mission timeline to Apophis with launch in 2015 and all defined phases

Name of Phase	Start date	Duration
LEOP Phase	11/01/2015	2 weeks
Coast Phase #1	25/01/2015	2 months
Thrust Phase #1	26/03/2015	16 months
Coast Phase #2	26/07/2016	1 month
Thrust Phase #2	27/08/2016	8 months
Far Approach Phase	27/04/2017	3.5 months
Close Approach Phase	09/08/2017	1 week
Hovering Phase	16/08/2017	1 week
In-Orbit Phase	23/08/2017	1.5 months
Transfer to RTE Orbit	07/10/2017	5 days
RTE Phase	12/10/2017	3 months
End of operations	10/01/2018	-

7. PROGRAMMATIC PROPOSAL

The inter-planetary mission for Proba-IP is based in this phase high-level scheme, where two programs are running in parallel.

- Spacecraft development:
  - Phase A is scheduled in 9 months.
  - Phase B has a duration of 1 year.
  - Phase C/D is scheduled in 3 years, before the launch operations.
  - Phase E1 includes the support in the launch vector facilities, and also the support to LEOP and IOC operations in the Customer facilities.
- A parallel phase for the development of Technology Payload is need with the appropriate control points for the technology reviews and design updates, and also for the reception of qualification and flight models of TP hardware and/or software equipments. The identified payloads to be developed and monitored are:
  - Technology payload.
  - Guidance, navigation and control management (GNCM).
  - On-board resources autonomous management (OBRM). FDIR management (FDIRM). Electrical propulsion s/s management (SEPM)
  - Opportunity or complementary payload.
  - A reference date for the second quarter of 2010 is proposed only for clarification purposes, and without any contractual intentions.

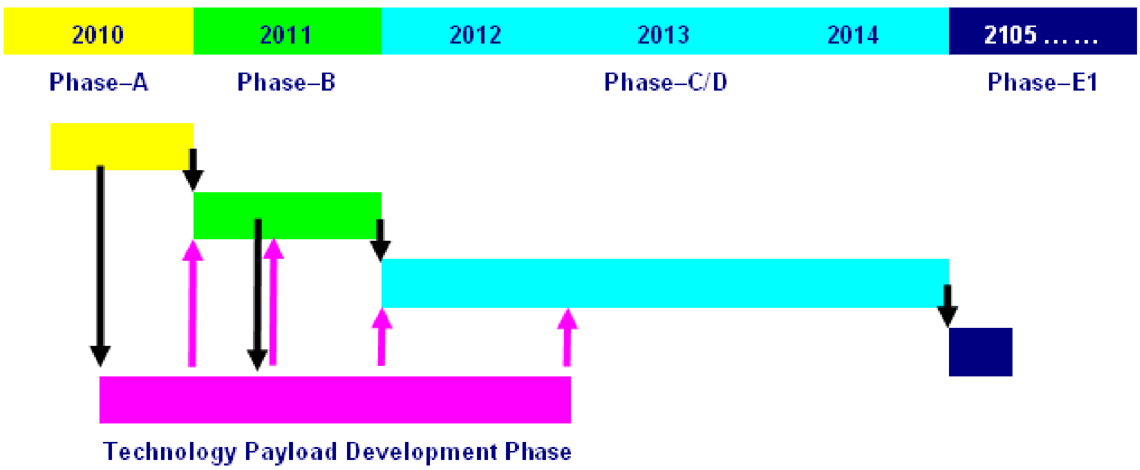

Figure 7-1: Scheme for the Proba-IP phases

8. CONCLUSIONS

Following conclusions can be derived from this study:

- Missions to Apophis in 2015 and 1989 UQ in 2017, launched by different types of European launchers, supported by different upper stages and propelled by different low-thrust propulsion solutions have been analysed in detail within this study.
- A solution based on the use of a VEGA launcher plus a recurrent use of an LPM is feasible with HEMP 3050 engine and compatible with the presented platform design.
- After consolidation of the EP engine and S/C power subsystem models it was possible to define a baseline scenario targeted to Apophis launching in January 2015 and with escape velocity of 1.892 km/s and a maximum wet mass of 434.7 kg. Arrival would occur in July 2017.
- Regarding the back-up scenario, it was possible to define a case targeted to 1989 UQ launching in October 2017 and with escape velocity of 1.839 km/s and a maximum wet mass of 438.0 kg. Arrival would occur in April 2010.
- Moreover, it has been shown that a large flexibility exists in the selection of target asteroids for other opportunities.
- A platform design associated to the proposed missions has been achieved with a S/C dry mass of 300 kg and 85 kg of propellant.
- A Ground Segment and operational profile have been defined for the mission compatible with ESA rules.
- Relevant technology developments for the mission have been identified in the fields of AGNC, OBRM, SEPM and FDIR.
- A programmatic plan associated to an earliest launch date in 2015 has been established both for the platform and the required technologies
- Platform development costs have been evaluated
- Technology development activities have been identified for each of the main mentioned technology areas
- Scientific payloads of opportunity have been listed in case the system resources allow for their inclusion in the mission
- A number of areas where particular attention is required in next mission phases have been identified and justified

END OF THE DOCUMENT