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CHANGE RECORD

ISS	REV	Date	Affected Pages	Description
A	0	03/04/2019	All	Initial draft for PM2
A	1	22/05/2019	All	Post PM2 update
B	0	03/06/2019	All	Pre-SRR delivery

Text amendments with respect to the previous issue are identified by a vertical bar in the outer margin. Minor text amendments like underlining, punctuation, spelling, page numbers and deletions are not necessarily marked however.

List of Figures

List of Tables

Abbreviations List

CI	Configuration Item
AD	Applicable Document
CDMS	Command & Data Management System
DART	Double Asteroid Redirect Test
DCP	Detailed Characterization Phase
DIP	DART Impact Phase
DOR	(Delta) Differential One-way Ranging
DSM	Deep Space Manoeuvre
DSN	Deep Space Network
ECP	Early Characterization Phase
FDS	Flight Dynamics Simulations
FOP	Flight Operations
LEOP	Launch and Early Operation Phase
MOC	Mission Operation Centre
MOCD	Mission Operations Concept Document
OBC	On-Board Computer
OBCP	Ob-Board Control Procedures
PDP	Payload Deployment Phase
RARR	Range and Range Rate
RD	Reference Document

Table of contents

1. Introduction

The Mission Operations Concept Document (MOCD) is prepared within the phase B1 study of the HERA mission led by OHB System AG. It describes the operations concept for the various phases of the mission covering both the space segment and the ground segment.

The document details the operational activities of the platform, payload and ground segment in order to drive the generation of the use cases (CDMS), the requirements for the ground segment (Mission Operation Centre, MOC), and also the generation of the procedures presented in the flight operations plan (FOP)

The document is structured as follows:

- Chapter 3 gives an overview of the HERA mission. It describes the scope of the mission and its objectives, as well as top-level requirements and constraints that drive the definition of the operational concepts and ground segment. The chapter also describes the relationship of the HERA mission with other missions or programmes, and agencies.
- Chapter 4 defines the space segment of the HERA mission by describing its capabilities and characteristics of the space-ground network. It focusses on the system architecture for the telecommand and telemetry to describe in detail the method in which the data will be generated, stored, and downlinked to end users.
- Chapter 5 defines the ground segment of the HERA mission by first presenting the overall space-ground system architecture, followed by the functionalities and roles of individual ground segment components and operational entities involved. It also presents various operational considerations for each mission phase as key design drivers.
- Chapter 6 presents the mission operations concept for each mission phase in detail. For each phase, key operational activities are described, together with relevant subsystems that are involved.
- Chapter 7 describes various support processes, mainly pertaining to flight dynamics required during the operations.

2. Documents

2.1 APPLICABLE DOCUMENTS

AD	Doc. Number	Issue/Rev	Title/Applicability
AD01	ESA-TECSP-RS-009704	1/0	HERA Mission Requirements Document
AD02	ESA-TECSP-RS-009703	1/2	HERA System Requirements Document

2.2 REFERENCE DOCUMENTS

RD	Doc. Number	Issue/Rev	Title/Applicability
RD01	HERA-GMV-MIS-RP-0001	1/0	[E01] Mission Analysis Report

3. Mission Operations Requirements and Constraints

This chapter presents a high-level overview of the HERA mission. The overall scope and constraints are described and then the relationship with other missions is presented.

3.1 MISSION DESCRIPTION

HERA is a small ESA mission that is part of the Asteroid Impact & Deflection Assessment (AIDA) mission. AIDA is a joint mission with NASA to demonstrate a deflection of a binary asteroid, Didymos, and perform close proximity observation of the outcome, while demonstrating technologies for future missions and addressing planetary defence.

NASA's contribution to the AIDA mission is the Double Asteroid Redirection Test (DART), whose primarily objective is to crash an impactor on the smallest object of the binary asteroid system, Didymoon.

To that aim, the HERA mission's motivation is three-fold:

1. To demonstrate new technologies
2. To provide science of characterisation of the geophysical properties and dynamical state of the binary asteroid, Didymos
3. To contribute to planetary defence through reduction of asteroid impact risk

The first part of the AIDA mission will be fulfilled by the DART, whose launch is foreseen to take place in 2021, with its arrival at the Didymos asteroid system in 2022, followed by the impact on Didymoon.

HERA will then constitute the second part of the mission, starting with the launch envisioned for 2023 and consequential arrival at Didymos in 2026. Upon arrival, HERA will start its early characterisation phase of the surface of Didymos in order to assess the shape and the gravity field of asteroid.

A binary asteroid is a suitable target to demonstrate the ability for deflection, as the kinetic effects of the impact can be measured accurately through the change in relative motion between the two asteroid's components. The HERA satellite can greatly improve the outcome of the experiment by providing close proximity observations to better assess the impact, as well as to provide additional scientific return.

3.2 PROGRAMMATIC AND OPERATIONAL CONSTRAINTS

Several major drivers have been identified for the HERA project that have impact on the spacecraft design, the ground operations and the project schedule. These drivers are individually addressed in the following subsections.

3.2.1 LIMITED RESOURCES ASSOCIATED TO DEEP SPACE MISSION

During the mission, the spacecraft distance to the Earth and the Sun will vary largely. In the middle of the cruise phase, the distance will even reach 3.3 AU for the Earth and 2.3 AU for the sun. As a consequence, the solar flux will be reduced and the communication link will be stressed. Additionally, the large variations in distance will have an important impact on the spacecraft thermal behaviour.

In order to accommodate the spacecraft to these large and varying distances, while respecting the limited launch mass and limited budgets, it was decided to implement a Solar Array Drive Mechanism (SADM) and to have fixed communication antennae.

This has implications for the operations. While the SADM ensure that the spacecraft will receive sufficient incoming power during all phases of the mission, it won't be possible to guarantee the best communication link (i.e. highest data rates) all the time: when pointing the optical camera to the asteroid, there will be no flexibility to point the antenna towards the Earth. Hence, the operations will need to respect these constraints and specific observation and communication slots will need to be scheduled.

3.2.2 DEEP SPACE NETWORK

In order to support the communication link from the Earth, the ESA Deep Space Network will be required. This needs to be taken into account in the ground operations. Additionally, regular use of Delta-DOR campaigns will be required for the precise orbit determination of the satellite.

3.2.3 PROXIMITY OPERATIONS

The main driver for the proximity operations is to acquire and maintain a desired formation between the spacecraft and the binary asteroid, while minimizing the risk for collision. This implies the definition of collision-free trajectories, dedicated on-board FDIR functionality, precise navigation (based on AFC images, with on-board processing) and precise execution of propulsive manoeuvres.

It also calls for a staggered approach, where the distance to the asteroid is gradually decreased in order to better assess the gravitational properties of the system and to better predict the dynamical behaviour of the spacecraft.

Moreover, this poses specific challenges for the ground operations, as a good balance will need to be found between observations periods, data downlink periods, on-board processing (flight dynamics), and scheduling of propulsive manoeuvres for formation maintenance.

3.3 RELATIONSHIP WITH OTHER MISSIONS

The HERA mission on its own has its benefits and will allow to better understand the dynamical and geophysical properties of binary asteroid. However its full potential is reached in tandem with NASA's DART mission, which together makes up the AIDA mission. HERA spacecraft will arrive after the collision of DART on Didymoon to determine the momentum transfer given by the kinetic impactor, and also perform observations of the impact crater to assess the surface properties of the body.

3.4 EXTERNAL DEPENDENCIES OR INTERFACES WITH THIRD PARTIES

The HERA and DART mission will benefit from each other and will require exchange of information to coordinate the operations, and improve the scientific finding, such as:

- Updates on Didymos model as a result of proximity operations
- Updates on Didymos and Didymoon orbit thanks to delta-DOR measurements of HERA in the vicinity of the asteroid

The precise interfaces between ESA and NASA are yet to be defined.

4. Space Segment Characteristics

This chapter describes the capabilities and characteristics, commanding and monitoring of the space segment.

4.1 TELECOMMAND DESCRIPTION

This section details the command routing and describes the different types of telecommands. Note that the cubesats have their own on-board scheduler.

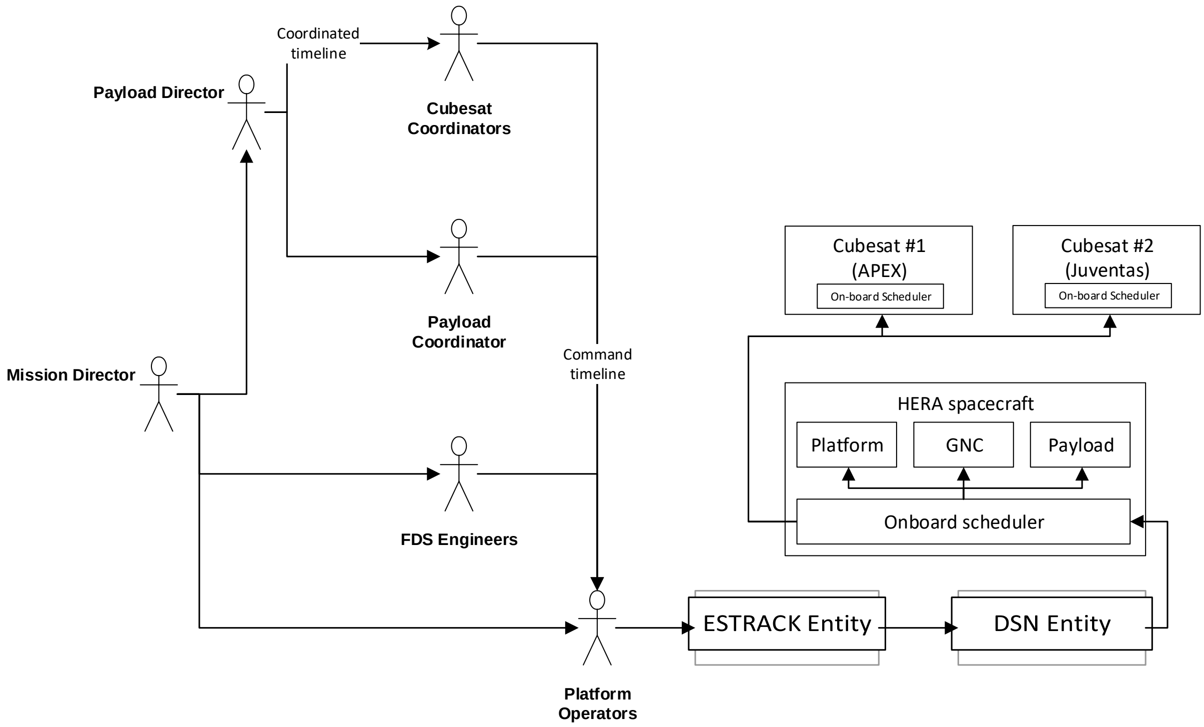

Figure 4-1: Data flow for the telecommanding

4.1.1 COMMAND ROUTING

The telecommands sent to the spacecraft can be split into the following categories based on destination:

- Platform telecommands
- GNC telecommands
- Payload telecommands

The diversity in commanding requires a specific approach to handle the different streams properly. This is typically the case for science and deep space mission (e.g. ROSETTA).

Referring to Figure 4-1, the approach for the commanding is as follows:

- The mission director gathers all inputs from the different actors (platform operators, FDS engineers and payload coordinator) and derives a coordinated timeline for the next TBD days/weeks

- The mission director actor distributes the coordinated timeline to the different actors
- Each actor provides a command timeline consistent with the coordinated timeline towards the platform operators
- The platform operators combine the different timelines, check the consistency and reformat it for (X-band) uplink. Different priorities and upload slots are assigned.
- The commands are uplinked through the ESTRACK entity and the DSN entity towards the spacecraft scheduler.
- On-board the spacecraft, different sub-schedules are used to clearly separate the different command categories. This allows independent enabling/disabling/deletion, and furthermore, PUS-C group ID functionality is envisaged to be added.
- Based on the sub-schedule, timestamp and APID, the commands are forwarded to the correct destination at the selected time.
- A read back of the on-board scheduler is performed.
- The read back is used by the platform operators to report on the upload success and possible rescheduling of activities in case of upload problems.

4.1.2 REAL-TIME AND TIME-TAGGED TELECOMMANDS

Real-time and time-tagged command sequences will be available for HERA operations.

The real-time commands are commands sent from ground for immediate execution on-board. This category is only used for specific activities during a satellite contact and specific contingency recovery procedures. Due to the long communication time with the spacecraft (up to TBD minutes), the result of the execution will be delayed, but still available within the contact time window with the spacecraft.

The time-tagged command sequences are for nominal tasks in the mission timeline. They include all actions which are part of the platform or payload planning. For these sequences, the ground tools will derive a conflict-free timeline and generate the corresponding command sequences. Each command will have a specific time-tag for execution on-board at the required time. On-board, the commands are inserted in the on-board scheduler. Several sub-schedules are available to ensure that activities can be separated and independently enabled/disabled (from ground or by FDIR on-board). The execution of these specific command sequences is not different from a LEO mission, where the same concept is used. Through the use of high-level telecommands, the required amount of commands is limited.

Alternatively, OBCPs can be used that are generated on ground instead of loading the individual commands into the scheduler. The OBCP is loaded on-board and a single command from ground can start it at specific times/intervals (or event based) to execute the entire activity. This concept is used for several deep space missions like ROSETTA, Venus and Mars Express.

Upload of commanding shall be performed by using command sequences packaged in a file. These command files are uploaded using a file transfer protocol (CFDP) to ensure correct reception and efficient usage of the bandwidth in nominal and non-nominal conditions, such as resending of part of the data. The uplinking of files is also envisaged for specific memory operations, upload of OBCPs and uplink of specific tables.

For the payload commanding concept, the following is baseline:

- Upload of timeline to OBC (72 hours is considered a sensible baseline)
- Commanding of payloads from OBC (separate timelines for each payload/activity). Real-time commanding of the payload from ground would be technically feasible, but not the nominal way of commanding.
- In case of anomaly with a payload, the payload is de-activated and the timeline for that particular payload is halted. The same can be done in case of anomaly with geometry or the execution of a safe mode.

In case of safe mode (SM), the nominal MTL is automatically disabled and a specific safe mode MTL is loaded. This implementation is "a la carte" for HERA. It requires detailed analysis together with the payload providers to determine the state of all instruments during a SM. In general, the instruments are autonomously switched OFF (or put in a very low power mode) and the nominal MTL is disabled.

4.1.3 COMMAND VERIFICATION

For each command, the standard PUS acknowledges are generated on-board. This includes successful or failed acceptance and successful or failed completion, execution progress reports can be requested for specific telecommand which require long execution times. Due to the long round-trip time, any acknowledge will arrive at the ground segment minutes or even hours (during next contact period) after the generation of the acknowledge packet.

For commands loaded into the on-board scheduler, a verification is required that the command was correctly loaded. This is also based on command acknowledges for the loading itself (wrapper command). In addition, after the loading of a command sequence into the on-board scheduler, a read-back could be performed to validate the content on ground. The read-back can be based on the number of commands, or a partial (TC header) or full read-back of the scheduler, depending on the available bandwidth. The baseline is to read-back the TC headers of the scheduler.

4.1.4 VIRTUAL CHANNELS AND MAP IDS

The spacecraft accepts telecommands on different virtual channels and using different MAP IDs. MAP IDs are used to assign addresses to direct the command to different hardware/software.

The utilisation is currently based on the PROBA approach, which can be modified as the system design becomes more detailed. Summary of the virtual channels and MAP IDs is presented in Table 4-1.

Table 4-1: Virtual channels and MAPIDs

VC	MAP	Utilisation
1	0	High Priority SPDU commands via software PTD
1	1	Low priority commands via software PTD to CDMS (PUS packets)
1	2	Low priority commands via software PTD to CDMS (PUS packets)

VC	MAP	Utilisation
3	0	High Priority CPDU commands to emergency (HW) EPTD of primary OBC
4	0	High Priority CPDU commands to emergency (HW) EPTD of redundant OBC

4.1.5 ON-BOARD CONTROL PROCEDURE

The use of the on-board scheduler (MTL) allows for the execution of static sequences, which is success-oriented as there is no immediate action in case a command fails the execution. On-board autonomy allows the spacecraft to react to unexpected inputs or events, while automation processes allow the scheduling and execution of routine operational activities. An On-Board Control Procedure (OBCP) is a means to combine these two functionalities, and thereby to improve spacecraft operations. Moreover, OBCPs can be combined with on-board surveillances to enhance the autonomous reaction to a failure.

As a baseline, the OBCPs is foreseen for two different applications: execution of routine command sequences, and the execution of specific on-board FDIR.

For the execution of *routine command sequences*, the OBCP is seen as a flight control procedure executed on-board. It can execute a specific activity by sending specific telecommands to different on-board applications in a static or dynamic timeframe. The OBCP can be made such that the command execution is verified, specific telemetry is checked or until wait until a specific parameter is within certain limits. Extensive experience has been build up with using OBCPs on the PROBA satellites, both in AIV and operations. As an example, a very simple OBCP is used on-board to perform a bake-out of a payload detector, commanding the heater based on the read back of the temperature to control the bake-out temperature accurately within the set boundaries. The use of OBCPs for deep space mission is common. The tools to implement, test and validate the OBCPs shall be provided to the MOC.

The different interfaces of the OBCP can be found in Figure 4-2. The OBCP can have the exact same functionality as the ground flight control procedure, but can react immediately instead of being bound to the round-trip delay. The OBCP can be started from ground in real-time or from the on-board scheduler.

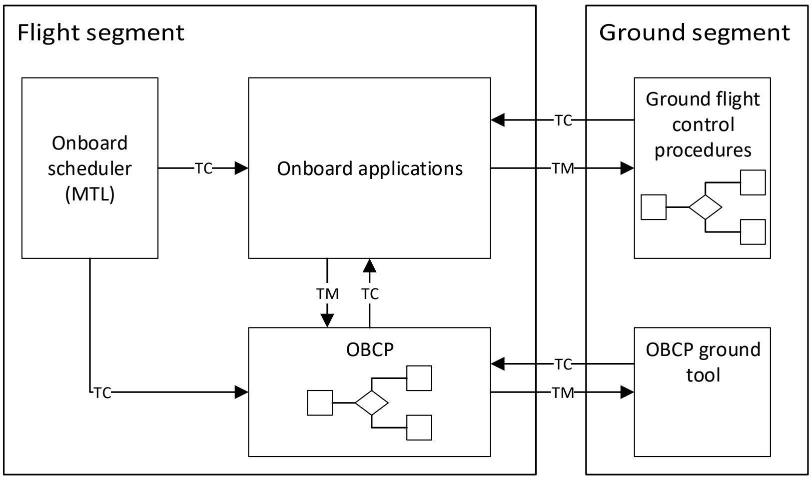

Figure 4-2: OBCP interaction scheme with on-board and on-ground components

A second application of OBCPs is *FDIR*. In reaction to an event, a specific TC can be sent (typical PUS event -> event-action service). The event can however also trigger the execution of an OBCP. This opens a huge range of possibilities to recover from the anomaly using the on-board available information. This avoids the long ground loop, which depends on available ground contacts, operator presence, etc. The use of OBCPs for FDIR is regularly used (e.g. Herschel/Planck) and enables relatively complex actions in response to specific FDIR.

A typical example is an FDIR event sent by the payload which should result in multiple actions: request diagnostic data from the payload, graceful shutdown of the payload, disabling of the on-board scheduler related to this payload, download of the diagnostic information. All this can easily be coded in an OBCP, which can be validated during AIV and tuned using a satellite simulator.

The proposed OBCP engine is the Euclid µPython solution developed by Spacebel, who is the envisaged OBSW provider for HERA. The µPython solution is the current baseline, not only because it offers the most functionality on-board, but mainly because it ensures a rich preparation environment on ground, which can be run on any PC with Python. This is believed to be vital for missions that utilise OBCPs for many different functions.

4.2 TELEMETRY DESCRIPTION

This section presents the telemetry routing concept and the different types of telemetry.

4.2.1 TELEMETRY ROUTING

There are two main types of telemetry that will follow a different end-to-end routing: platform telemetry and payload telemetry.

The platform telemetry is all data generated by the platform which is not related to the payload. This is a variety of packets, for which the main blocks are: TC verifications, events, housekeeping, memory dumps/checks and read backs/reports of on-board applications and units.

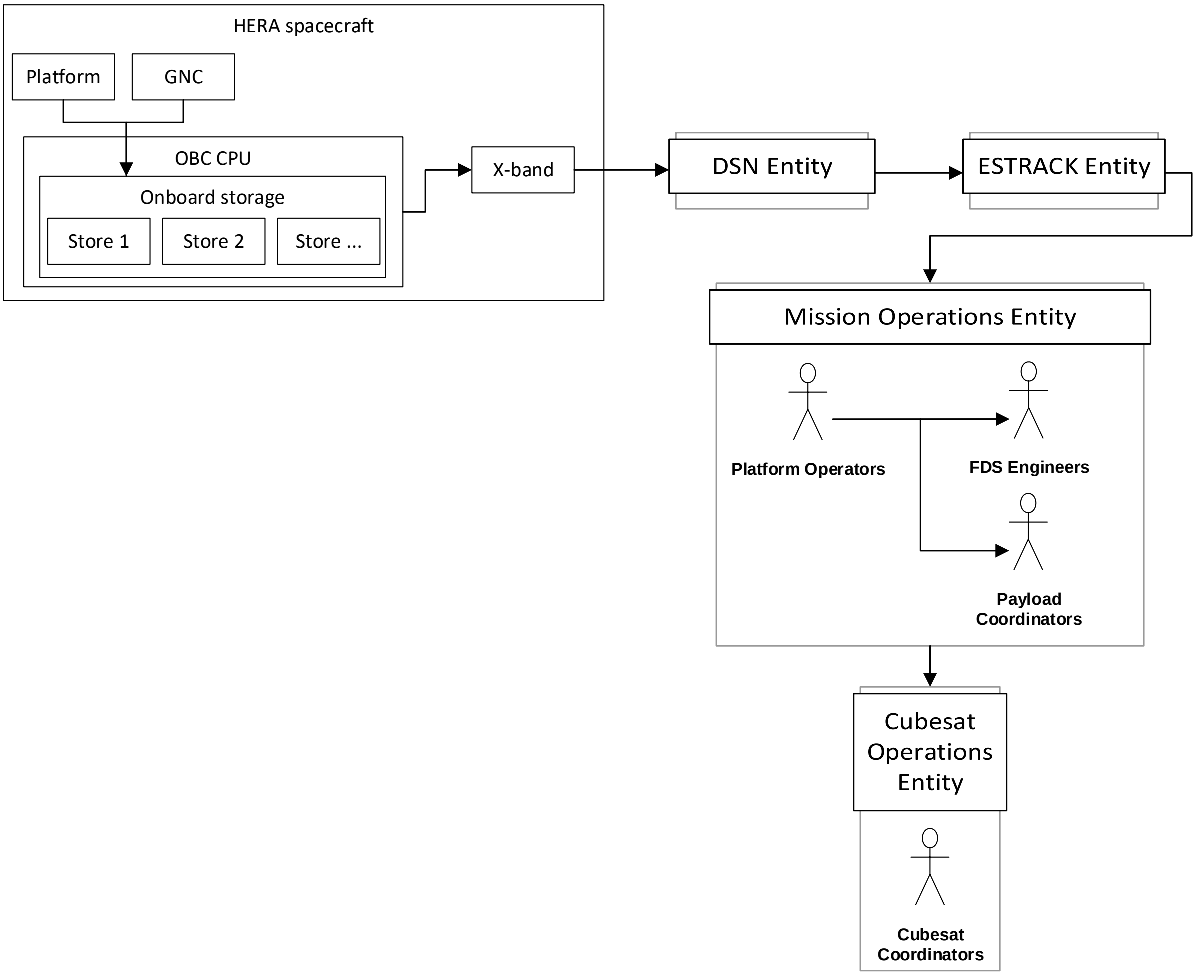

Figure 4-3: Data flow for platform telemetry

The platform housekeeping is gathered and stored in the on-board computer (OBC) prior to downlink. Different on-board stores in the non-persistent memory of the OBC are created to separate the different types of data and allow independent download. The size of the non-persistent memory (RAM) capacity of the PNEXT OBC, which is the current baseline, is 256 Mbyte. Typically, an OBSW requires 8 to 16 Mbyte, and the remaining RAM capacity can be used for on-board packet stores, which is sufficient. Nominal PUS services are used for the on-board store management.

During the download, the data is downlinked over X-band on a specific virtual channel (see Sec. 4.1.4) towards the DSN ground stations, from which it is forwarded to the platform operators. From there, data is decoded, unpacked and displayed on the mission control displays. Archiving and further data distribution towards other entities is also performed at this point. For the GNC data, the FDS engineers receive the data as soon as it is available (filtering based on APID can be used). For specific mission phases like the asteroid approach and certain proximity operations, a low data latency is required for the data towards the FDS.

Real-time telemetry is also generated and downlinked over X-band via a specific virtual channel.

The payload data is archived in the on-board mass memory integrated in the OBC, which is the current baseline configuration. The payload data is archived as standard CCSDS packets with the APID as payload identifier. This allows straightforward data filtering and distribution on ground. The payload data is downlinked in a particular virtual channel to allow straightforward filtering of the data from the other data on the X-band link. Specific virtual channel prioritisation is implemented on-board to ensure minimum bandwidth allocation for the different data streams.

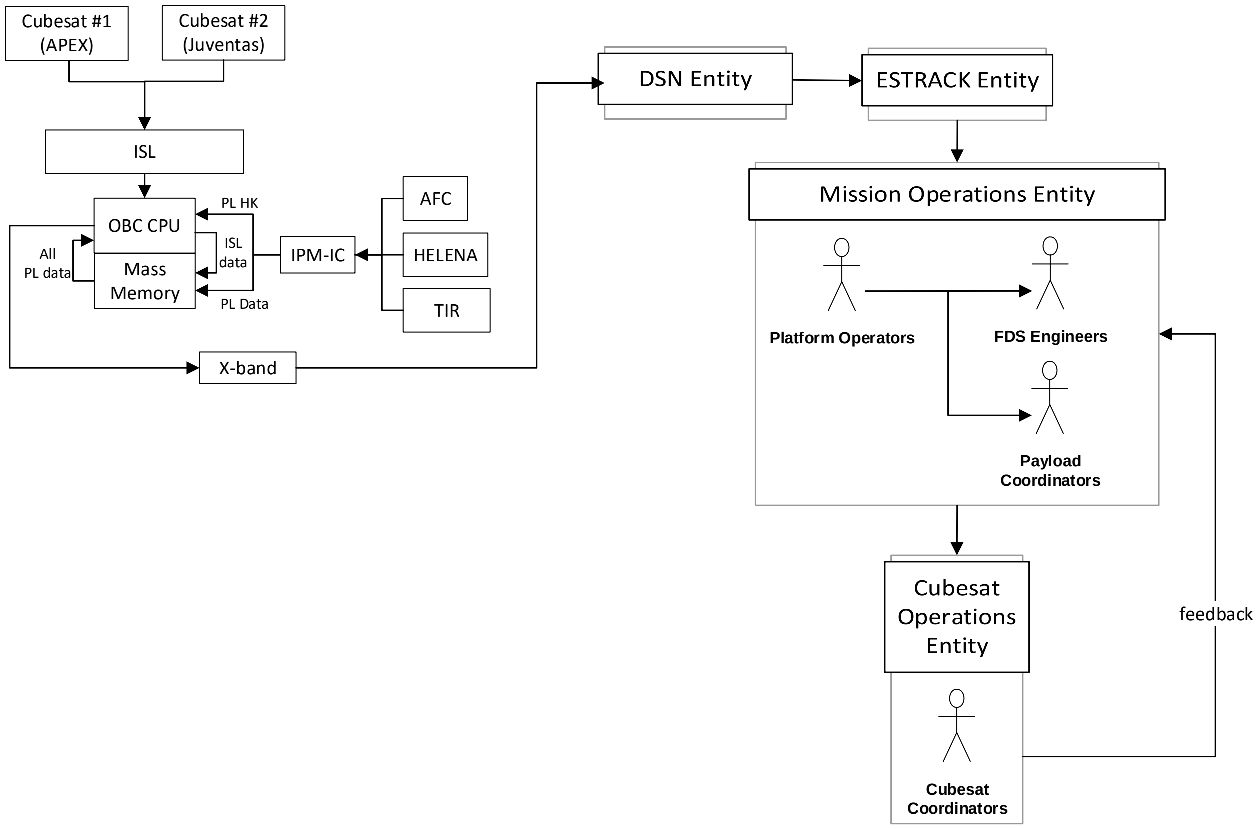

Figure 4-4: Data flow for payload telemetry

It is important to note that the current baseline of the DHS subsystem includes an Image-Processing & Instrument Control Unit (IP-ICU), where the image data from the AFC and TIR are processed on-board to assist with the autonomous GNC of the spacecraft.

It is important to note that some of the payload data, especially AFC, will provide crucial information on the asteroid(s) to detail FDS models. Therefore, some of the payload data shall be transferred to ESOC so that FDS engineers can analyse them directly, or alternatively, the payload operations entity can feedback the results of the payload data processing to the FDS engineers.

4.2.2 VIRTUAL CHANNELS

The allocation of telemetry streams to different virtual channels is presented in Table 4-2.

Table 4-2: Virtual channel allocation for telemetry

Virtual Channel	Description	Origin
VC#0	Essential telemetry	Processor module in OBC
VC#1	Real-time HK telemetry	Processor module in OBC
VC#2	Stored platform HK data	Produced by the ASW
VC#3	Payload data	Flexible implementation (can be linked to packet stores in mass memory)
VC#4	Payload data	Flexible implementation (can be linked to packet stores in mass memory)
VC#5	Payload data	Flexible implementation (can be linked to packet stores in mass memory)
VC#6	Payload data	Flexible implementation (can be linked to packet stores in mass memory)
VC#7	Idle frames	Produced by the telemetry module in the GRTM core in GR712RC ASIC

The OBC sends data on VC#0, #1, #2 and #7. The bandwidth can be allocated to these virtual channels. PNEXT OBC will be updated to include virtual channel prioritization (currently a configurable bandwidth allocation table is used).

VC#0 is so-called essential telemetry and is accessible via hardware, i.e. the telemetry data transmitted through VC#0 is mainly useful for anomaly investigation, when the on-board software telemetry is not available.

VC#1 and VC#2 are accessible via the on-board software. It will be used to differentiate between the real-time telemetry (VC#1) and stored telemetry (VC#2). On VC#2, apart from stored telemetry, also science data from the low-speed payload will be send.

VC#3 to VC#6 is used to dump data from the mass memory to the ground through the OBC. As the payload data is read by the OBC from the mass memory, before being sent on the downlink to the X-band transmitter, there is a lot of flexibility in the usage of the virtual channels. It could be envisaged to group all payload data, which is useful for the FDS, in a dedicated virtual channel to allow quick segregation of this data from the other payload data. These 4 virtual channels can be implemented with lots of flexibility, even while already in-flight.

4.3 ON-BOARD AUTONOMY

The deep space nature of the HERA mission necessitates a certain level of autonomy in order to fulfil the proximity operations. Additionally, the autonomy concepts as used in the PROBA missions may be re-used to simplify the ground operations. The HERA autonomy concept can be summarized by the following aspects:

- High level commanding (mode changes and payload/platform activities can be performed using a single telecommand)
- Autonomous FDIR approach
- Autonomous LEOP activities (solar panel deployment, priming of propulsion subsystem, survival mode acquisition)

If necessary, any autonomous activity can be deactivated in order to perform tasks from ground instead.

One important component in the HERA autonomy concept is the use of the HERA optical payloads to demonstrate autonomous visual navigation in the vicinity of the asteroid and for low altitude fly-bys. The baseline for achieving autonomous GNC is to use Asteroid Framing Camera (AFC) to evaluate the dynamic properties of the binary system and its physical properties, which will then be processed by navigation algorithm. Thermal Infrared Imager (TIRA) and Planetary Altimeter (PALT) may also employed for this as options.

The on-board autonomy is mostly encapsulated within the system mode definition and the FDIR concept. Both are discussed in the proceeding sections.

4.3.1 AUTONOMOUS FDIR

Following the FDIR design of previous missions at OHB and QS, a FDIR hierarchy consisting of five layers is proposed (see Sec. 4.3.1). In this way the failures are classified in groups that are subject to a common failure handling approach. Besides, a correlation between the levels and the failure criticality exists in general, i.e. level 4 events are typically more critical than level 0 events.

Figure 4-5: FDIR hierarchy

4.3.1.1 Level 0

A L0 failure is associated to an internal single failure in a platform or payload unit that can be automatically isolated and recovered by the unit itself without any impact on the rest of the system (H/W or S/W).

L0 failures are typically EDAC single bit errors. The EDAC device can detect and correct one bit flip in data read from RAM memory. There is no impact on the data reading operation when the corrupted data in RAM is re-written locally by the background function.

Any platform or payload unit is responsible for the detection and recovery of its L0 failures.

4.3.1.2 Level 1

A L1 failure is associated to a failure in a platform unit or payload that cannot be recovered by the unit/payload itself. Therefore the Satellite Control Software (SCSW) is in charge of the failure detection and recovery process. Since the failure does not affect other satellite subsystems, the recovery is compatible with the current AOCS and spacecraft mode.

An analogous approach applies to the payload which is composed of several payloads controlled by the SCSW, ie. a failure of a payload unit that can be recovered by the payload itself is regarded as a L1 failure as well.

The following L1 failures can be distinguished:

- Platform unit or payload failure: The failure detection is performed by the SCSW via simple health checks. When it detects an anomaly, the SCSW triggers the recovery sequence. Usually the recovery sequence is simply a power cycle and if necessary a switch-over to the redundant branch
- Payload unit failure: Analogous to a platform unit failure
- Internal communication I/F failure: The failure detection and recovery mechanism depends on the communication link

4.3.1.3 Level 2

L2 failures are associated to a unit or complete subsystem, so that the recovery is not confined to the failed unit/subsystem, but involves other subsystems as well. These failures are managed completely by the SCSW, resulting in an AOCS and/or spacecraft lower level mode change.

A L2 failure is typically related to one of the following S/C vital functions:

- Command and control function (including communications with ground),
- Attitude and orbit control functions (including propulsion),
- Power supply function,
- Thermal monitoring and control function

Several spacecraft vital functions are monitored by a two-stage process, i.e. by the SCSW using a low limit and by HW using a higher limit. Therefore an objective of the L2 FDIR is to process these anomalies by the SCSW before they result in a more severe system alarm with major impacts on mission operation.

4.3.1.4 Level 3

A L3 failure is considered as a severe internal SMU computer failure which cannot be handled autonomously by the Processor Module (PM) HW or SW, but require a higher HW instance – the Reconfiguration Module – for recovery. These failures are detected either by PM HW or SW that trigger the recovery by raising a related SMU-internal alarm. The actual start of the recovery is then initiated by the Reconfiguration Module of the SMU in response to this alarm.

A suitable recovery action is a PM reset. The recovery is completed by the SCSW which reads the spacecraft context in the Safe Guard Memory during run-up in order to establish the proper spacecraft configuration and mode.

The following are the SMU-internal alarms included in Level 3:

- Watchdog alarm: The watchdog has to be cyclically refreshed by the SCSW, otherwise the alarm is triggered. A standard configuration is e.g. a watchdog expiry duration of 1s and a SCSW refresh period of 0.5s.
- PM HW alarm: SMUs usually provide one or more HW alarms to flag severe HW errors of the PM module (e.g. processor or additional ASICs/FPGAs) that result in a loss of the processing function.
- PM undervoltage alarm

- SW alarm: The SW is able to issue an alarm in case of SW traps or a FDIR escalation due to persistent L2 failures. Common SW traps are:
  - Uncorrectable (double bit) error in memory
  - Memory access violation (e.g. due to illegal memory address or write access to a write-protected memory area)
  - Arithmetic error (e.g. division by 0, overflow, underflow)
  - Processor illegal instruction
  - Cache error

4.3.1.5 Level 4

In case of a major anomaly on the persistent memory where the SCSW images are retrieved from during boot, or repetitive attempts to reboot the nominal PM fail, a switch-over of the PM is performed by the REM. The failure detection is performed by the REM's watchdog or bios. For recovery, the OBC is reconfigured to the secondary PM lane.

4.3.1.6 Critical System Level Alarms

Certain alarms are foreseen to detect critical anomalies that endanger the spacecraft health or safety. Each failure indicates a severe anomaly in the spacecraft control function, or may be caused by two or more anomalies occurring in different units/subsystems about the same time.

The following system level alarms are foreseen for HERA:

- Attitude anomaly alarm: the attitude anomaly alarm is triggered by the sun sensors on the solar arrays if a pre-defined offset is reached (approximately 10° TBC). In this case, the SCSW changes to Safe Mode to reacquire Sun pointing of the solar arrays and subsequent Earth reacquisition in Safe Mode. Nominal PM re-configuration is performed.
  - In case entering Safe Mode does not lead to switching off the attitude anomaly alarm, survival mode is entered and PM reconfiguration is initiated.
  - The use of dedicated solar generator sun sensors is foreseen to cover incorrect SADM actuation, which cannot be detected by using body mounted CSSs only.
- Battery undervoltage alarm: This alarm is generated by the PCDU. If the battery voltage decreases below first HW threshold, transition into Safe Mode plus PM reconfiguration is foreseen. A part of the recovery is the HW Disconnect of Non-Essential Loads (HW-DNEL). If a second threshold is reached, survival mode is entered and PM is re-configured. As a last resort, PM switchover is performed in case a third threshold is reached.
- Thruster ON alarm: This alarm signals that at least one thruster is switched on for an anomaly long duration. The alarm shall trigger a direct switch of propulsion branches as well as of transient into safe mode to enable detumbling with the thrusters.

Note that in the LEOP phase usually parts of the FDIR are disabled, most specifically the system level alarms. This is assessed as uncritical since entering survival mode is foreseen after launcher separation.

Table 4-3: Summary of critical system level alarms

Alarm	Mode	Detection	Recovery Action
Attitude Anomaly	Nominal mode	Solar generator CSS SCSW	Transition to Safe Mode (L2 + L3)
	Safe mode + [TBD] mins	Solar generator CSS SCSW, REM	Survival mode transition PM recovery (L3 + L4)
Battery Undervoltage #1	Nominal mode	SCSW and HW	Transition to Safe Mode (L2 + L3) Switch to non-essential units
Battery Undervoltage #2	Survival mode	SCSW, REM	Survival mode transition (L2 + L3)
Battery Undervoltage #3	Survival mode	SCSW, REM	PM switchover (L4)
Thruster ON Alarm	Any mode	IMU, Valve BSM, SW	Switch to redundant branch, transition to Safe Mode (L2)

A summary of the FDIR levels is presented in Table 4-4.

Table 4-4: Summary of description of FDIR levels

Failure			Function Affected	Detection By	Recovery Trigger	Impact on Mission
Level	Description	Severity				
0	Internal failure	4 – negl.	Platform unit	Platform unit	Platform unit	No outage
	Internal failure	4 – negl.	Payload unit	Payload unit	Payload unit	No outage
1	Failure of platform or payload unit	4 – minor	Platform unit, payload	SCSW	SCSW	Limited outage
	Communication IF failure	4 – minor	Communication with platform or payload unit	Platform or payload	Platform or payload	Limited outage
2	Vital function anomaly	3 – major	Vital satellite function	SCSW	SCSW	SC mode change, mission interruption
3	OBC HW or SW failure	3 – major	OBC	OBC, SCSW or REM	OBC	SC mode change, mission interruption
4	Global satellite malfunction	2 – critical	OBC	REM	ERM watchdog or BIOS	Survival mode, mission interruption

4.3.2 BASELINE SYSTEM MODES

Figure 4-6 illustrates the system modes for the HERA mission. It consists of seven modes, where two fall-back modes are shown in black and two operational modes in blue, with three special manoeuvre modes in red to reflect the autonomous guidance capability. In nominal conditions, the spacecraft will be either in the operations mode or the propulsion mode.

The spacecraft will enter the fall-back modes (survival and safe modes) autonomously in case of major/critical anomalies. They guarantee stable and safe conditions for the platform and the payload and communication capability. Safe mode supports high data rate communication with earth, while survival mode is used as a last resort when inertial attitude measurements are not available to the platform.

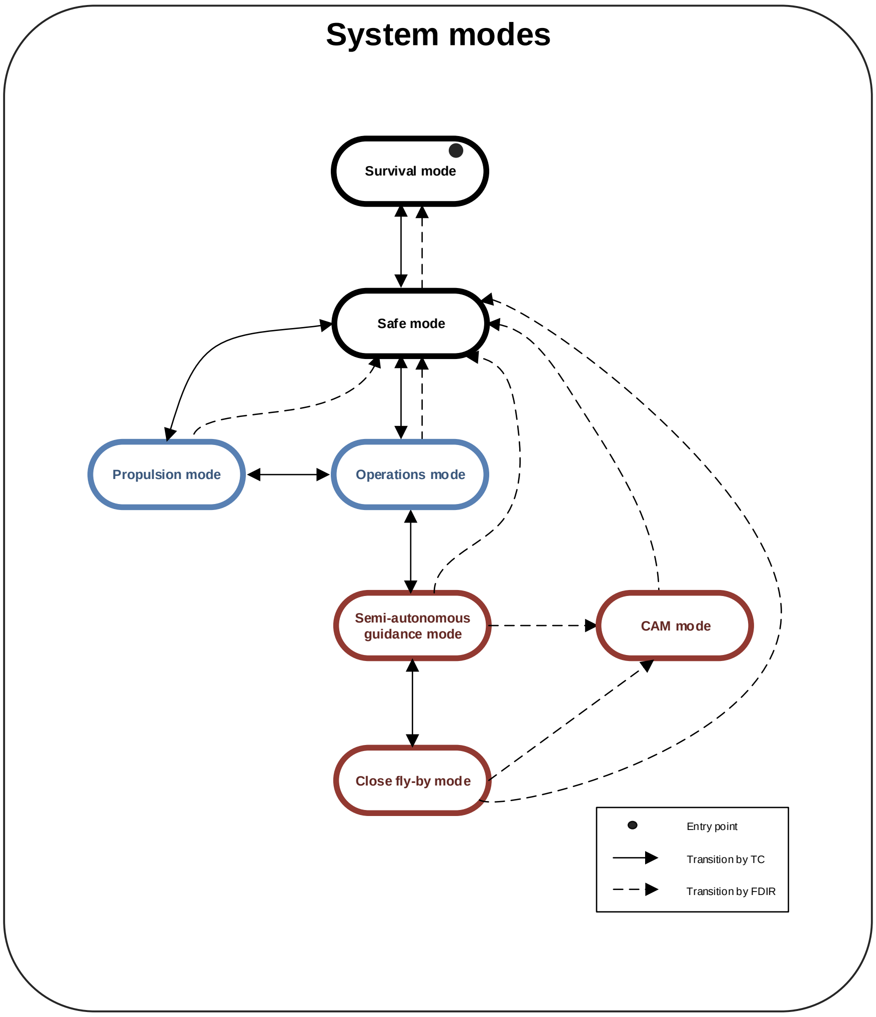

Figure 4-6: HERA system mode overview

4.3.2.1 Safe Mode

The safe mode is the main fall-back mode for the spacecraft in case of major anomalies, i.e. when high level system modes cannot be maintained. In safe mode, the safety of the spacecraft is guaranteed in terms of power availability, thermal conditions, dynamic stability and payload protection. Moreover, it ensures an uninterrupted high data rate communication link with the Earth through the HGA.

The safe mode however relies on the full on-board knowledge of the spacecraft attitude, implying that the star trackers must have a fixed navigation. In case of incapability to maintain the on-board attitude knowledge estimation, a transition to survival mode occurs.

Below are the main characteristics of the safe mode:

- Non-essential equipment and payload are deactivated and inhibited
- Payload protection measures are taken (closing of covers)
- Attitude control such that the HGA is pointed to the Earth and the SADM-axis is perpendicular to the Sun direction
- Control of the SADM angle such that the solar array is pointed directly to the sun
- Activation of the X-band transmitter and selection of the HGA, such that ground contact can be established (note: X-band receiver is *always* active)
- Removal of excess angular momentum when necessary, this includes the ability to de-tumble the spacecraft in case of high angular rate

The safe mode relies on star trackers and the IMU as sensors. Temporary outage of one or more sensors can be overcome by the on-board attitude Kalman filter, at least for a limited period of time. The sun sensors are also available in this mode, but for FDIR purposes only (i.e. the measurements are not considered in the attitude control loop).

The reaction wheels are nominally used for attitude control, while the GNC thrusters are activated for angular momentum offloading. In case of malfunction of the reaction wheel assembly, the GNC thrusters can take over the authority for attitude control. Note that a single reaction wheel anomaly will only lead to that wheel being deactivated, while the other three wheels will continue to provide 3-axis attitude control. Reconfiguration to the attitude control by the GNC thrusters is hence only required in case of a second wheel anomaly.

The deployable solar array will create an unbalance in the solar radiation pressure and will therefore result in build-up of the angular momentum of the spacecraft. In order to compensate for this effect, the spacecraft will perform a 180° rotation about the earth axis at pre-defined intervals (e.g. once every 12h).

In order to point the HGA to the Earth, the spacecraft must be aware of the Earth's direction. This information is provided by ground and stored in the context memory, such that it is still available after an OBC reboot. Likewise, the Sun's direction profile over time is provided by ground and stored in the context memory.

A configurable parameter will determine whether survival mode or safe mode must be entered upon an OBC (re)boot. At launch, this parameter will specify survival mode entry as required for the LEOP phase. After declaration of successful LEOP, ground will change the parameter such that safe mode becomes the nominal entry point after a reboot.

4.3.2.2 Survival Mode

The objectives of the survival mode are essentially the same as for safe mode, but it needs to cope with the additional constraint that no 3-axis attitude estimation is available on-board. A consequence thereof is that an uninterrupted communication link with the Earth is no longer possible as the spacecraft does not know where the Earth is.

In order to overcome this incapacity, the spacecraft will be controlled with respect to the Sun such that the LGA is pointed towards the Sun and the Earth, as it is omnidirectional, to achieve constant communication to Earth. There could be a slow controlled rotation about the sun axis, which will yield a small strobing effect for stabilization, but there will be no strobing when pointing the LGA towards Earth.

Below are the main characteristics of the survival mode:

- Non-essential equipment and payload are deactivated and inhibited
- Payload protection measures are taken (closing of covers)
- Attitude control such that the LGA line-of-sight describes a cone about the sun with angle corresponding to the sun-spacecraft-earth angle
- Control of the SADM angle such that the incoming power is maximized
- Removal of excess angular momentum when necessary, this includes the ability to de-tumble the spacecraft in case of high angular rate

The survival mode is based on sun sensors and the IMU. They provide on-board knowledge of the sun direction (2-axis attitude, not 3-axis) and of the spacecraft angular velocity.

The GNC thrusters are the primary actuators for attitude control (and for angular momentum offloading).

In order to accomplish the survival mode attitude control strategy, it is necessary to have on-board knowledge of the sun-spacecraft-earth angle. This information will be provided by ground and will be stored in the context memory, such that it will still be available after a reboot or OBC lane switch-over.

Besides being the fall-back mode of last resort, survival mode will also be the nominal system mode for the LEOP phase, and will be acquired directly after separation from the launcher. The main reason for that, is that there is no guarantee that the star trackers will be able to obtain a fixed navigation (e.g. because of sun/earth blinding, high angular velocity, unfavourable temperatures or any other malfunction). Sun sensors are more robust and are therefore baselined as main sensor for the LEOP.

The configuration of survival mode for LEOP will be somewhat different than later in the mission:

- The LGA will be selected for X-band up- and downlink, as they guarantee omni-directional coverage and support sufficient data rate at close distance from earth (i.e. during LEOP and begin of commissioning phase)

- The on-board knowledge of the sun-spacecraft-earth angle will be set to zero, such that the LGA will be pointing to the sun and the SADM will be remain its default configuration (zero angle)

This configuration will be set pre-launch and will be updated from ground after successful execution of the LEOP.

During LEOP, the deployment sequence will be executed, i.e. the deployment of the SADM and the priming of the propulsion subsystem. These activities will be performed by means of start-up telecommands, which will be deleted after confirmation of successful first execution. They are hence not part of the survival mode itself.

As the communication link may be cumbersome to establish in survival mode, it is sensible to have an autonomous transition to safe mode, when possible. Therefore, the star tracker will regularly be checked and upon successful navigation fix, safe mode will be autonomously recovered by the spacecraft. Note that this autonomous transition will be disabled by default for LEOP, and will be enabled by ground upon successful LEOP.

4.3.2.3 Operations Mode

Most of the time, the spacecraft will be in operations mode. This is a versatile mode that can support a myriad of activities:

- Scheduled downlink to earth
- Asteroid observation according to user-defined pointing profiles
- Fine propulsion manoeuvres for formation acquisition and maintenance with the binary asteroid
- Commissioning activities such as equipment health checks and system/ subsystem calibration tests

The operators will have almost full control on the spacecraft activities, within the operational and FDIR constraints.

Nominally, the spacecraft will be 3-axis stabilized and will accept ground-provided inertial attitude profiles for its pointing. Ground will however also have the option to use the sun or earth pointing functions available in safe and survival mode.

Below are the main characteristics of the operations mode:

- Payload and equipment are all available
- 3-axis attitude control based on user-provided inertial attitude profiles
- Autonomous control of the SADM angle such that the incoming power is maximized (this can be overridden by ground such that a user-defined profile is imposed, e.g. in order to avoid attitude perturbations during critical phases)
- Antenna selection and activation/scheduling of X-band transmitter under responsibility of ground
- Autonomous angular momentum offloading (this function can be enabled and disabled by ground, such that offloading occurs at desired periods of time)
- GNC thrusters can be activated in open-loop in order to perform small delta-V manoeuvres

The default sensors in operations mode are the star tracker and the IMU. The sun sensors are used for FDIR purposes. The reaction wheels are used for attitude control, with the GNC thrusters for angular momentum management. Here as well, ground has the option to revert to attitude control with GNC thrusters only (although not recommended for propellant consumption reasons).

The payload will be operated by ground command, no specific autonomy related to payload operations is foreseen (except for FDIR).

4.3.2.4 Propulsion Mode

The execution of large delta-V manoeuvres, such as the braking manoeuvres to rendezvous with the asteroid, requires a dedicated system mode. This is called propulsion mode.

Below are the main characteristics of the operations mode:

- Payload is deactivated and inhibited
- Payload protection measures are taken (closing of covers)
- Transfer thrusters can be commanded to generate a specified delta-V
- Attitude control is based on the GNC thrusters as the reaction wheels do not have sufficient capacity to absorb the generated propulsion perturbations
- Attitude guidance is based on ground-provided inertial attitude profiles
- Control of the SADM angle such that the incoming power is maximized
- X-band transmitter can be activated (over either LGA or HGA), depending on the needs and attitude constraints

The star trackers and IMU will be available for attitude estimation, and the sun sensors for FDIR. As mentioned in the list above, the reaction wheels will not be used for attitude control (wheel speed will be maintained) but the GNC thrusters will be employed instead.

4.3.2.5 Semi-Autonomous Guidance Mode

If an image processing module would be included in the spacecraft design, the optical images taken by the AFC payload could be processed to provide an on-board estimation of the relative position and attitude of the spacecraft with respect to the asteroid. A centre of brightness algorithm or feature tracking algorithm can be used, depending on their performance as function of the distance to the asteroid and of the illumination conditions.

The semi-autonomous guidance mode allows to keep the asteroid within the field-of-view of the camera, even in case of large ground-based prediction errors in the relative position. The idea is to provide a desired inertial attitude profile from ground (as for the operations mode), which is called the primary attitude. The secondary attitude is the one derived from the on-board image processing and relative state estimator. As long as the difference between primary and secondary attitudes remains with a certain limit (e.g. 10°), the secondary attitude will be used for attitude control. In other words: the predicted asteroid pointing will be overruled by the measured asteroid pointing. Whenever the limit is crossed however, the primary (predicted) attitude will be used.

This strategy allows to provide flexibility to ground for the design of attitude profiles, while benefiting from more accurate pointing performance.

Following sensors will be used for the semi-autonomous guidance mode:

- Star tracker of the inertial attitude
- IMU for increased performance and robustness in the inertial attitude estimation
- Sun sensors for FDIR
- AFC camera for relative navigation (this includes the on-board image processing) and for collision risk assessment

The reaction wheels will be used for attitude control.

Ground will have the capability to command the GNC thrusters in this mode, as for the operations mode. This can be used for small delta-V manoeuvres or for angular momentum management. The latter functionality can be performed autonomously on-board, but it may be better to leave this responsibility to ground in close proximity, in order to better control the perturbations on the relative orbit.

4.3.2.6 Close Fly-by Mode

In case a close fly-by doesn't need to be performed (<2km), e.g. for high resolution imaging, or in order to deliver a certain payload (COPINS), a specific system mode will be needed that is able to autonomously update the trajectory in order to better control the point of closest approach (perigee altitude) while ensuring that the spacecraft is never in a collision course. This would be the responsibility of the close fly-by mode.

The same sensors and actuators are baselined as for the semi-autonomous guidance mode.

The main difference with the previous mode is that the also the spacecraft trajectory is now controlled autonomously on-board, whereas only the attitude pointing is controlled for the semi-autonomous guidance mode. This implies, for the close fly-by mode, that several small delta-V manoeuvres will be executed autonomously on-board.

4.3.2.7 Collision Avoidance Manoeuvre (CAM) Mode

The on-board availability of feedback on the relative state of the spacecraft with respect to the asteroid, enables the assessment of a collision risk and the derived mitigation measures.

The collision risk is monitored in close fly-by mode. Upon detection of a real collision threat, an escape manoeuvre is defined and CAM mode is entered for execution of that manoeuvre.

Note that the CAM mode does not compute a collision avoidance manoeuvre by itself, it only executes that manoeuvre. The parameters that define the CAM manoeuvre are to be provided by the collision risk monitoring module that is active during close fly-by mode. Alternatively, the CAM parameters can be provided by ground.

The CAM parameters (in fact simply a delta-V and direction) are to be provided in with respect to the inertial reference frame, such that the CAM can be executed also in case of loss of relative navigation, i.e. based on star tracker only.

Following sensors and actuators are baselined for the CAM mode:

- Star trackers inertial attitude estimation
- IMU for robustness in the attitude estimation
- Reaction wheels for attitude control
- GNC thrusters for the actual CAM

After a successful CAM manoeuvre, a transition to safe mode is autonomously commanded.

5. Ground Segment Characteristics

The ground segment configuration and operations are detailed in this chapter.

5.1 GROUND SEGMENT CONFIGURATION

Before identifying and describing the individual functions of the ground segment components, specific entities are defined. Ground segment is composed of operational entities, which are organisations or groups of people focussed on providing a coherent set of functions in the ground segment. The use of entities ensures a clear split of responsibilities.

The following high level mission segment architecture is proposed:

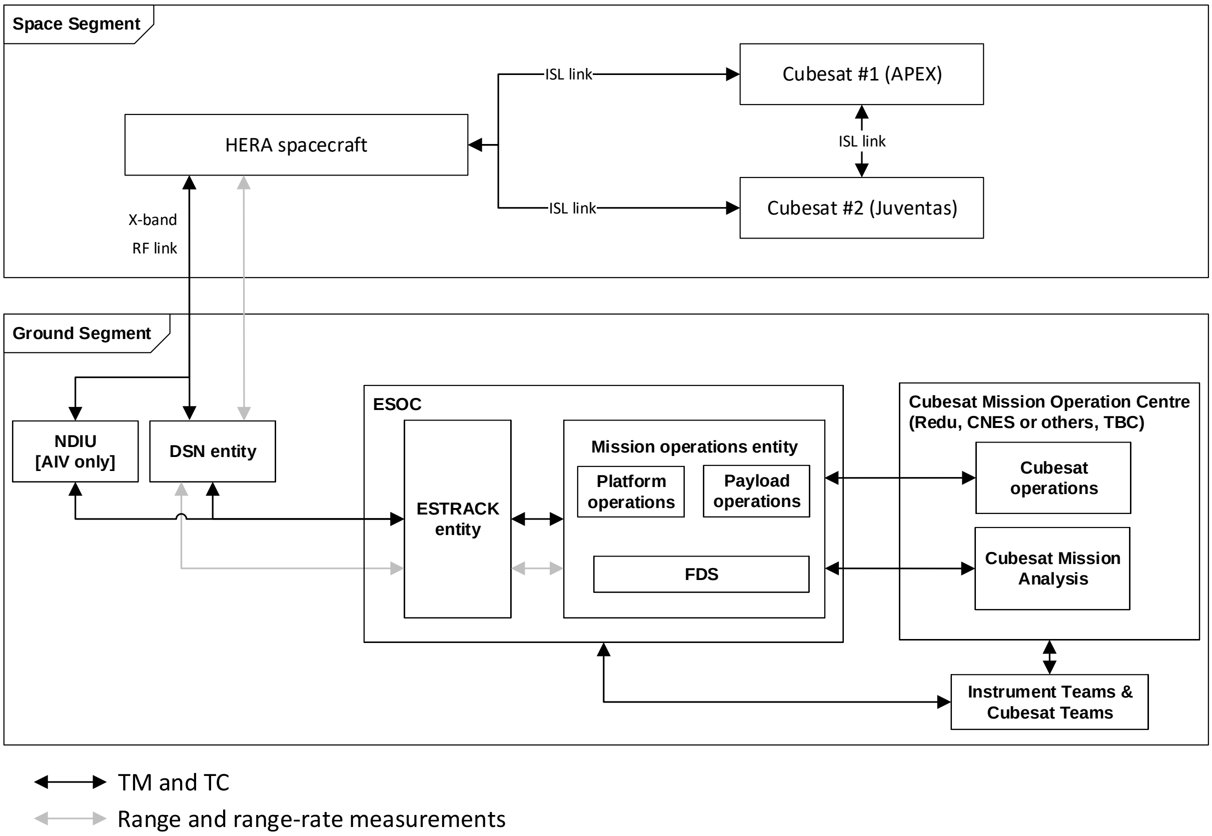

Figure 5-1: The overall mission architecture presenting the different ground segment entities

The specific relationships between these entities are presented in a schematic diagram in Figure 5-2.

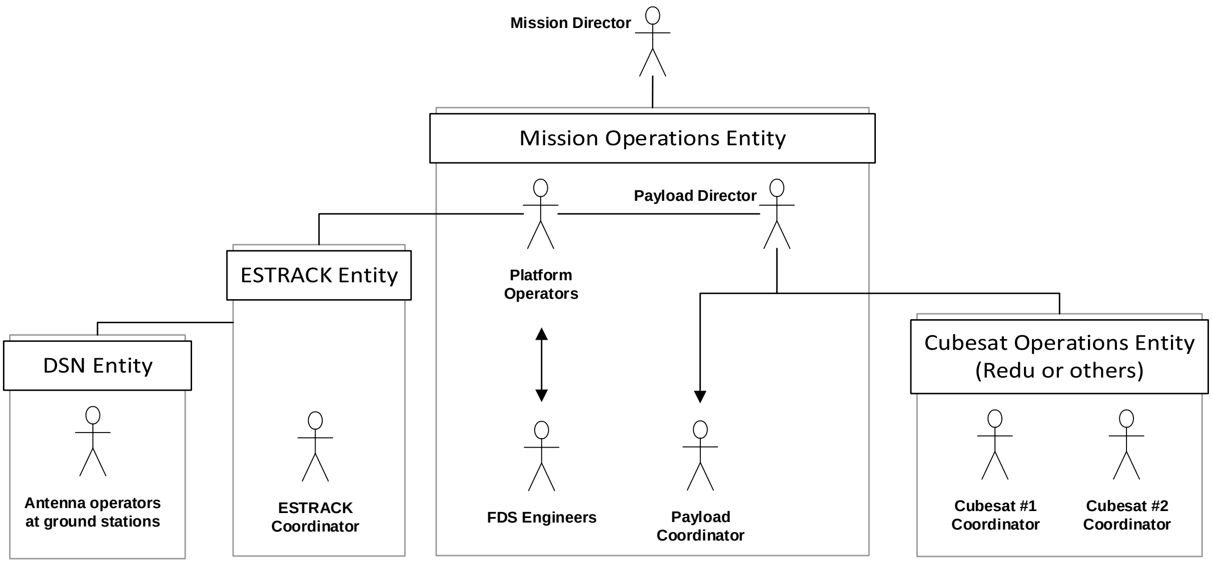

Figure 5-2: Overview of the entities and the actors and the relationships to one another

A description of the different entities and their responsibilities is given in the proceeding sections.

5.1.1 MISSION OPERATION ENTITY

The mission operations entity is the central entity in the overall ground segment. It is located at ESOC and consists of 2 main actors; the platform operators and the FDS engineers (with supporting industry engineers if applicable). The entity is under the control of the mission director.

The entity interfaces with the ESTRACK entity for sending commands to the spacecraft and receiving the telemetry. Through the ESTRACK entity, the ground antenna planning and coordination is performed. Also range and range-rate measurement data is received from the ESTRACK entity.

It's important to note that payload and cubesat operations are done outside of ESOC, and therefore, the mission operation entity will interface with another, external cubesat and/or instrument operations entity.

Exact configuration of this arrangement is still to be defined, nevertheless, the current baseline assumes the payload operations at ESOC are limited to providing FDS inputs for both HERA payloads and cubesats, and merging the TC sequences received from the HERA payloads and the cubesats into the uplink towards HERA. This proposed external cubesat mission operation centre is currently being considered to be located at either the ESEC in Redu or CNES, or combination of the two entities, and will be elaborated in future phase studies.

A schematic summarising the functions at the mission operations entity is given in Figure 5-3.

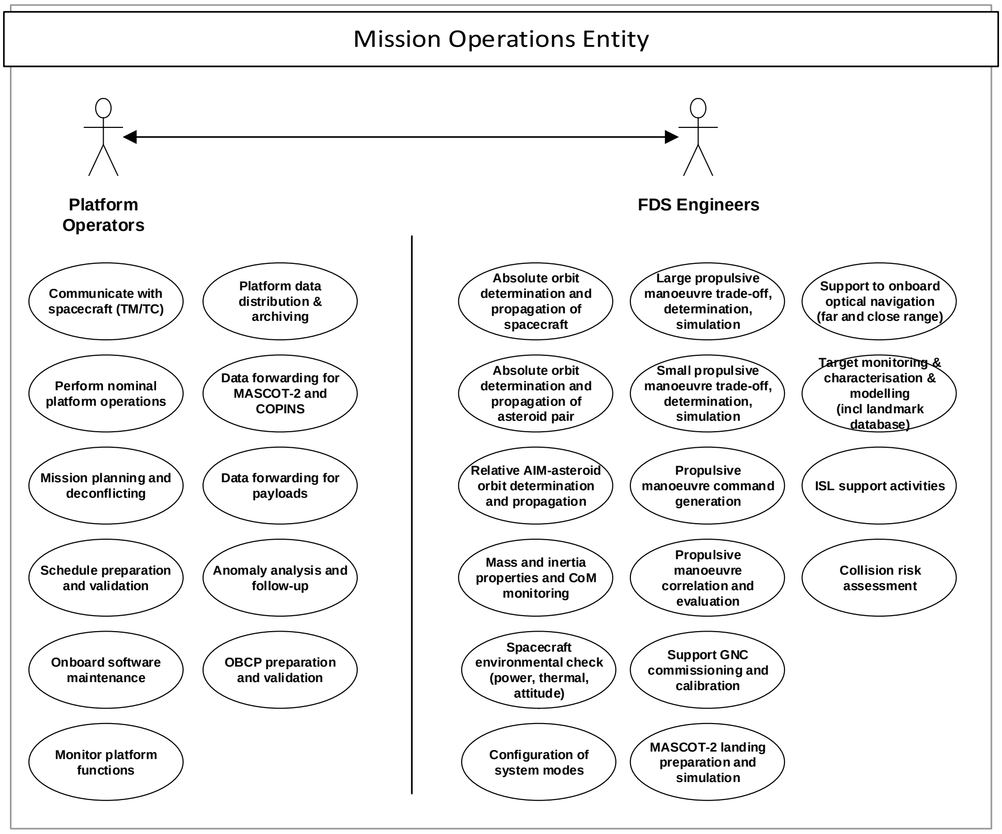

Figure 5-3: Mission operations entity with actors and functional blocks

Each of the responsibilities of the platform operators are discussed in detail in the proceeding subsections. The FDS functionality is presented in Section 7.1.

Note that the functions of the platform operators are typically not mission specific, and it should be very similar to any deep space mission.

5.1.1.1 Communication with Spacecraft (TM/TC)

One of the primary functions of the mission operations entity is to communicate with the spacecraft, which involves sending telecommand and receiving telemetry. For this function, the existing mission control systems (e.g. SCOS) can be used. Specific deep space features are required, for example to cope with the correct correlation of command and command verification messages downlinked with a significant time delay. Specific tools are required to provide the operator with an up-to-date status information on all commands (based on the command acknowledges), whether the commands were send in real-time or from the on-board scheduler.

It is recommended to re-use the work performed during AIV for the operational phase. The satellite database (MIB) is an excellent example of this. The MIB can be populated and validated during AIV and re-used during operations. This requires the use of the same or similar mission control system in AIV and in operations. A second example is the operational procedures, which can also be developed and tested during AIV by industry and later re-used in operations. This is more complex and requires a specific combined approach between industry and ESA. This was the baseline on previous PROBA missions, so the option can be considered.

5.1.1.2 Nominal Platform Operation

This function covers all operations required for routine, day-to-day operations of the spacecraft. It involves a diverse set of activities, such as power and thermal control of the spacecraft, downloading of the platform and payload data. Due to the deep space nature of the mission, these activities are not performed in real-time, but loaded in the scheduler for time-tagged execution on-board.

No specific requirements from the platform have been identified. All of the required platform functions will be available through a telecommand. The on-board scheduler size will be sized to incorporate all required commands (platform and payload operations) for at least 72 hours (TBC). This has to be taken into account also for the uplink capability.

5.1.1.3 Mission Planning and De-conflicting

The planned platform and payload activities need to be combined into a single mission planning. Specific activities incorporate constraints for other parts of the system. All these constraints need to be taken into account. For this function, existing mission planning tools can be used.

No specific requirements from the platform for the mission planning tool itself have been identified. During further detailing of the mission, the constraints of different activities will become clear and will serve as an input for the definition of the mission planning tool. This customisation is always mission specific.

5.1.1.4 Schedule Preparation and Validation

The planned platform and payload activities need to be combined into specific TC sequences which can be uploaded efficiently towards the spacecraft. This is done by the mission planning and de-conflicting function (see above). The output is a consolidated planning and the corresponding TC sequences. A consistency check of the resulting sequences is required before upload.

No specific requirements for the function have been identified, other than the uplink capacity and on-board scheduler size.

5.1.1.5 On-board Software Maintenance

The on-board computer will be running software that can be (re)-configured to the demands of the mission or specific mission phases. This is usually done by:

- Patching of configurable parameters
- Patching of code of hard-coded parameters on the running software (non-persistent memory)
- Patching of code of hard-coded parameters on the offline software (persistent memory) – requires reboot to activate the patch
- Uploading of new on-board software (entire image in persistent memory)

All of the above functions require a ground function to keep track of the changes and facilitate the upload process. Existing software maintenance tools, tuned towards the mission specific software can be used.

Apart from the nominal mission specification, no specific requirements from the platform have been identified.

5.1.1.6 Monitoring of Platform Functions

The mission operations entity has to monitor all on-board systems to assess the health and detect issues whenever they become apparent. This includes the analysis of generated events, software statistics, power, thermal and data budgets, etc. This function has a clear link to the “spacecraft environmental check” function of the FDS engineers.

Existing tools and applications can be used for this. Apart from the nominal mission specification, no specific requirements from the platform have been identified.

5.1.1.7 Platform Data Distribution and Archiving

From the mission operations entity, platform data is extracted and forwarded to the payload entities and support teams in industry or remote sites.

A second part of this function is proper archiving of the platform data.

Existing tools and applications can be used for this. Apart from the nominal mission specification (define clear IFs and data formats), no specific requirements from the platform have been identified.

5.1.1.8 Data Forwarding for Payloads

The data from the payloads is received through X-band and made available to the mission operations entity through the ESTRACK network. The science data needs to be forwarded to the payload operations entity for detailed processing.

Existing tools and applications can be used for this. Apart from the nominal mission specification (define clear IFs and data formats), no specific requirements from the platform have been identified.

5.1.1.9 Anomaly Analysis and Follow-up

When anomalies are detected, the necessary actions need to be initiated to gather all information, perform detailed analysis and propose solutions (if required) to resolve the anomaly. A specific anomaly tracking system is required for this.

Existing tools and applications can be used for this. No specific requirements from the platform have been identified.

5.1.1.10 OBCP Preparation and Validation

The baseline is to use OBCPs as on-board flight control procedures and for specific FDIR. This requires a ground tool to prepare the OBCP and validate it on the spacecraft simulator. OBCPs are used for many different missions, such that there is an extensive experience within the operations team. The baseline OBCP implementation is based on µPython, which provides extensive and rich preparation tools. Basically any computer can be equipped with Python and can therefore serve to produce and test OBCPs. Final validation will need to be performed with the spacecraft simulator, running the flight onboard software.

5.1.2 ESTRACK ENTITY

The ESTRACK entity is the main interface between the different ground stations (antennas) located at different sites in the world and the mission operations entity. It uses the existing facilities available at ESOC to distribute the data (e.g. OPSNET). The ground station currently identified are 15m X-band up-/down-link antenna (e.g. Kourou) for LEOP/commissioning and ESA DSN antennae for the remainder (i.e. most) of the mission.

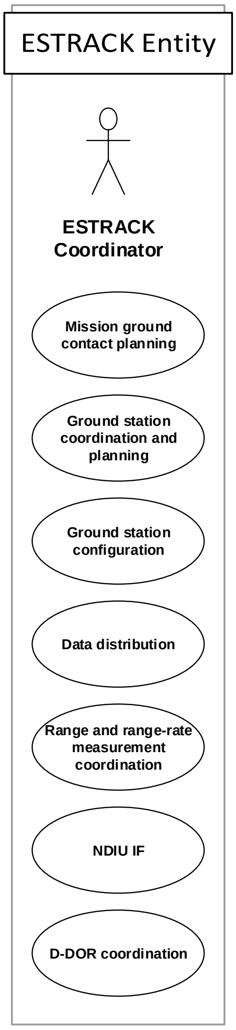

Figure 5-4: The ESTRACK entity and its actors and functional blocks

5.1.2.1 Mission Ground Contact Planning

The ESTRACK entity will be responsible for the planning of the mission ground contacts. This complex functions combines the mission planning and mission ground contact need with the station availability and visibility.

Existing tools and applications can be used for this. Apart from the nominal mission specification no specific requirements from the platform have been identified. Due to the need for specific 15m and DSN antennas, the availability of the ground stations shall be carefully assessed.

5.1.2.2 Ground Station Coordination and Planning

Based on the above, a detailed ground station planning is made. Details for the planned contacts are shared with the ground station. The ESTRACK entity will coordinate the ground stations to ensure correct services when required by the mission operations entity.

Existing tools and applications can be used for this.

5.1.2.3 Ground Station Configuration

For ground stations which are operated remotely, the ESTRACK entity will provide all the functions to configure the antenna and station for the spacecraft contact.

Existing tools and applications can be used for this.

5.1.2.4 Data Distribution

The forwarding of telecommands towards the ground stations on one side and the forwarding of telemetry and science data from the ground stations on the other side is the key function of the ESTRACK entity. Specific systems might need only real-time or replayed data from selected virtual channels or APID's.

5.1.2.5 Range and Range-rate Measurement Coordination

The ESTRACK entity coordinates the range and range-rate campaigns with the different ground stations. It forwards the measurements to the mission operations entity (FDS engineers) who have the applications to interpret the data and translate it into the orbit determination inputs.

Existing tools and applications can be used for this.

5.1.2.6 NDIU Interface

During the AIV campaign, an NDIU will be provided to Industry. The NDIU will provide a TC/TM interface for ESOC towards the spacecraft when in the cleanroom at the Prime's premises, testing facilities and launch site. The unit is mostly used for operations preparations tests from ESOC. The connection and configuration of the NDIU and the data exchange between spacecraft and the mission operations entity is under the control of the ESTRACK entity. This ensures a similar interface for the mission operations entity during AIV and later during operations.

Existing tools and applications can be used for this.

5.1.2.7 D-DOR Coordination

The D-DOR measurements require specific coordination activities as multiple ground stations are used at the same time. The coordination is under control of the ESTRACK entity, which forwards the data towards the mission operations entity for detailed processing.

Existing tools and applications can be used for this.

5.1.3 DSN ENTITY

The DSN entity provides the communication capability with the space segment. On one side it interfaces with the ESTRACK entity for coordination and configuration, and on the other side it communicates with the space segment on X-band. The communication link is bi-directional (up- and downlink). It is assumed that the DSN entity makes use of the SLE protocol for data exchange towards the mission operations entity. The different functional blocks are described in the schematic shown in Figure 5-5.

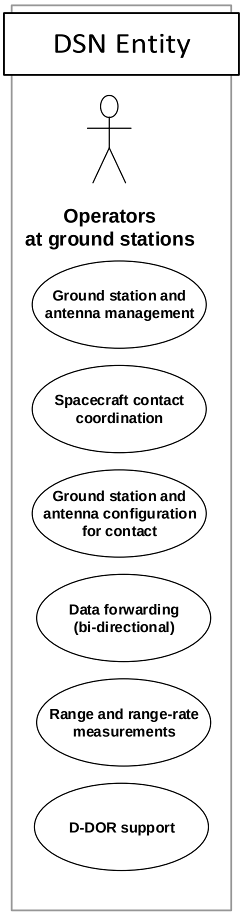

Figure 5-5: The DSN entity and its actor and functional blocks

5.2 GROUND SEGMENT OPERATIONAL CONSIDERATIONS

This chapter presents the design drivers for the ground segment, as seen from the mission design. The focus is on specific recommendations and assumptions from the mission design, mainly related to:

- Satellite operations
- GS availability
- Mission and payload operations planning
- Data volumes to be processed
- System performance requirement

Recommendations are structured per mission phase. The detailed description of the mission phases can be found in Chapter 6.

5.2.1 LEOP

Two aspects of the Launch and Early Operations phase (LEOP) are retained as potential drivers for the Ground Segment: the activation state of the S/C during launch and the availability and selection of ground stations. Both aspects are elaborated below.

5.2.1.1 Launch Condition

The baseline is to launch the satellite in OFF condition.

There is no technical reason to have the spacecraft ON during launch. This baseline is used for all PROBA launches so far, which means that the existing design of the launcher separation circuits within the OBC can be re-used. It is also advised from launch authorities and significantly simplifies the spacecraft-launcher compatibility (no EMC during launch).

This has the following impact for the operations and ground segment:

- Access to the spacecraft is restricted some days before the launch
- Industry operators at launch site can be released prior to launch and support LEOP activities at the MOC (these operators can be available to support the LEOP activities from ESOC)
- No data I/F is required to the spacecraft through the launcher (although battery charge I/F and monitoring of propulsion is required in any case)

A preliminary analysis does not show major issue when launching in OFF state for the platform itself. Detailed analysis of -for example- the thermal behaviour of the system in this phase shall however be performed in the next project phases.

5.2.1.2 Ground Station Selection and Ground Segment Availability

The following mission aspects play a role in the management of the Ground Segment:

- The spacecraft is injected into a heliocentric orbit after launch, the distance from the Earth hence increases rapidly.
- The spacecraft has only X-band up- and downlink. There is no other communication system available for the LEOP.
- The spacecraft has omni-directional uplink and downlink capability to allow commanding and reception of telemetry at all times during the LEOP. The data rates depend on ground antenna characteristics and varying distance during the LEOP, and are driven by the use of low gain antennae (LGA).
- The spacecraft has no means to perform absolute navigation on-board. This puts additional requirements on ground stations, contact times and tracking capabilities to determine the orbit and define possible launcher injection correction manoeuvres.
- During the LEOP, the spacecraft is brought into the nominal state (deployment of solar panel, priming of the propulsion systems and later the acquisition of 3-axis stabilized attitude). Although most of these activities are triggered by the release from the launcher and are executed autonomously on-board, sufficient ground contact time is required to monitor the health of the spacecraft, retrieve the necessary telemetry and react quickly in case of anomaly.

For the LEOP, several X-band up- and downlink stations will be required. A similar approach as used for GAIA (SpaceOps 2014 - GAIA mission operations concept and launch and early orbit phase – In-orbit experience) and LISA Pathfinder (LISA Pathfinder Acquisition of Signal Analysis after Launch Injection and Apogee Raising Manoeuvres), which use 15m X-band antenna’s during LEOP is baselined. As also used for LISA Pathfinder, X-band Acquisition Aid (XAA) could be used to find the spacecraft after launch injection, in the absence of an S-band system.

XAA is an antenna with small diameter (1.2m) and consequently with large antenna 3dB beam-width (1.5 deg), installed on the LEOP 15m ground station at ESOC. It is used to acquire the X-Band downlink, auto-tracking the satellite based on the same signal and steering the 15m antenna pointing until achievement of acquisition and autonomous tracking.

During the LEOP, spacecraft operations can be time critical and therefore require a high priority on the ground stations, the selection of backup ground station(s) and appropriate ground segment availability at ESOC.

5.2.2 COMMISSIONING

During the commissioning, the different subsystems will be tested and calibrated where required. Not only the space segment is commissioned, but also the different ground segment systems are heavily involved and tested in operational conditions.

The following aspects play a key role:

- The propulsion system commissioning needs to be executed relatively soon after LEOP. A launcher injection correction manoeuvre needs to be executed within 7 days (TBC) after LEOP. For the commissioning of the propulsion system, several test burns will need to be executed to properly health check and calibrate the propulsion system prior to this manoeuvre. Sensors on-board the spacecraft will provide part of the information (e.g. accelerometers, star tracker), but specific range and range-rate measurements from ground will be required in addition. These tests will also provide necessary inputs to refine the spacecraft models in the FDS applications.

- For the commissioning of the communication system, different settings will need to be tested. The intention is to cover the different communication configurations which will be used during the mission. Specific ground segment support will be required for these tests.
- Some of the techniques required for close-proximity operations will be (preliminarily) tested during the commissioning phase (e.g. characterization of sensors). The commanding will use the specific ground segment applications and also the returned data will be inserted in the FDS applications.
- Other techniques required for close-proximity operations may not be tested prior to the commissioning phase.
- For payload commissioning, it is the intention to involve the payload teams and use the standard ways to command and retrieve the science data.
- Specific commissioning activities will require increased availability of the ground segment.

5.2.3 CRUISE PHASE

For the cruise phase, the availability requirement of the ground segment is considerably lower than for LEOP, commissioning and for the close-proximity operations. During the cruise phase, Doppler information, for instance, can be acquired when receiving TM via a coherent link such that, together with ranging, a single 8-10 hrs pass every month during a coasting cruise is sufficient for deep space navigation. This has to be traded-off, however, against the need for ground visibility and the risk of a spacecraft issue requiring ground intervention. The accepted compromise is one pass every week and will be used as the baseline for this mission during the Deep Space Cruise Phase.

It is assumed that the tracking campaign for the DSM at launch plus two months can be done with RARR (Range and Range Rate (Doppler)) data alone, especially because it will be done at the end of the Commissioning Phase during which regular and extensive tracking data will have been collected. A ΔDOR is however baselined, in order to get the best possible accuracy and to optimize the DSM.

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 the platform has been designed such that there is no such constraint for the HERA mission. A Deep Space Hibernation Phase is therefore not considered.

5.2.4 APPROACH PHASE

During the approach phase the relative orbit determination becomes increasingly important. This requires the download of AFC images and interpretation in the ground segment. During this phase, also the asteroid(s) characterisation based on in-flight data will start on ground.

During the approach phase, several braking manoeuvres are required. This inherently puts timeliness requirements on absolute and relative orbit determination and determination of the next manoeuvres.

Δ-DOR campaigns are considered as baseline for orbit determination around the major events, such as DSM and braking manoeuvres, in order to obtain the best accuracy. RARR-only campaign withheld as backup option.

5.2.5 CLOSE-PROXIMITY OPERATIONS

The close proximity operations phase is clearly the most demanding phase for flight operations, including the ground segment. The following key aspects are identified:

- Detailed modelling of the asteroids is required based on the AFC images
- Many FDS functions can only be commissioned once the spacecraft is in close proximity of the asteroids. This requires that the FDS is commissioned, while accurate outputs are required at the same time.
- During this phase, the payload operations are a significant part of the operations. Detailed timelining is required between the payload teams and the mission operations teams to combine scientific requests, mission and platform constraints and operational feasibility.
- For the delivery of the COPINs cubesats, the spacecraft may need to perform specific manoeuvres prior to release of the payload. Extensive ground segment activities (mainly simulations) will be required in preparation and planning of the deployment manoeuvres.
- Due to increased data volume generated on-board, additional ground segment capacity (more or longer contacts) may be required to downlink the platform and science data and distribute it to the relevant teams.
- Timeliness requirements will be applicable in this phase, which will require regular spacecraft contact periods. The data required to perform the relative navigation has to be downlinked for on-ground processing regularly. This is to ensure a proper status and health monitoring.
- Additional monitoring activities (e.g. collision, thermal and power impact) become very important in this phase.
- The communication with the COPINs cubesats using the ISL and the HERA spacecraft as relay will need to be taken into account in the on-board data management, the payload operations, required downlink time, on-ground data distribution, analysis capabilities, etc.

Table 5-1: Design drivers per phase

Phase	Duration	Contact planning
LEOP	3 days	15m Antenna Day 1: Visibility on first 6 hours after separation, 10 hours total contact time in first 24h. No gaps longer than 4 hours. Day 2: 8h contact/day spread over at least 3 contacts. No gaps longer than 6 hours. Day 3: 8h contact/day spread over at least 2 contacts. No gaps longer than 10 hours. Ground response time: 10h Backup station required for LEOP only XAA support for acquisition High priority on ground antenna required.
Commissioning + DSM	2 months	DSN Antenna (35m) 4h to 8h contact/day, 5 days per week Ground response time: 72h Specific ΔDOR and RARR campaigns around DSM (before and after)
Initial Cruise Phase	1 month	DSN Antenna (35m) Gradual decrease from 4h to 8h/day 5days/week to 4h to 8h/day 1day/week Ground response time: 7 days Nominal RARR once a week
Deep Space Cruise Phase	11 months	DSN Antenna (35m) One 8h contact/week Ground response time: 7 days Nominal RARR once a week Asteroid/arrival ops. Preparations continue

Phase	Duration	Contact planning
Asteroid Approach Phase	2 months	DSN Antenna (35m) 4h to 8h/day 3 to 5 days/week Ground response time: 72h Specific ΔDOR and RARR campaigns around approach manoeuvres (before and after) First far range AFC images in support of relative navigation Instrument commissioning well before first arrival manoeuvre Test burn two weeks before first burn
Close-proximity Operations	6 months	DSN Antenna (35m) 4h to 8h/day 5 days/week Ground response time: 72h Regular ΔDOR and RARR campaigns required Critical ops at approach prior to autonomous phase, and for COPINS proximity operations.

6. Mission Operations Concept

In this chapter, each of the mission phases is highlighted individually and the operations are presented together with their implications on the flight and ground segments. Figure 6-1 shows the orbits of HERA spacecraft, the earth and the asteroid about the sun, and visualises the different mission phases.

For each of the mission phases, the following aspects are discussed (if relevant):

- Payload operations
- Orbital state and required orbital manoeuvres
- Distance and orientation wrt binary asteroid (including effect on visual camera and lighting conditions of the asteroids)
- Guidance/system mode (attitude) and required slew manoeuvres
- Power and communication constraints
- Ground segment involvement (ground contacts, processing, orbit determination campaigns)
- Specific events

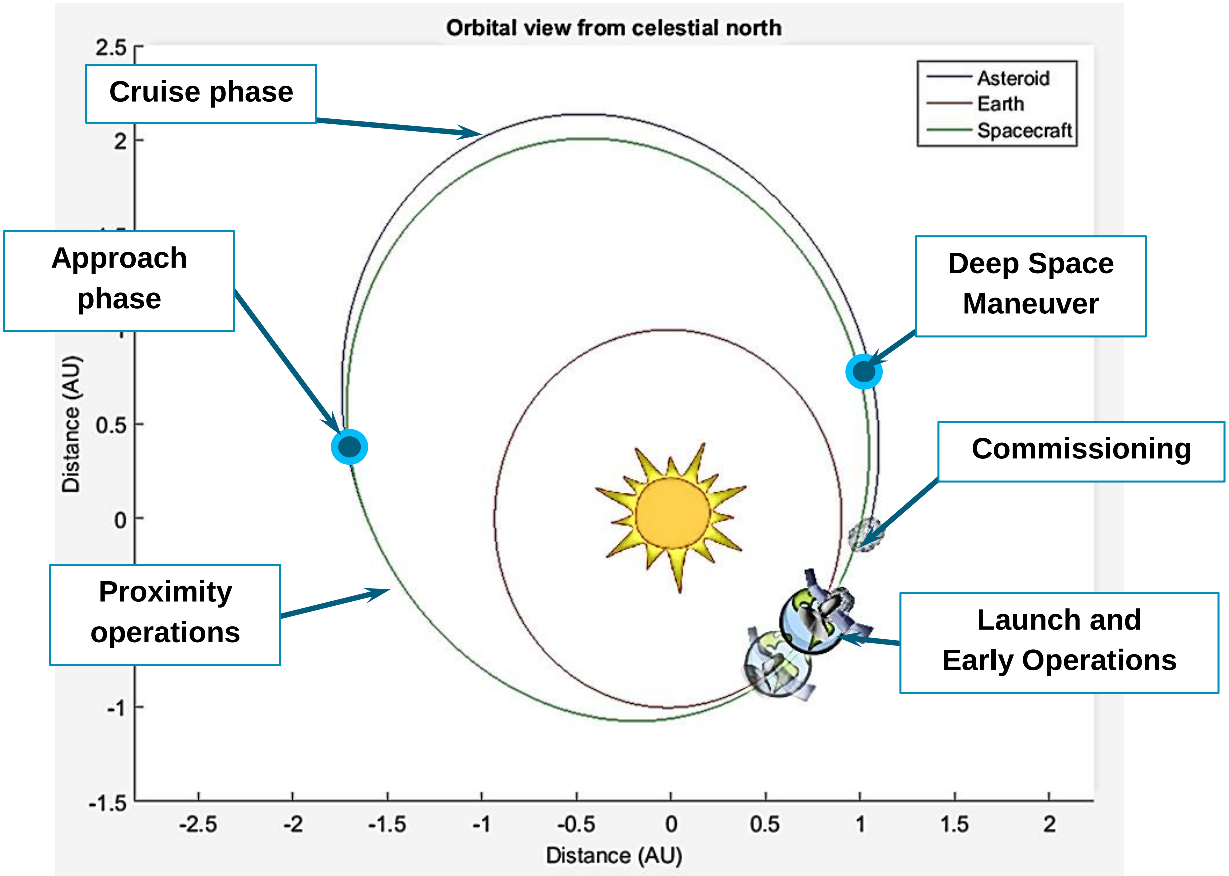

Figure 6-1: HERA orbit and mission phases

6.1 LAUNCH CAMPAIGN

The baseline mission scenario foresees a launch in 2023 (backup in 2024) on a Soyuz 2.1b with Fregat MT upper stage from either Baikonour or Vostochny. The back-up launcher is Ariane 6.2 from Kourou.

The launch campaign typically starts two to three months before the launch date. It involves the following spacecraft activities:

- Transfer of the spacecraft to the launch site
- Final integration of the spacecraft (e.g. mounting of deployable solar panels)
- Functional testing of the spacecraft
- Filling of propulsion and pressurant tanks
- Removing of covers and integration of arm connectors (if any)
- Integration of spacecraft on launcher
- Monitoring of the battery voltage and the propulsion tank pressure and temperature

The launch campaign is added to the operations timeline because it may have an impact on the LEOP and commissioning phase.

6.2 LAUNCH AND EARLY OPERATIONS (LEOP)

Following the launch, the spacecraft is inserted into a an intermediate, circular parking orbit at 185 km altitude and 51.6° deg inclination. By that moment, the first two launcher stages have been separated and the Fregat has performed the first burn. Once in the parking orbit, a coast arc of roughly 1 hr 20 min takes place after which the Fregat upper stage performs the second and final burn to inject the spacecraft into an earth escape hyperbolic arc for the interplanetary transfer.

The duration between lift-off and separation of the spacecraft from the launcher is about 1 hr 45 min, assuming that the separation occurs up to a few minutes after the final Fregat burn.

Upon detection of the separation from the launcher, the spacecraft is powered on and autonomously initiates a sequence to de-tumble, deploy its solar array, and prime its propulsion subsystem (GNC thrusters). It is assumed that the star trackers will not be able to acquire a fixed navigation immediately after separation from the launcher. Hence, the spacecraft will not have full knowledge of its attitude. This is the reason why survival mode is entered first after separation: it has de-tumbling capability and it imposes a slow rotation about the sun vector once stabilized. Survival mode is not a backup for safe mode, as it is not earth pointing. During LEOP however, earth pointing is not required as the LGA provides omni-directional coverage and relatively high data rates. It is the intention to use the reaction wheels right after separation (as is done on PROBA satellites with same wheels), although the survival mode will be able to work with thrusters only as well. The baseline reaction wheels do not need any warm-up time and can be activated autonomously without constraints.

The LEOP ends when the spacecraft is in a safe situation and ready for the commissioning activities. The LEOP is estimated to last one to three days.

During LEOP, the ground segment will be responsible for tracking the satellite, such that telemetry can be received and the health of the satellite can be assessed. A series of ground stations can be involved, including smaller X-band antennae as the distance between the spacecraft and the Earth will still be limited. Regular contacts will be needed to follow the progress of the spacecraft and for contingency. The baseline is to have as close as possible to 24/7 contact with the spacecraft during LEOP.

It is recommended to use a DSN ground station with high northern latitude during this phase, as it would provide 22 hr of contact/day. A 15 m (U/D) X-band antenna could be used as support of the DSN antennae.

Note that the launch window lasts for about 3 weeks. The exact launch date will have an impact on the overall planning, but sufficient margins are available to perform all operations.

Table 6-1: Summary of LEOP phase

Start date	2023/22/10 (LPO) – 2023/12/11 (LPC)
Duration	1-3 days
Spacecraft activities	Launch Acquisition of parking orbit (by Soyuz Fregat) Injection in interplanetary orbit (by Soyuz fregat) Separation from the launcher Deployment of solar panels Priming of propulsion subsystem (GNC thrusters) Detumbling Acquisition of stable and safe attitude
Distance to earth	0.0 – 0.01AU
Distance to sun	1.0 AU
Distance to asteroid	~0.9 AU
System modes	Survival mode
Payload operations	N/A
Events	Launch
Constraints	Launch conditions
Propulsive manoeuvres	None (performed by spacecraft)
Communication	LGA: up- and downlink, high data rate, omni-directional
Ground segment activities	Tracking of spacecraft (ground stations TBD, 15m antenna)

	Reception of first telemetry (LGA) Uplink of first telecommand(s) Assessment of spacecraft health

6.3 COMMISSIONING

During the first week after launch, the launch injection errors should be corrected at the boundary of the Earth's sphere of influence. This requires an orbit determination campaign from ground, and the commissioning of the transfer thrusters to perform the propulsive manoeuvres.

During this short period of time, the following commissioning activities should be performed:

- Commissioning of 3-axis attitude control capability
  - Verification and calibration of related sensors and actuators
  - Testing of on-board attitude determination and attitude control
- Commissioning of propulsion subsystem
  - Verification of sensors
  - Priming of transfer branch of propulsion subsystem (bi-propellant)
  - Test burns and calibration

The Deep Space Manoeuvre (DSM) is assumed to occur 2 months after launch, in order to enter the interplanetary transfer trajectory on an intersection course with Didymos. This period of time is available for the remaining commissioning activities, of which the most important are:

- Commissioning of communications subsystem (HGA)
- Extended health check of the platform
- Health checks of payload (if possible)
- Check-out of time correlation chain

Note that the exact DSM time, direction and amplitude are dependent on the launch date (and possible further optimization) but remains within a time frame of 10 days. The execution of the DSM and possible clean-up manoeuvres is assumed to last less than 10 days.

It is highly recommended to attempt a preliminary commissioning of the AFC camera with the Earth or the Moon as target. These targets may not be representative for an asteroid, but they present a unique opportunity to perform some tests and calibration.

A 2 x 4 hr/day for 5 days per week ground contact baseline is proposed for the commissioning activities, at least for the first month. Frequent passes are necessary to perform and assess all the commissioning activities. During the first and last weeks of the commissioning phase, ΔDOR and Range and Range Rate (RARR) measurement campaigns will be conducted in order to estimate the spacecraft orbit for the LIC and DSM manoeuvres. For each of the RARR campaigns, the spacecraft will be tracked over an arc of several hours (TBC).

As for the LEOP phase, it is recommended to use a DSN ground antenna at high northern latitude to perform the commissioning activities.

Table 6-2: Summary of commissioning phase

Start date	2023/25/10 - 2023/14/11 (depending on launch date)
Duration	~2.5 months
Spacecraft activities	Commissioning of platform units and subsystems Priming of transfer thrusters Commissioning of payload (TBC) Correction for launch injection errors (LIC) Deep Space Manoeuvre (DSM)
Distance to earth	0.01 – 0.2AU
Distance to sun	1.0 – 1.2AU
Distance to asteroid	0.9 – 0.7AU
System modes	Operations mode: commissioning activities and scheduled downlinks (i.e. most of the time) Propulsion mode: execution of LIC and DSM
Payload operations	Commissioning (TBC)
Events	N/A
Constraints	LIC correction to be executed within one week (TBC) from launch
Propulsive manoeuvres	Correction of launch injection errors (LIC) Execution of Deep Space Manoeuvre (DSM) and clean-up manoeuvers
Communication	LGA: up- and downlink, moderate data rate HGA: up- and downlink, high data rate
Ground segment activities	Reception of telemetry Scheduling of commissioning activities Assessment of commissioning activities Orbit determination campaign (ΔDOR and range and range rate) for LIC and DSM manoeuvres

	Determination, scheduling and execution of launch injection correction, deep space manoeuvre and possible clean-up manoeuvres

6.4 CRUISING PHASE

The cruising (or transfer) phase comes down to waiting for the intersection with the binary asteroid. The transfer takes approximately 3 years during which the second DSM will be executed for target correction. This phase is characterized by a long period of limited activity imposed by the large distances from the Sun and the Earth, as well as to the asteroid, and the resulting restrictions on available resources.

The spacecraft will be in operations mode during this phase, but little activities will be scheduled on the mission timeline (MTL): weekly LGA downlinks and regular health checks only. The spacecraft is hence effectively be in standby most of the time, simply gathering housekeeping telemetry every month. The battery status of the COPINS payload will be monitored and battery charging will be commanded if necessary.

During the cruise, the spacecraft will be 3-axis stabilized using star trackers, and will be mainly earth-pointing (attitude profiles provided by ground). A 180° slew manoeuvre about the earth axis will be commanded up to twice a day using reaction wheels, in order to minimize the angular momentum build-up caused by solar radiation pressure asymmetry.

In case of major anomaly, the spacecraft will enter safe mode, which ensures that the LGA is pointed to the Earth and that the RF transmitter is switched on. If a failure occurs that prevents maintenance of the 3-axis stabilized mode, the spacecraft will revert to survival mode. In this mode, the spacecraft is sun-pointing with an angular offset that ensures that the LGA is in omnidirectional with the Sun and the Earth. The spacecraft then uses sun sensors (an IMU) for its navigation.

The cruise phase will stop about one month before the expected first braking manoeuvre. This allows sufficient time for further commissioning and monitoring activities, and for performing the first on-board observation of the binary asteroid to prepare for the next propulsive manoeuvre.

As the distance between the spacecraft and the Earth will vary over a long range, different uplink and downlink data rates will be available. The default data rates will need to be updated on-board as the mission progresses.

The ground segment activities will be limited to health monitoring and tracking (RARR campaigns) of the spacecraft. One 8-hr pass per week is envisaged as baseline. This should be compatible with the requirements for ranging, and should enable a regular check of the evolution of the spacecraft health. More frequent contacts can be scheduled if needed in case of emergency.

A high solar conjunction period is expected in the middle of the cruise phase, which is likely to compromise the RF communication for up to a few weeks. Therefore, it will be important to provide a sufficiently long MTL and to foresee periods of autonomous angular momentum offloading.

The Deep Space Antennae will be baselined for this phase (as for most of the mission).

Table 6-3: Summary of cruising phase

Start date	T0+2.5 months
Duration	~3 years
Spacecraft activities	Spacecraft is in standby
Distance to earth	0.2 – 3.2 AU
Distance to sun	1.2 – 2.2 AU
Distance to asteroid	0.7 – 0.02 AU
System modes	Operations modes: stand-by, scheduling of X-band downlinks and health checks
Payload operations	N/A
Events	Solar conjunction between 2021/10/23 and 2021/11/11
Constraints	Largest distance from the sun Largest distance from earth (including solar conjunction)
Propulsive manoeuvres	None (only angular momentum offloading)
Available data rates	LGA: N/A HGA: uplink and downlink at low to moderate data rates
Ground segment activities	Reception of telemetry Assessment of spacecraft health Orbit determination campaign (RARR) at regular intervals

6.5 RENDEZVOUS

The approach (or insertion) phase consists of the navigation relative to the asteroid based on the on-board visual camera, and of braking manoeuvres to obtain a final relative position of about 30 km (TBC) from the asteroid.

Each braking manoeuvre will be composed of the following activities and will last one week each:

IMU calibration (or sensors calibration), target detection campaign AOCS commands, RW desaturation, and Health Checks

- Sensors and IMU calibration
- Target detection campaign
- RW desaturation
- AOCS commands
- Spacecraft health checks

- Imaging of target
- ΔDOR and RARR campaigns
- Determination of relative orbit (on-ground processing)
- Calculation of propulsive manoeuvre
- Execution of propulsive manoeuvre

The braking manoeuvres will be performed once a week. Subsequent manoeuvres will be of decreasing amplitude. More detailed analysis on the implementation of the braking manoeuvres can be found in E01.

Table 6-4: Summary of approach phase

Start date	T0+39 months
Duration	2 months
Spacecraft activities	S/C health monitoring and final commissioning before braking manoeuvres Target localisation Approach manoeuvres
Distance to earth	2 – 1 AU
Distance to sun	2 – 1.7 AU
Distance to asteroid	0.02AU – 30 km
System modes	Operations mode: far range asteroid imaging and scheduled downlink periods Propulsion mode: execution of braking manoeuvres
Payload operations	Far range AFC imaging
Events	First on-board detection of the asteroid (unresolved) First resolved images of primary asteroid at the end of the approach phase (TBC)
Constraints	Still a large distance from the sun Still a large distance from earth Braking manoeuvres are critical to the mission
Propulsive manoeuvres	Braking manoeuvres (baseline: 5)
Communication	LGA: N/A HGA: uplink and downlink, moderate data rates

Ground segment activities	Reception of telemetry Sensors and IMU calibration Target detection campaign RW desaturation AOCS commands Spacecraft health checks Actuator test Orbit determination campaigns (ΔDOR, RARR and AFC images) Scheduling, commanding and execution of braking manoeuvres

The distance from the Earth will still be considerable, such that the LGA will not be useful. Earth-pointing slots will hence be necessary during the approach phase to make use of the HGA. The use of deep space ground stations will be mandatory to allow for the downlink of about 10 daily images from the AFC camera.

Note that the far range observations made during the approach may already provide useful information of the characterization of the asteroid's dynamics (e.g. removing any remaining uncertainty on the secondary's pole direction).

6.6 PROXIMITY OPERATIONS

The proximity operations phase involves performing a set of hyperbolic flyby arcs to characterise the Didymos system and operate the payload, including the two cubesats.

Various phases within the proximity operations are listed below in the order of occurrence:

- ECP: Early Characterization Phase
- DCP1: Detailed Characterization Phase
- PDP: Payload Deployment Phase (COPINs), including rehearsals
- DCP2: Detailed Characterization Phase
- DCP3: Detailed Characterization Phase (post-impact observation)

Table 6-5: Summary of proximity operations phase

Start date	T0+19months
Duration	6 months
S/C activities	Early characterization of the primary asteroid and binary dynamic state at moderate distance (~35km)

	Experiment with light curve analysis for range estimation (TBC) Detailed characterization of the primary and secondary asteroids at close distance (<10km) Payload (COPINS) deployment rehearsals Deployment of COPINS High resolution imaging during flybys (TBC) Radio Science Experiment (RSE) (TBC) Data relay for COPINS through ISL Post-impact detailed characterization of the asteroids
Distance to earth	1.0AU-0.07AU
Distance to sun	1.7AU-1.0AU
Distance to asteroid	35km (preliminary characterization) - <10km (detailed characterization-TBD (COPINs deployment)
System modes	Proximity operations mode: all nominal activities can and shall be performed in this mode (payload activities, formation maintenance, downlink)
Payload operations	Visual camera (AFC) image acquisition COPINs deployment Inter-satellite link for relaying COPINs data
Events	COPINs deployment
Constraints	Collision risk (binary asteroid, COPINs)
Propulsive manoeuvres	Orbit maintenance, asteroid flyby for payload release (TBC km, refer to E01 for detail)
Communication	LGA: uplink, low data rate HGA: uplink and downlink, moderate to high data rates
G/S activities	Reception of telemetry Sensor and camera calibration RW desaturation AOCS commands

	GNC Systems performance checks Processing of AFC images (and possibly other science data) to: • Determine the precise dynamics of the binary asteroid • Build a landmark database of the primary and secondary asteroids • Determine the relative position of the S/C wrt the asteroid Scheduling and commanding of delta-V manoeuvres for orbit maintenance (collision free trajectories) Scheduling of scientific observations and downlink periods Planning and definition of strategy for COPINS deployment (including rehearsals) Monitoring of the S/C behaviour Reception of scientific data and forwarding to the SOC

6.6.1 EARLY CHARACTERIZATION (ECP)

The early characterization phase presents the first activities in close-proximity (at about 35km relative distance). Its objective is a first assessment of the binary's orbit and dynamical behaviour.

The relative distance between the S/C and the binary asteroid can (and shall) be assessed during the approach phase, but with limited accuracy. In order to improve the range estimation, a pyramid orbit is imposed to the satellite, resulting in a motion that lies in a horizontal plane wrt the asteroids. This allows to image the target under different angles and to obtain useful measurements for the range estimation, while remaining in a safe (collision free) trajectory. The pyramid's trajectories are constructed on-ground and commanded to the satellite. Transitions from one arc to the other will be executed using dog-leg manoeuvres, such that the S/C velocity will never go under the escape velocity of the binary system, hence ensure safety of the spacecraft.

6.6.2 DETAILED CHARACTERIZATION (DCP1, DCP2 AND DCP3)

After the Early Characterization phase, the satellite will be brought to a closer distance (<10km) from the primary asteroid in order to perform a more detailed characterization and to build a landmark database to navigate the S/C. This will be the responsibility of the FDS.

A similar approach as for the pyramid orbits during ECP is chosen, but a somewhat different shape is selected in order to obtain legs of 3 or 4 days while being under influence of the higher gravitational forces of the asteroids.

Three periods of detailed characterization are envisaged to gather sufficient science data over the course of the proximity operations. DCP1 represents the first detailed characterization, with emphasis on improving the parameters obtained during the ECP, particularly the asteroid's dynamics properties and the landmark databases. DCP2 focusses on deployment and operation of COPINs cubesats to further extend the data set. Finally, DCP3 continues to perform detailed characterization, but more focused on the DART impact crater by using trajectories closer than those in DCP1. The goal there is to assess the effect of the impact on the asteroid's orbit and to investigate the crater formation.

6.7 EXTENDED OPERATIONS

Further observations may be possible while the distance to the Earth and the Sun remains within operational limits, as long as on-board resources are available. The extended operations are not defined at the moment.

6.8 END-OF-LIFE DISPOSAL

Different approaches can be envisaged for the end-of-life disposal of the S/C:

- Force the spacecraft to drift away from Didymos
- Attempt to land the spacecraft near the polar regions of Didymain or Didymoon
- Continue operations until all spacecraft resources are depleted
- Bring the spacecraft in orbit about Didymain

Alternatively, the spacecraft could be transferred to an industry, private consortium to test spacecraft and/or payload operations to support future asteroid resource utilisation activities. Nevertheless, no specific requirements or mission objectives are identified at the moment that would favour one of the proposed approaches.

7. Support Processes

This chapter elaborates on the support processes that will be required during the operations, such as flight dynamics, maintenance of databases, ground and on-board software maintenance. At this point of the project, only the flight dynamics support is described.

7.1 FLIGHT DYNAMICS SUPPORT

The role of the FDS is very mission specific and is critical for the HERA mission. It is important to re-use the experience, tools and approaches used in ROSETTA as much as possible.

Taking all the existing ROSETTA systems into account, the following 3 key aspects have been identified as most critical, as they were never implemented before in the same context. They mainly relate to the binary asteroid context.

- Image processing of binary asteroid (only done at ESOC with one body in the FoV)
- Binary system upgrade of all the existing SW
- Relative determination w.r.t. a binary system

In the following subsections, a list with major responsibilities of the flight dynamics system are highlighted.

7.1.1 ABSOLUTE ORBIT DETERMINATION AND PROPAGATION OF SPACECRAFT

Absolute orbit determination of the spacecraft is required for, amongst other things, ground antenna steering, orbit propagation and determination of large propulsive manoeuvre. The main emphasis will be during

- LEOP, to find the spacecraft, analyse the injection orbit and determine the required orbit insertion correction manoeuvres
- commissioning, to aid in calibrating the propulsion system characteristics
- cruise phase for the execution of the deep space manoeuvres (DSM)
- approach phase together with relative navigation for the execution of the different approach manoeuvres

For the absolute orbit determination, range and range-rate measurements will be required with Δ-DOR campaigns when higher accuracy is need. The processing of these measurements requires specific applications.

7.1.2 ABSOLUTE ORBIT DETERMINATION AND PROPAGATION OF ASTEROID PAIR

During the approach phase, and possibly also during some of the proximity operations phase, relative and absolute measurements (images for relative and RARR/ Δ-DOR for absolute) are used to improve the ephemerides errors of the asteroids.

7.1.3 RELATIVE SPACECRAFT-ASTEROID ORBIT DETERMINATION AND PROPAGATION

During the approach and proximity operations phases, relative navigation will assume the main role as in these phases the orbits are designed w.r.t. the asteroids. Thus, for achieving a proper guidance of the S/C, i.e. that it follows the designed trajectory, a proper navigation w.r.t. the asteroids must be ensured as well.

The relative orbit determination will be based on on-ground processing of acquired images and S/C telemetry.

7.1.4 MASS AND INERTIA PROPERTIES AND COM MONITORING

The spacecraft mass, inertia properties and CoM will change after each propulsive manoeuvre. Especially for the large manoeuvres (LEOP, cruising and approach phase), the impact will be high, but also for the smaller manoeuvres, the changes are still significant. It is important to keep track of this on ground for the calculation of upcoming manoeuvres, but also to regularly upload the properties to the spacecraft to ensure efficient attitude and orbit control. It is suggested to have an on-ground estimation of mass and inertia, because 99% of the manoeuvers will be of the orders of cm/s so the online update of these parameter is not of great advantage. Hence, GNC algorithms shall be capable of accepting update of parameters like mass, CoM and inertia communicated by ground.

7.1.5 SPACECRAFT ENVIRONMENTAL CHECK (POWER, THERMAL, ATTITUDE)

In supplement to the satellite operators, the FDS will monitor the overall health of the spacecraft. The FDS will focus on the current status and make predictions taking the upcoming manoeuvres, orbit configuration and spacecraft configuration into account. This requires accurate models of the spacecraft and its environment.

7.1.6 CONFIGURATION OF SYSTEM MODES

The different system modes require specific configurations for the different mission phases or activities. The configuration of the system modes comes out of the overall mission planning and is translated into telecommands which are put in the timeline.

A tool is required for this purpose. This function could be integrated into the overall mission planning or can be a stand-alone tool.

7.1.7 LARGE PROPULSIVE MANOEUVRE TRADE-OFF, DETERMINATION, SIMULATION

The large propulsive manoeuvres include the Deep Space Manoeuvres and the braking manoeuvres during the cruise and approach phases. From the absolute and relative orbit determination and propagation, the list of possible manoeuvres can be deduced. A trade-off need to be made to select the optimal manoeuvre taking the overall mission timeline and platform constraints into account. The selected manoeuvre needs to be simulated and eventually incorporated into the mission planning.

The applications prepared and validated for ROSETTA are baseline, with expected minor modifications.

7.1.8 SMALL PROPULSIVE MANOEUVRE TRADE-OFF, DETERMINATION, SIMULATION

The smaller propulsive manoeuvres are spread over the entire mission. They include de-tumbling, momentum offloading, pointing and small orbital manoeuvres in close proximity of the asteroids. Most of these manoeuvre are autonomously performed on-board and require no specific ground interaction. For the manoeuvres in close proximity however, detailed ground analysis and assessment is required. Similar to the large propulsive manoeuvres, they need to be simulated and planned.

The applications prepared and validated for ROSETTA are baseline, with expected minor modifications.

7.1.9 PROPULSIVE MANOEUVRE COMMAND GENERATION

The planned propulsive manoeuvres need to be translated into time-tagged telecommands and forwarded to the spacecraft operator which will load them into the on-board scheduler. The command generation shall also include a consistency check. An additional check of the uploaded commands on-board the spacecraft is equally important.

7.1.10 PROPULSIVE MANOEUVRE CORRELATION AND EVALUATION

After each propulsive manoeuvre, a dedicated evaluation is performed of the performance. This will feedback into the spacecraft models and navigation, i.e. spacecraft state estimation, which are used to determine the next propulsive manoeuvres.

7.1.11 SUPPORT GNC COMMISSIONING AND CALIBRATION

The FDS shall support the calibration of the different GNC units, both sensors and actuators. This will also include the calibration of these units which is an input to the spacecraft model.

7.1.12 TARGET MONITORING & CHARACTERISATION & MODELLING

Specific tools will need to be developed for the characterisation and modelling of the asteroids based on the acquired optical images and other data. Similar tools have been developed for ROSETTA so a high re-use is envisaged.

7.1.13 ISL SUPPORT ACTIVITIES

The FDS will support the overall mission planning in providing inputs on the “formation” characteristics of the spacecraft with respect to deliverable payload (COPINs). The input will mainly consists of antenna angles, distances and possible obstruction times by the asteroids which will translate into optimal positions for these bodies.

7.1.14 COLLISION RISK ASSESSMENT

By design, all propulsive manoeuvres will be designed to avoid risk of collision (collision-free trajectories). A dedicated function should nevertheless monitor the collision risk of the spacecraft with the asteroids and any other known objects (e.g. ejecta). The function shall provide an assessment on the short and long term.