Design Definition and Justification Report

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1 INTRODUCTION

This document was prepared within the phase B1 study of the HERA mission led by OHB System AG. It describes the current status of the HERA design and provides relevant justification.

The description is limited to the trade-offs performed within phase B1 and the resulting deviations from the AIM baseline. Please refer to [RD01] for an overview on the AIM trade-offs.

The document is structured as follows:

- Chapter 3 provides the requirements discussion
- Chapter 4 gives the mission overview
- Chapter 5 introduces an overview on the spacecraft
- Chapter 6 details the analyses and trade-off performed on the different architecture levels
- Chapter 7 concludes the document

2 REFERENCES

2.1 Applicable Documents

This document shall be read in conjunction with documents listed hereafter, which form part of this document to the extent specified herein. In case of a conflict between any provisions of this document and the provisions of the documents listed hereafter, the content of the contractually higher document shall be considered as superseding.

Table 2-1: Applicable Documents

AD	Doc. No.	Issue	Title
[AD01]	ESA-TECSP-RS-009704	1.0	HERA Mission Requirements Document (MRD)
[AD02]	OPF-GFA-TN-119	1.0	Mission Analysis Guidelines (MAG)
[AD03]	ESA-TECSP-RS-009703	2.0	HERA System Requirements Document (SRD)
[AD04]	-	5.2	Didymos Reference Model
[AD05]	-	1.0	HERA Cost Estimate template
[AD06]	-	1.0	HERA Product Assurance Requirements
[AD07]	DOPS-ESTR-OPS-MAN-1001-OPS-ONN	2.0	DOPS-ESTR-OPS-MAN-1001-OPS-ONN_EFM
[AD08]	-	-	Soyuz-Users-Manual-March-2012
[AD09]	ESA-TECSP-RS-009695	1.0	HERA Margin Philosophy
[AD10]	ESA-AMCO-RS-0008	1.1	HERA Operations Implementation Requirements Document (OIRD)
[AD11]		1.0	HERA Payload Operations Document (POD)
[AD12]	ESA-TEC-SOW-009533	1.0	ESA-TEC-SOW-009533

It should be noted that all requirements listed in the documents of Table 2-1 are applicable unless noted otherwise or exceptions are identified and agreed.

2.2 Reference Documents

The following documents contain additional information that is relevant to the scope of this document.

Table 2-2: Reference Documents

RD	Doc. No.	Issue	Title
[RD01]	HERA-OHB-SYS-RP-0003	0.1 Draft	Delta Assessment HERA/AIM
[RD02]	AIM-OHB-SYS-DDJ-03	1.1	Propulsion DDJR
[RD03]	AIM-OHB-MAN-RP-01	1.0	AIM Final Report
[RD04]	-	3.6	AIM Payload Operations Document
[RD05]	-	-	Micheletal_Hera_CorrectedProof_ASR2017
[RD06]	ISIS.LVPOD.TN.006	0.1	LV_POD ICD of CRS
[RD07]	ISIS.LVPOD.TN.005	1.0	LV-POD Payload ICD_CubeSat Deployer ISIPOD
[RD08]	ISIS.LVPOD.TN.008	1.1	LVPOD Executive Summary Report
[RD09]	-	1.1	AIM HGA Preliminary Sizing v1.1
[RD10]	-	-	Arianespace launch analysis
[RD11]	-	0.1	AIM ISL Requirements Specification v0.1 (draft)
[RD12]	-	-	Ariane 6 User's Manual v0.0 (May 2016)
[RD13]	ESA-TECSAG-TN-011315	1.0	HERA proximity operations v1.0
[RD14]	-	-	B. D. Lucas and T. Kanade. An Iterative Image Registration Technique with an Application to Stereo Vision. International Joint Conference on Artificial Intelligence, pages 674-679
[RD15]	-	-	C. Tomasi and T. Kanade. Detection and Tracking of Point Features. Carnegie Mellon University Technical Report CMU-CS-91-132.
[RD16]	-	-	J. Shi and C. Tomasi, Good Features to Track. In Proceedings of the IEEE conference on computer vision and pattern recognition (CVPR'94), pp. 593–600.

RD	Doc. No.	Issue	Title
[RD17]	RO-ESC-TN-5562		Navigation Analysis for comet operations up to SSP delivery
[RD18]	-	-	Asteroid Impact Mission: Mission Analysis Guidelines
[RD19]		1.0	AIM Proximity Operations Trajectories
[RD20]	HERA-OHB-SYS-RP-0001	0.2	System Budgets Document
[RD21]	HERA-OHB-SYS-TN-0003	0.1	Spacecraft Configuration Alternatives
[RD22]	HERA-GMV-GNC-DDJ-0001	0.2	GNC Design Definition Justification File
[RD23]	HERA-OHB-SYS-TN-0002	1.0	LGA based survival mode
[RD24]	HERA-GMV-MIS-RP-0001	1.0	Mission Analysis Report
[RD25]	HERA-OHB-SYS-RP-0004	1.0	System Autonomy and FDIR Concept
[RD26]	HERA-OHB-MAN-LI-0001	1.0	Hera Acronyms and abbreviations

2.3 Abbreviations & Nomenclature

For all terms, definitions and conventions used, if available. A complete list of the current abbreviations and acronyms used in the Hera Project can be found in [RD26].

3 REQUIREMENTS DISCUSSION

3.1 Discussion of Main Drivers

This chapter discusses the main driving requirements as included in the SRD 2.0. In some cases these requirements have been mutually agreed for change. This is reflected in the discussion.

The discussion of each individual requirement is part of the Delta HERA/AIM document and can be found in [RD01]. Also repetitive to that document a short statement on the AIM requirement is given for convenience.

3.1.1 Baseline and back-up launcher

Req ID	Req. Text	AIM Baseline
SY-ARC-40	**Baseline Launcher** The Hera Spacecraft shall be compatible with a Soyuz 2-1b Fregat launcher to be launched from the Baikonur or Vostochny (TBC) spaceport.	Soyuz Kourou
SY-ARC-50	**Backup Launcher** The Hera Spacecraft should be compatible with an Ariane 6 launcher to be launched from the Kourou spaceport.	Falcon 9
SC-ENV-40	**Mechanical Environment for Backup Launcher** The Hera Spacecraft foresee the design, verification and test activities as required by the launch service provider in accordance to [RD10] for the Ariane 6 backup launch.	

The Hera spacecraft shall be launched preferable with a Soyuz Fregat 2-1b launcher, as the AIM spacecraft. However, the launch site will be different, which yields some impacts on the resulting mechanical requirements and the launch environment. The back-up launcher is the A6 instead of the previously envisaged falcon. Comparing all, the Hera spacecraft needs to withstand a harsher launch environment compared to AIM. The differences are summarized in the following:

- Slightly more severe sine loads, which are even exceeded by Ariane 6.2 (no detailed information about Ariane 6.2 launcher performance available for a satellite below 2t)
- Slightly higher shocks
- Higher but also more detailed clamp band release loads
- QSL, acoustic and random vibration loads similar between Soyuz from Kourou and Baikonur and better for Ariane 6
- Horizontal transportation to launch site represents new QSL load case

Reviewing the to be considered loads and the AIM design, the launcher selection will not cause spacecraft level design modifications, but might impact unit level decisions. Either the design loads might need to be adapted or the units might be moved to reduce e.g. the propagation of shock and sine loads.

During the HERA phase B1 the option of an alternative launcher was discussed and considered programmatically infeasible. These requirements are driving, but are understood and considered mandatory.

3.1.2 Launch and Transfer

Req ID	Req. Text	AIM Baseline
SY-PHA-10	**Launch Window** The Hera Spacecraft shall be designed to be compatible with the launch window starting on 8 October 2023 up to 28 October 2023 (referred to as EEMA2023 in [AD02]).	17.10.–6.11. 2020 (reduced launch window)
SY-PHA-20	**Backup Launch Windows** The Hera Spacecraft shall be designed to be compatible with at least two more backup launch windows, a main one in 2024 and a secondary not later than 2027.	-
SC-PRO-60	**Transfer to Asteroid System Delta-V** The Hera Spacecraft propulsion sub-system shall be able to accommodate the Delta-V necessary for the interplanetary transfer for the baseline and backup options according to [AD02], not higher than 1250 m/s without margins and not including asteroid proximity operations.	1285 m/s incl. margins and launcher dispersion correction
SC-PRO-70	**Proximity Operations Delta-V** The Hera Spacecraft propulsion sub-system shall be able to perform all planned manoeuvre to meet the mission objectives considering an additional margin of 20 % on top the HERA margin philosophy.	69 m/s incl. margin and contingency
SC-GEN-90	**Total Mass at Launch** The Hera Spacecraft total wet mass (MWET), including all applicable margins, shall not exceed 700 kg (TBC) including launcher adapter. Note: [AD02] suggests that, for the current baseline, this figure might be lower considering additional margins due to the early stage of analysis.	770 kg excl. launch adapter

The baseline launch window is driving for the mission in several aspects. On the one hand, the development approach and schedule are tightly constrained by a launch end of 2023, especially considering a final approval of this mission not before the conference at ministerial level in autumn 2019. The development schedule for phase B2/C/D has to fit within this about four years. The use of standard/qualified solutions is seen as mandatory to meet the schedule, combined with project management approaches based on co-engineering and co-location between customer and industry.

As it is discussed in chapter 6.2.1 in detail it was agreed to take the Mars Swing-by transfers and baseline and first back-up. A suitable back-up was found in 2027 although yielding a much larger mission duration and a considerable ground storage time.

- EEMA2023, EMA2024
  - Reduced deltaV demand of less than 600 m/s
  - Medium transfer durations (2023: ~1200 d, 2024: ~800 d)
  - Maximum distances between Earth and asteroid during proximity operations (up to 2.8 AU)
  - Theoretical wet mass 650 kg according to MAG

More details on the transfers, like the manoeuvre timeline, deltaV demand but also the propellant demand, expected wet mass and resulting dry-mass at launch are given in chapter 6.2.1.

The transfers impact severely the overall mission design and feasibility and also imply modifications to the AIM design, due to the propellant demand resulting from the deltaV requirement of 1250 m/s but also the distance to Earth during the proximity operations implying the need for a more power full communications system. After agreeing to the baseline and back-up trajectories the relevant deltaV for those transfers are considered the new requirement for tank capacity. It needs to be noted, that for the 2023/2024 launches the tanks will only be partially filled with propellant and that the 2027 back-up requires a much bigger filling ratio. Both is compatible to the new tank. The updated requirement should take this into account.

The wet mass expected at launch considerably varies depending on the selected trajectories. A target of 700 kg is unlikely to be met. This si further elaborated in chapter 6.2.1. It seems also rather conservative with respect to the expected launcher performances for the different launch cases.

Nevertheless mass has to be closely monitored as the secondary back-up trajectory presently does not provide a positive launcher margin and the 2023/2024 yield less than 8 % additional launcher margin.

Mission Geometry

Req ID	Req. Text	AIM Baseline
SC-GEN-40	**Safe Spacecraft Design** The Hera Spacecraft shall be designed to ensure the spacecraft remains energy positive in all nominal operational phases and modes, even under failure conditions.	-
SC-GEN-41	**Safe Spacecraft Design (II)** The Hera Spacecraft shall be designed to ensure the spacecraft remains thermally safe in all nominal operational phases and modes, even under failure conditions.	
SC-GEN-110	**Data Volume** The Hera Spacecraft shall support a minimum data return volume of 100 Gbits of scientific data since the arrival at the asteroid system until end of the mission. Note: For details on the assumptions behind this figure, please refer to [RD04].	25 Gbit science data
OP-MOD-10	**Spacecraft Safe Mode Definition** The Hera Spacecraft Safe Mode shall be designed such that the platform and payloads can remain stable until depletion of consumables, able to send at least essential telemetry to ground and waiting ready to receive commands from ground.	No major delta to AIM

The expected distance to the Sun during important mission phases itself is not considered to be a major driver for the system compared to AIM. During the six months proximity phase a high amount of data needs to be downlinked. Compared to AIM, the link budget for this case is severely impacted for two reasons. The data volume to be downlinked was increased by a factor of four to 100 Gbits and the distance to Earth was increased from the order of 0.1 AU to up to 2.8 AU, depending on the transfer scenario, respectively the corresponding arrival time at the asteroid.

It was assessed that meeting this data volume within the 6 months of proximity operations would require one of the following options (not exclusive)

- 70 W TWTA, 1m HGA
- 35 W TWTA, 1.4 m HGA

This will be further discussed and traded in chapter 6.3.2.

3.1.3 GNC

The most challenging requirements from GNC point of view resulting from SRD and MRD are the following:

- SC-GNC-230: 300m 1mm/s (each axis, 99.73% confidence level) during nominal close operations
- SC-GNC-240: 30m, 0.2mm/s (each axis, 99.73% confidence level) at altitudes below 5km

On the other hand the MRD [AD01] states the following requirements in terms of position knowledge error, already mapped onto the proposed GNC modes:

- T2.1, applicable to ODCS-MODE-1: Relative Position Knowledge Error: 100m (95% probability, 90% confidence)
- T2.2, applicable to ODCS-MODE-2: Relative Position Knowledge Error: 10% of distance to target asteroid (95% probability, 90% confidence)
- T2.3, applicable to ODCS-MODE-3: Relative Position Knowledge Error: 10% of distance to target asteroid (99.7% probability, 90% confidence)

The requirements stated by the MRD are derived from the scientific objectives of the mission and well justified. It is suggested to follow this approach with the requirements derived for each of the proposed GNC modes.

The results of AIM-CP show that the performances requested in MRD are challenging (especially T2.1), but within the capabilities of the proposed technologies, particularly when possibility of use of the altimeter at lower altitudes is considered.

Req. ID	Req. Text	AIM Baseline
SY-PHA-80	**Spacecraft Disposal** The Hera Spacecraft should be disposed at the end of the mission operational lifetime by means of a descent towards the surface of Didymain.	This option implies another GNC landing mode that was not present in AIM.
PL-PAL-20	**PALT as Navigation Altimeter** It should be possible to use the PALT payload as a navigation altimeter. Note: The goal is to enable using PALT as complementary sensor to trigger collision avoidance manoeuvres or test navigation strategies during proximity operations.	To include PALT in the GNC sensor suite is a different approach from the AIM-CP.

Req. ID	Req. Text	AIM Baseline
SC-GNC-20	**GNC Function** The Hera Spacecraft GNC sub-system shall provide all on-board hardware and software to measure and control the spacecraft attitude throughout all mission phases.	It is understood that the same AIM baseline is compliant to the HERA concept
SC-GNC-30	**GNC Fail-Safe Operation** The Hera Spacecraft GNC sub-system operation shall be fail safe.	
SC-GNC-80	**Excessive Rate Protection** The Hera Spacecraft GNC sub-system shall ensure the spacecraft safety by preventing excessive angular rates throughout all mission phases.	Delta with respect to AIM. To clarify the word excessive. Eventually to replace the word excessive with a hard constraint.
SC-GNC-230	**Spacecraft Nominal Relative State Knowledge** During Proximity Operations he Hera Spacecraft position and velocity relative to the Didymos System shall be known on-board to an accuracy of 300m (TBC) and 5mm/s (goal: 1 mm/s), with a 99.73% confidence level, in every axis (each axis independent of the rest).	The requirement may be too optimistic according to the results of AIM–CP, but thorough analysis will be performed to try to achieve this performance.
SC-GNC-240	**Spacecraft Relative State Knowledge during Critical Close Operations** During critical close operations below 5km (TBC), knowledge error of the relative distance to the COM of the target body shall be lower than 10% of the actual distance with 99.7% probability at 90% confidence level.	The performance on the relative velocity determination achieved in AIM would not satisfy the requirement. To be further investigated during HERA.
SY-GEN-30	**Passively Safe Trajectories during Proximity Operations** During proximity operations, the Hera Spacecraft shall be in passively safe trajectories guarantying no risk of collision with the asteroids and safe conditions for the platform and payloads for 1 month. Note: Passively safe means without thruster actuation and considering all worst case forces acting on the spacecraft. For further background information, please refer to [RD19].	No delta to AIM

Req. ID	Req. Text	AIM Baseline
T2.1	**Autonomous visual based navigation for semi-autonomous attitude guidance** Position knowledge error (any axis) lower than 100 m relative to the centre-of-mass of Didymos with 95% probability, 90% confidence level. Contribution of attitude guidance to APE lower than 0.5 deg (95% probability, 90% confidence level). NOTE: This includes the autonomous navigation error.	The requirement may be too optimistic but thorough analysis will be performed to try to achieve this performance.
T2.4	**Sensor data-fusion for robust navigation** At least the same as the equivalent vision-based navigation solution.	Enlarged sensor suite wrt AIM (PALT and TIRA)

The HERA mission includes highly autonomous phases, with large use of data-fusion techniques, independent validation of the nominal GNC solution and close trajectories up to landing on Didymain for the SC disposal. It is important to remark that, even if these updates were not part of the AIM baseline, GMV has been working on these technologies several years and it has the in-house experience to achieve the maturity level required at the end of the HERA phase B1. An extensive survey, design and implementation of data-fusion techniques has been also addressed in the DAFUS activity, fully applicable to the AIM scenario, which will be used as a starting point for the HERA phase B1.

As baseline for AIM, GMV understands that it shall be possible to estimate on-board and control the inertial attitude during all the mission phases (degraded 2 axis stabilization during survival mode). However, this does not apply to the SC orbit which shall be estimated on-board only during the close proximity operations.

The requirement SY-GEN-30 limits the possible trajectories to be used during proximity operations.

3.1.4 Avionics

Req ID	Req. Text	AIM Baseline
SC-DHS-120	**On-Board Data Storage Size** The Hera Spacecraft data handling data storage shall be sized to be compatible with the Ground Segment coverage and the mission's operational scenario throughout all mission phases considering worst case scenario of housekeeping data generation and payload product data acquisition with margins.	HERA payload product data volumes highly unknown
SC-DHS-130	**On-Board Payload Data Safety** The Hera Spacecraft data handling sub-system shall be designed to acquire and store concurrently the data from all payloads.	

The above requirements are considered a major design drivers for three reasons:

- The current Payload Operations Documents [AD11]**Fehler! Verweisquelle konnte nicht gefunden werden.** & [RD04] do not provide many details on the expected payload data volume for the HERA payload set, while this was much clearer for AIM. Detailed analysis and dedicated discussions with the lead scientists and payload providers as early as possible in the study are required to determine the science requirements and derive from that the payload operations, resulting in the expected data rates and data volumes. Especially for the hyperspectral payload (CHITY), which typically provide massive amounts of data, a detailed and careful assessment needs to be performed to ensure that it fits both the science requirements and the spacecraft constraints. More recent information indicate up to 22 Gbits/s data generations. This would be a serious driver and need further discussion.

- The new PROBA-Next avionics do not include a payload mass memory (non-volatile). There are several options to include this based on heritage solutions but a good view on the required storage capacity is required to trade-off the different solutions. In addition, it shall be discussed with ESOC whether a file-based storage and download can be de-scoped from HERA as it was agreed for AIM. File based storage and download has an important impact on the changes required to the hardware and software.
- Some payloads have really high peak data rates, which drive the data handling architecture (CHITY). This requirement enforces to take more complex data handling solutions to follow the payload parameter. It is understood, that the idea of HERA is to find a good compromise between all mission elements and avoid shifting complexity to one element only.

3.1.5 Operations

Req ID	Req. Text	AIM Baseline
SY-MOD-10	**Safe Mode** The Hera Spacecraft shall support a Safe Mode that ensures two way communications for system diagnosis, and start or recovery of system nominal operations. *Note: Ensuring two way communications is to be understood providing that Ground Station coverage is available.*	In line with new requirement
SY-MOD-20	**Survival Mode** The Hera Spacecraft shall support a Survival Mode that, in case of contingencies that prevent nominal operations and communications, ensures the spacecraft can autonomously achieve the conditions to establish two way communications.	

Where for AIM, a safe mode was required which was sun-pointing, the HERA requirement (SY-MOD-10) provides more flexibility in the definition of the mode itself. The SAFE mode as defined for AIM remains valid for HERA. Also a different relative spacecraft-Earth-Sun position during the transfer will have a significant impact. The above might lead to an altering of the on-board communications architecture and a change in the safe mode configuration, possibly requiring different pointing scenarios in different mission phases.

The above is equally important for the definition of the survival mode as covered by SY-MOD-

Req ID	Req. Text	AIM Baseline
SY-MOD-40	**Collision Avoidance Mode** The Hera Spacecraft shall support a Collision Avoidance Mode that, in case of risk of collision with any of the asteroids in the Didymos system, ensures the spacecraft can autonomously perform a Collision Avoidance Manoeuvre (CAM) to achieve safe conditions.	Only optional semi-autonomous CAM mode foreseen

Where in AIM, an optional CAM mode was provided in the close fly-by mode, for HERA this is more explicit in the requirements (e.g. SY-MOD-40). Important in this matter is that the altimeter can be used as an additional sensor, compared to the more limited set of sensors available on AIM. The addition of the altimeter should provide more capabilities to detect a risk of collision and perform an appropriate CAM.

Req ID	Req. Text	AIM Baseline
OP-GEN-10	**Operations Implementation Requirements** The Hera Spacecraft operations shall be compliant with the requirements established in [AD10] unless otherwise specified in this document.	OIRD replaced by new, standard OIRD deviating from AIM.

The SRD also includes several requirement enforcing the existence of specific modes as well as their functionalities and properties. Driving requirements in this context are listed below.

Req ID	Req. Text	AIM Baseline
OP-AUT-60	**Autonomy for Collision Avoidance** The Hera Spacecraft, when in Collision Avoidance Mode, shall be able to determine the collision risk and compute and execute a manoeuver that prevents collision	No major delta to AIM
OP-AUT-120	**Autonomy Period** The Hera Spacecraft shall autonomously ensure safety of the spacecraft for the duration of the longest solar conjunction + 10 days (TBC) without ground intervention. Note: It is assumed here that no critical operations will be performed during conjunctions	72 h

These two requirements result in on-board autonomy and necessary awareness of spacecraft environment. This needs to be computed on–board. This requires the necessary sensing and processing capabilities. A sufficient memory is required to store time tagged command for 15 days autonomous operations including possible contingencies.

3.2 Payload

Req ID	Req. Text	AIM Baseline
SY-PAY-10	**Payload Accommodation** The Hera Spacecraft platform shall accommodate the following payloads: • 2x Asteroid Framing Camera (AFC) • Planetary Altimeter (PALT) • Thermal Infrared Instrument (TIRA) • 2 x 6U Hera Cubesat including their deployment mechanism	Different payload

Req ID	Req. Text	AIM Baseline
SC-GEN-110	**Data Volume** The Hera Spacecraft shall support a minimum data return volume of 100 Gbits of scientific data since the arrival at the asteroid system until end of the mission. Note: For details on the assumptions behind this figure, please refer to [RD04].	25 Gbit science data
SC-DHS-130	**On-Board Payload Data Safety** The Hera Spacecraft data handling sub-system shall be designed to acquire and store concurrently the data from all payloads and guarantee.	

The payload is so far not sufficiently defined to derive all potential drivers.

The mass and power demands could be accommodated. However, considering the mass criticality of the spacecraft, effort for mass reduction is required, especially for APEX (14+kg demand against 12kg allocation).

The data return volume of 100GB is driving the communication architecture as well as the operation of the spacecraft, which leads to implementing a 70W TWTA and dedicated downlink operation (up to 10 months in case of 70W TWTA failure) after the science operation.

The discussion of payload drivers within this document will be extended afterwards.

4 MISSION OVERVIEW

4.1 Mission Architecture

The Hera mission architecture to achieve the mission objectives is depicted in Figure 4-1. The mission is broken down in four segments, including the launch-, space-, ground- and target asteroid- segments. The interaction with the DART mission is included for completeness, although this is not part of the baseline mission. No direct interface will occur between HERA and DART.

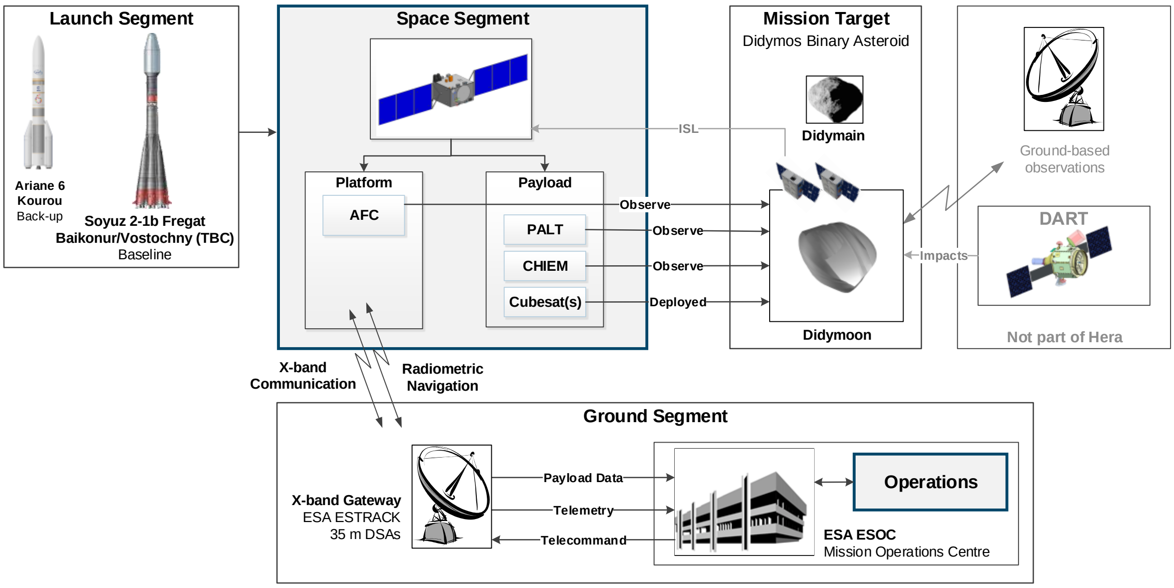


Figure 4-1: HERA Baseline Mission Architecture

The primary elements of the architecture are:

- **Space Segment**, the main subject of this activity, which comprises of the:
  - Payload, which is composed of several individual instruments that are to be integrated onto the spacecraft. The payloads fulfil the asteroid science, resource utilisation and planetary defence objectives by performing experiments and measurements. The spacecraft will also host two 6U CubeSats.
  - Platform, which is the spacecraft bus and provides all the functions to successfully perform the mission. The AFC and PALT are used for navigation measurements, and also acts as a science payload. The spacecraft maintains an inter-satellite link to the deployed CubeSats, providing a deep space network.
- **Launch segment**
  - Soyuz 2-1b Fregat, launched from either Baikonur or Vostochny, as the baseline. The back-up launcher is an Ariane 6 from Kourou. The launcher provides direct injection into the escape trajectory.
- **Ground Segment**, provided by the Agency, with definition supported by industry:
  - 35 m Deep Space Antennas (DSA), providing X-band space-to-ground interface for nominal operations. Larger antennas, e.g. Goldstone are assumed in case of spacecraft emergency situations.

  - Mission Operations Centre (MOC), which will be responsible for the all the mission planning, and the control of the spacecraft. No dedicated Science Operations Centre (SOC) is foreseen for HERA, yet.
- **Mission Target**, the binary asteroid system 65803 Didymos (1996 GT). Didymos consists of two components: a primary body informally referred to as Didymain, and a secondary body that is informally called Didymoon. Both definitions are used throughout this document. Didymoon essentially orbits Didymain due to its large mass, but strictly speaking the complete system exhibits a barycentre motion. This enables high-priority science measurements (i.e. mass-determination through wobble measures) to be performed. Didymoon is the primary target for Hera, and will also be impacted by the DART spacecraft few years prior to Hera arrival.

4.2 Mission Phases

This chapter provides an introduction of the operations concept for the Hera mission. Figure 4-2 provides an overview of the Hera timeline, which can be subdivided into the following distinct phases:

- **Launch**: baselined on a Soyuz 2.1B Fregat from either Baikonur or Vostochny.
- **LEOP Phase**: within the launch and ascent phase, the spacecraft will be switched-off. After an initial coast phase in an intermediate parking orbit, the Fregat upper stage injects the spacecraft into its transfer trajectory and separates it. The spacecraft will autonomously detect the separation and initiate a switch-on and power-up sequence. At its end the spacecraft enters safe mode. It:
  - Is in a detumbled state
  - has acquired a 3-axis stabilised sun-pointing attitude
  - has deployed its solar arrays
  - has initialled a two way contact with MOC
  - has performed the health check of essential units.
- **Commissioning Phase**: the spacecraft performs in-orbit verification and functional check-outs of the platform’s equipment to demonstrate full readiness for nominal operations of the platform, as well as for the MOC. The commissioning of the payloads will be started as early as possible after platform commissioning and in line with the payloads needs (including the CubeSats and initial ISL functions). The payload commission will also support the verification of the interfaces between the payload operations centres and the spacecraft
- **Transfer Phase**: baselined transfer is of type EEMA2023 and thus includes swing-bys at Earth and Mars and one deep space manoeuvres prior to the arrival at the Didymos system. In case of the 2023 transfer the spacecraft will arrive at Didymos in early 2027.
- **Asteroid System Rendezvous**: insertion into the asteroid system with a braking manoeuvre, supported by orbit determination from ground and optical navigation.
- **Proximity Operations**: with the detailed characterisation of Didymoon (including the DART impact crater), the release and operations of the CubeSats, and technology demonstration activities (GNC, autonomy etc). Nominal operation can be broken down into four separate stages. Optional science and impact observations activities are dependent on the available resources and timeline (not currently baselined). Nominal operations include:
  - Detailed Characterisation Phase 1
  - Payload Deployment Phase

    - Detailed Characterisation Phase 2
    - Detailed Characterisation Phase 3

- **Mission Extension** presently three reasons for extension of the proximity phase are envisaged
    - Finalisation of science data downlink due to a bottleneck in data rate due to the large distance between Hera and Earth in case of the 2023/2024 missions proximity operations
    - (in case of sufficient health status, remaining consumable on-board the spacecraft and available funding) Extended science programm, e.g. fly-through in between Didymos and Didymain
- **End of Life Phase (ELP)**: either an descent towards the surface of Didymain or transferred to private operators.

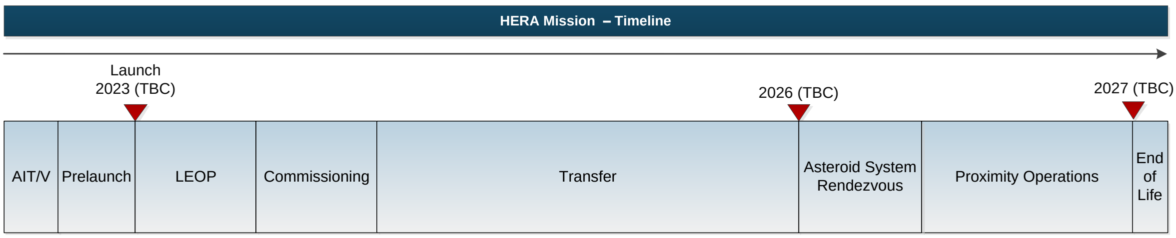


Figure 4-2: High Level Mission Timeline. Durations Not to Scale

4.2.1 Launch

The spacecraft baseline scenario foresees a launch in 2023 (primary backup in 2024) on a Soyuz Fregat from the Baikonur (or Vostochny, TBC). The spacecraft will be off at launch. The alternative back-up launcher is an Ariane 6.2 from Kourou. The launcher, launch date and timeline is a mission level requirement that drives the mission architecture and design. The launcher drives the maximal available launch mass, useable volume and the possible escape velocity for an interplanetary transfer. It should be noted, using the Fregat upper stage for injection into an interplanetary trajectory is still relatively unusual within the aerospace industry. The different launch locations will also impact on the on-ground transportation, mating conditions, mechanical requirements and environmental loads. A launch from Baikonur on the Soyuz Fregat is slightly more severe (i.e. sine loads, shock), although not immediately significant, than compared to a launch from Kourou on an Ariane 6.2. The characteristics of the launcher will be evaluated during the phase B1 study, and the comparable worst-case conditions (i.e. sine loads, shock, clamp band release loads, quasi-static loads, acoustic loads and vibration loads) will be used throughout.

4.2.2 LEOP Phase

Following launch, the spacecraft will be inserted into a circular 185 km intermediate orbit, inclined at 51.6 deg [AD02]. After an initial coast phase, the Fregat upper stage will perform a second burn, injecting the spacecraft into its interplanetary transfer. After separation, the spacecraft performs the following functions automatically:

- Spacecraft separation from the upper stage triggers the PCDU start and check-up, which ensures the correct operation of the several systems needed during LEOP. The spacecraft then establishes a two-way communication link with the ground segment through the omnidirectional low gain antenna (LGA).

- The spacecraft's angular momentum is automatically damped until the residual angular momentum is low enough to safely perform Sun acquisition. This activity uses the spacecraft RCS thrusters and rate sensors.
- Once the angular velocities are nulled, the slew manoeuvre to acquire safe power supply can begin. This manoeuvre will point the spacecraft's +X panel towards the Sun to recharge the batteries through the solar arrays once these are deployed.
- The solar array deployment will occur after Sun acquisition. This reduces the effort on the actuators as well as the time it takes to slew the spacecraft towards the Sun.
- A preliminary health check of all essential units will be performed
- Finally, upon establishing a safe power supply and ground communication, the spacecraft will wait for a command from the ground segment to enter the commissioning phase, which will perform the necessary operations to start and check-up of the star tracker. Once the star tracker is functional and the AOCS fully defines the spacecraft's attitude with respect to the inertia frame, the reaction wheel attitude control hand-over begins. This procedure is closely monitored by the ground segment to ensure the spacecraft's safety.

4.2.3 Commissioning Phase

The commissioning phase includes a number of activities to perform the in-orbit verification and functional check-out of the platform's equipment, the payload and the ground station. Please note in case of e.g. the AFC an early full commissioning is not desired. It is preferred to give the spacecraft time for outgassing prior to first exposure of the optics.

The commissioning activities would include, but are not limited to:

- Verification of the system's platforms
  - Verification of the X-band communication, including telecommunication, transmission and/or interface to the ground stations
  - Verification of transmitted data
  - Activation and checking of platform subsystems (thermal, electrical etc.)
  - Test of telemetry data store for platform TM and instrument data
  - Verification of system modes
- Verification of the payload/instruments
  - Verification of instrument commandability and measurement data transfer to platform
  - Functional tests of instrument operating modes
- Payload data storage and transmission
  - Verification of the ground segment
  - Verification of link margins for communication and of tracking/ranging capabilities
  - Verification of signal acquisition and antenna tracking performance
  - Check of link margins in transmission for ground stations
  - Test of on-ground data retrieval and storage
  - Verification the functionalities of stations, including the generation of source packet annotation data
  - Verification of contents of generated telemetry data files (validity of source packet contents, including annotated error quantifiers)
  - Dissemination of telemetry data files

4.2.4 Transfer Phase

Hera is baseline for a launch in 2023. The trajectory includes one Earth and one Mars swing-by before the rendezvous with the Didymos system. This transfer takes approximately 3.3 years and requires one deep space manoeuvres in between the swing-bys.

About two weeks after launch a correction manoeuvre is performed to compensate for any launcher dispersion errors and constraints on the injection elevation. The deep space manoeuvre performs target correction, after the Earth swing-by. The spacecraft arrives at Didymos in early 2027 after 3.3 years of transfer. The backup trajectory in 2024 excludes the Earth-swing-by, and so only includes one deep space manoeuvre and the Mars Swing-by. The spacecraft would arrive at Didymos always the same day.

During the transfer phase, the spacecraft is in an operation mode. The spacecraft maintains an attitude that allows sufficient power generation and communication to Earth. The solar generator faces Sun up to an accuracy of 5° and provides sufficient electrical power. The thermal control system ensures the temperature range of all equipment in the changing thermal environment during the transfer. Daily communication windows are planned making use of HGA after leaving the sphere or omni-directional LGA coverage capability or in case of high data needs. A daily downlink is typical for a deep space mission, and is assumed to be between four and eight hours each (mission phase dependent). It is desirable to perform payload- and system- checkouts, as well as for software updates of the spacecraft and CubeSats. In case of in-flight opportunities, science measurements should be taken i.e. during Mars swing-by or close flybys at other asteroids, as long as this can be performed with the available resources in terms of power available at the relevant point of the trajectory. The transfer phase ends once the spacecraft is in close proximity to, and inserted into the Didymos system.

4.2.5 Asteroid System Rendezvous

The interplanetary transfer concludes with the asteroid system rendezvous. Here, during the last few months of the transfer phase, the asteroid system will likely become visible against the background stars imaged by the AFC instrument. This allows for significant orbit refinement and determination, especially relative to the asteroid system. A large arrival braking manoeuvre is needed to adjust the spacecraft's orbital elements to those of Didymos. The braking manoeuvre is therefore spilt into multiple burns with several days between each of them. The spacecraft arrives at the first observation station (Early Characterisation Phase 1) at approximately 30 km) from the asteroid. The first size, shape and dynamical characterisation – including orbital and rotational periods - of Didymos will be performed by remote sensing imagery. This phase is called Early Characterisation Phase (ECP) and marks the start of the science activities.

The first of a set of rendezvous manoeuvres is performed after the Didymos detection. At this stage, the target's brightness is very faint, so a large integration time for the camera is used in order to identify the target. This already allows to introduce a correction component (due to the Didymos' ephemeris refinement) in the first braking manoeuvre. The manoeuvres are designed with safety and navigation requirements in mind – being able to recover the spacecraft if one of the manoeuvers is missed; perform an arc like approximation so the optical measurements provide information in all directions; and; gradually introduce a correction components to account for the ephemeris refinement.

Proximity Operations

The high uncertainties in the Didymos system make navigation with respect to the target difficult. Accordingly, a set of hyperbolic flyby arcs are designed (all passively safe). They first occur at a far distance (approximately 30 km, Early Characterisation Phase 1) from the Didymos system, with a high margin of safety above the escape velocity. Shape and the gravity
4.2.6 field is first determined. Then, upon refinement of the dynamical system, the distance between the asteroid and Didymos is reduced. The spacecraft performs a number of different Detailed Characterisations Phases, each lasts approximately six weeks (which still include a safety margin above the escape velocity). Each flyby arc enables the spacecraft to access different viewing angles with varying latitudes and longitudes from different local times. Detailed operational ground and on-board strategies are required in order to achieve a feasible and realistic navigation strategy.

The proximity operations include the following phases:

- **Detailed Characterisation Phase 1** (DCP1) with close proximity operations at approximately 10 km from the system's barycentre. The AFC camera provides accurate characterisation of Didymoon mass (via the wobble period of Didymain), which is combined with density and size measurements, medium resolution imaging and operations of the Planetary Altimeter (PALT).
- **Payload Deployment Phase** (PDP) with the release and commissioning of the 6U CubeSats (APEX and Juventas), supported by the spacecraft.
- **Detailed Characterisation Phase 2** (DCP 2), with the same operations for DCP1, and the operations of the CubeSats. The spacecraft not only supports the deployment of the CubeSats, but also maintains an inter-satellite link with a TM/TC interface between the ground stations and CubeSats. The inter-satellite creates a deep-space local network, ensuring communication between Hera and the CubeSats. Hera will support the operations of the CubeSats for a maximum duration of TBD months. The Cubesat mission ends with their forced landing (or bouncing) on Didymoon. The end-of-life event, and subsequent landing locations, will be observed by the AFC and Thermal Imaging instruments on-board the Hera spacecraft.
- **Detailed Characterisation phase 3** (DCP 3), with the detailed characterisation of the DART impact event. The spacecraft will progressively approach Didymoon, reducing the spacecraft-to-asteroid distance to below 10 km (with full payload operations). A number of close flyby trajectory (6 km distance) will enable high resolution images of the DART crater site. The phase concludes with a high risk autonomy technology, based on on-board landmark navigation.

4.2.7 Optional phases that include additional science activities and a possible DART impact phase may be included. Implementation depends on the available resources and timeline/schedule. It shall therefore not drive the baseline mission design.

End of Life

Two possible end-of-life scenarios exist. The spacecraft either descents towards the near the polar regions of Didymain or is transferred to an industrial, private consortium. The latter could be used to test spacecraft and/or payload operations. This could potentially support future asteroid resource utilisation activities.

The performed orbit propagation show no risk of return to vicinity of Earth (0.1 AU) within 100 years or the risk for impacting any planet within 50 years. Space Debris Mitigation requirements (at SRR) compliance can thus also be achieved without a successful landing on Didmain.. In this case this manoeuvre needs to be achievable with an acceptable probability and relevant constraints need to be imposed on a potential industrial or private team.

Regardless on the end-of-life scenario the spacecraft need to be passivated.

4.3 Functional Architecture

Figure 4-4 shows the function analysis for the different elements of the spacecraft bus. The functional analysis identifies, describes and relates the functions to a system, which the system must perform in order to be successful. It does not address how these functions will be performed. Where necessary traceability is shown to the original requirement given in the SRD. This assists in mapping the requirements of specific bus function, and will be used to define the later sub-functions.

The functions are therefore compatible with the requirements given in the SRD [AD03], and have been derived from the mission architecture and the proposed configuration for the space segment. Individual functions are then defined for each subsystem of the spacecraft.

The SC mode architecture is largely the same as AIM. The overall mode diagram is seen in Figure 4-3.

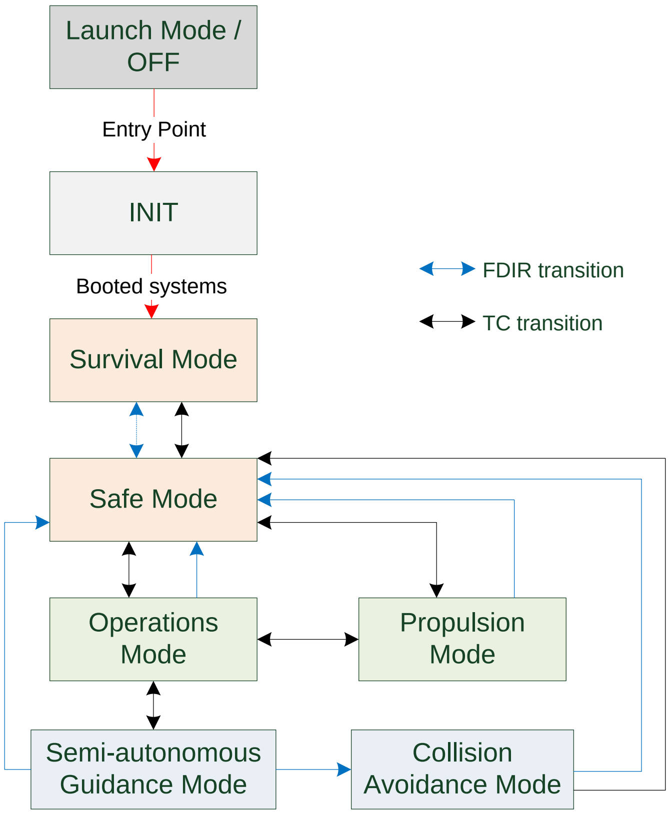


Figure 4-3 - SC modes diagram

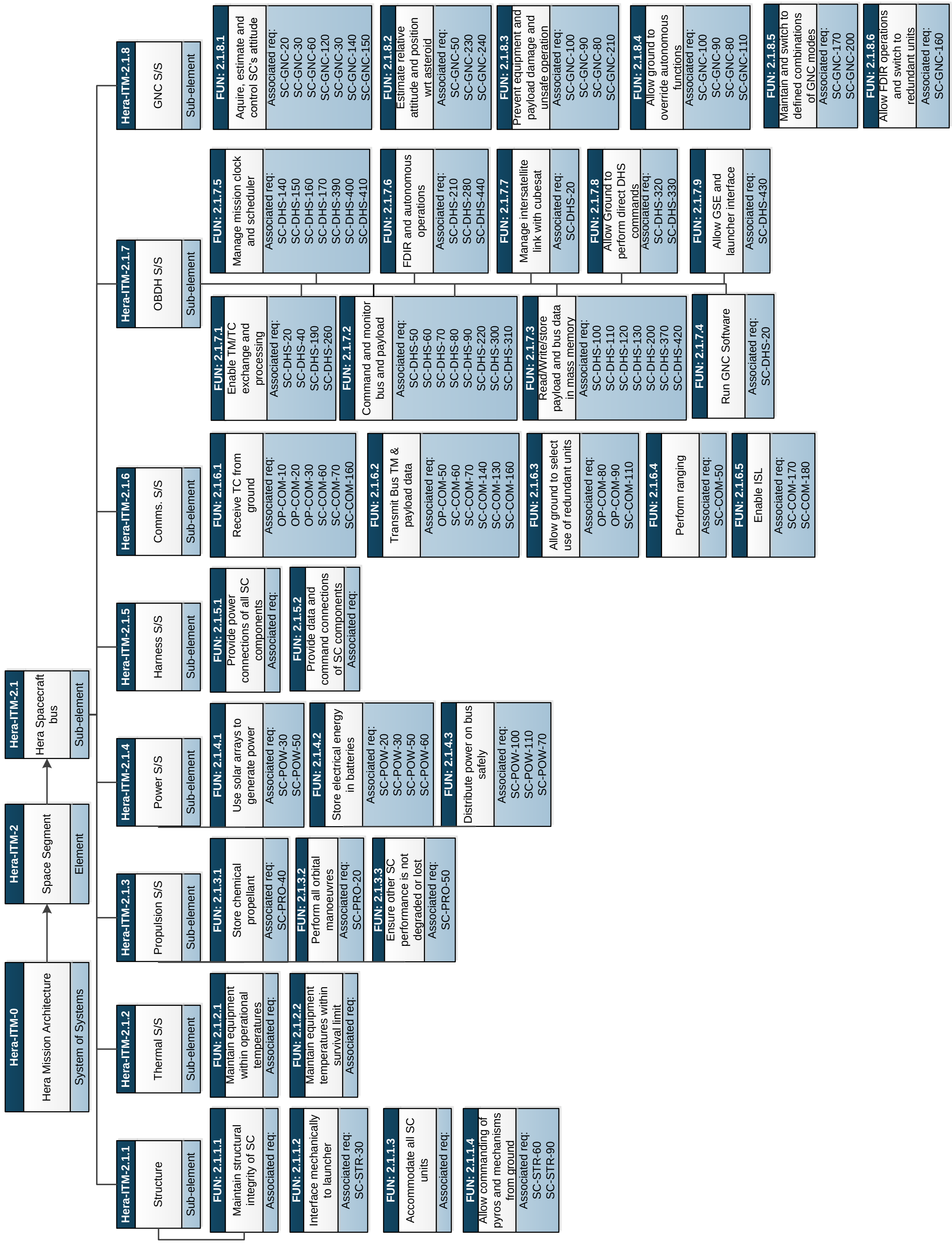


Figure 4-4 - Spacecraft Bus functional Diagram

Mode Descriptions

A description of each mode is provided here. For each mode, a list of the active GNC actuators and antenna is provided. Note that the list is a high level description, and in reality the GNC functionality and mode of operation is described by the specific GNC mode.

4.3.1 Launch Mode

This is not strictly a mode, as the SC is kept switched off during this period. Upon launcher release is initiates the boot sequence and the SC mode diagram is entered via the "entry point" red arrow in Figure 4-3. The sequence will involve autonomous switch on and power up, and
4.3.2 deployment of solar arrays.

Survival mode

In case of major system anomaly, i.e. when the actual system mode cannot be maintained, a
4.3.3 fall-back mode will be commanded by the OBC. By default, the system safe mode will be entered. In case of inability to maintain the safe mode, most likely because the S/C cannot compute a 3-axis attitude estimation, the survival mode will be entered instead.

In survival mode, the spacecraft rotates itself towards Sun ensuring the LGA pointing towards Sun. Due to the orbital constellation, even without omni-directional coverage capability a slow but constant communications link can be established.

In survival mode, the S/C relies on sun sensors for sun acquisition and keeping. No full 3-axis attitude estimation is available on-board, only 2-axis (i.e. sun direction is known, but not the rotation angle about the sun direction).

The star tracker will be activated in survival mode. If the star tracker is able to provide valid attitude measurements, the OBC will autonomously command a transition to safe mode.

Upon Launcher separation, the SC also enters survival mode. This is so it can immediately perform attitude corrections and sun acquisition immediately, without having to utilize wheels or star trackers. During this LEOP, the LGAs are used. Once attitude is stabilized and communications are established the SC moves to Safe mode, in compliance to requirement OP-PHA-30.

Actuators in use: ACS thrusters

4.3.4 **Sensors in use:** Sun sensors

Antenna in use: LGA

Safe mode

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 earth (through the HGA).

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).

Nominally, reaction wheels are used in safe mode (with RCS used for wheel offloading). This allows to be more fuel efficient, and maintain accurate pointing. Here, the HGA is used to maintain ground contact with Earth – the higher data rates allow to diagnose the system faster.

In case there is a problem with the wheels or some AOCS S/W, RCS thrusters are used instead. In this case, pointing performance is degraded, and the MGA is used instead to guarantee link (TBC).

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.

In order to point the HGA/MGA to the earth, the S/C must be aware of the earth direction. This information is provided by ground and stored in the context memory, such that it is still available after an OBC reboot.

Actuators in use: Reaction Wheels if possible. Otherwise ACS thrusters

Sensors in use: Star Trackers + IMU

Antenna in use: HGA in case Reaction Wheels are in use. If ACS thrusters are used, switch to MGA (TBC)

4.3.5 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 S/C 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.

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).

Actuators in use: Reaction Wheels

Sensors in use: Star Trackers + IMU

Antenna in use: HGA

Propulsion mode

The execution of large delta-V manoeuvres, such as the braking manoeuvres to rendezvous with the asteroid, deserve a dedicated system mode called propulsion mode. In this mode, the transfer thrusters can be commanded (by ground) to deliver a given delta-V, while the GNC thrusters are in charge of the attitude control. The reaction wheels are indeed not designed to
4.3.6 absorb all perturbation torques induced by the transfer thrusters.

Actuators in use: OCS thrusters for manoeuvre. ACS for attitude control.

Sensors in use: Star Trackers + IMU. Accelerometer in closed loop to control manoeuvre

Antenna in use: Baseline none. However if possible (depending on scenario) MGA/LGA.

Semi-autonomous guidance mode

As mentioned in the Hera Mission Requirements document, a demonstration of semi-
4.3.7 autonomous guidance is envisaged.

It requires on-board processing of the optical images taken by the AFC, which allows an on-board estimation of the relative position and attitude of the spacecraft wrt the asteroid. A centre of brightness algorithm or feature tracking algorithm can be used, depending on the specific situation (distance to the asteroid, required performance and robustness and 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. At the same time, robustness is ensured by comparing the on-board estimation with the ground prediction. When the difference crosses a certain limit, the ground prediction is used.

This strategy allows to provide flexibility to ground for the design of attitude profiles and to have a predictable S/C behaviour, while benefiting from more accurate pointing performance.

Actuators in use: Reaction Wheels

Sensors in use: Star Trackers + IMU + AFC

4.3.8

Antenna in use: None

Collision Avoidance Mode

Collision avoidance mode is triggered via the AFC or/and the PALT (TBC). The CAM is activated when Didymoon falls outside AFC field-of-view and/or a distance deviation of TBD m is detected by the PALT (TBC) or distance from size estimation using AFC images. It is intended as a type of "emergency mode" in which the SC must be able to quickly and reliably move away from the asteroid. Once CAM is triggered (for instance from Close Fly-By Mode) the system has already deduced that it is in danger of collision, thus CAM does not need to determine this – it just needs to avoid said collision.

In the collision avoidance mode a pre-defined sequence is executed by the GNC module to ensure that the spacecraft increases its distance towards the asteroid system and is on a safe trajectory. Entering CAM indicates loss of correct position and/or attitude information. To recover the correct current state the current attitude and position information shall be saved with a flag indicating its consistency with all sensor data. In case CAM is triggered, the last state with a correct flag is loaded and used to initiate collision avoidance actions (i.e. bring the SC onto a collision free trajectory).

After a successful CAM manoeuvre, a transition to safe mode is autonomously commanded. Ground can also force out of CAM, in case of erroneous trigger.

Actuators in use: ACS thrusters

Sensors in use: IMU + PALT

Antenna in use: None

Table 4-1: Summary of Actuator and Sensor sue during different spacecraft modes

Mode	STR	IMU	SS**	AFC	PALT	ACS	OCS	RW
SURM		x	x			x		
SFM	x	x				x***		x
OPM	x	x						x
SAGM	x	x		x				x
CAM		x			x	x		
PRM	x	x				x	x	

* : actuators used during close fly-by mode are TBC, and could require ACS thrusters for added manoeuvrability

** : sun sensors are almost always used as inputs to FDIR to verify that the current attitude estimate is correct. However they are not actively part of the attitude estimation process, other than during survival mode.

*** : Nominally, reaction wheels are used during safe mode. In case this is not possible, ACS thrusters can be used in safe mode as well.

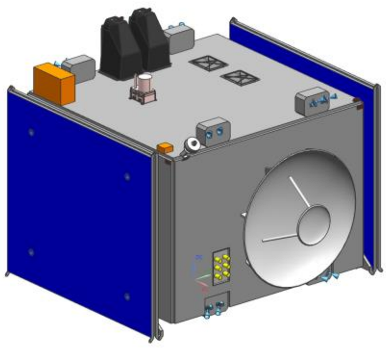
5 SPACECRAFT DESIGN

5.1 Design overview

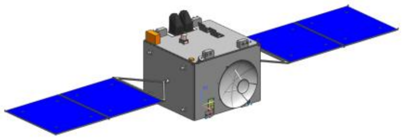
Table 5-1 on the next page provides a quick overview on the overall Hera spacecraft baseline.

Table 5-1: HERA System Overview

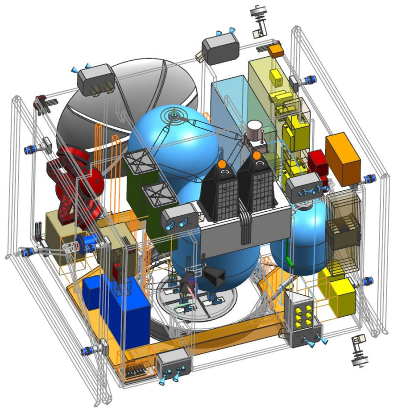
Spacecraft Design		
Payloads	2 × CubeSats 6U	
	2 × AIM Framing Camera (AFC)	
	1 x PALT	
	1 x Thermal Infrared Instrument	
Payload Support	CubeSat deployers	
	Inter Satellite Link	
Dimensions	Stowed	2199 × 2072 ×1830 mm^3
	Deployed	2199 × 8887 × 1830 mm^3
Mass	Dry (w/ margin)	505 kg
	Propellant	<365 kg + 1 kg pressurant
Power	Max. consumption	4108 W @ 2.33 AU to Sun 514 W @ 1.9 AU to Sun
Structure	Double Shear web structure	
Propulsion	Bipropellant pressurized system	
	Thrusters	16 × 10 N thruster (ACS N+R) 4 × 22 N thruster (OCS N+R)
	Tanks	1×MON PMD tank (MTA PTP-166) 1×MMH PMD tank (MTA PTP-166) 2×Helium Vessel 50 l
Communication	Frequency	X-band Earth communications S-band ISL for Cubesats
	Antennas	2 × X-band LGA (omnidirectional), 2 × ISL, HGA (1m), 2 x LGA
	RF Chain	2 × X-DST (N+R) 1 × 35 W TWTA 1 x 70 W TWTA
OBDH	OBC	QinetiQ Proba Next
	RTU	Platform RTU internally redundant Propulsion RTU internally redundant Limited cross-strapping
AOCS & GNC	Three-axis stabilized platform	
	Sensors	2 × Star Tracker 6 × Coarse Sun Sensors 2 × IMU
		2 × AFC, used as NavCam (N+R)
	Actuators	4 × RW + RCTs
	Image Processing	IP-ICU
Thermal	passive: Radiators, heat pipes, MLI interface fillers; active: heaters	
Power	Solar Array	2 × SADM, deployable
		2 wing, 2 panels, 8.7 m^2 total
		MPPT
	Battery	8S4P ABSL 18650 NL
	Bus	28 V unregulated
Mechanisms	Solar Panel HDRMs, SADM	



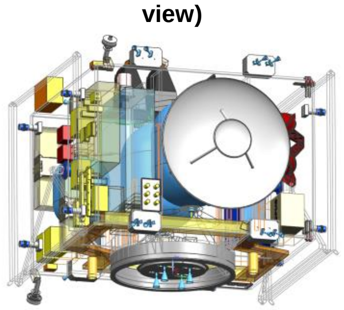
Stowed configuration



Deployed configuration



Stowed configuration (internal view)



Stowed configuration (internal view)

Figure 5-1 provides the system block diagram. The rational leading to this architecture is discussed in chapter 6.6.

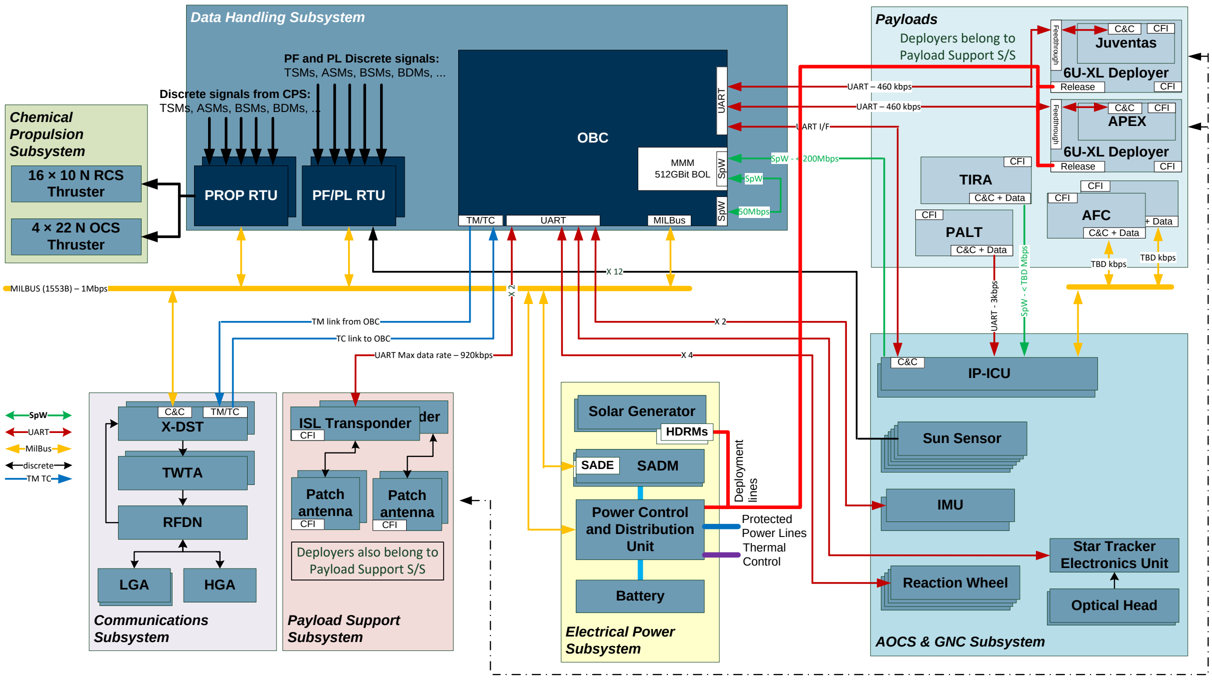


Figure 5-1: HERA System Block Diagram

The present Hera product tree can be found in Figure 4-4.

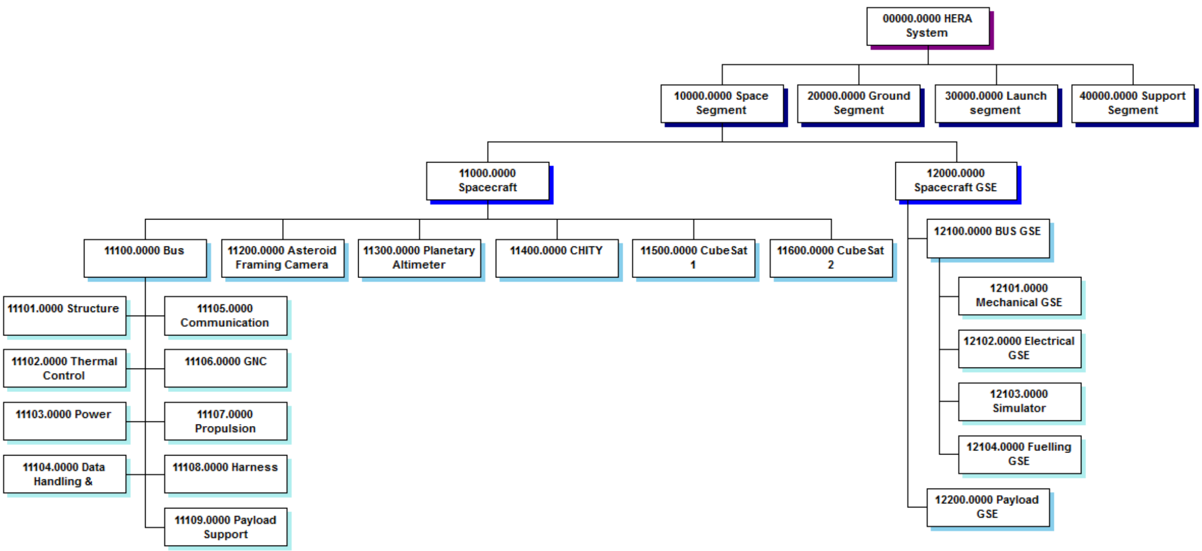


Figure 5-2: HERA product tree

5.1.1 Spacecraft Accommodation and Layout

The satellite in the stowed configuration is shown in Figure 5-3. From the dimensional point of view, the spacecraft accommodation within the Soyuz fairing is uncritical, see Figure 5-4.

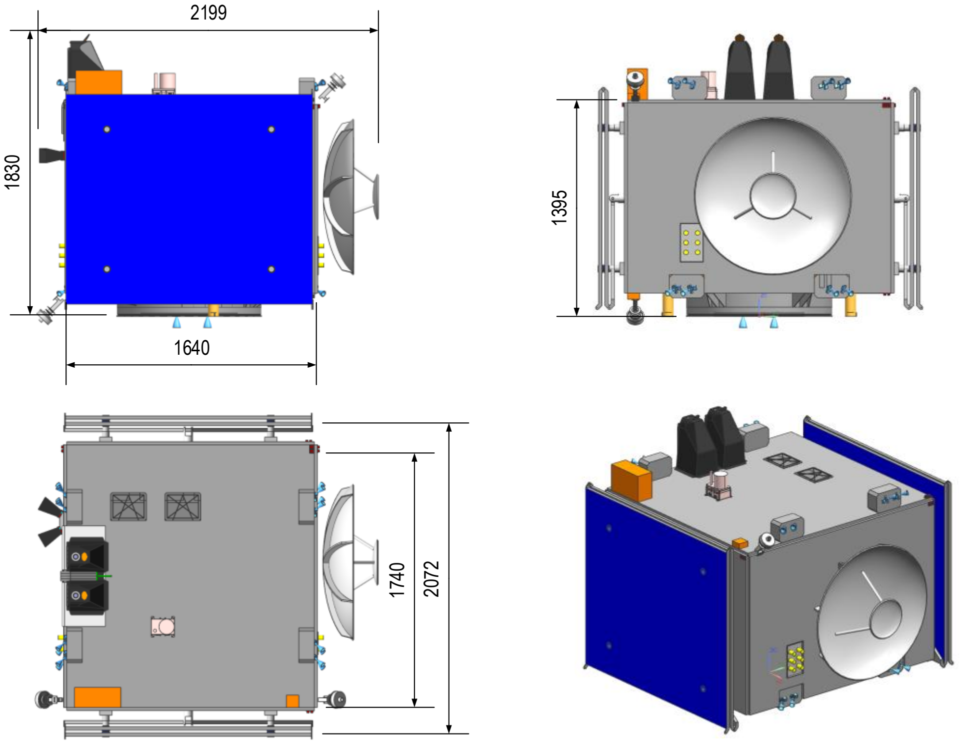


Figure 5-3: HERA Spacecraft in stowed configuration

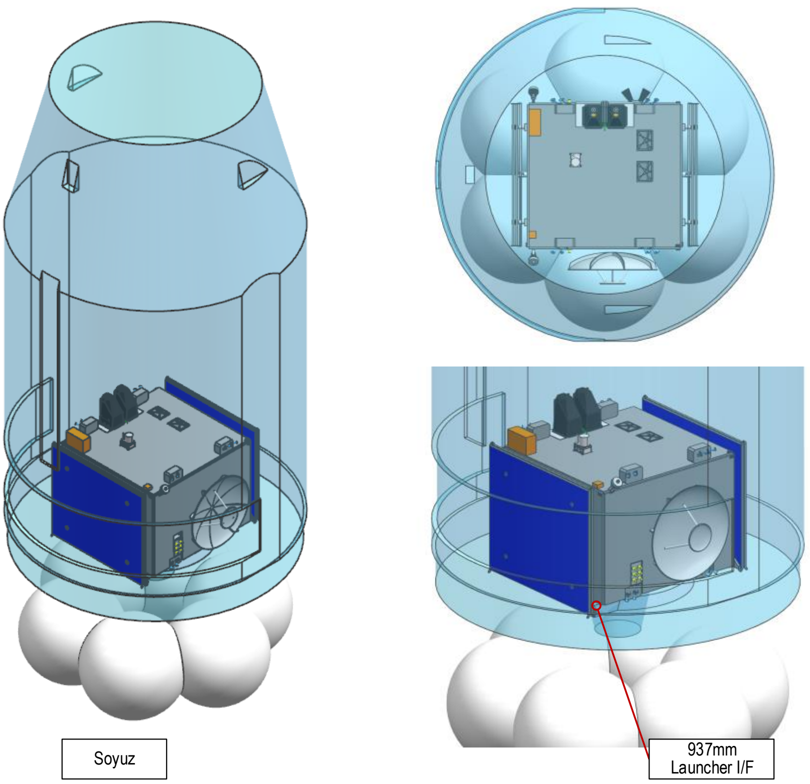


Figure 5-4: HERA SC in Soyuz launcher fairing

The deployed configuration is shown in Figure 5-5. The XY-plane of the global reference coordinate system coincide with the separation plane. The X-axis points towards the HGA and is parallel to its normal vector. The dimensions of the deployed spacecraft configuration are shown in Figure 6-3. The only elements to be deployed after launcher separation are the solar generator wings. The solar arrays is subdivided into two wings with each two panels and a SA area of 8.7 m².

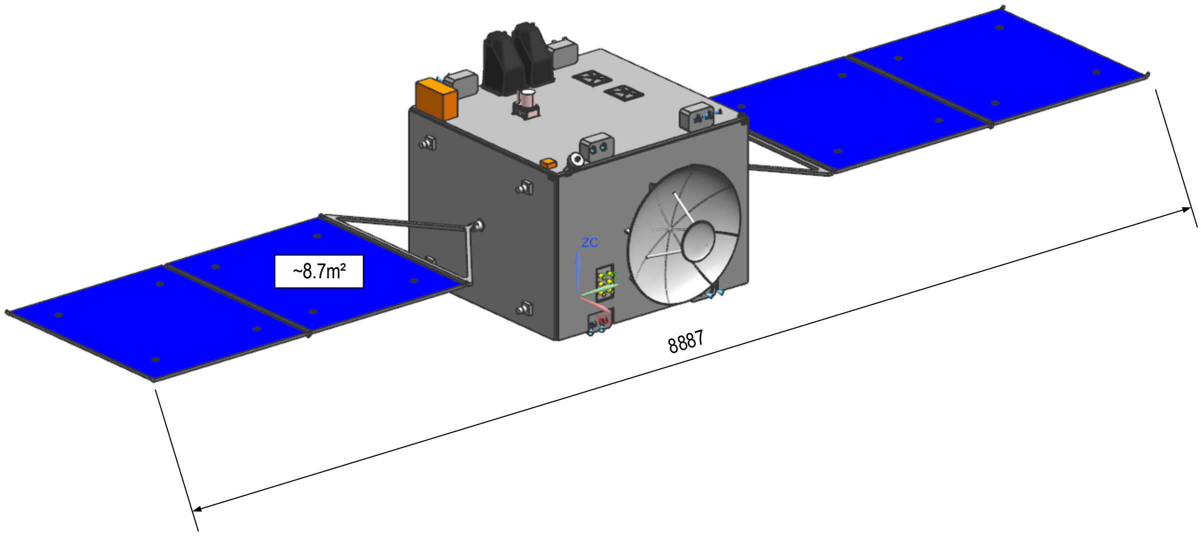


Figure 5-5: Deployed configuration of HERA spacecraft

The spacecraft bus is driven by the two tanks and the payload accommodation, as well as by the mission geometry constraints (antenna and solar generator accommodation). Due to the required ox-to-fuel ratio, the two propellant tanks are accommodated asymmetrically, which leads to an overall asymmetric SC geometry around the launcher's longitudinal axis. However, the spacecraft CoM is maintained close to the Z-axis. The two propellant tanks are accommodated in the central compartments. Most electrical units are located on the two radiators in +/- Y direction.

The spacecraft internal accommodation is illustrated in Figure 5-6. The +X and –X panels are equipped with the HGA and the star tracker heads only. Two LGAs provide hemispheric coverage. Equipment on +/-X phase requires support brackets as the panels are designed as closure panels. The –X face is constantly oriented to deep space throughout the mission.

All payloads are mounted on the +Z panel. In case of the CubeSats deployers the mounting is supported via the central shear webs, which can be seen in the middle of the pictures (Figure 5-6 and Figure 5-3). The doors are opened in +Z direction and do not interfere with any payload or platform specific FOV. While no radiators have been explicitly marked in the CAD model, Figure 5-3 shows that no units are accommodated in the immediate vicinity of the dispensers. Any potential radiator blockage that might occur due to the dispenser lids opening is thus safely avoidable.

The location and orientation of AFC, PALT and TIRA are driven by the FOV, which shall have an optimum alignment to the target objects to be analysed. All instrument boresights are aligned with reference to the AFC boresight.

Regarding the deployable payload, the Hera CubeSats are mounted on a location that allows a safe release without embedding a risk of FOV obstruction of the observation payloads. It is equipped with its own separation mechanism.

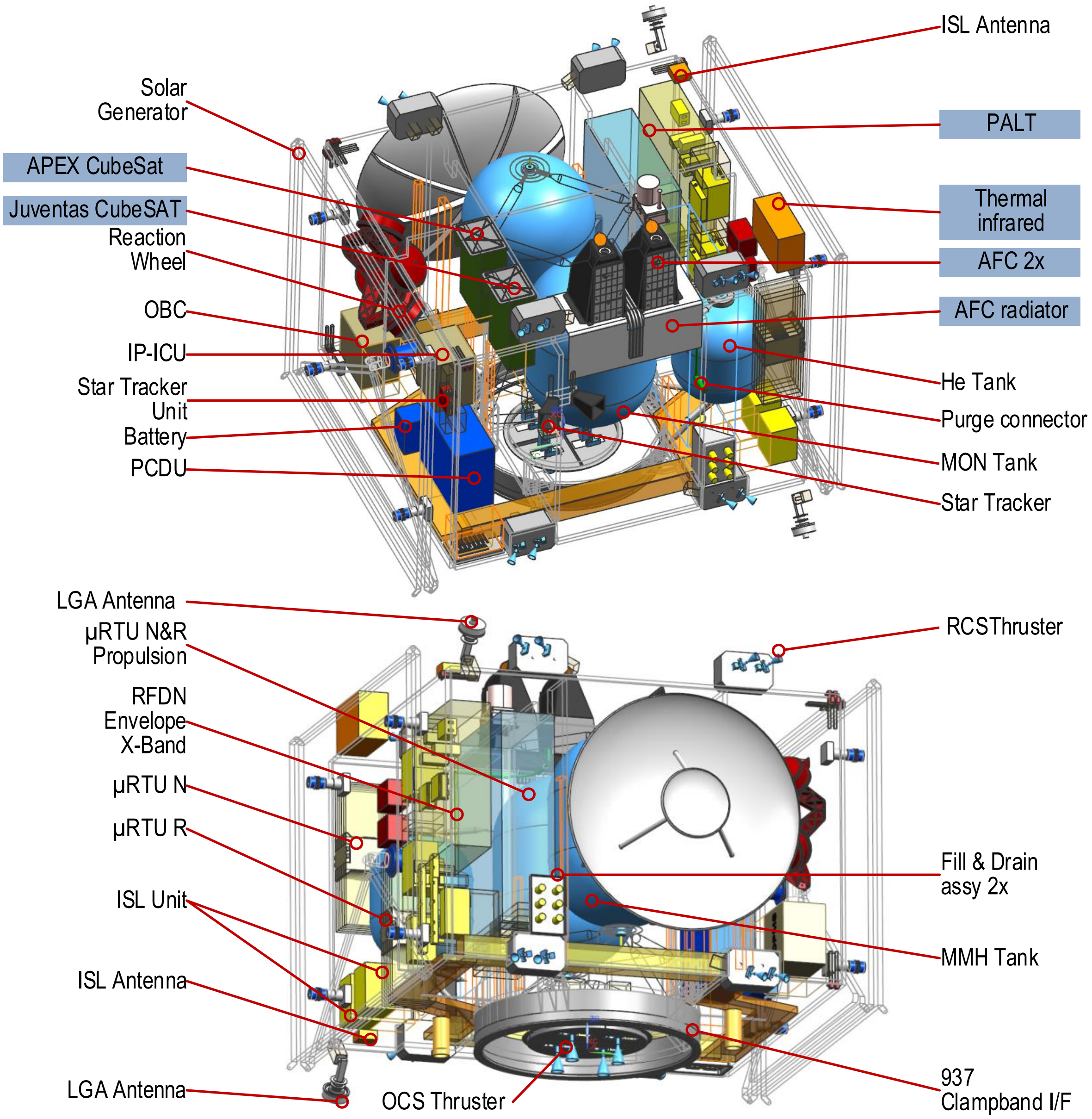


Figure 5-6: Internal accommodation of HERA spacecraft

5.1.2 Summary of main budgets

All budgets are summarised in [RD20].

The present design of the HERA spacecraft has a dry-mass of 505 kg including all margins. Depending on the transfer scenario, a variable amount of fuel is added leading to different launch masses. For a launch in 2023 and 2024 a propellant mass of about 303 kg is required (considering 100 m/s for proximity operations, leading to a wet mass of 808 kg. This is in line with the launcher performance of 857 kg after subtraction of launcher margin (50 kg) and launch adapter (110 kg). 2.4 % additional launcher margin are available.

The maximum power demand of the Hera spacecraft is 514 W and occurs during payload data downlink from the Didymos system. Although this case has to be considered for PCDU design, the solar array is driven by the 408 W required during safe mode at largest distance to Sun, which occurs during the transfer.

5.2 Payload

This chapter provides a description of the payload components carried on HERA, their role in the asteroid operations and impact on the spacecraft design, e.g. in terms of mass, power, data rates, data interface, pointing accuracy, vibration, etc. (Please note that AFC, PALT, TIRA, and Cubesats are the baseline payloads for HERA. CHITY, a hyperspectral camera, is not baselined anymore). Nevertheless, in the SRD [AD03] the payload is insufficiently defined since several requirements were missing. In order to fully assess and identify the impact of each payload component on HERA spacecraft is necessary to have this missing information to finally judge on the impact on the spacecraft and avoid late changes in the design.

The following payload requirements are provided in the SRD [AD03] for all the payload components: mass, volume, power and field of view. They affect the HERA spacecraft mass, volume and power budgets, the EPS architecture and sizing, and the spacecraft accommodation.

The payloads' components will also perform simultaneous operations having an optimum alignment to the target objects to be analysed. Given that, the AFC, PALT and TIRA boresights will be positioned along the launch direction and the payload components' configuration will exclude any possible risk of FOV mutual obstruction. Moreover, the payloads need to operate within the specific temperature ranges provided in Table 5-2. In this direction, a thermal control strategy for AFC, PALT and TIRA will be established to ensure that these devices can always operate with their required short- and long- term pointing stability.

Table 5-2: Operation temperature ranges of payload components

	AFC	PALT	TIRA	Hera CubeSat
Power-off Mode	-30 to 50°C	-50 to 70°C	-TBD to TBD°C	-40 to 50°C (before release)
Stand-by and Nominal Mode	-20 to 40°C	-40 to 60°C	-30 to 70°C	-30 to 40°C (before release)

The objectives of the HERA payload do not only cover asteroid research aspects, but also
5.2.1
technology demonstration. Hence, it is possible to divide the HERA payload into scientific instruments for asteroid research and technology demonstration elements.

Asteroid Research Payload

Asteroid Framing Camera (AFC). The AFC is similar to the framing camera carried on the NASA Dawn spacecraft in 2007. It is a multispectral imager, optimised to work both as scientific and navigation camera in the proximity of the asteroid. Its scientific objective is to support the understanding of the origin and evolution of large asteroids by the optical determination of their dynamical parameters (e.g. mass, rotation and gravitational field) and the mapping of the surface (e.g. to determine the shape and volume of the asteroid). By means of direct analysis of the imagery, it will be possible to determine the crater morphology and distribution, as well as the volume, shape and surface topology of the binary system. On the other hand, the multispectral images will provide data on the elemental and mineralogical composition of the asteroid. The AFC will also provide information on the dynamics of the binary system and it will verify the environmental safety prior to reaching the close vicinity with the asteroid, through long-exposure measurements capable of characterising the particles' environment.

As navigation camera, AFC will support the landmark navigation, based on different strategies which require on-board or on-ground image processing. The Asteroid Framing Camera comprises a refractive lens system, a baffle with a door in front of the optics tube, a set of 7 band-pass filters and a clear filter with wheel mechanism, a Charged Coupled Device (CCD) at the focal plane, a thermal stabilization system and supporting electronics. In terms of accommodation constraints, the optical axis of the AFC is required to be in launch direction. This drives the configuration of the other payload components since they need to have the same orientation.

Planetary Altimeter (PALT). PALT is a laser altimeter, which will mainly support the creation of digital elevation models of Didymos system's objects and the determination of Didymoon mass by measuring the distance between HERA spacecraft and Didymos. During the mass determination, PALT shall be operated in parallel with AFC and CHITY.

PALT will also measure the distance between HERA and the surface elements on Didymoon, contributing to the evaluation of the volume of Didymoon, and it will determine the shape of areas permanently shadowed on Didymoon. To perform this task, PALT needs to be very close to Didymoon (less than 2 km). In case the instrument can perform Doppler measurements, it will be possible to direct measure the wobble and hence to estimate gravitational parameters of the asteroid.

Thermal InfraRed imager for Asteroids (TIRA). TIRA is a Thermal Infrared Imager, with which thermal infrared images of the surface of the asteroid even for the shadow or night-side can be obtained.

It observes the temperature variation (day and night) and thermal inertia of the asteroid by imaging the thermal radiation from the asteroid. By measuring the variation, the physical condition of the surface, if it is covered with fine particles or solid rocks, can be observed.

5.2.2 Technology Demonstration Payload

6U HERA CubeSat. HERA spacecraft design can accommodate two 6U CubeSats (named APEX and Juventas), which shall be deployed in proximity of the Didymos system barycentre. Currently, the CubeSats payload is not fixed, however at the end of their lives, they will land on Didymoon. The landing dynamics and location will be observed by the AFC and CHITY; each CubeSat will be equipped with accelerometers to measure the landing dynamics.

The spacecraft will support the CubeSats activities via an inter-satellite link, which will overcome the limited computation capability of a CubeSat and the high complexity given by a fully independent on-board GNC visual-based navigation system. Moreover, it will enhance the autonomous position determination of the CubeSat, providing ranging capabilities. Lastly, the inter-satellite link will allow a continuous exchange of data between the two systems.

Inter Satellite Link (ISL). The ISL shall insure the communications between CubeSat and the HERA spacecraft in deep space, including the transmission of payload, transfer of housekeeping data and telecommands from/to the CubeSat. S-Band is selected as bandwidth for the ISL.

Using ISL systems, HERA spacecraft will act as a mother-ship spacecraft with full communication capability with ground and the CubeSats will rely on HERA for all communications. As mentioned earlier, ISL shall also improve the navigation system of the CubeSat by providing relative positioning.

GNC and Autonomy. HERA will be the first European satellite to demonstrate an autonomous visual navigation in the vicinity of an asteroid and for low altitude fly-bys. For very low altitude fly-bys, it shall also prove autonomous GNC. It is intended to demonstrate the advantages of an autonomous and advanced navigation system, which uses information from different types of images, i.e. hyperspectral, vision and thermal infrared images. HERA spacecraft will be equipped with a set of sensors and navigation algorithm independent from the GNC system to investigate the collision risk of the trajectory being flown.

6 ANALYSIS AND TRADE-OFFS

6.1 Overview

The present Hera design is a tight follow-up of the previous AIM design. In some cases changes were mandatory and let to new trade-offs. The summary of those trade-offs is provided in the following table. For each trade-off the main options, as well as the selection is provided. The comment gives a high level justification for the selection as well as a outlook on the system impact. The column chapter provides the section of this document where this trade-off is discussed in more detail.

Trade-Off	Options	Selection	Chapter	Comment
Mission Trade-Off				
Baseline and back-up trajectories	MAG transfers + several additional options provided by ESOC	EEMA 2023, EMA 2024, secondary back-up TBD	6.2.1	Trajectory selection impacts deltaV demand, power and link sizing cases
Orbit Control Thrusters	2+2 x 10 N 2+2 x 22 N	2+2 x 22 N	6.2.2	Both options feasible
Survival Mode	Strobing + MGA Sun-pointing and LGA	Sun-pointing and LGA		Simplified survival mode operations
System Trade-Offs				
Solar Wings	1 Solar Wing 2 Solar Wings	2 Solar Wings	6.3.1	Symmetry and SA size
Downlink approach at 2.33 AU	Continuous (SA powered) Interval (battery powered)	Continuous (SA powered)	6.3.2	Safe mode robustness Avoid battery sizing case SA sizing case
Antenna Pointing	Fixed Pointing mechanism	fixed	6.3.3	Simplicity, cost, mass
Attitude Control Thrusters	8+8 x 1 N 8+8 x 10 N	8+8 x 10 N	6.3.4	Both options feasible, but 1N would require dual mode, which would result in additional mass, cost and complexity.
MTP Trade-Offs				
	Covered by system TO			
Electrical Trade-Offs				

Trade-Off	Options	Selection	Chapter	Comment
Solar array conditioning	S³R MPPT	MPPT	**Fehler! Verweisquelle konnte nicht gefunden werden.**	Better efficiency
PCDU	ADPMS PCDU Independent PCDU	Prel. Selection Independent PCDU	**Fehler! Verweisquelle konnte nicht gefunden werden.**	Higher maturity
Avionic Trade-Offs				
RTU Configuration	1 RTU 2 RTU	2 RTU (platform + Propulsion)	6.6.1.1	Improved AIT schedule and increased system reliability
Image Processing	On one OBC core Separate unit Image Processing Unit + other GNC software in OBC	Separate unit	6.6.1.2	All in OBC not feasible. Mixed solution to be assessed
Mass memory	In IPM unit In OBC Separate unit	In OBC	6.6.1.3	Heritage solution expected to be more cost effective
Antenna Technology	MOMETA Dish Antenna	Dish Antenna	6.6.1.5	Reduced schedule risk

Please note that the AIM trade-offs, which have been considered applicable in terms of assumptions and results have not been repeated and are also not discussed in this document. Please find a complete list in the Delta Assessment HERA/AIM document [RD01].The trade-off discussion can be found in the relevant AIM documentation.

6.2.1

6.2 Mission Trade-Offs

Identification of Baseline Transfers

The baseline launch date for the Hera mission is in early Q4 of 2023 with a primary back-up in Q4 2024. Out of the different options in the mission analysis guidelines and the additional trajectories provided by ESOC a Mars-Swing-by type of trajectory was selected offering an appealingly low-deltaV demand. It even allows increasing the launcher performance by reducing the required Vinf. This delta in velocity needs to be corrected by an additional transfer manoeuvre ‘shortly’ after launch. The deltaV on board and the required Vinf allow even a late tuning of the optimal launch scenario.

Due to the tight implementation schedule for a launch in 2023 or even 2024 a secondary back-up is required. Unfortunately the orbital constellation between Earth and Didymos (and Mars) does not allow to find another similar transfer. All later back-up require significantly higher deltaVs for the transfer operations up to a factor of two or three and significantly longer transfer times. In addition most of these trajectories require only one major propulsive manoeuvre, namely the arrival. Even though the launcher Vinf requirement is significantly less than the maximum launcher performance a shift of deltaV from spacecraft to the launcher is not feasible, as the arrival manoeuvre cannot be compensated. A vice-versa shift is also not possible due to the limitations of the spacecraft as further discussed in the following.

It is the intention to fit all possible launch scenarios with one spacecraft design. In most elements of the spacecraft this does not create a big impact. Critical for the selection of the set of trajectories is on the one hand the launcher performance and the tank selection.

Based on the analyses performed in AIM a bi-propellant architecture for the chemical propulsion subsystem was selected. This limits the possibly available tanks for such a mission in the respective volume range to PMD tanks.

Given the baseline launcher to be a Soyuz 2.1B Fregat from Baikonur or Vostochny the spacecraft will face horizontal integration and transport to launch side. Consequently not only a maximum filling ratio for the tank has to be considered but also a quite high minimum filling ratio to ensure a constantly wetted PMD. This filling ratio is typically in the order of 80% unless specific design solutions, like asymmetric PMD and matching accommodation within the spacecraft and on the launcher allow a lower minimum filling ration (as for PTA-166 ~50-60%).

6.2.1.1 Spacecraft Designs

Based on AIM analyses three different spacecraft designs have been established to support this trade-off.

- 2x PTA-125 (125 l)
- 2x PTA-166 (166 l)
- 2x OST-31/0 (177 l)

These designs are all building on the same architecture, as it is described and justified within this document, but are sized to accommodate the different tanks. Mass budgets and MCI computations have been performed for all of them. To increase the readability of this document, these designs are summarized in [RD21]. Table 6-1 provides the overview of those parameters mainly driving this trade-off, like dry-mass, tank volume and deltaV capability.

Table 6-1: Analysed spacecraft options [RD21]

	125 l	166 l	177 l
Spaecraft Dry Mass [kg]	493.8	505.0	525.0
Minimum Propellant Capacity [kg]	232.0	231.1	287.4
Maximum Propellant Capacity [kg]	273.3	365.5	390,1
Maximum DeltaV capacity [m/s]	1006	1279	1295

	125 l	166 l	177 l
Minimum Launch Mass [kg]	725.8	735.9	812.4
Maximum Launch Mass [kg]	769.3	870.5	915.1
Dimensions mm³ - **Stowed** - **deployed**	2072 x 2006 x 1687 8887 x 2006 x 1687	2072 x 2199 x 1818 8887 x 2199 x 1818	2072 x 2180 x 1827 8887 x 2180 x 1827

6.2.1.2 Trajectory Assessments

The available trajectories, six discussed in the MAG and 167 provided as one liners from ESOC have been assessed. For each year of potential secondary back-up(2025, 2026 and 2027) the one or two trajectories with lowest deltaV, have been identified and short listed.

For each shortlisted trajectory the launcher performance has been derived and the propellant mass computed. The result is provided in Table 6-2.

In case of the trajectories in 2023 and 2024, as well as the 2027-5r the ESOC deltaV has been replaced by the deltaV resulting from more detailed computations described in [RD24]. This is marked in orange.

Table 6-2: Trajectory options short list

Trajectory Definition										No additional fuel					
Transfer	Transfer Duration	Launch Date	Arrival Date [month]	Type	Vinf	Launcher Perf. [kg]	Tank size [l]	Dry-mass [kg]	Transfer deltaV [m/s]	DeltaV shift from Transfer to V_inf [m/s]	Transfer DeltaV incl.margins [m/s]	Wet Mass [kg]	Launcher Margin [kg]	Filling Ratio MON [%]	Filling Ratio MMH [%]
2023	3,3	45207	46412	EEMD	5,670	822	124	493,8	565	0	619	644,7	17,5	0,53	0,52
2024	2,3	45573	46413	EMD	5,670	822	124	493,8	562	0	616	644,0	18,1	0,52	0,52
2023	3,3	45207	46412	EEMD	5,100	1016	124	493,8	1089	-524	1169,385	790,1	66,1	1,03	1,02
2024	2,3	45573	46413	EMD	5,100	1016	124	493,8	1086	-524	1166,55	789,3	66,9	1,03	1,02
2025-1r	7,8	45802	48651	EMEED	5,250	965	124	493,8	1245		1333,5	839,9	-34,9	1,21	1,20
2025-30r	5,2	45977	47876	EED	5,040	1034	124	493,8	1383		1478,4	886,6	-12,1	1,37	1,36
2026-34r	6,3	46347	48648	EMEED	5,120	1010	124	493,8	1250		1338,75	841,5	8,6	1,21	1,20
2027-2r	5,3	46701	48637	EED	5,050	1031	124	493,8	1281		1371,3	851,8	19,6	1,25	1,24
2027-5r	7,8	46563	49412	EMEED	5,250	965	124	493,8	1090,7		1171,485	790,7	14,3	1,04	1,03
2027-2r_new	5,4	46690	48653	EED	5,174	992	124	493,8	1303		1394	859,2	-26,8	1,27	1,26
2023	3,3	45207	46412	EEMD	5,670	822	166	505	565		619,185	659,3	2,8	0,40	0,40
2024	2,3	45573	46413	EMD	5,670	822	166	505	562		616,35	658,6	3,5	0,40	0,40
2023	3,3	45207	46412	EEMD	5,100	1016	166	505	1089	-524	1169,385	808,0	48,2	0,79	0,78
2024	2,3	45573	46413	EMD	5,100	1016	166	505	1086	-524	1166,55	807,2	49,0	0,79	0,78
2025-1r	7,8	45802	48651	EMEED	5,250	965	166	505	1245		1333,5	858,9	-53,9	0,92	0,91
2025-30r	5,2	45977	47876	EED	5,040	1034	166	505	1383		1478,4	906,7	-32,2	1,05	1,04
2026-34r	6,3	46347	48648	EMEED	5,120	1010	166	505	1250		1338,75	860,6	-10,5	0,93	0,92
2027-2r	5,3	46701	48637	EED	5,050	1031	166	505	1281		1371,3	871,1	0,3	0,95	0,95
2027-5r	7,8	46563	49412	EMEED	5,250	965	166	505	1090,7		1171,485	808,7	-3,7	0,79	0,78
2027-2r_new	5,3744	46690	48653	EED	5,174	992	166	505	1303		1394,4	878,7	-46,3	0,97	0,97
2023	3,3	45207	46412	EEMD	5,100	1016	177	525	1089		1169,385	840,0	16,2	0,77	0,76
2024	2,3	45573	46413	EMD	5,1	1016	177	525	1086		1166,55	839,1	17,1	0,77	0,76
2025-1r	7,8	45802	48651	EMEED	5,25	965	177	525	1245		1333,5	892,9	-88,0	0,90	0,89
2025-30r	5,2	45977	47876	EED	5,04	1034	177	525	1383		1478,4	942,6	-68,2	1,02	1,01
2026-34r	6,3	46347	48648	EMEED	5,12	1010	177	525	1250		1338,75	894,7	-44,6	0,90	0,90
2027-2r	5,3	46701	48637	EED	5,05	1031	177	525	1281		1371,3	905,6	-34,2	0,93	0,92
2027-5r	7,8	46563	49412	EMEED	5,25	965	177	525	1090,7		1171,485	840,7	-35,7	0,77	0,76
2027-2r_new	5,3744	46690	48653	EED	5,174	992	177	525	1303		1394,4	913,5	-81,1	0,95	0,94

The feasibility of the individual transfer and spacecraft configuration is assessed on the three last columns. Technically only transfer/spacecraft configurations, which are all green are feasible without modifications to the present mission design, however the possible measures to achieve feasible transfers will be addressed in the following as part of the overall discussion.

6.2.1.2.1 Additional Launcher Margin

To obtain the addition launcher margin, the spacecraft wet mass was compared to the launcher performance after subtracting a launcher margin of 50 kg and 110 kg accounting for the standard Soyuz launcher adapter.

Options to improve the additional launcher margin are listed below:

- Reduce dry mass and by this also the required propellant mass (1 kg dry links to about 2 kg wet mass).
- Use lightweight launch adapter: The considered Soyuz standard adapter (110 kg) is very heavy compared to the adapter developed for Euclid (78kg). Using the different adapter yields the potential for an increase of the possible dry mass by 32 kg. For the time being this adapter is not taken as baseline, as it will be a costly item for the project. Nevertheless, it is considered as contingency option and decision milestones for its procurement will be derived.
- Improved launcher performance: The launcher performance assessment is based on Soyuz performance data from 2012. It is considered conservative. On top a 50 kg margin is considered. In parallel an RFI to Glavcosmos on the performance of Soyuz for the 2023/2024 launch slot resulted in a performance of 1015 kg instead of 822 kg. Even considering the 50 kg launcher margin, a significant performance increase results. A request for information on the other launches is on-going. Using the beneficial Glavcosmos performance would solve most launcher performance issues.
- Reduction of payload. The decision to remove e.g. one CubeSat including its deployer has the potential for more than 19 kg saving in dry mass.

In case the decision for the light weight adapter is taken, the allowable wet mass is rising by 32 kg for the cost of money, without any technical modification in the spacecraft. The Soyuz performance need to be modified closely, but an increased launcher performance at least by the 50 kg margin presently assumed is likely. This would then mean having the delta of Glavcosmos estimate and ESOC estimate as launcher margin.

6.2.1.2.2 Tank Filling Ratio

The tank filling ratio depends directly on the spacecraft dry mass and can only be reduced, if the spacecraft dry-mass is reduced. Most trajectories are presently feasible with respect to tank filling ratio, except for the 2025-30r EED transfer.

Please note that this parameter is rather important for the Hera mission. Some cases are close to the maximum tank filling ratio of 95 %. In case further analysis show a deltaV or propellant demand increase, which is not covered by the margins there is only a limited growth potential left before a bigger tank needs to be selected, which then has some snowball effect on e.g. structural mass.

6.2.1.3 Conclusion

The present analysis show a preference for the 166 l tank option, as they yield the best compromise of spacecraft mass and stowable propellant volume. The baseline launches in 23/24 are feasible with a good additional margin on the launcher performance.

Several back-up transfers in 2027 are found which also do not require further modification to the spacecraft.

Transfers in 2025 and 2025 can be made feasible by the described measures on launcher performance. The more lightweight adapter could be procured to enable the 2026 transfer case. In case of 2025 either the Glavcosmos launcher performance estimate need to be considered or one payload needs to be removed.

Orbit Control Thruster Selection

10N thrusters for the AOCS have been the baseline architecture during Phase A. The most
6.2.2 driving sequence for the attitude thrusters is at the asteroid during PL operations. The pointing requirements are driving the maximum allowed thrust inaccuracy per manoeuvre. The requirement is also induced by the trade-off S/C autonomy vs. continuous ground station command. The lower response time for the on-board autonomy is beneficial for the 10N thrusters. Thus the accumulated pointing uncertainty/error is smaller. The 10N thrusters can be operated in pulsed mode to reduce the net thrust.

The result of the trade-off 1N vs. 10N is shown in **Fehler! Verweisquelle konnte nicht gefunden werden.**. The evaluation reveals the strong benefits of the 10N thruster. The compliance to the required pointing accuracy at the asteroid performed with the AOCS thrusters has to be assessed in the next step. 1N thrusters are only available for mono-propellant, which requires the use of a dual-mode propulsion system. The mission is mass critical and the concept to use a mono-propellant architecture for the transfer is not feasible. Furthermore, OHB has a strong heritage in the use of 10N RCTs which is beneficial wrt. AIT schedule and risk. An additional benefit of using the 10N RCTs for AOCS is to use identical components to the transfer and the AOCS.

Table 6-3 Thruster Trade-Off 1N vs. 10N

Criterion	1N RCT	10N RCT	Justification
Propellant	- (mono-prop.)	++ (bi-prop.)	Bi-propellant architecture used for OCS and AOCS thrusters. 1N RCTs would require a dual-mode architecture
Propulsion Architecture	- (dual mode)	+	1N thrusters are only available for mono-propellant. However, the transfer thrusters are bi-propellant. Thus a dual-mode would increase the architecture complexity.
Pointing Accuracy at Asteroid	++	O	Achieved pointing accuracy with RCTs is better with the 1N thrusters. However, the required performance can be achieved also with the 10N RCTs.
Propellant Demand/Mass	-	+	Lower propellant demand for the 10N RCT due to higher ISP of the thruster and therefore a lower S/C wet mass.

Criterion	1N RCT	10N RCT	Justification
Company Heritage	-	+	OHB has no heritage in dual-mode architectures for 1N RCTs. The 10N RCT is very common for OHB.
Cost	-	+	The more complex propulsion architecture (dual mode) of the 1N concept increases the overall S/S costs.
Spare Philosophy	-	+/O	Identical parts can be used for the transfer and AOCS if the 10N thrusters are for both the new baseline
Redundancy	o	+/O	Additional redundancy would be gained for the case the AOCS and OCS thrusters (10N) are identical.

The trade-off reveals the strong benefits of the 10N thruster. At the end, the trade-off is depending on the analysis from GNC and OBDH. However, the feasibility to perform the AOCS manoeuvres with the 10N thrusters is discussed in [RD22]. Therefore the 10N RCTs remain as the baseline for HERA.

6.2.3 Survival Mode Strategy

The spacecraft enters survival mode in case of a major system anomaly. This potentially includes a loss of attitude information and the need to recover the spacecraft pointing.

Two main options are considered for survival mode implementation.

- **Option 1**. In survival mode, the spacecraft performs a slow rotation about the sun vector, such that the MGA describes a strobing pattern wrt Earth. Once every S/C revolution, a MGA will be pointed to Earth such that ground can pick up a signal. This allows ground to predict the motion of the spacecraft and to stop the rotation when the MGA is Earth pointing, such that a stable and uninterrupted communication link can be established. This option requires sun sensors and IMU active to identify the sun-direction and to control the strobing velocity. An uninterrupted communication link can only be established as soon as the rotation has been stopped by ground command. However, the required MGA will provide a significantly higher data rate compared to a LGA based communication.
- **Option 2**. Upon entry to survival mode, the spacecraft orients itself to a sun-pointing attitude and maintains it to an accuracy of +/- 5°. Communication will be established by LGA.

  This option requires sun sensors to recover the sun-direction. As soon as this is achieved and the LGA are switched on, a continuous communication on a low data rate is available, provided that the Spacecraft/Earth link can be closed.

During cruise the Hera spacecraft reaches distances to Earth, where an omnidirectional coverage is not feasible with the present communication subsystem design. A TC data rate of 15.625 bps can be ensured even using the New Norcia ground station. The minimum TM data rate varies from about 8 bps for New Norcia or Sardinia to a minimum of 26 bps using Goldstone ground station. The analysis presented in [RD23] shows relevant details for the 2023/2024 baseline and primary back-up.

Due to the robustness of the solution, the low complexity, the smaller amount of required equipment (sun sensors and LGA vs sun sensors, IMU and MGA) and the slow but continuous communications link not requiring an addition MGA the second options is the preferred scenario and present baseline.

6.3 Spacecraft

6.3.1 Solar Wings

Throughout the AIM activity different solar array concepts have been used. Phase A/B1 made use of a symmetric two solar wing configuration, while during the consolidation phase a reduction to one solar wing was introduced. In both cases the solar arrays are steerable via solar array drive mechanisms.

Body mounted or fixed panels can be excluded from the following discussions, as the mission goals in terms of data acquisition and data downlink require a primary pointing towards Didymos or Earth and the Sun-S/C-Earth angles varies up to 60° over the mission.

Main drivers for the selection of the appropriate solar wing concept are the following requirements:

- Required size of 8.7 m²
- Storable for launch
- Asymmetric solar array configurations cause parasitic torques, which have to be considered in GNC and proximity operations design

As the solar array size for HERA is significantly bigger than the requirement for AIM during the consolidation phase due to the increased power demand in the sizing case, the effects caused by an asymmetric solar array will become more severe than in the AIM consolidation phase configuration.

It was consequently selected to go for a two solar wing configuration to facilitate the proximity operations. The drawback are additional HDRMs, a second Yoke and a second SADM.

6.3.2 Downlink Approach at 2.33 AU

6.3.2.1 TWTA output power

Hera has two contradicting but driving requirements/constraints. One demanding a high transmit power, one yielding the need to reduce the power demand of the spacecraft.

It is required to downlink 100 Gbit of science data plus navigation and housekeeping data within the six months of proximity operations. Please note that Earth Didymos distance is increasing during proximity operations phase for the 23/24 transfer and thus the available data rate is decreasing over the science phase and reaches a minimum about three months after nominal end of science data acquisition

This drives the communication subsystem towards the selection of a high transmit power. As presented in [RD20] a 70 W TWTA does not even fulfil this target. The proximity operations need to be extended by about 1.5 months to downlink all data (considering no further science data acquisition). Employing 35 W transmit power would half the achievable data rate and thus extend the proximity operations by about 9.5 months.

This tendency is contradicted by the solar array sizing case and the mass limitations of the mission. The sizing case for the solar array is the entry to safe mode at maximum distance to sun (in case of the 2023/2024 trajectory 2.3 AU). Within safe mode operations only essential units will be active including the communications subsystem.

The top four power consumers in that mode are listed below (figures including component but no system margin and harness losses):

- Communications: 135 W in case of 70 W transmit power and 76 W transmit power in case of 35 W transmit power
- TCS: 96 W
- GNC 56 W
- Data Handling 50 W

Data handling power demand is mainly driven by the OBC and cannot be heavily reduced. Power saving will be investigated during the subsystem analyses.

During safe mode the spacecraft is required to point the HGA towards Earth for high data rate telemetry downlink and tele command upload. The attitude control is performed by reaction wheels. Without relaxing the pointing requirement, no major power saving potential is seen in the GNC.

The power demand for TCS has to be verified in detailed subsystem analyses but it seems unlikely, that a major reduction will be possible.

The overall power demand in that phase sums up to 470 W with high transmit power and data rate and 411 W with the lower transmit power and half of the data rate. The higher transmit power increases the power consumptions and thus the solar array size by more than 15 % or about 2.7 kg. The data rate increase linked to it, does not provide any major benefit, as the safe mode telemetry can easily be downloaded also with the lower transmit power as shown in [RD20].

A solar array which provides more than 411 W power to the spacecraft at 2.33 AU from is sufficient to cover the case of highest power consumption (520 W @ up to 1.8 AU).

As mass and cost saving measure it is selected to use two different TWTAs on-board the Hera spacecraft. The 35 W TWTA will be used during transfer communication windows. A 70 W TWTA will be used during science data downlink. In case of failure of one or the other TWTA the spacecraft will have to use the other to enable communication. This either results in an extended science data downlink phase or in a less efficient amplification if the 70 W TWTA is used at lower output power. A redundancy by diversity is anyhow feasible.

6.3.2.2 Interval vs. continuous downlink

Another option investigated for further reduction of power demand in safe mode is the idea to abandon the continuous downlink strategy in favour of an interval downlink supported by battery power. It was investigated, as the 500 W @ 1.8 AU requires only 75 % of the SA size as the 411 W @ 2.33 AU.

While performing TMTC downlink, no further reduction of the power demand is feasible, however a continuous downlink strategy could be replaced by an interval downlink approach in which downlink phases are interrupted by non-downlink phases. These phases are used for recharging the battery. The spacecraft will always be capable of receiving TC.

A top level assessment has been done to understand the possible saving in SA size, the affect on battery sizing and the data volume downlinked per day. It is reported in Table 6-4. The following figures are unaffected from the different scenarios:

- TC power demand: 37.8W (incl. comp but exl. System margin & losses)
- TM/TC power demand: 135 W (incl. comp but exl. System margin & losses)
- TM data rate @ 3.3 AU: 9 Kbit/s
- Present battery capability 165 Wh@28V (sized for LEOP only)

Table 6-4: Assessment of effect of interval downlink

DL duration per day [h]	averaged power demand [W]	DL data volume per day [Mbit]	Avg. power saving [W]	Avg. power saving [%]	SA size saving [m"]	Energy demand from battery [Wh]
1	41,9	32,4	93,2	22%	1,9	93
2	45,9	64,8	89,1	21%	1,9	178
3	50,0	97,2	85,1	20%	1,8	255
4	54,0	129,6	81,0	19%	1,7	324
5	58,1	162,0	77,0	18%	1,6	385
6	62,1	194,4	72,9	17%	1,5	437
7	66,2	226,8	68,9	17%	1,4	482
8	70,2	259,2	64,8	16%	1,4	518
9	74,3	291,6	60,8	15%	1,3	547
10	78,3	324,0	56,7	14%	1,2	567
11	82,4	356,4	52,7	13%	1,1	579
12	86,4	388,8	48,6	12%	1,0	583
13	90,5	421,2	44,6	11%	0,9	579
14	94,5	453,6	40,5	10%	0,8	567
15	98,6	486,0	36,5	9%	0,8	547
16	102,6	518,4	32,4	8%	0,7	518
17	106,7	550,8	28,4	7%	0,6	482
18	110,7	583,2	24,3	6%	0,5	437
19	114,8	615,6	20,3	5%	0,4	385
20	118,8	648,0	16,2	4%	0,3	324
21	122,9	680,4	12,2	3%	0,3	255
22	126,9	712,8	8,1	2%	0,2	178
23	131,0	745,2	4,0	1%	0,1	93
24	135,0	777,6	0,0	0%	0,0	0

Table 6-5 provides an overview on the pros and cons of an internal versus a continuous downlink strategy during safe mode.

Table 6-5: Pros and Cons of continuous and periodical downlink strategy

	Continuous downlink	Periodical downlink
Pro	• Continuous spacecraft telemetry available on ground (32 Mbit/h) • Violation of requirements SY-MOD-10 • Battery sizing case is LEOP • Lower operational complexity	Smaller SA size due to reduced SA size, up to 1.9 m² and relevant cost could be saved
Con	8.7 m² SA and resulting cost	• No continuous TM available on-ground. Available data volume depends on selected ratio • Safe mode is battery sizing case unless a ratio of 1:23 or 23:1 is chosen. Impact on battery cost • higher operational complexity

It is understood, that cost is presently the overarching argument driving trade-offs. However an interval downlink is not in line with the requirements baseline and introduces operational risk by limiting the telemetry while the spacecraft is in a failed state. A continuous downlink scenario during a potential safe mode during the transfer is selected.

6.3.3 Antenna Reflector Pointing Mechanism Trade-Off

The intention of this trade-off is to assess the need for a reflector pointing mechanism and to analyse the pros and cons of a fixed reflector. The complete trade-off shall to be only based on the reflector architecture.

The current S/C design is considering two degrees of freedom. Reflector is axis-symmetrical and the spacecraft can be rotated along the x-axis to achieve the first degree of freedom. Hera uses a Solar Array Drive Mechanism (SADM).

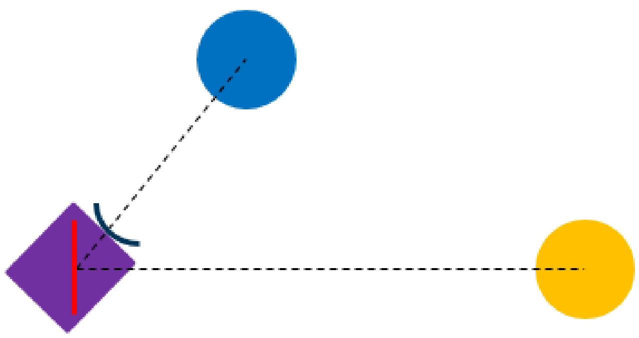


Figure 6-1 S/C Orientation (purple) with Sun (yellow) and Earth (blue)

The communication reflector during communication modes shall always point to Earth and the Solar Array shall direct to the Sun. The spacecraft, Earth and Sun define one plane (see Figure 6-1). During transfer the S/C is in an inclined orbit with Didymos. However, Didymos has a very low inclination of 3.4° (see Table 6-6). Minor deviations wrt. SA pointing to the Sun can be accepted, because the impact on the solar flux is negligible.

Orbital Elements at Epoch 2457800.5 (2017-Feb-16.0) TDB
Reference: JPL 134 (heliocentric ecliptic J2000)

Element	Value	Uncertainty (1-sigma)	Units
e	.3838328705983211	9.8107e-09	
a	1.644588561816618	7.7121e-09	au
q	1.013341413181381	1.8414e-08	au
i	3.407679182063813	3.275e-06	deg
node	73.22185799635652	2.3681e-05	deg
peri	319.2515891346795	2.6016e-05	deg
M	110.7962199046193	1.7727e-05	deg
t_p	2457563.413531463186 (2016-Jun-23.91353146)	3.6274e-05	TDB
period	770.3433271164755	5.4187e-06	d
	2.11	1.484e-08	yr
n	.4673240973573958	3.2872e-09	deg/d
Q	2.275835710451855	1.0672e-08	au

Orbit Determination Parameters

# obs. used (total)	863
# delay obs. used	6
# Doppler obs. used	0
data-arc span	7686 days (21.04 yr)
first obs. used	1996-04-11
last obs. used	2017-04-27
planetary ephem.	DE431
SB-pert. ephem.	SB431-N16
condition code	0
fit RMS	.32267
data source	ORB
producer	Davide Farnocchia
solution date	2018-Apr-04 12:07:24

Additional Information

Earth MOID = .0406808 au
Jupiter MOID = 3.16266 au
T_jup = 4.200

Table 6-6 Orbital Elements for Didymos [NASA/JPL]

The necessary S/C orientation to Sun and Earth is valid for the transfer sequence. However, during LEOP, thrust manoeuvres and at close-approximation this approach is not sufficient.

During LEOP the use of LGA is requested to ensure that constant communication can be ensured. The HGA is not needed during this phase and therefore not affected by the 2 degree of freedom constraint. During the thrust manoeuvers the thruster has to point in the required direction. Though, the HGA pointing to Earth is not required during this sequence. At Didymos an additional pointing requirement has to be met for the instruments. Then the communication has to be limited during the observation sequence at Didymos. Constant tracking requirements of Didymos (camera field of view) can limit the feasibility and detailed assessments are necessary.

In addition, an off-pointing of the Solar Arrays in the order of the inclination is negligible with respect to the reduced electrical power.

Criterions	Fixed Antenna	Movable Antenna	Justification
Accommodation Space Restrictions	++	o	The restricted design space for the fixed HGA is smaller than for the movable option.
Mass (unit and system impact)	++	-	The additional components for the pointing mechanism are increasing the S/S mass. HERA is mass-critical and the overall goal should be to minimize the system mass.
Operational Restrictions	-	+	The reflector pointing mechanism is providing two additional degrees of freedom. Therefore a downlink during close fly-by at the asteroid can be achieved while PL pointing to Didymos and the SA pointing to Sun.
Technical Risk / Reliability	++	o	The simpler architecture with a fixed reflector has a less probability to fail or malfunction.

Criterions	Fixed Antenna	Movable Antenna	Justification
Cost	++	-	The mechanism is adding costs wrt. additional components as well as the compensation of losses between TWT and reflector by a power increase of the TWT requires a resizing of the EPS system.
Overall Schedule Impact	++	-	More components for the moveable antenna leads to higher procurement, integration and testing time.

Table 6-7 Trade-Off for Fixed vs. Movable HGA

A pointing mechanism for the HGA is not required to fulfil the requirement. The two degrees of freedom (SADM and axis-symmetricity of the HGA) are sufficient. An inclination of the HGA on the S/C is not foreseen for HERA as well. Further assessments have to be performed if Didymos has to be in the camera field of view during flyby. Then its impact on system and subsystem level (e.g. communication data rate, available electrical power, camera field of view…) has to be assessed in more detail.

6.3.4 Attitude Control Thrusters

The purpose of this trade-off is to evaluate, if the identical thruster type can be used for GNC and for the transfer thrusters. The Phase A baseline is the 22N configuration with 2 nominal and 2 redundant thruster accommodated in the LIR. The redundancy concept shall be identical for both concept. The assessment of this trade-off is shown in **Fehler! Verweisquelle konnte nicht gefunden werden.**.

The GNC assessment for the technical feasibility of the 10N thrusters for the transfer is ongoing, but the preliminary results show that the concept is achievable. The GNC thrusters are not affected by this trade-off.

Criterion	22N RCT	10N RCT	Justification
OCS Architecture	O	-	The single-seat valves of the 10N RCT requires additional branch valves to be compliant to single point failure requirement.
Redundancy concept	O	O	Both options are using the same redundancy concept 2N+2R, but for the 10N thrusters with single-seat valve additional branch valves have to be considered.
Total propellant mass	+	-	Slightly higher propellant demand due to lower ISP for the 10N RCT
Total S/C mass	+	-	Slightly higher total S/C mass due to higher propellant mass for 10N RCT.
TRL / heritage	O	+	Components are flight-proven & the thruster is well known for the 10N

Criterion	22N RCT	10N RCT	Justification
Cost	O	+	The overall OCS component cost for 10N configuration is slightly lower and the AIT and procurement effort is lower as well for the 10N option.
Spare Parts	O	+	The approach of using the identical components as GNC and transfer thrusters is beneficial wrt. spares and therefore cost.
AIT	O	+	Less different parts improve the AIT process for shared configuration
Procurement (GeoR)	-	+	The shared conf. is based only on European components

Table 6-8 Thruster Trade-Off 10N vs. 22N

The major benefits of the 10N thruster concept is reduced technical and schedule risk due to the company heritage and less complex AIT process, the spare part concept and the procurement from European suppliers. The GNC analysis revealed that the minimum thrust level for the transfer would be achieved with 2 nominal 10N thrusters. Therefore the 2 nominal and 2 redundant thrusters for the 10 and 22N thruster concept could be accommodated in the LIR.

HERA is a mass critical mission. The strict wet mass limit is induced by the launcher and the propellant capacity. Because the ISP of the 22N thruster is higher (about 3,8%) , the resulting propellant mass is lower than for the 10N configuration. The mass difference wrt. different components is only minor and therefore the propellant mass is driving the overall S/C mass.

At the end, both concepts should be feasible. Each solution has its pros and cons. The 22N thrusters are the preferable solution for a mass-critical approach and the 10N thrusters are favorable regarding schedule, cost and risk. However, the GNC feasibility, as described in the text above, has to be shown by analysis. The final decision regarding the type of thruster will be done within the next weeks. Thus, the baseline is still the 22N thruster until the analysis is finalized.

6.4 MTP Architecture

6.4.1 Structure

6.4.1.1 Design Drivers

The launch loads [AD708/AD709] induce the main drivers for the structural design. The sizing parameters are structural strength, structural life, stiffness, damping, dynamic envelope, mass, and mechanical interface. From these drivers, the basic requirement of the structure can be derived; the structure should support the payload and the other subsystems with enough strength and stiffness to prohibit any failure that may keep them from working successfully. In addition, structural loads can also be induced by other components on the S/C such as HDRMs or Reaction Wheels.

The structure shall meet the requirements for the primary (Soyuz) and secondary launcher (Ariane 6.2). Therefore envelope cases are defined to consider the worst environment of both launchers.

6.4.1.2 Structure Architecture

Figure 6-2 shows the mechanical configuration of HERA. The double shear web primary structure is connected to a 937 mm interface ring to the LV. This concept allows accommodating the two polar-mounted bi-propellant tanks with adequate stiffness and has significant heritage by both OHB and ESA (i.e. Mars Express). The shown cut-outs enable harness routing along the bottom panel. The complete structure of the spacecraft is made of aluminium sandwich technology (CFRP sandwich panels will be investigated in the study, as potential mass saving option):

All panels are honeycomb panels (20 mm thick), aluminium core 3/16'' – 5056- .0007 as well as 3/16" – 5056- 0.001. . All shear webs and closure panel have face sheets made of CFRP and all remaining panels are built with aluminium skins 2024 T81 (0.3-0.5 mm thick).

The LIR (Launcher Interface Ring) and shear brackets are made of Al 7075-T7351.

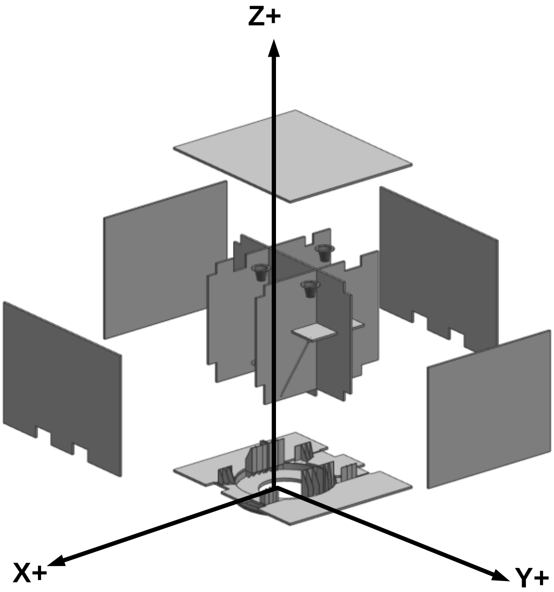


Figure 6-2: HERA spacecraft mechanical configuration

Six brackets connect the core structure to the launch adapter, which have been designed based on OHB's heritage in an ongoing project. Within the LIR, an inner aluminium sandwich plate is used to support the pushing of the brackets in the bending modes. The bi-propellant tanks are connected at the lower end with four CFRP rods to the shear webs via dedicated brackets and at the upper end to the +Z panel. A FEM model has been developed in Phase A/B1 form AIM to verify the structural performance versus the launcher requirements. The detailed design is described in the dedicated structure design report [RD234].

6.4.1.3 Structure Mass Budget

The analysis revealed full compliance to the Soyuz 2-1b requirements with a structural mass of 135.05kg with component margins. A mass budget is provided in Table 6-9. However, this budget is just the current preliminary status for PM2. Iterations to optimize the structural mass wrt. requirements will be performed afterwards. The detailed measures to reduce the S/S mass is described in [AD234] Based on a preliminary assessment, the results remain valid despite change of payloads and unit location.

After PM2 the measures have been taken into account to reduce the subsystem mass. The detailed mass breakdown is described in [AD234]. The analysis highlighted potential field for mass optimisation such as bracket design, unit support structure, LIR as well as primary and secondary structure. The shear webs are now made out of CRFP, which is also beneficial regarding thermo-elastic deformation.

HERA Structure S/S Mass Summary	Mass TOTAL (kg)	Component Margin (kg)	Mass TOTAL with Margin (kg)
LIR	7.10	1.42	8.52
LIR_Brackets	5.83	1.17	7.00
Primary structure	19.64	3,93	23.57
Secondary structure	36.87	7.38	44.25
Unit Support / Rods	9.73	1.95	11.68
Tertiary Structure (other BRKs, Cleats, Harness Support, Fasteners)	10.42	2.08	12.50
Structure S/S total mass			**107.5**

Table 6-9: Structure Mass Budget

6.4.2 Thermal

6.4.2.1 TCS Design Drivers

The thermal design of HERA is driven by the payload duty cycles, the environmental conditions during transfer and at the asteroid as well as by constraints in terms of mass and schedule.

- During transfer, the S/C reaches a distance of 2.6 AU to the Sun which induces an electrical power reduction to more than one sixth of the available electrical power at Earth. This necessitates a restriction in heater power demand during transfer to alleviate the burden on the EPS.
- The complete mission is mass critical which requires a mass-optimized TCS design.
- TCS requires a constant Sun pointing of the +X panel during transfer orbit. Further on, a constant Nadir-pointing in asteroid orbit mode is requested by the payload. Continuous power supply can be achieved by usage of SADMs. The mission profile requires pointing deviations of several degrees without endangering the low view factor of the radiator.
- Already partially degraded thermo-optical surfaces at the begin of nominal operations due to long transfer phase
- The main drivers for the heater power demand are the PL units and the reaction wheel heat dissipation and associated radiator sizing. Since a simple and robust thermal control design is desired, the high dissipations close to the Sun lead to relatively large radiators.

6.4.2.2 TCS Architecture

HERA has a quite ambitious schedule. Thus, a simple passive thermal control concept has been chosen as baseline design with heaters in closed-loop regulation as only active elements. The chosen thermal control concept is held flexible enough for possible future satellite-design changes and provides sufficient margins in accordance with the margin philosophy. A trade-off has been performed in Phase A/B1 of AIM between simple radiators, louvered radiators and heat switches for critical units. The results show that these options do not prove to be mass effective but also added risk and complexity to the spacecraft. The HERA thermal design includes the following major features:

- The unit dissipation is radiated by individual radiator sections on the respective panels.
- Grouping of units with identical temperature requirements simplifies the heater control.
- External units such as HGA and radar antennas have to be thermally decoupled to reduce impact on the S/C.
- Heater patches are located close to the units. The heater lines are grouped for units with the same temperature requirements.
- All equipment, which does not need access to space, is located within the satellite body. These components are mounted to the satellite's internal wall surfaces and heat is conducted to the mounting structure and/or radiated from the component surfaces to the internal spacecraft environment.
- The AFCs are mounted on a thermal double that functions also as a radiator plate. Heat pipes are implemented to transfer the heat from the units to the radiative area and by this improve the radiator efficiency. The top surface will be covered with white paint. The doubler is thermally decoupled from the S/C main body. Additionally, the double is required for the alignment of the unit wrt. S/C main axis.

6.4.2.3 TCS Mass Budget

HERA is driven by the maximum wet mass of the spacecraft. Thus, the associated subsystem mass budget is limited as well. The TCS mass budget for HERA is listed in Table 6-10.

Components	Mass [kg]	Margin [kg]	Mass with Margin [kg]
External MLI	7.81	1.56	9.37
Internal MLI	1.23	0.25	1.47
Thermal Finishes	1.91	0.38	2.29
Heat Pipes	2.03	0.20	2.23
Thermal Doublers	2.92	0.59	3.51
Heaters / Thermistors	1.96	0.39	2.35
Interface Filler	0.13	0.03	0.16
Miscellaneous	7.81	1.56	9.37
TCS Mass			23.0

Table 6-10: HERA TCS Mass Budget

6.4.3 Propulsion

6.4.3.1 Propulsion S/S Design Drivers

The driving requirements for HERA's propulsion system result from the different functional requirements, ranging from deep space rendezvous manoeuvres to manoeuvring at the asteroid. The following manoeuvres have to be covered by the propulsion system:

- Collision Avoidance Manoeuvres
- Detumbling after Separation
- Attitude Control in Survival Mode
- RW Desaturation
- Attitude Control while OCS is active
- Deep Space Manoeuvre
- Insertion burn

Based on these manoeuvres, the following requirements are driving for the propulsion system design:

- High Δv demand for rendezvous with the asteroid system
- Accurate low Δv for local asteroid operations

Overall, a Δv demand of 1250 m/s is required for the rendezvous with the asteroid system. To cover this large velocity demand and thus long firing duration, engines with high thrust and high specific impulse are favourable to reduce propellant mass. On the other hand, certain manoeuvres at the asteroid ask for an increment of merely a few cm/s, which favours thrusters with low thrust and small minimum impulse bit. Based on these conflicting requirements, a propulsion architecture with two types of thrusters has been selected to optimize the performance in light of the demanding mission constrains

6.4.3.2 Propulsion Architecture

In Phase A/B1 of AIM, trade-offs between dual mode and MON/MMH as well as LAE utilization have been conducted. A MON/MMH bi-propellant system without LAE has been chosen for the design, which offers high specific impulse, favourable COTS tanks, a double shear web structural concept (two tanks) and a low system mass. In addition to these benefits, it has a high European heritage and also benefits from OHB's telecommunication product line, which provides robustness and a low cost solution.

Considering the mass criticality of HERA, the propulsion system has been designed with the main objective to reduce the mass. Nevertheless, all design requirements, like single-point failure tolerance and standards, have been maintained. The propulsion schematic can be found in Figure 6-3.

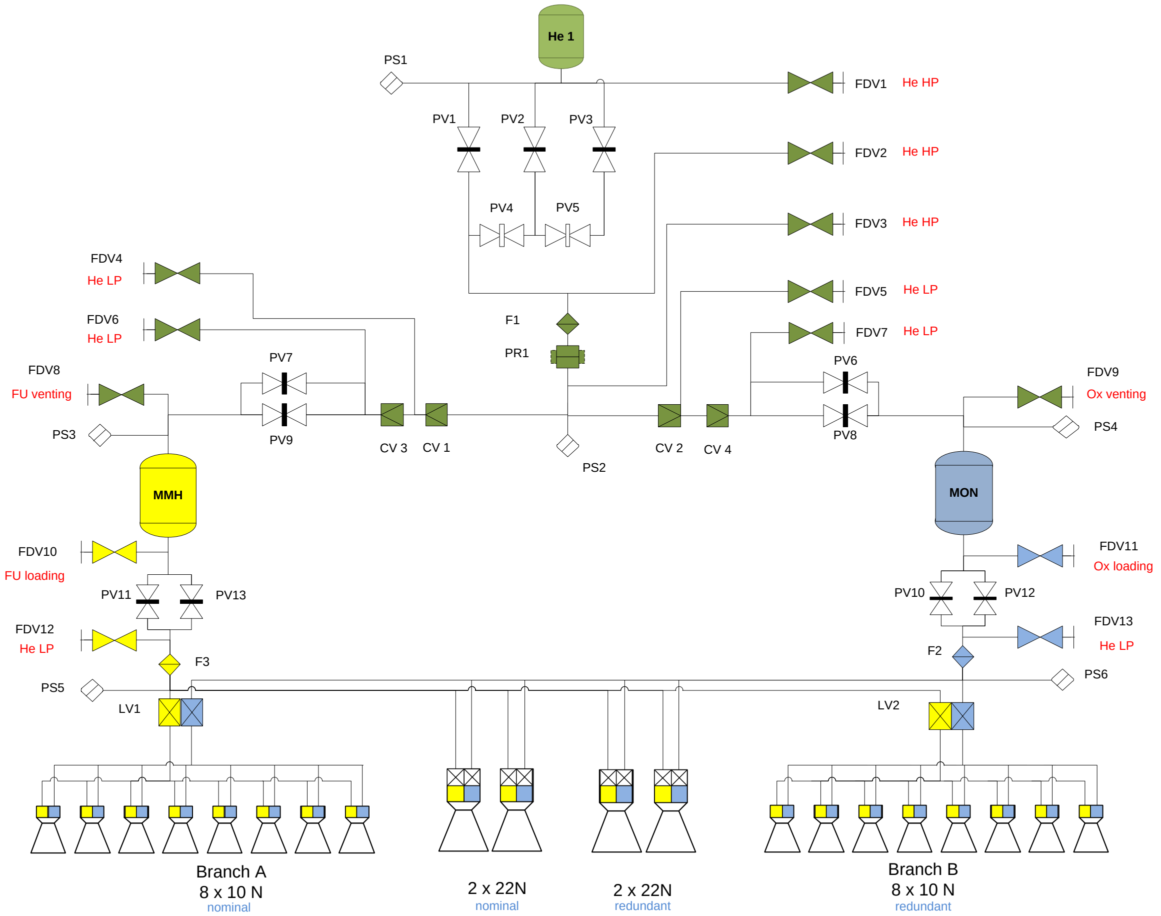


Figure 6-3: HERA propulsion schematic

The HERA propulsion system, as presented in Figure 6-3, is based on telecommunication product line bi-propellant systems and thus benefits from heritage, robustness and series production. All components selected for the baseline are flight qualified hardware and need no delta qualification. One 50 litre helium tank stores the pressurant at up to 310 bar. An internally redundant mechanical pressure regulator controls the pressure to the propellant tanks. MMH and MON are separated from each other and the helium reservoir by two series redundant check valves. The propellant is stored in two 166 litre tanks. For handling and launch safety, the propellant and pressurant tanks are isolated by normally closed pyro valves. The pressure in the tanks is measured by two redundant sensors each.

As main orbital control thruster, two DST-12 22 N thrusters from MOOG have been chosen with two additional thrusters as redundancy. These thrusters have a specific impulse of 300-304 s, which reduces the propellant demand by 8 kg compared to 10 N thrusters with a dry mass difference of only 30 g per thruster. The higher thrust allows completing the manoeuvres in less than half the time. Due to the needed manoeuvre accuracy, the 22 N thrusters are not well suited as reaction control system. Therefore, 8+8 10 N thrusters have been chosen based on their smaller minimum impulse bit performance. The canted accommodation allows a full 6 DoF force-free torques and torque-free forces. To reduce mass, single-seat 10 N thrusters from Airbus (ADS RCT S10-13/21) have been chosen with dual-line latch valves. This design enables single-point failure redundancy and three barriers on-ground, but reduces the overall mass without margin by 3.8 kg. Compared to a conventional design, the robustness is reduced because one thruster failure leads to a loss of the entire branch and necessitates a migration to the redundant branch.

The pressurant assembly is isolated via a pyro ladder. This allows to prime the pressurant assembly at BOL for the DSM and afterwards close the assembly until the arrival at Didymos. There, the pressurant assembly can be re-opened to fulfil the rest of the mission. In the sixteen months transfer period in between the manoeuvres, the system is operated in blow-down mode, which only leads to minor propellant penalties due to the very limited Δv required by wheel offloading. Using a pyro-ladder approach allows utilizing the high performance for high Δv manoeuvres as, well as to avoid pressure build-up in the tanks and the hazard to reach the MEOP. Currently, a second pressure regulator is not considered required due to the low duty cycle.

For the propellant tanks, two polar-mounted PTP166 tanks from MT Aerospace have been selected. With a current filling ratio of up to 98 %, the 166 litre tanks provide limited margins for evolution of the propellant demand, however they fit the deltaV requirements for the required launch mass. This model has been used on Eurostar2000 for MON/MMH storage and therefore provides strong heritage.

6.4.3.3 Design analysis

6.4.3.3.1 MEOP Analysis

The maximum accepted operation pressure of the propellant tanks is at 20 bar. The tanks are nominally pressurized during operation to 15.5 bar. Due to deviation of the propellant temperature the pressure in the tank may increase above this value. An analysis has been performed to assess the allowed maximum temperature deviation to prevent over pressurization.

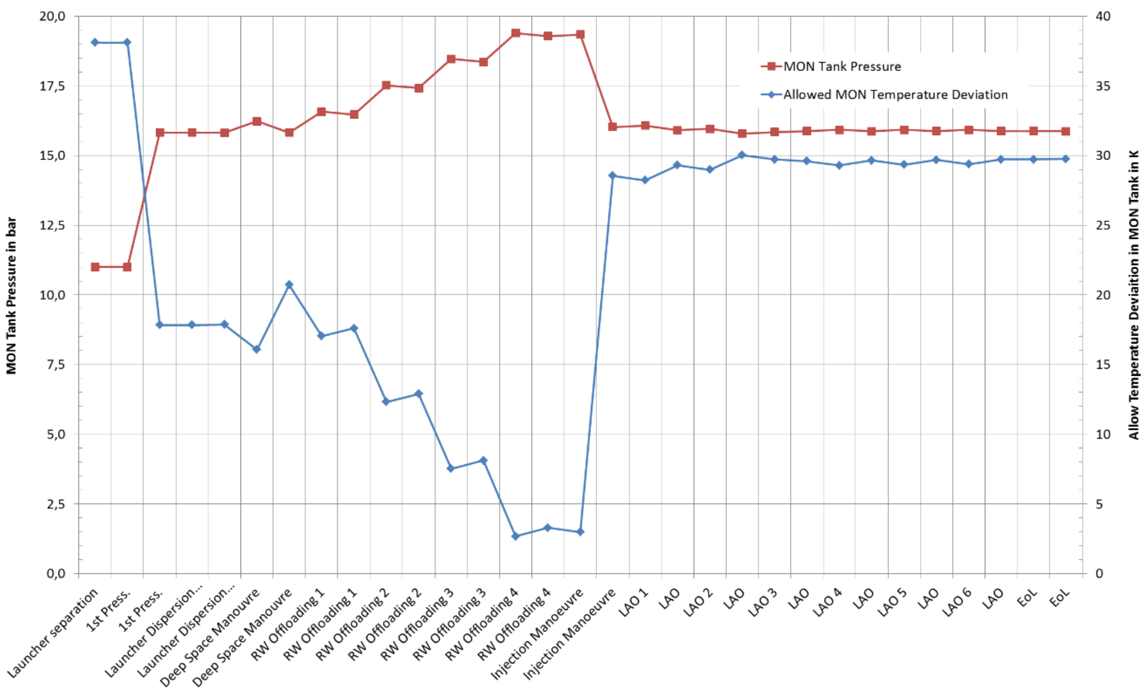


Figure 6-4: Worst case pressure evolution over time

6.4.3.3.2 Preliminary Pressure Drop Analysis

In the following a preliminary pressure drop analysis based on ExoMars is presented. The ExoMars propulsion system is selected as worst reference bi-propellant system incorporating several additional propellant latch valves and filters as presented below.

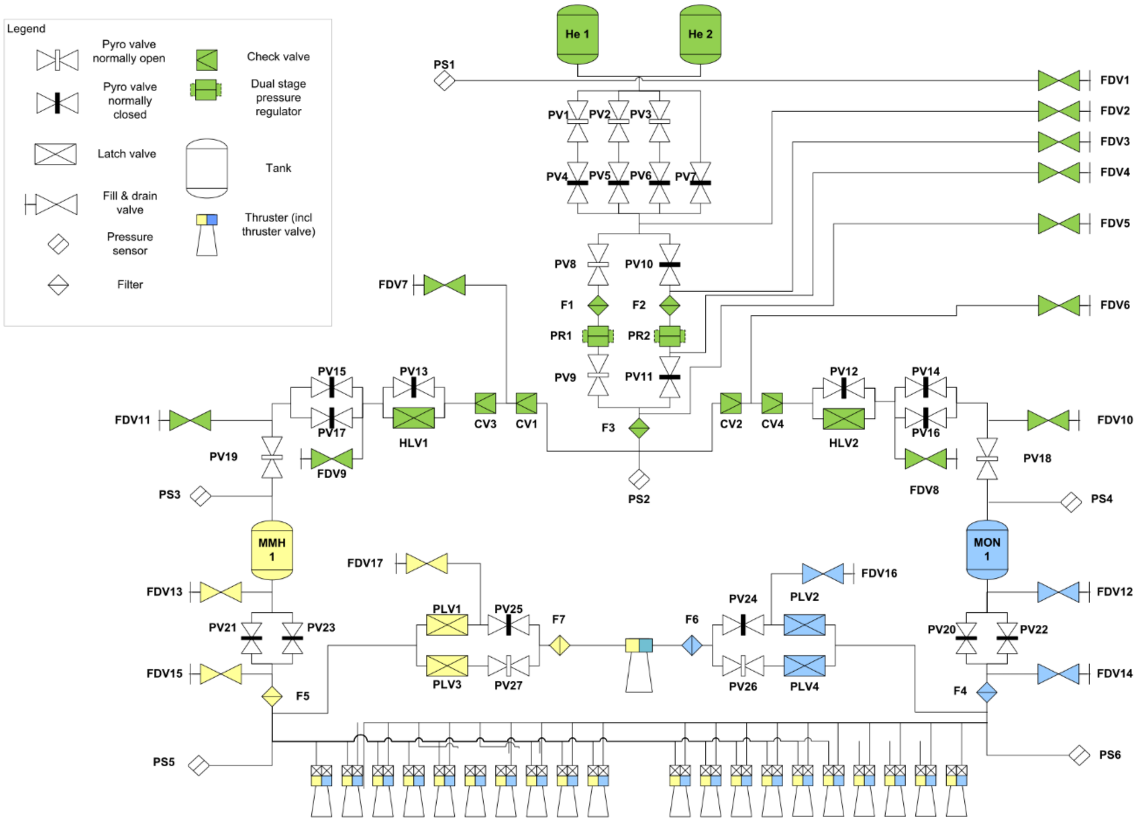


Figure 6-5: ExoMars Propulsion System

The ExoMars propulsion system has a pressure drop between pressure regulator (PR1) and main engine of 0.9 bar in the MON line. The check valve cracking pressure contributes to this pressure by 0.1 bar per valve independend of the mass flow. The remaining pressure drop of 0.7 bar can be scaled to the actual AIM mass flow (quadratic).

$$\Delta p_{Aim} = \Delta p_{Exo} \frac{\dot{m}_{Aim}^2}{\dot{m}_{Exo}^2}$$

ExoMars main engine requires 135 g/s propellant corresponding to 82 g/s MON. The worst case AIM mass flow is based on two 22 N RCTs operating continuously with 2 additional 10 N thrusters serving for reaction control. This corresponds to 22 g/s propellant or 13 g/s MON. Thus, the ExoMars pressure drop of 0.7 bar is scaled to 0.0184 bar for AIM.

Thus, for the low mass flow of the AIM propulsion system the pressure drop is driven by the non-scaled check valve cracking pressure of 2x 0.1 bar, resulting in an overall pressure drop of 0.22 bar. This pressure drop is sufficiently low to be covered in the preliminary operation pressure trade.

6.4.3.3.3 Propulsion Subsystem Redundancy Scheme

All active components of the propulsion system are fully redundant.

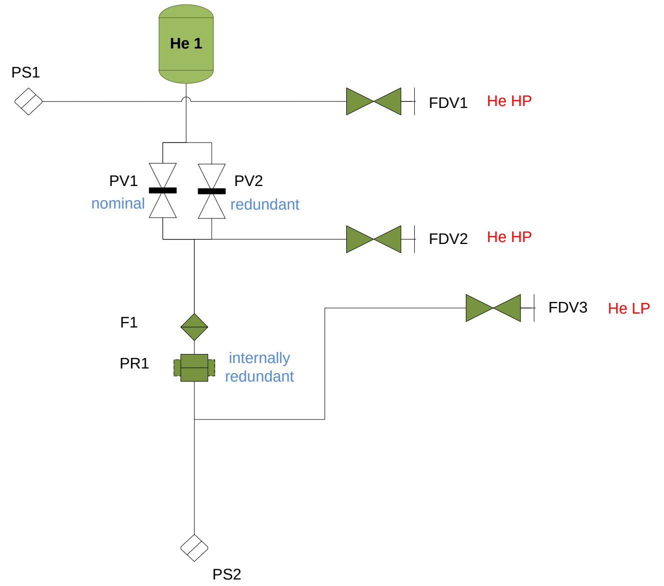


Figure 6-6: Pressure Storage and Regulation

The helium is stored in a 50 l helium tank from MTA. The high pressure is measured by the pressure transducer PS1 and indirectly by the downstream low pressure transducer PS2. The tank is filled by the FDV1. The tank is isolated on ground and during launch by parallel redundant normally closed pyro valves (PV1 and PV2).

The pressure is regulated from 310 bar down to 18 bar by the regulator PR1. The regulator includes series redundant seats against leakage and two independent pressure stages. The regulator is tested on ground by FDV2 and FDV3. It is protected against any pyro-valve particles by the helium filter F1.

No passivation assembly is required for the HERA mission as it will not stay in protected regions (LEO, GEO) after EOL.

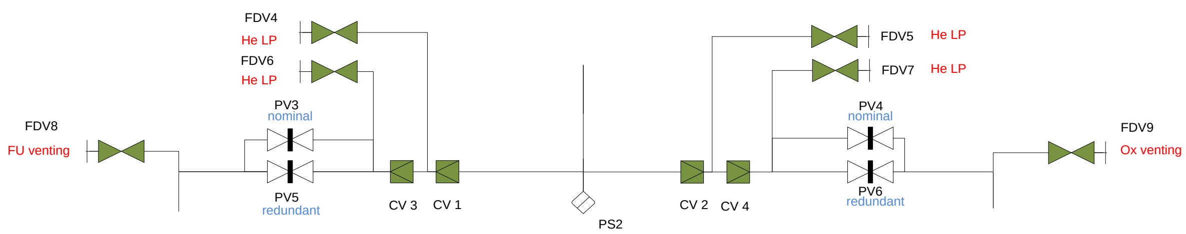


Figure 6-7: Propellant Isolation

On ground and during launch the Pressure Control Assembly (PCA) is isolated from the propellant tanks by redundant pyro valves in each branch (PV3 & PV5, PV4 & PV6). In orbit series redundant check valves (CV1 & CV3, CV2 & CV4) prevent propellant vapour migration into the gas section. All these sections are tested and pressurised on ground by FDV 4 to 7.

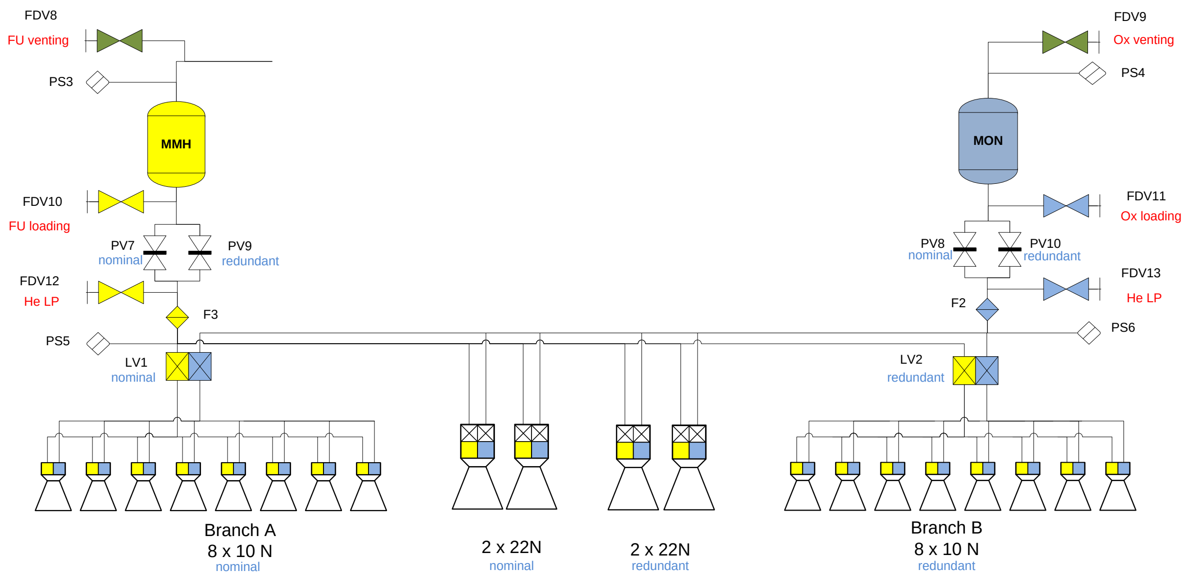


Figure 6-8: Propellant Storage and Thrusters Branches

The propellant tanks are filled by the fill drain valves FDV10 and FDV11 and pressurized on ground by FDV8 and FDV9. The tank pressure is measured by PS3 and PS4. After opening of the pyro valves (PV7-PV10) the pressure transducers PS5 and PS6 function as redundant sensors for the tank pressure. The propellant filters F2 and F3 protect the thrusters and filter all potential particles generated from the upstream pyro valves. Two latch valves LV1 and LV2 open and close the nominal and redundant branch of single seat 10 N thrusters. At least three barriers are included between propellant and ground by the thruster or branch latch valves, the thruster flow control valves and the pyro valves which are considered as internally two barriers.

Two nominal 22 N thruster perform main orbital manoeuvres. A fully redundant set can perform the manoeuvre in case of any failure of the nominal thrusters. For nominal operation only eight 10 N reaction control thrusters (RCTs) are required with additional eight thrusters for redundancy.

6.4.3.3.4 Plume Impingement Assessment on HGA

The accommodation (**Fehler! Verweisquelle konnte nicht gefunden werden.**) of the HGA reflector is limited by the plume impingement of the RCTs. The maximum heat flux induced by the thruster for a typical reflector shall not be exceeded. The type of thruster, distance and angle between the RCTs and HGA is driving the heat flux. Minor deviations wrt. thruster orientation can have a significant impact on the heat flux. The analysis is based on the current accommodation and orientation. The preliminary analysis shall assess the general feasibility of the baseline concept and evaluate the technical feasibility of a bigger reflector.

The position of the RCTs is predefined, because the orientation is driven by GNC and the location is predefined by the location of the structural interfaces to the shear webs. Therefore, the possibilities for adaptations is quite limited.

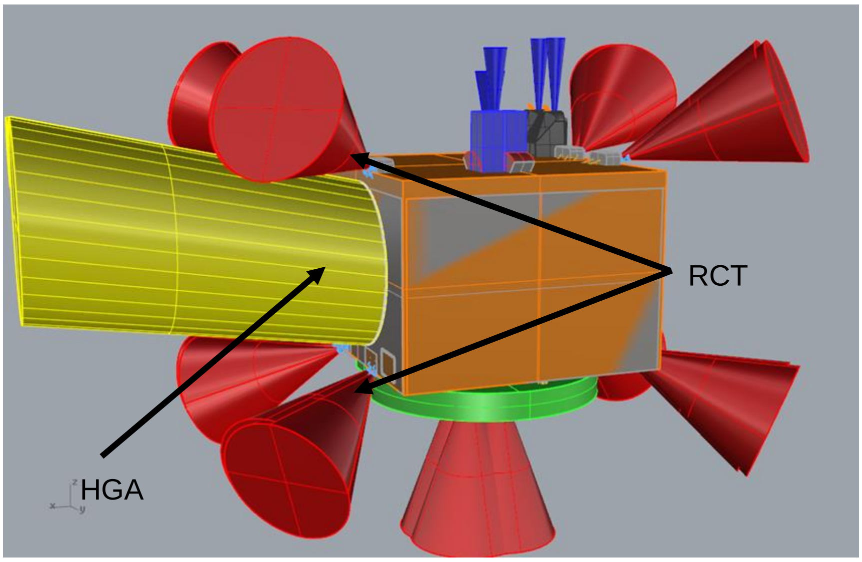


Figure 6-9 HGA and RCT accommodation including field of view

The 1m HGA reflector is the baseline for HERA. With increasing reflector diameter the gain of the antenna increases. Thus, the transmitter power can be reduced with constant data rate and signal-to-noise ratio. The lower power demand is beneficial for the electrical power budget and for the thermal control subsystem. However, the larger reflector might interact with the plume effects of the RCTs.

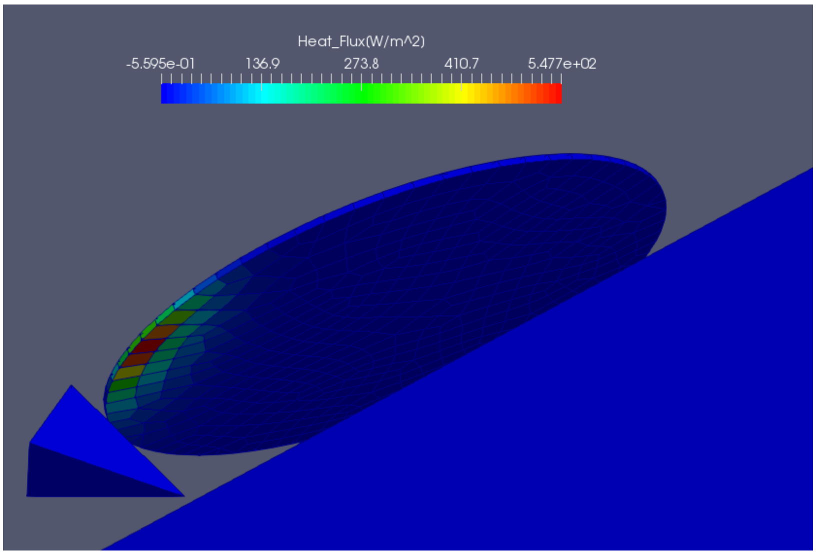


Figure 6-10 Heat Flux on 1m HGA

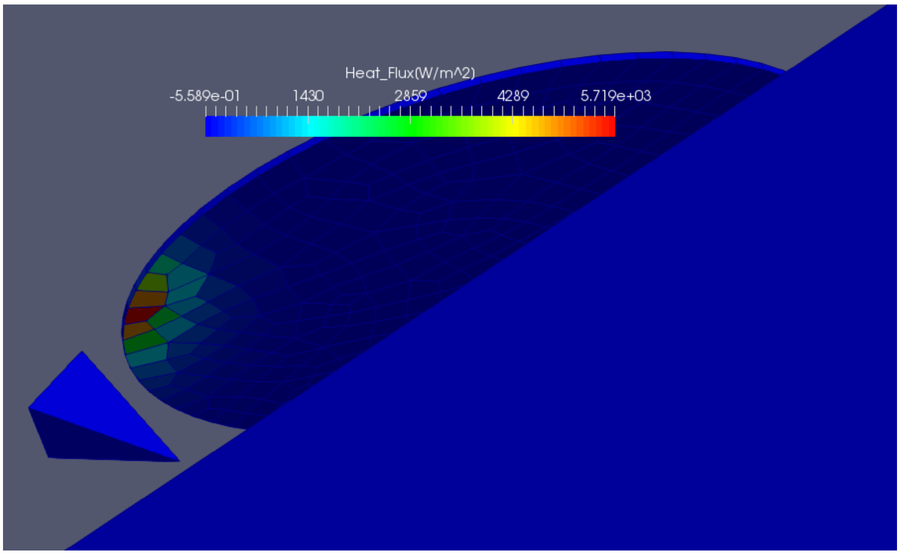


Figure 6-11 Heat Flux on 1.2m HGA

It can be revealed that the maximum heat flux on the 1m reflector is less than 550W/m² (**Fehler! Verweisquelle konnte nicht gefunden werden.**). For the analysis the worst case conditions are selected with the RCT closest to the reflector to evaluate the general technical feasibility. Typical reflectors with dedicated MLI are designed for a heat flux up to 1200W/m². Therefore, the baseline concept has sufficient margin especially wrt. the early phase and its preliminary analysis. The interference between the 1.2m reflector and the RCTs can be seen in **Fehler! Verweisquelle konnte nicht gefunden werden.**. The maximum heat flux is more than 10 times as high as for the 1m reflector. The technical feasibility is therefore not give. Therefore, the 1m reflector is still the baseline for HERA.

A parametric analysis could be performed in a later phase if the reflector size has to be increased. A minor re-accommodation of those components can have a significant benefit as well. At the end, those analysis are very time-intensive and will only be performed if essential for the mission success.

6.4.3.4 Propulsion S/S Mass Budget

The propulsion subsystem mass budget is listed in Table 6-11. The total subsystem mass including component margins is less than 54kg.

Table 6-11 Propulsion Subsystem Mass Budget

Component	Quantity [-]	Unit Mass [kg]	Component Margin	Unit Mass with margin [kg]	Subtotal with margin [kg]
10 N Thruster	16	0.31	5%	0.33	5.21
20 N Thruster	4	0.65	5%	0.68	2.73
MON tank	1	9.8	5%	10.29	10.29
MMH tank	1	9.8	5%	10.29	10.29
Pressurant Tank	2	5.4	5%	5.67	11.34
Branch Valves	2	0.48	5%	0.50	1.01
Fill an Drain Valve Single Seat	9	0.06	5%	0.06	0.57
Fill and Drain Valve Double Seat	4	0.09	5%	0.09	0.38
Pyro Valve	13	0.16	5%	0.17	2.18
Check Valve	4	0.1	5%	0.11	0.42
Pressure Sensor	6	0.27	5%	0.28	1.70
Pressure Regulator	1	1.25	5%	1.31	1.31
Propellant Filter	1	0.1	5%	0.11	0.11
Propellant Filter	2	0.43	5%	0.45	0.90
Tubing Set	1	4.5	20%	5.40	5.40
Total Subsystem Mass [kg]					53.84

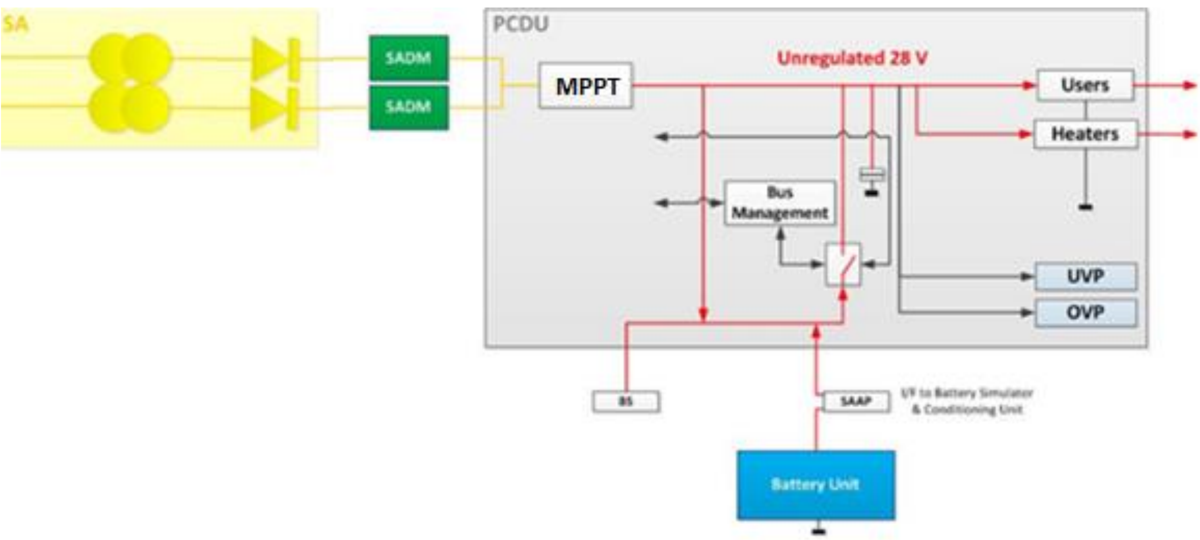
6.5 Electrical Architecture

6.5.1 Power

6.5.1.1 EPS Design Description

The EPS is required to provide electrical power during the whole mission without any degradation of the main and secondary power lines to all connected loads during all mission phases. The system will be single point failure tolerant and will be based on a 28 V unregulated main bus.

The EPS overview describes how units are connected together.



EPS is composed of:

- Solar Array,
- Solar Array Drive Mechanism,
- Power Conditioning and Distribution Unit,
- Battery unit,

The electrical power system shall be dimensioned to supply all active satellite subsystems throughout the entire mission without any degradation of mission performance.

6.5.1.2 Functional Description

6.5.1.2.1 Functional architecture

During all the mission phases with the Solar Arrays deployed, electrical power is provided by the two Solar Arrays, composed of 2 deployable panels each, during all mission modes without battery support. Battery supply is only foreseen during the separation slot where the arrays are not deployed yet, or contingency scenarios. No eclipse periods are foreseen.

The SADM provides one degree of freedom to optimize solar array sun pointing, therefore optimizing solar array power input to the S/C.

The battery supplies the S/C during separation and contingency scenarios during nominal mission phases.

The PCDU distributes power to S/C subsystems with an unregulated 28 V bus through:

- R-LCLs/LCLs for the platform and payload units,
- LCL/switches for heaters,

The PCDU integrates a MPPT power transfer for the solar array power that would then allow a maximum power available on the main bus to provide requested power by the S/C.

All main power circuits are completely redundant, and the whole subsystem is capable of activating all the protections required to guarantee that main bus can survive a single failure inside or outside of the EPS.

In addition, protections are implemented against battery under-voltage caused by excessive battery discharge, and battery overcharge. If the battery is excessively discharged down to a predefined level the battery will be disconnected by an isolating switch inside the PCDU.

The EPS functions are hardware controlled with default values. TM/TC and parameter changes are performed by on-board computer software intervention via the 1553 MIL-bus.

6.5.1.3 EPS Design Justification

The following requirements have been considered for the dimensioning of the power subsystem:

Req ID	Description
SC-POW-10	Power General The Hera Spacecraft electrical power sub-system shall be designed, developed and verified in accordance to ECSS standards as specified in [SD7] and [SD9], unless otherwise specified in this document or tailoring agreed in writing.
SC-POW-20	Power Function The Hera Spacecraft power sub-system shall generate, store, condition, protect and distribute the electrical power needed by all platform units and payloads throughout all mission phases, including on-ground activities and contingencies.
SC-POW-30	Power Sources The power for the Hera Spacecraft, all platform units and payloads, shall be supplied through the use of solar cell arrays and secondary batteries.
SC-POW-50	Solar Array Worst Case Scenario The Hera Spacecraft power sub-system shall be sized considering the solar array worst case conditions at the end of life.
SC-POW-60	Battery Worst Case Conditions The Hera Spacecraft power sub-system shall be sized considering the battery worst case conditions at the end of life.
SC-POW-70	Power Failure Protections The Hera Spacecraft power sub-system shall be equipped with suitable protection devices preventing failure propagation from the power bus to any other user of the bus, both during on-ground activities, launch and in-orbit operations.
SC-POW-80	Solar Arrays Monitoring It shall be possible to monitor the current generated by each section of the Hera Spacecraft solar arrays.
SC-POW-90	Battery Monitoring It shall be possible to monitor the voltage, and the charge and discharge currents of each module of the Hera Spacecraft battery.

Req ID	Description
SC-POW-100	Start-Up Power Source The Hera Spacecraft power sub-system shall be able to start up from any of the power source (battery or solar array). Note: The rational is to cover not only in-orbit operations but also on-ground activities e.g. testing of partial spacecraft assembly during the AIV/AIT campaign.
SC-POW-110	Restart Capability The Hera Spacecraft power sub-system shall be capable of restarting automatically after the occurrence of a power interruption.
SC-POW-120	Power Switch-Off with Integrated Battery The Hera Spacecraft power sub-system shall provide the possibility to disconnect the power sources from the main bus and the external power sources at spacecraft skin level, with batteries being integrated, fully charged and connected, during the on-ground activities.
SC-POW-130	Battery Interface The Hera Spacecraft power sub-system shall allow the possibility to recharge the battery during on-ground activities, including while the spacecraft is integrated with the launcher.

The general EPS performances and functions given by the above requirements are currently covered by the elements defined hereunder:

EPS Element	Design Summary	Supplier
Solar array	2 wing with 2 solar panels each (4.25 m² and 18.3 kg per wing including HDRMs and deployment elements) 3G30C cells in series, TBD parallel strings	TBD
Battery	8S4P 18650 NL (Moli ICR-J) battery (incl. redundant string)	ABSL
PCDU	18 modules, based on PROBA-Next design 28 V unregulated bus, MPPT	Qinetiq
SADM	2 x Septa-31 class or similar	RUAG

6.5.1.4 Power lines allocation

The PCDU provides R-LCL/LCL protected lines to the consumers. No fuses are used for the distribution sections.

PCDU Design Description and Justification

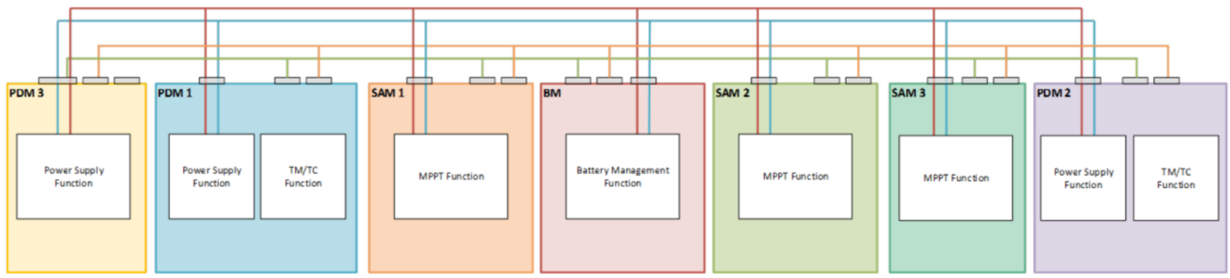
6.5.2.1 PCDU Design Description

The PCDU is based on a 28V unregulated bus. Its technical specification requirements are mainly based on ECSS-E-ST-20C requirements for BUS quality. According to ECSS-E-ST-
6.5.2 20C - Electrical and electronic, chapter 5.7.2 g. 2. , 28 V may only carry power up to 1.5 kW.

The number of module would be 7 modules (3xSolar Array Module, 3x Power Distribution Module and 1x Battery Module)

The key features of the PROBA-Next PCDU are:

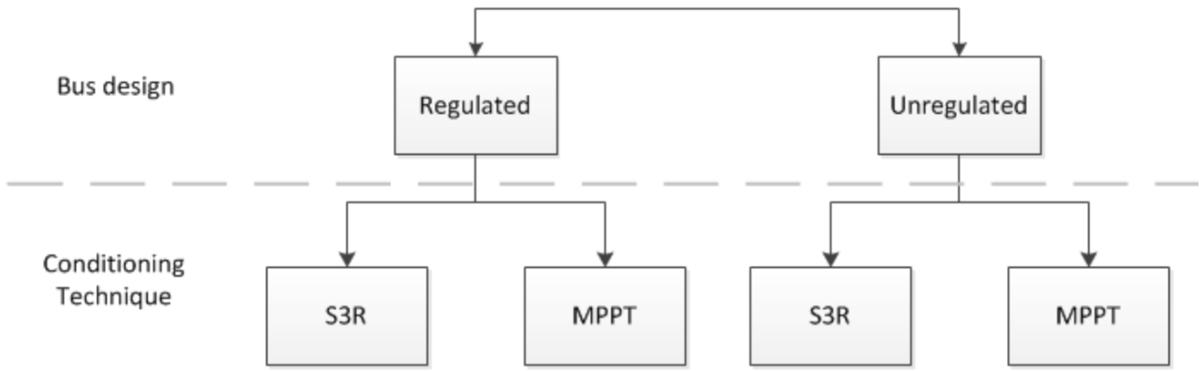
- Battery Module
  - Battery charge/discharge regulation (triple redundant)
  - ~15A maximum charge/discharge current
  - 28-32V bus (24-34V nominal bus voltage)
- Solar Array Module
  - Use of a potential off-the-shelf solution (TBC)
- Power Distribution Module with TMTC interface + redundant OBC
  - TMTC interface towards nominal + redundant OBC
  - 9 LCL protected power outputs (up to 4A per output)
  - 10 LCL protected power outputs (up to TBD A per output)



6.5.2.2 PCDU Design Justification (trade-off)

In the selection of the PCDU electrical architecture, every option comprising a main bus of 50V. According to ECSS-E-ST-20C - Electrical and electronic, chapter 5.7.2 g. 2. , 28 V may carry power up to 1.5 kW. This hard limit is far above the estimated power consumption for the mission, currently in the range of 600 W.

Regarding the unit architecture, different solutions are proposed:



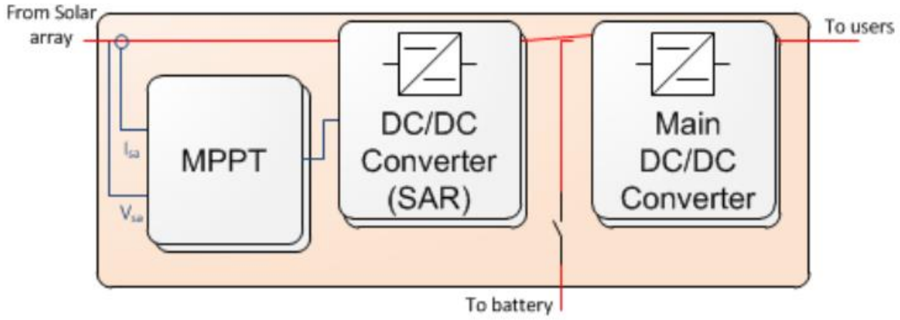
6.5.2.2.1 Regulated S3R

The S3R conditioning technique is constituted by a Shunt regulator connecting directly the solar array to the main bus through diodes. Each solar array section is connected to one shunt / diode pair. The bus regulation is performed by shunting not needed sections to the ground, during sunlight, and by the battery discharged regulator DC/DC converter, during eclipse. The battery charge is controlled by the Battery Charge Regulator in current mode.

The number of BCR and BDR converters are a function of the power needed from the battery. The number of shunts is related to the amount of solar array sections needed.

6.5.2.2.2 Regulated MPPT

The regulated MPPT option has following generic architecture:



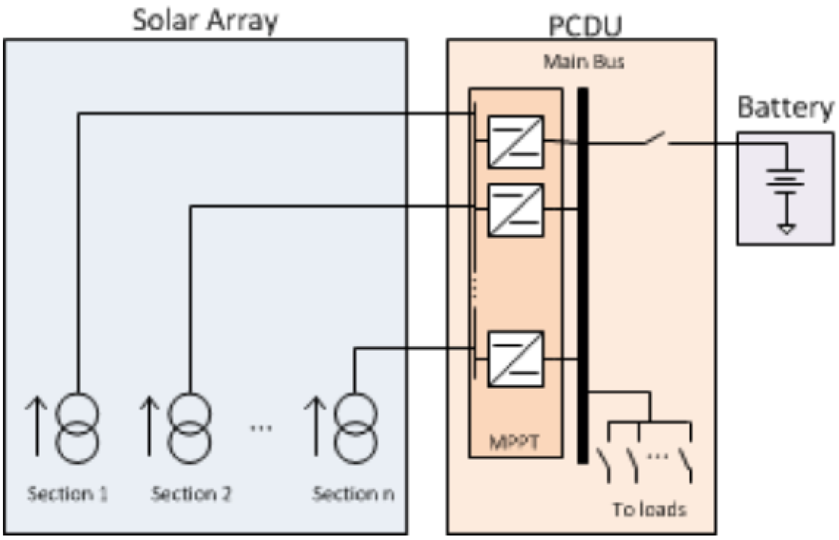
While being possible to implement it is comprised a very complex design. In total there are 2 different DC/DC converters, the Solar Array regulator (SAR) with a Maximum Power Point Tracking (MPPT) algorithm and the main bus regulator. This solution is not normally advisable from the suppliers, since it suffers from high technical challenges due to its complexity, mass and difficulty to manage two DC/DC converters in series.

6.5.2.2.3 Unregulated S3R

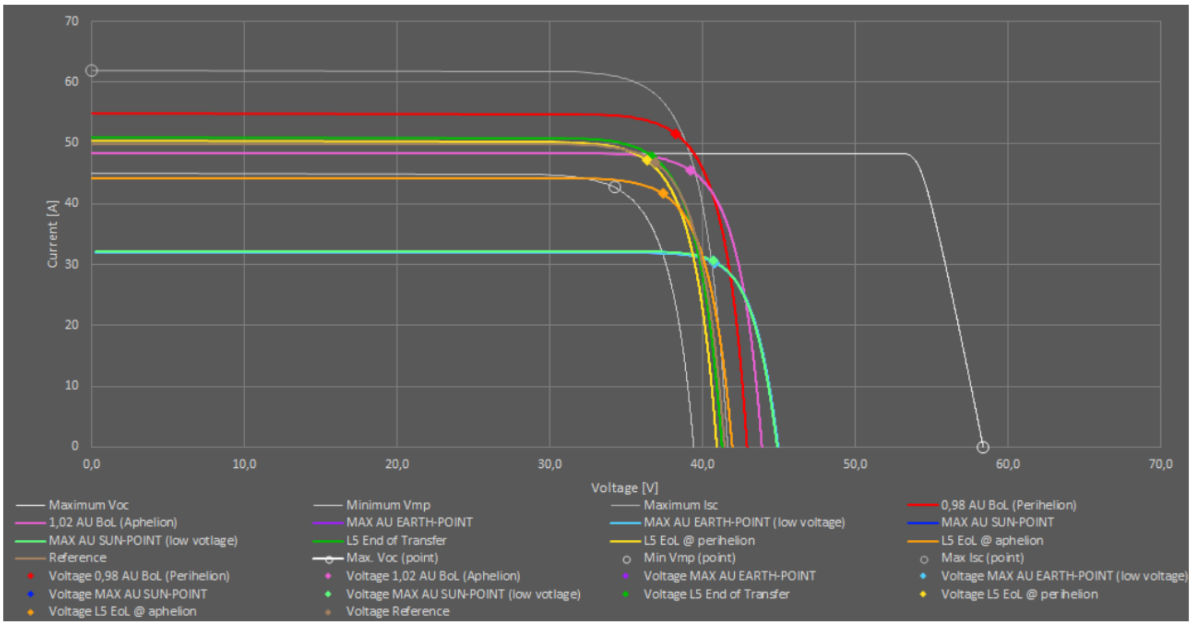
This option is used in OHB programs such as EnMap and SARah. It is a simple solution, with no DC/DC converters, low dissipation, low mass and cost. However, for the generic Hera platform it leads to oversized solar arrays because of the fact that the solar array voltage depends on the state of charge of the battery. For example, for lower state of charge, the battery voltage will be lower, hence the generated power from the solar array also lower. This design leads to when the satellite is in need of more power, in this case to charge the battery, coincides with when less power can be generated from the solar array

6.5.2.2.4 Unregulated MPPT

The MPPT technique connects the solar array to the main bus through a DC/DC converter denominated Solar Array Regulator (SAR). This DC/DC converter regulates the voltage of the solar array so that the power generated is the same as the power demand, thus maximizing the power conversion efficiency. All solar array strings are connected in parallel and then connected to the SARs. The number of SARs in parallel is set by the total power needed from the solar array.



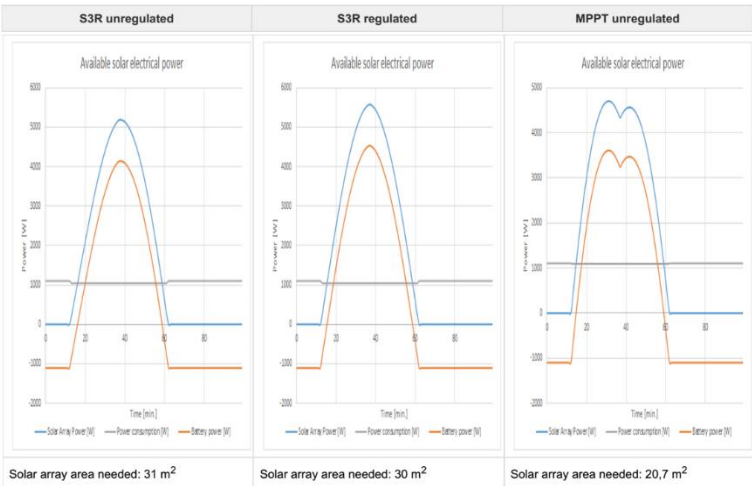
The solar array is operated either on the right or left side of the MPP. However, for stability reasons the right side is preferred, since the voltage remains approximately constant. The chart below shows the maximum power points for the Lagrange Space Weather mission example for the MPPT design and trade-off being used for this architecture for HERA.



6.5.2.2.5 Trade-off parameters

TCS design	It is true that the regulated bus simplifies the design process of the Thermal Control Subsystem. This is due to the fact that the heaters are sized for only one voltage level. With unregulated bus there are two design cases that need to be taken into account. The heater has to be design to provide the needed heater power for the lower voltage, while at maximum voltage the heat density has to be lower than the design limits for the heater and the glue fixing it. This is more complicated the bigger the cap between the minimum and maximum voltages, as the difference between the minimum and maximum heater powers increase even further. Lessons learned from past projects suggest that delays are created when the voltage limits change throughout the project development. These changes cause sometimes a complete redesign of the entire TCS. While this problem existed in the past, currently the Thermal Engineers at OHB have learned from the experience and have identified what design adaptations are required when working with unregulated bus. Programs like SARah, OptSat and ExoMars2020 have all used unregulated bus without any setback directly associated with the interaction of the TCS with unregulated bus. Also the unregulated bus voltage range is directly associated with the battery voltage range. Projects in the past, such as EnMap, used ABSL HCM and HC battery cells that have a wide voltage

	range. The battery cells currently being used for new designs, i.e. SAFT VES16 and ABSL NL, are built with a different chemistry than the HCM and HC cells. These cells have a flatter voltage vs. charge curve, which causes a smaller voltage range, and thus an even smaller difference in heater power between minimum and maximum. The following chart illustrates the differences between the obsolete HCM cell and the modern cells: Also the unregulated bus voltage range is directly associated with the battery voltage range. Projects in the past, such as EnMap, used ABSL HCM and HC battery cells that have a wide voltage range. The battery cells currently being used for new designs, i.e. SAFT VES16 and ABSL NL, are built with a different chemistry than the HCM and HC cells. These cells have a flatter voltage vs. charge curve, which causes a smaller voltage range, and thus an even smaller difference in heater power between minimum and maximum.
Power dissipation during sunlight	The comparative tense refers to the architecture regulated with S3R, against unregulated MPPT. It is true that MPPT has higher power dissipation during sunlight than S3R, because of the Solar Array Regulator.
Unit Mass	The values presented in the table above show that in order to provide a peak power much higher than the average, the MPPT is a preferred solution since with S3R the BDR DC/DC converter has to be sized to supply the peak power, where as in the MPPT the battery supplies the peak.
Circuit complexity	Having two different DC/DC converters (BCR and BDR) and the Shunt regulator leads to more circuit complexity than just the main DC/DC converter with the MPPT algorithm.
Dissipation during eclipse	Dissipation during eclipse is higher with S3R solution because of the BDR
Peak power sizing	In order to reach the capability to provide peak power of 4 kW, if the solar array is not designed for this peak the S3R regulated solution requires much higher mass because this power has to be passed through the BDR from the battery. This generates high power dissipation and also the DC/DC converter has to be sized for the peak power. While on the unregulated solution, the power is directly supplied to the bus from the battery without any DC/DC converter in between.
Bus noise	The Solar Array Regulator in the MPPT is a buck converter when considering the Solar Array as the input. However, during voltage control mode the regulation is done at the solar array side, hence the converter behaves like a boost converter. Boost converters have typically higher noise at the output and lower noise at the input, hence lower noise is generated at the main bus.

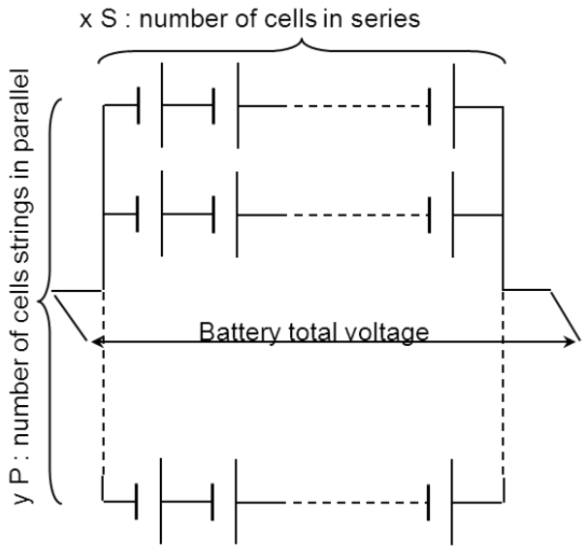
	For the regulated bus architecture, the current technology used for the BDR is a boost converter with the battery on the input, which causes the main bus to be the output and hence more noise. There is also the low frequency noise of the S3R (DC/DC Converters) which is difficult to filter and requires a huge capacitor bank.
Solar array size	S3R solutions lead to bigger solar array sizes. This is due to the fact that the solar array voltage is not at the maximum power point and the excess power is shunted away. The difference becomes more predominant at lower solar array temperatures, because at this condition the voltage generated by the solar array is higher than the S3R regulation voltage. Incidentally the lower solar array temperature, thus more power wasted in shunting, coincides with when the satellite needs more power to charge the battery, i.e., when the spacecraft is coming out of eclipse. This is especially true when designing a spacecraft with a non-sun-pointing solar array, such as Earth Explorer 9, and non-sun-synchronous orbit, like Sentinel 9. The angle between the solar array normal and the sun vector causes a lower solar array temperature, making the voltage higher, thus decreasing the efficiency of the S3R solution. This is illustrated in the following charts.  The S3R solution leads to considerably higher solar array area, up to 50% for unregulated and 45% for regulated.

Based on the above given comparatives, **the MPPT architecture provides more advantages for HERA over the S3R**.

Battery Design Description and Justification

Based on the current power budget, bus and SA architecture, an 8S4P battery design is selected. The number of strings is based on the mission required energy, the maximum DoD, one string failure requirement, mission losses and cell nameplate energy. Since, nominally, only one major discharge will occur in flight and a minimal number of battery cycles is foreseen
6.5.3 during the ground tests, a maximum DoD of 80% is allowed. It is possible to further discharge the battery until the H/W UVA is triggered in case of any contingencies during the LEOP phase.

The selected battery uses the Li-Ion rechargeable technology and an S-P architecture (Design 1). They are a smaller version of the ABSL's 8S52P 18650 NL battery, which have been already used for other OHB projects. A trade with the SAFT VES has been also conducted.



Driving scenario: LEOP/Transfer 330W, 30 minute → Required W/h: 165 Wh @ 28V

Design 1 -Custom ABSL 18650NL		
Variable	**Number**	**Unit**
Nominal capacity	2.4	Ah
Nominal cell voltage	3.7	V
Required cells series	8	
Parallel strings	4	
Total capacity watt hours	**268.8**	**Wh**
Usable capacity watt hours	**215.04**	**Wh**
Total battery mass	**1.86**	**kg**
Energy density	144.8	Wh/kg
Total battery width	0.089	m
Total battery length	0.089	m
Total battery height	0.195	m

Design 2 - Based on Saft VES 16		
Variable	**Number**	**Unit**
Cell capacity	4.5	Ah
Mean voltage per cell	3.6	V
Needed cells in series	8	
Needed strings in parellel	3	
Total capacity watt hours	**378**	**Wh**
Usable capacity watt hours	**302.4**	**Wh**
Total battery mass	**4.59**	**kg**
Energy density	82.3	Wh/kg
Total battery width	0.163	m
Total battery length	0.122	m
Total battery height	0.183	m

6.5.4 SADM Design Description and Justification

There are two Solar Array Drive Mechanisms used on the HERA spacecraft, one for each SA wing. The SADMs will be mounted on each side of the spacecraft, and will be independently controlled by software executed by the OBC via Milbus.

The SADM function is to:

- drive the solar array for optimal power input,
- provide the solar array reference positions,
- transfer power from the solar array to the spacecraft,
- transfer signals between the solar array and the spacecraft with double insulation.

The SADM assembly is composed of the following elements:

- Shaft and spherical bearing assembly,
- Motor and gearbox assembly,
- Power slip ring unit,
- Signal slip ring unit,
- Position sensors (potentiometer).

The SADM is powered using a two phase hybrid stepper motor with redundant windings giving 36000 stable (un-powered) positions of the output shaft over one revolution. For each power and signal transfer, all the brushes are redundant. The unit consists of: an actuator, a slip ring assembly (composed of a collector for power and signal transfer) and two potentiometers for position feedback.

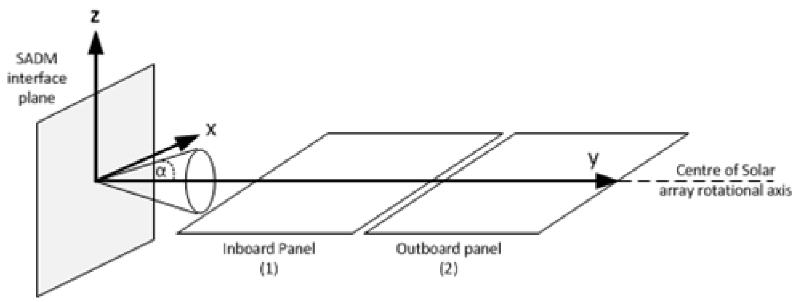
6.5.5 Solar Array Design Description and Justification

HERA is a three-axis stabilised spacecraft. The SA is characterised by:

- Two (2) deployable Solar Array Wings (SAW) each employing an extension arm and two (2) deployable panels. Each wing with 4.25 m² and 18.3 kg mass consider also in this figure the deployment mechanisms such as HDRMs and hinges per wing.

- Solar cells maximum voltage 4.1 V
- 4 electrical sections of 2.5A are foreseen per panel (TBD)
- The number of solar cell assemblies in series must optimized with respect to the degradation due to space radiation, the solar array temperature, the voltage drop of the harness (incl. blocking diodes) and the minimum Vmp requirement respectively (≥ 57V @Top, EOL).
- Six (TBD) Hold-Down brackets and redundant release mechanisms per wing
- Deployment mechanism
  - Root hinge
  - Two (TBD) panel hinges per wing (i.e. single hinge per hinge line)
  - Deployment synchronization system (TBD)
- Grounding of the panel and yoke structure via bleed resistors.
- Signal
  - ✓ 2 (TBD) temperature sensors on each outboard panel.
  - ✓ Deployment switches:

The figure below show one fully deployed wing:



The sizing of the SAW has been considered for the max power demand during nominal mission phases at asteroid rendezvous (1.8 AU). Due to this, once the SAW is fully deployed in LEOP, the power generated at SAW output is estimated to be around 3 kW, much higher than the consumption foreseen at this stage.

This shall not be an issue for the EPS, and PCDU in particular, since the MPPT shall ensure the power distributed along all the spacecraft consumers is only the necessary one (as per LCLs and relays activated). Therefore, the PCDU shall only work at the point of the consumption requested, not at the max power generation point of the solar arrays (as explained in 6.5.2.2.4).

Hence, no dedicated on-board operation shall be necessary to reduce the power generated at Solar Array output (i.e solar array de-pointing).

6.5.5.1 In-flight configuration and deployment

The HERA SA has 3 in-flight deployment configurations as specified:

- Stowed configuration,
- Partially deployed configuration (1 panel) (need to be confirmed),
- Fully deployed configuration (2 panels).

Following micro-switch signals are proposed to verify the deployment status: one for partial deployment latching, one for 180° latching of the outer panel and one for latching of the yoke hinge (indicating full deployment of wing).

6.5.5.1.1 Stowed Configuration

During launch and first transfer phase, the SA is stowed to the S/C sidewall with 6 (TBD) hold-down units. In this configuration the solar cell side of the outboard panel is facing towards space in S/C´s Y-direction.

6.5.5.1.2 Partially Deployed Configuration

During transfer orbit the outboard panel has deployed by 90° and its cell side is facing in –Z-direction of the S/C. In this configuration the SA is powering the transfer phase of the S/C. For this partially deployed configuration, 5 (TBD) primary hold-down units will be released. The remaining panel stack is kept stowed via the 6th, the secondary, hold-down unit, which is connected to the second outermost panel. The partial deployment is a free, i.e. after release an "uncontrolled", deployment of the outboard panel, motorized by springs in both panel hinges.

6.5.5.1.3 Full Deployed Configuration

The full deployment of the wing is achieved by releasing the secondary hold-down unit and is initialized by switching on the motor gear unit (MGU), which controls and constrains this final deployment. The MGU gears into the deployment via the free, rotary closed-cable-loop pulley system (CCL), which connects all hinge-lines of the SA and acts as a synchronisation gear to ensure that the wing deploys straight out from the S/C in a homogeneous motion. The deployment energy is provided by springs in each panel hinge. MGU acts as a slow-down unit during full deployment in order to minimise latching moments without the need to reduce
6.5.6 deployment torque margins. It stays activated until all hinges have latched.

Harness

The following requirements have been considered for the dimensioning of the power subsystem:

Req ID	Description
SC-HAR-10	Harness General The Hera Spacecraft harness shall be designed, developed and verified in accordance to ECSS standards as specified in [SD7], unless otherwise specified in this document or tailoring agreed in writing.

Req ID	Description
SC-HAR-20	Harness Function The Hera Spacecraft harness shall provide electrical connection between all electrical equipments, test connectors, safe and arm brackets and connectors, and umbilical connectors, and adequate distribution and separation of all: • Power supply lines • Analogue and digital signal lines • Command and actuation pulses and stimuli lines between all units
SC-HAR-40	Harness Used as Mechanical Support No piece of the Hera Spacecraft harness shall be used as a mechanical support.
SC-HAR-50	Incorrect Mating Prevention The possibility of incorrect mating of any Hera Spacecraft connectors shall be avoided by design.
SC-HAR-60	Harness Routing of Redundant Functions Wiring of Hera Spacecraft redundant systems, sub-systems, units and redundant functions shall be routed through separate connectors and wire bundles.
SC-HAR-70	Cross Strapping Cross strapping of any line should not be carried out in the Hera Spacecraft harness.
SC-HR-80	Safe and Arm Connectors Safe and arming plugs shall be provided for disabling of functions with hazardous, catastrophic or critical consequences

6.5.6.1 Cables

To reduce magnetic susceptibility and emission, twisted cable will be used (R-SC-3200 & R-SC-3210). Usage of twisted shielded cables according to ESCC 3901/019 (if atomic oxygen is a problem for the mission, the usage of cables according to ESCC 3901/018 has to be checked) (R-SC-2850).

6.5.6.2 Shielding

All HMUs overall shielded. Shielded with aluminium foil (2 wrappes with at least 50% overlap TBC). Additional grounding of the overall shielding each 20 cm (TBC)

The overall shield and the cable shields will be connected with a clamp band to the backshell to guarantee a good connection to the spacecraft structure.

6.5.6.3 Connectors and Backshells

Non-magnetic connectors, class TBD, according to ESCC 3401 will be used (R-SC-3210). Non-magnetic, EMC tight (TBC) backshells will be used (R-SC-3210).

Additional: as an option filter connectors could be used, but should be avoided

6.5.6.4 Routing

Cables falling into different EMC classes will be assembled to different (separate) cable bundles and connectors – as far as possible. All cable bundles will be routed as close as possible to the structure. Routing at the outside of the spacecraft will be avoided as far as possible.

If the routing at the outside of the spacecraft cannot be avoided additional protection has to be taken into account (TBC).

6.6 Avionics Architecture

The systems block diagram was already introduced in Figure 5-1. For convenience it is repeated here, as the justification of the avionics architecture is provided in the following.

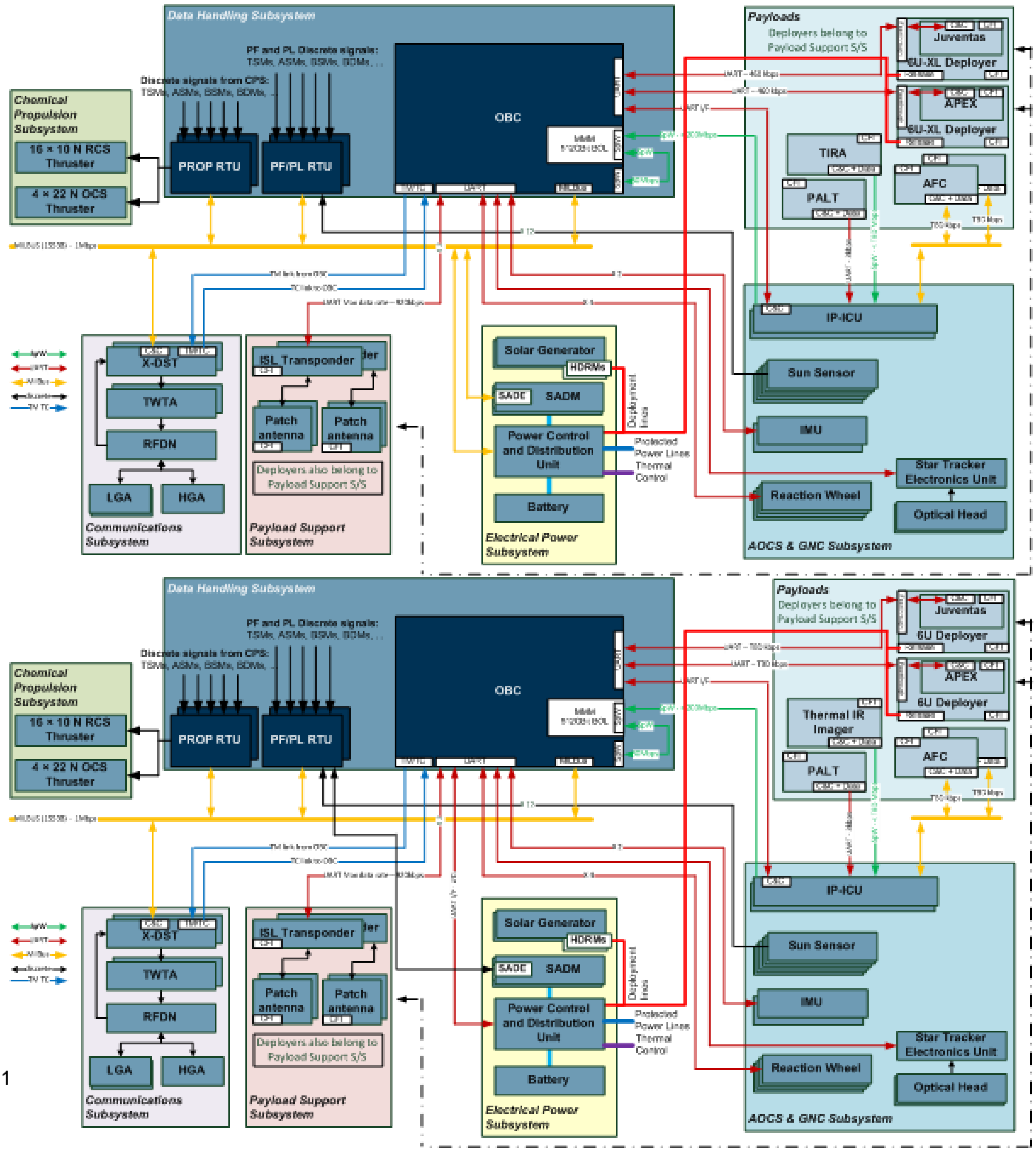


Figure 6-12: Systems Block Diagram (repeating Figure 5-1)

6.6.1 Trade-Offs

6.6.1.1 RTU Configuration

During Phase A, a single RTU configuration for both platform I/O and propulsion interfaces was baselined in the proposal. Nevertheless, considering the latest mass budget which can accept minor mass increase in favour of other benefits, we have evaluated another RTU configuration: 2 RTUs for platform I/O and propulsion interfaces respectively.

For this analysis, we focused on the following factors to determine the appropriate configuration: Cost, Mass, Schedule, and Reliability. The result of the trade-off is shown in Table 6-12.

Table 6-12 RTU Configuration Trade-Off

Criterions	2 RTUs	Single RTU	Justification
Cost	-	+	Cost can be increased for the 2- RTU configuration. Nevertheless, the increase is expected to be minor due to the shortened AIT activities.
Mass	-	+	The additional RTU increases the mass by a few kg, due to having additional housing.
AIT/Schedule	++	-	With 2 RTUs, AIT for the platform and the propulsion subsystem can be conducted in parallel. Design/Verification of RTUs themselves before joining AIT can also be performed in parallel, shortening and de-risking RTU development and AIT activities.
Reliability	++	-	With 2 RTUs dedicated to control RWs and Thrusters respectively, if one of the RTUs fails, the attitude can be controlled either with RWs or Thrusters single point failure can be avoided.

According to the trade-off above, although we have minor mass increase (and possibly slight cost increase) by having 2 RTUs, the increase of a few kg in mass is worth accepting considering increased reliability and de-risking/shortening AIT.

Thus we conclude that the favourable RTU configuration is the 2-RTU configuration as the baseline.

6.6.1.2 Image Processing

In this section, we discuss image processing architecture considering the processing load for autonomous navigation proposed for HERA, which involves following different techniques:

- Relative navigation (feature tracking) using AFC images, allowing determining the displacement of different relevant points selected between consecutive images.
- Centroid navigation using CHITY data (hyperspectral camera), is determining the centre of brightness of the image in order to determine the position of the asteroid in the FoV.
- Maximum correlation with a Lambertian sphere is an IP algorithm used to determine the position of the asteroid in the FoV of AFC camera.

A trade-off has been made in order to define the IP (Image Processing) architecture in terms of HW and interface implementation. The intension of this trade-off is to evaluate the possibility of re-using processing HW of other existing elements in the already foreseen architecture, and the necessity of a new dedicated processing component.

In the trade-off, the following three options have been considered:

- **Option 1**. OBC-only using both cores (run full GNC SW incl. IP) (pure SW option)
- **Option 2**. OBC single core and spare resources in the available RTG4-FPGA (if any)
- **Option 3**. Extra FPGA dedicated for IP + OBC using a single core

Refer to HERA-GMV-GNC-DDJ-0001 “IPM Architecture” for detail, including the analysis and calculation made for this evaluation. According to this analysis evaluating computational resources for GNC incl. image processing, we need to simply discard the first to options due to lack of resources necessary for HERA scenarios:

- **Option 1**. OBC-only using both cores (run full GNC SW incl. IP) (pure SW option)
  In this configuration, one of the cores will be devoted to the data-handling functions of the OBC while the second core would be devoted to the GNC. Considering the 1 Hz frequency for the GNC ASW and the feature tracking computational load, the resources are not enough. On top of the pure analysis of performances, there is no space-qualified multiprocessing OS (RTEMS-SMP or any other) which could allow safe operations using a multicore processor.
- **Option 2**. OBC single core and spare resources in the available RTG4-FPGA (if any)
  The performed analysis shows that feature tracking cannot be implemented purely in SW in a LEON 3 core for the HERA scenario, and it shall be process on an FPGA. However, the occupation required for IP is incompatible with the RTG4 available resources.

Thus we conclude that the baseline for IMP configuration is Option 3, a dedicated FPGA accommodating IP + OBC using a single core architecture.

The unit accommodating the dedicated image processing FPGA, together with additional function of interfacing with all the payloads excluding CubeSats, is called IP-ICU (Image Processing and Interface Control Unit).

Please note, the IP-ICU is presently the unit within the satellite baseline with the lowest TRL and thus highest associated risk. Within phase B2 an alternative configuration will be investigated as back-up, which reduces the functionality of the dedicated unit to optional tasks like feature tracking and use high TRL components for all mandatory functions. A latest decision milestone to select the final baseline will also be identified.

6.6.1.3 Mass Memory

Here in this section, the configuration of MMM in terms of hardware and interfaces is discussed. Notice that this configuration can be modified and updated as the payload data volume requirements are being finalized.

The following 3 options have been considered: A dedicated MMM component, which was the baseline during Phase A, an integrated MMM together with IP-ICU, and an integrated MMM together with OBC. The result of the trade-off is shown in Table 6-13.

Table 6-13 MMM Configuration Trade-Off

Criterions	Dedicated MMM	Integrated IP-ICU/MMM	Integrated OBC/MMM	Justification
Cost	-	+	++	Cost can be reduced for the integrated solutions. For OBC integrated solution, OBC processor can be used to control MMM, thus no dedicated processor for MMM.

Criterions	Dedicated MMM	Integrated IP-ICU/MMM	Integrated OBC/MMM	Justification
Mass	-	+	+	Integrated solutions reduce mass by avoiding additional housing.
Heritage	+	-	+	For integrated OBC/MMM, same approach already taken in the ADMPS for PROBA series.

The evaluation shows the integrated OBC/MMM configuration is the most favourable option to choose.

Compared to the dedicated MMM configuration, of which the cost and mass are expected to be high due to having a “dedicated” component, the integrated OBC/MMM configuration has a strong cost/mass benefits. Furthermore, the integrated OBC/MMM configuration does not require a processor dedicated for MMM (OBC can be used for controlling MMM), which results in further cost reduction.

Regarding heritage, we had the same OBC/MMO integrated approach already for ADMPS OBC of PROBA2, PROBAV, and PROBA3.

Thus, we conclude that the integrated OBC/MMM configuration to be the baseline of HERA.

6.6.1.4 GNC

The objective of this section is to summarize the present Guidance, Navigation and Control Design, focusing on the GNC and Image Processing (IP) algorithms design and the justification behind their selection for the HERA mission in the frame of the Phase B1 system study. Also a dedicated section to FDIR will be reported.

For details, please refer to [RD22].

6.6.1.4.1 GNC Strategy and Modes

The proposed strategy includes an incremental autonomy level, from a spacecraft manually flown during the interplanetary, approach and early characterization phase, up to a fully autonomous GNC (Level of autonomy E3) during the close fly-by.

The layout of the GNC modes is reported in Figure 6-13, where nominal transitions are indicated as continuous lines and FDI(R) transitions are indicated as dashed lines:

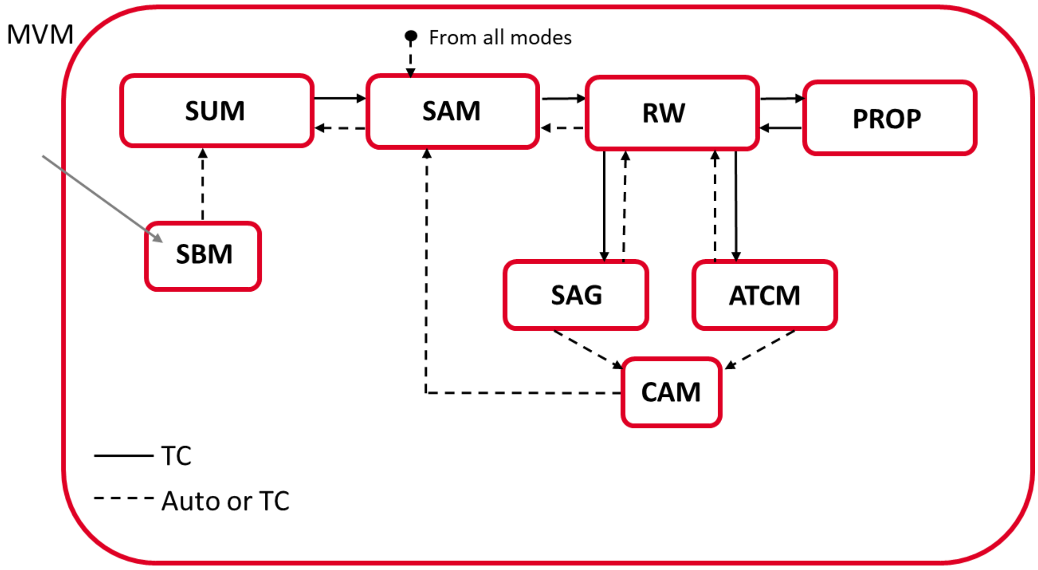


Figure 6-13: GNC modes

Figure 6-13 shows that there are 7 major GNC modes and a stand-by mode to be used as initialization/testing mode. A preliminary descriptions of the GNC modes and sub-modes is given below:

- **SBM (Stand-by GNC mode):**
  The GNC MVM enters this mode during S/C verification, before launch, when all the GNC has to stand-by
- **SUM (Survival GNC mode):**
  The GNC MVM enters this mode at the boot of the system (including LEOP) and every time the S/C mode is set to Survival. It is composed by three sub-modes:
  - DETUMB-SUM, to stop any rotation of the S/C (first sub-mode triggered);
  - SAC (Sun acquisition), to achieve Sun primary pointing (started after attitude navigation and guidance convergence). Attitude navigation in this mode consists only of IMU and Sun sensors;
  - SAP (Sun pointing), mode in which, thanks to the SADM, it will be possible to point the so-lar panel to the Sun and the HGA/MGA with the same offset angle between the Sun and the Earth. During LEOP this is not required (LGA will be almost omni-directional), but during the next mission phases this configuration together with a slow rotation around the Sun direction will allow for communicating with the Earth once every rotation (starting after attitude guidance convergence). Recent iterations at system level paved the way towards the possibility of using the LGA also during the advanced phases of the mission and not only in LEOP. If that is baselined this sub-mod of GNC will be deleted (i.e. no rotation will be required).
- **SAM (Safe GNC mode):**
  The GNC MVM enters this mode every time the SC mode goes to Safe. It is composed by three sub-modes:
  - RCS-EP is the Earth pointing mode using RCS
  - RW-EP is the Earth pointing mode using RW
  - WOL is the Wheels-of-loading mode, necessary when RW-EP is active.
- **RW (Operational Reaction wheels attitude based mode):**

The GNC MVM stays in this mode most of the mission life time. It is composed by two sub-modes:

- RW-AC where the attitude is controlled using RW
- WOL is the Wheels-of-loading mode.

- **PROP (GNC Propulsion mode):**
The GNC MVM enters this mode both when the SC mode is set to Operations and an RCS manoeuvre is planned by the ground station and when SC mode is set to Propulsion for main engine manoeuvre execution. Two main sub-modes are considered:
    - RCS-MAN where RCS manoeuvres are performed (as current baseline the attitude is controlled by RCS and RW are maintained with a constant speed – open loop translational control with pulse counting)
    - MP where Main Engine manoeuvres are performed (as current baseline the attitude is controlled by RCS and RW are maintained with a constant speed – closed loop translational control using accelerometers).
- **SAG (Semi-autonomous Attitude Guidance mode):**
This is part of the experimental modes/technology demonstrations that will allow the SC to achieve distances of the order of 10 km with respect to Didymos. Two main sub-modes are considered:
    - SAG where no manoeuvres are performed and the attitude is controlled using RW. A delta quaternion (limited in angular size) is added to the reference ground based attitude profile to maintain the target in the FoV. An autonomous translational navigation filter that uses centroiding measurements is required to maintain a good estimation of the SC position
    - WOL is the Wheels-of-loading mode.
- **ATCM (Autonomous Translational Control Manoeuvres):**
This is part of the experimental modes/technology demonstrations that will allow the SC to achieve distances smaller than 5 km with respect to Didymos. Two main sub-modes are considered:
    - RW-AC where no manoeuvres are performed and the attitude is controlled using RW. An autonomous translational navigation filter that uses feature tracking measurements is required to maintain a good estimation of the SC position
    - RCS-MAN where re-targeting autonomous manoeuvres are performed using RCS in order to reduce the pericenter of the SC hyperbolic arc, allowing a closer distance with respect to the asteroids.
    - WOL is the Wheels-of-loading mode. The goal is to avoid the wheels off-loading when performing this close operations and preliminary analysis shows that the pointing profile and the torque disturbances do not require an off-loading procedures more often than 3/5 days.
- **CAM (Collision Avoidance Mode):**
The GNC MVM enters this mode when the SC mode is set to CAM. Two main sub-modes are considered:
    - STABILIZED_CAM during which the SC is prepared to perform a collision avoidance manoeuvre and it is attitude stabilized (could be necessary if a tumbling is triggered by a thrust failure open).
    - CAM_EXE when the pre-planned manoeuvre is executed

6.6.1.4.2 GNC algorithms Design

For reasons of clarity the Attitude Determination and Control System (ADCS) will be separated from the Orbit Determination and Control System (ODCS). The Attitude Guidance is also considered as a separated block, not part of the ADCS.

6.6.1.4.2.1 Attitude Determination and Control System

The overall architecture related to the ADCS is illustrated with Figure 6-14. The figure shows that the ADCS functions are assumed to be executed as one single task taking raw unit-level measurements from the sensors and issuing unit-level commands to the actuators. The sensor measurements are assumed to have passed basic validity checks where this validity information also is available to the ADCS functionality.

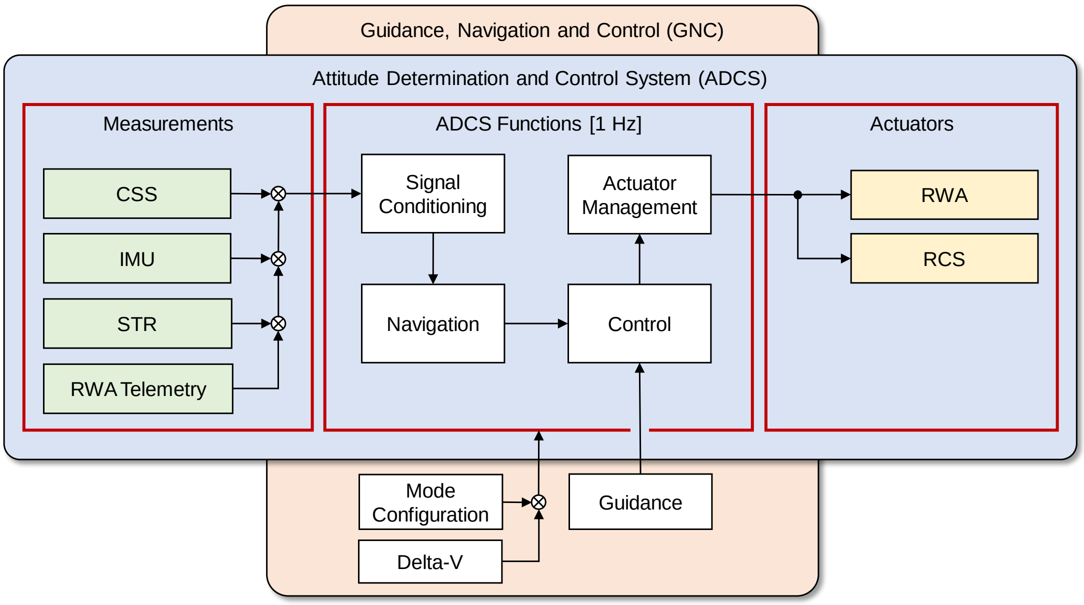


Figure 6-14: ADCS HW/SW Architecture

The ADCS functions are divided into Signal Conditioning, Navigation, Control, and Actuator Management functions. The Guidance module is external to the ADCS and is under GMV responsibility. It is also assumed that mode configuration and requested delta-V are provided from source external to the ADCS described in this section.

The different functional groups are described as follows:

- Signal Conditioning functions handle the raw unit-level measurement coming from the sensors and actuators and converts these into engineering values. The measurements are also transformed into spacecraft body coordinates.
- Navigation Functions include all estimation functions.
- Control Functions include all the attitude and momentum control functionality.
- Actuator Management functions handle the conversion from the desired actuation in spacecraft coordinates into individual raw value commands ready for distribution to the individual actuators.

The ADCS can be configured into a Survival Mode or a Nominal Mode. The survival mode handles the detumbling, sun acquisition and sun pointing. The Nominal Mode is a three-axis stabilised mode that supports the scientific operations as well as the system Safe Mode by choice of attitude guidance function.

6.6.1.4.2.2 Attitude Guidance

The main objective of the attitude guidance module is to provide the reference attitude profile, which is then tracked by the control loop.

During design phase, four different modes have been proposed for the attitude guidance module:

- Ground based attitude guidance
- Autonomous attitude guidance
- Semi-autonomous attitude guidance
- Survival attitude guidance

For the ground based and semi-autonomous attitude guidance, the reference attitude profile comes in form of attitude quaternion and angular rate of the spacecraft. For the survival attitude guidance, only the Sun-Spacecraft-Earth angle is provided.

6.6.1.4.2.3 Orbit Determination and Control System

The purpose of the ODCS system is to have an autonomous on-board estimation of the relative position of the S/C with respect to the binary asteroid system and to navigate safely during close proximity operations. The system is composed by a guidance (autonomous only during the very close fly-by for the retargeting manoeuvres) a navigation (vision based with centroiding, feature tracking and data fusion with the PALT) and a control system.

The relative position estimation is used both to correct the pointing during the ECP/DCP and to compute the corrections manoeuvres during the very close fly-by.

The ODCS consists of the following major components (see Figure 6-15):

- Image processing
- Measurement Management
- Translation navigation (state estimation)
- Translation guidance
- Translation control

Each of the components can be used in different configurations, depending on the mode being utilized. Details regarding the component configuration are provided in the subsequent chapter. The architecture is also flexible, enabling information processing from different sensors, depending on the need of the given mission phase.

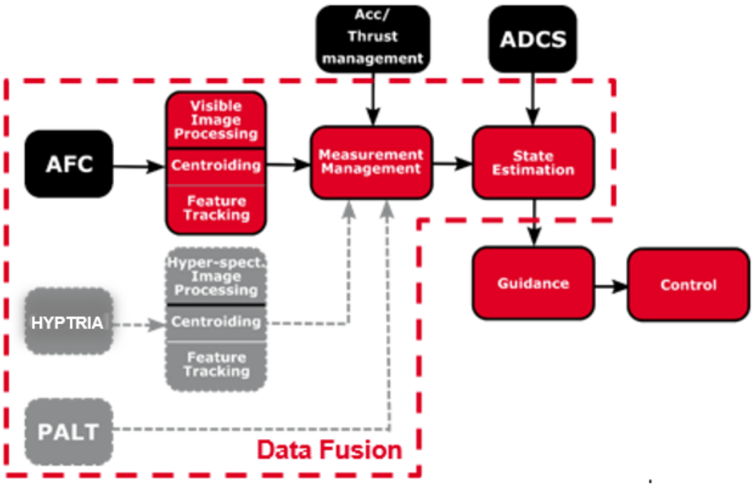


Figure 6-15: Preliminary ODCS architecture.

The required functionality drives the modes in which the ODCS shall operate:

- **ODCS-MODE-1**: Centroid-based navigation. As requested by T2.1 Autonomous visual based navigation for semi-autonomous attitude guidance [AD12]. The mode shall be used during ECP and DCP phases, at distances from 30 to 8 km, when full asteroid can be seen in a single image frame. Navigation in this mode is based on centroid measurements obtained from AFC. The purpose of this mode is to enable (semi-)autonomous attitude guidance. Translation guidance is based on the ground computed manoeuvres. This mode is based on the approach that has already been demonstrated during AIM Consolidation Phase.
- **ODCS-MODE-2**: Feature tracking navigation. As requested by T2.2 Autonomous visual based navigation for low altitude fly-bys [AD12]. This mode shall be used during low altitude fly-bys when asteroid is occupying entire FOV (below 8km). Features tracked in the AFC images are used as an input to the navigation filter to compute relative position with respect to the asteroid. This approach was already demonstrated during AIM Consolidation Phase, including the capability of switching pointing towards secondary and changing navigation reference frame.
- **ODCS-MODE-3**: Feature tracking with altimeter. As requested by T2.3 Autonomous visual based navigation for very low altitude guidance [AD12]. This mode shall be used during very low altitude fly-bys (below few hundreds of meters, the final range TBC). In addition to feature tracking used in ODCS-MODE-2, the data fusion with PALT will be investigated. The guidance and control for execution of corrective manoeuvres can be implemented. This mode will be based on the previous developments of AIM-CP and past experience of GMV based on other activities (e.g. NEOGNC, PHOOTPRINT). The mode can be used both by navigating with respect to the primary or secondary, depending on final mission scenario, the fly-by altitude and the altimeter range.
- **ODCS-MODE-4**: Disposal/landing mode. As requested by SY-PHA-80 [AD03] a controlled descent and landing shall be performed in order to be compliant with the SC disposal strategy. Therefore, another GNC mode should be dedicated to performing a controlled descent to the surface of the asteroid. It is proposed to extend the ODCS-MODE-3 in such a way to enable soft landing on the asteroid. The guidance and control for D&L trajectory will be studied. GMV has already a relevant experience with development of GNC algorithms for descent and landing on asteroids.

6.6.1.4.3 Image Processing algorithms Design

The Image Processing (IP) algorithms can be divided into three main classes depending on the mission phase:

- Maximum correlation with a Lambertian sphere for images in the visible range (baseline when the entire primary body can be seen in the images – according to the body size and the AFC characteristic this occurs when the range is higher than 8.5 km).
- CoB algorithm (baseline when the entire primary body can be seen in the images – according to the body size and the hyperspectral camera characteristic this occurs when the range is higher than 8.5 km).
- Feature tracking algorithms (baseline for close fly-by).

6.6.1.4.4 GNC-FDIR Architecture and Strategies

The algorithms of the GNC-FDIR have to be considered as complementary to the system FDIR and dedicated only to the inputs to the GNC subsystem and to the commanded actions. The recovery actions will be only suggested to system, which will take the decision having all the S/C related information fully available. Two GNC-FDIR chains are developed, a Nominal GNC-FDIR chain that is the baseline for the GNC subsystem and an experimental alternative chain GNC-FDIR that uses data-fusion in order to validate the GNC chain.

6.6.1.4.4.1 Nominal GNC-FDIR strategy

Different strategies are employed for each mission phase in order to adapt to the specific needs of each phase. The level of GNC-FDIR authority also depends on the mission phase, thus for the interplanetary phase a low level authority is required as the communication with ground is efficient and fast enough. For the close proximity operations a high level of authority is required due to long communication time and the need of a fast reaction. If a failure is present, the GNC-FDIR will identify the failure and suggest the recovery at system level.

6.6.1.4.4.2 Alternative GNC-FDIR Data Fusion Strategies

An alternative high autonomy GNC-FDIR chain using data fusion techniques will be designed in order to independently validate the GNC nominal chain (to verify the GNC performances) and to ensure operations' safety, which involves assessing the collision risk, and computing and executing a collision avoidance manoeuvre (CAM) if necessary.

For the HERA mission, the alternative chain will be based on the data fusion of information received from both nominal and non-nominal sensors, such as the AFC, the HYPTIRA, and the PALT.

6.6.1.4.4.3 FDIR CAM strategies

The objective of the Collision-Avoidance Manoeuvre (CAM) is to ensure that, when activated in the presence of a failure, the spacecraft exits the sphere of influence of the system, minimizing the collision risk with any of the asteroids.

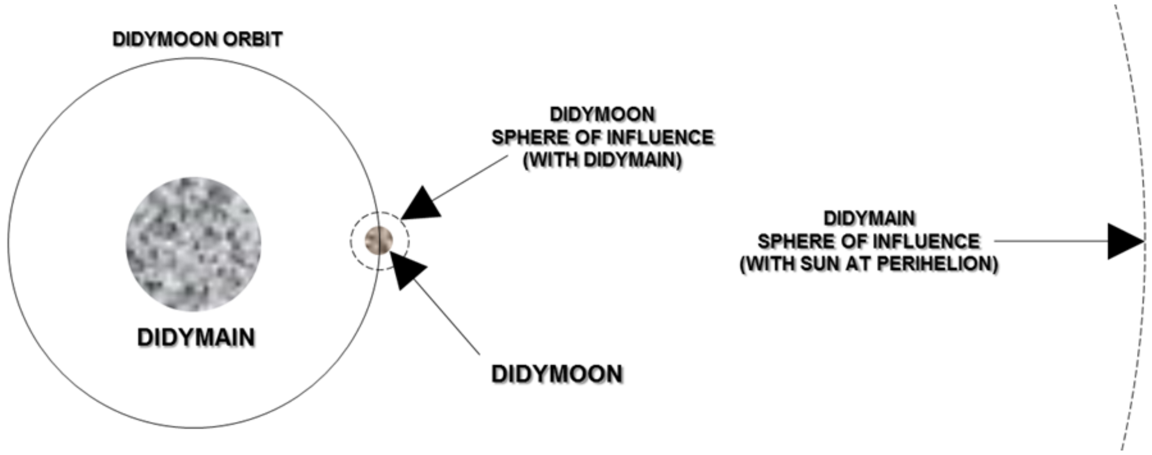


Figure 6-16: Visual approximation of the Didymos spheres of influence

6.6.1.4.5 GNC-FDIR algorithms Design

The pyramidal level of the GNC-FDIR level are described below.

6.6.1.4.5.1 Level 1: Unit/local level

The unit level consists of four detectors: frozen input detector, discontinuity detector, sensor total malfunction monitor, the RW constant output monitor and RW friction loss detector. At this level, if a fault is detected, it is forwarded to the system FDIR that shall take the reconfiguration decision and command it if necessary.

6.6.1.4.5.2 Level 2: Application level

In this section the GNC-FDIR level 2 algorithms are described in detail. The functions of GNC-FDIR level 2 breaks down in the following components: *thruster consistency check*, *reaction wheel consistency check*, *navigation filter covariance monitor* and *time, navigation solution consistency monitor, attitude determination consistency check, attitude global monitor* and *torque request check.* If a fault is signalled by one of the monitors, the fault is forwarded to the system FDIR that shall take the reconfiguration decision.

6.6.1.4.5.3 Level 3: Sub-system level

During close proximity operations phase the GNC FDIR will estimate the collision risk of the S/C according to its current state. The collision risk is computed by comparing the difference of the current estimated state with the reference state. When this difference, in terms of both position and velocity is greater than a predefined threshold, the collision assessment flag is raised (for more details about CAM triggering see Section 6.6.1.4.4.3.

6.6.1.4.5.4 Level 4: Recovery level

The recovery level of the system is part of the GNC-FDIR recovery action. The recovery can be done autonomously by S/C (decided at *system* level) or by *ground* intervention. At GNC-FDIR level, the recovery (CAM action) is suggested at system level FDIR where the decision shall be taken due to higher overall system visibility.

The GNC-FDIR level 4 main function is to trigger the CAM mode, described in section 6.6.1.4.4.3, when a collision assessment flag is raised at level 3. Basically level 4 contains a pre-loaded lookup table. The table contains the mission phase, a time vector and a specific set of inertial delta-Vs for each moment of time, which ensures it will take HERA in a safe trajectory.

6.6.1.4.6 Conclusions

In this section the Technology requirements coming from the [AD01] are analysed. It has to be taken into account that MIL, SIL, PIL and HIL have been cross validated. Therefore, if a test in this document works as MIL, it is demonstrated that we are compliant with the requirements in SIL, PIL and HIL as well.

In the tests shown in the following subsections the worst case scenario have been chosen. This way, if the test shows that the requirement is fulfilled it is proved that is fulfilled in any other scenario.

6.6.1.4.6.1 T2.1.

6.6.1.4.6.1.1 ECP

6.6.1.4.6.1.1.1 Position knowledge error req.

The worst case arc is the 8th one. For this arc the requirement plots are as follows:

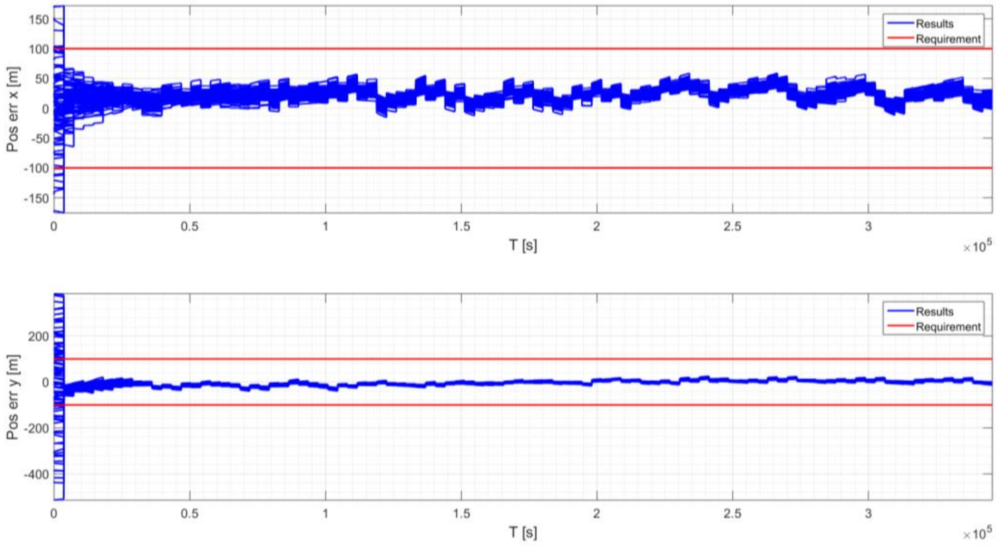


Figure 6-17 Position knowledge error req. for 8th ECP arc – only the XY axes of the camera plane are shown as they are the ones applicable for the semi-autonomous AG

As it could be seen in the presented plot, both x and y position knowledge error are lower than 100 m relative to the centre-of-mass of Didymos during more than 95% of the time. Being compliant with the requirement.

6.6.1.4.6.1.1.2 Attitude Pointing Error req.

The worst case arc is the 8th one. For this arc the requirement plots are as follows:

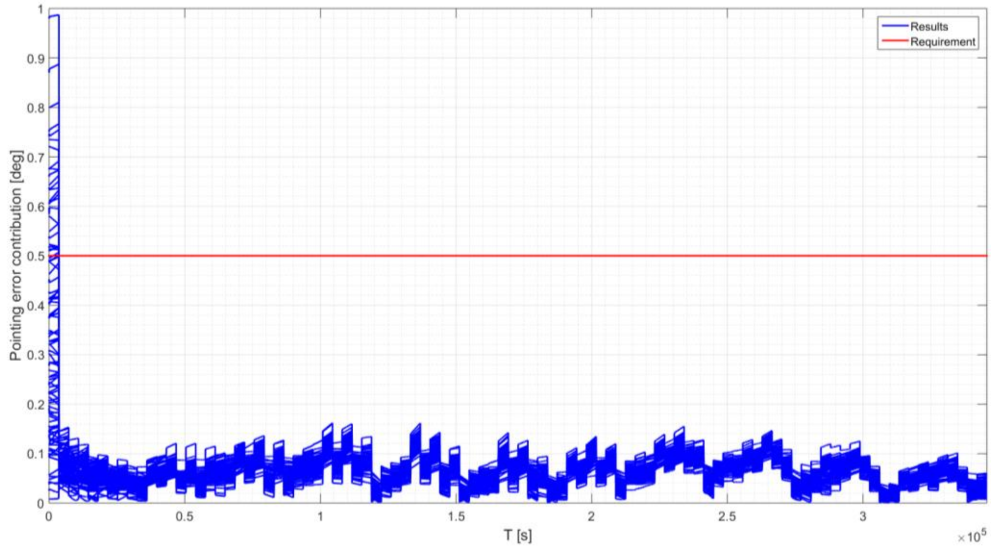


Figure 6-18 Attitude Pointing Error req. for 8th ECP arc.

As it could be seen in the presented plot the Contribution of attitude guidance to APE is lower than 0.5 deg during more than the 95% of the time. Being compliant with the requirement.

6.6.1.4.6.1.2 DCP12

6.6.1.4.6.1.2.1 Position knowledge error req.

The worst case arc is the 8th one. For this arc the requirement plots are as follows:

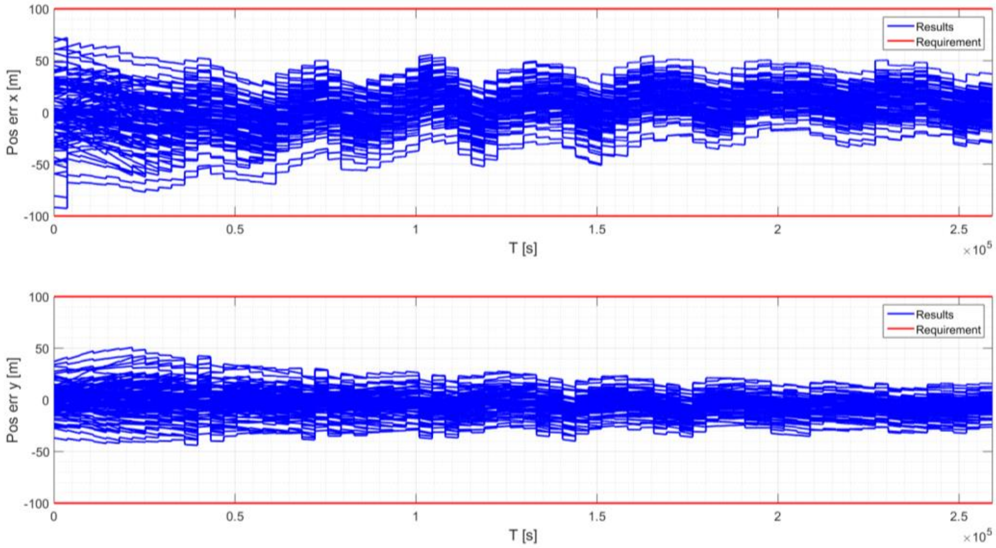


Figure 6-19 Position knowledge error req. for 8th DCP12 arc - only the XY axes of the camera plane are shown as they are the ones applicable for the semi-autonomous AG.

As it could be seen in the presented plot, both x and y position knowledge error are lower than 100 m relative to the centre-of-mass of Didymos during 100% of the time. Being compliant with the requirement.

6.6.1.4.6.1.2.2 Attitude Pointing Error req.

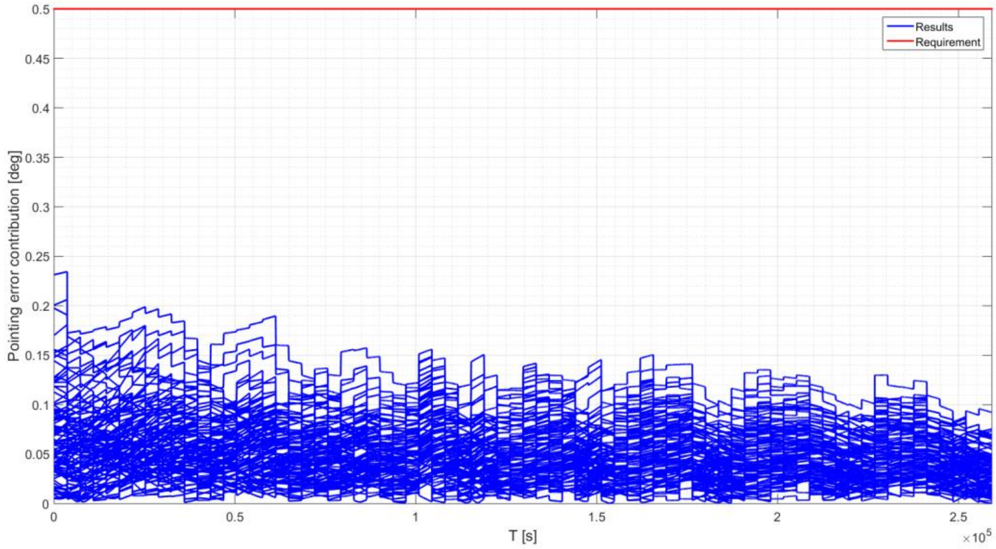


Figure 6-20 Attitude Pointing Error req for 8th arc.

As it could be seen in the presented plot the Contribution of attitude guidance to APE is lower than 0.5 deg during 100% of the time. Being compliant with the requirement.

6.6.1.4.6.2 T2.2.

6.6.1.4.6.2.1 Fly-by approach.

6.6.1.4.6.2.1.1 Position knowledge error req.

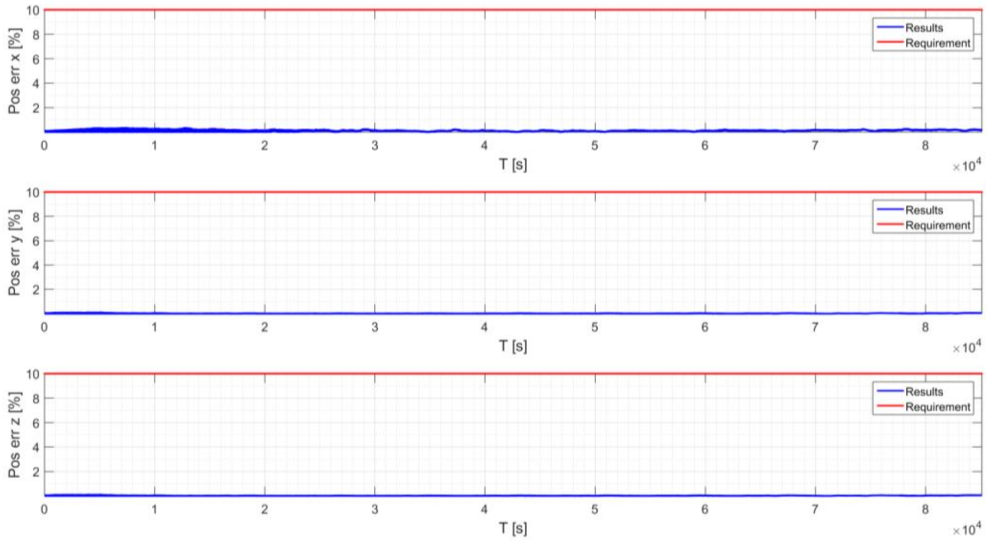


Figure 6-21 Position knowledge error req. for Fly-by aproach.

As it could be seen in the presented plot, all x, y and z position knowledge errors are lower than 10% relative distance to the centre-of-mass of the target during 100% of the time. Being compliant with the requirement.

6.6.1.4.6.2.1.2 Attitude Pointing Error req.

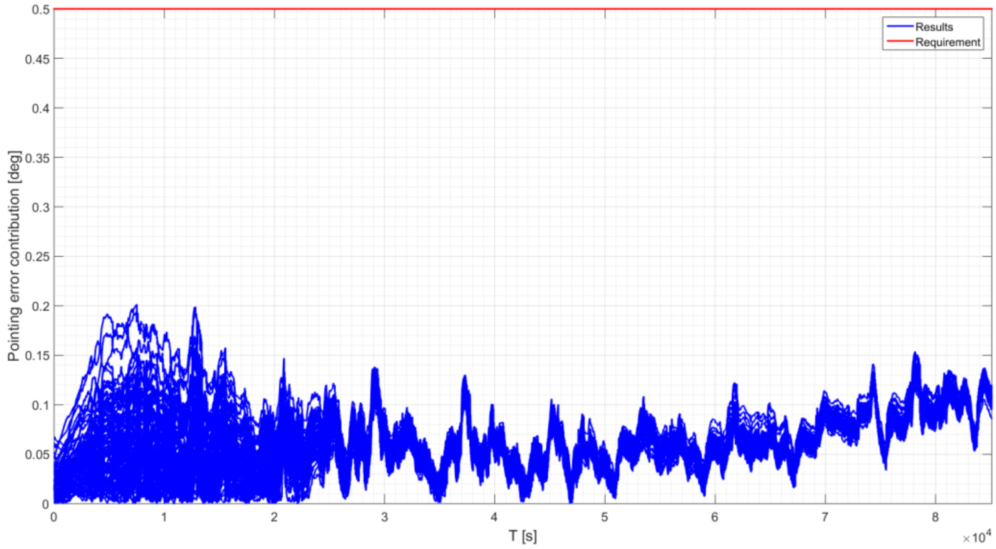


Figure 6-22 Attitude Pointing Error req. for Fly-by approach.

As it could be seen in the presented plot the Contribution of attitude guidance to APE is lower than 0.5 deg during 100% of the time. Being compliant with the requirement.

6.6.1.4.6.2.2 DCP3

6.6.1.4.6.2.2.1 Position knowledge error req.

Excluding the central part, which belongs to pure propagation (centroiding IP cannot be used being the body bigger than the FoV), both x and y position knowledge errors are lower than 10% of the distance relative to the centre-of-mass of Didymos during 100% of the time (to be compliant also in range PALT is necessary for such low altitudes).

If it is desired to be compliant with the requirement without excluding the central part, the Feature Tracking can be activated in this part to avoid pure propagation, as it could be seen in the first part of the plot of 6.6.1.4.6.3.1.3 (up to $T = 3.95 \times 10^4$ sec), when the requirement was respected (the last part of the 6.6.1.4.6.3.1.3 plot, due to the proximity of the target asteroid, is not representative for this scenario). Another possibility would be to use the thermal camera that has a wider FoV and allows to use the centroiding algorithm also in the central part of the arc. In this case the performance would be similar to the one reported in Section 6.6.1.4.6.1.2.

6.6.1.4.6.2.2.2 Attitude Pointing Error req

Excluding the central part, which belongs to pure propagation (centroiding IP cannot be used being the body bigger than the FoV), the contribution of attitude guidance to APE is lower than 0.5 deg during 100% of the time.

If it is desired to be compliant with the requirement without excluding the central part, the Feature Tracking can be activated in this part to avoid pure propagation, as it could be seen in the first part of the plot of 6.6.1.4.6.3.1.3 (up to $T = 3.95 \times 10^4$ sec), when the requirement was respected (the last part of the 6.6.1.4.6.3.1.3 plot, due to the proximity of the target asteroid, is not representative for this scenario). Another possibility would be to use the thermal camera that has a wider FoV and allows to use the centroiding algorithm also in the central part of the arc. In this case the performance would be similar to the one reported in Section 6.6.1.4.6.1.2.

6.6.1.4.6.3 T2.3.

6.6.1.4.6.3.1 Close fly-by.

It is important to remark that in order to navigate the close fly-by using feature tracking based on-board navigation, the Altimeter is required. The ground based navigation analysis demonstrated that it is not possible to initialize the autonomous GNC with the required precision for this phase, so the strategy has been updated and a *bridging phase* (named fly-by approach in this document) successfully takes care of the authority transfer between ground and on-board.

It has been demonstrated that an on-board centroid based navigation can be initialized with the precision available/given by ground navigation. During this *bridging phase* the altimeter shall be activated, when distance constraint is respected, and this will allow to converge towards the required initialization for the feature tracking phase (named close fly-by in this document).

6.6.1.4.6.3.1.1 Position knowledge error req.

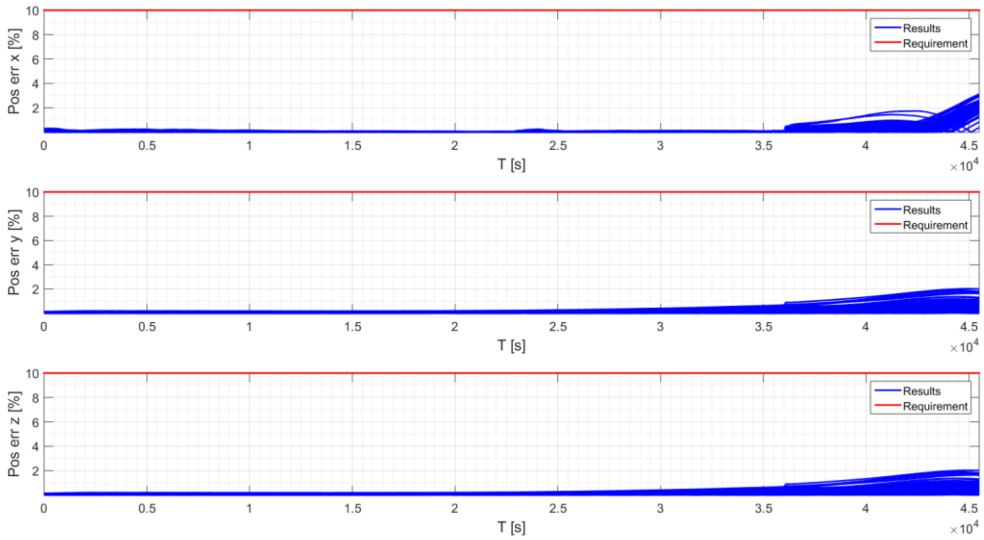


Figure 6-23 Position knowledge error req. for Close fly-by.

As it could be seen in the presented plot, all x, y and z position knowledge errors are lower than 10% relative distance to the centre-of-mass of the target during 100% of the time. Being compliant with the requirement.

6.6.1.4.6.3.1.2 Altitude error at closest distance req.

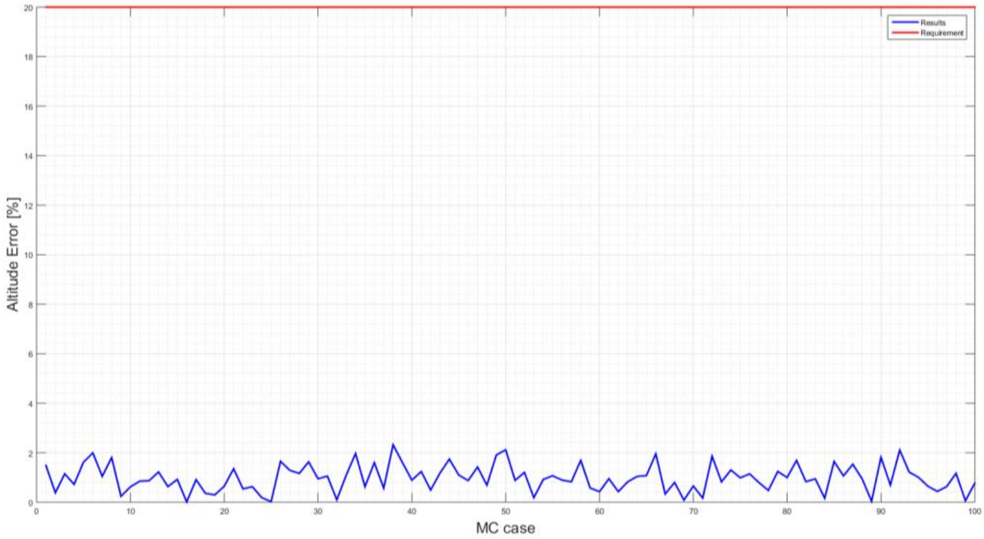


Figure 6-24 Altitude Error as percentage of Altitude at closest distance.

As it could be seen in the presented plot, 100% of the MC cases have an altitude error at closest distance below the 20% of nominal altitude, most of them below 1 order of magnitude.

6.6.1.4.6.3.1.3 Attitude Pointing Error req.

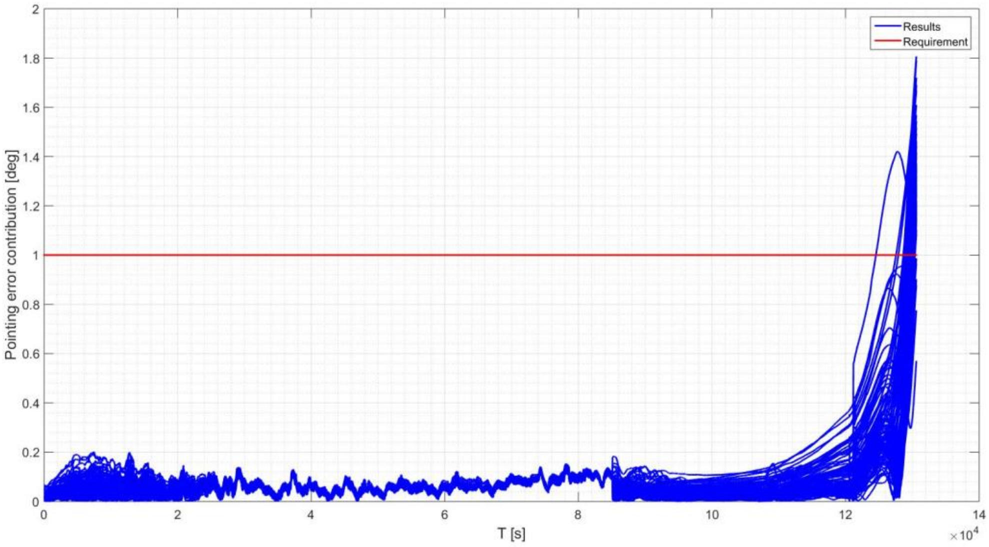


Figure 6-25 Attitude Pointing Error req. for Close fly-by.

Figure 6-25 showed the attitude pointing error for the full fly-by (meant the close fly-by and the fly-by approach). As it has can be seen, the pointing error is below the 1 deg requirement for almost the entire arc (95.4% of the whole fly-by), being compliant with the requirement. The increased pointing error towards the end of the phase (up to 1.8 deg) is due to the deterioration of the illumination conditions. It is indeed mandatory to optimize illumination conditions when the autonomous manoeuvres have to be performed in order to minimize navigation error in the moment it is needed to safely perform the manoeuvre. This will cause degraded illumination conditions later in the arc, reducing autonomous navigation performance, but it will not compromise spacecraft safety.

6.6.1.4.6.4 T2.4.

When using sensor data-fusion the position error in camera frame shows even better performance than in the equivalent vision-based navigation solution. While the velocity error in camera frame is usually of the same order of magnitude. The thermal camera measurements allow to have IP results independent from illumination conditions, while fusion with altimeter guarantees a good precision in the radial channel, which is the less observable in vision based navigation. For details, see E-19: GNC Analysis report.

6.6.1.4.6.5 T2.5.

This requirement is redundant with T2.3 as both the case with and without the altimeter has been investigated and therefore, the results have already been checked, being compliant with the requirement.

If in the next phase other sensors can be included for this experiment (TIRA) there will be the possibility to test a different data fusion strategy and an independent algorithm for collision risk estimator.

It has been demonstrated that the alternative chain investigated offers results “At least the same as the equivalent vision-based navigation solution” (see T2.4)

6.6.1.5 Communications (AWS)

The HERA communications subsystem is made of two independent parts: an X-band TT&C section and an S-band inter-satellite link (ISL) section. The block diagram of the HERA communications subsystem is presented in Figure 6-26.

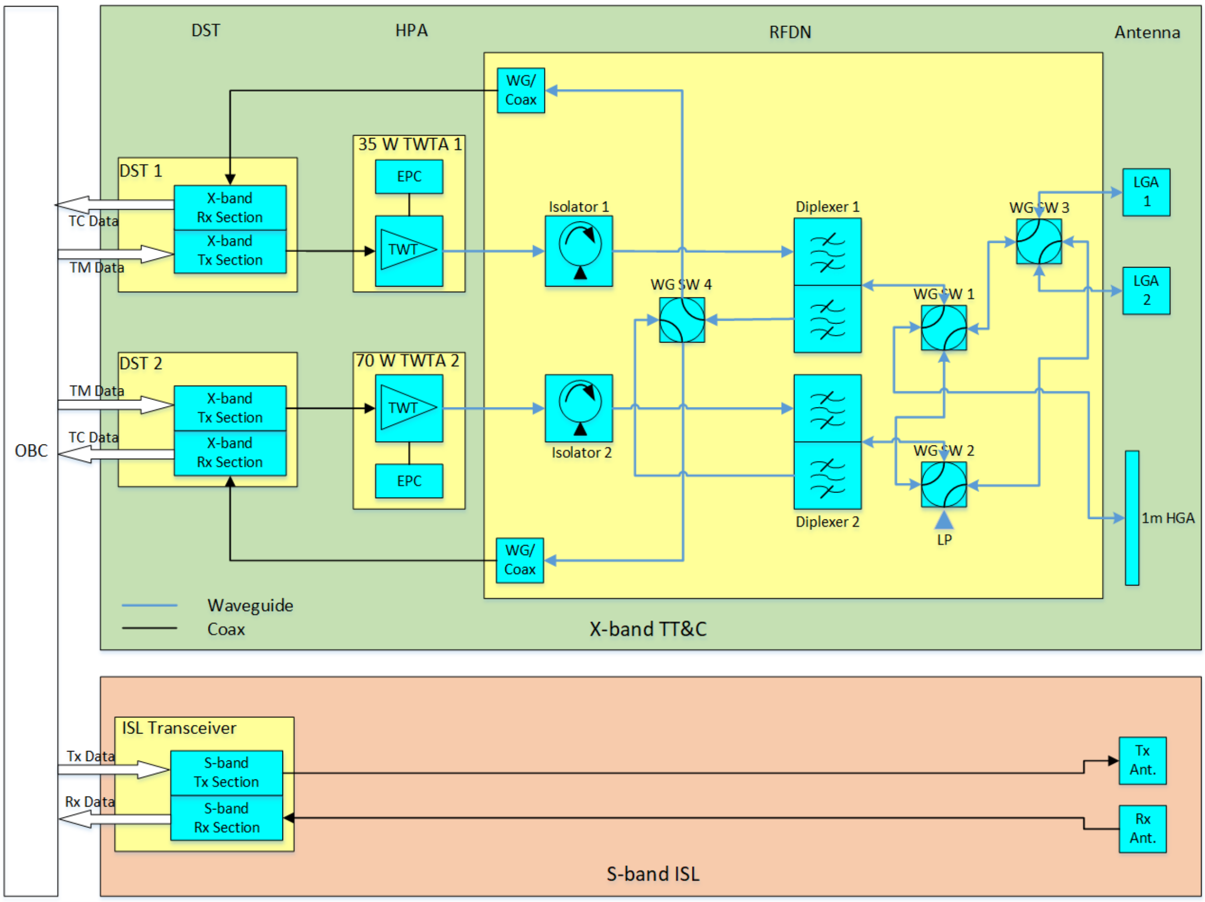


Figure 6-26: Preliminary block diagram of the HERA communications subsystem

6.6.1.5.1 X-band TT&C section

The X-band TT&C section provides the communication link with Earth during the HERA mission. It enables scientific data download and spacecraft commandability and observability.

The TT&C section is made of different units operating at X-band: deep-space transponder (DST), high power amplifier (HPA, based on TWTA), RF distribution network (RFDN), and antennas (hemispherical LGAs and directional 1 m diameter HGA). It consists of two communication chains in parallel, for redundancy purposes, with a cold redundant transmit path and a hot redundant receive path. It operates at X-band, in the 7145 – 7190 MHz frequency range for the uplink and in the 8400 – 8450 MHz frequency range for the downlink.

The design of the X-band TT&C section is derived from the following requirements:

- The Tx and Rx communication chains shall be redundant, in order to ensure single failure tolerance and achieve a high reliability
- The RFDN shall connect any antenna to the transmitting chain and any antenna to the receiving chain. Different Tx and Rx antennas shall possibly be connected to the nominal DST
- A trade-off has concluded that the communications subsystem only needs two different kinds of antenna: a high gain antenna (HGA) with a diameter of 1 m for the nominal operation and two hemispherical low gain antennas (LGA), pointing in opposite directions in order to provide an isotropic coverage, for the emergency and safe modes

- A trade-off has concluded that the redundant TWTAs shall have a different saturated output power: 35 W to be used during the cruise phase and when power consumption has to be limited and 70 W to be used during the proximity phase in order to (almost) double the data rate for the download of scientific data.

The TT&C section interfaces with the On-Board Computer (OBC) and provides the following functionalities:

- It receives and demodulates Telecommands (TC) transmitted at X-band by the ground station, and transmits them to the TC decoder of the OBC
- It modulates the X-band downlink carrier with the telemetry (TM) stream received from the TM encoder of the OBC, amplifies the modulated signal to the required RF level and transmits it towards the ground station
- It turns around the ranging signal received at X-band from the ground station, using transparent ESA standard ranging and/or regenerative PN ranging
- It operates in coherent mode: the transmitted downlink carrier is derived coherently from the received uplink carrier through a turn-around ratio (f_{up}/f_{down}) of 749/880, in order to enable two-way Doppler tracking
- It generates DOR tones at X-band.

6.6.1.5.1.1 X-Band Deep Space Transponder

The X-band Deep Space Transponder (DST) supports the following operational modes:

- uplink operating modes
  - unmodulated carrier
  - telecommand only
  - ranging only
  - telecommand + ranging
- downlink operating modes
  - unmodulated carrier
  - telemetry only
  - ranging only
  - telemetry + ranging
  - DOR tone generation.

The DST can support both transparent ESA standard ranging and regenerative PN ranging. Only PN ranging can be used simultaneously with GMSK telemetry.

The DST is a single source unit, being only manufactured by TAS-I (Italy). The proposed DST is therefore the TAS-I X-band Deep Space Transponder, with strong heritage in deep space missions. The HERA DST will be largely recurring from the ExoMars and Solar Orbiter programs.

The TAS-I DST is made of five modules, as shown in Figure 6-27:

- The baseplate module includes the Rx and Tx DC-DC converters.
- The digital module implements the heritage ASIC for receiving, transmitting and turn-around functions. It is devoted to the digital processing, as well as to the unit management tasks through the LEON microprocessor embedded in the ASIC. The digital module also includes the FPGA implementing the baseband GMSK processing.

- The receiver module consists in the analogue part of the DST receiver: uplink signal down-conversion, filtering and amplification in order to provide the proper level to the ADC input of the digital module. It also includes the internal frequency reference (OCXO).
- The transmitter module up-converts to X-band the low-frequency modulated signal generated by the digital module.
- The modulator module implements the vector modulation and the DOR tone generation.

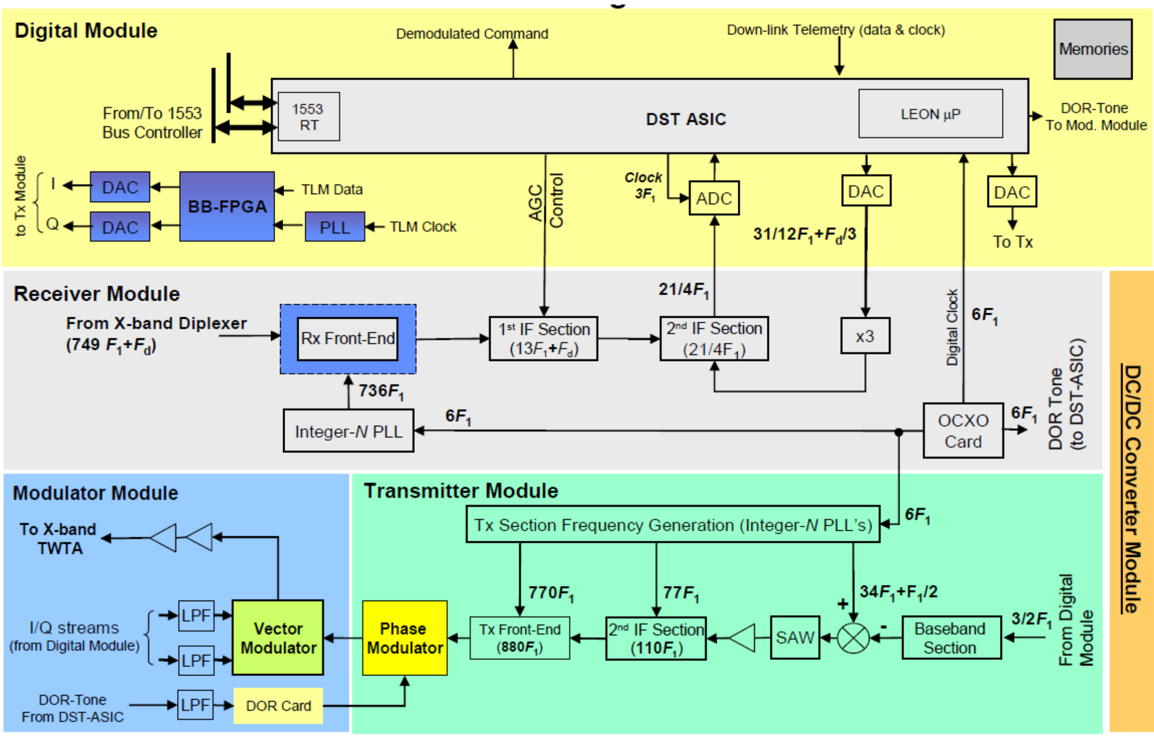


Figure 6-27: Heritage DST block diagram, manufactured by TAS-I

The main technical specifications of the TAS-I DST are summarized in Table 6-14.

Table 6-14: Main performance and budgets of the TAS-I DST

X-DST: General Features	
Mass	3.6 kg
Dimensions	268 mm x 142 mm x 170 mm
Power Consumption	Receiver Only: 17.9 W Receiver + Transmitter (5 dBm output): 36 W
Power Bus Interface	User defined
Overall Frequency Stability	± 1 ppm (both for the receiver and transmitters sides)
Telemetry and Telecommand Interface	1553
X-DST: Receiver Side	
Nominal up-link carrier frequency	X
Modulation format	PM/BPSK/NRZ, PM/SP-L
Receiver noise Figure	< 1.8 dB
Signal Dynamic Range	-60 ÷ -153 dBm
Carrier Loop Bandwidth	Tunable by firmware
Tracking Range	±1.2 MHz at minimum carrier level
Carrier Acquisition Probability	>99%
Command Data rates	PM/BPSK/NRZ: -Selectable by command (7.8125 bps, 15.625bps, 250bps, 1 kbps, 2 kbps, 4 kbps)
Implementation loss	< 2 dB
X-DST: Transmitter Side	
Nominal down-link carrier frequency	X
Modulation format	Residual Carrier: PM/BPSK/NRZ (from 21 bps to 52 kbps), Suppressed Carrier: GMSK (from 62 kbps to 10 Mbps in X-Band)
Modulation Indices	Selectable by command, both for Telemetry (0.2 to 1.2 rad-pk with 0.1 rad-pk step). and Ranging (0 to 1 rad-pk with 0.0061 rad-pk step).
Transmitted Power	Selectable by command from -50 dBm to +4 dBm (0.5 dB step)
Spurious and Output harmonics	Better than 50 dBc
Integrated Phase Noise (1 Hz - 1 MHz)	X-band: ≤1.5 deg-rms,
Allan deviation at tracking threshold	Integration Time = 1000 s: $<10^{-15}$

6.6.1.5.1.2 X-band TWTA

The proposed X-band TWTAs amplify the X-band downlink signal delivered by the DST, in order that the equivalent isotropic radiated power (EIRP) radiated by the communications subsystem antennas enable to close the downlink budget with the intended data rates. Each TWTA is composed of one Electronic Power Conditioner (EPC), one Traveling-Wave Tube (TWT), and one high-voltage cable connecting the EPC and the TWT.

The TWTA operates according to the following operational modes:

- OFF (EPC OFF): No TWTA function is powered. The RF input signal can be present at TWTA input.
- Pre-heating (EPC ON, HV OFF): The TWTA remains inactive as long as the tube preheating is not completed. The preheating period is less than 4 minutes.
- No drive: The TWTA is active and no RF signal is present at its input.
- Saturation: The TWTA is active and operates at saturated output power, with 0 dB input back-off. This is the nominal operational mode. It enables the highest power efficiency of the TWTA. As already mentioned, one TWTA will have a saturated output power of 35 W end-of-life, while the other one will have a saturated output power of 70 W end-of-life.

There are two possible suppliers for the X-band TWTAs, with many flight heritages: TAS-B (Belgium) and TESAT (Germany). The X-band TWTA consists of an EPC designed and manufactured by either TAS-B or TESAT and of a TWT (including the HV cable) designed and manufactured by TED (France). The EPC is made of a high voltage section that provides the supply voltages for operating the TWT and a low power section that provides the power bus, telemetry and telecommand interfaces to the satellite. Two different X-band TWTs will be used for HERA: the 35 W TWTA shall use a TH4604C TWT designed for a saturated output power ranging from 26 to 45 W and the 70 W TWTA shall use a TH4704C TWT designed for a saturated output power ranging from 55 to 80 W. Both TWTs have the same housing. Each EPC will be configured and tuned (minor variations in some component values) for the specific TWT and output power.

6.6.1.5.1.3 X-band RFDN

The HERA X-band RFDN is made of the necessary passive components for ensuring

- The receive routing of the RF signal from the antennas to both DST receivers
- The transmit routing of the low power RF signal from the DST transmitters to the TWTAs
- The transmit routing of the high power RF signal from the TWTAs to the antennas
- The selection of the nominal and redundant antennas for both receiving and transmitting.

It consists basically of coaxial cables (with SMA connector) and waveguide-to-coax transitions for the low-power part and waveguide components (WR-112 flange) for the high-power part. The list of the RFDN components is detailed in Table 6-15. The high-power RFDN components (WR-112 waveguide technology) shall have an RF power handling capability of at least 80 W (maximum TWTA beginning-of-life saturated power), while a maximum power of 1 W is intended for the low-power section components (waveguide-to-coax transition and coaxial cable).

Table 6-15: List of the X-band RFDN components

X-band RFDN Component	Quantity
X-band isolator (including load)	2
X-band diplexer (including filter)	2
X-band switch	4
X-band low-power load	1
WG to coax transition	2
Coaxial cable	4
Internal waveguide section	13
External waveguide section	3

The internal waveguides interconnect RFDN components and TWTAs. They will be accommodated on the communications subsystem panel. The external waveguides interface with the X-band antennas. Low-power coaxial attenuators (not mentioned in the above list) could be needed at the input of the TWTAs in order to avoid overdrive operation.

Possible suppliers of the X-band RFDN, with flight heritages for most or all of the components, are Cobham (France), Honeywell (UK), TAS-E (Spain), TESAT (Germany) and TRYO Aerospace (Spain).

6.6.1.5.1.4 X-band LGA

The HERA X-band communications system is composed of two identical X-band low-gain antennas. The LGAs are mechanically fixed passive devices. They have a hemispherical radiation pattern and will point in opposite directions in order to provide omnidirectional coverage.

Figure 6-28 shows a typical radiation pattern of the LGA antenna, as per heritage measurements (ExoMars 2020).

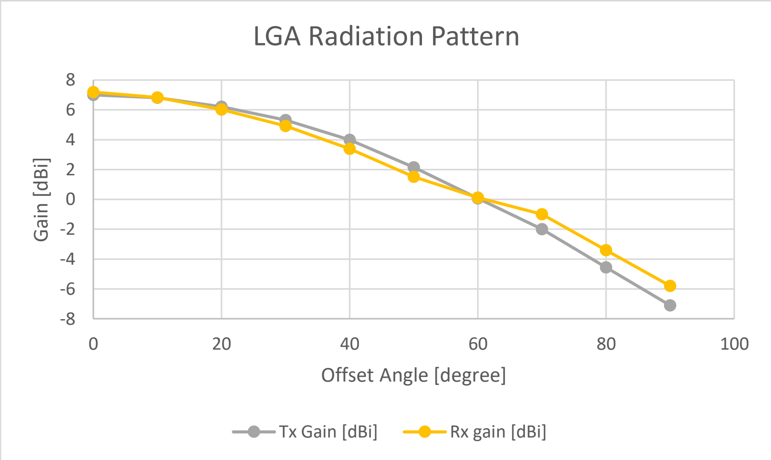


Figure 6-28: Radiation pattern of heritage LGA

Possible suppliers for the LGA, with flight heritage, are RUAG Sweden (Sweden) and TRYO Aerospace (Spain).

6.6.1.5.1.5 X-band HGA

The HERA high-gain antenna is used for nominal operations, during the Cruise and Proximity phases. The X-band HGA is a fixed passive antenna. The expected radiation pattern is shown in Figure 6-29. It has been theoretically assessed on the basis of a 1m diameter dish antenna assuming a uniform circular aperture.

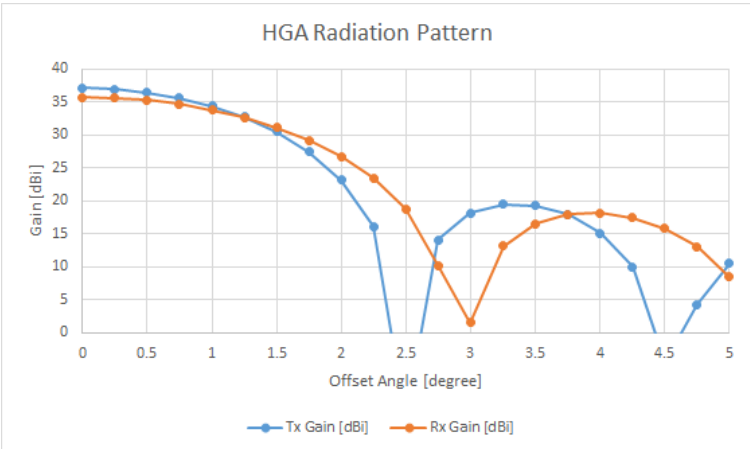


Figure 6-29: Expected radiation pattern of the HGA

It has been confirmed by potential suppliers that an antenna with such a radiation pattern and a diameter of about 1 m can be manufactured on the basis of heritage technology. Possible suppliers for the HGA are MDA (Canada), Sener (Spain) and TAS-I (Italy).

6.6.1.5.2 S-band ISL section

The S-band inter-satellite link (ISL) section will provide the communication link with the CubeSats of the HERA mission and will therefore enable the HERA spacecraft to relay the communications between CubeSats and Earth. The ISL communication link will enable to control (by sending commands) and monitor (by receiving telemetries) the CubeSats, as well as to download scientific data collected by the CubeSats. Cubesat ranging could also be provided by the ISL.

The S-band ISL consists of one ISL transceiver, two S-band antennas (one Rx antenna and one Tx antenna), and two coaxial cables.

The ISL transceiver interfaces with the on-board computer (OBC) for both data and TM/TC.

Possible suppliers of the ISL transceiver are Syrlinks (France) and Tekever (Portugal). Possible suppliers for the S-band antennas are SSTL (UK) and STT (Germany).

6.6.1.6 OBDH

Figure 6-30 shows the OBDH architecture, which consists of an OBC, mass-memory and RTUs.

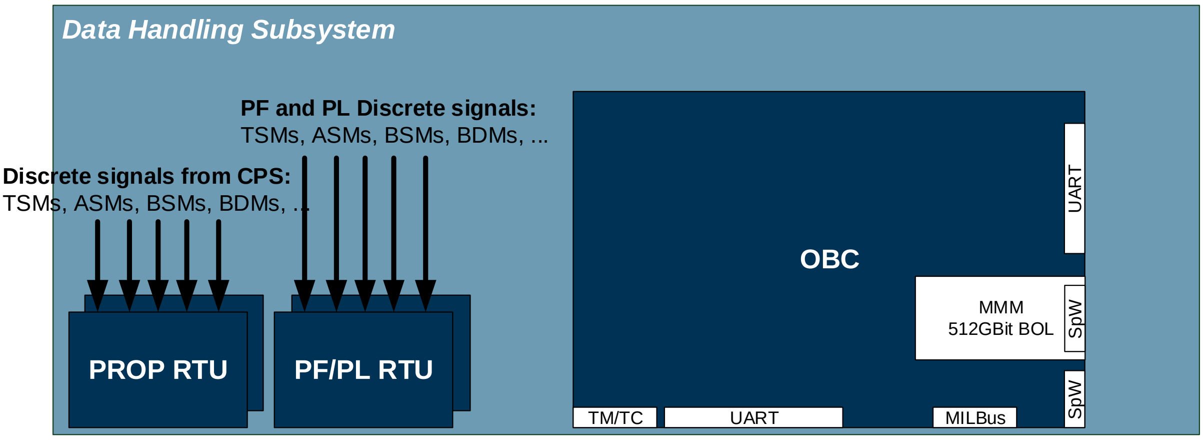


Figure 6-30: Systems Block Diagram

The driving paradigm behind the avionics architecture is to maximize re-use of existing hardware and software for the sake of cost and schedule efficiency. Moreover, it is intended to simplify the architecture as much as possible to streamline the AIT/AIV process and reduce costs.

Most components have been used on previous missions by the industrial team before. This heritage allows software reuse for a major part, minimizing costs and reducing schedule risk. Moreover, existing industrial relationships and experience with suppliers can be harnessed to support a smooth procurement and AIT process, if compatible with the geo-return for this mission.

Taking into consideration these factors, the HERA avionics architecture is centred around PROBA-Next OBC developed by QinetiQ Space.

PROBA-Next OBC is an evolution of ADPMS, which was also developed by QS and has flown on all the PROBA satellites. Under HERA Ph.B1, the redundant lane EM is currently under development at QS to boost the TRL to 6, which is required bin the ESA SoW before the start the start of Ph.B2.

The PROBA-Next OBC provides significantly increased performance, memory capacity and interfaces compared to its predecessor. Key features include:

- The OBC is built around the powerful GR712RC which includes a LEON3FT Fault-Tolerant processor and a large scale companion FPGA that controls peripheral functions and handles OBC communication interfaces
- Dual-core LEON3 processor to replace the single core LEON2 in ADPMS
- Doubled the on-board memories to 512 Mbyte of SDRAM from 64 Mbyte in ADPMS
- Modular design to allow expanded functionalities and interfaces through attachment of additional modules

PROBA-Next OBC consists of the following modules:

- Processor Module (PM)
  - Contains the main OBC capabilities i.e. processing function, memories, and BIOS
- Extension Module (ExM)
  - Contains auxiliary interfaces and functions, as well as the primary power conversion
- Reconfiguration Module (RM)
  - Performs the switch-over between nominal and redundant lane in case of a failure

At system level, there will be 2 OBCs for redundancy, running in cool redundant configuration. In a cool redundant system, the critical functions (CF) are always active (Hot) on both OBCs. Therefore, both OBCs are powered in this configuration, but the non-active OBC is configured such that only the CF are active, while the remaining functions are all in standby and no software is running on the non-active OBC.

The configuration of which lane is active and how a reconfiguration is handled is all controlled by a dedicated module called the Reconfiguration Module, which is a purely analog module.

6.6.1.6.1 Data Handling Design

The data handling subsystem design is built around the On-Board Computer unit and include Mass Memory Module and Remote Terminal Unit.

6.6.1.6.1.1 Interconnection Diagram

The following figure gives more details about the Data handling subsystem interconnection diagram. The other interconnection to the S/C are not listed in the figure.

All the different type of internal DHS interface are listed. Green arrow are linked to the SpW interface. Orange and purple colour are used to define the DHS MIL-BUS for Control and Command both RTU (PL/PF and PROP).

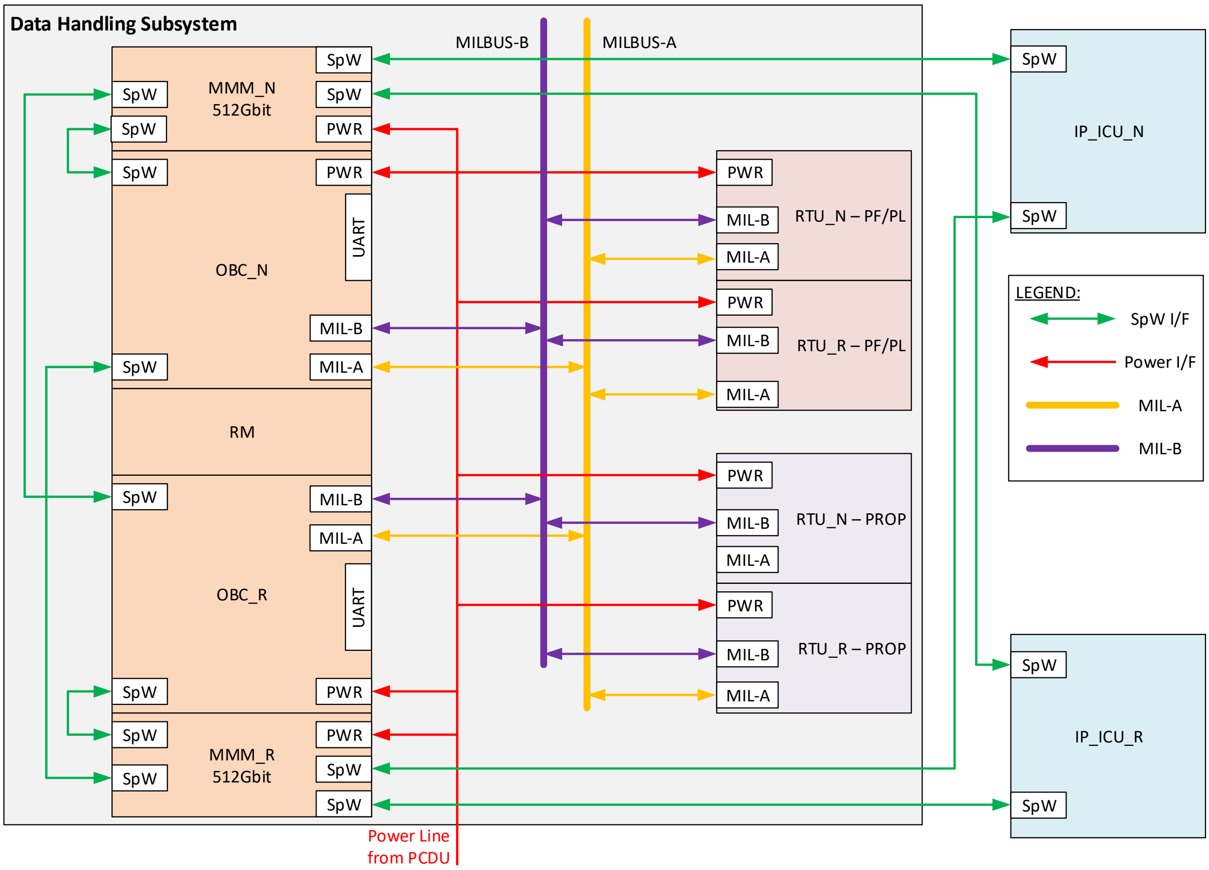


Figure 6-31 – Data Handling Subsystem interconnection diagram.

The DHS redundancy concept is based on the interface cross-strapping between the nominal and the redundant lane of the DHS units.

6.6.1.6.1.2 Functional Architecture

6.6.1.6.1.2.1 OBC – IP-ICU/Payload

Communication architecture:

The DHS communication architecture is composed of the OBC and the MMM. Functionally speaking the IP-ICU is also part of the Data Handling Subsystem.

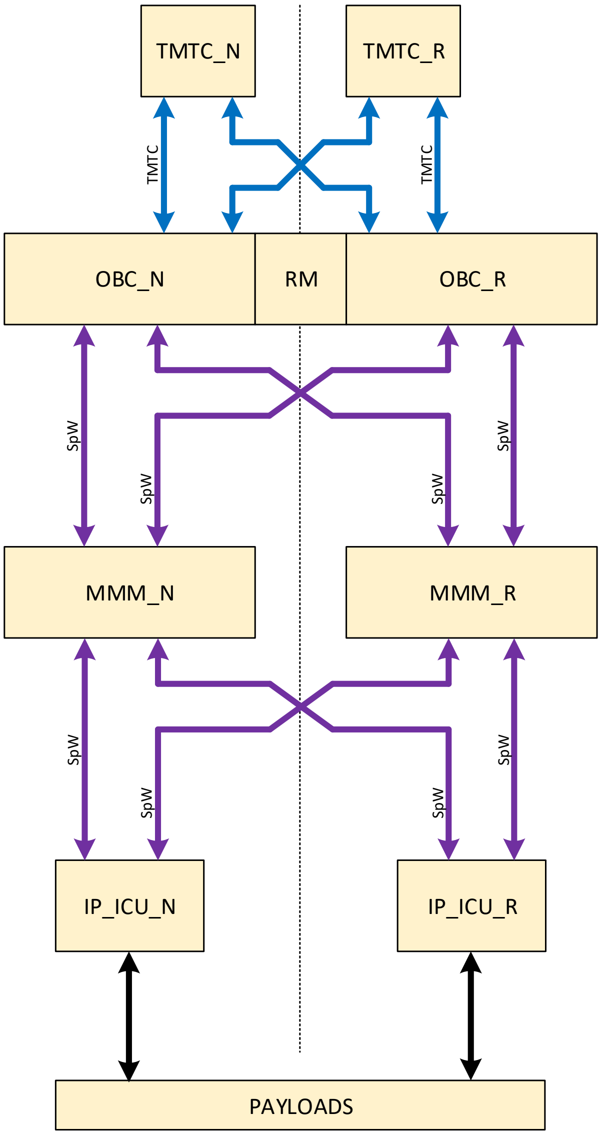


Figure 6-32 - DHS communication architecture between OBC and IP-ICU/Payload Subsystem

Data Flow example:

The HERA mission characteristics does not allow to have a dedicated TM link from the MMM to the RF subsystem for the transmission of the Payload and Platform data to ground. Indeed, all the data stored in the MMM will have to be transferred via the C&C SpW interface to the OBC and then formatted.

An example of a Communication between the OBC and Payloads is given below:

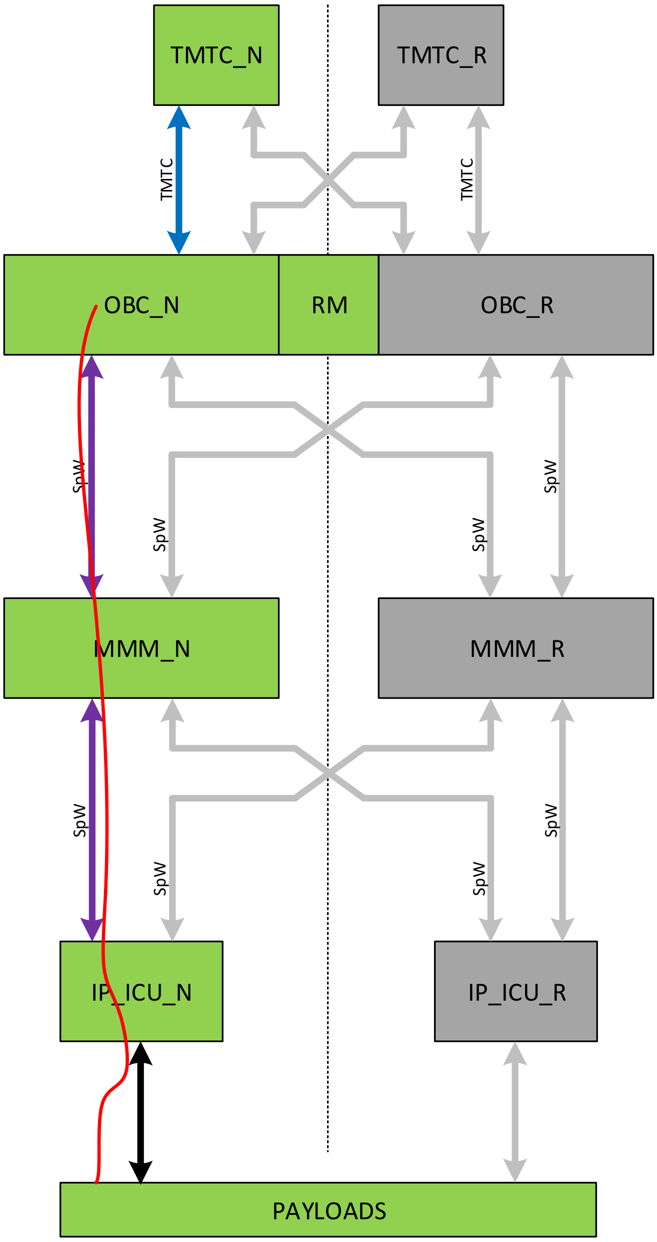


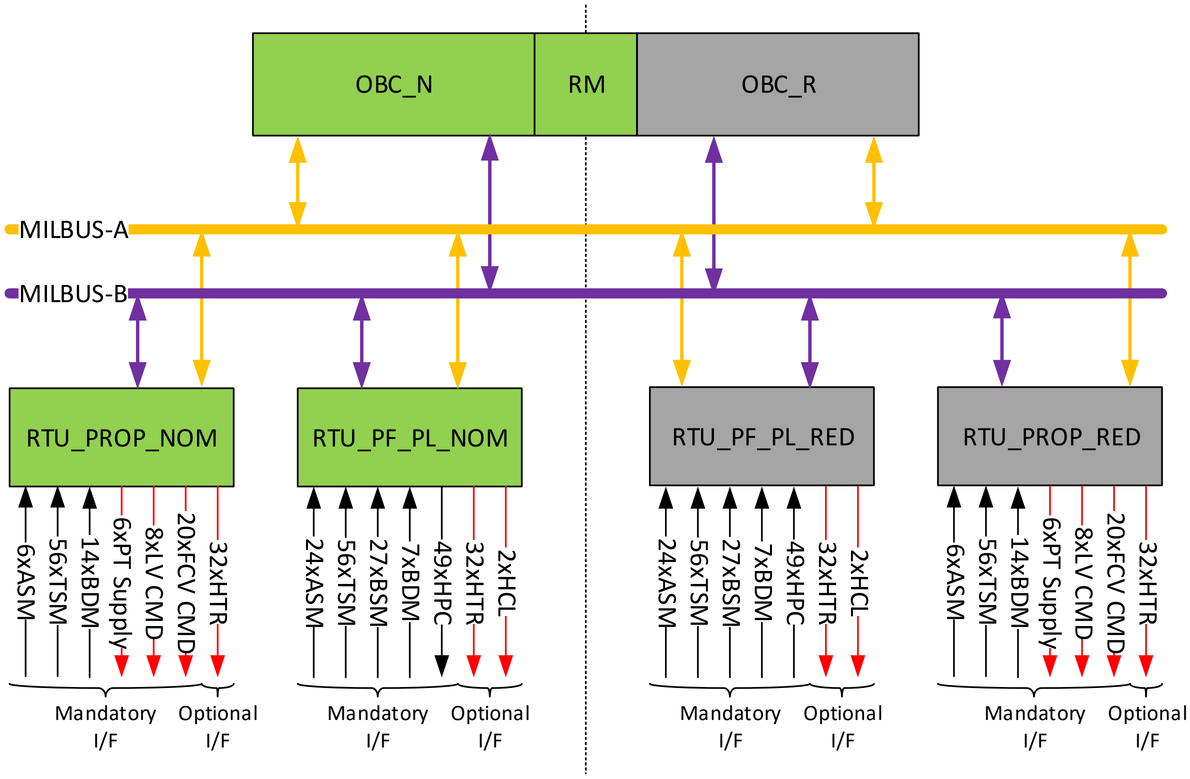
Figure 6-33 - Communication flow between the OBC and Payloads:

Note: Nominal communication flow is marked in red in the figure.

6.6.1.6.1.2.2 OBC – RTU

The OBC communicates via MILBus interface with the two types of RTU needed for the HERA mission. On RTU is dedicated to the propulsion control and command. The second type is dedicated to the Payload and the Platform housekeeping and direct command.

The DHS communication architecture with the RTU is given in the following figure:



6.6.1.6.2 Data Handling Unit Design

6.6.1.6.2.1 On-Board Computer

The OBC concept is built around a platform controller and a large scale companion FPGA that controls peripheral functions and handles OBC communication interfaces. The FPGA is connected to the platform controller by an internal SpaceWire link. In the standard configuration of the OBC, memory blocks used for OBSW storage, boot memory, and safeguard memory will be connected to the FPGA while memory blocks used for working memory and BIOS are connected to the platform controller. Furthermore, the OBC contains UART (RS422), CCSDS TM/TC, SpaceWire, JTAG, MIL1553, analogue and digital IO interfaces as shown in the figure hereafter. The OBC also performs primary power conversion, providing internal 5V, 3V3, 2V5, 1V8 and 1V2 power rails. EMI filtering blocks out disturbances from the unregulated 28V power bus and filters switching noise originating from the DC-DC converters. Additional overvoltage protection circuitry will be provided. For the analogue inputs required, an ADC with SPI interface can be included in the OBC architecture as a companion to the FPGA.

The figure below presents the generic diagram of the OBC system level architecture:

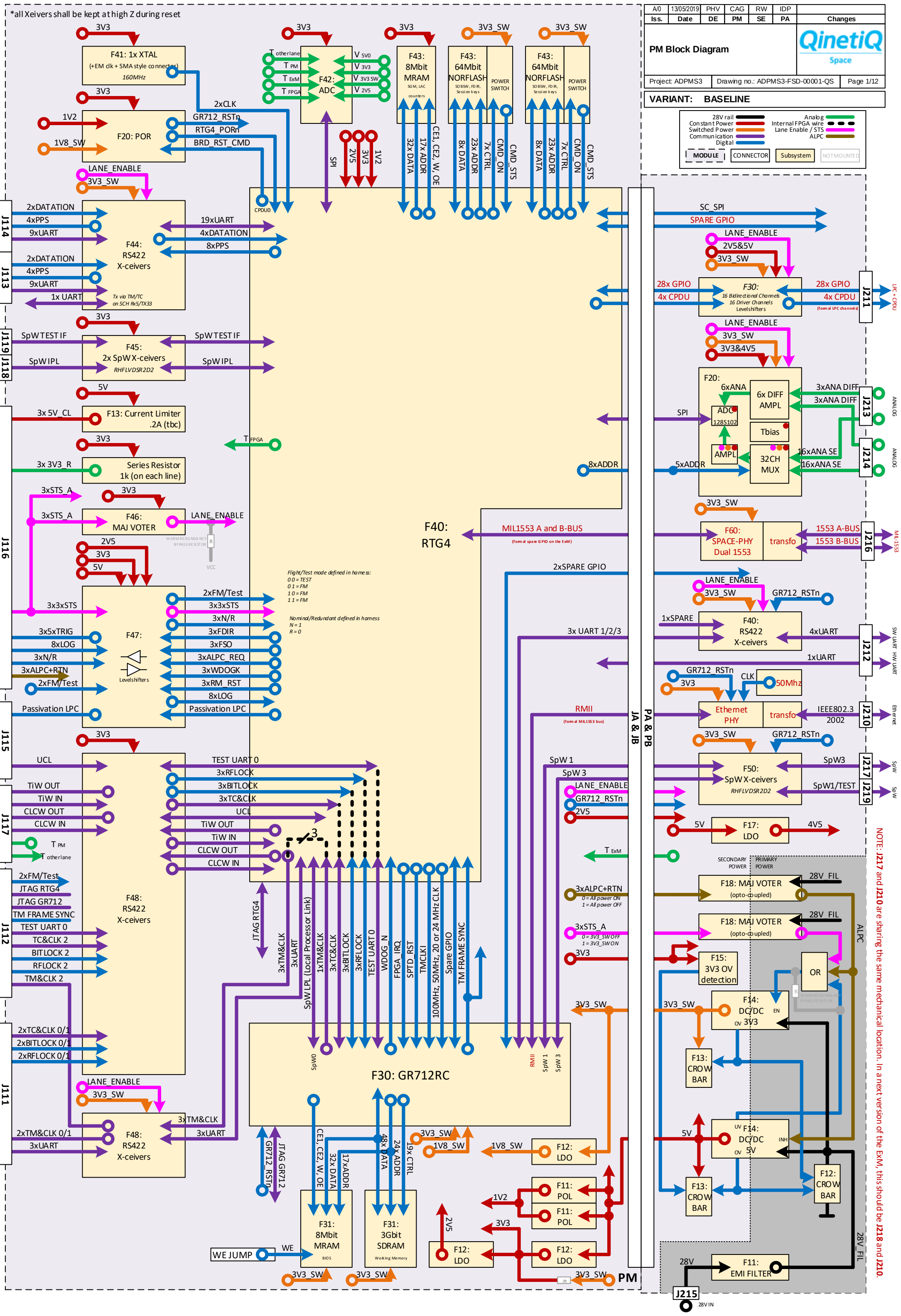


Figure 6-34 – OBC system level architecture.

Processor Module (PM) and the Extension Module (ExM).

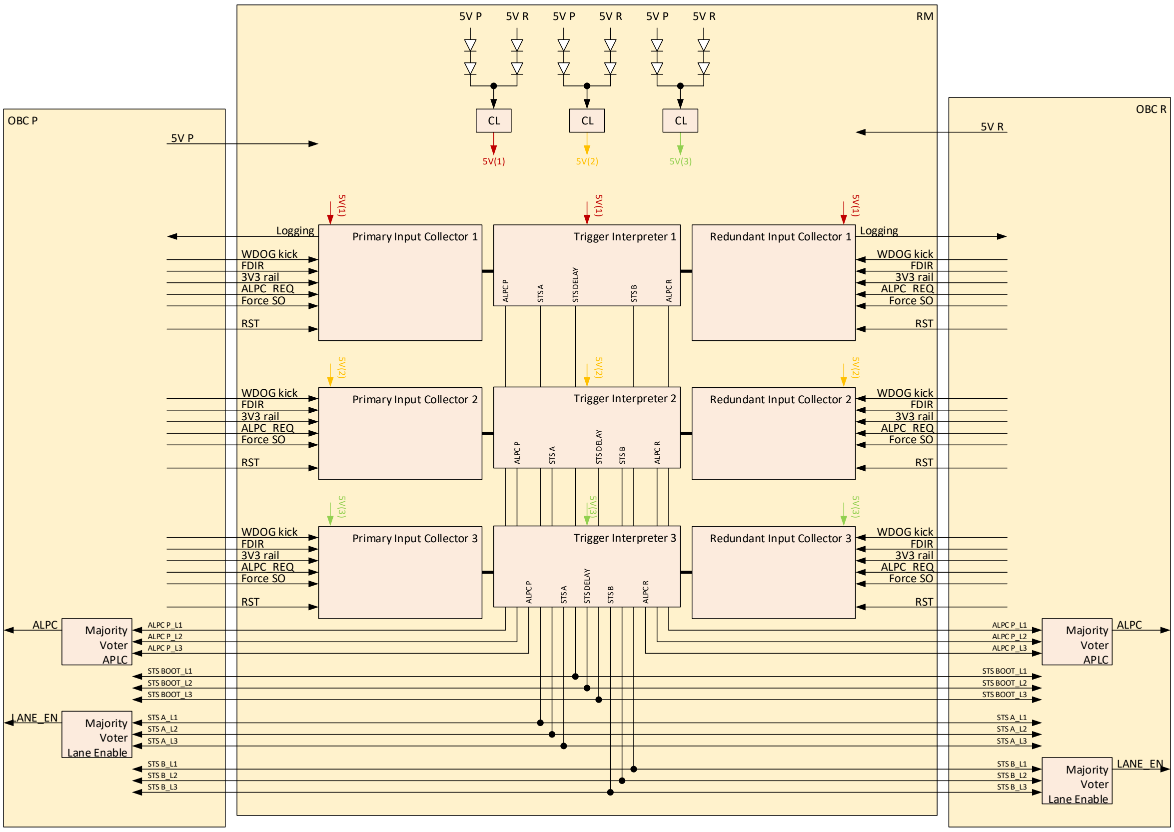
The Processor Module (PM) contains the main OBC processing and interface functionalities. Therefore, essential components are located on the PM: the platform controller with companion FPGA along with safeguard, OBSW storage and reduced data storage capacity (e.g. for science data from demonstrators), depending on the board configuration. UART (RS422), SpaceWire and CCSDS TM/TC communication interfaces are provided on the Processor Module. For signal integrity, the generation of the lower voltage rails such as 3V3, 2V5, 1V8 and 1V2 (mainly used by the FPGA and the Microprocessor) is also located on the Processor Module.

Primary power conversion along with EMI filtering and overvoltage protection is done on the Extension Module (ExM) and power will be fed to the Processor Module (PM) through the board-to-board connector interconnecting both modules. Additional UART, SpaceWire (shared with an optional CAN), MIL1553 and SPI communication interfaces are reserved on this connector for future use. The ADC and analog front-end electronics is located on the Extension Module and interfaces with the companion FPGA on the Processor Module over SPI. The Extension Module will also contain supplementary GPIO and LPC interface electronics.

Reconfiguration Module

The Reconfiguration Module is part of the Proba Next On-board Computer (OBC) and is responsible for management of the active OBC lane. It comprises three independent Lanes, the outputs of which are sent to the OBC for majority voting. The three lanes are identical so their outputs should also be identical under normal operating conditions.

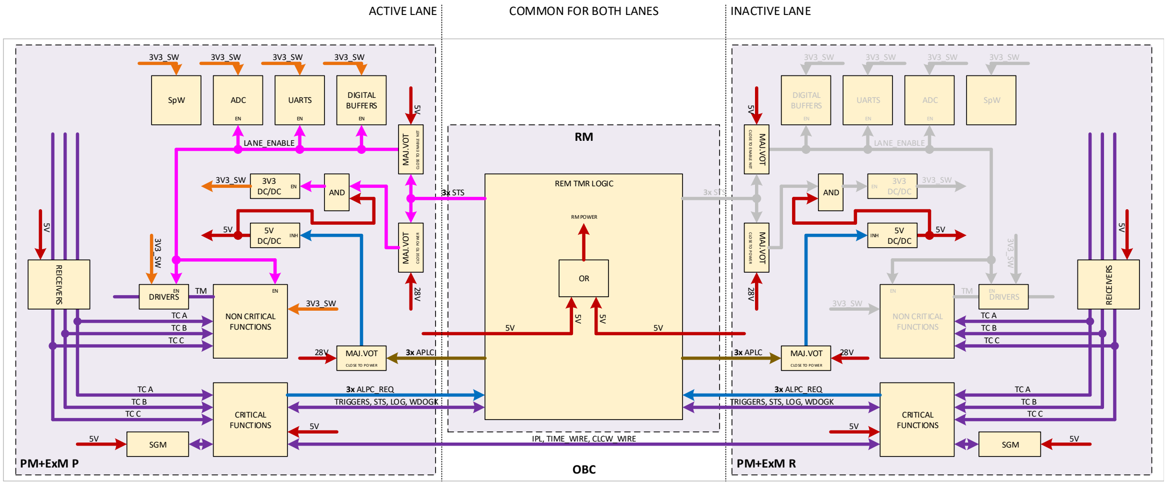
The following diagram shows a top level functional representation of one of the three Lanes of the Reconfiguration Module (RM):



The reconfiguration module has two main parts; the critical functions and the switch-over mechanism. The first part contains all critical functions such as essential telecommand, essential telemetry, time management and safeguard memory. The second part – the switch-over mechanism – is the part that collects the different trigger (alarm) inputs and orchestrates the switch-over.

HERA OBC Configuration

The following figure gives the baseline On Board Computer configuration that fit the HERA requirements:



The table below shows the distribution of data handling boards in the Pnext OBC for the HERA satellite.

Pnext OBC module	Envisaged number
PM	2 (primary and redundant)
ExM	2 (primary and redundant)
RM	1

Note the MMM is not included in the above table.

6.6.1.6.2.2 Mass Memory Module

The following section gives more details about the Mass Memory Module design part of the OBC unit. The MMM functions are listed below:

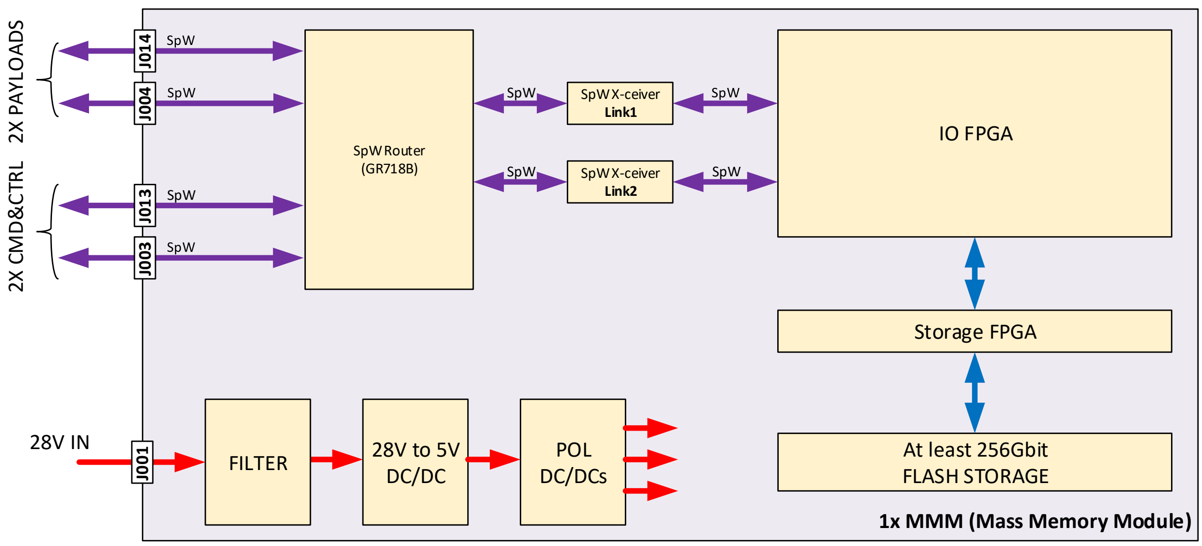
- The reception and storage into the mass memory of payload data (from the IP-ICU)
- The reception and storage into the mass memory of platform data coming from the OBC.
- Servicing as a routing point between OBC and the IP-ICU and the Payloads hence allowing the OBC to command & control the Payloads and the IP-ICU via the MMM.
- The transfer of the PL and PF data through the OBC (TM/TC I/F – RS422) to the Communications Subsystem (from ASW).

The MMM receives and stores data from the spacecraft payload units and the nominal and redundant OBC units. When commanded, the MMM will transfer stored data via the embedded TM encoder to the nominal or redundant downlink units on the spacecraft. The MMM also routes data from the OBCs and payloads via an embedded SpaceWire router. Additionally, the OBC can directly control payloads via the embedded SpaceWire router without the use of a dedicated Instrument Control Unit.

The MMM is composed of the main functional block listed hereafter:

- 1x FPGA for payload interface
- 1x FPGA for Flash storage and support functions (C&C SpW, Watchdog)
- Flash Memory Partition (512Gbit net BOL) for non-volatile data storage plus power switch, using 128Gbit devices
- SDRAM buffer plus power switch
- Interface circuitry (SpaceWire PHYs)
- Reset circuitry component
- Provision of secondary voltages (DC/DC Converter, PoL)

The MMM design overview is given in the figure below:



MMM Key parameters are:

- Reception and storage into the mass memory of payload data plus platform data coming from the OBC
- Monitoring & Control of the MMM by the OBC
- Output of stored data
- Simultaneous recording (up to 200 Mbps) and playback (up to 100 Mbps physical)
- Storage capacity: 512 Gbit BOL
- Secondary power provision for the MMM
- Single point failure-free design (for a combination of two MMMs (i.e. 1 x nom, 1 x red)
- Interfaces:
    - 1 x nom & 1 x red Payload Interface (SpaceWire, up to 200 Mbps physical)
    - 1 x nom & 1 x red C&C + TM data (*) Interface (SpaceWire up to 100Mbps physical)
    - 1 Power Input Interface (22V –33.6V)
- Power: < 7.8 W (full operating, with margin), < 4.8 W (standby, with margin)

(*) TM data are Payload and Platform data to be downloaded to ground via the HERA RF subsystem.

Possible suppliers for the MMM are MDA (Canada), DSI (Germany), SYDERAL (Poland) and TECNOBIT (Spain).

6.6.1.6.2.3 Remote Terminal Unit

The outcome of the RTU configuration trade-offs show the necessity to implement in the HERA spacecraft platform two types of Remote Terminal Unit. The first RTU type provides the necessary I/O interface to the Platform and the Payloads. The second RTU type is dedicated to the control and command of the Spacecraft propulsion subsystem.

The RTU design are all based on a modular RTU system. The modularity enables easy assembly of a large variety units with different functionalities to match the needs at satellite level.

6.7 System Level FDIR Approach

The system level FDIR concept has been tailored from the OHB ‘Harmonised FDIR Concept and Analysis’ report (GL-0046-SYS, in preparation). The fundamental objective is to ensure the integrity of the spacecraft. This is achieved by isolating the failure and performing the required recovery, guarantying the spacecraft’s safety and survivability. Interruption to the mission is limited using the available redundancy

Figure XXX shows the FDIR hierarchy. It consists of five layers, which builds on the heritage of previous missions at OHB System (e.g. MTG, Electra, Galileo). Failures are classified into groups, which are subject to a common failure handling approach. A correction exists between the levels and the failure criticality. The higher the number, the more critical the failure – i.e. level 4 events are more critical than level 0 events.

FDIR is performed, based on the hierarchical approach, on the lowest possible level (as close as possible to the origin of the failure). Although it should be noted that the origin of where the failure is detected is not taken into account. This might differ from the recovery action of the FDIR mechanism, and would be confusing. It therefore does not provide an advantage when applying the FDIR hierarchy on the FDIR concept.

The FDIR approach has three main advantages. It provides:

- A decentralised software FDIR approach – i.e. the software can judge from the FDIR level by which software instance the FDIR function needs to be covered.
- A modular test approach, which facilitates the judgement on which level (unit, subsystem, spacecraft) the FDIR functions need to be tested
- Compliance to the typical FDIR rule that comes hand-in-hand with the FDIR hierarchy (from customer experience): “Failures shall be recovered on the lowest possible FDIR level”.

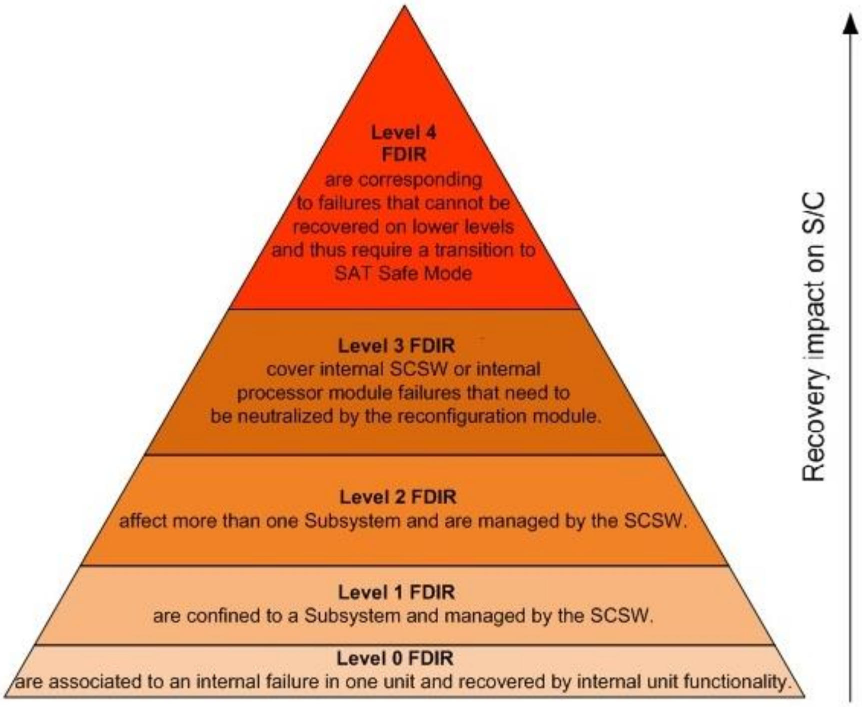


Figure 6-35 - FDIR levels, recovery impact on S/C

The system level FDIR concept is therefore harmonised to include inputs from multiple subsystems. These includes:

- Launcher Separation, where no special FDIR scenario is currently foreseen. Some alarms might have to be disabled in LEOP (trigger during de-tumbling), which needs to be confirmed.
- LEOP as applied to survival mode after separation
- GNC FDIR concept
- Thruster FDIR concept
- Payload FDIR concept including unit monitoring & recovery action (e.g. a power cycle) via the OBC, dedicated internal FDIR unit functions (detect and isolate an internal anomaly), alarms and watchdogs.
- MIL-BUS FDIR including the MIL-BUS error detection, reaction time and recovery sequence (also applied to SpaceWire and UART)
- OBC FDIR
- FDIR verification and validation approach including the design, specifications, flow and testing

Certain alarms – at system level - are foreseen to detect critical anomalies that would otherwise 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
- Battery Under-voltage Alarm
- Thruster On Alarm
- Sun presence alarm(TBC)

FDIR will also be based on enabled/disabled validity checks. This prevents inadvertent FDIR activation. Possible validity conditions include, data validity, equipment operational state, configuration conditions and proximity checks for the Cubesat(s)

PUS-is central to the implementation of FDIR, including:

- Event Reporting Service [PUS5, OIRD-EVT-1]
- Monitoring Service [PUS12, OIRD-OBM-1]
- Event Action Service [PUS 19, OIRD-EAS-1]
- Request Sequence Service PUS 21

More detailed information is given in the FDIR Concept and Autonomy Report [RD25].

7 COMPARISON DESIGN TO MISSION OBJECTIVES

This document provides a consistent overview on the Hera spacecraft design considering system architecture and major subsystem architectures based on the provided requirements baseline in SRD issue 2.0. To the present level of analyses the spacecraft is compliant to this agreed set of requirements.