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ARTEMIS: A complete mission architecture to bridge the gap between humanity and near-Earth asteroids

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Abstract

Asteroid retrieval missions have recently attracted increasing interest from the community and could provide opportunities for scientific exploration, resource utilisation and even the development of planetary defence strategies. This paper was developed as a result of a 6-month MSc group project, realised by a total of 14 students at Cranfield University pursuing the Astronautics & Space Engineering degree. An overall system design is proposed for a technology demonstrator mission to move a near-Earth asteroid into an easily-accessible location where it could be further explored by future missions. The target final orbit is a southern halo orbit around the Lagrange point (L2) on the Sun-Earth system. ARTEMIS (Asteroid Retrieval Technology Mission) abides by ESAs constraints for a Large (L) mission call: realised in only one launch with Ariane 64, an operational duration of less than 15 years and a cost at completion of at most €1100M.

The proposed mission combines the design of optimal trajectories, employs advanced solar electric propulsion and introduces a befitting level of spacecraft autonomy. The target is the 2006 RH_{120} asteroid, with an approximate diameter of 6.5 m and mass of roughly 350 tons. To refine existing data, the ARROW CubeSat mission (Asteroid Reconnaissance to Research Object Worthiness) is to be launched a year prior to the main mission to probe the asteroid via a fly-by. ARROW will provide valuable information, such as the asteroids spin rate, rotational axis and better mass estimate, increasing the overall chance of mission success. The main mission will then capture and secure the asteroid using a mechanism of arm-like booms with xenon-filled Vectran^{TM}bags. To allow for proper adaptability to the objects shape and mass distribution, as well as preserve the asteroid unaltered, the mechanism is fully contained in fabric that encapsulates the asteroid.

The paper concludes that such a mission is conditionally feasible, and summarises the design process resulting in the final overall mission baseline design. It also examines the practicality of the suggested design for future missions such as space debris removal or its ability to retrieve celestial bodies with variable mass and shape. Proper adaptation of the design could allow for retrieval of similar size or smaller objects. The future implementation of this mission may further the understanding of the origin of the solar system and act as a catalyst to a new celestial body exploitation industry.

1 Introduction

The origin of the Solar System was marked by turbulence and violence, where interstellar dust, particles and compounds collided forming larger and larger bodies due to gravitational forces. Once big objects, like planets, were formed, heavy metals and complex compounds fell towards their cores making them inaccessible to study.

However, due to their small size, asteroids maintain their ancient composition almost intact. Near Earth Asteroids (NEAs) represent an accessible piece for understanding the composition of the universe and the formation of the Solar System.

Besides their scientific interest, NEAs could be a solution for a sustainable interplanetary transport system. It is undeniable that great efforts are underway in the space exploration sector, but the current approach of carrying all the fuel needed from departure is still considered an unsustainable model. There is still a great gap between the current space technology and the future possibility of in-space refuelling using asteroids as potential fuel stations and this mission is a step to bridge that gap.

In addition, asteroid mining and resources derived from them are a potential market waiting to be exploited. First movements have been done to promote this industry, USA "SPACE Act of 2015" [28] and Luxembourg "Law of 20 July 2017 on the exploration and use of space resources" [29] are the first legal frameworks specifically designed for this purpose, they provide the legal framework for the companies based in these countries to keep their profit, as opposed to other countries in which the ownership is debatable. New companies like "Deep Space Industries", "Planetary Resources" or "The Asteroid Mining Corporation" were born in consequence, primarily investing on asteroid mining technologies.

1.1 Precursor Missions

From the first asteroid fly-by achieved by Galileo in 1991 to the latest 2016 launch of the OSIRIS-REx sample retrieval mission, NASA has been the agency showing most interest in the field of asteroid exploration participating in no less than 10 missions.

The first mission targeting a NEA was NASA's DSPSE - Clementine, a probe mission launched in 1994. The ones that followed were either fly-by: NEAR Shoemaker (NASA/APL - 1996), Chang'e-2 (CNSA - 2010), impactor missions: Deep Impact (NASA/ JPL - 2005) or sample return missions: Hayabusa (JAXA - 2003), Hayabusa 2 (JAXA - 2014) and OSIRIS-REx (NASA - 2016).

Studies that have considered missions with purposes other than scientific observation or sampling include the ESA/NASA candidate Asteroid Impact Mission (AIM) meant to demonstrate a deflection technique, and the Near-Earth Asteroid Retrieval Mission (ARM) study proposed by NASA/Caltech. The aim of the latter was to reach a NEA and capture a part of it which would then be brought to a Distant Retrogade Orbit around the Moon to be accessible by future manned missions [30]. Although the study was selected for launch, it eventually was cancelled due to NASA's 2018 budget. The ARM was an inspiration for the present study, which presents a mission to capture and redirect a NEA to a more accessible orbit, abiding by ESA's constraints.

1.2 Objectives

ARTEMIS aims to develop an overall mission design to study the feasibility of a NEA retrieval mission in the ESA framework.

The ARTEMIS project, as part of the Astronautics and Space Engineering MSc, aims to mock-up an ESA large class mission proposal with the theme: "Move an asteroid for science, defence and resources". From this statement, the main objective of the mission has been derived:

- To move a NEA from its current orbit and state to a more accessible one in order to allow future missions to make use of the body for science, defence and resource acquisition purposes.

ESA proposals open the opportunity for the next steps on the "global exploration roadmap" [21] and establish the boundary conditions that shall be maintained for a European mission. This approach creates a starting point for the project, which uses [19] and [20] as reference documents. With these factors in mind the secondary objectives were derived:

- To perform non-destructive high-quality science on the captured asteroid.
- To demonstrate to the public that positive actions are underway that test technologies to use celestial bodies to defend the planet against extra-planetary hazards.
- To demonstrate basic in-space resource utilization technology.

2 The Asteroid

The entire NEA population has been considered, obtained from the MPC [1] database on January 24, 2018. Only asteroids 2006 RH_{120}, 2013 RZ_{53} and 2018 AV_2 can be retrieved within the maximum propellant mass in this design. However, due to the retrieval opportunities, only by selecting 2006 RH_{120} the mission can be completed by 2050.

2006 RH_{120} is a small asteroid situated in an orbit with semi-major axis 1.033 AU and eccentricity 0.025 as of 2018 [3]. Approximately every 20 years it has a close-approach with the Earth, and its similarity with the Earth's orbit presents the opportunity to both observe and retrieve it. Asteroid 2006 RH_{120} has a mean diameter between 2.8 m and 6.4 m, and mean mass between 31 and 366 tonnes, obtained from its brigthness and approximate albedo distribution [12, 13].

While its trajectory is well known, its composition is not, and the current information on the asteroid is limited. To achieve an insight to the asteroids unknowns, a cubesat mission will be launched to rendezvous with the asteroid in November 2028 and take high resolution images to determine its shape, size, and composition.

2.1 ARROW Mission

The ARROW mission is a 6U cubesat proposed to close fly-by asteroid 2006 RH_{120} and capture images. Using a BUSEK Iodine fuelled ion thruster that can provide up to 700 m/s ΔV, the cubesat would piggyback on a launch to L_2 and hibernate in a deployment capsule until ready to begin its trajectory. Candidate missions to do so are ESA's ARIEL and PLUTO (both M class), to be launched between 2026 and 2028 [14]. The accurate attitude control system, consisting of two star-trackers, six sun-sensors, and three reaction wheels, will be complemented by multiple trajectory updates from the ground. With these, the cubesat will be able to adjust its orbit and approach the asteroid within a few kilometres, or possibly closer, for high resolution photographs.

3 Mission Analysis

The design of the mission has been divided into several phases. First, the spacecraft is launched and put into a heliocentric trajectory to approach 2006 RH_{120}. Then, a series of manoeuvres are executed for science and defence purposes and are followed by the rendezvous and grabbing of the asteroid, after which the latter is hauled into its final orbit. During the hauling phase, the spacecraft has to thrust with increased mass which critically impacts the required propellant mass and, thus, drives the choice of the retrieved body.

3.1 Launch

The spacecraft designed in order to meet the mission objectives, named ZEUS-15, is planned to launch on board an Ariane 64, a heavy-lift launch vehicle currently under development by the ESA [1] from the Guiana Space Centre on the 23rd of November 2029. The rocket provides an estimated mass capability to Earth-escape of 6,789 kg, including an integration adapter, within an 18 m tall fairing. ZEUS-15 is inserted into a 0° declination direct escape trajectory with right ascension 142°, and velocity at infinity of 1.08 km/s.

3.2 Outbound Trajectory

The outbound phase of the spacecrafts trajectory directs its path from the escape trajectory until reaching 2006 RH_{120}. After the spacecraft is commisioned and has departed the Earth's SOI, some manoeuvres are executed to correct launcher insertion errors and a non-optimal declination of the hyperbolic escape asymptote (due to geo-political constraints of the launch from French Guiana). Then, ZEUS-15 thrusts during 1 complete revolution around the Sun before approaching 2006 RH_{120} at a distance of 2000 km where the proximity phase begins. Trajectories involving more than 1 revolution were ruled out due to time constraints. A porkchop plot with the launch opportunities in the 2025-2045 interval is shown in Figure 1.

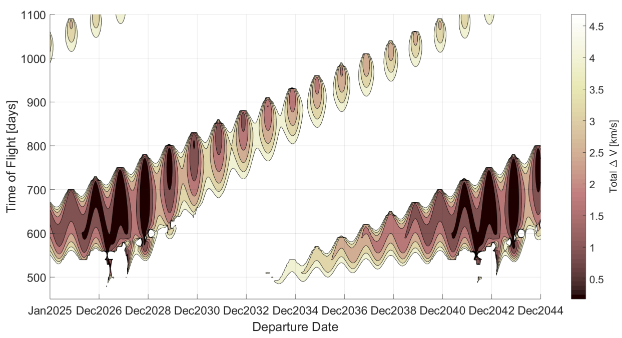


Figure 1: 1-revolution transfer opportunities to 2006 RH_{120}

[1] https://www.minorplanetcenter.net/data

3.3 Proximity Operations

The proximity phase concerns all manoeuvres that occur while the spacecraft is in close proximity to the asteroid. To model the asteroid approach and rendezvous, the Clohessy-Wiltshire equations [7], typically used for satellite rendezvous, were used.

The phase begins when the asteroid can first be seen by the spacecraft's on board camera, at a distance of 2000 km. The spacecraft executes a manoeuvre relative to the target that brings it to 30 km away from the asteroid at which point it orbits the asteroid 5 times for science-data gathering purposes. Once completed, the third operation begins, where the spacecraft targets a point trailing 500 m behind in the asteroid's orbit to perform the planetary defense investigation (see section 4.3.2), while determining the asteroid's spin axis and alignment. Finally, for a successful rendezvous to be completed, the main axis of the spacecraft must be aligned with the asteroid's spin axis [9]. In the worst case scenario, where the asteroid spin axis is orthogonal to its plane of motion, the spacecraft needs to travel through a right angle triangle to complete the rendezvous, until the asteroid is grabbed and secured. A detailed definition of the final rendezvous operation is included in 5.1.

3.4 Hauling

The hauling of the asteroid entails an insertion into a hyperbolic invariant manifold leading to a periodic orbit in the L_2 point, where families of planar, vertical and halo orbits and their associated manifolds have been considered [11]. The insertion point with the lowest associated ΔV corresponds to a southern halo orbit and is selected as the target for the hauling of 2006 RH_{120}. The obtained trajectory has a thrusting phase and a cruise phase. During the first, the engines of ZEUS-15 will be firing to reach the desired branch of the manifold. Afterwards, during the cruise phase, ZEUS-15 will follow the slow dynamics of the manifold until the arrival at the final orbit, presented in Figure 2.

4 The Spacecraft

The ZEUS-15 was a spacecraft designed to move a small asteroid regardless of its composition within the requirements of the mission set out in earlier sections.

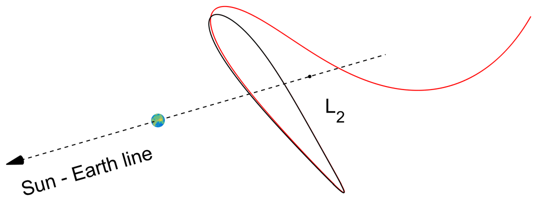


Figure 2: Insertion Halo Orbit at L_2 (black) and its associated manifold (red). The Earth's size is exaggerated for visualisation purposes.

4.1 Propulsion & Attitude Control

Propulsion is achieved using electric thrusters. Mass consumption for the hauling manoeuvre with chemical propulsion would be 6.5 times higher, requiring either the selection of a smaller asteroid or multiple launches. For that reason a more complicated low-thrust trajectory was designed. Moreover, since illumination is not a problem for the mission, a stable amount of solar electric power is available for the engines.
Therefore, the engines chosen were one RIT 2X by Ariane Group and 12 (due to their short lifetime in a 15 year mission) SPT 140 by OKB Fakel, plus 2 more SPT 140 on arms used for planetary defence (described in section 4.3.2). The total propellant (xenon) mass is 3290 kg which is stored at -1 Celsius in 7 tanks by MT Aerospace (4 of M-XTA/180l and 3 of L-XTA/900l): the biggest commercially available xenon tanks on the market. The propulsion operations are described in Table 1.

4.2 Power

Power is generated by two rectangular solar panel arrays of 72.8 m^2 combined area and a combined mass of 318.8 kg. The arrays are sized according to the largest expected power load; 17,237 W during the planetary defence experiment. 17,676 W shall be available at BOL while EOL shall have 16,396 W. This design meets the power demands of the mission.

Energy is stored by 4 ABSL 8s104p 28V 156Ah Li-Ion rechargeable batteries. A fifth battery is added for redundancy. The 4 batteries provide 18,870 Wh to accommodate the energy demands of asteroid capture phase of 18,889 Wh; assuming no solar power generation. Limited solar generation is expected during this phase, explained in section 5.1.2

Power distribution is handled by a PPT and decentralised system. Fuses connected in series with the power bus and fault detection circuits are included to

Phase	Duration	ΔV (m/s)	Propellant Mass (kg)	Thrust (N)	Engines Firing	Power Required (W)
Inclination Change	50.72 days	347	138	0.562	SPT 140	9000
Outbound	762 days	734	153	0.00776	RIT 2X	2300
Initial Chase	4 days	11.8	2.43	0.088	RIT 2X	2500
Pseudo-orbits	7 days	20.5	4.24	0.088	RIT 2X	2500
Proximity approach	11.8 hours	1.44	0.297	0.088	RIT 2X	2500
Planetary Defence	16.2 days	-	62	0.193 x 4	SPT 140	13314
Hypotenuse	2 hours	0.264	0.101	0.306	SPT 140	5750
Orientation and Spinning	5.5 hours	-	1.58	-	SVT 01	0
Opposite	1.66 hours	0.222	0 .0846	0.306	SPT 140	5750
De-Spinning	23 hours	-	56.1	-	SVT 01	0
Hauling	1390 days	135	2800	0.416	SPT 140	7400
ADCS	2500 hours	-	25.2	-	HT100	600

Table 1: Thrusting manoeuvres and fuel consumption.

enable isolation of faults and electrical shorts. A top-level mass estimation of the system places the mass of the electrical harness as 115 kg, power regulation 90 kg and PCU 15 kg.

4.3 Payload

To achieve the objectives of the Science Committee, which provides the three secondary objectives of the mission, and to explore possible methods of additional funding for the mission, three separate payload elements are included: science, defense and ISRU.

4.3.1 Science

A demand from the mission is the ability to visually spot the asteroid as soon as possible, allowing for a close approach through orbital corrections. This optical manoeuvring will use a camera with a high resolution chosen as 1.34 μRads. However, once close to the asteroid, the camera‘s field of view would not allow for full frame images of 2006 RH120, requiring a second camera for close-up imaging.

Planetary science covers a wide range of fields including solar physics, volcanism, geophysics and atmospheric science. The scientific instruments on board will provide information of the physical, compositional and environmental properties of 2006 RH120. A LIDAR, UV Spectrometer, X Ray Spectrometer, IF Spectrometer, Sample Distributor and Analyser, Magnetometer and Dust Imager are the instruments selected to perform this task.

4.3.2 Defence

An Ion Beam Shepherding experiment shall be conducted to demonstrate a slow push asteroid deflection technique. A small modification in the orbit of an asteroid heading towards Earth, with enough time in advance, could deter a collision from occurring.

Designed for a 3 standard deviation from $2006RH_{120}$'s mass and size, the experiment shall fire two SPT-140 Ion thrusters at the asteroid while using another two thrusters to counteract this motion, holding the spacecraft stationary relative to the asteroid. The momentum of the quasi-neutral beam of particles shall impart a deflection of 1 mm/s onto the asteroid. The experiment shall last 16 days with a total cost of 62 kg Xenon propellant, based on a mass of 366.2 tonnes and geometric diameter of 6.5 m. The deflection is measured using a combination of star trackers and sun sensors and a LIDAR system. The latter measures the distance between the asteroid and the spacecraft and uses this information to maintain the 500 m distance, feeding the information into a control loop with the thrusters. With a fixed distance between asteroid and spacecraft, changes recorded in the orbital elements of the spacecraft by the star trackers and sun sensors represent a change in the orbital elements of the asteroid-spacecraft system.

This technique is used to increase the maturity of this technology and proving its usability for future hypothetical planetary defence mission.

4.3.3 ISRU

The ISRU payload was used as a technology demonstration opportunity for ISRU techniques, exploring the functionality of technologies which could extend the life of spacecraft visiting asteroids in-situ and potentially provide the mission with extra funding. The technologies demonstrated consist of:

- Using the asteroid's water content, extracted from its minerals using a drill, as a method of propulsion through the use of a hydrolysing thruster.

- Using the asteroid's regolith (the loose surface material) as a propellant by inserting it into a pulsed plasma thruster.

- Using asteroid regolith as a structural member, such as a brick, by melting the material with microwave radiation and moulding it into a relevant shape.

These payload technologies explore just a limited use of asteroid materials, de to the imited payload space available. The equipments used for ISRU experiments are: a water gathering Auger Drill, a HYDROS hydrolysing thruster by Tethers Unlimited, a BUSEK Pulsed Plasma Thruster, a solid state microwave emitter, and the Sample Distribution System shared with the science payload.

4.4 Communications

Various values used were deduced for simplicity. These included passive loss (-5 dB), system noise (Ka:80 K, X:30 K), phase modulation index (1.2) energy per bit-to-noise density ratio (1 dB) and margin (3 dB) [2].

A 20 GBit hard drive was selected for storage based on the data generated by the payload instruments. Due to demands of the deep space network, downlinks are expected to be transmitted in one hour slots. This requires 6 Mbps which is reasonable for science data [2]. For housekeeping, 1 Gbit weekly download is planned, resulting in 278 kbps data rate. Running the link budget with all previously discussed considerations lead to output of a 0.5 m K-band antenna and a 0.5 m X band antenna. From the output RF power predicted from the link budget, Ka antenna input power exceeds 500 W after asteroid arrival where more power is made available to communications as thrusters are not operational.

4.5 OBDH

The driving requirement put on the OBDH subsystem for the mission was the spacecraft's autonomy: its ability to carry out difficult tasks such as the close approach without intervention from the Earth, and being able to do so while performing updates on its main computer or if the main computer is non-functional. This led to the complex bus-star hybrid architecture that will connect a star network of the AOCS subsystem, centered on a pair of processors, and all of the other subsystems to the two main computers via a bus. Components for this architecture were determined to meet these requirements, and were chosen from BAE Systems' range of space-ready hardware [4]:

- Main computing unit: RAD750 CPU (processor), HISTACK RAM (payload data), Magnum RAM (OS), CRAM (Software) and PROM (OS)
- AOCS computing unit: RAD750 CPU (processor), TITAN STACK RAM (image processing).

4.6 Capture Mechanism

The capture mechanism has to capture the asteroid so that relative movement between itself and the spacecraft is prevented. Material should not escape at any time and the spacecraft including the capture mechanism has to fit in the Ariane's 64 fairing in stowed position.

The mechanism and its configurations is shown in Figure 3 and is composed of the following parts:

Structural beams There are 8 structural beams, 1 in each corner of the top of the spacecraft, designed to withstand and transfer the loads from the asteroid and the friction bags, giving integrity to the mechanism. The beams have a hollow cylindrical cross-section of 1.5mm thickness to adapt and withstand forces coming from all directions and availability to use the inside part for the feed systems and cables. The beams are divided in 5 sections joined by 1DOF joints that allow the mechanism to radially open and close to capture the asteroid. Each of the arms composing the mechanism has a total deployed length of 12.6 m.

Friction bags The friction bags are made of Vectran^{TM}, a material that has been used in planetary landing systems in several missions [5], has good resistance to abrasion and works well in a very wide range of temperatures. The capture mechanism includes 2 different types of friction bags:

Firstly, 24 small friction bags, 3 in each structural beam, hold the asteroid in place inside the mechanism and transfer the forces and momentums to the structural beams. Such bags are 0.75 m in diameter and protect the structural beams from any contact with the asteroid.

Secondly, 2 large friction bags with 1 m diameter on top of the spacecraft protect it from a possible collision with with the asteroid.

All the friction bags are inflated with Xenon during the capture phase, the same gas used by the propulsion system. After the asteroid is secured, the bags are be inflated with Urethane foam to avoid leaks and outgassing of the Xenon gas.

MLI fabric The mechanism is covered from the outside using 390 m^2 of MLI [6] , preventing the leakage of asteroid material and protecting the inside from radiation and temperature gradients, a critical aspect due to the quick degradation of Vectran ™under UV radiation [5].

String-bag closure systems In order to ensure that the asteroid is completely encapsulated, a mechanism based on string-bags is used. Specifically, 2 cables pulled by a rotary actuator firmly close the capture mechanism.

4.7 Structures - Configuration

The design of ZEUS-15 have been influenced by nearly the totality of the requirements of the mission, as their proper evaluation will be key to ensure the completion of the mission objectives.

4.7.1 Requirements

The main requirements that directly have affected the configuration of the spacecraft are:

- The S/C (spacecraft) shall allocate and ensure all the components and instruments of the system without disturbing their performance.
- The primary/secondary/tertiary structure shall not experience any plastic deformation, break or buckling at any stage of the mission (specially during launch).
- The S/C structure and mechanisms shall ensure the grabbing of the asteroid during manoeuvres and hauling trajectory.
- The S/C shall fit in the Ariane 6 launcher fairing and meet at the same time their requirements (including static and dynamic loads, mass and CoG limits and safty factors).

The configuration of ZEUS-15 has been then driven by the launcher selection, the multiple susbsystem requirements (to determine their optimal location in the S/C) and the propellant mass and volume.

4.7.2 Configuration Design

The three possible configurations of ZEUS-15 are "fully deployed", "launch" and "hauling" (figure 3) and have been designed taking into consideration the different stages of the mission.

The most limiting elements (the ones that have determined the final S/C width and height) have been the fuel tanks, with more than 50% of the total S/C mass and more than 14% of the internal S/C volume. Once the distribution of those has been determined the rest of components have been allocated either in the interior or the exterior of the main S/C body.

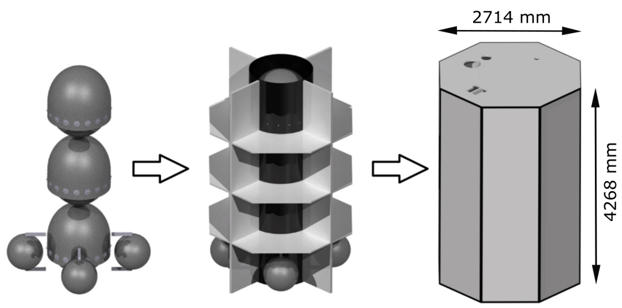
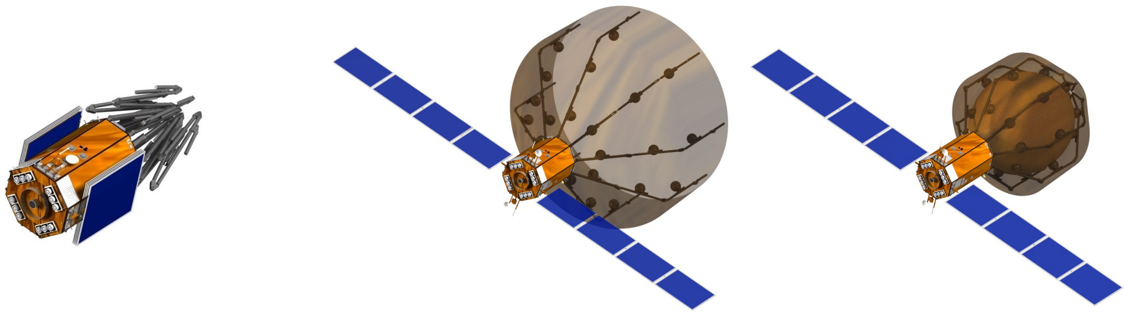


Figure 4: Fuel tanks and ZEUS-15 skeleton. Representation of how the tanks have driven the final dimensions of the S/C.

Capture Mechanism The grabbing mechanism is placed at the top of the S/C mainly to distance it from the LV adapter (which will be at the aft end) and reduce loads and vibrations produced during launch.

Solar Arrays ZEUS-15 will possess a total solar panel area of 72.8 m^2 consisting in 8 rectangular panels (4 small panels in each side) connected by a single hinge. The rectangular shape has been considered optimum for ZEUS-15 (instead of a circular shape) taking into account both structural and launcher constraints.

Antennae The S/C will carry 4 antennas (2 X-band low gain and 2 X/Ka-band high gain ones), which have been located at the bottom part of the S/C increasing their coverage and minimizing the blockage with the vehicle. Their respective amplifiers have been allocated in one side of ZEUS-15, reducing the amount of wires and signal losses.



(a) Launch configuration (b) Fully deployed configuration (c) Hauling configuration (+ aster.)

Figure 3: Spacecraft configurations overview

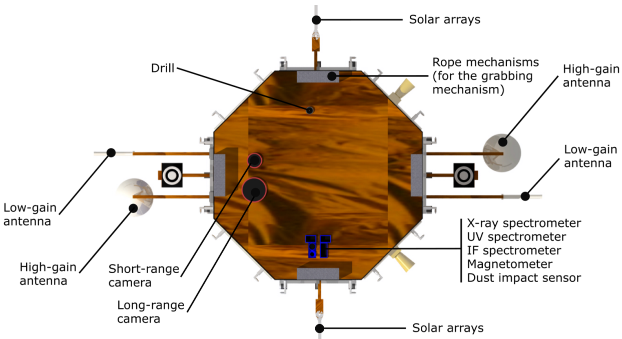


Figure 5

The antennas, as well as the solar panels, will use deployable arms made of composite (avoiding thermal gradients and minimizing pointing errors) which have been kept as low as possible to keep fundamental frequencies high enough.

Payload Equipment Most of the instruments have been placed at the top of the S/C in order to have a direct line of sight to the target, especially during near asteroid operations. After grabbing, the asteroid will block the cameras which will no longer be needed for visual navigation, but will be accessible to the auger drill and sample acquisition system.

In red (figure 6), the long/short-range cameras; in blue, the spectrometers, magnetometers and dust impact sensors; in grey, the drill (top) and in green, the PPT thruster, microwave and sampler distribution and analyzer. At the bottom of the S/C, in yellow, a small box can be seen which correspond to the LIDAR system used for planetary defence purposes.

ACS ZEUS-15 will be equipped with 2 star trackers, 4 sun-sensors, 12 electric thrusters (ACS), 16 cold gas thrusters (to spin the S/C) and 2 IMU's. All the items have been placed over the external surface ex-

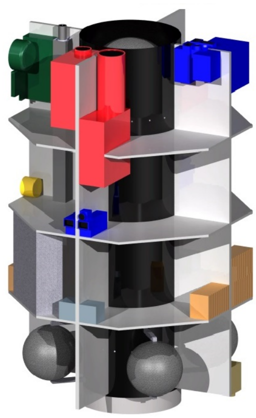


Figure 6: Perspectives of ZEUS-15 internal configuration

cept for the 2 IMU's, that have been located in its interior.

OBDH The OBDH items (processors and memories) have been centralised in a common box in order to reduce cabling and protective material. These components are of utmost importance, so they are placed as far as possible from the external walls to reduce the damage from possible impacts or radiation.

Power Subsystem The 5 batteries of ZEUS-15 (orange boxes in Figure 6) are quite heavy elements (more than 50 kg each) and therefore have been placed in the aft of the S/C to lower the CoG height. In addition, they are located in the same sides as the solar panels, reducing cabling and electrical losses.

Propulsion Subsystem 12 SPT140 engines (3 different generations) and 1 central RIT2X have been placed at the bottom of the S/C (for propulsion) while 2 additional SPT140 have been placed pointing towards the asteroid (for planetary defence purposes). Placing the engines at the back of the S/C prevents possible interactions with other components and can be thermally isolated more easily. The fuel tanks will be distributed in a mixed configuration, with 3 big tanks inside the central tube (L-XTA/900l Family) and 4 smaller ones radially at the bottom (M-XTA/180l Family), again for CoG placement.

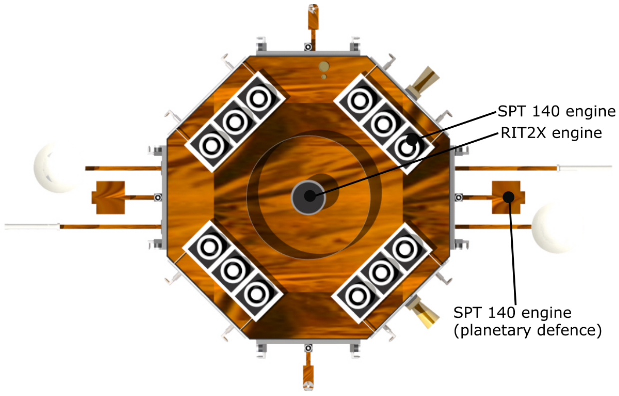


Figure 7: Propulsion module configuration, back view

Thermal Subsystem The Thermal subsystem is formed by the multi-layer insulation (MLI) that covers the exterior of ZEUS-15, tanks and little arms, and also the different radiators that dissipate the heat from the rest of components.

4.7.3 Structural Properties

For the centre of mass, there are two main constraints in terms of position imposed by both the launch vehicle adapter and the launcher selected. The final CoG of ZEUS-15, for the 3 possible configurations, is gathered in the following tables:

Folded configuration (launch)		
CoG x	2120	mm
CoG y	-6.0	mm
CoG z	7.3	mm
ϵ	0.48	$^\circ$
CPz (Sun side)	3840	mm
CPz (90° side)	3845	mm

Table 2: S/C properties for folded configurations

Max. aperture configuration		
CoG x	2123	mm
CoG y	-9.3	mm
CoG z	-0.9	mm
ϵ	1,78	$^\circ$
CPz (Sun side)	5860	mm
CPz (90° side)	6750	mm

Table 3: S/C properties for max. aperture configuration

Grabbing configuration (+ 336 t asteroid)		
CoG x	8780	mm
CoG y	-0.08	mm
CoG z	-0.04	mm
ϵ	-	$^\circ$
CPz (Sun side)	3860	mm
CPz (90° side)	6150	mm

Table 4: S/C properties for grabbing configuration

ϵ represents the angle between the S/C longitudinal geometrical axis and the principal roll inertia axis, and should be kept below 1° (in launch configuration) for good dynamic balance. In addition, the CoG in the lateral axes is at no more than 30 mm from the launcher longitudinal axis, ideal for good static balance.

The structure of ZEUS-15 will be composed of a central tube made of graphite/epoxy and different panels made of aluminium honeycomb [8] that will distribute the forces and deformations homogeneously. The design consists of 8 vertical walls in the exterior (one for each side of the octagon), 4 vertical panels in the interior (connecting the central tube with the external walls) and 5 horizontal panels, 3 in its interior and 2 for the bottom and top parts of the S/C.

Finally, the entire ZEUS-15 structure has been modelled and analysed through a simple longitudinal load case (throughout a NASTRAN analysis) and the results showed that the S/C is perfectly able to sustain the critical loads and deflections during launch and orbital manoeuvres.

4.8 Thermal

ZEUS-15 spacecraft's thermal control system (TCS) mainly relies on passive thermal insulation and passive thermal control with thermal straps and heat pipes. The system is also controlled by an active heating system that assures being above the allowed lowest temperature.

The spacecraft is insulated with MLI blanket [6] with 15 layers to a total external covered surface of 45.4 m^2. In addition, 5.4 m^2 of radiators are placed in all its walls to reject the heat. This radiators are made by Optical Solar Radiators (OSR) or white painting depending on the performance and heat loss of the instruments.

The most critical instruments are the electrical engines due to the high thermal dissipation that they offer. For these instruments, a heat pipe is used as well as a thermally decoupled system in order to insulate them from the main S/C body.

4.8.1 Thermal Architecture

Figure 8 presents a simplified overview of all the instruments on the S/C and the thermal interface of these with the radiators and the main body. The instruments are accommodated inside ZEUS-15 (close to the external walls) and can be connected to their radiators by using thermal straps or heat pipes. Additionally, high dissipation instruments are decoupled from the main body.

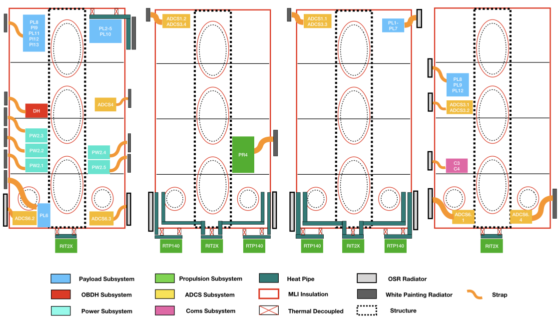


Figure 8: Thermal architecture of ZEUS-15

4.8.2 Thermal Analysis

A thermal analysis of the S/C has been done using ESATAN-TMS software including four major scenarios: Near Earth Outbound Trajectory, Outbound/Hauling Trajectory, Close Approach, and Grabbing.

The results from the thermal analysis showed that the worst case scenario is the grabbing manoeuvre but the S/C is able to sustain all the cases with the present thermal control systems and isolators.

5 Systems

The systems engineering approach applied in this project has been divided in three main packages: Operations, Budgets and Risks.

5.1 Operations

The mission is launched from CSG, Kourou (French Guiana), on 12/11/2029 using the Ariane 64 launcher on a direct escape trajectory from Earth. ZEUS-15 is to rendezvous with the asteroid 2006RH_{120} on 20/07/2032 and spend 823 days within its vicinity, until the capturing manoeuvre is initiated on 21/10/2034. After the asteroid is encapsulated, the spacecraft is to spend another 1346 days until it commences its return on 07/07/2038. In addition, a solar conjunction will occur between 17/12/2035 and 06/09/2036 (264 days) where all communications with the spacecraft will be at a halt. It shall reach the south halo orbit around L2 on 28/06/2044.

The different mission phases can be viewed in table 10 with only the proximity operations being elaborated in the subsection that follows.

Date	Duration (Days)	Mission Stage	Mission Phase	Overview
12/11/2029	8	Outbound Stage	Launch & Earth Escape	S/A - HGA Deployment, S/C stabilisation, Instrument Status Checks
21/11/2029	180		Commisioning & Early Cruise	Continuous Thrust (Inclination Change & Outbound), Commissioning of Payload & Capture Mechanism
21/5/2030	723		Late Outbound Cruise	Continuous Thrust until halt - Day 824 (Outbound), Correction Manoeuvers, Payload Checkouts
14/5/2032	42		Asteroid Detection	Insertion into 2006RH120's vicinity (2000km), Establish Visual Contact, Asteroid Ephemeris Measurements & Refinement
26/6/2032	54	Near Asteroid Stage	Initial Approach & Asteroid Recon	Approach Manoeuvre (30km), Instrument Commissioning & Asteroid Characterisation, 5 Fly-Bys (180m), Approach Manoeuvre (500m)
20/8/2032	760		Science & Planetary Defence	1st Science Phase - 365 days, Ion Beam Shepherding Technique Demonstration - 30 Days, 2nd Science Phase - 365 days
20/9/2034	22		Proximity Ops & Capture	Capture Mechanism Deployment & Checks - 14 days, Approach to Capture Manoeuvers - 1 day (Match 2006RH120's Spin Rate - Approach & Capture), Despin & Stabilise - 1 day
13/10/2034	1363		Safekeeping & ISRU Science	Drill Samples, ISRU Experiments, Solar Conjunction - 264 days
7/7/2038	1400	Hauling Stage	Initial Return Cruise	Continuous Thrust until halt - Day 4549 (Hauling)
8/5/2042	190		Interim Return Cruise	Precision Tracking, Correction Manoeuvres
15/11/2042	591		Late Return Cruise to Halo	S/C Follows Manifold, Insertion into South Halo Orbit around L2

Figure 10: Important Dates and Operational Phases of ARTeMis.

Initial Approach & Asteroid Reconnaissance

An initial manoeuvre is performed to bring the spacecraft 30 km away from the asteroid. A week is then spent for instrument commissioning and calibration to ensure that the instruments are fully operational in the environment of the asteroid's vicinity.

Asteroid characterisation then begins from a leader-follower formation for an initial estimate of its shape, size and rotational axis. This should last for 10 days, resulting in approximately 5500 revolutions of the asteroid.

ZEUS-15 then proceeds to realise the 5 data-gathering pseuo-orbits around 2006RH_{120}, passing within 180 m at its closest. During the 7 days of controlled fly-bys, most payload instruments operate intermittently: the magnetometer, the UV, IF and X-ray spectrometers as well as the dust analyser.

After orbiting the asteroid, the spacecraft returns to a position 30 km trailing the asteroid, and approaches to 500 m away in a manoeuvre taking 11.8

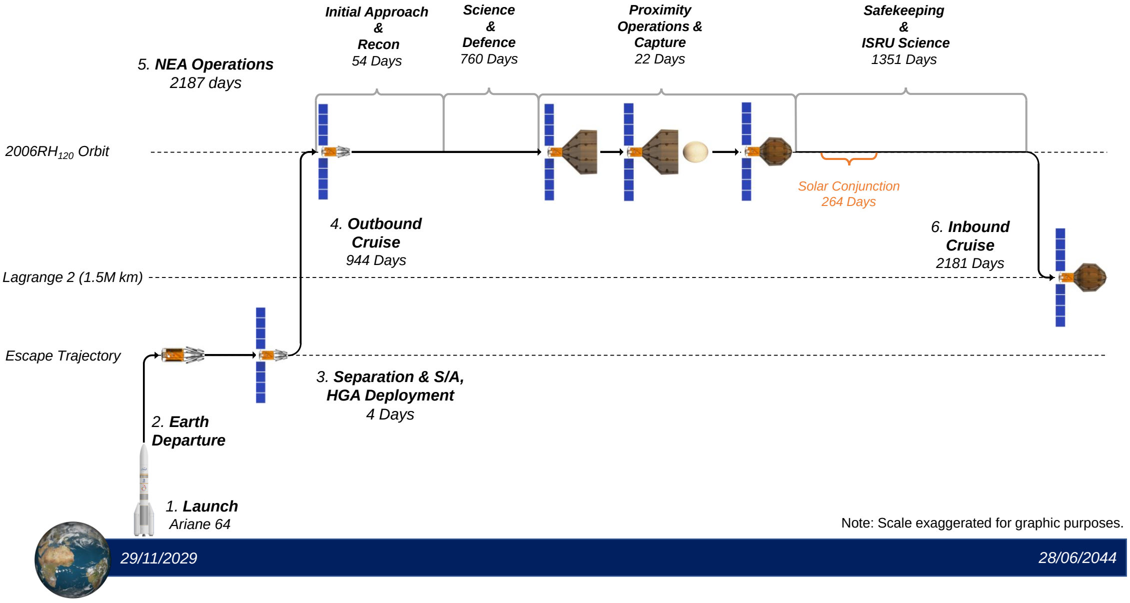


Figure 9: Mission timeline of ARTeMis

hours. Allowing a margin of two days before this operation, 24 h coverage is established. During this approach, the long range optical camera (LROC) will take images of the asteroid at various shutter speeds, thereby allowing more accurate estimations of its spin rate.

A margin of 20 days is applied due the complexity of this phase and the numerous factors that might contribute to hindrances.

5.1.1 Science & Planetary Defence

With ZEUS-15 hovering 500 m away from $2006RH_{120}$ the first science sub-phase begins. The payload instruments will be operating on a 12 weekly rotation schedule, as seen below:

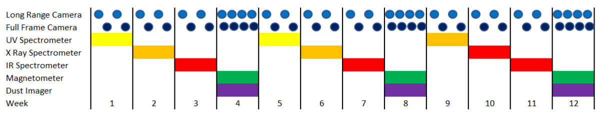


Figure 11: Science Instruments Duty Cycle

The LROC and the full frame optical camera (FFOC) build up a 3-D mapping model. Since a resolution of millimetres will be possible from that distance,some idea about the regolith composition can be obtained. Due to the proximity to the asteroid in addition to the long duration of this phase the data generated will provide a thorough knowledge of the asteroid's surface.

Following the science phase, the planetary defence demonstration will take place, trying to deflect the asteroid using an Ion Beam Shepherding technique. The spacecraft pivots, its rear facing the asteroid. It then simultaneously fires its main engines and its planetary defence engines (PDE) to achieve a balanced thrust vector. This way, the spacecraft remains stationary with respect to $2006RH_{120}$ and a deflection technique is demonstrated: the xenon ions that collide with the asteroid cause a minor displacement resulting in alteration of its trajectory, which is measured using a LIDAR system situated in the rear of the spacecraft. For this process to be measurable, approximately 16 days of operations are required. Due to the flexible time constraints at this stage of the mission, a 100% margin is applied, resulting in a 30-day total operation. 24h daily coverage is required, since malfunctions in the propulsion system might result in catastrophic mission failure.

After the planetary defence demonstration another identical science sub-phase is realised for one year. Comparison of the results of the two science phases can provide valuable data on the radiation degradation of the equipment in addition to the altered environment of the asteroid's surface due to the xenon ion bombardment.

5.1.2 Proximity Operations & Capture

The grabbing mechanism deploys and is tested thoroughly to minimize the faults that might occur through the process. With a detailed 3-D map of the asteroid, simulations of the capture procedure are run at the MOC multiple times. This stage should take approximately 2 weeks.

At 500m away, a linear approximation can be applied to the motion between the spacecraft and the asteroid. However, this can only be valid when the proposed linear motion doesn't deviate largely from the orbital path. It was assumed that motion taking a longer time than it takes for the asteroid to move along 1° of its path will invalidate the linear approximation, implying a limit of 27 hours for proximity operations. If given a go command, the spacecraft will act autonomously throughout the whole capture procedure. Its dedicated on-board processor for capture will take over and perform all required actions, including encapsulation of the asteroid in the capturing mechanism, secure closure of the open end of the mechanism, and controlling the pressure of the Vectran bags accordingly. This process normally lasts for 4 h with 100% percent margin in case of difficulties. Since ZEUS-15 will be spinning at this stage, the solar arrays cannot be relied on to be the main power source. Thus the on-board batteries will take over. If power levels drop below a certain threshold, the spacecraft will retract to its initial position 500m behind the asteroid and despin, so as to regain power and controllability, while the situation is assessed by ground control. Possible safe modes have been identified to tackle such issues, found in 13. The required manoeuvres are shown in figure 12.

As the rotation of the spacecraft to match the asteroid's spin rate can result in intricate illumination patterns for the solar arrays, an assumption has been made of zero power generation as a worst case.

Once $2006RH_{120}$ is firmly encapsulated in the grabbing mechanism, the cold gas thrusters are used to despin the spacecraft in a total of 36.7 hours. Reaching the latter stages in this operation, the spacecraft will be spinning extremely slowly, therefore the assumption is made that power can be fully generated by the solar arrays.

5.1.3 Safekeeping & ISRU Science

With the asteroid firmly in place, the time window until the optimal departure date is given to ISRU science which can keep being performed at a lesser rate throughout the hauling trajectory.

At 17/12/2035 a solar conjunction will occur, which due to lack of noise modelling has been assumed to disrupt all communications between Earth and ZEUS-15 within a 5° window [?]. It ends in 6/9/2036, lasting a total of 264 days. During this period, the extra magnetometer, UV and X-ray spectrometer on the side of the spacecraft will point at the Sun, carrying out investigations of the Solar environment. Measurements will be taken in allocated time periods until the maximum storage limit of 20 Gbit is reached.

5.2 Budgets

It is crucial to identify the properties that will affect the mission and define a representative list of budgets, taking into account the key drivers and the requirements: Mass, Volume, ΔV, Power and Data.

The need of a common margin policy rises as part of systems engineering effort to manage risk and performance. ESA margin philosophy [22] has been applied at different levels: at design level, at equipment level, and at system level. The ARTeMis design is intended to achieve phase 0 maturity, therefore conservative allocations, design and sizing have been applied.

5.2.1 Mass

Spacecraft mass is not only one of the main drivers of ARTeMis, but is an input for calculations involving propellant, structures, the ADCS, launcher capabilities and parametric cost model. A bottom-up model is adopted based on the list of components necessary for each subsystem of the S/C. The margins applied to the mass calculation are applied at equipment and system level. The mass data shown in table 5 has incorporated the equipment level margins.

Subsystem	**Amount [kg]**	**Percent of total.**
Structures	1330	41.20%
Thermal	231	7.14%
ACS	25	0.77%
Propulsion	615	19.03%
Power	731	22.62%
OBDH	97	3.01%
Comms	42	1.30%
Payload	159	4.92%
Nominal dry mass	3230	100%
Total dry mass	3877	120%
Propellant	3285	
Total mass	7162	

Table 5: Spacecraft mass breakdown

The structures subsystem stands out from the rest, taking around 41% of the nominal dry mass of

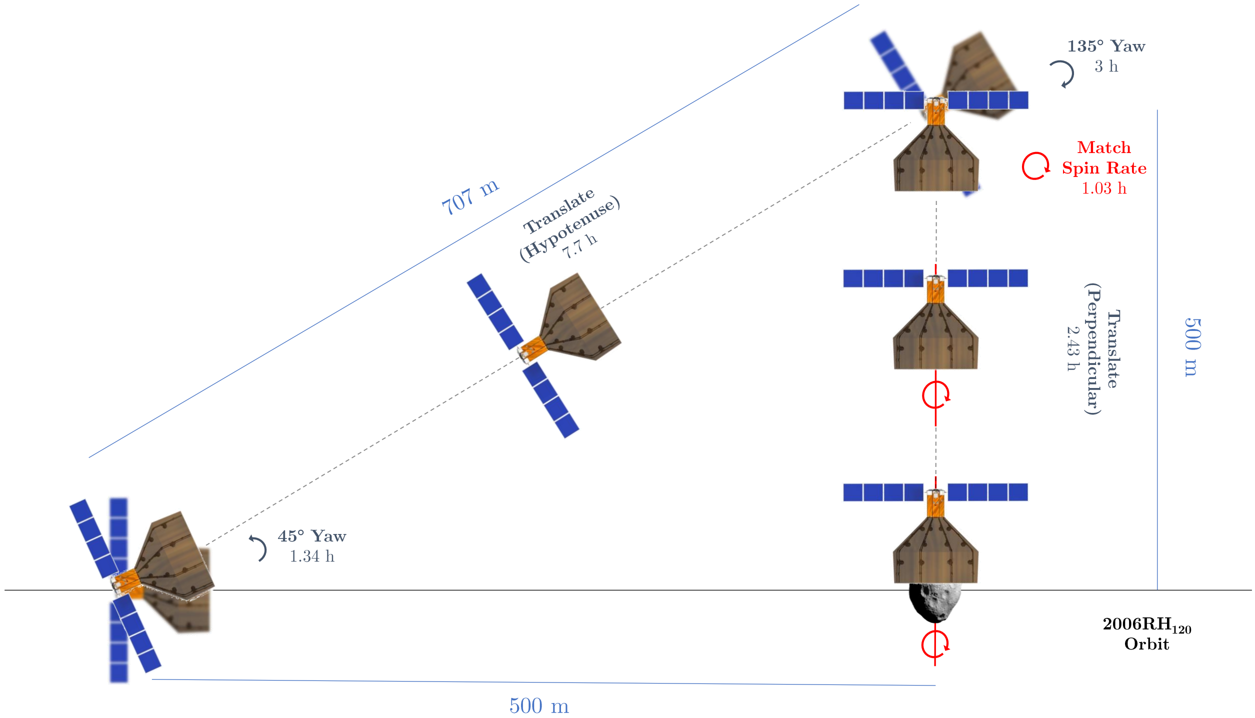


Figure 12: Approach to Capture Manoeuvers

the S/C. The grabbing mechanism, which has a mass of 485 kg, is designed from scratch with the objective of being highly adaptable and strong. It is a complex mechanism critical for the success of the mission and its unique nature leads to a large mass.

The propulsion subsystem has also great importance in the mission. Taking into account that ARTeMis has as objective to move an asteroid which is more than 50 times more massive than the S/C itself, it is also a critical part of the concept with a lot of redundancy. Due to the use of electric propulsion, the power subsystem needs to achieve really high performance, leading to around 23% of the nominal dry mass.

It is also remarkable the amount of propellant carried in the S/C. It is calculated, with all the margins applied, that the amount of propellant needed is 3285 kg, close to the dry mass. Even considering that hauling trajectory is one of the phases with the lowest ΔV, most of the propellant is expended due to the increase of mass when the asteroid is grabbed.

5.2.2 Cost

Cost budget is one of the most important factors that affect the success or failure of engineering projects of different sectors. ESA states the cost it is willing to cover is 1.1 billion €; however, the opportunity of additional funding is also opened.

Unmanned Space Vehicle Cost Model is one of the most popular and studied models for space industry. Developed for the United States air force in 2002, it is a parametric model that has the mass of the different subsystems as input to estimate recurring and non-recurring costs of the subsystems of the mission.

The model requires some modifications and improvements to adjust it to the desire mission. The main changes implemented for ARTeMis have been: currency change, inflation factor ([2] table 11-31), cost Construction model (COCOMO 81) and other minor factors at programmatic and management level.

The cost breakdown of the elements of the mission and system is shown in table 6.

The final estimated cost of ARTeMis is approximately 1.36 B€, which leaves a gap of 260 M€uncovered. A usual practice in L-class mission is obtaining additional funding looking for additional stakeholders; for example, national agencies of the member states, institutions or universities develop and integrate the payload instruments. The involvement of general public in the mission is a recent method; for example, the "Hot ticket" [27] initiative by NASA, where you can send your name to the Sun in the "Parker Solar Probe" mission.

System	Amount [k€]	Percent.
S/C	473,082	42.25 %
Structures and Thermal	159,797	
ACS	15,970	
Power	84,217	
Propulsion	48,977	
OBDH	53,916	
Comms	40,494	
Payload	69,711	
Integration and testing	141,925	12.67 %
Project management	94,616	8.45 %
Systems enginering	70,962	6.34 %
Assurance	52,039	4.65 %
Launch	90,000	8.04 %
Operations	141,925	12.67 %
Cubesat	1,344	0.12 %
Propellant	3,942	0.35 %
Reserves	50,000	4.46 %
Total cost	1,119,836	100 %
Total cost + margins	1,360,119	

Table 6: Mission cost breakdown

Risk	Mitigation technique
Launch vehicle failure	Ensure selection of reliable launch vehicle: Arianespace ensures a high reliable of Ariane 6, backed up by their very successful launch history.
Solar panels: individual cell failure causing string of cells to fail	We have introduced diodes to bypass any failed cells avoiding the failure of a full string of cells.
Data bus failure due to debris or electrical short	We have placed the data bus as far as possible from the walls of the spacecraft for protection.
Loss of navigation data for extended periods of time	Added additional batteries for Random-Access Memory (RAM).
Main engine failure	There are two extra SPT 140 engines for redundancy.

Table 7: Five highest individual risks and the mitigation actions taken

5.3 *Risks*

Individual team members identified risks and failure scenarios with a defined probability, and impact on the mission. The risks posing the greatest threat to the mission's objectives, cost, and schedule, and their mitigation strategies are shown in Table 9.

5.3.1 *Impact with The Earth*

Analysing potential failure scenarios within subsystems allows the identification of the areas of the mission with the highest risk. The Trajectories, Configuration and Close Approach & ACS work packages proved to have the greatest threat and therefore these risks are mitigated using safe modes. Safe modes are an automated process on board the spacecraft during which all non-essential systems are shutdown and the spacecraft follows very specific commands. A brief overview of the safe modes can be seen in Figure 16, with the inclusion of a defence safe mode, for use in the very unlikely event that during hauling our spacecraft and the asteroid have a probability of colliding with the Earth.

The design of the spacecraft while holding the asteroid features a wide blunt nose with a high mass concentration. This gives the spacecraft a typical reentry vehicle design and a high risk the spacecraft will not burn up in the atmosphere should a system failure lead to an uncontrollable Earth collision trajectory. Pyrotechnics are employed to detach the arms from the main body of the spacecraft; releasing the asteroid. This will produce only a few discrete parts, reducing the risk of hitting a satellite, which are likely to burn up individually in the Earth atmosphere. Burn up in the atmosphere protects terrestrial infrastructure and populations.

6 Discussion

6.1 *Back-up Missions*

With the current mission architecture and the constraints placed on the operational time and the end date, no back-up missions can be planned in case the launch date is delayed or 2006 RH_{120} cannot be reached. However, by moving the allowed mission end-date back to 2055, two back-up missions for asteroids 2013 RZ_{53} and 2018 AV_2 could be planned and successfully completed with the same spacecraft configuration.

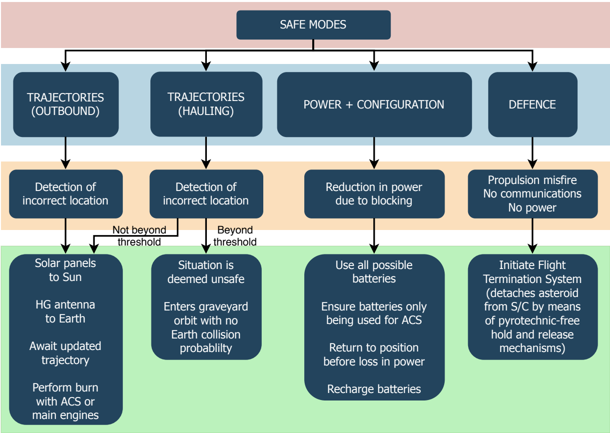


Figure 13: Safe modes to mitigate subsystems with greatest risk

6.2 End-of-Life Strategy

Although safe modes have been designed to prevent any risk of hitting the Earth during the hauling of the asteroid, no considerations have been taken on the effects of the perturbations (e.g. gravitational, solar radiation pressure) in an unstable periodic orbit. On that matter, the work by Georgantas [16] defines a model to obtain the probability of a body in a planar periodic orbit to hit the moon or the Earth, by applying random errors to its position and velocity (bounded between 10 km and 1 km and between 10 cm/s and 1 cm/s, respectively) through the patched 3-body method [15] and a Monte-Carlo simulation (N=20000). Using such simulation model for the selected southern halo orbit shows that the majority of the resulting trajectories (>65%) would cause an escape to a heliocentric orbit. In the cases of impact, for the most unfavourable moon positions (worst-case scenario), the probabilities of an Earth impact are $< 2.63\%$, while $< 0.12\%$ for the moon.

7 Conclusions

The mission's objective was primarily to move a Near Earth Asteroid to an orbit where science, planetary defence and in-situ resource utilisation investigations could be carried out, with secondary investigations carried on board the spacecraft itself. This investigation found that the feasibility of the overall mission depends heavily on the capability of the chosen launcher, Ariane 64. As limited information on the launcher is available, the estimations and assumptions on its capability to escape are variable and thus it cannot be definitively said whether Ariane 64 will be able to meet the mission's requirements. The Ariane 64 will also require a launch from the equator, needing an inclination change to match the plane of the ecliptic, a ΔV that needs to be provided by the launch vehicle that has not been accounted for so far in these investigations.

However, if the Ariane 64's capabilities to escape are demonstrated to be greater than those estimated in this project then the mission can go ahead as planned. In addition, the asteroid is estimated with a 99% certainty of a mass below 366 tonnes, and thus the design is over-engineered for the likely asteroid mass. This innovative design has led to a conditionally feasible mission concept that can complete an objective that was hitherto impossible. Particularly, the techniques employed in the mission analysis, combined with the propulsion system chosen, and the asteroid capture device, are ground-breaking. The risk of total mission failure due to asteroid mass is negligible, as the CubeSat scout will be able to determine a more accurate estimation of the asteroid mass and in the unlikely event it is too large the launch can be postponed until a more suitable target asteroid is found. However, a different launcher with a better known and more effective launch capability could take the ZEUS-15 into an escape orbit where the Ariane 64 cannot. This mission concept successfully blends futuristic innovations with mission feasibility to unearth a unique and useful solution to the asteroid capture problem.

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