ANDROID EXECUTIVE SUMMARY

ANDROID ACTIVE DEBRIS REMOVAL IOD

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1.0	03/07/2014	34	First official issue of the document.

TABLE OF CONTENTS

LIST OF TABLES AND FIGURES

1. INTRODUCTION

1.1. PURPOSE

This document is one of the deliverables from the Phase 1 of the AnDROiD project (ESTEC contract 4000110200/14/NL/MV), devoted to the definition of a conceptual design of active debris removal for a small satellite mission. In more detail, this document is the output of WP4100, "Management" under GMV responsibility.

The purpose of this document is to provide an executive summary of the results and the activities carried out in the course of the project.

1.2. SCOPE

In the frame of the AnDROiD project, system level definition of the mission is to be carried out during the first phase of the study (see [AD.1] and [AD.2] for details). This phase was followed by a simulation campaign of the main GNC functionalities and a final phase devoted to the programmatic assessment of the proposed mission. The scope of the present document is limited to the reporting of the activities carried out during the course of the project and its main results.

2. APPLICABLE AND REFERENCE DOCUMENTS

2.1.1. APPLICABLE DOCUMENTS

The following documents, of the exact issue shown, form part of this document to the extent specified herein. Applicable documents are those referenced in the Contract or approved by the Approval Authority. They are referenced in this document in the form [AD.X]:

Table 2-1: Applicable documents

Ref.	Title	Code	Version	Date
[AD.1]	SoW Active Debris Removal for a small satellite mission	GSP-SOW-13-700	1.0	12/06/2013
[AD.2]	AnDROiD Proposal	GMV 11169/13V1/13	1.0	23/09/2013
[AD.3]	Space sustainability. Adoption of Notice of ISO 24113: Space systems – Space debris mitigation requirements	ECSS-U-AS-10		10/02/2012
[AD.4]	Negotiation Meeting MoM	AND-GMV-MoM-0001	1.0	22/10/2013

2.1.2. REFERENCE DOCUMENTS

The following documents, although not part of this document, amplify or clarify its contents. Reference documents are those not applicable and referenced within this document. They are referenced in this document in the form [RD.X]:

Table 2-2: Reference documents

Ref.	Title	Code	Version	Date
[RD.1]	Conceptual Design of Active Debris Removal for a small satellite mission	GMV.ANDROID.D1	1.1	16/06/2014
[RD.2]	De-Orbit Simulation Results	GMV.ANDROID.D2	1.0	30/05/2014
[RD.3]	Cost Report For The Conceptual Design Of Active Debris Removal For A Small Satellite	GMV.ANDROID.D3	1.0	23/06/2014

2.2. ACRONYMS

Table 2-3: Acronyms

Acronym	Definition
ABAG	Absolute Attitude Guidance Mode
ABSM	Absolute Mode
ABSN	Absolute Navigation Mode
ACS	Attitude Control System
ADPMS	Advanced Data and Power Management System
ADR	Active Debris Removal
AIV	Assembly Integration and Verification
AMM	Autonomous Mission Management
AOCS	Attitude and Orbit Control System
ASAG	Angular Synchronization Attitude Guidance Mode
CAM	Collision Avoidance Manoeuvre
CAMM	Collision Avoidance Manoeuvre Mode
CATG	CAM Translational Guidance Mode
CDEC	Robotic Arm Capture and De-orbiting Control Mode

Acronym	Definition
IRLN	Intermediate Range Relative Navigation Mode
LAR	Large Angle Rotations
LEO	Low Earth Orbit
LIDAR	Light Detection And Ranging
LTAN	Local Time of the Ascending Node
LVLH	Local Vertical Local Horizontal
MCI	Mass Centering and Inertial
MEC	Mechanical frame
MIB	Minimum Impulse Bit
MOI	Matrix of Inertia
MPC	Model Predictive Control
NETM	Net capture and de orbiting Mode
NMTG	Nonlinear Impulsive Manoeuvres Translational Guidance Mode
NOAG	No Attitude Guidance Mode

Acronym	Definition
CDF	Concurrent Design Facility
CFRP	Carbon Fiber Reinforced Polymer
COM	Center of Mass
CPC	Coarse Pointing Control Mode
CRLM	Close Range Relative Mode
CRLN	Close Range Relative Navigation Mode
DDV&V	Design, Development, Validation and Verification
DOF	Degree of Freedom
DSLP	Dual Segmented Langmuir Probe
ECSS	European Cooperation for Space Standardization
EEE	Electronic, Electrical and Electromechanical
EMF	Electromotive Force
EXPM	Experiments Mode
FDIR	Fault Detection, Isolation and Recovery
FMTG	Forced Motion Translational Guidance Mode
FOV	Field of View
FPC	Fine Pointing Control Mode
FRLM	Far Range Relative Mode
FRLN	Far Range Relative Navigation Mode
GNC	Guidance Navigation and Control
GPS	Global Positioning System
GSP	General Studies Programme
ICD	Interface Control Document
IMTG	Impulsive Manoeuvres Translational Guidance Mode
IMU	Inertial Measurement Unit
IOD	In Orbit Demonstration
IRLM	Intermediate Range Relative Mode

Acronym	Definition
NOC	No-Control Mode
NOTG	No Translational Guidance Mode
OBSW	On Board Software
PCI	Peripheral Component Interconnect
PTFE	Polytetrafluoroethylene
RAAN	Right Ascension of the Ascending Node
RDV	RendezVous
ROBM	Robotic Capture and de-orbiting Mode
RTEMS	Real-Time Executive for Multiprocessor Systems
RTU	Remote Terminal Unit
SAFC	Safe Control mode
SAFM	Safe Mode
SAFN	Safe Navigation Mode
SCAG	Scanning Attitude Guidance Mode
SPAG	Safe Pointing Attitude Guidance Mode
SPOUA	South Pacific Ocean Uninhabited Area
SRP	Solar Radiation Pressure
SSA	Space Situation Awareness
SSO	Sun Synchronous Orbit
TBC	To Be Confirmed
TBD	To Be Decided
TEC	Tethered Control Mode
TETG	Tethered Translational Guidance Mode
TLE	Two Line Element
TPAG	Target Pointing Attitude Guidance Mode
TRL	Technology Readiness Level
TTC	Telemetry and Tele Command

2.3. DEFINITIONS

N/A

Table 2-4: Definitions

Concept / Term	Definition

3. MISSION SUMMARY

In recent years the concern about the future exploitation of space has been growing due to the risk that uncontrolled space debris poses to the space environment and therefore to the survivability of operational spacecraft. Two main regions of concern exist, GEO, where most of the commercial telecommunications satellites orbit, and LEO, where many scientific missions observing the Earth fly. Of special concern is the sun synchronous orbit, a natural resource of special interest for Earth sciences. The population of debris in this region has been growing, increasing the risk of a collision and hence the exponential increase in the number of debris.

One of the possible solutions to this problem is Active Debris Removal. Recent studies show that the situation could be contained if 5 to 10 debris objects are removed per year. According to IADC guidelines, initially those debris objects to be removed first should be the ones posing more risk, usually large objects (high mass, high surface, high energy).

The goal of the AnDROiD mission is to demonstrate the critical technologies that would be required for active debris removal missions. During the course of the project an investigation of the possible alternatives, technologies required and system level design for such a mission has been carried out, paying attention to the most critical technologies, namely the guidance, navigation and control system and the debris capture mechanisms (robotic arm and net system).

AnDROiD mission proposes to attempt at least two different capture techniques before the actual deorbiting of the target, with the objective of maximising the mission return in terms of development and in-orbit demonstration of key ADR technologies. Another output of this mission will be the results and the lessons learned from the ground operations of an ADR mission, which has been designed in order to maximise on-board autonomy thereby reducing ground operations costs.

The selected capture technologies, a rigid method (robotic arm) and a flexible one (net system), are somehow generic and could be scaled for different type of targets. The robotic arm capture will be performed first, followed by a set of secondary experiments and final capture with the net system to later deorbit the compound.

As case study PROBA2 has been selected as target debris. PROBA2 was launched on the second of November 2009 by a Rockot launch vehicle. The main goal of the mission was technology demonstration while at the same time providing scientific observations of the Sun. Currently it is still operational in a sun-synchronous orbit at 718km altitude with a local time of ascending node of 06h24 AM. For the study it has been studied that by the time AnDROiD is launched, PROBA2 will likely be non-operational and considered as debris. Initial analysis indicates that it should be spinning at an angular rate of 5 revolutions per orbit (corresponding to 0.3°/s). Two grasping points have been identified. The adaptor ring, selected as baseline point due to its generality with respect to other missions, and the DSLP antenna as backup (TBC), as per Figure 3-1.

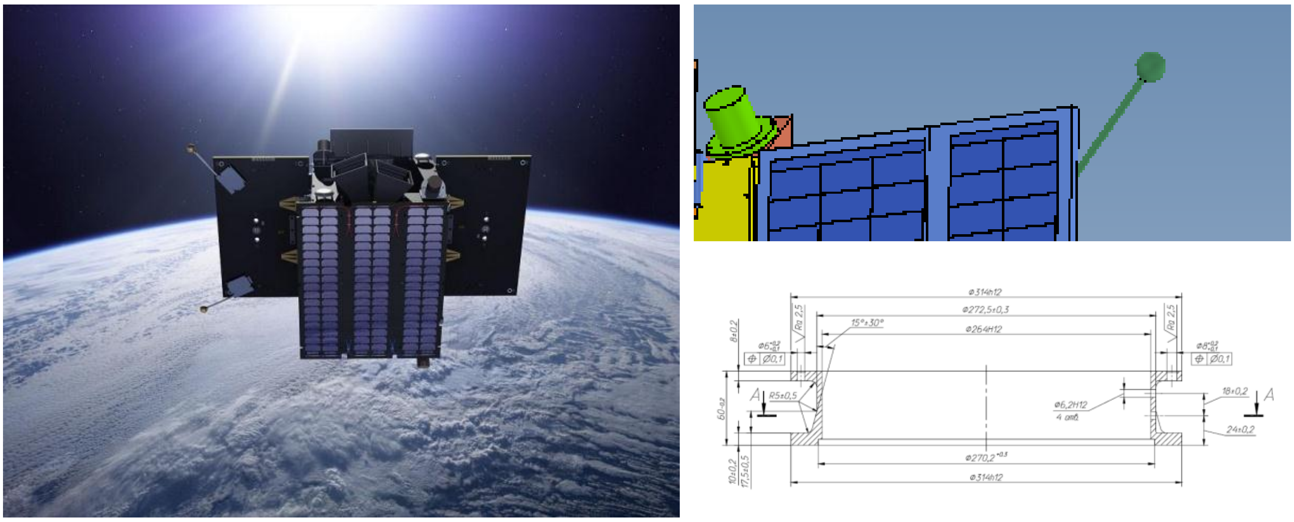


Figure 3-1: PROBA2 image and grasping points

The mission is to be launched in a shared launch into LEO. The total mass of the system is expected to be 353kg, with a total envelope with appendages of 1188(D)x1133(W)x1145(H)mm³. In terms of mass there should be no problem in finding a candidate launch, while in terms of envelope the situation could be tight. Right now it would be marginally feasible to launch AnDROiD as single passenger under the VESPA adaptor of VEGA launch vehicle, though the situation could be improved in further design iterations and clarification of the launch interface.

In terms of launch opportunities, SSO is a popular orbit for which several flights are done per year. The most popular orbits are around 650km altitude dawn dusk (quite close to the target orbit of AnDROiD) and 820km altitude 10:30 LTAN. A launch opportunity should be selected so that the ∆V and time required to arrive to the final orbit is minimised. Changes in inclination and right ascension of the ascending node can be extremely expensive in terms of propellant need and should be avoided. Initial assessment at this stage has indicated that enough opportunities for such launch should exist in the coming years. Therefore the mission has been designed taking into account an allocation of ∆V of 100m/s for final orbit acquisition (and target synchronisation), which should cover the needs of almost any launch into a dawn dusk orbit.

In addition to the demonstration of ADR mission technologies, a number of potential additional experiments have been identified and proposed.

The following table summarises the main mission elements for AnDROiD.

Table 3-1: System description

Parameter	Value
Proposed launch date	Before 2018 (goal)
Operational lifetime	1 year
Launch system	Vega launcher, secondary payload as single passenger under VESPA adapter
Mission goals	Demonstrate ADR technologies using PROBA2 as target
Capture mechanisms	Robotic arm Net system
Deorbit strategy	Direct re-entry with net system
Target orbit	SSO, 718 km, inclination 98.285°, eccentricity 0.0013, LTAN 6:24 AM
Relative sensors	Navegación camera DVS TSD LIDAR RVS3000 Jena Optronik
Mission demonstration	Rendez vous with non-cooperative target Proximity operations around non cooperative target Capture with robotic arm Capture with net system Direct deorbit with net system System autonomy

Parameter	Value
Proposed additional experiments	Spin synchronisation Collision avoidance manoeuvre Hop – rendezvous Deorbit rehearsal with robotic arm COBRA ADR related HW like IR camera or flash LIDAR SSA related HW like debris monitoring/cataloguing sensors Other ESA programmes like small payloads for science
System wet mass	353 kg including margins
System dimensions	1188(D)x1133(W)x1145(H)mm³
Communications	S-band downlink: 827kbit/s S-band uplink: 64ksps X-band downlink: 33Mbit/s
Power	Solar panels: 1 body-mounted and 1 deployable GaAs solar array with 28% efficiency cells Battery: Li-ion, 28V, 12Ah Bus: 28V battery regulated voltage
Total mission ΔV	400 m/s
Ground segment	MCC with single ground station (i.e. Kiruna)
Operational concept	High level of autonomy implemented in the system. Only early rendezvous phase (far rendezvous) executed from ground. Operator supervision could be required for go/no go points

4. MISSION REQUIREMENTS

The following table contains the mission requirements as defined in statement of work [AD.1]. These requirements have been analysed and flown down to the system requirements of the mission.

Table 4-1: Mission Requirements

Req ID	Requirement text
MIS -010	The system shall be designed to be launched as piggyback (i.e. sharing launch cost to minimise overall mission budget)
MIS -020	The system shall perform a rendezvous with a PROBA platform as target for a variety of target waiting conditions
MIS -030	The system shall be compatible with targets, where no a-priori knowledge of the magnitude and orientation of the attitude motion vector is available
MIS -040	The target object shall be an ESA owned object (non-operational satellite) in the LEO region (e.g. SSO), not heavier than 200 kg. For the purpose of the demonstration PROBA 1, 2 or V should be considered
MIS -050	The system shall be able to capture and manoeuvring the target satellite without generating any extra debris that do not decay in less than 25 years
MIS -060	The mission shall comply to Space debris mitigation requirements stated in [AD.3]
MIS -070	The total operational lifetime of the system will be less than one year
MIS -080	The mission shall be launched before 2018

It has to be noted that at NM/KO ([AD.4]) PROBA 2 was selected as target for the mission. Furthermore it was agreed that it should be assumed that PROBA 2 will be non-operational by the time of AnDROiD mission. These considerations slightly modify requirements MIS-020 and MIS-040.

5. MISSION DEFINITION

5.1. TARGET DEFINITION

PROBA2 has been selected as the target for the AnDROiD mission. At this moment PROBA2 is an operational mission and it has been assumed that will be in a non-operational status by the time AnDROiD approaches it. The current orbit of PROBA2 is:

- Altitude: 718km
- Local time of ascending node (LTAN): 06h24 am
- Inclination: 98.285°
- Eccentricity: 0.0013
- Orbit period: 99.16min

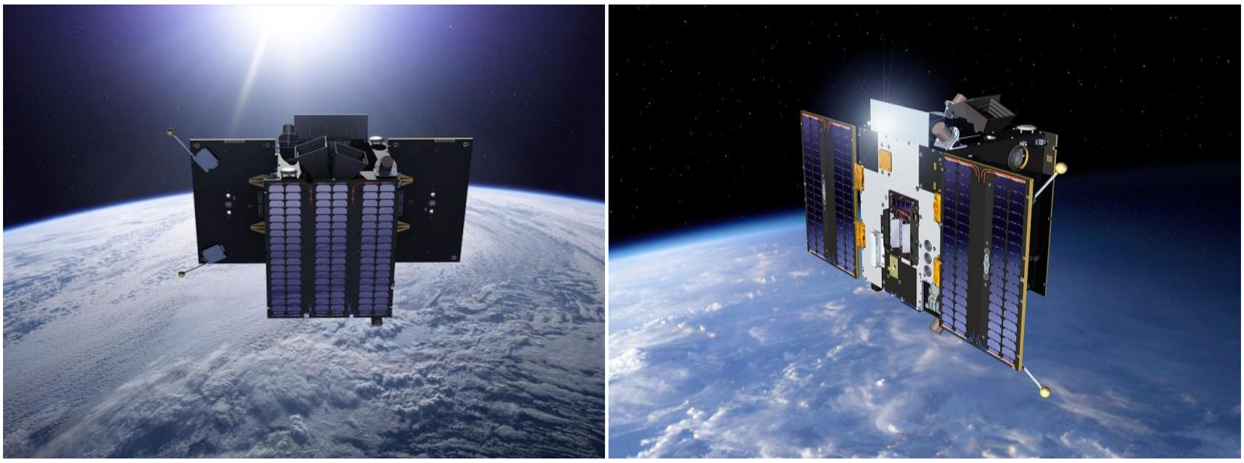


Figure 5-1: Artist's impression of PROBA 2

The following table summarises the mass and inertia properties of the spacecraft.

Table 5-1: Mass and inertia properties of PROBA 2

Mass (kg)	**124.80**		
COM (mm)	X	Y	Z
	-16.1	**-5.8**	**381.4**
MOI (kg m²)	X	Y	Z
X	**13.42**	0.27	-0.29
Y	0.27	**11.60**	-0.52
Z	-0.29	-0.52	**10.06**

The dimensions of the S/C are roughly 590mm(D) x 1603mm(W) x 790mm(H) (including deployed panels, but excluding appendages like baffles and antennae). The launcher interface ring has an internal diameter of 264mm and an external diameter of 314mm. Its height is 65mm (TBC). This interface ring has been identified as the grasping point for the robotic arm. DSLP antennae could be analysed as an option in case of need.

Simulations have been performed to assess what would be the rotational status of the spacecraft once it is decommissioned. The main perturbation affecting the attitude dynamics will be the magnetic dipole. Unfortunately the remanent magnetic dipole was not measured on ground before launch, hence several simulations with reference values have been carried out using the best guess available, the actual dipole while in operation (flight data) and no dipole. The results indicate that a spin rate of up to 5 revs/orbit could be expected, with no clear indication of the orientation of the spin axis.

5.2. MISSION ANALYSIS

AnDROiD is to be launched as a piggy back in a non-dedicated launch, as per mission requirement. Given the orbit of the target, PROBA 2 in dawn dusk SSO at around 720km of altitude, it will be desirable to get launched in a similar orbit. Fortunately, most of the upcoming missions in LEO will be launched into SSO, and within those missions, two orbits are more popular than the rest (over 70% of the satellites will go in one of these two orbits, 35% to each one):

- SSO 820km, 98.65deg, LTAN at 10:30 AM
- SSO 650km, 97.95deg, LTAN 6:00 AM

For the study it has been assumed that AnDROiD will be injected into a dawn dusk orbit and a provision of 100m/s has been made to adjust the altitude and inclination of the final orbit (PROBA2). This allocation should allow a launch in altitudes between 550km and 850km.

The following table summarises the ΔV budget and the timeline of the mission. As can be seen, ample margin exist to meet the requirement of 1 year of operational lifetime. Indeed, together with the proposed level of autonomy, this could lead to operations only during working hours.

Table 5-2: Summary timeline and ΔV

phase	time (h)	time (h) w. margin	ΔV (m/s)	Total ΔV (m/s)
Orbit synchronization	9.91	2232	100.00	110.00
Commissioning	-		0	0
Rendezvous	25.79	108	6.33	6.96
Commissioning	56.18	236	10.18	11.20
Proximity Operations and Target inspection I	40.70	171	0.06	0.07
Additional experiments I	67.79	285	10.34	13.38
Robotic arm capture	9.61	40	0.02	0.02
Combo experiment	1.83	8	10.24	11.31
Target release	1.50	6	0.02	0.02
Additional experiments II	73.25	308	6.53	9.19
Proximity Operations and Target inspection II	30.44	128	0.05	0.05
Net capture	3.50	15	0.14	0.16
System stabilisation	1.65	7	0.01	0.02
Deorbit	11.26	47	182.30	236.53
total	333.43	3591	326.21	398.89

As can be seen in the table above, the main contributor for the ΔV budget is the deorbit ΔV. In order to minimise the gravity loses the burn has been split in 3 manoeuvres to be executed in consecutive orbits with a thrust level of 35 N (2x22N thrusters with 37 deg of de-pointing wrt tether, to be optimised)

Table 5-3: Sequence of deorbit manoeuvres

Targeted perigee altitude [km]	ΔV [m/s]	Diff (%)
500	59.919	1.19
300	56.471	1.017
80	64.748	1.36

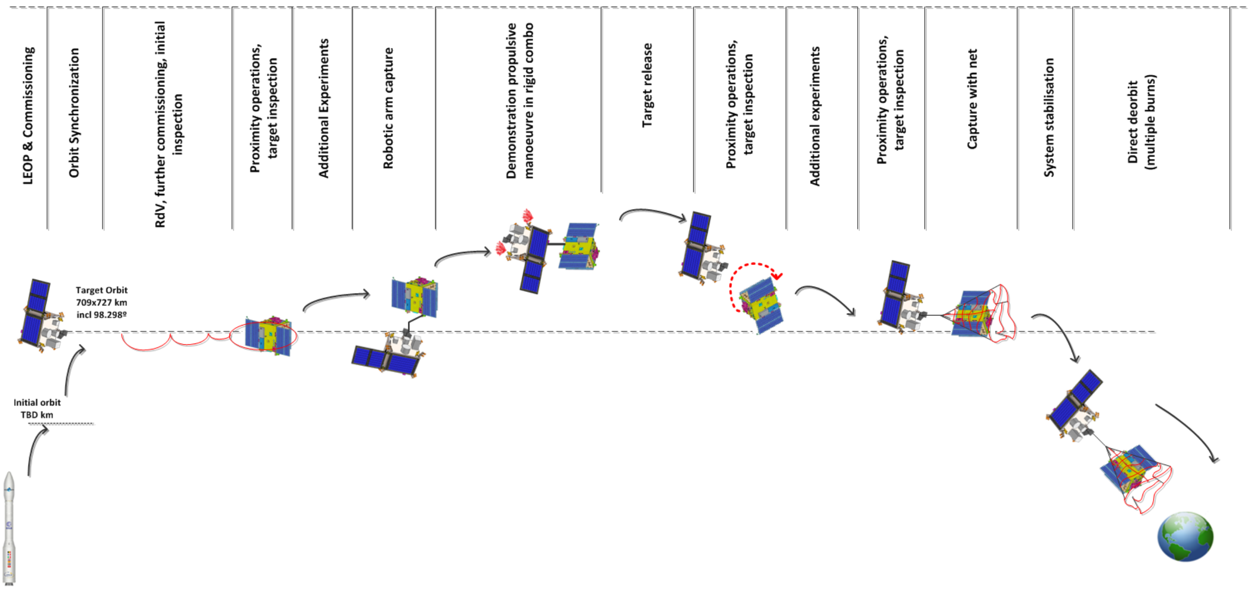


Figure 5-2: Mission timeline

Apart from the main technology demonstrations, additional experiments have been proposed for the AnDROiD mission, to be selected once the constraints of the mission have been better defined. They have been divided into software experiments, not modifying the design of the mission but making use of consumables and hardware experiments, requiring some kind of modification to the proposed design.

Table 5-4: Additional experiemnts

Type	Name	Objective
SW	Spin synchronisation	Demonstrate capability to synchronise with the spin axis of the target, either along it or perpendicular to it, using different control techniques
SW	CAM	Collision avoidance manoeuvre, needed for all proximity operations, part of the baseline but may not be nominally triggered, hence it should be forced
SW	Hop rendez vous	AnDROiD strategy is based on the use of safe drifting trajectories to perform the rendez vous with the target, other strategies like hops are to be tested
SW	Deorbit with robotic arm	Nominal deorbit is to be performed with the net, but short manoeuvre with the robotic arm is to be performed to emulate the deorbit
SW	COBRA	Contactless technique to control the target attitude (de-tumbling) by using the plume impingement fo chemical propulsion engines. Could also be used for deorbiting a target debris
HW	ADR-SSA	Different relative sensors could be tested (IR camera, flash LIDAR) or payloads included to monitor the debris environment (visual sensors, impact sensors) or space weather payloads
HW	Equipment	In line with previous PROBA missions, different equipment could be tested in flight, like new batteries, SA, EEE components
HW	Payload of opportunity	Small payloads of opportunity in support of other ESA programs could be accommodated, like small scientific payloads

Finally, five contingency cases have been identified:

- CAM, required throughout all mission cases. A manoeuvre shall be performed in case a collision risk is detected. Such manoeuvre shall stop the relative motion and induce a drift with respect to the target. The system should be based on an independent set of sensors (TBC) and navigation filters
- Retreat after robotic arm failure. In case of a failure of the capture with the robotic arm, the system shall retreat to a safe position. If the failure in the robotic arm can be corrected, a second capture attempt could be carried out
- Net catching failure. In this case it will be required to cut the tether and let the net drift away. Preliminary calculations indicate that the ner should re-enter in less than 25 years (TBC). In this situation capture with the robotic arm could be performed to finally deorbit PROBA2.
- Non-execution of de-orbit burn. If the original problem can be solved, it will become just an issue of planning the next attempts at the correct times. The perigee of the last orbit is still high enough as to provide ample margin for problem resolution and final burn scheduling.
- Chaser enters into safe mode while connected to the target via the net. In this situation the safer option would be to cut the tether and drift away. An option to be studies at a later stage could be to spin the system in the orbital plane.

5.3. PLATFORM DESIGN

The table below gives a short overview of the AnDROiD platform and its subsystems. The platform design is based on the PROBA-NEXT platform.

Table 5-5: Components of AnDROiD platform based on PROBA architecture

	AnDROiD platform
Avionics	ADPMS (Advanced Data and Power Management System) Processor: LEON2-E (SPARC V8) Mass Memory Module : 11 GByte Interfaces: RS422, TTC-B-01, analogue and digital status lines, Packetwire, compact PCI
Power	Solar panels: 1 body-mounted and 1 deployable GaAs solar array with 28% efficiency cells Battery: Li-ion, 28V, 12Ah Bus: 28V battery regulated voltage
Structure	Aluminium outer panels Aluminium milled bottom board CFRP outer panels with solar arrays
AOCS	3-axis stabilised satellite Actuators: • 3 magnetotorquers (internally redundant) • 4 reaction wheels • 1N Hydrazine propulsion system • 20N Hydrazine propulsion system Sensors: • 2 magnetometers • 2 star tracker (with 2 camera head units) • 2 GPS receivers • 1 navigation camera • 1 inertial measurement unit • 3 sun sensors (TBC) • 1 rendez-vous sensor (TBC)

	AnDROiD platform
Communication	S-band downlink: 827kbit/s S-band uplink: 64ksps X-band downlink: 33Mbit/s
Software	Operating system: RTEMS Data handling/application software: based on PROBA-V OBSW
Thermal	Mainly passive thermal control, heaters for the battery, the propulsion subsystem and the payload

Accommodation and dimensions of the spacecraft are mainly driven by the propellant tank and the accommodation needs for the robotic arm, relative sensors and net system. The following figures show the proposed configuration. The total envelope with appendages however is 1188(D)x1133(W)x1145(H)mm³.

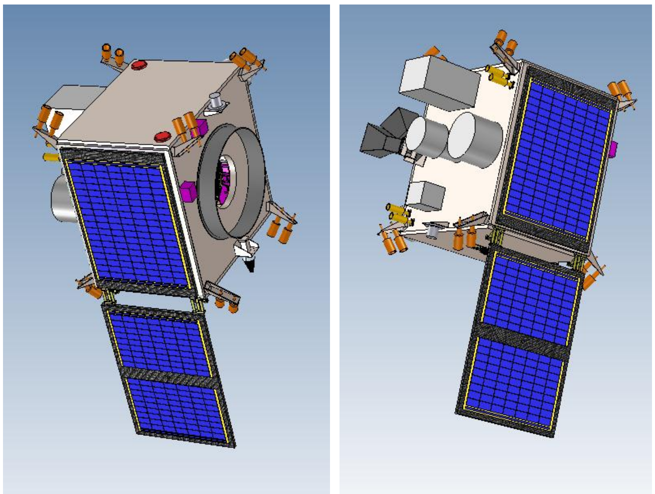


Figure 5-3: External view of the spacecraft with deployed panel (solar side)

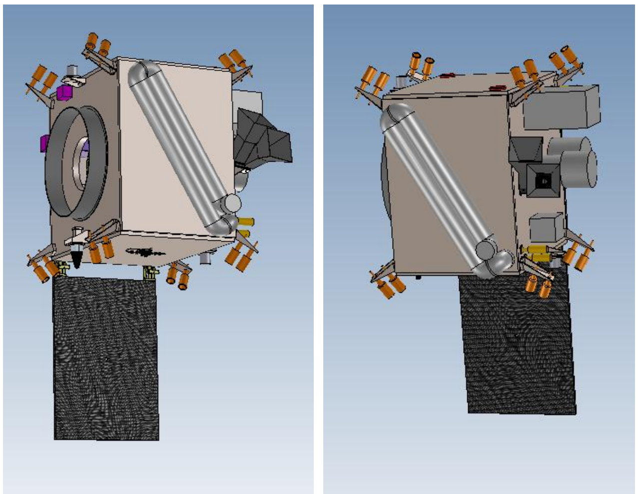


Figure 5-4: External view of the spacecraft with deployed panel (anti-solar side)

The total mass of the system is 353 kg, including margins, launcher I/F ring and propellant (up to 68 kg of propellant are loaded). The total dry mass with margin is less than 280kg.

5.4. GNC DESIGN

In an active debris removal mission the GNC system is one to the key technologies to be demonstrated. In this contest the GNC shall provide all the required functionalities to perform the attitude and translational movements to approach, capture and deorbit the target. Analysis of the required functionalities has been carried out and an architecture of the system defined, performing several trade-offs, especially for the interaction between the GNC and the robotic arm. In this respect, and taking into account the dynamics of the target it has been decided to implement independent control systems for the GNC and the robotic arm and to perform the capture in free floating mode.

The modes and submodes required to cover the required functionalities for the different mission phases have been defined as well as the transitions between them. The following figure summarises the GNC modes.

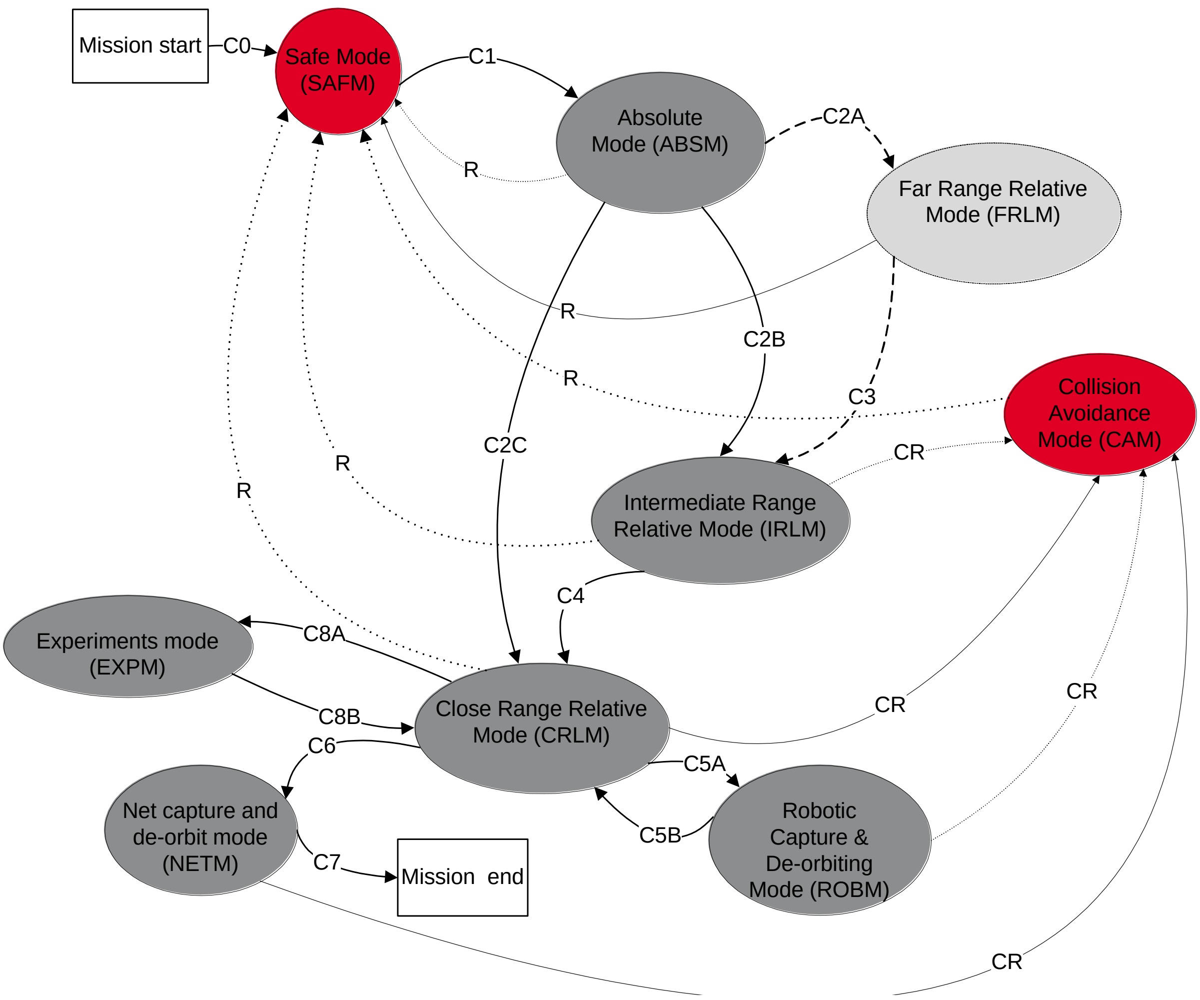


Figure 5-5: High-level GNC mode diagram

Hardware selection has been carried out with the main constraint of selecting elements with flight heritage, both for the sensors and the actuators.

Actuators have been sized according to the mission needs in terms of thrust level, accuracy, agility and momentum storage capabilities. Existing hardware meeting these requirements have been selected.

In terms of sensors, standard AOCS sensors have been selected. For relative sensors, a navigation camera (DVS) has been selected to cover the mission needs, complemented with a LIDAR (RVS3000). More detailed analysis is required to verify if a mission without the LIDAR would be feasible, as it would lead to a significant cost reduction.

5.5. ROBOTIC ARM DESIGN

Different architectures have been analysed and simulated for the design of the robotic arm. Length selection, mass budget, singularities, required angular speeds and generated torques during the different mission phases have been taken into account.

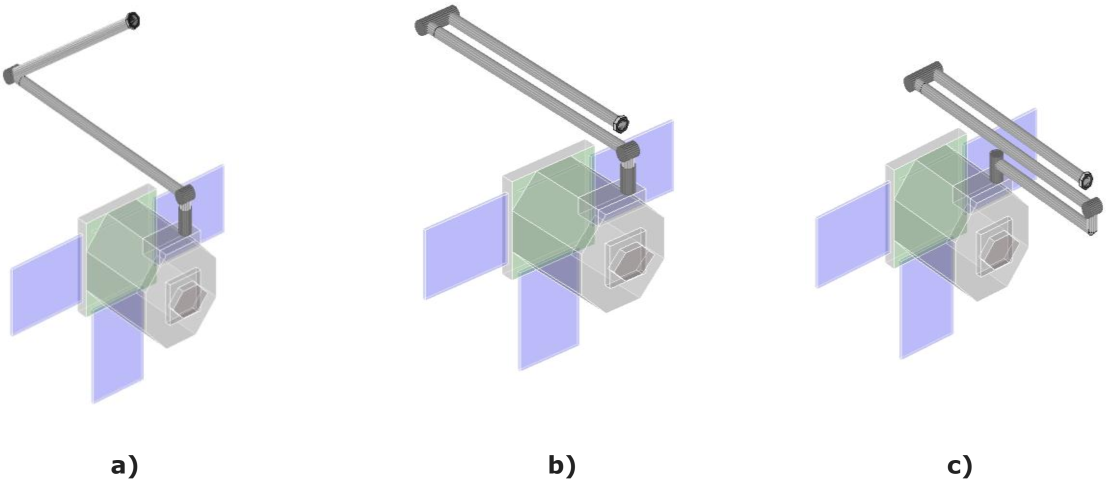


Figure 5-6: Robotic arm architectures

The analysis of the results has concluded that two architectures are feasible for performing the assumed task: "2" and "3" in Figure 5-6, though the architecture "2" has better control properties. The "2" architecture is less compact than the "3" and occupies more space at the satellite, but when using the deployed arms technology it can be stored at the satellite in a compact form. The length of the manipulator is 3 meters and it was assessed in the workspace analysis. The budgets are presented in Table 5-6.

Table 5-6: Mass and power budgets. Joint efficiency 50%.

Option	Total mass [kg]	Electric power consumption [W]		
		Caputre	Rigidizing phase	Deorbiting
Joint without brake	20,0	0.6	8	400
Joint with brake	22,2	0.6	8	0

The mass budget of the manipulator is 20 kg. The maximal (peak) mechanical power consumption is about 4 W. The peak electrical power is about 8W (assuming the joint efficiency 50%). The manipulator needs the breaking gears in joints for managing the large torques while the deorbiting phase. When sensing and control elements are taken into account, a total of 50W is envisaged for the system.

The following figure and tables summarise the characteristics of the selected architecture.

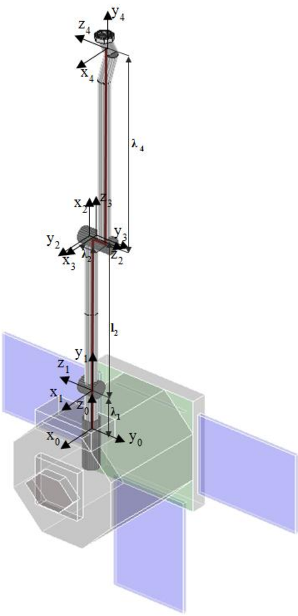


Figure 5-7: Architecture „I1" and Denavit-Hartenberg coordinate systems

Table 5-7: Denavit-Hartenberg parameters of architecture „2"

lp	θ	λ	l	α
1	θ_1	$\lambda_1 =$ 0,21m	0	$\frac{\pi}{2}$
2	$\theta_2 + \frac{\pi}{2}$	λ_2 = 0,1m	l_2 = 1,28m	π
3	$\theta_3 + \frac{\pi}{2}$	0	0	$\frac{\pi}{2}$
4	θ_4	λ_4 = 1,28m	0	$\frac{\pi}{2}$
5	θ_5	0	0	$-\frac{\pi}{2}$
6	θ_6	0	0	0

Table 5-8: Manipulator parameters (Architecture „2")

Link no.	Length [m]	Location of joint of link n+1 [m]			mass [kg]	Centre of mass location [m]			Inertia tensor [kg·m²]		
		p_a(x)	p_a(y)	p_a(z)		p_rho(x)	p_rho(y)	p_rho(z)	Ixx	Iyy	Izz
1	0,21	0	0,21	0	1,40	0	0,105	0	0,00865	0,00117	0,00865
2	0,64	0	0,64	0	4,27	0	0,32	0	0,15630	0,00356	0,15630
3	0,64	0,64	0	0,1	4,27	0,32	0	0,05	0,00356	0,15630	0,15630
4	1,28	0	0	1,28	8,53	0	0	0,64	1,18642	1,18642	0,00711
5	0,15	0	0,15	-0,04	1,00	0	0,075	-0,02	0,00438	0,00083	0,00438
6	0,059	0	-0,04	0,059	0,39	0	-0,02	0,0295	0,00110	0,00110	0,00033
7	0,021	0	0	0,021	0,14	0	0	0,0105	0,00036	0,00036	0,00012

5.6. NET SYSTEM DESIGN

In contrast with rigid capture mechanisms, tethered-net solutions are characterized by capturing debris from a safety distance, by passive angular momentum damping and by establishing a tethered connection between the chaser and the target. Moreover, tethered-nets are general-purpose removal systems: they could effectively intervene on objects different in configuration, materials and possibly in dimensions.

A generic tethered-net capturing system is composed by:

- Net
- Tether
- Net storage and deployment mechanism
- Tether reel mechanism

The deployment of net is performed by impulsively accelerating a number of corner weights (bullets) attached to the net edges or mouth. The bullets will perform a dual role by opening the net gradually (due to their momentum) in such a manner that the net is fully extended just before reaching the target debris and afterwards by closing in and entangling on the target due to the same momentum. Additionally the use of two mechanisms (rotors) located in the bullets with the role of rolling in the cord that encompasses the net mouth shall assure the full closing of the net around the target. The net is linked to the tether and implicitly to the chaser S/C through a central vertex (knot) which has the role of absorbing/distributing the loads. During net deployment the tether is left slacked in order to reduce the interference on the dynamics of the net and avoid significant reaction forces on chaser satellite. After debris capturing is successfully performed the tether is gradually tensioned and unwind in order to minimise longitudinal oscillations. A separation of 20 m between the two satellites has been selected for safety reasons resulting in a bullet divergence angle of 7°.

A model has been built up to support the system sizing and the different trade-offs performed to define the net system. The main trade-offs have been:

- Net system design and sizing; planar or 3D (pyramidal or pseudo conical), mesh type and size, manufacturing technology.

  A planar net of 10x10m has been selected, using knotting and thermo welding for manufacturing with a mesh size of 0.25m.

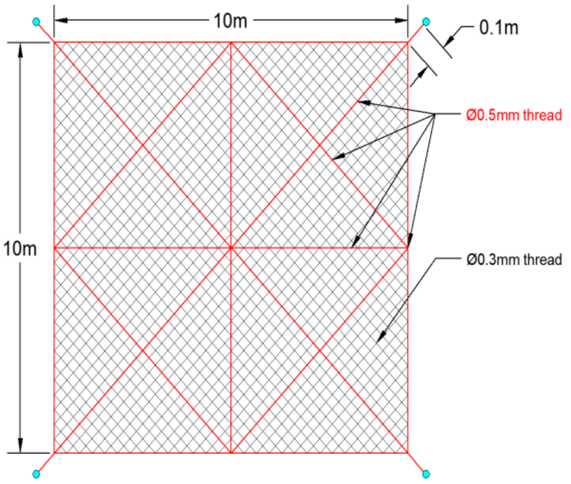


Figure 5-8: Net design

- Material, different materials have been analysed both for the tether and net. The following table summarises the materials taken into account.

  Dyneema SK75 has been selected as the material for both the net and the tether. It shall be noted that the last part of the tether will be covered with carbon fiber jacket to protect it from the plume impingement

In terms of mechanism, two main elements have been analysed, the deployment mechanism including the storage canister and the tether reel mechanism. The design of the deployment mechanism has been based on the breadboard made in the frame of the Patender activity (an ESA TRP activity within CleanSpace program), developed by Prodintec. It is composed of a central canister with an hinged opening lid and the bullets firing mechanism, based on `pneumatic technology. It has been found out that a controllable reel mechanism would help in controlling the dynamics of the system (tension control and reconfigurations).

The following table summarised the baseline design:

Table 5-9: System Description

Design parameter	Option
Material	Dyneema SK75 (net+tether)
Tether thermal protection	CF T1000G jacket
Net configuration type	Planar 3
Tether-net link	Inter-weaving
Corner masses	4 bullets, 2 with spring driven reels
Storage canister	Breadboard inherited from Patender (Prodintec design)
Ejection mechanism	Pneumatic (inherited from Patender)
Tether reel	Active control reel
Net Size X,Y,Z [m]	10, 10, 0
Mesh Size [m]	0.25
Net Threads Diameter [m]	0.3x10-3 / 0.5x10-3
Bullet Link Length [m]	0.1
Bullet Link Diameter [m]	1x10-3
Initial capture distance [m]	20
Divergence Angle [deg]	7
Initial Velocity [m/s]	2
Net Mass [Kg]	0.210
Bullet Mass [Kg]	1.25
Total Mass with Bullets [Kg]	1.45
Estimated Net Volume (100% percent margin to account for knots) [m3]	1x10-3
Total system mass (with margins) [kg]	15.5

5.7. GROUND SEGMENT AND OPERATIONS

AnDROiD mission could benefit from previous PROBA missions experience. In terms of system geometry, it will indeed be very similar to PROBA 2, so the same architecture for ground stations could be used. In terms of functionalities, the approach proposed for AnDROiD is also in line with previous PROBA missions.

As discussed in the previous sections, it is proposed to perform most of the AnDROiD phases in an autonomous way. The only phase performed under ground control is the orbit synchronisation, which is an offline operation that could be considered as routine by the operations and flight dynamics teams. The main reasons for this approach are:

- Being a technology mission, it should also be used to advance in autonomy technologies, both in terms of mission/spacecraft management, scheduling and FDIR
- High level of autonomy should reduce the operational costs of the mission and enable the possibility of operating the mission only during "office hours" to further reduce the costs. This possibility is also supported by the available margins in the timeline of the mission.
- In terms of safety and risk of collision, having the operator in the loop will not provide any added value to the mission. The operator will have the same information on ground as the spacecraft will have in flight and will have to operate a similar software than the one in flight to check the safety of the mission with the same time constraints. Furthermore, in order to provide the data in real time to the operator a complex and expensive net of ground stations should be used, therefore increasing the cost of the mission.

Therefore it has been decided to eliminate the operator from the loop and perform the mission design in such a way no direct intervention of the operator will be required. In any case, system monitoring will be performed at each pass and go/no go points could be inserted at different points in the sequence of events.

The mission control center could be co-located at REDU with the rest of the PROBA missions control center. With respect to the ground stations required, REDU could be used for TM/TC in S band complemented with Kiruna or Svalbard for X band telemetry (experiments data), depending on the final data volume required.

A first analysis has been carried out to assess the contact times with the different ground stations. The results are summarised in the following table.

Table 5-10 Contact times with ground stations

Station	Min contact per pass [min]	Mean contact per pass [min]	Max contact per pass [min]	Max time without contact [hours]	Min contact per day [min]	Mean contact per day [min]	Max contact per day [min]
REDU	0	3.773	11.833	11.753	47.000	54.583	58.000
Kiruna	0	7.207	12.000	8.361	97.000	104.883	110.833
Svalbard	4.667	9.848	12.000	1.594	136.500	143.239	149.167

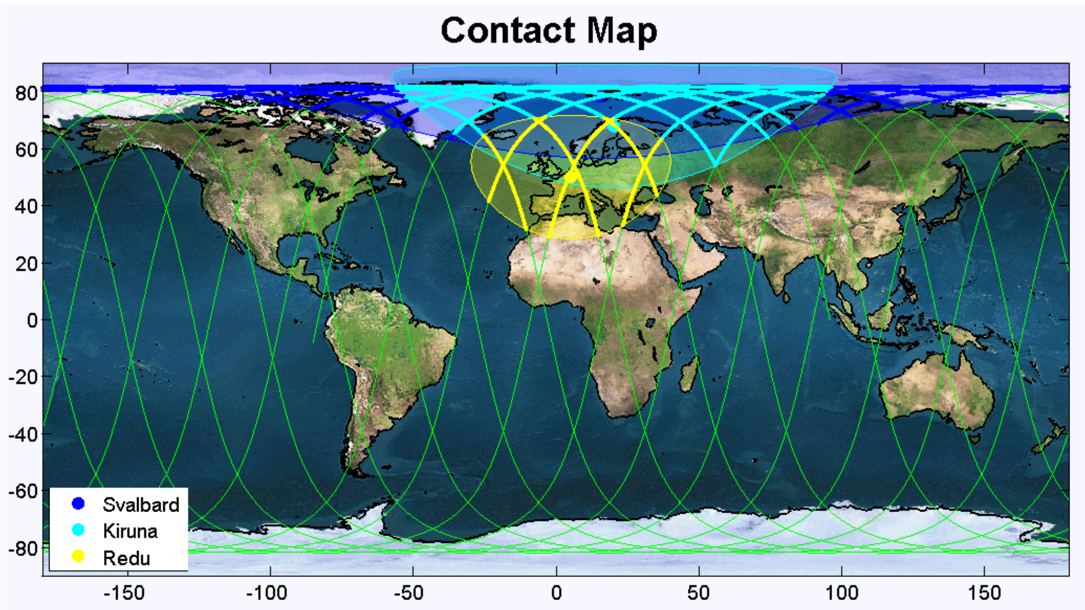


Figure 5-9: Contact areas with the different ground stations

As can be seen from the table and the figure above, REDU will provide limited coverage, short passes and around 7 blind orbits per day. It could be still suitable for TM/TC, but not for receiving experiment data. Kiruna has better performances but it also can have up to 5 blind orbits per day. This limitation would introduce an operational/planning constraint on the mission. Detailed analysis of the operations should be performed to see whether if this constrained can be dealt with by proper planning of the different mission phases, as it looks in principle. Finally, Svalbard would provide the largest amount of data available (contact time) with no blind orbits.

The final location of the ground stations shall be decided once detailed analysis on the required TM data volume and operations planning is performed. Initial estimations indicate the Kiruna could be a suitable solution, with Svalbard as back up.

5.8. SCALABILITY ANALYSIS

A scalability analysis has been carried out at different levels. A system like the one proposed could evolve into a system capable of deorbiting larger targets.

A scaling exercise has been performed using as design points the proposed AnDROiD system (without the robotic arm) and the CDF e.Deorbit net option. The system mass has been interpolated taking as parameter the required propellant mass. In terms of ΔV requirements 50m/s have been assumed for proximity operations and capture and the required ΔV for re entry has been computed as a function of the target altitude. The following table summarises the results:

Table 5-11 Deorbit System total mass (without margins)

		Target mass [kg]							
Target altitude [km]	**ΔV [m/s]**	**100**	**250**	**500**	**750**	**1000**	**2000**	**4000**	**8000**
600	**147**	257	280	317	355	392	542	843	1443
700	**173**	264	290	333	376	419	591	936	1625
800	**200**	271	301	350	399	447	643	1035	1818
900	**224**	278	310	365	419	474	691	1126	1995
1000	**248**	285	321	381	440	500	740	1218	2176

As can be seen from the table above, small targets (up to 750kg) could be deorbited with a relative small system in all the orbital regimes (mass under 400 kg). This deorbit system could be launched as a piggy back in a non-dedicated launch. For larger targets a dedicated launch may be required. If VEGA is taken as reference launcher, any target (up to 8000kg) could be deorbited for altitudes as high as 700km. For higher orbits, the target mass should be reduced. Targets as heavy as 4000kg could be deorbited in altitudes as high as 1000km

In terms of platform, if the same design is kept, the proposed system could be used to deorbit a target of up to 150kg, assuming that ΔV for orbit acquisition and target capture is limited to 120m/s, i.e. no additional experiments are carried out (propellant tank limited to 68kg). A different strategy could be to implement natural decay as deorbiting strategy instead of direct re-entry and only the robotic arm is used with no further experiments. The following figure shows that a target of up to 1 ton could be removed at 800 km altitude.

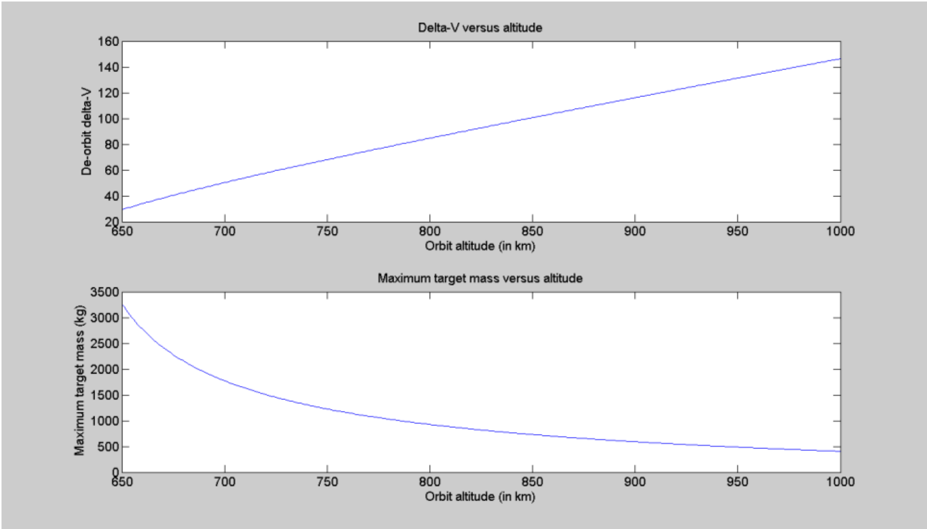


Figure 5-10: Maximum target mass that can be de-orbited

In terms of GNC, the proposed architecture is perfectly scalable. In terms of SW, the same architecture could be maintained with small variations depending on the selected strategies (capture and deorbit). These strategies will be mainly driven by the physical properties of the target, its orbit and the rotational state.

Larger targets will lead to different selection of sensors and actuators:

- FOV selection for close operations will be dictated by the minimum distance to the target (target size) and having the full target in the image (IP)
- Actuators will be sized according to the mass and inertia of the target. Larger target will require higher thrust level, though same accuracy

Higher spin rate will lead to changes in strategy for capture (fixed base manipulator) and selection of actuators in line with the above

The architecture of the GNC system and algorithms will remain basically unchanged (except for change in strategy, though most of the strategies are already demonstrated in Android). Tuning of the different algorithms will be required:

- Relative Navigation shall be updated according to the updated target characteristics and sensor suite
- Guidance shall be scaled according to scenario size (minimum distances, spin rate, hold points,…)
- Control shall be updated with new MCI and strategy (spin sync required, fixed base)

The net system can be sized depending on the target mass, dimensions and the required thrust level for deorbit. During the course of the study a sizing tool has been developed to help in this exercise. Larger targets will require a bigger net and hence higher mass and volumetric needs.

The robotic arm can be sized depending on the dimensions of the target (length of the links), mass, inertia matrix and rotational status. In general terms higher MCI will lead to higher torques for the arm rigidisation and control. The same will occur in case of higher spin rates. Higher torques will translate into a higher mass of the system and larger power needs.

6. SIMULATION CAMPAIGN

A simulation campaign has been performed to validate the GNC strategies proposed for the mission. The different mission phases have been analysed and three scenarios defined, taking into account the criticality of the involved elements and the resources availability. A simulator has been built up for each of the different scenarios.

6.1. SCENARIO 1

Scenario 1 is a rendezvous scenario based on cotangential transfers, drift orbits and the safe orbit. This scenario focuses on the translation guidance required for performing the rendezvous. The manoeuvres are performed as impulsive feed-forward acceleration commands, and the control required for this is minimal.

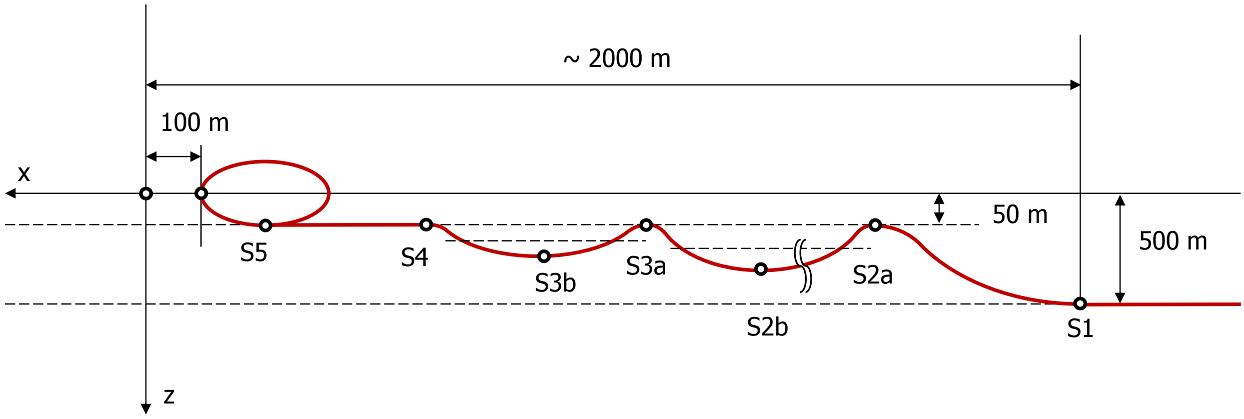


Figure 6-1: Rendezvous strategy

Test campaign has been run in for this scenario and the resulting trajectory and manoeuvres monitored to check that the guidance function performs as expected.

Figure 6-2 left shows the LVLH trajectory of the rendezvous. The guidance reference trajectory is shown in red, and the true trajectory is shown in black. The figure shows that the trajectories nearly overlap, and that the rendezvous is performed correctly down to a range of 100 metres. Figure 6-2 right shows a zoom of the rendezvous trajectory, showing the final 100 metres and the inspection flight. This figure shows that in-plane trajectories of the rendezvous and the inspection flight are correctly performed.

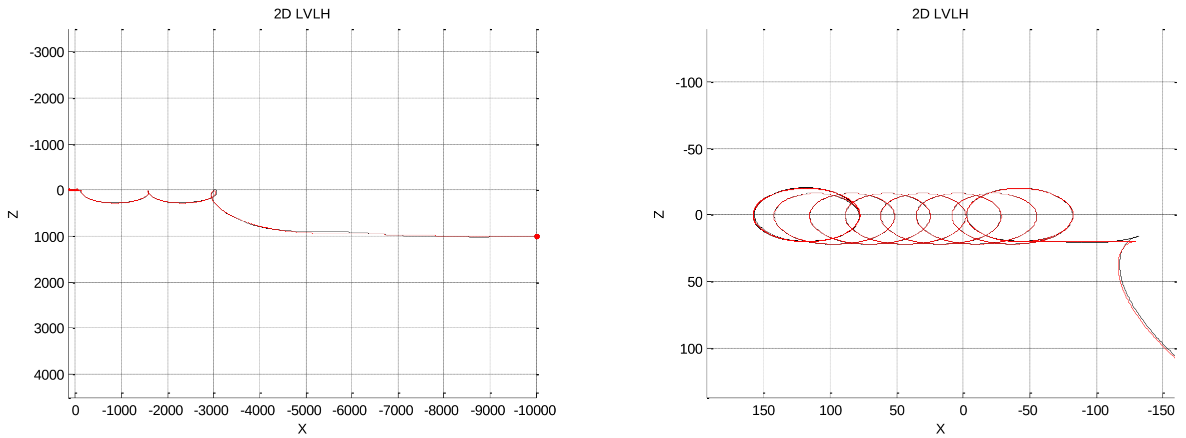


Figure 6-2: LVLH trajectory, rendezvous from drift orbit and inspection

Figure 6-3 shows a 3D figure of the inspection flight, showing that the trajectory indeed consists of two safe orbits connected by a cork-screw drift orbit that lasts for 6 orbits.

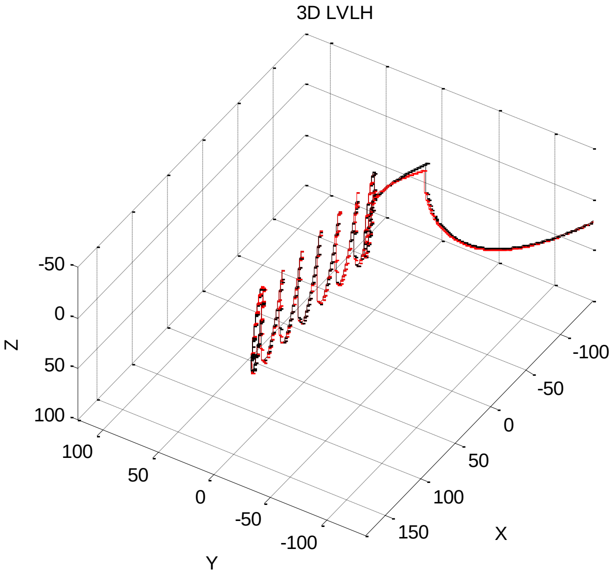


Figure 6-3: 3D LVLH trajectory, inspection

The simulation results have shown that the proposed strategy is working as expected, with only small errors due to the limitations of the models in use (i.e guidance models could incorparate J2, use more effective linearization, and the guidance plan could be reset, refreshing reference orbit). In terms of ΔV, the dimensions of the executed manoeuvres are as expected. It should be noted that the proposed strategy leads to a reduction in the required ΔV of a factor 10 when compared with a hop based strategy.

6.2. SCENARIO 2

The synchronization with the target consists of an approach in the target body fixed frame and a synchronization of the attitude of the chaser with the attitude of the target spacecraft. The approach in the target body fixed frame requires continuous thrust acceleration (i.e., forced motion trajectories) and occurs below a distance of 20 metres. The approach in the target body fixed frame is considered proximity operations. Figure 6-4 shows several approach strategies for approaching a spinning satellite, labelled A, B and C. The base reference frame of the figure is the LVLH frame, but certain parts of the trajectory are computed in other reference frames, such as the inertial reference frame or in the target body reference frame. This is done because the formulation is simpler and because the trajectory defined in a certain frame is invariant. For example, a fly-around in the target body frame is more easily expressed in the target body frame than in the LVLH frame. In addition, the fly-around trajectory can be fully pre-computed, while the attitude of the target with respect to LVLH can and will be updated through measurements. That is to say, the trajectory in the target body frame remains the same while the trajectory in the LVLH frame changes because the attitude of the target with respect to LVLH changes.

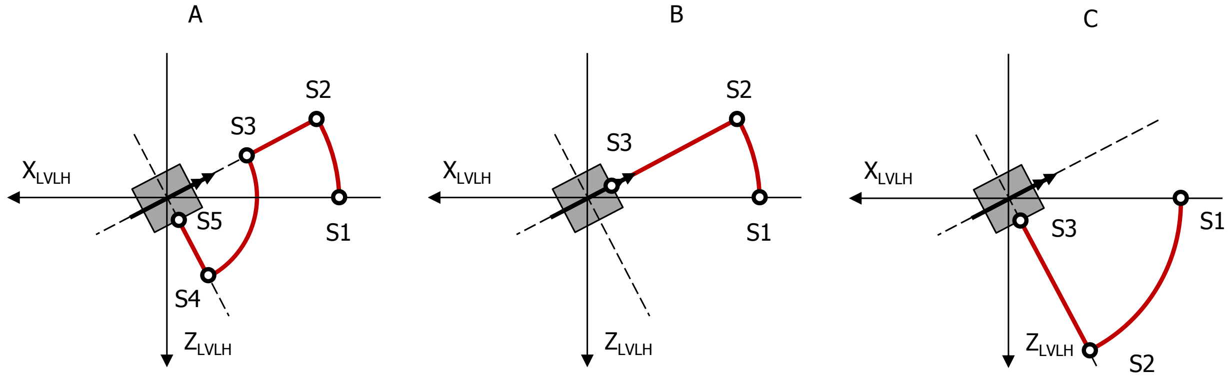


Figure 6-4: Proximity operations for a spinning satellite

Strategy A consists of the following elements:

- S1 to S2: Fly-around to a point on the body spin-axis
- S2 to S3: Perform straight-line approach over spin axis
- S3 to S4: Fly-around in target body fixed frame to grasping point
- S4 to S5: Perform straight-line forced motion to grasping point contact

Strategy A is designed for a fast spinning satellite. If the debris object is spinning fast, it will be costly to maintain a position perpendicular to the spin axis of the body, because the centrifugal acceleration needs to be compensated for by the thrusters. So if the target object is spinning fast it may be necessary to perform a first approach over the spin axis of the body to get fairly close to the target, followed by a fly-around to the grasping that is as short as possible to save propellant.

Strategy B and C are simplifications of strategy A. In strategy B, the debris object is grasped at a point on or very close to the spin axis of the body. In strategy C, the chaser performs a fly-around in the LVLH frame to the expected location of the projection of the approach direction on a sphere with radius 20 m at S2. At S2, the chaser starts following the approach direction in the objects body frame and performs the approach to the target object from S2 to S3 in the target body frame.

The strategy implemented here is strategy A. This strategy contains all the manoeuvres and trajectories that are also present in strategy B and C.

Test campaign has been run in for this scenario and the resulting trajectory, the error in the state vector, the acceleration, quaternion components, angular rates and manoeuvres monitored to check that the guidance and control functions perform as expected.

Figure 6-5 shows the LVLH trajectory and the rotated spacecraft at a specific instant in time. The Proba-2 spacecraft is at the centre of the frame, AnDROiD at the top right. The trajectory traced out by Proba-2's angular velocity vector on a sphere with radius 10 m is shown in green. A sphere of 10 m is plotted in grey for reference. The guidance reference trajectory is plotted in blue, and the true trajectory is plotted in red. From this figure it can be concluded that the guidance trajectory is followed correctly.

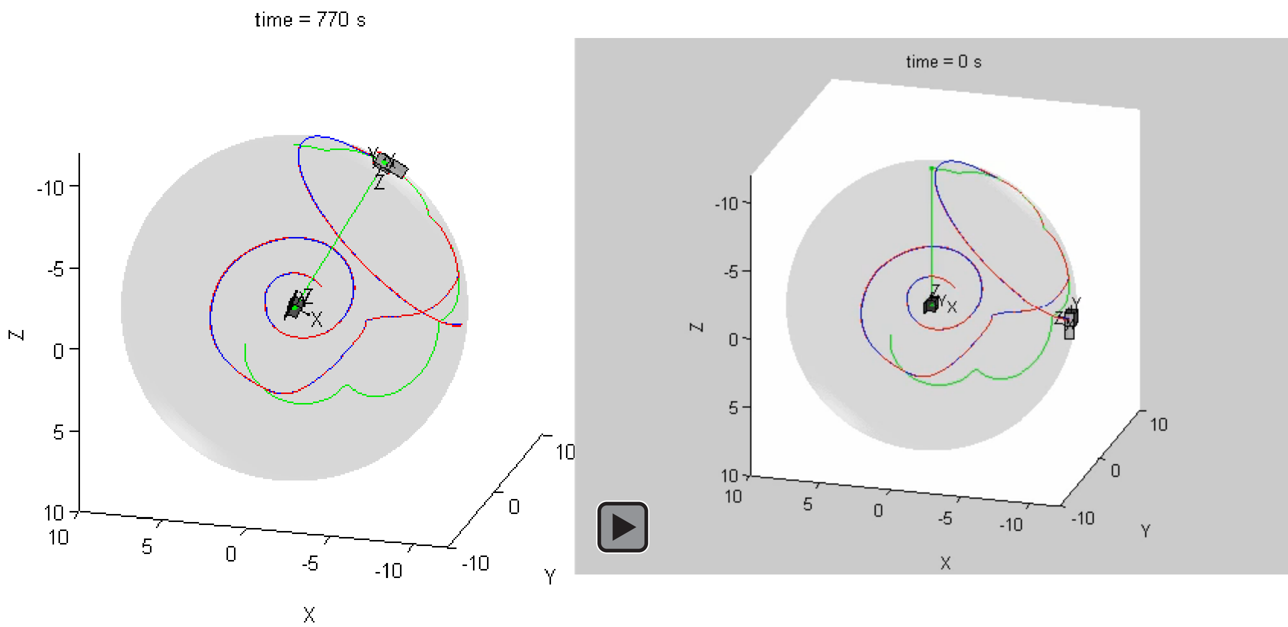


Figure 6-5: LVLH trajectory

Figure 6-6 shows a simulated camera view for a camera field of view of 20° taken at the same time as Figure 6-5 above.

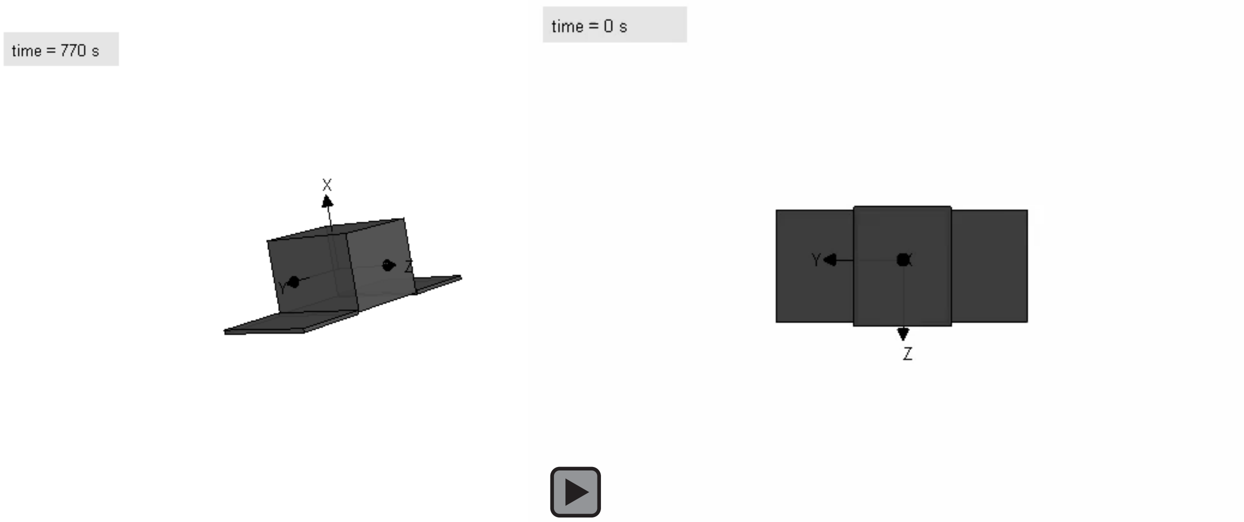


Figure 6-6: Simulated camera view

Figure 6-7 shows a simulated camera view near the end of the simulation, showing that the attitude of the chaser correctly follows the attitude of the target.

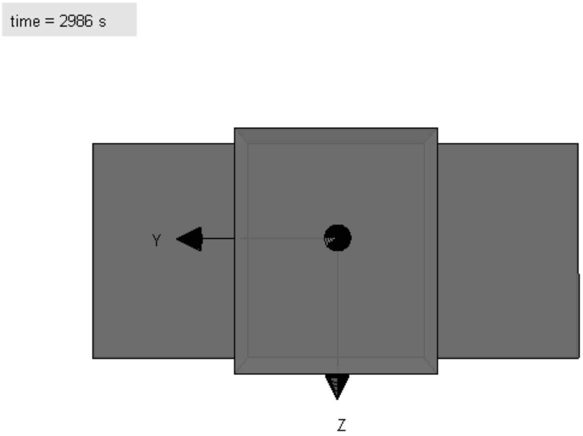


Figure 6-7: Simulated camera view near the end of the simulation

Small differences have been found in between the executed ∆V and the reference ∆V. The source if this discrepancy is due to the considered actuator efficiency and the controller design.

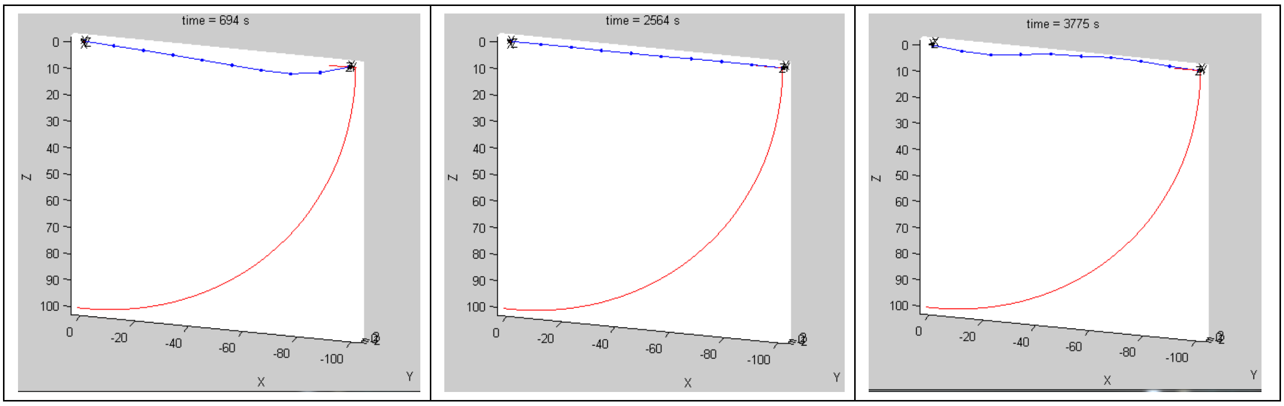
The guidance function behaves as expected with centimetre accuracy level and pointing errors under 0.1deg.In order to follow the reference trajectory "high" forces are required of up to 0.55N, which in turn translates into these attitude errors. The controller itself could be made more aggressive, leading to smaller errors in position and velocity, at the expense of ∆V.

6.3. SCENARIO 3

Scenario 3 focuses on tethered operations. The two main phases that are simulated are the system stabilisation phase and the deorbit. During the stabilisation phase the combo is stabilised after the tether has been attached (the process of attaching the tether is not simulated, but it is assumed that this is done by means of a net). The stabilisation could include moving to an equilibrium point on R bar. The deorbiting is a sequence of three manoeuvres to lower the perigee of the orbit of the combo till final direct re-entry in the atmosphere. The scenario has been divided into two tests:

- Single de-orbit burn with safe guidance mode
- Sequence of three de-orbit burns

Both tests have been run successfully. The next figure depicts the behaviour of the system for test 1.



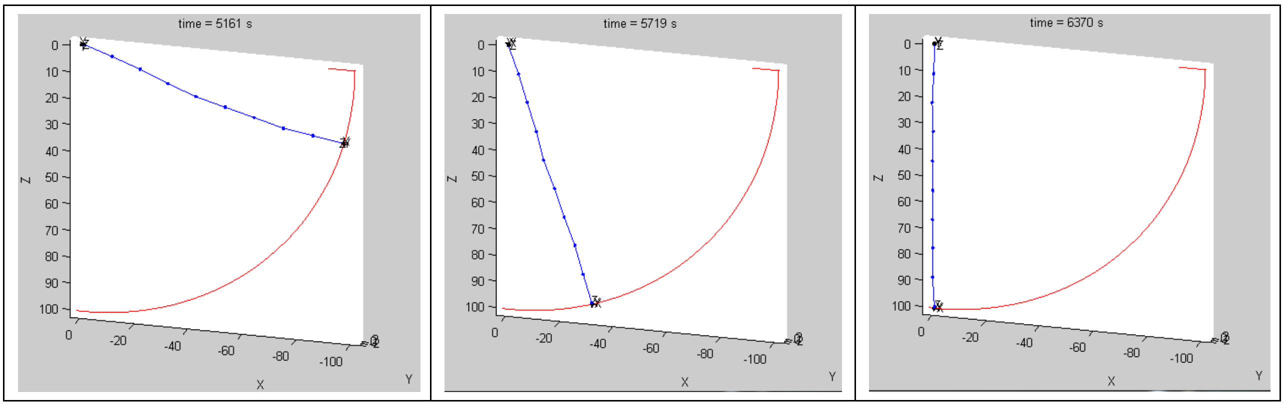


Table 6-1: Simulation results for Test 1 – Trajectory

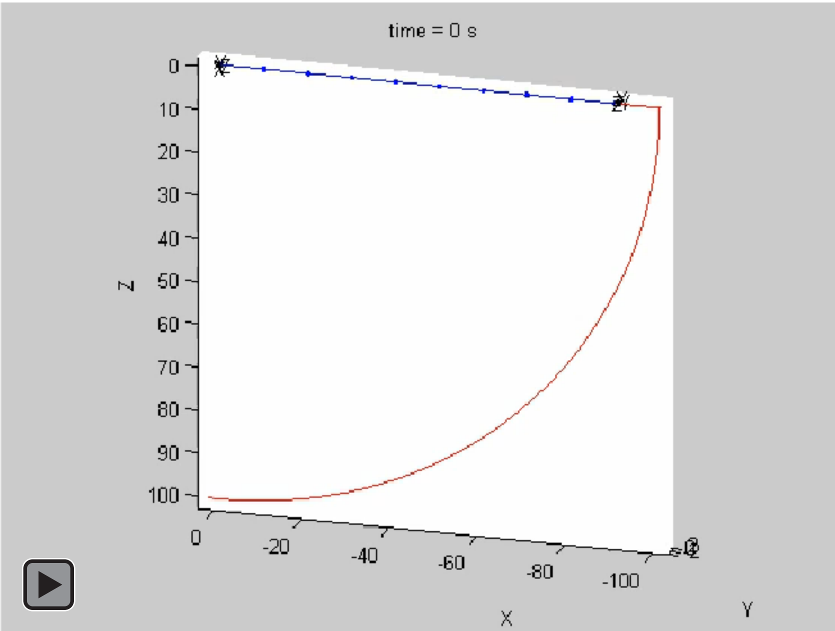


Table 6-2: Simulation results for Test 1 – Trajectory animated

The scenario initial conditions can be considered as initial errors from the no-pushing Vbar equilibrium that are properly handled by the controller in the simulations. When the operational phase starts there is a switching-on transient that excites the tension mode causing oscillations. But these oscillations seem not be a great issue to damp. Then, during the operational phase, the controller guarantees good performances in stabilizing the chaser in the proper equilibrium position.

Considering the low pushing force of 35.2 N, also the switching-off transient is not a problem. Indeed, during this transient the amount of potential energy that has been stored inside the tether is not high, and the "bouncing-back" effect is negligible.

As far as the transfer between Vbar and Rbar is concerned, no major problems have been encountered. However, it is important to remark the importance of maintaining the tether slightly elongated in order to maintain the control of the target attitude. This way, also the tether oscillations are minimized.

The equilibrium on Rbar has been noticed to be considerably stable, as expected considering the preliminary study. It should also be noted that the "bouncing back" effect is small (V_bar, at 2600 s) and that the motion is maintained in-plane (the motion along Y_bar is of the order of nm).

The following figure shows the results of the second test in which the three deorbit burns have been simulated.

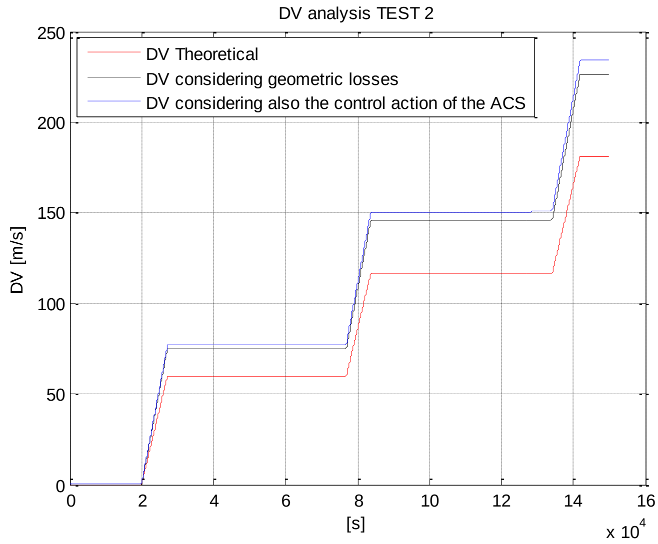


Figure 6-8: ΔV for the de-orbiting phase of three burnings

The first thing to notice from Figure 6-8 is the amount of ΔV that has to be taken into account for the propellant consumption of AnDROiD (blue line) is considerably higher than the nominal one (red line). The main reason for this is that the ideal one does not include the losses due to the geometry of the system (around 30%). The black line shows the orbital manoeuvres ΔV taking into account this geometrical losses and the blue line indicates the total ΔV, including the GNC control ΔV. It has to be noticed that the GNC ΔV is below 8m/s for a total ΔV of around 230m/s.

The results are satisfactory. The bounce back after a burn is small and can be controlled with the rdv thrusters. The main open point is related to the navigation system. Tensiometer will be required to better estimate the length of the tether and maybe a tether tension based controller.

7. PROGRAMMATIC ASESSMENT

Programmatic assessment has been carried out to estimate the system development time and cost, based on the mission design proposed at this stage (which has a low level of definition, not even a Phase A has been carried out). In an active debris removal mission the most critical technologies are those directly involved in the goal of the mission, which is capture and deorbit a non cooperative object. That is, the capture mechanism and the guidance navigation and control system. In the present study special attention has been paid to these areas. Two capture mechanisms have been analysed, a rigid one consisting in a robotic arm and a flexible one consisting in a net system. The GNC system has also been analysed in detail.

Identification of critical technologies in all these areas has been carried out together with an assessment of the current TRL level and the development plans required to bring those technologies to a flight ready status for the presented mission. These analyses have been integrated into a mission development plan, leading to a mission master schedule.

Required launch date before 2018 could be achieved if no margins are considered, which is not realistic. The robotic arm development is at this stage in the critical path. System could be delivered for integration with the platform by mid-2017, leading to a launch by mid-2018. It should be noted that given the launch strategy, the final launch date will be set by the main passenger.

A cost assessment exercise has been carried out at the end of the study over the proposed baseline. It shall be noted that the exercise has been carried out over a system with a low level of definition, and without taking margins into account. This exercise should be re-evaluated during the course of next phase as the level of definition is increased, so that a value with a higher confidence level can be achieved. The result of this first costing exercise indicated that the mission could fit into the cost envelope of an IOD mission.

8. CONCLUSIONS

A mission design for AnDROiD Active Debris Removal demonstration mission has been presented. The mission is mainly devoted do the demonstration of following key ADR technologies:

- GNC system for rendezvous, capture and deorbiting
- Robotic arm for debris capture and deorbiting
- Net system for debris capture and deorbiting

The mission “debris” target, PROBA 2, has been characterized both in terms of physical properties (including possible grasping points) and of expected dynamic status. Dedicated observation campaign will be desirable to validate assumptions made during the course of the study, mainly in terms of spin rate.

AnDROiD mission timeline has been analysed together with the DV budget. In terms of timeline, comfortable margins are available to carry out all the requested demonstrations and additional experiments. Proper scheduling could lead to the removal of operations outside regular working hours, an effort that goes in line with AnDROiD goal of reducing ground costs by increasing system autonomy. In terms of ΔV (driver for spacecraft size), 85% is due to orbital manoeuvres, mainly target orbit matching and final direct deorbit. There is room for optimisation of the strategies and therefore bring the requirement down.

. In order to maximise mission output, a large list of experiments has been proposed both in terms of hardware and software. Once the envelope of the mission is better defined, a selection of additional experiments could be carried out. Note that, nonetheless, the currently defined mission timeline accounts for both the demonstration of the main mission goals (GNC, robotic arm capture, net system capture and final deorbit) plus a set of selected additional experiments

A conceptual platform design has been proposed, based on PROBA-NEXT. Accommodation, mass, power, memory and interface aspects have been discussed and a solution presented with no major issues. Further design iterations should be carried out to refine the solution.

The GNC preliminary design has been presented including architecture of the system, strategies definition, modes and mode transition definition and hardware selection for both actuators and sensors based on existing equipment. Main functionalities have been prototyped and simulated, and the proposed strategies verified.

A net system design has been presented tailored to the needs of the mission (target). The solution proposed is based on already proposed designs and on-going activities (i.e. Patender).

A robotic arm design has been proposed based on existing developments. The design to be further refined mainly in terms of power consumption, configuration and sensor suite to be used.

Preliminary analyses indicate that the AnDROiD proposed mission is feasible within the given timeframe and budgetary envelope of an ESA IOD mission, though more in depth analyses will be required in subsequent phases of the programme to increase the level of confidence and refine the programmatic assessment. Mission costs have been contained thanks to re-use of heritage platform hardware, in order to concentrate efforts on development of critical ADR technologies and to enable including additional technology experiments.

END OF DOCUMENT