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# SBSS-DM and ANDROID: two small missions for Space-Based Space Surveillance and Active Debris Removal Demonstrations

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## Abstract

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, 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.

This paper intends to recall and resume the results of two projects carried out by GMV in collaboration with QinetiQ in which two different demonstration missions' concepts targeting two central aspects of the debris remediation problem were outlined: 1) Space-Based Space Surveillance (SBSS-DM) and 2) Active Debris Removal (AnDROiD).

In the frame of the SBSS-DM assessment study, different architecture of operational mission were evaluated and several trade-off performed so to determine which solution/s could better fulfil the users requirements with reduced costs and while minimising the associated technological risks.

Similarly the AnDROiD mission was proposed as small scale missions focusing on the feasibility of removing small debris objects (100-200kg) while testing on orbit at least two different capture techniques before the actual deorbiting of the target, a net and a robotic arm.

## 1. Introduction

Due to the intensive activities in the space during the last half century, the population of man-made space objects is playing an increasingly important role in the space environment. Today more than 6000 satellites are orbiting around the Earth but only 900 are operational and the problem is going to grow in the future: almost 1200 new satellites are expected to be launched in the next 8 years based on a forecast by Euroconsult. The population of man-made space objects consists of approximately 6% operational spacecraft, 22% non-functional spacecraft, 17% rocket upper stages, 13% mission-related debris and 42% fragments from explosions or collisions [1].

The total mass of the population is estimated at 6300 tons. Figure 1 shows the distribution of debris in LEO. The highest concentration can be found at an inclination of 82-83° and around the sun-synchronous inclination. A large portion of the population at inclination 82-83° consists of objects launched from Plesetsk using the Cosmos-3M launch vehicle. The sun-synchronous orbit is of particularly high importance because of its usefulness for remote sensing and Earth observation purposes.

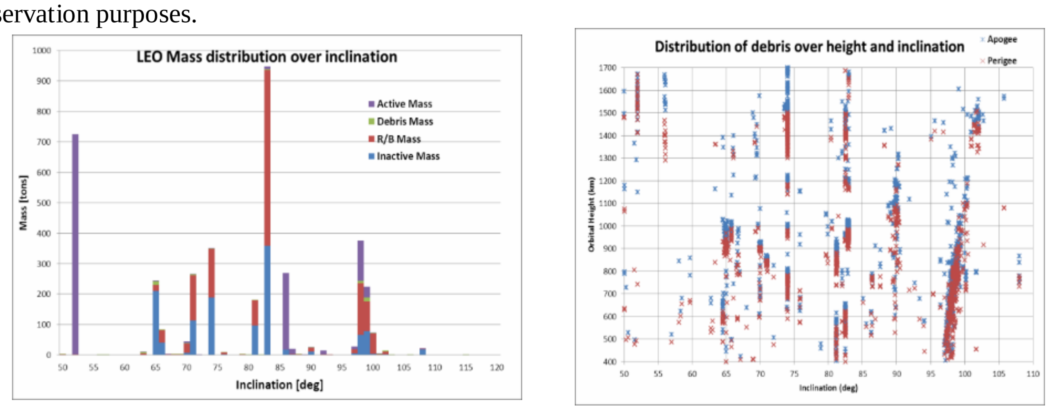


Figure 1: Debris distribution in LEO orbits

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In the last years almost the totality of space actors have devoted a not negligible effort towards the analysis of the effective risk posed by the increasing number of space debris, the evaluation of possible both mitigation and remediation techniques. Nevertheless significant challenges are faced by spacefaring organisations to implement these measures. Currently it is not yet practical to remove anything but the largest debris objects. Such an approach would nevertheless make sense, because the large objects tend to be the primary source of new small debris and because 99% of the total mass of the debris is concentrated in the large objects.
Nevertheless still a part of the required technologies, mostly for what concerns the active removal of space debris, have not all yet demonstrated in space or even in representative environments. This fact together with the still existing limitation in terms of legal regulation are still slowing dawn the process of setting up a debris removal service.
Demonstrating critical technologies on small missions could potentially be an asset to give momentum to the same.
The following paragraphs will concentrate in defining two small demonstration missions which concepts intend to depict two central aspects of the debris remediation problem being the Space-Based Space Surveillance and an Active Debris Removal.
Both concept have been the results of a collaboration between GMV and QinetiQ under two separate ESA's contracts.

## 1.1 SBSS-DM: Space Based Space Surveillance Demonstration Mission [1]

The SBSS-DM mission is the results of an assessment study financed by ESA to define a demonstration mission for space-based space surveillance. The project team included QinetiQ Space as responsible for the platform and RAL Space for the payload activities.
The study has been organized into two phases. The first one was devoted to the high-level definition of a possible Space Based Space Surveillance (SBSS) operational service.
During the second phase of the study the effort was devoted to the definition of a precursor mission to demonstrate the SBSS operational needs and the fulfilment of the same user requirements used to dimension the operational service.
Even if this paper will offer a high level view of the SBSS operational mission, the focus of the same will be the SBSS-DM demonstration mission. Details about the operational scenario may be found in [1].
In line with this, the main goal of this SBSS demonstration mission is the demonstration of the most critical aspects of the complete SBSS operational mission with the scope mitigate/reduce the risks of a full operational system. Nevertheless the proposed demonstration mission is fully representative (except in the number of spacecraft) of the operational system and in line with typical ESA IOD requirements.

### 1.1.1 OUTLINE OF THE SBSS OPERATIONAL MISSION REQUIREMENTS AND NEEDS

*User Requirements*
The user requirements to be used as dimensioning case and starting point for the SBSS-DM mission definition have been derived taking advantage of the experience of the team in different ESA's SSA projects. The most relevant implications of the user requirements discussed and agreed with ESA were related to the priority assigned to the space surveillance in the geosynchronous orbit (GEO) region. As consequence, even if a potential SBSS operational system had to be used for the observation of objects beyond low Earth orbit (LEO) both in surveillance and tracking mode, the priority was granted to the GEO region. That is, objects in medium Earth orbit (MEO), geosynchronous transfer orbit (GTO) and high elliptical orbits (HEO) shall be observed as well. In order to complete the proposed observation, it was imposed as requirement also the feasibility of observing NEO objects during nominal as well as during ad-hoc observation campaigns.
In the GEO region, objects larger than 70cm shall be observed and pre catalogued in less than 72 hours. Once pre catalogued, they should become part of the full catalogue in less than 3 days. The system shall provide revisit times shorter than 72 hours for catalogued objects. In order to include a detected object in the catalogue, the accuracy envelope shall be better than 2.5km. These requirements shall be achieved by the SBSS operational system in conjunction with the ground based segment of the SST (Space Surveillance and Tracking) system that would be also available in the future.
In terms of reactiveness, the SBSS operational system shall be able to fulfil a tracking request for any GEO "catalogued" or "pre-catalogued" object before 48 hours after the request is issued. This requires that the space segment shall be able to be re-planned or re-programmed in less than 24 hours to accommodate the active tracking request. During planned operations, the system shall have autonomy of 7 days.
Finally, the system shall be available the 90% of the lifetime once the system becomes operational. In terms of lifetime, 50 years have been initially defined, which implies that the space segment will have to be replaced in a regular basis. Each spacecraft shall be designed for a target lifetime of 7 years.

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*Observation strategy*

In order to provide enough data for the orbit determination process, an object in the GEO ring shall be observed in different positions along its orbit during at least three days.

The observation strategy consists [1] of the quick scanning of the four declination stripes, changing the pointing of the spacecraft approximately every minute to place the FOV (4x4 degrees) at different declination. Each declination stripe is divided in 9 fields equal to the extent of the FOV. The declination stripe is covered 4 times every half orbit, changing the spacecraft the target stripe to the symmetrical one every time it crosses the polar region (so that the illumination conditions are optimised and it is avoided that the Earth enters in the FOV of the payload).

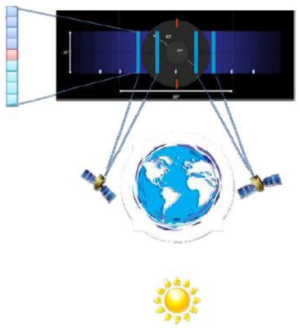

Figure 2: Observation strategy

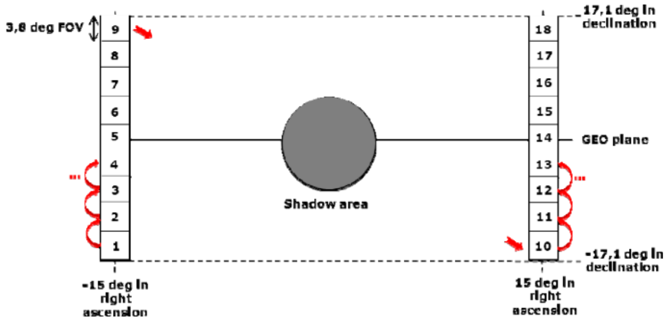


Figure 3: Strategy of pointing dividing the strip in 9 fields

This strategy has been designed to avoid that an object could be missed and cross the stripe being undetected (leak proof). The time allocated to the scanning of a stripe is set so that a GEO object will not completely cross the stripe width during the scanning period. Each field will be observed during almost one minute, which will enable the acquisition of at least 5 images. Each spacecraft will acquire at least 360 images per orbit (over five thousand per day). This fact drives the need to perform some image processing on board to reduce the size of the required communications link with ground.

This observation strategy implies that the spacecraft will have to perform frequent pointing manoeuvres: change from field to field (4 degrees), re-scanning the declination stripe (32 degrees), changing to the symmetrical stripe (up to 77 degress+32 degrees). The manoeuvre times and tranquilisation times have been taken into account in the definition of the observation strategy. Four more manoeuvres of 90 deg around the Line of Sight (LoS) have been also defined to homogenise the thermal environment along a single orbit.

*SBSS operational mission architecture outline*

The proposed SBSS operational system is composed of a constellation of 4 spacecraft placed at the same dawn-dusk (local time of the ascending node at 6:00 AM) sun synchronous orbit (SSO) at 888.3 km of altitude and 98.91 degrees of inclination. Dawn dusk orbit provides the best illumination conditions for the detection of objects in the GEO ring while providing a stable thermal and power generation environment for the platform. The altitude is selected so that the earth does not interfere with the field of view (FOV) of the instrument in any of the observation configurations and the orbital period is such that the movement of GEO objects and the movement of the SBSS spacecraft is synchronised (integer number of orbits per day).

The satellites will be equally spaced in this orbit, separated by 90 deg in true anomaly and work in pairs separated by 180 degrees. The four satellites will be launched in a single VEGA launch using a modified VESPA multi-payload adapter.

Three of the satellites will be devoted to the continuous surveillance of the GEO ring, scanning four stripes (fixed right ascension, declination between -18/18 degrees) located at +/-15 degrees and +/- 38.5 degrees with respect to the Sun-Earth direction, opposite to the Sun. The first couple of satellites, separated 180 degrees in true anomaly, will scan the stripes at +/-15 deg. The third spacecraft will scan alternatively the stripe at + and – 38.5 deg. The fourth spacecraft will share the surveillance tasks (complement the third spacecraft) with the response to active tracking requests from ground segment.

*Payload*

Table 1 below shows the key parameters of the SBSS payload, being this the same for the operational as well as the demonstration mission.

A SNR of at least 4 is required for detection of a 0.7 m object in GEO. The telescope that is defined for the mission has a square FOV of 4° x 4° and an aperture of 350 mm. It includes a detector of 4k x 4k elements, giving a plate scale of 3.5 arcsec per pixel. To avoid the sky background swamping the signal, the exposure time must be limited to the time taken for an object to cross one pixel, with the object track being reconstructed from a series of consecutive

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images. For a typical GEO object the pixel crossing time is 0.23 s. To meet the observation strategy, an image must be acquired every 5 seconds, implying a read-out rate of 0.8 Mpixels / s if the pixels are read out through 4 parallel ports.

Table 1: Key parameters of the payload

| | |
|---|---|
| SNR (0.7m object in GEO) | Min 4 |
| Entrance pupil diameter | 350 mm |
| Field of view | 4° x 4° (square, full angle) |
| No. pixels | 4k x 4k |
| Exposure time | 0.23 s (= one pixel crossing) |
| Plate scale | 3.5 “/pixel |
| Read out rate | 0.8 M pix / s (one image / 5s) |

The telescope optical design, selected after an extensive trade-off, is a Ritchey–Chrétien telescope with a three-element refractive field flattener. Figure 4 shows a cross section through the telescope. A baffle arrangement has been developed and analysed for stray light performance, with the results feeding into the SNR model.
The telescope structure is made from carbon fibre reinforced plastic in order to maximise stability while minimising mass. The mirrors are made from Zerodur and light-weighting techniques are applied to reduce their mass. The telescope is equipped with a door in order to maintain cleanliness; in the open position the door also acts as baffle for stray light from the Earth. A piezo-electrically driven focus mechanism, actuating the detector, is included to allow re-focussing on orbit.
The detector is a custom CMOS detector. CMOS is chosen over a CCD because of the absence of frameshift smear, its greater radiation tolerance, its lower power consumption and simpler read-out electronics.

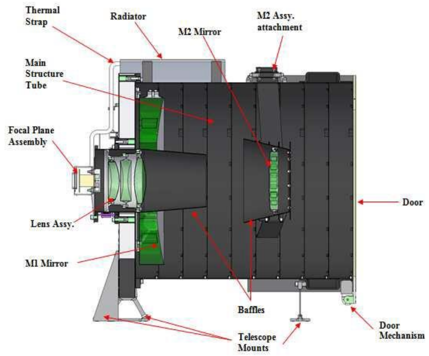


Figure 4: Strategy Cross section through the payload

*Performances*
During the first part of the project, the performances of the complete SBSS system have been evaluated. Several simulations have been run for different epochs to take into account the effect of the illumination conditions long the year.
The results of the simulations are summarised in the following table. The results are based on 14 days of simulation, neglecting the data of the first 4 days for cataloguing purposes.

Table 2: SBSS operational system performances

| **Detectability** | **Low LEO** | **High MEO** | **GEO** | **HEO** |
|---|---|---|---|---|
| Spring Equinox | 40.7% | 60.4% | 99.3% | 73.5% |
| Summer Solstice | 51.1% | 76.6% | 99.5% | 69.9% |
| Winter Solstice | 53.4% | 78.7% | 99.1% | 71.6% |

| **Pre-cataloguing** | **Low LEO** | **High MEO** | **GEO** | **HEO** |
|---|---|---|---|---|
| Spring Equinox | 0.0% | 9.8% | 98.1% | 6.8 % |
| Summer Solstice | 0.4% | 12.8% | 97.9% | 7.7 % |
| Winter Solstice | 1.2% | 15.1% | 97.9% | 7.7% |

| **Cataloguing** | **Low LEO** | **High MEO** | **GEO** | **HEO** |
|---|---|---|---|---|
| Spring Equinox | 0.0% | 6.2% | 94.7% | 3.7 % |
| Summer Solstice | 0.0% | 7.7% | 90.0% | 4.5 % |
| Winter Solstice | 0.1% | 6.4% | 94.1% | 4.3% |

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As can be seen, the performances are maximised for the detection and cataloguing of GEO objects, as initially requested. Indeed, a full catalogue can be built in less than 2 weeks. For the other orbital regimes the performances are pretty low in terms of cataloguing, but quite good in terms of detection. In order to improve the situation, specific observation strategy could be envisaged for the fourth satellite.

1.1.2 THE SBSS DEMONSTRATION MISSION

The main objective of the SBSS Demo Mission is to demonstrate the system performances of the operational system, mainly in terms of detectability and cataloguing. Therefore the main characteristics of the payload shall be replicated as well as the payload supporting services, like platform agility.

*SBSS-DM Observation strategy*
In order to ensure that the observation geometry of the operational system is maintained and open the possibility of using this first spacecraft as a component of the operational system the SBSS-DM will be injected in the same orbit as the S/C of the operational mission. , being this a dawn-dusk (local time of the ascending node at 6:00 AM) sun synchronous orbit (SSO) at 888.3 km of altitude and 98.91 degrees of inclination.
The same observation strategy as the one followed by each operational spacecraft will be exercised by the SBSS-DM. New observation strategies will be implemented to be able to demonstrate the detectability performances during different epochs along one year. In this way the exact illumination conditions of the complete service will be replicated.

*SBSS platform*
The platform design is based on heritage from ESA's PROBA family [1-4], developed by QinetiQ Space. Some adaptations are necessary to accommodate the relatively large payload.
It shall be stressed that the same platform will be used both for the operational as well as the demo mission.
Such as its predecessors, the SBSS will be an accurate and agile platform, allowing to meet the pointing requirements and to perform frequent scanning manoeuvres needed to change the FOV declination.
During each observation period, the telescope will be pointing to the same point on the GEO region, minimising at the same time the rotation of the FOV around the LOS direction. Four 90° manoeuvres per orbit (two of them matching the polar manoeuvres above mentioned) will be performed to improve the S/C thermal conditions and to avoid star tracker blinding by the earth (see Figure 5).

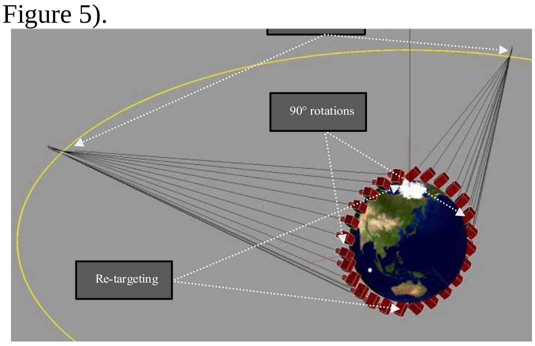


Fig. 5: S/C attitude over one orbit, with GEO targets at 38.5° sun illumination

The SBSS platform is represented in Figure 6 and Figure 7. It has a mass of about 170kg for dimensions of about 1.4mx0.9mx0.9m. Three body-mounted solar panels will provide power to the bus and the payload, while the Advanced Data and Power Management System (ADPMS), will take care of the power conditioning and distribution. Depending on the observed GEO-target and the presence of eclipses (in winter time) the platform will deliver between 160W and 200W average power. The ADPMS, developed by QinetiQ Space, also serves as on-board computer, it hosts a LEON2-processor. The ADPMS will host the mission software and be in charge of main operations, but a dedicated electronic unit will be implemented to perform the on board image processing tasks and interface between the payload and the computer.

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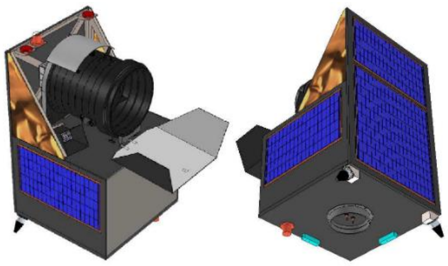

Fig. 6: SBSS platform with telescope (external view)

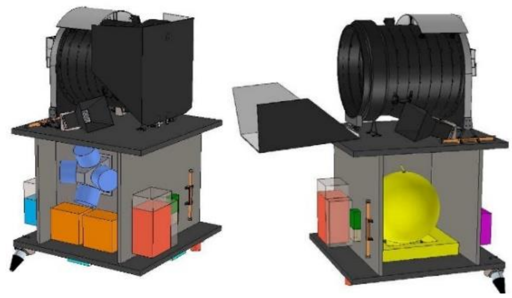

Fig. 7: SBSS platform with telescope (internal view)

The SBSS platform also accommodates a 1N (HPGP) propulsion system. About 140m/s of propellant will be needed in order to correct for launch injection errors, to perform orbit maintenance, for collision avoidance and finally to de-orbit the spacecraft at end-of-life (which requires over 75% of the total Delta-V capabilities of the spacecraft).
An S-band communication system is used for uplink to the spacecraft and for downlink of housekeeping telemetry. The S-band antennas are accommodated such as to guarantee omni-directional coverage throughout the mission. Payload data is downlinked via X-band, at a rate of 33Mbps.
The thermal design is mainly passive. The payload is radiatively and conductively decoupled from the platform. Survival heaters are installed for the critical components, such as the battery, the propulsion system and payload.
A summary of the main platform characteristics is given in Table 3.

Table 3: SBSS platform characteristic summary

| **SBSS platform** | |
|---|---|
| Avionics | ADPMS (Advanced Data and Power Management System) Processor: LEON2-E (SPARC V8)<br>Mass Memory Module : 16 Gbit (baseline), 11 GByte available<br>Interfaces: RS422, TTC-B-01, analogue and digital status lines, Packetwire, compact PCI |
| Power | Solar panels: body-mounted GaAs solar cells with 28 % efficiency<br>Battery: Li-ion, 28V, 12Ah<br>Bus: 28V battery regulated voltage |
| Structure | Aluminium inner H-structure Aluminium milled bottom board CFRP outer panels with solar arrays<br>Aluminium payload and anti-sun panels |
| AOCS | axis stabilised satellite Actuators:<br>3 magnetotorquers (internally redundant)<br>4 reaction wheels<br>1N HPGP propulsion system Sensors:<br>2 magnetometers<br>2 star tracker (with 2 camera head units)<br>2 GPS receivers Pointing performances:<br>AKE (Absolute Knowledge Error) - 4 arcsec<br>APE (Absolute Pointing Error) - 27 arcsec<br>RPE (Relative Pointing Error) - 1 arcsec over 1.5s - 5 arcsec over 60s |
| Communication | S-band downlink: 1Msps S-band uplink: 64ksps<br>X-band downlink: 33Mbit/s |
| Software | Operating system: RTEMS<br>Data handling/application software: based on PROBA OBSW |
| Thermal | Mainly passive thermal control, heaters for the battery and the payload |

The SBSS spacecraft is completely redundant spacecraft with the exception of the payload. The spacecraft bus is single point failure safe. Hot redundancy is foreseen for the S-band receivers, the star tracker cameras and certain parts of the power management in ADPMS.
Cold redundancy is foreseen for the transmitters, the on-board computer and the AOCS sensors and actuators.
High levels of reliability and autonomy are also achieved by implementation of an advanced Failure Detection, Isolation and Recovery (FDIR) approach. Anomalies are handled on-board without need for ground intervention.

*Ground Segment*

The main components of the both SBSS operational mission and DM ground segment will be:

- Ground station, preliminarily selected at Kiruna. It provides enough visibility to download the payload data on a regular basis (only 4 blind orbits per day) and to update the satellites tasking with enough time to respond to active tracking requests (the requirement is to be able to respond to these requests in less than 48 hr, no tough requirement has been imposed to data timeliness).

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- Flight Operation Segment (FOS), containing the spacecraft monitoring and control, flight dynamics, mission planning, data acquisition and external data acquisition functions.
- Payload Data Ground Segment (PDGS), containing the data processing, calibration and validation, archive and cataloguing, dissemination, and performance monitoring functions.

*SBSS-DM Performances*
In order to demonstrate the cataloguing performances, specific observation strategies have to be designed.
With a single spacecraft it is not possible to define a leak proof strategy for the GEO region (a minimum of two spacecraft are required for detection of all GEO objects and three for cataloguing purposes).
Nevertheless the SBSS-DM intends to be representative of the operational services in terms of cataloguing performances. Consequently the focus has been set in observing a subset of the GEO population in such a way that three different observations can be obtained per day.
The first strategy consists in scanning declination stripes at +38.5 deg, +15deg and -15 deg in a synchronised way with respect to the movement of a GEO object along a day. With this strategy it is not possible to avoid that the Earth enters the FoV of the instrument at some points, and therefore the achieved cataloguing performances are not very high, although it is the most representative strategy.
A second strategy has been designed in which three responsiveness declination stripes are observed at +45deg and +/-19deg. Each stripe will be swept four times in half the satellite´s period, and it will change to a different stripe in the Earth´s poles. In this strategy, the detectability becomes slightly worst due to the illumination conditions, but it provides a more systematic observation for cataloguing purposes.
Figure 8 resumes the performances achieved applying the mentioned strategies to the SBSS-DM, while compared with the results obtained for the SBSS operational service.

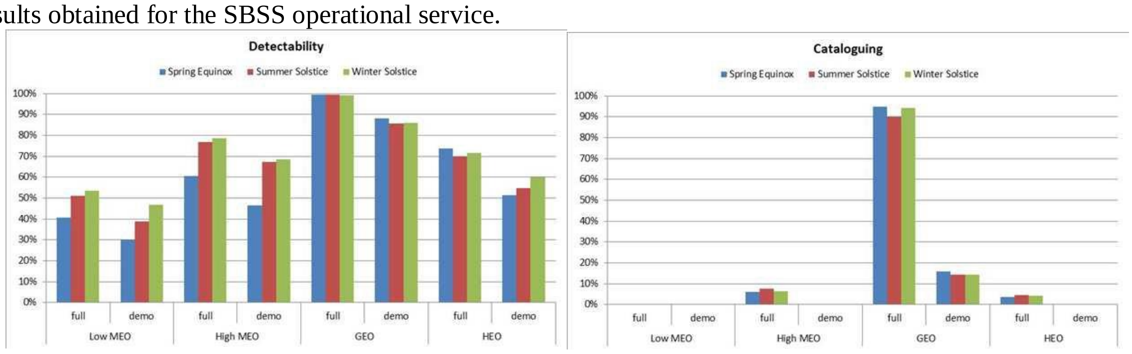


Fig. 8: SBSS-DM performances

## 1.2 AnDRoID: Active Debris Removal In-orbit Demonstration mission [3], [4]

*The AnDRoID mission is proposed as a small scale mission focusing on small debris objects (100-200kg) active removal. The goal of the AnDRoID mission is than to offer an affordable solution for removing these small targets. It could also potentially serve as a test platform to exercise technologies and strategies required for other missions with different class targets so to reduce the costs and the risks of a bigger mission.

AnDRoID has been studied by GMV with the support of QinetiQ Space for platform design, Space Research Centre Polish Academy of Sciences for the robotic arm design and GMV Romania for the net system design under ESA's contract.

### 1.2.1 AnDRoID MISSION REQUIREMENTS AND ANALYSIS

In line with its objectives of testing as much as possible technologies to optimize in terms of costs and risk future ADR operational/service missions, 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. Thus a deep 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).

Similarly the concept has been investigated such to obtain as output a ground operations methodology/procedures designed in order to maximise on-board autonomy thereby reducing ground operations costs.

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With the scope of reducing as much as possible the risk of de-orbing a dead body, the PROBA-2 satellite has been identified as target debris. It shall be recalled that PROBA-2 is a Belgium satellite build by QinetiQ, being this last also system responsible of the proposed concept.

The main goal of the PROBA-2 mission was also technology demonstration while at the same time provide scientific observations of the Sun. PROBA-2 was placed in a sun-synchronous orbit at 718km altitude with a local time of ascending node of 06h24 am. At the time of the AnDRoID study it was hypnotised that PROBA2 was likely to be non-operational and thus considered as debris. Initial analysis indicates that it should be spinning at an angular rate of 5 revolutions per orbit.
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).

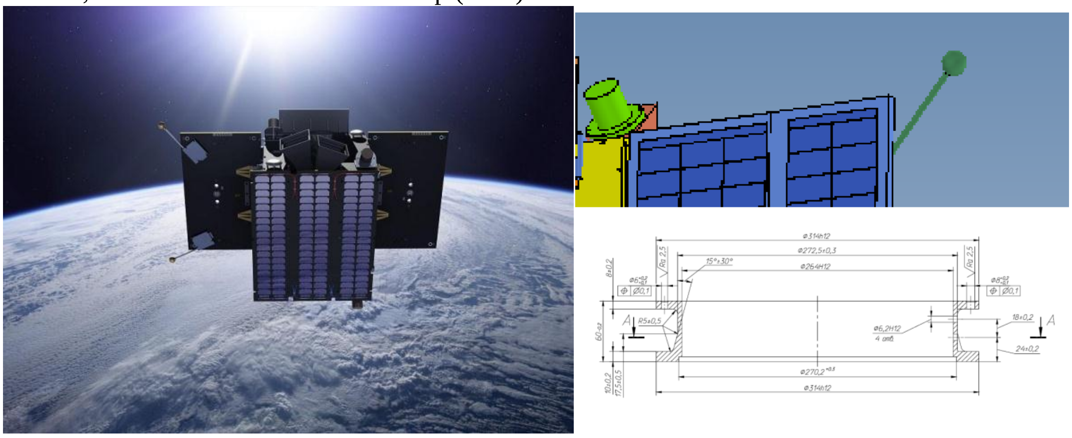

Fig. 9: PROBA2 image and grasping points

AnDRoID is to be launched in a shared launch into LEO. The total mass of the system is expected to be under 350kg, with a total envelope with appendages of 1188(D) x 1133(W) x 1145(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.
The following table summarises the **Δ**V 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 4. AnDRoID mission timeline and **Δ**-V budget (without margins)

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

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).

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Table 5: Sequence of deorbit manoeuvres Targeted

| Targeted perigee altitude [km] | ΔV [m/s] | Diff (%) |
|---|---|---|
| 500 | 59.919 | 1.19 |
| 300 | 56.471 | 1.02 |
| 80 | 64.748 | 1.36 |

Fig. 10 summarizes the AnDRoID Mission timeline. As from below figures, the capture with the robotic arm will be performed from a safe orbit with the caser in free floating mode, while the capture with the net system will be performed from a hold point in V bar. Finally the deorbit burn will be performed following a direct control re-entry. In order to reduce the gravity losses the manoeuvre will be split in three consecutive burns.

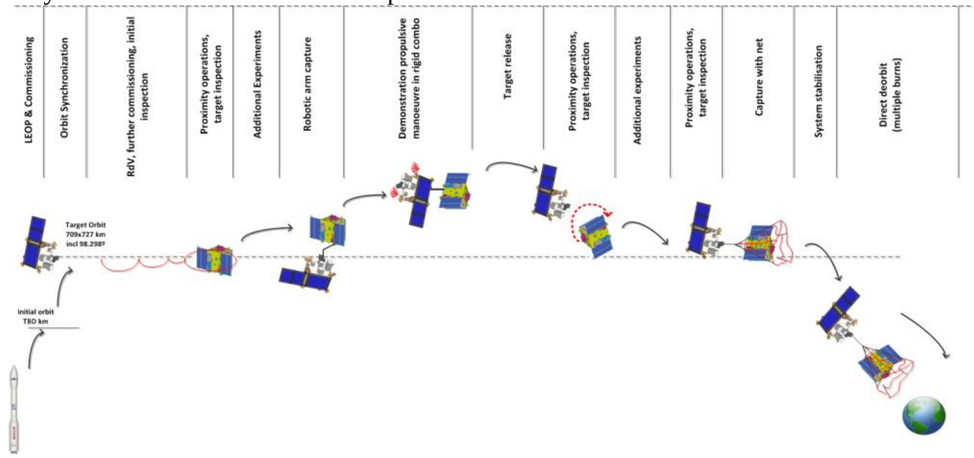


Figure 10: AnDRoID Mission timeline

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 6).

Table 6: Additional experiments

| 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 net 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.

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- 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.

### 1.2.2 THE AnDRoID DEMONSTRATION MISSION

*System Design*
The system is composed of a single spacecraft with a total wet mass of about 350kg including margins (see Tab. 2 below). Out of this mass 68 kg are propellant, about 20% of the launch mass. In case of larger propellant needs, the platform could be enlarged accommodating a larger hydrazine propellant tank. The main contributors to the mass of the system are the GNC equipment and the capture mechanisms comprising the robotic arm and the net system.

Table 7: AnDRoID mass budget

| **Subsystem** | **Total mass [kg]** | **Mass margin [kg]** | **Total mass w margin [kg]** |
|---|---|---|---|
| Structures | 71.73 | 13.33 | 86.06 |
| Thermal control | 0.77 | 0.15 | 0.92 |
| Communications | 9.22 | 1.54 | 10.76 |
| ADPMS | 15.40 | 1.54 | 16.94 |
| GNC | 34.02 | 3.11 | 37.13 |
| Propulsion | 17.66 | 2.19 | 19.85 |
| Power | 5.46 | 0.53 | 5.99 |
| Harness | 6.50 | 1.30 | 7.80 |
| Capture systems | 35.50 | 7.10 | 42.60 |
| Total [kg] | 196.26 | | 228.05 |
| System Margin (20%) [kg] | | | 45.61 |
| **Total dry mass [kg]** | | | **273.66** |
| Propellant mass [kg] | | | 68.00 |
| **Total wet mass [kg]** | | | **341.66** |
| Launcher IF ring [kg] | | | 6.00 |
| **Total launch mass [kg]** | | | **347.66** |

*Platform Design*
The ANDROID spacecraft design is based on the PROBA-NEXT platform, which is the successor of the PROBA1, PROBA2 [1] and PROBA-V(egetation) [2] satellites developed by QinetiQ Space.

The PROBA-NEXT platform is a fully redundant all-purpose and generic platform that can host payloads in the range of 150kg and that can deliver more than 600W of power. It is a 3-axis stabilized platform providing a high pointing accuracy, with pointing errors below 30arcsec (95% confidence level). While the baseline configuration of the PROBA-NEXT platform offers upgraded downlink capacity (200Mbps) and mass memory storage (2Tbits) compared to the other members of the PROBA-family, it was decided to re-use the subsystems from PROBA-V, as they are in line with the requirements of the ANDROID mission.
The PROBA-NEXT platform has a 30 liter propulsion tank (23kg propellant) capacity, resulting in a delta-V of 200m/s for a typical 220kg S/C satellite. The ANDROID mission however calls for a significantly larger amount of propellant, as it needs to fully de-orbit the target satellite as well. This imposes the use of a propulsion tank with a volume of 90 liters. Therefore, the PROBA-NEXT platform structure was scaled up to fit the dimensions of the tank. The resulting total wet mass is about 275kg.

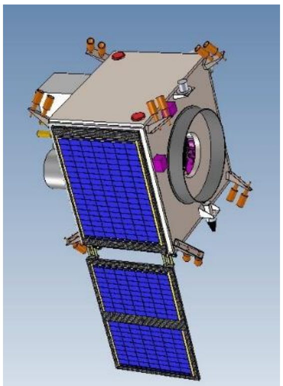
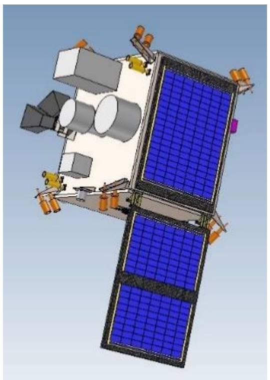

Figure 11: External view of the spacecraft with deployed panel (solar side)

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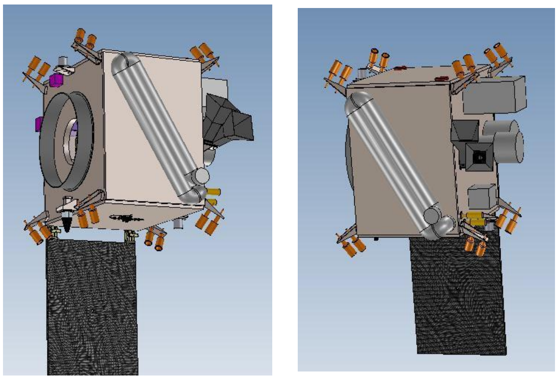

Figure 12: External view of the spacecraft with deployed panel (anti-solar side)

Table 8: AnDRoID platform

| AnDROiD platform | |
|---|---|
| Avionics | ADPMS (Advanced Data and Power Management System)<br>Processor: LEON2-E (SPARC V8)<br>Mass Memory Module : 11 GByte<br>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<br>Battery: Li-ion, 28V, 12Ah<br>Bus: 28V battery regulated voltage |
| Structure | Aluminium outer panels<br>Aluminium milled bottom board<br>CFRP outer panels with solar arrays |
| AOCS | 3-axis stabilised satellite<br>Actuators:<br>• 3 magnetotorquers (internally redundant)<br>• 4 reaction wheels<br>• 1N Hydrazine propulsion system<br>• 20N Hydrazine propulsion system<br>Sensors:<br>• 2 magnetometers<br>• 2 star tracker (with 2 camera head units)<br>• 2 GPS receivers<br>• 1 navigation camera<br>• 1 inertial measurement unit<br>• 3 sun sensors (TBC)<br>• 1 rendez-vous sensor (TBC) |
| Communication | S-band downlink: 827kbit/s<br>S-band uplink: 64ksps<br>X-band downlink: 33Mbit/s |
| Software | Operating system: RTEMS<br>Data handling/application software: based on PROBA-V OBSW |
| Thermal | Mainly passive thermal control, heaters for the battery, the propulsion subsystem and the payload |

*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 sub-modes 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.

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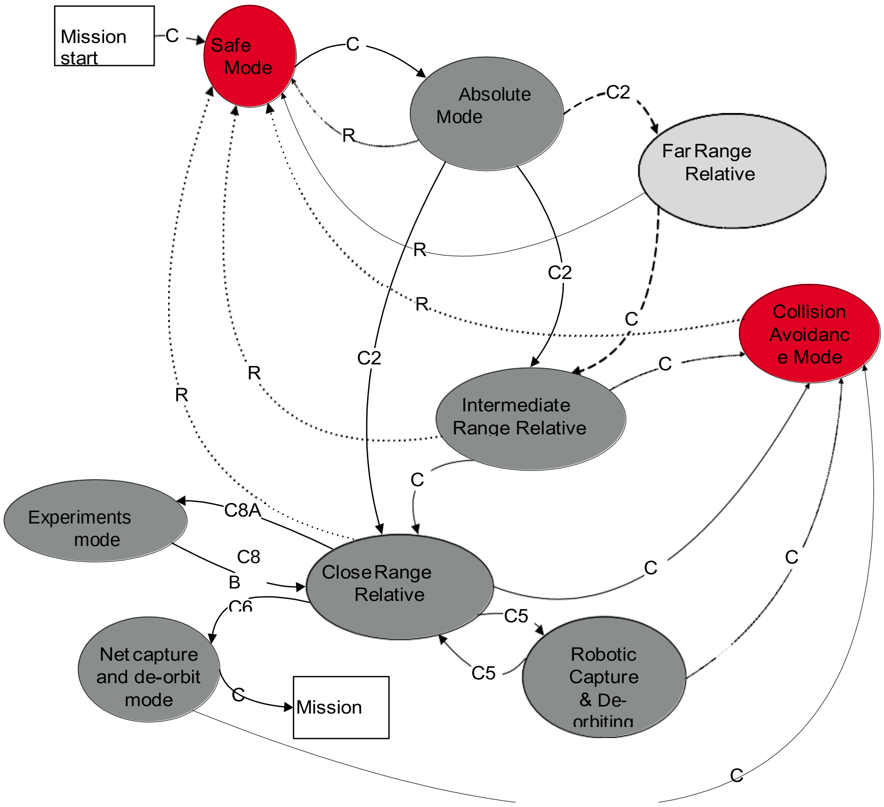


Figure 13: High-level GNC mode diagram

For the capture with the robotic arm a strategy based on a free floating platform has been selected. This approach leads to some simplifications for at GNC level and at robotic arm control level and is perfectly applicable to the target considered. Should the spin rate of PROBA2 be higher than expected, a strategy based on spin synchronisation should be selected.

With respect to the GNC equipment selection, the main driver has been the TRL level. In order to keep the mission cost as low as possible, equipment with flight heritage has been selected, provided that the performance requirements are fulfilled.

In terms of actuators, apart for the magneto-torquers and reaction wheels, the main elements are the monopropellant thrusters. Two different set of requirements are imposed on this system, one for the proximity operations where high accuracy is required and a second for the main orbital manoeuvres (target orbit acquisition and deorbit burn).

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For the proximity operations a system composed of 16 1N thrusters is selected (redundant free torque and free force manoeuvres) with a minimum impulse bit of 2.6 mNs. For the orbital manoeuvres a system composed by fur 20N thrusters (2+2 redundant) will be required to reduce the gravity losses. Furthermore this thruster will also have geometrical losses, as they will need to be depointed with respect to the deorbit burn direction so that the plume impingement on the tether is minimised.

The sensor system is composed of standard attitude control elements (star trackers, magnetometers, inertial measurement unit) plus the relative navigation sensors, composed of an optical camera and a LIDAR to support the proximity operations and provide robustness to the system to non-optimal illumination conditions on the target.

The cameras flown in the PRISMA mission (VBS from DTU Denmark and DVS from TDS Italy) would be perfect candidates for this mission. Performances would be better with the DVS, but performances achieved with the VBS should suffice once integrated with the LIDAR data. Accuracy provided will be enough from the initial range of 4km.

From the LIDAR available technologies, a flash LIDAR would be preferred to a scanning LIDAR (lower perturbances and power demands in general), but unfortunately no flash LIDAR is available in Europe at the moment. Therefore it has been opted for the RVS developed by Jena Optronik and flown in ATV. It shall be noted that a new equipment is under development (RVS3000) that could improve the performances of the mission (similar precision but lower mass, power and volume).

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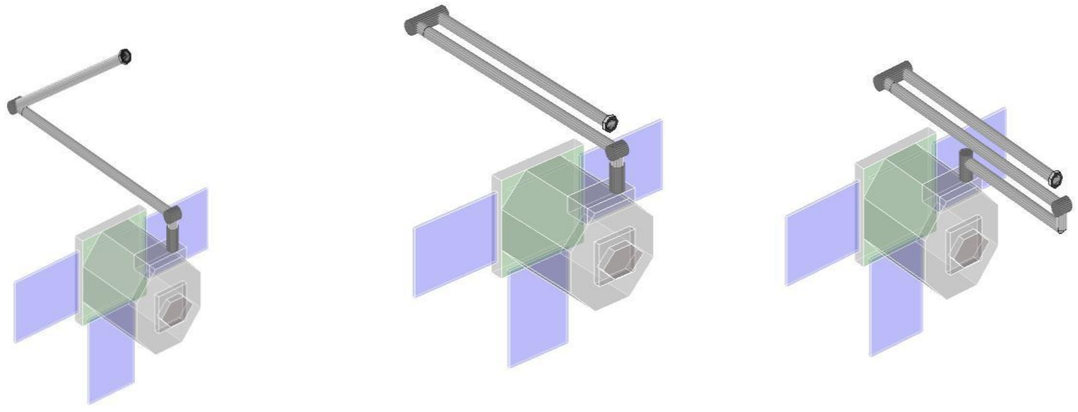
*Robotic Arm Design*

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.



**a)** **b)** **c)**

Figure 14: 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 14, 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 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 summarise the characteristics of the selected architecture.

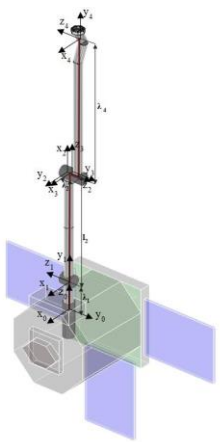


Fig. 15: Architecture „I1” and Denavit-Hartenberg coordinate systems

*Net 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

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- Tether reel mechanism

The conceptual design of the tethered-net system has been carried out aiming at determining the preliminary budgets associated with the proposed ADR scenario. Attention has been invested in analysing the scalability of such a system to larger targets. One main advantage of this technology is that it could be effectively applied to debris with various configurations and differences in characteristic dimensions. A generic tethered-net capturing system is considered to be composed of two main elements – net and tether – accompanied by the corresponding mechanisms: net folder/storage canister, bullet ejection mechanism, tether reel.

The proposed solution involves capturing the Proba 2 debris from a safety distance through the ejection of a tethered-net and by establishing a solid but flexible connection between the chaser and the target. The net deployment is performed by impulsively accelerating four corner weights (bullets) attached to the net mouth (perimeter ring). The bullets shall perform a dual role firstly 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 secondly 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 through a central vertex (knot) which has the role of absorbing/distributing the loads. During net deployment the tether is left slack in order to reduce the interference on the dynamics of the net and avoid significant reaction forces on the chaser satellite. After the debris capture is successfully performed the tether is gradually tensioned and unwound 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° for firing the deployment bullets.

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.

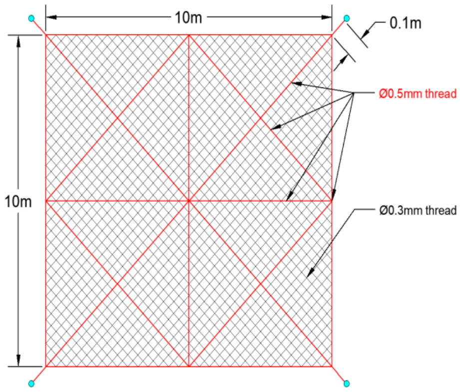

Fig. 16: Net design

- 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 a 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 9: Net 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 |

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| | |
|---|---|
| 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<br>(100% percent margin to account for knots) [m3] | 1x10-3 |
| **Total system mass (with margins) [kg]** | **15.5** |

*Ground Segment*

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.

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 enter. 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.

### 1.2.2 AnDRoID SCALABILTY 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. The below results could be outlined:

- 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 **Δ**V for orbit acquisition and target capture is limited to 120m/s, i.e. no additional experiments are carried out (propellant tank limited to 68kg).
- 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.
    - 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.
- 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.

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