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Trajectory analysis of a CubeSat mission for the inspection of an orbiting vehicle

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Abstract. The paper describes the analysis of deployment strategies and trajectories design suitable for executing the inspection of an operative spacecraft in orbit through re-usable CubeSats. Similar missions have been though indeed, and one mission recently flew from the International Space Station. However, it is important to underline that the inspection of an operative spacecraft in orbit features some peculiar characteristics which have not been demonstrated by any mission flown to date. The most critical aspects of the CubeSat inspection mission stem from safety issues and technology availability in the following areas: trajectory design and motion control of the inspector relative to the target, communications architecture, deployment and retrieval of the inspector, and observation needs. The objectives of the present study are 1) the identification of requirements applicable to the deployment of a nanosatellite from the mother-craft, which is also the subject of the inspection, and 2) the identification of solutions for the trajectories to be flown along the mission phases. The mission for the in-situ observation of Space Rider is proposed as reference case, but the conclusions are applicable to other targets such as the ISS, and they might also be useful for missions targeted at debris inspection.

Keywords: CubeSats; in-orbit inspection; mission analysis; formation flight; rendezvous and docking; deployment analysis; simulation

1. Introduction

In the last years, many ideas came up for missions targeted at the close observation of spacecraft in orbit. Cubesats and nanosatellites might well serve the purpose of inspecting orbiting spacecraft, in the form of free-flyers operating in the vicinity of the target for a certain amount of time while observing the target with suitable sensing equipment.

The present study aims at 1) identifying the design drivers and technical requirements related to the trajectories of a nanosatellite flying in the vicinity of the target vehicle, and 2) proposing possible solutions for the deployment strategies and the trajectories of the inspector relative to the target during all phases of the proximity operations.

The inspection of spacecraft in orbit at a close distance from the vehicle itself might be useful

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with respect to several applications and targets, which mainly fall into one of the following two categories:

• monitoring operative space assets for supporting their operations and/or increasing their capabilities.

• inspecting space debris for preparing to and/or directly executing active removal missions

Few studies have been conducted so far for the inspection of the ISS using free-flyers to increase responsiveness and efficacy while reducing the cost with respect to currently available inspection systems (cameras mounted on the exterior of the station or used by astronauts during ExtraVehicular Activities - EVAs, images taken by the visiting vehicles and through complex telerobotic operations (Caron and Keenan 2005, Caron and Mills 2012). A mission was already accomplished during STS-87 in 1997 with the Autonomous Extravehicular Activity Robotic Camera Sprint (AERCam Sprint) when a small sphere was flown around the payload bay of the Orbiter remotely controlled by astronauts in the Shuttle aft flight deck (Williams and Tanygin 1998). The AERCam Sprint was a prototype intended to demonstrate a light and camera system that could have been used for the remote inspection of the ISS. Unfortunately, the operational version of this system, the Mini AERCam spacecraft, was built and tested on ground but never flew (Lorenzen *et al.* 2016). In 2003, the eXperimental Small Satellite (XSS) of the Air Force Research Laboratory was launched and successfully completed its mission of imaging the Delta II upper stage. More recently, NASA successfully flew the Seeker mission to demonstrate advanced autonomous inspection capability at low cost (1.8MUSD budget) and fast delivery (14-month schedule from sketch to operations). Seeker is a 3U CubeSat that operated around Cygnus on September 2019, taking images of the vehicle and performing a set of manoeuvres (including detumbling and target tracking, station-keeping, translations) and communication tests with its paired CubeSat Kenobi (Pedrotty *et al.* 2019). Studies on this topic have been carried out also by the European Space Agency (ESA) (Nichele *et al.* 2018).

Many organisations are working on the preparation of Active Debris Removal (ADR) missions as the issue of increasing space debris population is one of the challenges of the space ecosystem. The studies conducted so far (e.g. the e.Deorbit mission as in Biesbroek *et al.* 2015) show that having detailed information about the condition of the spacecraft prior the removal mission is flown would facilitate the implementation of such a mission, and it would increase safety and enhance mission success (Liang and Ma 2011, Corpino *et al.* 2017). For this reason, ESA started the study of a precursor mission based on CubeSats, called e.Inspector, which is intended to gather information on the target for better designing the final ADR mission and/or any other mission of interest (e.g. refuelling or re-boosting) (ESA 2019). The Ecole Politecnique Federal de Lausanne is leading the CleanSpace One project with the objective of removing the 1U Swiss Cube satellite from orbit using a CubeSat (Richard-Noca *et al.* 2016). The CleanSpace One mission will also demonstrate in orbit technologies needed for other ADR missions.

NASA, ESA and other organisations and companies are also working on CubeSat missions for demonstrating capabilities related to formation flight and proximity operations, which have relevance for the purpose of inspecting vehicles in orbit (e.g. NanoAce 3U CubeSat flown in 2017, GOM-x-4B 6U CubeSat flown in 2018, Cubesat Proximity Operations Demonstrator (CPOD) to be launched in 2020 (Bowen *et al.* 2016), Rendezvous Autonomous CubeSats Experiment (RACE) Phase A study. Information about these missions and studies can be found at the URLs given in the bibliography at the end of the paper.

In conclusion, the idea of including small platforms to support the abovementioned mission objectives has been considered by several organizations, and missions have been already flown and are under development in the US and EU, with the participation of research centres and universities, and private companies. All the missions and studies conducted so far brought to light that several challenges related to proximity operations and formation flight need to be addressed to accomplish the future missions with the adequate level of safety.

The study presented in this article has been conducted for a case of interest, which is the inspection of the Space Rider (SR), next ESA's transportation vehicle with re-entry capability expected to fly in 2022 (Fedele *et al*. 2018). The focus of the paper is upon the mission analysis of the baseline concept of the Space Rider Observer Cube (SROC) mission, which requires the deployment and the retrieval of the inspector from/in the target space vehicle (i.e., Space Rider). Many considerations and results hold also for missions to other targets and/or for debris inspection, but some differences might exist, which will deserve further analysis beyond the scope of the study presented in this article. Section 2 describes briefly the mission concept, providing fundamental facts about the concept of operations and the architecture of the SROC mission. The SROC mission consists of a CubeSat deployed from the rider itself, flying around the mother-craft, taking images of the vehicle and getting back in the cargo bay for returning to Earth with Space Rider. Section 3 recalls the theoretical background applicable to proximity operations and rendezvous & docking missions. Section 4 presents the mission analysis, including the definition of the deployment strategy and the design of trajectories elaborated within a feasibility (Phase 0/A) study conducted for the European Space Agency. Section 5 concludes the paper with discussion of the results obtained so far in the perspective of future activities.

2. SROC mission concept

The Space Rider Observer Cube (SROC) mission aims at demonstrating the critical capabilities and technologies required for successfully executing an inspection mission in a safety-sensitive context. This in-orbit demonstration has the potential of opening a wide spectrum of novel applications for CubeSats in the area of inspection missions. Furthermore, the development of the advanced technologies needed for the SROC mission will have a positive impact also for pursuing other mission objectives, especially in the domains of nanosatellites for debris inspection and space exploration. the following primary mission objectives have been derived:

- To observe Space Rider with unprecedented imaging for engineering and outreach purposes
- To demonstrate critical technologies and functions related to formation flight missions, in terms of:
    - Proximity Navigation
    - Guidance and Control capabilities
    - Imaging capabilities
    - Communication architecture
- To demonstrate CubeSats in-orbit retrieval and reuse capabilities

Due to the novelty and perceived complexity of this last objective, the SROC programme might require multiple sequential missions at incremental level of complexity.

The SROC Design Reference Mission (baseline scenario) consists in a 12U CubeSat deployed from the Space Rider cargo bay with the purpose of imaging the vehicle from its vicinity (approximately at a distance of 250 to 700 metres) with a multispectral camera when flying safety ellipse trajectory for about one week. After the observation phase is completed, the CubeSat rendezvous and docks with the Rider through a mechanism developed ad-hoc (the DARM system)

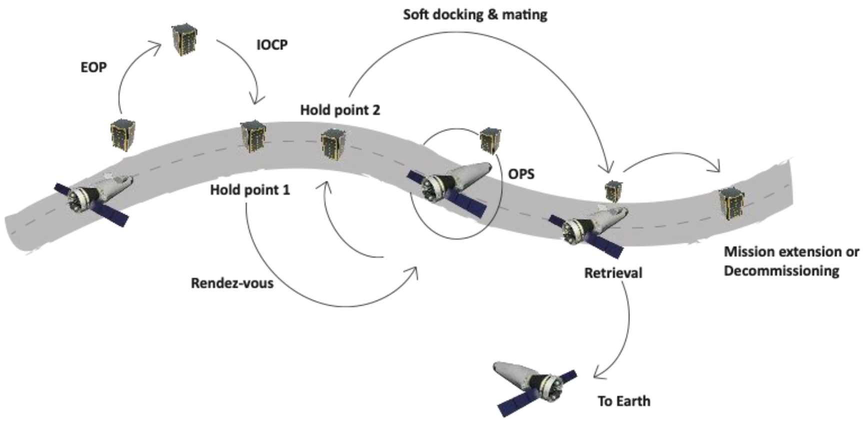


Fig. 1 SROC mission – Operative scenario

for the mission, and re-enter Earth with the mother-craft. The SROC platform features advanced CubeSat technology already available on the market and novel technology needed for the safe accomplishment of proximity operations. The preliminary development plan of the entire project has been defined, aiming at a flight opportunity in early 2022 (i.e., with the Maiden Flight of Space Rider). Should any failure occurs preventing safe docking with the rider during operations, the SROC reduced scenario is implemented.

Fig. 1 shows the design reference mission related to the SROC nominal operations. The nominal SROC ConOps involves preliminary operations, namely early operation phase (EOP) and in-orbit calibration phase (IOCP), in which verification and calibration of subsystems and payloads are carried out. To reduce the risks to the SR, two hold points have been considered, one of which also with the role of safe point in case of off-nominal events. Generally, the aim of these hold points is to guarantee a safe and reliable manoeuvre, initiating approaches only after a “go” from SROC operators. The HP#1 can be also used as virtual point in case rehearsal manoeuvres need to be accomplished. This option would be an action towards risk mitigation of the close proximity operations and docking phase. SROC could carry out the critical manoeuvres around a virtual point (HP#1) before accomplishing them in the vicinity of the rider later in the mission. The current ConOps baseline does not include any rehearsal phase, but the margin adopted for the delta-V calculation allows this option to be considered in later design stage without major impact on the system design.

3. Proximity operations

Spacecraft proximity operations is the tracking or maintenance of a desired relative separation, orientation or position between or among spacecraft. In this situation, there is not just one orbit (or location on the orbit) to be controlled, but there are many, and the typical approach consists in controlling the orbit of one of the spacecraft (the leader) and regulating the others (the followers) relatively to it. The leader is also called target or chief while the followers are also called chasers or deputies.

While the leader’s orbit is handled with an absolute reference frame, for the followers a relative frame is considered: this is a local orbital reference frame in which the motion is described

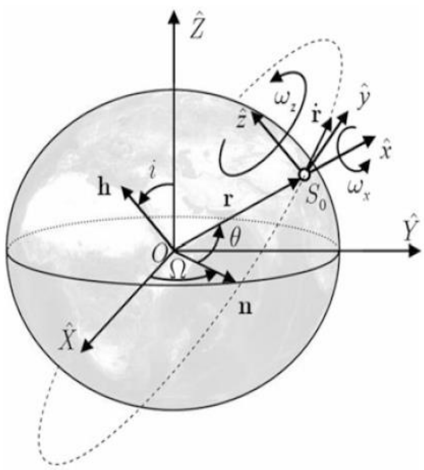


Fig. 2 ECI frame (absolute) and LVLH frame (relative)

relatively to a particular point in orbit or to another spacecraft; in this way the local orbital frame for both the leader and the follower can be defined, but the trajectories of the chaser are defined relatively to the target. This frame is usually called local-vertical/local-horizontal (LVLH) frame: the origin is the centre of mass of the spacecraft, the first axis is in the direction of the orbital velocity vector (V), the second axis is in the opposite direction of the angular momentum vector (H) of the orbit and the third one completes the triad. In rendezvous literature (Fehse 2003), these coordinates are also called Vbar, Hbar and Rbar respectively (the last one refers to the radial direction in case of a circular orbit). Fig. 2 shows the reference frames used in the analysis.

The most important condition to keep a formation is that all the spacecraft shall have the same orbital period so that, after completion of one orbit they are back in the same relative position. According to the third Kepler's law, it is known that same orbital period means same semimajor axis, which also means same specific energy. The state of a spacecraft is described by a system of non-linear differential equations; fortunately, under certain conditions, it is possible to linearize these equations. In fact, assuming that the orbit of the leader is circular and the orbit of the follower is just slightly elliptic or inclined with respect to it, the motion of the two spacecraft looks very similar and the system of equations can be simplified, obtaining the Hill-Clohessy-Wiltshire (HCW) equations.

Not considering perturbations or other forces, the HCW equations are homogeneous and assume the following form:

$$\begin{aligned} \ddot{x} + 2n\dot{z} &= 0 \\ \ddot{y} + n^2 y &= 0 \\ \ddot{z} - 2n\dot{x} - 3n^2 z &= 0 \end{aligned} \tag{1}$$

where $n = 2\pi/T$ is the mean motion, and T is the orbital period.

A first important observation is that the equations of motion in the orbital plane (x,z) are uncoupled from the equation of motion in the normal direction (y). Given an initial state, the solutions of Eqs. (1) are:

$$x(t) = -[6nz(0) + 3\dot{x}(0)]t + \left[x(0) - \frac{2\dot{z}(0)}{n}\right] + \left[6z(0) + \frac{4\dot{x}(0)}{n}\right]\sin(nt) + \frac{2\dot{z}(0)}{n}\cos(nt)$$

$$y(t) = \frac{\dot{y}(0)}{n}\sin(nt) + y(0)\cos(nt)$$

$$z(t) = \left[4z(0) + \frac{2\dot{x}(0)}{n}\right] + \frac{z(0)}{n}\sin(nt) - \left[3z(0) + \frac{2x(0)}{n}\right]\cos(nt) \tag{2}$$

$$\dot{x}(t) = -[6nz(0) + 3\dot{x}(0)] + [6z(0)n + 4\dot{x}(0)]\cos(nt) - 2\dot{z}(0)\sin(nt)$$

$$\dot{y}(t) = \dot{y}(0)\cos(nt) - y(0)n \ \sin(nt)$$

$$\dot{z}(t) = \dot{z}(0)\cos(nt) + [3z(0)n + 2\dot{x}(0)]\sin(nt)$$

Analyzing these solutions, the variation of the different components leads to the following effects:

• A displacement of the initial position in the direction of motion x results in a stationary condition: the position is different, but the relative velocity does not change;

• A displacement of the initial position in the y direction results in a harmonic oscillation along this direction with period T;

• A displacement of the initial position in the radial direction z results in a periodic oscillatory motion along both radial and along-track directions, this drift produces an elliptical relative trajectory with period T;

• A variation of the initial velocity in the along-track direction produces a periodic oscillatory motion in the radial direction and a drift along the x direction; changing the along-track velocity means changing the semimajor axis and so breaking the main requirement to keep a formation, that is why after one orbit the follower will be drifted;

• A variation of the initial velocity in the y direction has a similar effect of the variation of initial position in this direction;

• A variation of the initial velocity in the radial direction results in a periodic elliptical relative motion on the orbital plane with period T. Depending on the variation and the starting position, the semimajor axis of the ellipse will grow or become smaller, but its period will not change.

3.1 Main phases of a rendezvous mission

In general, a Rendezvous and Docking mission can be divided into four main phases: phasing, far-range rendezvous, close-range rendezvous, and mating.

The phasing is the reduction of the orbital phase angle between the chaser and target and it ends with the acquisition of the ‘entry gate’ (or ‘trajectory gate’) which shall satisfy a set of margins for position and velocity values at a certain range. The ‘gate’ (or ‘aim point’) will be on the target orbit, or very close to it, and represents the beginning of the far-range relative rendezvous operations.

The major objective of the far-range rendezvous phase is the reduction of trajectory dispersions; therefore, main tasks are the acquisition of the target orbit, the reduction of approach velocity and the synchronisation of the mission timeline. At the end of this phase, the chaser reaches a point near the target in which it can stay indefinitely at zero DeltaV cost, and this point can be a Vbar holding point, or a forward and backward drifts below or above the target orbit, or an elliptical motion with the mean orbital height equal to the target orbit.

The close-range rendezvous phase includes the closing, which is the reduction of the relative distance, and the final approach, which consists on the achievement of the mating conditions. This phase is safety critical and, because of the resulting relative trajectory, pure tangential thrust manoeuvres are rarely used while radial approaches are preferred. Radial approach starts from a Vbar hold point and precedes flying around the target. The final approach depends on the docking system and shall fulfil the requirements of attitude and relative position and velocity.

The mating includes capture, which is the prevention of escape of capture interfaces and the attenuation of shock and residual motion, and the achievement of rigid structural connection.

For the SROC mission, in which the CubeSat will be deployed by the target vehicle itself, the phasing phase does not exist. All other phases have been analysed and they are described in Section 4. It shall be highlighted that the most critical part of the SROC mission is related to the last part of the rendezvous, i.e., final approach and mating. These two phases require capabilities and technologies which have never been demonstrated in orbit between nano/micro satellites nor between nano/micro satellites and bigger spacecraft.

3.2 Orbit inspection

There are several strategies to realize inspection of objects in space using a satellite, and they can be broadly divided into three main groups:

• Close observation: the inspector satellite intercepts and rendezvous with the space object to be observed and continues station-keeping about the space object while collecting sensor data. After data collection is complete, the inspector satellite moves off.

• Far observation: the inspector satellite is placed in a fixed orbit from which it can observe the target space object. The inspector employs long-range sensors to collect data. No attempt is made to manoeuvre closer to the space object.

• Fly-by: the inspector satellite is manoeuvred to an intercept orbit that brings it close to the space object; during the close approach, the inspector satellite collects sensor data then continues along its own trajectory.

In the first two strategies, the inspector satellite keeps its position relative to the target (closer or farther respectively) for a longer time than in the third case, in which the distance from the target is variable as the inspector travels along its orbit. For the SROC mission, the first option has been chosen because: 1) compared to the long-range strategy, it gives the opportunity to stay closer to the target, while 2) compared to the fly-by solution, it allows a longer inspection time.

When the inspector is in proximity of its target, different strategies for the geometry of the proximity operations have been considered for the SROC mission:

• Vbar holding at TBD distance from SR: the inspector satellite is in the same orbit of the target with a small phase angle in true anomaly. The chaser stays stationary near the target, observing always the same portion of the target.

• Out of plane motion at TBD distance from SR with radial components: the orbit of the inspector satellite has a different inclination with respect to the orbit of the target. The chaser swings around the leader's orbital plane, increasing the observable area of the target.

• Safety ellipse with in and out of plane fly around components: the inspector satellite stays in an orbit with the same period of the target but with different eccentricity and inclination, in order to make the inspector to follow an elliptic trajectory around the target; varying the inclination of the orbit of the inspector, the inclination of the ellipses also changes, and this allows the chaser to

Table 1 Trade-off between observation strategies

Criteria	Weight	Vbar holding	Out of plane	Safety ellipse
SR coverage	0.25	0.75	2.25	2.25
Distance from SR	0.25	1.75	1.50	2.00
Response time	0.30	0.60	1.50	2.40
Complexity	0.10	0.50	0.80	0.30
Cost (delta-V)	0.10	0.40	0.80	0.60
	1.00	4.00	6.85	7.55

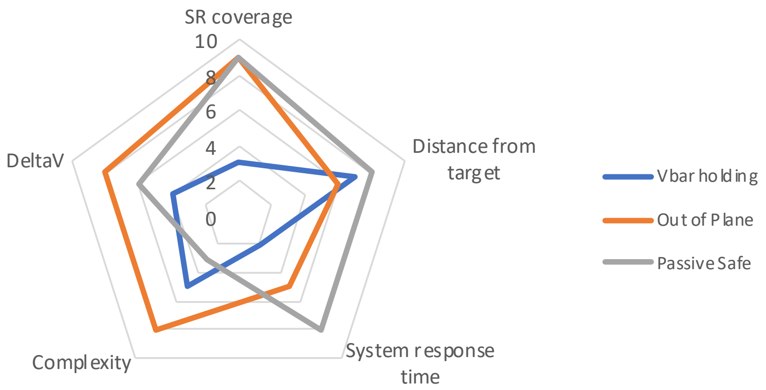


Fig. 3 Trade-off of operative strategies

access every point of the target (it maximises the portion of the target area observed by the inspector, and gives full control about the area to be observed)

A trade off study has been made for the selection of the trajectory of the operative phase; to evaluate the different strategies the following parameters have been considered:

• Coverage of the target → coverage is given as a percentage of the surface of SR imaged by SROC during observation. The higher the coverage is, the better the solution is ranked;

• Distance from the target → it refers to the distance between SR and SROC. A short distance is positive for achieving higher quality of the images, but smaller distance also implies higher risk of collision between the vehicles. A compromise between image's quality and safety shall be found;

• System response time → the response time is a measure of agility of the spacecraft and responsiveness to commands, and it is relevant for the effectiveness of Collision Avoidance Manoeuvre execution. It is measured in term of delay from the time of the command and the time of execution of the manoeuvre;

• Complexity → it refers to the complexity of implementing the trajectory, and it is measured in terms of number of manoeuvres needed to maintain the trajectory for a given duration of the observation;

• DeltaV → it is the deltaV required to maintain the selected trajectory for the duration of the observation. It is the measure of energy cost of the trajectory, and it has significant impact on system design.

Each possible strategy is assessed against the abovementioned figures of merit (Fig. 3) and ranked according to Table 1. The result shows that the V-bar solution is the worst choice while the other two options are similar; the difference between the Out of Plane strategy and the Safety Ellipse is that the latter has strong constraints in terms of attitude control during the maneuvers, but it has a better response time and allows to reach shorter distances from the target. The final choice for the SROC mission is to implement the safety ellipse trajectory, for which the detailed analysis has been carried out and it is reported in section 4.5.

4. Mission analysis

In this section the analysis of orbit geometry and trajectories for the SROC mission is described. Assumptions about Space Rider orbits and system configuration have been made in order to develop formation and rendezvous strategies, derived from public documentation available on the web.

4.1 General assumptions and definition of scenarios

In order to properly evaluate the operational orbit of SROC, the following assumptions have been made.

- Space Rider orbit is circular at 400 km of altitude:
  - Different inclinations have been considered, in particular special orbits such as Sun Synchronous (about 97 deg), quasi-equatorial (about 5 deg) and mid-inclination (about 37 deg) orbits
  - Both noon-midnight and dawn-to-dusk Sun Synchronous orbits have been studied for assessing the impact on observations
- The attitude of Space Rider is fixed with TPS towards nadir direction
- The Space Rider mass is assumed to be 4165 kg
- The following properties of the SROC spacecraft are assumed:
  - mass: 24 kg (according to a 12U form factor)
  - drag coefficient: 2.2
  - reflectivity coefficient: 1.3
  - drag area: 0.06 m^2 (no deployable solar panels)
  - SRP area: 0.06 m^2 (no deployable solar panels)
- Deployment conditions are uncertain: standard ΔV of 1.0 m/s has been considered, based on existing technology, for the baseline scenario. ΔV of 0.2 m/s has been considered for simulations of deployment conditions with customised deployment system
- Holding points are considered for go/no go commands and for possible rehearsal operations in order to increase mission safety
- Impulsive manoeuvres are assumed
- Orbit propagators and environmental models:
  - Integrator: RungeKutta89
  - Gravitational perturbation: JGM-2 at order J4
  - Atmospheric model: MSISE90
  - Solar radiation pressure: spherical models
  - Third body: Sun and Moon gravitational effects
- Perturbations:
  - Space Rider motion is considered controlled (i.e., not perturbed, except

for gravitational effects)

- Solar radiation pressure and third body perturbations influence the trajectories of SROC varying its orbital parameters, but they are much less effective than the atmospheric drag, which slows down the CubeSat. This effect is significant in hold points because varying the semimajor axis, the proximity operations condition is not satisfied
- To guarantee the correct evolution of the mission, each manoeuvre studied in the ideal case has been corrected with another manoeuvre to contrast the effect of the perturbations. This manoeuvre is the sum of the ideal and the correction manoeuvres.

The following different scenarios have been simulated, for considering all possible orbits of Space Rider and for assessing the impact on mission geometry (the starting time for each scenario is the UTC Gregorian time 1st of March 2022 11:00:00):

• Scenario 1: sun synchronous midday/midnight orbit with inclination of 97.03 deg and RAAN of 94.79 deg.

• Scenario 2: sun synchronous down/dusk orbit with inclination of 97.03 deg and RAAN of 4.79 deg

• Scenario 3: intermediate orbit with inclination of 37.00 deg and RAAN of 4.79 deg

• Scenario 4: intermediate orbit with inclination of 37.00 deg and RAAN of 94.79 deg

• Scenario 5: quasi-equatorial orbit with inclination of 5.00 deg and RAAN of 0 deg

Every scenario starts with SR true anomaly set at 0 deg and, to define the starting position of SROC and to handle the output data, a reference LVLH frame centred on SR has been created. To represent SROC inside SR cargo bay, null starting relative position and velocity have been set. Scenario 1 is assumed for the baseline design, as it is currently the most likely orbit for the SR Maiden Flight of 2022. All plots reported in the following sections refer to this scenario, but the results of simulations of all cases are commented in the text where relevant differences exist. The simulations have been carried out using the General Mission Analysis Tool (GMAT) (NASA open source software system).

4.2 Deployment analysis

The deployment of SROC from SR is executed after 2000s from the beginning of the scenario.

A preparatory simulation has been conducted to understand the most suitable direction for deploying the CubeSat. The analysis showed that, after the propagation of one orbit, the CubeSats with positive V-bar component of the deployment velocity find themselves in a negative V-bar position and vice versa; this is because a positive component increases the velocity relative to the Earth and results in a more energetic orbit, which means that the orbital period increases, while a negative component decreases it. When the main spacecraft has completed one orbit, the CubeSats with a longer period still have to complete their orbit while the ones with a shorter period already completed it. The CubeSats with a pure radial deployment velocity change their orbit without varying the orbital period, so, after one orbit, they reach the same position, with the risk of collision with the mother-craft (Fig. 4). Observing the evolution in time of the displacements along the V-bar and R-bar, it can be also observed that every CubeSat has a symmetric behaviour with respect to the CubeSat with an opposite deployment velocity.

The refined deployment analysis has been conducted for all scenarios and for different deployment velocities: 1 m/s assuming current technology, and 0.2 m/s, which would be

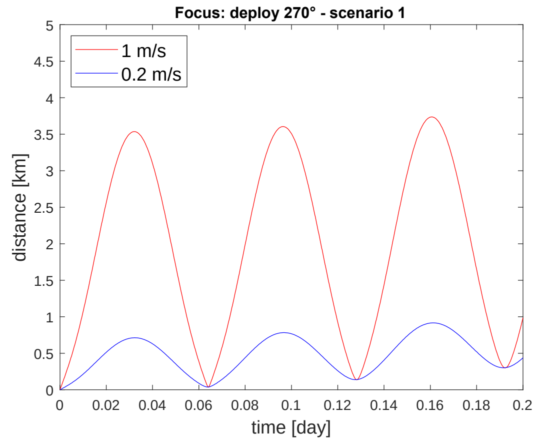


Fig. 4 SR-SROC distance during first 3 orbits after radial deployment

achievable with customised deployment system. The study aims at understanding the variation of the relative distance of SROC from Space Rider after a certain period of time considering the effect of the deployment angle (in the orbital plane, and with respect to the +Vbar direction) and the deployment velocity. Ten days of free drift is considered in the simulation, to take into account the commissioning of the satellite after deployment. Fig. 5 shows the trend of SROC-SR relative distance after deployment for the baseline scenarios (SSO midday-midnight):

• Every plot represents the distance from SR as a function of time.

• Every plot represents two deployment velocities with the same deployment direction.

• The deployment velocities are 1 m/s and 0.2 m/s.

• Deployments are assumed to be in the orbital plane.

• Plot titles identify the deployment direction; the angle is measured from positive Vbar, clockwise.

• Starting time for simulation is UTC Gregorian time March 1st 2022 11:00:00.

• The time is propagated for 10 days using a Runge-Kutta89 propagator.

The deployment analysis showed that, regardless the specific scenario:

• the distance of SROC from SR always increases with time for deployment with negative component of velocity along Vbar (deployment angles from 90 to 270 deg).

• if a positive component exists, the SROC-SR relative distance first increases then decreases. The SROC orbit becomes higher, then the effect of the perturbations makes it to decay, so the orbit is lowered. After a certain time from deployment there is a (local) maximum distance and then the spacecraft returns near SR. Then, the distance just increases because the orbit keeps being lowered by drag.

• the maximum relative distance (around 3500 km) is achieved when deploying in the -Vbar direction, the shortest is instead in the range 800-900 km for deployment in the ±Rbar direction, for deployment velocity of 1 m/s.

• the maximum relative distance is achieved when deploying in the -Vbar direction, the shortest is instead reached for deployment in the +Vbar direction, for deployment velocity of 0.2 m/s. The values of maximum and minimum distance vary depending on the scenarios (max: 1300-1500 km, min: 250-400 km).

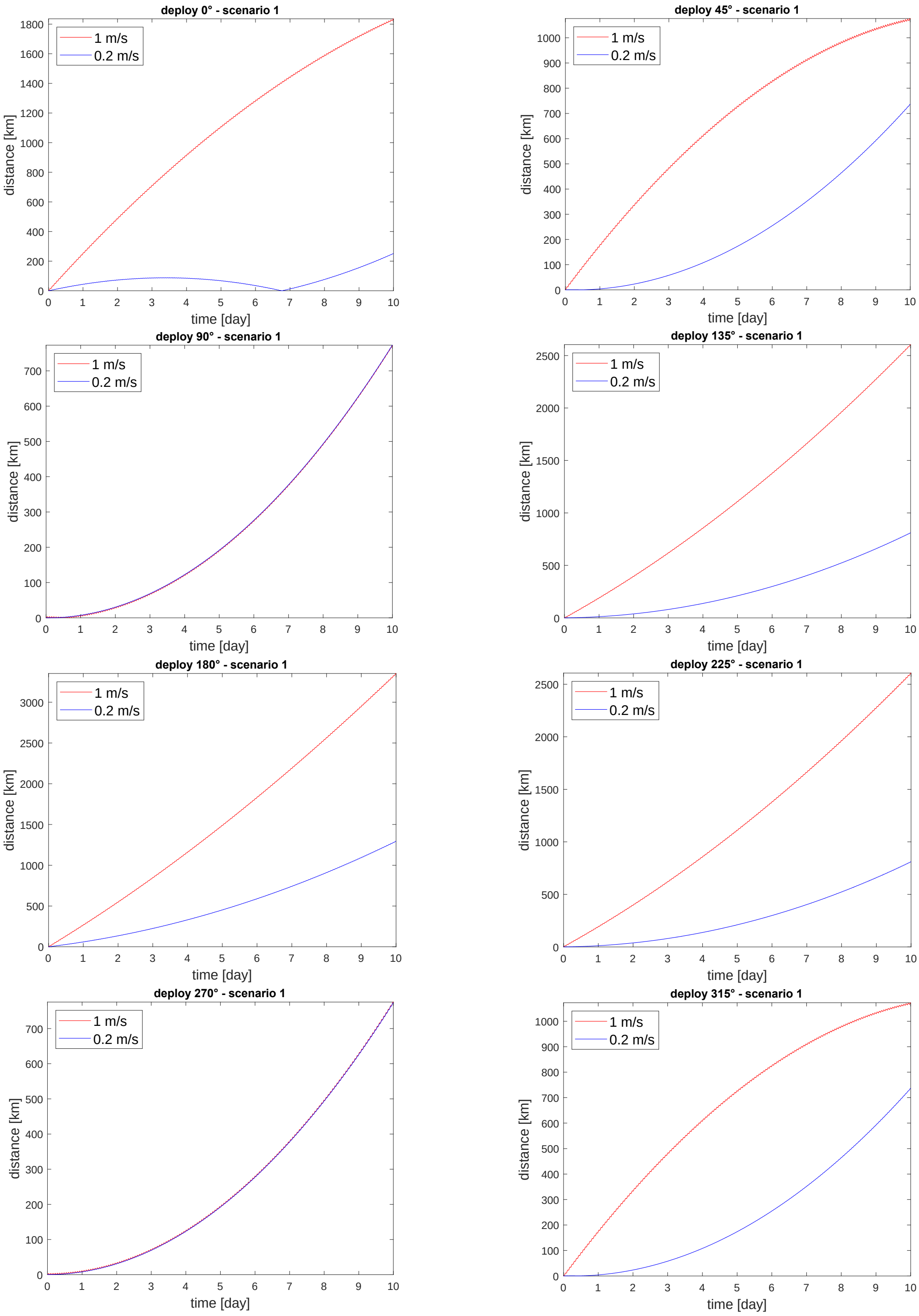
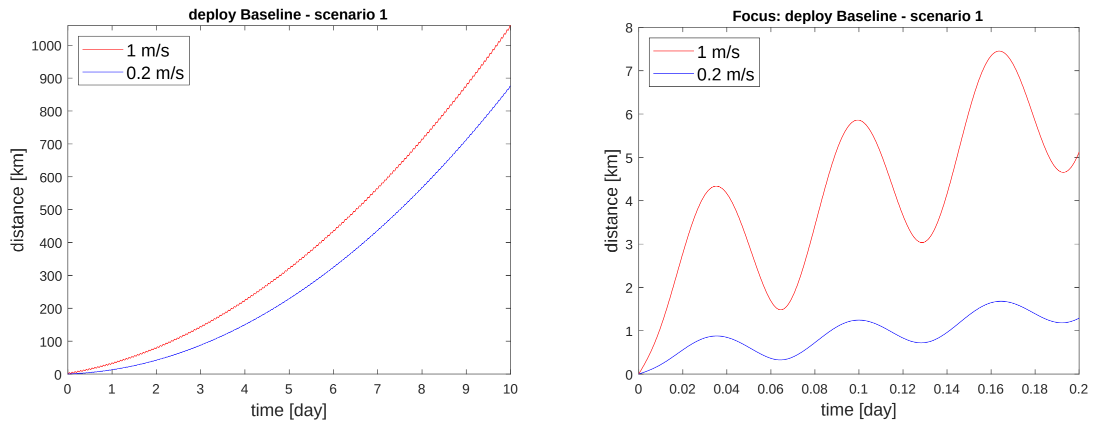


Fig. 5 Distance of SROC from SR in 10 days after deployment with different deployment velocity (Scenario 1 - SSO Midday-Midnight)



(a) SR-SROC relative distance during EOP-IOCP (b) SR-SROC relative distance during first three orbits

Fig. 6 SR-SROC relative distance after deployment and during EOP-IOCP

• when deploying along ±Rbar direction, the deployment velocity has negligible effect over the relative distance.

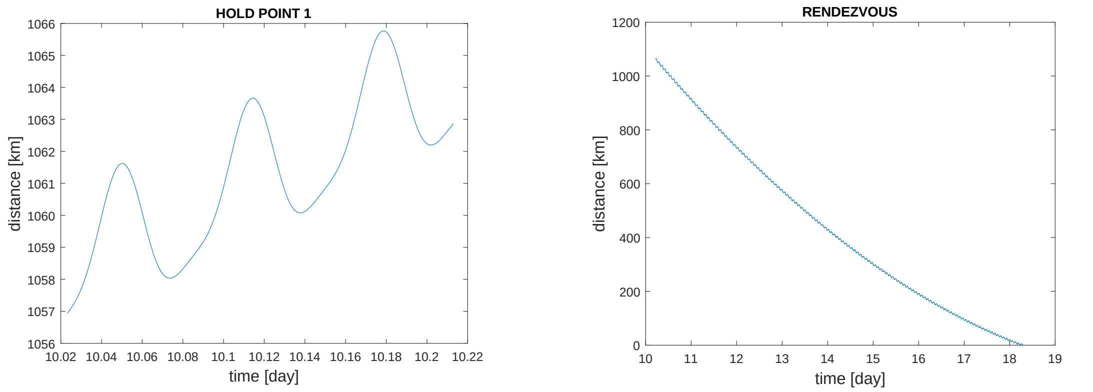
• when deploying along ±Rbar direction, it can also be noticed that the relative distance increases very slowly, as it is caused only by perturbations. In particular, in the first orbits the relative distance is very small. For this reason, a pure radial deployment is not recommended, to avoid possible collision between SROC and SR.

Another interesting result of the analysis is the different relative distance achieved after 10 days in the scenarios depending on the deployment velocity for deployment angles of 45 deg and 315 deg (±45 deg wrt +Vbar): for SSO (see Fig. 5) the difference is maximum (around 300 km), while it shrinks as the inclination decreases (less than 200 km for intermediate orbits, around 50 km for the quasi-equatorial orbit). This is ascribable to the effect of perturbations due to the combination of solar pressure and gravitation. Instead, the influence of the RAAN is negligible, as seen from comparison of simulation of scenarios 1 and 2, and 3 and 4.

From this analysis it can be noticed that, when considering a certain deployment impulse in the orbital plane, the higher the V-bar component, the higher is the drift after one orbit, but, if the impulse is purely radial, there is the risk of an impact of SROC on SR after one orbit. Therefore, the best choice is to deploy SROC with a velocity with a small V-bar component, in order to have a small drift and a certain margin of safety.

4.3 EOP-IOCP

The final value of the deployment angle has been selected over a trial and error process, guided by the duration of the free-fly phase and maximum relative distance during the Early Orbit Phase (EOP) and the In-Orbit Calibration Phase (IOCP). At the end of the design process, i.e., for the baseline, the deployment angle has been assumed to be 5 deg with respect to Rbar (anticlockwise wrt nadir vector) with velocity 1 m/s (imposed by current deployment technology), and components -0.0872 m/s along Vbar, 0 m/s in Hbar, and 0.9962 m/s along Rbar. It is important to notice that the V-bar component is negative, resulting in a decrease of the semimajor axis of



(a) SR-SROC relative distance at HP#1 (b) SR-SROC relative distance during rendezvous

Fig. 7 SR-SROC relative distance during holding and rendezvous phases (baseline scenario)

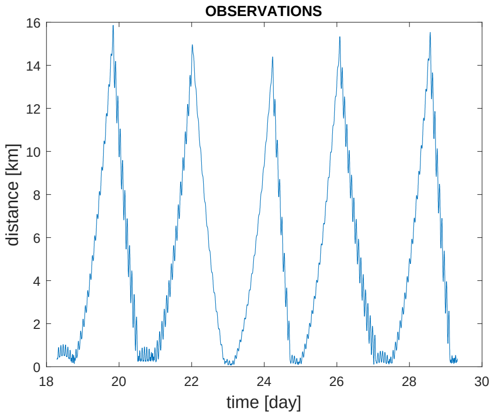
SROC's orbit; this choice has been made to increase safety in case of failure; in fact, atmospheric drag will naturally slow down the CubeSat, decreasing its orbit altitude. Further iteration of the deployment analysis will include optimisation of the velocity magnitude (constrained by the deployment mechanism) and the deployment angle (constrained by SR interface).

Fig. 6(a) shows the SROC-SR relative distance during whole EOP+IOCP, while Fig. 6(b) focuses on first 3 orbits after release. It can be seen that the effect of the deployment velocity is small even if not negligible in the long period, i.e., at the end of the phase. At deployment, the higher velocity allows the CubeSat to move away faster. However, even with a small deployment velocity (0.2 m/s), a safe drift is guaranteed.

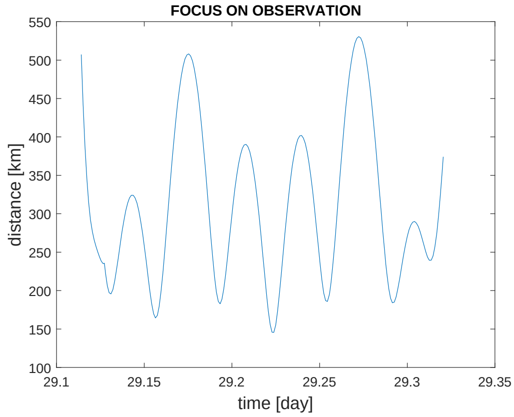
4.4 Hold point and rendezvous

A hold point consists in a stationary point relatively to the target and, for this reason, the condition of proximity operations must be fulfilled; to reach the same orbital period of SR, SROC shall execute a manoeuvre which makes its semimajor axis increase of the same quantity it was decreased with the deployment, so SROC shall execute exactly the same manoeuvre of the deployment but in the opposite direction (positive V-bar). It is important to notice that increasing the separation time, not only a drift on the V-bar, but also a drift on the R-bar appears; that is why, even if the proximity operations condition is fulfilled, the R-bar displacement does not result in a stationary point, but in an elliptical motion around the stationary point.

The scenarios continue propagating for a certain time (in these cases 16400s) to simulate the waiting time in HP#1 before starting the long-range rendezvous phase in which SROC drifts back near SR; to do this the CubeSat has to execute a manoeuvre that will increase its orbital period and this manoeuvre has been calculated imposing the achievement of a certain position at the end of the phase. It should be considered that this phase can take a shorter or longer time depending on the DeltaV of the manoeuvre at the beginning of the phase: the higher the cost the shorter is the time. For this reason, a preliminary trade off study has been conducted. At the end of the approach, SROC shall enter in formation again, reaching an elliptical motion relative to SR, therefore, a manoeuvre is required and the faster the CubeSat approaches the target, the higher is the DeltaV required to enter in formation.



(a) SR-SROC relative distance for observations



(b) SR-SROC relative distance during close observation cycles

Fig. 8 SR-SROC relative distance during the observation phase

The plots in Fig. 7 illustrate the relative distance SROC-SR as a function of mission time for the baseline scenario during the hold point #1 and rendezvous phase. It can be noticed from Fig. 7(a) that SROC follows an elliptical motion around the hold point #1, and the relative distance to SR varies of few kilometres (less than 10 km for all scenarios) due to perturbations. As far as the rendezvous phase is concerned, it can be accomplished in less than 8 days for all orbits, as shown in Fig. 7(b).

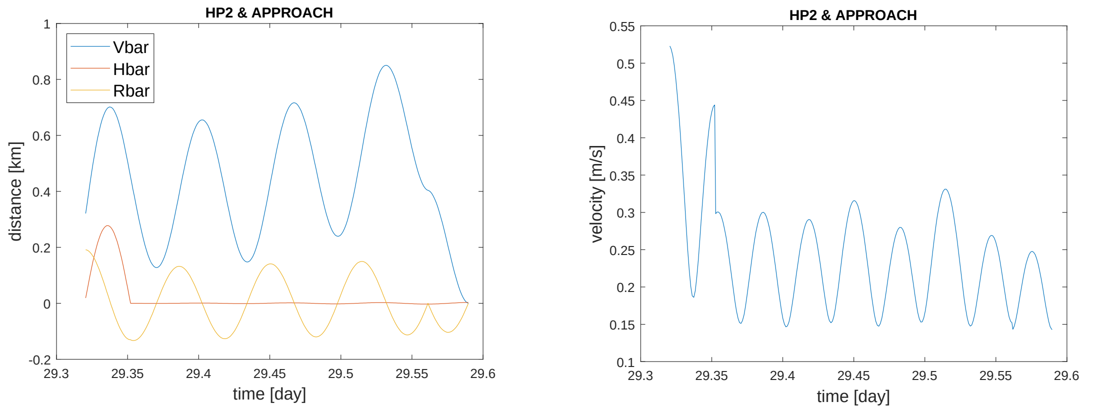
4.5 Operative phase

Fig. 8(a) illustrates the relative distance SROC-SR as a function of the mission time for the baseline scenario during the whole operative phase. In this phase, SROC stays in the vicinity of the rider for five orbits, during which the images of the vehicle are taken, then it moves away up to 20 km distance (the exact distance ranges from 16 to 23 km depending on the selected scenario). The observation cycle is repeated five times, over a total duration of 12 days. Fig. 8(b) shows the relative distance during the close-observation cycles: the distance ranges from a minimum of 145 metres up to a maximum of 535 metres (for other SR's orbits, not shown here, the range is 80-1000 metres). The distance is approximately 300 metres in average for most of the time.

4.6 Approach and docking

After the operative phase, SROC reaches Hold Point 2 (HP#2) to start the final phase of the mission; this last phase sees the CubeSat to rendezvous and dock with SR. If the rendezvous and docking is not achievable, a back-up plan is implemented exploiting SROC natural orbital decay.

To reach HP#2, the CubeSat executes a manoeuvre in the negative V-bar direction lowering its orbit and, after the desired time, enters in formation with SR; the choice of lowering the orbit is required for safety conditions, in fact, the effects of the perturbations make SROC naturally separate from SR and a lower orbit takes less time in case of atmospheric re-entry. The rendezvous phase is similar to the long-range rendezvous phase but, after entering in a safe elliptical trajectory



(a) SR-SROC relative distance during final approach (b) SR-SROC relative velocity during final approach

Fig. 9 SR-SROC relative distance and velocity during final approach for docking

in formation near the target, SROC waits for a go to execute a radial manoeuvre to achieve an intersect trajectory and perform the docking with SR.

As anticipated in Section 3.1, this is the most critical part of the whole SROC mission. The capabilities needed in terms of motion estimation and control of the inspector, driven by safety issues, require novel technology and the implementation of a failure management system (Fault Detection Isolation and Recovery system - FDIR), which are uncommon for CubeSats. Many studies and developments are ongoing to address some of these issues. Pirat *et al*. (2018) studied the problem of the precise determination of the inspector position relative to the target using visual based techniques for the upcoming RACE ESA's CubeSat mission. The attitude of the inspector also plays an important role, especially in the last phases of the approach prior to dock. The tolerance respect to angular misalignments strongly depends on the docking mechanism. Several teams are working on this aspect of the RVD problem for CubeSats, proposing different solutions: 1) based on classic probe-drogue configuration (Boesso and Francesconi 2013), 2) androgynous or semi-androgynous (Roscoe *et al.* 2018, Olivieri and Francesconi 2016), and electro-magnetic (Underwood *et al.* 2015). The FDIR capabilities shall be taken also into account when designing the final RVD trajectories. CubeSats usually implement very basic FDIR functions, which would deserve further research and development for guaranteeing adequate safety level (Franchi *et al.* 2018).

For the Close-Rendezvous and Approach, the correct strategy depends on the attitude of the target, and the docking mechanism, which are unknown at this point of the project. For this reason, for the simulations, a classic R-bar approach has been used. The soft-docking condition is achieved at a velocity of 0.15 m/s. The plots in Fig. 9 report the SR-SROC relative distance (a) and velocity (b) during the final approach phase up to the mating condition is achieved.

4.7 Atmospheric re-entry (decommissioning and disposal)

The atmospheric re-entry (natural orbit decay) has been simulated from all possible SR's orbits until SROC reaches the altitude of 100 km. The duration of these simulations varies from a minimum of 402 days for the quasi-equatorial orbit (scenario 5) to a maximum of 548 days (1.5 years) for the Sun-Synchronous orbit (scenario 1). The intermediate orbits (scenario 3 and 4) have

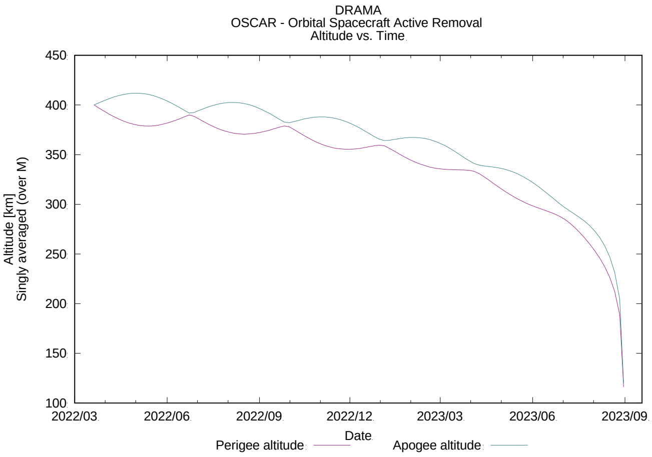


Fig. 10 De-orbit analysis

Sun-Synchronous Orbit (Scenario 1)

	Vbar	Hbar	Rbar	Execution time
Deploy [m/s]	-0,087	0,000	0,996	01/03/2022 11:33

Maneuvre	V [m/s]	H [m/s]	R [m/s]	Total [m/s]	Margin	Total with margin [m/s]	Execution time
HP1	0,617	0,000	0,000	0,617	100%	1,234	11 Mar 2022 11:33:20.000
Approach	0,901	0,000	0,000	0,901	100%	1,802	11 Mar 2022 16:06:40.000
Enter Formation	1,551	0,000	0,000	1,551	100%	3,102	19 Mar 2022 16:27:13.002
Get Operative	0,015	0,000	1,223	1,223	100%	2,445	19 Mar 2022 18:00:34.240
Re-Approach1	0,195	-0,463	0,000	0,502	100%	1,005	21 Mar 2022 06:50:00.394
Re-Operative1	-0,045	0,190	0,400	0,445	100%	0,890	21 Mar 2022 23:33:44.010
Re-Approach2	0,183	0,350	0,000	0,395	100%	0,790	23 Mar 2022 11:50:30.577
Re-Operative2	-0,033	0,100	-0,100	0,145	100%	0,290	24 Mar 2022 10:08:13.920
Re-Approach3	0,207	0,124	0,000	0,241	100%	0,483	25 Mar 2022 16:17:58.284
Re-Operative3	-0,085	0,170	0,400	0,443	100%	0,886	26 Mar 2022 04:22:32.433
Re-Approach4	0,178	-0,210	0,000	0,275	100%	0,551	27 Mar 2022 12:44:33.429
Re-Operative4	-0,010	-0,200	0,400	0,447	100%	0,895	28 Mar 2022 13:03:30.769
Re-Approach5	0,206	0,224	0,000	0,305	100%	0,609	30 Mar 2022 00:25:07.351
Re-Operative5	-0,080	-0,300	0,500	0,589	100%	1,177	30 Mar 2022 14:03:12.775
HP2	0,020	-0,314	0,027	0,315	100%	0,631	30 Mar 2022 19:26:57.418
Mating	0,000	0,000	0,294	0,294	100%	0,587	31 Mar 2022 00:28:13.359
CAM				2,500	100%	5,000	
RW Desaturation				6,000	100%	12,000	
						34,376	

Fig. 11 Delta-V budget for Scenario 1 (Sun Synchronous Orbit - Midday-Midnight)

de-orbit duration of 456 days (1.25 years). The analysis of this phase revealed that the stability of the orbit drastically decays below 300 km of altitude, as illustrated in Fig. 10 for the baseline scenario. A 12U CubeSat with mass of 24 kg and random attitude (uncontrolled tumbling satellite) has been considered for the analysis. The average cross section area is 0.08 m^2. The simulation started on March 20th 2022.

4.8 Delta-V budget

For every scenario, the actual manoeuvres executed by SROC consists in the vectoral sum between the values calculated in the ideal case and the corrections for the perturbations in LVLH reference frame. Additional values of 6 m/s for the attitude control (desaturation of reaction wheels) and 2.5 m/s for Collision Avoidance Manoeuvres are included in the delta-V budget. A margin of 100% has been considered for each manoeuvre. Main contribution is for the desaturation of the Reaction Wheels (12 m/s with margin). The maintenance/control of the observation trajectory requires approximately from 9.5 m/s to 12.9 m/s delta-V (including margin) depending on SR orbit. The delta-V required for the approach and docking manoeuvre varies from 0.5 m/s for the intermediate orbit (Scenario 4) to 1.8 m/s for the quasi-equatorial orbit (Scenario 5). It has been proven that, as expected, the delta-V has not a strong dependency from the specific scenario. Detailed values for all velocity items are given in Fig.11 or the baseline design.

The requirement regarding the delta-V has been as been finally set to 40 m/s.

A wide margin has been adopted for the following reasons:

• launch date is still uncertain, now set in March 2022. If the launch date is postponed, the atmosphere conditions might vary to a great extent due to changing solar cycle

• the baseline mission concept (Observe & Retrieve mission) has been considered for the delta-V calculation. Maintaining a higher delta-V requirement for the system design guarantees that other missions, which might require more manoeuvres for example for rehearsal operations or extended observations, can be implemented with the same system, thus reducing the need for re-design and modification should the mission objectives change at later design stage.

5. Conclusions

The close observation of orbiting objects with CubeSats can efficiently support a wide range of applications, such as the inspection of defunct satellites for preparing active debris removal missions, or the inspection of operative spacecraft (Space Stations, telecom satellites, transportation systems) for maintenance or assembly purposes. Deployment and retrieval conditions, and relative mission geometry are important aspects of this kind of missions.

In this paper, the preliminary assessment for the definition of suitable deployment conditions of a CubeSat from a mother-craft has been shown, considering two velocities and several deployment angles, for a set of possible inclination and RAAN of Low Earth Orbits. In the end, it has been drawn that deployment velocity magnitude and direction play a critical role, while the different orbits have little impact on the final result. The baseline deployment conditions have been set taking into consideration performance and safety requirements, but also the technical and programmatic feasibility within the timeframe imposed by the Space Rider project and current technology availability. Regarding the mission geometry, the preliminary selection for the formation flight strategy has been done according to a set of figures of merit and driven by mission safety. The chosen safety ellipse trajectories allow to meet excellent observation performance, even if at high complexity and delta-V cost, which is still affordable with the current system design. Considering available or high-TRL propulsion technology, the cost of the mission in terms of delta-V is pretty high. This implies that an important fraction of the total volume is allocated to the propulsion system, thus limiting the volume available for the payload. However, there is a high margin for improvement of trajectory design, to be further pursued in later design stage. The optimisation of trajectories can also exploit differential drag and lift, as presented in Horsley *et al.* 2013, which has not been considered in the preliminary analysis.

The retrieval of the spacecraft inside the mother-craft is the most critical task of the whole SROC mission, and it will deserve further in-depth analysis. In the present paper, this aspect is considered up to the achievement of safe mating conditions for soft docking. This part of the study needs to be re-iterated once the strategy and the mechanism for the retrieval will be finally decided. At the moment, Space Rider does not include the option to retrieve a payload, thus there is no interface ready to use for SROC. For this reason, the SROC concept of operations considers the option not to retrieve the CubeSat in the cargo bay should a safe mating be unachievable. In this case, the end-of-life analysis showed that the CubeSat orbit naturally decays in less than 1.5 year for the worst scenario.

The deployment and trajectory analysis demonstrated the feasibility of the SROC mission. Nevertheless, the development of the SROC mission requires the implementation of functions never tested in-orbit with CubeSats, which in turn will need novel technology in some critical areas. For this reason, the SROC spacecraft has been designed with a high level of redundancy for safety-critical tasks/items, but only passive safe trajectories have been considered for mitigating the risk of collision with the target vehicle in case CAMs are not executed correctly. The trajectories designed so far are safe, but they are not fuel-optimised. Advanced strategies and optimisation techniques can be applied to the trajectory design in later design stage to reduce the cost of the mission in terms of delta-V (Breger and How 2008). From the trajectory perspective, the key technologies for implementing the SROC mission are propulsion and Guidance, Navigation and Control (GNC) systems. Regarding propulsion, the only option for meeting the manoeuvrability requirements (imposed by the mission design and safety constraints) is using cold gas systems with full authority (6 degrees of freedom with thrusters only), high thrust (> 10 mN), and small minimum impulse bit (< 10 mNs). For the GNC, relative navigation is probably the most challenging function for which several sensors (visual cameras, GPS, lidar) will be necessary together with data fusion algorithms for extracting navigation information.

SROC represents a challenge for CubeSat missions, but has the potential to test advanced technology in relevant environment, which might be useful for many other missions to come.

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