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Mission Analysis and Trajectory Design for Space Rider Observer Cube

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Abstract

The work of this thesis focuses on two aspects of a 12U CubeSat mission: mission analysis and trajectory design and optimization. The project in question is Space Rider Observe Cube (SROC), an innovative ESA demonstration mission carried out by a CubeSat that will be deployed from Space Rider; the mission aims at demonstrating critical capabilities and technologies required to successfully execute a rendezvous and docking mission in a safety-sensitive context. Moreover, the project aims at demonstrating key technologies in the area of proximity operations, especially in the domain of in-orbit servicing, space exploration, and debris mitigation.

The trajectory design and optimization are achieved by the update and the enhancement of a Matlab code, previously created by Politecnico di Torino, which interfaces the user with an STK scenario through the STK Object model software interface. The synergized use of the two software enables the Matlab function to iterate different possible trajectory solutions on Astrogator, STK's tool for trajectory design. The relevant results and properties of these solutions are then saved by the Matlab function in dedicated structures or plotted in graphs to help the successive analysis and selection of an optimal Mission Control Sequence. This software analysis tool is used to set the optimal Mission Control Sequence for two ConOps for the SROC mission: the Observe and the Observe&Retrieve scenarios. Moreover, several deviations from the mission phases reported in the ConOps are analysed to assess how they affect the subsequent phases. For each of these possible deviations, it is then verified which respects the constraints on the total duration, total deltaV, and safety for Space Rider.

The mission analysis part of the thesis focuses on the analysis of the illumination conditions and the ground station coverage during the mission. An acceptable Line of Sight angle is required during several phases of the mission since SROC hosts different sensors operating in the visible spectrum to perform its navigation functions and to take pictures of Space Rider when in its proximity. The ground station coverage is fundamental to guarantee the downlink of the mission data and to send the send commands to SROC; during some safety-critical phases, such as the Final Approach, it is fundamental to guarantee a sufficiently long window of GS visibility.

Finally, several tools of the software DRAMA are described and used to evaluate the orbital lifetime of SROC (OSCAR tool), its re-entry survival prediction, and the associated on-ground risk for any object surviving the re-entry phase (SARA tool), and the deltaV cost to be allocated for the debris collision avoidance manoeuvres.

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Acronyms

ARES	Assessment of Risk Event Statistics
ASI	Agenzia Spaziale Italiana
ATD	Alternative Time-Down
CAM	Collision Avoidance Manoeuvres
ConOps	Concept of Operations
COTS	Commercial Off The Shelf
CPG	Centre Spatial Guyanais
CPVP	Commissioning and Performance Verification Phase
CROC	Cross Section of Complex Bodies
DD	DeltaV-Down
DOCKS	Docking System
DoE	Design of Experiment
DRAMA	Debtris Risk Assessment and Mitigation Analysis
DRP	Docking & Retrieval Phase
EMP	End of Mission Phase
EP	Encounter Point
ESA	European Space Agency
FF	free flight
GS	Ground Station
HCW	Hill-Clohessy-Whiltshire
HDRM	Hold Down & Release Mechanism
HP	Hold Point
IPA	In Plane Approach
IPLP	Integration & Pre-Launch Phase
KOZ	Keep Out Zone
LCM	Load Controller Module
LEO	Low Earth Orbit
LEOP	Launch & Early Operations Phase
LOS	Line Of Sight
MCC	Mission Control Centre
MCS	Mission Control Sequence
MPCB	Multi-Purpose Cargo Bay

MPCD	Multi-Purpose CubeSat Dispenser
OPA	Out of Plane Approach
OSCAR	Orbital SpaceCraft Active Removal
POP	Proximity Operations Phase
Qx	Quarter x (of a year)
SARA	Re-entry Survival and Risk Analysis
SR	Space Rider
SROC	Space Rider Observer Cube
STK	System Tool Kit
TD	Time-Down
UHF	Ultra-high frequency
WSE	Walking Safety Ellipses
ZeroRelVel	Zero Relative Velocity

1 Introduction

1.1 The CubeSat Standard

CubeSat is a small satellites class developed by Prof. Jordi Puig-Suari at California Polytechnic State University (Cal Poly) and Prof. Bob Twiggs at Stanford University's Space Systems Development Laboratory (SSDL) starting from 1999. It adopts a standard size and form factor, whose base unit is called 'U': as per CubeSat Design Specification [1], a 1U CubeSat is a cube with a 10 cm side and a maximum weight of 2 kg.

The adoption of this standard is due to the original goal of this project: to provide affordable access to space for the university and the science community [2]. Indeed, the standardized CubeSat platform can help reduce the cost and the duration of the development of a space mission, since it promotes a highly modular, highly integrated system where most, if not all, subsystems can be purchased as COTS products from many different suppliers. Moreover, standard dimensions enable the use of a container to store the CubeSat inside a launcher, thus minimizing flight safety issues and simplifying its accommodation.

In the last years, with the increase in the complexity and performance required by the recent CubeSat mission, bigger form factors were used, such as 6U and 12U (used by SROC). A comparison of the volume and shape of the most usual form factors is shown in Figure 1.1.

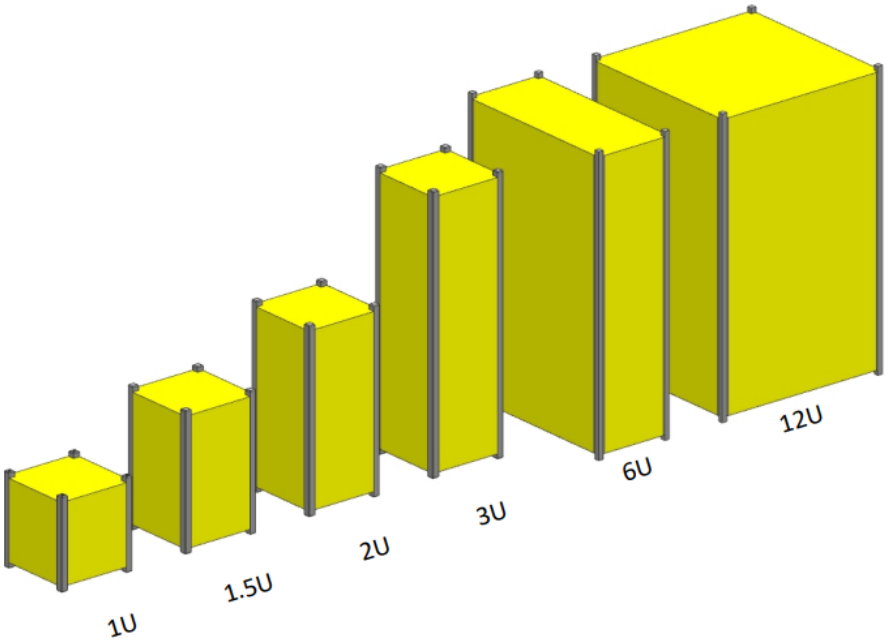


Figure 1.1: CubeSat family from 1U to 12U. Credits: The CubeSat Program, Cal Poly SLO [1].

Thanks to all these advantages and the introduction of miniaturized technologies, CubeSat have also gained increasingly more attention from government agencies and commercial groups. For example, ESA finds this technology very promising in the following applications [3]:

- Driving the drastic miniaturisation of systems, recurring to new approaches to packing and integration of subsystems
- Demonstrating, in an affordable way, new technologies and novel techniques for formation flying, proximity operations, rendezvous and docking (SROC falls within this category)
- Carrying out distributed multiple in-situ measurements
- Deploying small payload
- Augmenting solar system exploration

1.2 SROC Mission Introduction

Space Rider Observer Cube (SROC) is an ESA mission for in-orbit servicing developed by Politecnico di Torino, Tyvak International and the University of Padova.

SROC mission statement
To operate a CubeSat in LEO to demonstrate capabilities in the close-proximity operations domain in a safety-critical context, including rendezvous and docking with another operational spacecraft.

The SROC multipurpose space system is constituted by a 12U CubeSat (which will be referred to as SROC from here on out) and a deployment & retrieval system. The mission features Proximity Operations in the vicinity of Space Rider, then Docking with the mothership and re-entering Earth with it, while always ensuring the maximum safety for Space Rider.

To perform this mission, critical technologies and capabilities in the area of proximity operation will be developed and tested, thus advancing key technologies in the field of proximity operations. This in-orbit demonstration can provide a great drive forward for nanosatellites application in many fields, such as inspection missions, in-orbit servicing, space exploration and debris mitigation.

The SROC mission will advance current CubeSat technology and capabilities with respect to:

- formation flight, in terms of:
  - Proximity Navigation
  - Guidance and Control
  - Communications
  - Autonomous operations
- deployment, docking and retrieval of CubeSats:
  - Guidance, navigation and control algorithms for close approach up to docking
  - Deployment and retrieval mechanisms
  - Docking systems
- space targets observation:
  - Imaging

Since the project aims at demonstrating many in-orbit novelties in a very high safety-sensitive context, it may be imposed to implement the SROC programme through different missions with an increasingly high level of complexity and safety criticality to Space Rider. For this reason, two possible mission concepts have been defined:

- Baseline case: the Observe & Retrieve scenario is implemented. This means that SROC is deployed by Space Rider, performs inspection in its proximity, approaches it, docks with it and it is stowed inside its cargo bay to re-enter Earth. This scenario would also benefit the Space Rider programme, since it would demonstrate its capability to deploy and safely retrieve payloads.
- Reduced case: the Observe mission is implemented. It consists of a simplified ConOps where SROC is not retrieved by Space Rider. Instead, it is safely disposed into space after inspecting SR. This scenario could be used if the baseline case were considered too much complex or time-demanding. It is also possible to revert from the Baseline case to the Reduce case in case of off-nominal events which could prevent a safe docking with Space Rider.

Both the aforementioned ConOps are discussed in Section 2.3. Another possible mission concept, which was considered during the first phases of the project, involved the repetition of multiple deployments and retrievals during the same mission; however, this scenario, called Observe & Reuse, was excluded for the first SROC mission. It is noted that the work of this thesis focuses on the task proposed for the Phase B2 of the project.

1.3 Space Rider Mission Overview

Space Rider (**Space R**eusable **I**ntegrated **D**emonstrator for **E**urope **R**eturn) is an uncrewed orbital lifting body spaceplane developed by ESA to provide affordable and routine access to space [4]. The project is part of ASI's Programme for Reusable In-orbit Demonstrator in Europe (PRIDE) and has Avio and Thales Alenia Space as the main manufacturers. Its first flight is currently scheduled for Q4 2024 onboard Vega-C.

Space Rider will operate in LEO and it will be used to provide a space laboratory for many different types of payload to operate in orbit for a wide variety of applications in missions lasting for a maximum duration of two months. Space Rider's main fields are (bur are not limited to) [5]:

- Micro-gravity experimentation
- In-orbit Demonstration & Validation of technologies for exploration, orbital infrastructure servicing, Earth observation, Earth science, and Telecoms. The SROC mission falls within this category
- In-orbit Applications for Earth monitoring and satellites inspections
- Educational missions
- European pathfinder for commercial services in access and return from Space

The spacecraft is composed of two modules: the Service module (developed by Avio) and the Re-entry module (developed by Thales). The first one will provide power, thanks to the deployable solar panels, attitude control, and deorbit capability to the Re-entry module; the two modules will separate just before the atmospheric re-entry (as shown in Figure 1.2).

Figure 1.2: Space Rider mission. Credits: ESA [5]

The aerodynamic shape of the Re-entry module is a simple lifting body, which was chosen instead of operational wings or vertical fins to optimize the internal volume of the Vega rocket fairing. The 3-axis control is achieved using rear flaps. To guarantee the landing of this module, the lifting body shape will decelerate the speed below Mach 0.8, then one or two drogue parachutes will be deployed (at 15-12 km of altitude) to decrease the speed even more. Finally, a controllable gliding parachute, called parafoil, will be deployed to control the descent phase and guarantee a nearly horizontal touchdown (at approximately 35 m/s) with no wheels [6].

1.4 Thesis outline

This thesis focuses on the mission analysis and the trajectory optimization for the SROC mission. After the introduction, which aimed on giving some context about the CubeSat standard, the mission and the main actors involved in the SROC project, the following Chapters will be discussed:

- **SROC Mission Overview**: the SROC mission is further presented: the mission architecture (Section 2.1), a selection of requirements (Section 2.2) and the different concepts of operations (Section 2.3) are discussed. It is also explained why two different ConOps will be considered for this thesis: the Observe&Retrieve and the Observe scenarios. This chapter aims at giving more context to the reader, by highlighting the properties of each phase and their subphases, thus explaining the constraints or the goals which drive the successive mission analysis and trajectory optimization. Of course, it is not presented the entirety of the requirements, as well as the two concepts of operation, but only the portion of them that are interesting for the scope of this analysis. Regarding the requirements, only the following are reported:
    - ConOps Requirements (Sub-section 2.2.1);
    - Observation Requirements (Sub-section 2.2.2);
    - Orbit & Trajectory (Sub-section 2.2.3).
- **STK Scenario**: this chapter describes the reference system used for the analysis (Section 3.1) and the STK scenario (Section 3.2). This last treatment is divided into two parts: the first one is the description of the settings of the virtual models of SROC and Space Rider (SR) and the assumptions at the foundation of the orbital propagators used (Sub-section 3.2.1). The second part (Sub-section 3.2.2) describes the nominal Mission Control Sequence (MCS), which is the collection of different segments used by STK to simulate SROC's relative trajectory to SR.
- **Updated Matlab Functions**: it illustrates how the STK scenario and the Matlab functions are used together to define and optimize SROC's trajectory (Section 4.1). After describing the functioning of the software foundation for this analysis, the focus switches to two Matlab functions in particular: the IPA optimization one (Section 4.2) and HP definition one (Section 4.3). Only these two functions are detailly described since they are the ones that have been mostly changed. The other smaller changes that have been applied to the code are also listed at the beginning of the chapter.
- **Nominal Scenarios Analysis**: after describing how the different software is used to set, analyse, and optimize the mission, it is possible to illustrate the analysis performed to study the Nominal Scenarios. Two Nominal Scenarios are considered: the Observe and the Observe & Retrieve. The first one will be adopted for the first mission of SROC, while the second one will be the baseline for the successive mission. These tasks were already performed in previous cycles of the mission, however, they needed to be performed again to be updated, since they referred to an outdated orbit. The main tasks associated with this update are the following:
    - Study the ground station visibility analysis (Section 5.1) and propose different solutions to increase the duration of the longest visibility window;
    - Study both the visibility and the ground station coverage required to perform the Final Approach after the HP3 (Section 5.2);
    - Perform the Desing of Experiment (DoE) to define the best relative trajectory (specifically called Walking Safety Ellipse) during the observation phase;

  Finally, after updating the STK scenario, the Matlab functions, and the Walking Safety Ellipse, the deltaV budget and the total duration breakdown into the duration of the single mission segments (referred as "time budget") for the nominal scenarios are presented (Section 5.4).
- **Variant Scenarios Analysis**: here, one of the most important parts of the work is described: the study of the variant scenarios which could take place instead of the nominal ones. The main variant mission segments, caused by a programmatic or operational event, are analysed and discussed to assess the robustness of the SROC mission. For each mission segment considered, the origin of his divergence with the nominal scenario, its impact on the total deltaV and the total duration of the mission are presented. For a few variant scenarios which may cause relevant problems for the total duration of the mission, possible recovery manoeuvres are discussed. As it will be better explained

in the relative section, to simplify the analysis, the Hold Point 2 was used as a discontinuity point where the effects of the variant events prior to it do not impact the successive mission segments. Therefore, the Chapter is divided into the following sections:

- Variant mission segments before HP2 (Section 6.1);
- Variant mission segments after HP2 (Section 6.2), which also include the analysis of different manoeuvre to avoid or delay the encounter with Space Rider after the end of the proximity operation phase. In fact, because of the drag, SROC drifts more and more away from SR until it approaches it from behind;

Finally, all these variant scenarios are summarized in several tables to understand which ones must be labelled as off-nominal. The criteria by which a scenario is defined as off-nominal are also described (Section 6.3).

- **DRAMA Analysis**: this final chapter focuses on the different tasks performed using the ESA's software DRAMA and its following tools:
  - CROC: used to define the mass and volume properties of the spacecraft (Section 7.1);
  - ARES and Master used to evaluate the Collision Avoidance Manoeuvres (CAMs) from space debris (Section 7.2);
  - OSCAR: used to verify that the mission is compliant with ESA's space debris mitigation for agency projects [7] (Section 7.3).
  - SARA: used to verify that the whole spacecraft will burn in the atmosphere and to assess the risk of on-ground objects surviving the re-entry (Section 7.4).

2 SROC Mission Overview

2.1 SROC Mission Architecture

Summarizes the mission architecture, describing the baseline mission with some possible options which are still being analysed.

Table 2.1: SROC Mission Architecture

Mission elements	Description of baseline	Comments
Subject	Space Rider observations	For the baseline design it is required to achieve a 1 cm spatial resolution
	Close Proximity Operations demonstration	This demonstration will include the following manoeuvres: • Hold Points (HP) acquisition • Insertion into rendezvous trajectories to Space Rider • Insertion into Space Rider observation trajectories It will also determine the relative distance from Space Rider and the acquisition of Space Rider imagery
	Docking & Retrieval capability demonstration (occurs only for the Observe & Retrieve scenario)	SROC is deployed and retrieved
Payload	Visual camera	Visual camera with *ad-hoc* optics
Space Segment	1 CubeSat (SROC)	The CubeSat (Figure 2.1) has a 12U form factor and it is equipped with cold gas propulsion system and body mounted solar arrays
	1 Multi-Purpose CubeSat Dispenser (MPCD)	This deployer is used only for the baseline scenario, since it requires specific properties to guarantee the docking with SROC. In case the reduce scenario is considered, a standard 12U CubeSat deployer could be used instead
	1 Docking System (DOCKS)	DOCK is the interface between the MPCD and SROC; it includes: • Sensor suite for supporting the navigation function for relative distance minor to 1 m • Mechanisms to provide soft and hard docking of SROC to Space Rider

Orbit and constellation	Quasi-equatorial circular Low Earth Orbit at 400 km with i = 5.2 deg	
	Formation flight with respect to Space Rider	Rendezvous trajectories: • In-plane approach segment • Out-of-plane approach segment Space Rider observation: • Walking Safety Ellipses (WSE) with relative inclination change and variable geometry HP insertion and maintenance Potential Collision Avoidance Manoeuvres (CAM) to avoid space debris Docking: along the in-track axis
	Disposal orbit	Potential Collision Avoidance Manoeuvres (CAM) to avoid space debris, up to passivation of the satellite. Of course, this section only applies for the Observe scenario.
	Re-entry (uncontrolled) orbit	Natural decay within 2025-11-16; this applies only for the Observe scenario.
Communication Architecture	Store and Forward architecture	Direct link to Earth for communications purposes. Another option, although not baselined, is to use the crosslink between SROC and the MPCD to support the navigation function. *Note: a third option, described in Section 5.1, envisages the use of a GEO satellite constellation to perform data relay of SROC's data*
Ground Segment	Ground station network	Network of S-band and UHF ground stations; the compatibility with Estrack network is guaranteed
	Mission Control Centre (MCC)	SROC MCC is in Torino and will be in contact with the Space Rider MCC for specific mission phases or needs.
Operations	Mission Planning	Main driver for operations design: safety, reliability and autonomy
	Spacecraft Control	Compliant with ESA standard
	Flight Dynamics	Compliant with Space Rider operations
Launch Segment	Centre Spatial Guyanais (CSG) + Vega C + Space Rider	The launch was assumed to take plane during Q4 2024, during Space Rider maiden flight

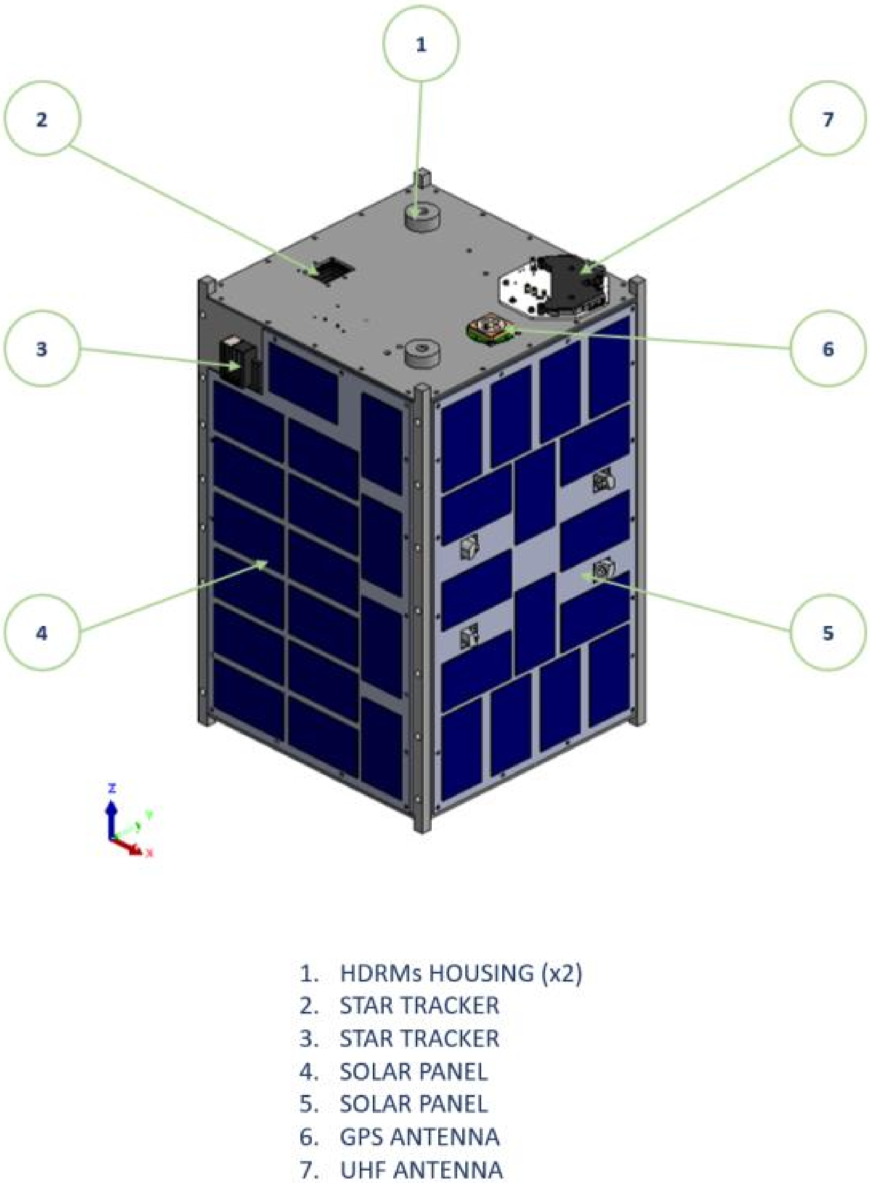


Figure 2.1: SROC external view

2.2 SROC Mission Requirements

The high-level requirements for the SROC mission were written considering:

- the Technology Traceability Matrix and the mission objectives
- the Statement of Work for the development of Phase B1 of the project
- the Mission Requirements Document made available during Phase B1
- Space Rider User Manual, and other requirements and constraints linked to the Space Rider project, such as the Payload Safety, Space Debris and Collision Avoidance Requirements
- the Space Debris Mitigation Policy for Agency Project [7]
- the trajectory design and mission analysis conducted as part of the Phase B1

Reporting the full SROC requirements specification would be unnecessary to understand the aspects of the mission concerning this thesis, which are mission analysis and trajectory design. Instead, a collection of the most relevant to the scope of this work is presented in the following format:

Requirement ID	Requirement Title
Requirement text	

2.2.1 ConOps Requirements

SROC-MIS-001	CubeSat in SR mission
The mission shall employ a CubeSat as a SR Deployable Payload (D-PL (KZ)) that can separate from Space Rider MPCB into its own free-flying mission with operations within the Space Rider Keep Out Zone	

SROC-MIS-002	Mission Scenarios
The mission shall be compatible with the mission scenarios defined as: • "Observe and Retrieve" (baseline scenario) • “Observe” (reduced scenario) *Note: the "Observe and Reuse" mission (enhanced scenario, considered in Phase 0/A) will be considered as a future development, but it is excluded as possible scenario for the first flight and it has not been studied in Phase B1*	

SROC-MIS-003	Launch date
SROC mission shall be compatible with the Space Rider's launch date on Q4 2024 (TBC). *Note: Compliance with other late launch dates shall also be guaranteed*	

SROC-MIS-006	Mission phases
The following mission phases shall be defined, listed chronologically: • Integration and Pre-Launch Phase (IPLP) • Launch and Early Operations Phase (LEOP) • Commissioning and Performance Verification Phase (CPVP) • Proximity Operations Phase (POP) • Docking and Retrieval Phase (DRP) - only for the "Observe and Retrieve Scenario" • End of Mission Phase (EMP)	

SROC-MIS-008	LEOP sub-phases
The LEOP shall be divided into the following sub-phases to support SROC release in space: • Launch • Deployment	

SROC-MIS-009	CPVP functions 1
During the CPVP, calibration and performance verification of all subsystems shall be performed	

SROC-MIS-010	CPVP functions 2
During the CPVP, compliance to performance specifications needed for safe proximity operations shall be	

demonstrated. *Note: are excluded functions that cannot be tested with the target at a certain distance (e.g. close proximity sensors performance) and/or around a virtual point instead of at the actual target (e.g. docking)*

SROC-MIS-011	**CPVP sub-phases**
The CPVP shall be divided into the following sub-phases to support SROC verification: • Commissioning • Verification	

SROC-MIS-012	**Commissioning duration**
The Commissioning phase shall take no longer than 7 (TBC) days *Note: target duration is 5 days*	

SROC-MIS-013	**POP functions**
During the POP, SROC shall perform on-orbit observations of Space Rider taken in its vicinity	

SROC-MIS-014	**POP sub-phases**
The POP shall be divided into the following sub-phases to support autonomous safe proximity operations: • Rendezvous • Observation	

SROC-MIS-015	**POP sub-phases**
During the DRP, the mission shall demonstrate in orbit CubeSat docking and retrieval capabilities	

SROC-MIS-016	**DRP sub-phases**
The DRP shall be divided into the following sub-phases to support safe docking and retrieval operations of SROC into Space Rider MPCB: • Closing • Final Approach • Mating • Retrieval	

SROC-MIS-017	**EMP functions**
The EMP shall consist of: • Moving SROC into a disposal orbit which does not interfere with Space Rider (for "Observe Scenario"); or • Retrieval and storage of SROC in the MPCD for Earth return within the Space Rider MPCB (for "Observe & Retrieve Scenario")	

SROC-MIS-018	EMP subphases
The EMP shall be divided into the following sub-phases according to the applicable mission scenario: -Observe and Retrieve scenario: • Re-entry • Post-landing • Post-flight -Observe scenario: • Disposal • Re-entry	

SROC-MIS-019	Scenario switch
In case of off-nominal performance during the "Observe & Retrieve Scenario", the mission shall be able to revert back to the "Observe Scenario" and SROC shall be decommissioned accordingly	

SROC-MIS-020	Hold points
The SROC approach trajectory towards SR shall include predefined hold-points where SROC can receive "go/no-go" commands from the SROC and SR mission control centres	

SROC-MIS-021	Collision Avoidance Manoeuvre (CAM)
SROC shall be able to perform CAMs, commanded by the SROC MCC, in case of high-risk conjunction events with spacecraft or space debris	

SROC-MIS-022	CAM capability
In case of off-nominal performance during the "Observe & Retrieve Scenario", the mission shall be able to revert back to the "Observe Scenario" and SROC shall be decommissioned accordingly	

SROC-MIS-023	ESTRACK compatibility
All aspects of the SROC mission shall be compatible with the network of ESA ground stations	

SROC-MIS-026	Space Debris Mitigation Policy
All aspects of the SROC mission shall be compliant with the Space Debris Mitigation for Agency Projects [7]	

2.2.2 Observation Requirements

SROC-MIS-040	SR observation phase coverage
The mission should achieve at least 90% (TBC) of Space Rider coverage mapping except for areas which might be permanently in shadow during the observation	

SROC-MIS-044	Observation Distance
The observation and imagery of Space Rider shall be taken from a relative distance between SROC and Space Rider > 200 (TBC) m, i.e. from outside the KOZ	

SROC-MIS-045	SR Single Inspection duration
Each observation cycle of Space Rider shall have a duration of at least 4 (TBC) hours	

SROC-MIS-046	Observation cycles
SROC shall perform at least 1 (TBC) observation cycle of Space Rider	

SROC-MIS-047	Relative velocity
The transversal component of the relative velocity between SROC spacecraft and Space Rider surface shall be less than 1.5 (TBC) m/s during the observation of Space Rider. *Note: considering an imaging system exposure time of 0.01 s.*	

2.2.3 Orbit and Trajectory Requirements

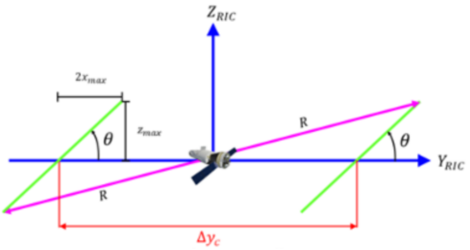
SROC-MIS-050	Operational orbit
SROC shall be compatible with an operational orbit in LEO (nominal 400 km circular) and inclination between 5-55 degrees, or SSO	

SROC-MIS-051	HP1 trajectory
SROC shall be able to acquire a trajectory around a virtual point (HP1) with null mean motion in the positive InTrack direction at a defined relative distance from Space Rider. *Note: the relative distance between HP1 and SR along the positive InTrack axis depends on the duration of the Commissioning phase. The range is approximately 330 – 1400 km*	

SROC-MIS-052	HP1 maintenance
SROC shall be able to maintain the HP1 trajectory for at least 3 (TBC) hours without manoeuvring *Note: the HP1 is useful to perform manoeuvres for demonstrating the required capabilities for proximity operations (e.g., orbit determination and control, attitude determination and control) and to decide whether to start the rendezvous or not*	

SROC-MIS-053	HP2 trajectory
SROC shall be able to acquire a hold point (HP2) at 2 - 5 (TBC) km from Space Rider along the positive InTrack axis *Note: the HP2 is useful to set up the navigation sensor suite for proximity operations and lock the target. The set up and locking can be also done during the rendezvous, i.e. without the need of HP2, but having a steady point in space is preferred from a GNC perspective*	

SROC-MIS-054	HP2 maintenance
SROC shall maintain the trajectory in the HP2 with null relative motion wrt SR for at least 3 (TBC) hours	

SROC-MIS-056	**WSE geometry**
SROC shall perform the observation of SR remaining within a passive safe and out of plane Walking Safety Ellipse (WSE) trajectory, whose geometry is defined by the following parameters:  *Note: see Section 5.3 for a more detailed description*	

SROC-MIS-057	**SROC KOZ**
SROC trajectories shall not cross the Space Rider KOZ defined as 200 (TBC) m radius sphere centred at the Space Rider vehicle centre of mass *Note: SROC is allowed to enter the KOZ during mission-specific phases (deployment, final approach and docking) agreed with Space Rider*	

SROC-MIS-058	**HP3 trajectory**
SROC shall be able to acquire one of the following holding trajectories (HP3) to reach the Radial or InTrack axis depending on the selected docking option: • InTrack docking: Holding consists of a trajectory with null relative motion wrt Space Rider < 150 (TBC) m along the positive InTrack axis • Radial docking: Holding consists of a passive-safe out-of-plane closing trajectory until reaching the radial axis/approach corridor. This trajectory maintains SROC < 150 (TBC) m mean distance from Space Rider.	

SROC-MIS-059	**HP3 maintenance**
SROC shall maintain the holding trajectory HP3 for at least 3 (TBC) hours	

SROC-MIS-060	**Maximum deltaV**
The ΔV for all SROC manoeuvres shall be less than 20 (TBC) m/s including margins	

2.3 SROC Concept of Operations

As mentioned before, three possible mission scenarios have been conceived: observe, observe & retrieve, observe & reuse. While the observe & reuse scenario was not evaluated, both the observe and observe & retrieve were analysed, although later on it was decided to implement the observe scenario for SROC's first mission. Figure 2.2 shows the main phases for both the scenarios, highlighting the fact that, until the completion of the inspection phase, the two missions are identical.

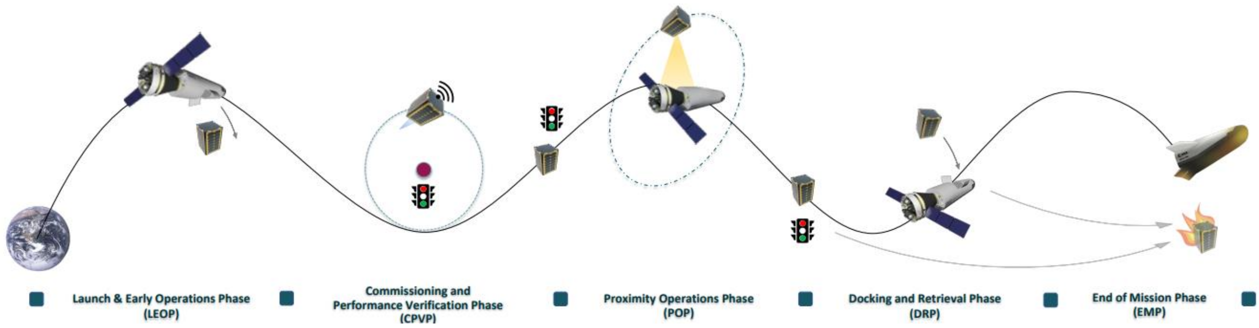


Figure 2.2: SROC mission for both the Observe and the Observe & Retrieve scenarios

In the baseline scenario, SROC will be launched inside Space Rider with Vega C (the target launch is the Space Rider Maiden Flight, which is scheduled for Q4 2024), then it will be deployed in orbit using the MPCD. Once deployed, SROC will finish the commissioning, then it will fly in formation with Space Rider and take pictures of it. Instead of performing the docking with SR, the SROC spacecraft will be decommissioned in orbit without further interaction with Space Rider. Table 2.2 and Table 2.3 describe the two ConOps and their relative mission phases and sub-phases.

Table 2.2: ConOps for Observe & Retrieve scenario

Mission phase	Mission subphases
Integration & Pre-Launch Phase (IPLP)	• Integration Phase • Pre-Launch Phase
Launch & Early Operations Phase (LEOP)	• Launch Phase • Deployment Phase
Commissioning and Performance Verification Phase (CPVP)	• Commissioning Phase • Verification Phase
Proximity Operations Phase (POP)	• Rendezvous Phase • Space Rider Observation Phase
Docking & Retrieval Phase (DRP)	• Closing Phase • Final Approach Phase • Mating Phase • Retrieval Phase
End of Mission Phase (EMP)	• Re-entry Phase • Post-landing Phase • Post-flight Phase

Table 2.3: ConOps for Observe scenario

Mission phase	Mission subphases
Integration & Pre-Launch Phase (IPLP)	• Integration Phase • Pre-Launch Phase
Launch & Early Operations Phase (LEOP)	• Launch Phase • Deployment Phase
Commissioning and Performance Verification Phase (CPVP)	• Commissioning Phase • Verification Phase
Proximity Operations Phase (POP)	• Rendezvous Phase • Space Rider Observation Phase
End of Mission Phase (EMP)	• Disposal phase • Re-entry phase

For the Observe and Retrieve scenario, the maximum duration from the deployment to the docking with Space Rider is less than 30 days (the duration of the nominal scenario in STK is 13.773 days considering also the margins). For the Observe scenario, the duration of the operation part is very similar, while it requires a maximum time of 1 year to lower its orbit and disintegrate in Earth's atmosphere. Any off-scenarios where one or more mission phases last longer than the nominal case are addressed in Chapter 6.

2.3.1 Observe & Retrieve Scenario

Table 2.4 describes with more details the Observe & Retrieve scenario phases, with their subphases, objectives, initial and final conditions.

Table 2.4: Detailed mission phases for the Observe & Retrieve scenario

Mission phase	Mission subphase	Phase description
Integration & Pre-Launch Phase (IPLP)	• Integration Phase • Pre-Launch Phase	*Objective*: SROC is ready for launch *Initial condition*: SROC/MPCD ready for integration into SR *Final condition*: SR ready for launch
Launch & Early Operations Phase (LEOP)	• Launch Phase • Deployment Phase	*Objective*: SROC is released from SR *Initial condition*: SR is launched *Final condition*: SROC is distant from SR of at least 200 m (TBC)
Commissioning and Performance Verification Phase (CPVP)	• Commissioning Phase • Verification Phase	*Objective*: SROC is commissioned and all its critical capabilities for proximity operation are verified *Initial condition*: SROC is distant from SR of at least 200 m (TBC) *Final condition*: SROC is travelling along a safe trajectory from SR (>300 km)
Proximity Operations Phase (POP)	• Rendezvous Phase • Space Rider Observation Phase	*Objective*: SR performs close observation of SR *Initial condition*: SROC is travelling along a safe trajectory from SR (>300 km) *Final condition*: SROC accomplishes the observation cycle(s)
Docking & Retrieval Phase (DRP)	• Closing Phase • Final Approach Phase • Mating Phase • Retrieval Phase	*Objective*: SROC goes back into SR's MPCB *Initial condition*: SROC accomplishes the observation cycle(s) *Final condition*: SROC is stowed into the MPCD into SR's MPCB
End of Mission Phase (EMP)	• Re-entry Phase • Post-landing Phase • Post-flight Phase	*Objective*: SROC returns to Earth inside SR's MPCB *Initial condition*: ROC is stowed into the MPCD into SR's MPCB *Final condition*: SROC and the MPCD are uninstalled from SR's MPCB and checked out

In the next sub-sections, the LEOP, CPVP and POP mission phases will be further analysed, while the LEOP, ILP and EMP will not be detailed because they do not present any manoeuvres in formation with Space Rider. The Final Approach Phase and the Mating Phase will also not be detailed and they are not analysed by this thesis, since they involve specific manoeuvre and navigation techniques that are easier to simulate and analyse in other software than STK.

2.3.1.1 Observe & Retrieve mission: Commissioning and Performance Verification Phase

This phase starts when SROC has left SR KOZ and the first signal generated by the satellite has been received by the ground segment. The KOZ is a fictitious sphere centred in the centre of Space Rider which separates the space which can be traversed by SROC and the space which cannot be used by the satellite; it is a constrained aimed at guaranteeing the safety of Space Rider, which can only be transgressed during previously accepted mission phases (such as the deployment and the Docking and retrieval phase).

The commissioning and performance verification phase consists of preparing the satellite for its nominal operations and verifying its critical capability for performing proximity operations in the actual operative environment. It is composed of two sub-phases: commissioning and performance verification. Figure 2.3 and Table 2.5 present a recap of this phase.

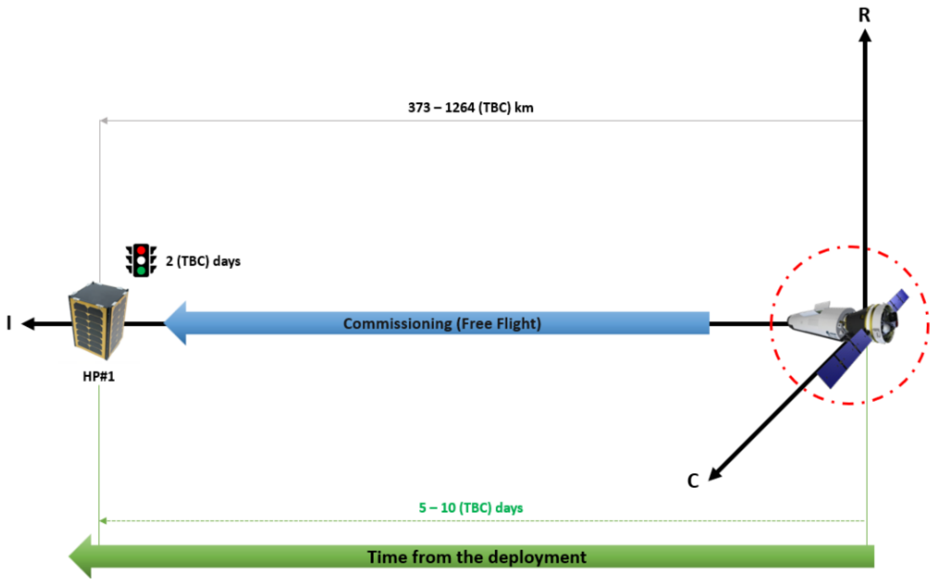


Figure 2.3: CPVP for nominal (5 days) and variant (10 days) commissioning

The commissioning phase duration was evaluated considering the commissioning of previous mission running the Tyvak bus and adding a safety margin. Since the maximum duration of the mission is relatively short, it is fundamental to reduce the maximum duration of this phase by evaluating the maximum number of ground stations able to communicate with SR; for this reason, Section 5.1 is dedicated to this analysis. For now, the nominal case considers a 5-day commissioning, while a variant longer than 10 days is considered in Section 6.1. In conclusion, the duration is yet to be confirmed, because it is necessary to confirm the following information:

- Time needed for checking and calibrating the components
- Number and duration of passes above the ground stations

During this sub-phase SROC is moving along a free flight (FF) trajectory which ends, for the nominal case, at approximately 373 km along the positive InTrack with respect to Space Rider (the definition of the RIC coordinate system can be found in Section 3.1).

The performance verification phase is fundamental to test some critical functions of SROC required to execute proximity manoeuvres; however, the performance of close proximity navigation sensors and of the payload cannot be tested this far from the target. Since during this phase Space Rider is very distant, the only capabilities that can be tested are the ones that can be performed using a virtual point. The exact sequence of operation is yet to be defined, however, the main capabilities to be tested should be the following:

- Insertion in a Hold Point (HP1) to stop the drift away from Space Rider;
- Collision Avoidance Manoeuvre: at least one artificial CAM, which would use the parameters calculated for a real CAM performed in proximity of Space Rider, should be tested to verify its correct execution;
- Insertion into the Walking Safety Ellipse: again, this manoeuvre uses the parameters that define a real insertion into a trajectory with specific geometrical features (see Section 5.3 for a more in-depth definition of the WSE) to observe Space Rider and performs it around a virtual point;
- Attitude change: different manoeuvres to control the attitude will be executed to test the system performances in terms of pointing accuracy, stability and slew rate. The exact number of manoeuvres to be tested is yet to be defined;

- Testing of Space-to-Ground and Ground-to-Space communication links and interoperations with Space Rider MCC;

Table 2.5: CPVP sub-phases - Observe & Retrieve - Nominal

Sub-phase	Characteristics	Description
Commissioning	*Objective*: to prepare SROC for nominal operations *Duration*: 5 days (target) *Environment*: LEO/FF *Relative distance*: 1-373 km	*Starting event*: SROC autobeaconing to ground (first signal acquisition) *Intermediate events*: • Commissioning procedures: RF link establishment, post deployment checkout of platform subsystems • Calibration of thruster and cameras • Test of critical equipment *Ending event*: post commissioning test passed
Verification	*Objective*: rehearsal of critical operations *Duration*: 2 days (target) *Environment*: LEO/FF *Relative distance*: ~ 373 km	*Starting event*: Command from ground to start experimental phase *Intermediate events*: • Test of HP insertion manoeuvre(s) • Test of CAM(s) • Test of attitude manoeuvre(s) • Test of WSE insertion manoeuvre(s) • Test of communication link (TBC) *Ending event*: Verification test passed

2.3.1.2 Observe & Retrieve mission: Proximity Operations Phase

This phase is one of the most critical and featuring of the SROC mission since it is when the satellite rendezvous Space Rider and then takes pictures of it, thus proving his capabilities of flying in formation with Space Rider and performing proximity operations. It is divided into two sub-phases: Rendezvous sub-phase (illustrated in Figure 2.4) and Observation sub-phase (Figure 2.5).

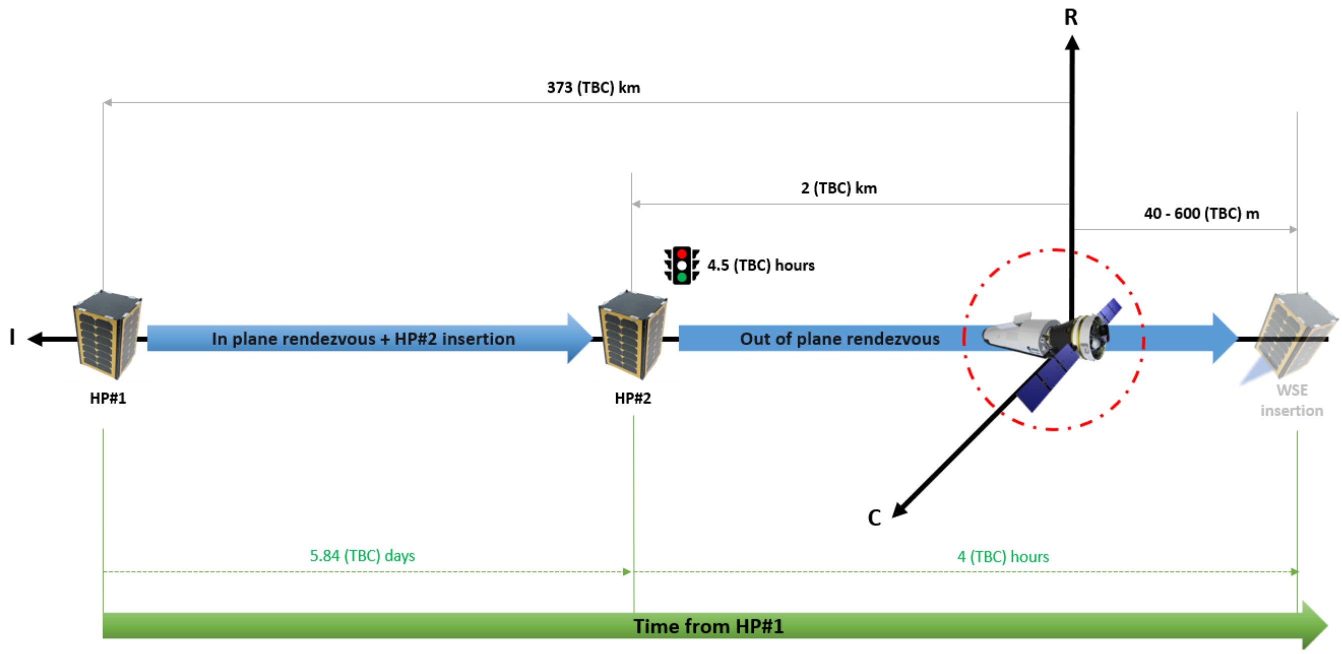


Figure 2.4: IPA + OPA Rendezvous

The rendezvous sub-phase first starts with an In-Plane Approach (IPA) where SROC uses its propulsion system to move from HP1 to a position in proximity to HP2 (from the STK simulation it is at 7 km InTrack), then SROC performs a Hold Point insertion manoeuvre to reach HP2 with the desired relative velocity. The HP2 was added to switch between the navigation sensor from far-range navigation to close-range navigation and as a go/no-go moment where SROC receives from the ground the command to proceed with the inspection phase. After the completion of the HP2 SROC performs an Out-of-Plane Approach (OPA) to move to negative InTrack and start the Observation phase.

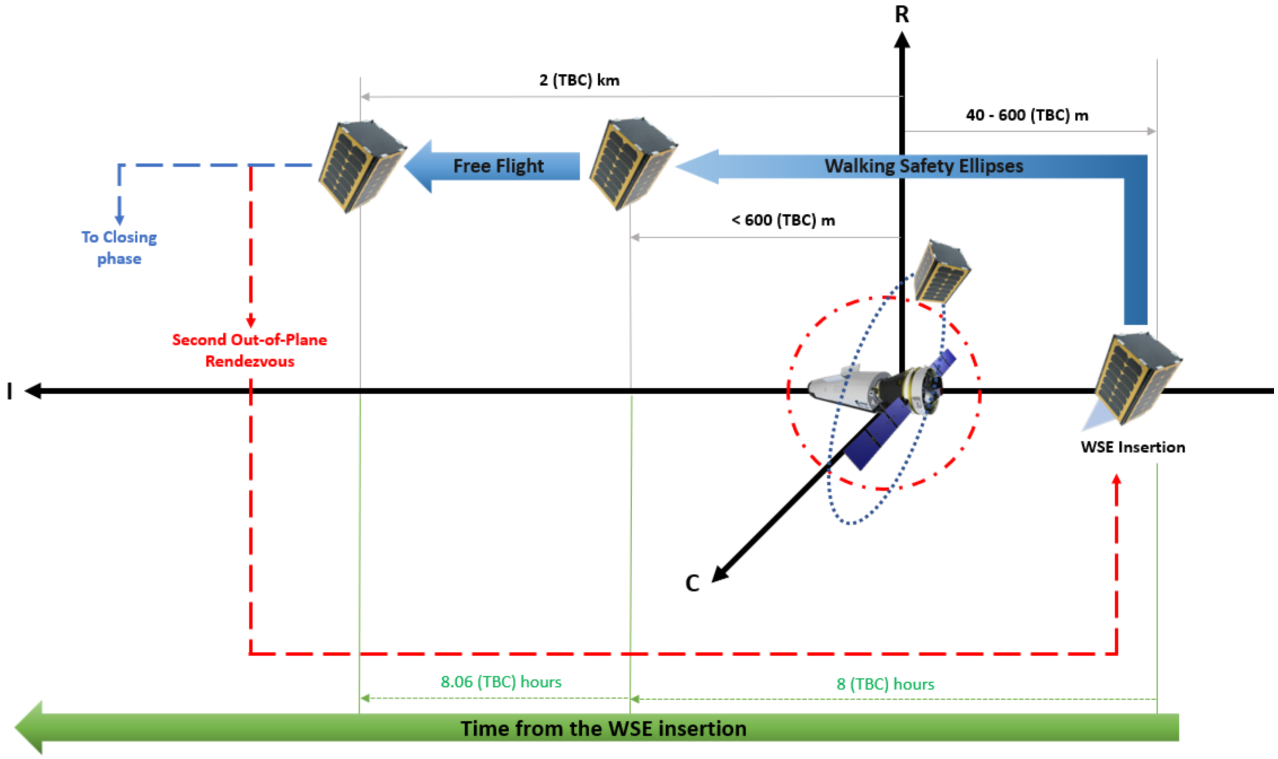


Figure 2.5: Observation sub-phase

The Observation sub-phase consists of one (or more) observation cycle(s), each composed of an inspection following a Walking Safety Ellipse and a free flight segment. To define the number of observation cycles, the following factors must be considered:

- The maximum deltaV available (requirement SROC-MIS-060) limits the maximum number of cycles
- The minimum SR surface to be covered defines a minimum number of cycles (requirement SROC-MIS-040)

From previous analyses performed on the WSE and the payload, it was proved that one observation cycle is enough to meet the requirement SROC-MIS-040. After the WSE insertion, SROC moves along a free flight relative trajectory and takes pictures of Space Rider when it is in payload range. Then after this segment, the satellite keeps moving in a free flight motion, but instead of taking pictures of Space Rider, it sends to the Ground mission data (this segment is called free flight). Defining the correct WSE was a complex task (discussed in Section 5.3) which involved considering many different parameters and constraints, such as the total access time of SROC to the Ground Stations during FF or the constraint of not surpassing the 2 km InTrack position during the FF to avoid losing the lock of SROC visual navigation sensors on SR. Here are just reported the results useful to the description of the ConOps:

- WSE observation duration: 8 hr
- free flight duration: 8.06 hr
- free flight final InTrack position: 2 km

After this phase, the satellite will either perform a second observation cycle (variant scenario) or pass to the successive phase (DRP).

Table 2.6: POP sub-phases - Observe & Retrieve - Nominal

Sub-phase	Characteristics	Description
Rendezvous	*Objective*: to reduce the relative distance from SR, reaching a precise position relative to it *Duration*: 5.76 (TBC) days *Environment*: LEO/FF *Relative distance*: 373 km to hundred meters (>200 m)	*Starting event*: SROC receives the command to start the rendezvous from the Ground *Intermediate events*: • IPA: trajectory in the Radial-InTrack plane to reach a specific position along positive InTrack (7 km) • HP2 insertion to move to the desired final position (2 km along positive InTrack) with the desired final relative velocity (null relative velocity) • HP2 maintenance: maintain of a hold point • OPA: trajectory out of the Radial-InTrack plane to reach a specific position along negative InTrack (<600 m) *Ending event*: Acquisition of the initial condition to start the observation sub-phase
Observation	*Objective*: insertion into the WSE to observe SR *Duration*: 16.06 (TBC) hours *Environment*: LEO/FF *Relative distance*: > 200 m to 2 km	*Starting event*: Command from ground to start the observation orbit *Intermediate events*: • WSE insertion manoeuvre • Observation of SR during WSE • Free Flight (FF) • OPA to start another Observation cycle (off-nominal scenario) • Manoeuvres to correct the trajectory if needed *Ending event*: Completion of the observation cycle(s)

2.3.2 Observe Scenario

For the observe scenario, the ConOps are identical to the Observe & Retrieve scenario until the end of the Proximity Operations Phase. After that, there is no Docking & Retrieval Phase, but a different End of Mission Phase. As mentioned before, this scenario was created in case the Observe & Retrieve scenario is considered too complex for the first mission; moreover, the Observe & Retrieve scenario was designed to revert back to the Observe scenario in case any off-nominal conditions occur in orbit. This switch can occur until the final approach is completed.

This scenario required to analyse its EMP phase (described in Sub-section 6.2.2) since it is fundamental to ensure that no encounter points with Space Rider will happen and that the spacecraft will still be able to perform CAMs to avoid hitting space debris and provide a more sustainable mission for the space environment. For this reason, SROC will not be passivated immediately after the proximity operation completions.

Table 2.7: Detailed mission phases for the Observe scenario

Mission phase	Mission subphase	Phase description
Integration & Pre-Launch Phase (IPLP)	• Integration Phase • Pre-Launch Phase	*Objective*: SROC is ready for launch *Initial condition*: SROC/MPCD ready for integration into SR *Final condition*: SR ready for launch
Launch & Early Operations Phase (LEOP)	• Launch Phase • Deployment Phase	*Objective*: SROC is released from SR *Initial condition*: SR is launched *Final condition*: SROC is distant from SR of at least 200 m (TBC)
Commissioning and Performance Verification Phase (CPVP)	• Commissioning Phase • Verification Phase	*Objective*: SROC is commissioned and all its critical capabilities for proximity operation are verified *Initial condition*: SROC is distant from SR of at least 200 m (TBC) *Final condition*: SROC is travelling along a safe trajectory from SR (>300 km)
Proximity Operations Phase (POP)	• Rendezvous Phase • Space Rider Observation Phase	*Objective*: SR performs close observation of SR *Initial condition*: SROC is travelling along a safe trajectory from SR (>300 km) *Final condition*: SROC accomplishes the observation cycle(s)
End of Mission Phase (EMP)	• Disposal Phase • Re-entry Phase	*Objective*: SROC is disposed according to ESA Space Debris Mitigation *Initial condition*: SROC accomplishes the observation cycle(s) *Final condition*: SROC burned in Earth atmosphere

3 STK Scenario

In this chapter, the settings of the STK scenario and the mission control sequence are described. Before diving into the description of the software functions, it is also given some context regarding the coordinate reference systems and the assumption behind the trajectory analysis.

3.1 Proximity operations

Spacecraft proximity operations are the maintenance or the targeting of a desired relative position, orientation and/or velocity between at least two satellites. This is a complex kind of analysis that requires the definition and study of the orbits of all the satellites involved: to simplify it and better understand the relative state of one satellite to another, a Satellite Coordinate System is used [12]. These systems have the origin in the centre of mass of a "leader" satellite and move with it; this means that the motion of the other satellite, called "follower", is evaluated with respect to the leader. Other treaties also use the term "chief" for the leader and the term "deputy" for follower.

One of these systems, called RTN (Radial Transverse Normal) or LVLH (Local Vertical, Local Horizon) is centred in the centre of mass of the "leader" satellite, moves with it and has the following axes:

- R axis points out of from the satellite along the geocentric radius vector; the Radial displacement is the one evaluated along this axis
- N axis is normal to the orbital plane; the CrossTrack displacements are the ones evaluated along this axis
- T axis is normal to the position vector and positive in the direction of the velocity vector; the AlongTrack displacement is the one evaluated along this axis

If the orbit is circular, the S axis is aligned to the velocity vector: this frame, called RIC (Radial InTrack CrossTrack) is the reference system that will be used from here on out (see Figure 3.1 for a comparison with RTN). This reference system is basically the same as the RTN one, with the CrossTrack axis coinciding with the T axis and with the InTrack axis coinciding with the S axis and parallel to the velocity vector.

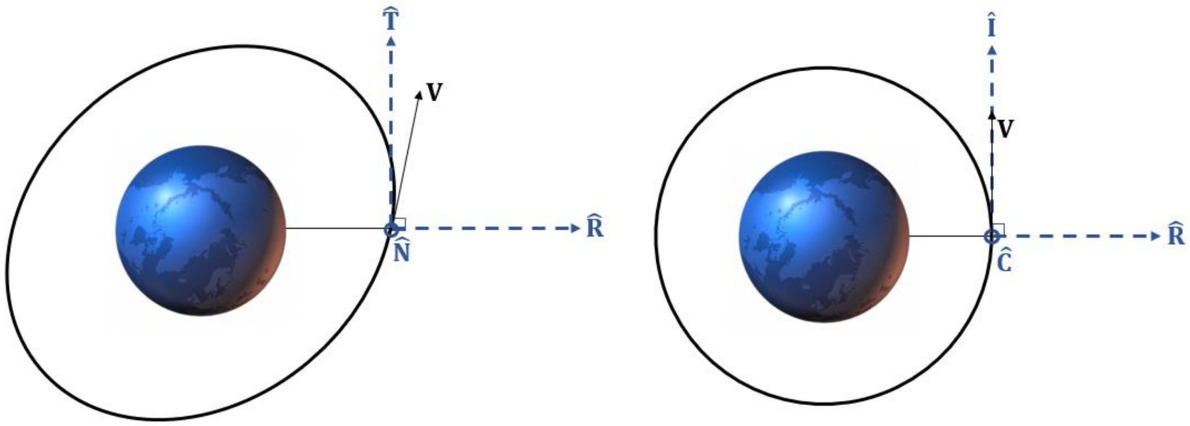


Figure 3.1: RTN (left) for an elliptical orbit and RIC (right) for a circular orbit

Of course, this is an approximation, since Space Rider's orbit is not perfectly circular because of the effect of the non-sphericity of the Earth (the maximum degree and order of the gravity model used for the propagator are described in Sub-section 3.2.1). However, as shown in Figure 3.2, the maximum angle between Space Rider velocity vector and the InTrack axis of the RIC reference system is approximately 0.08 degrees at most, so identifying the distance along the I axis as InTrack generates an almost negligible error.

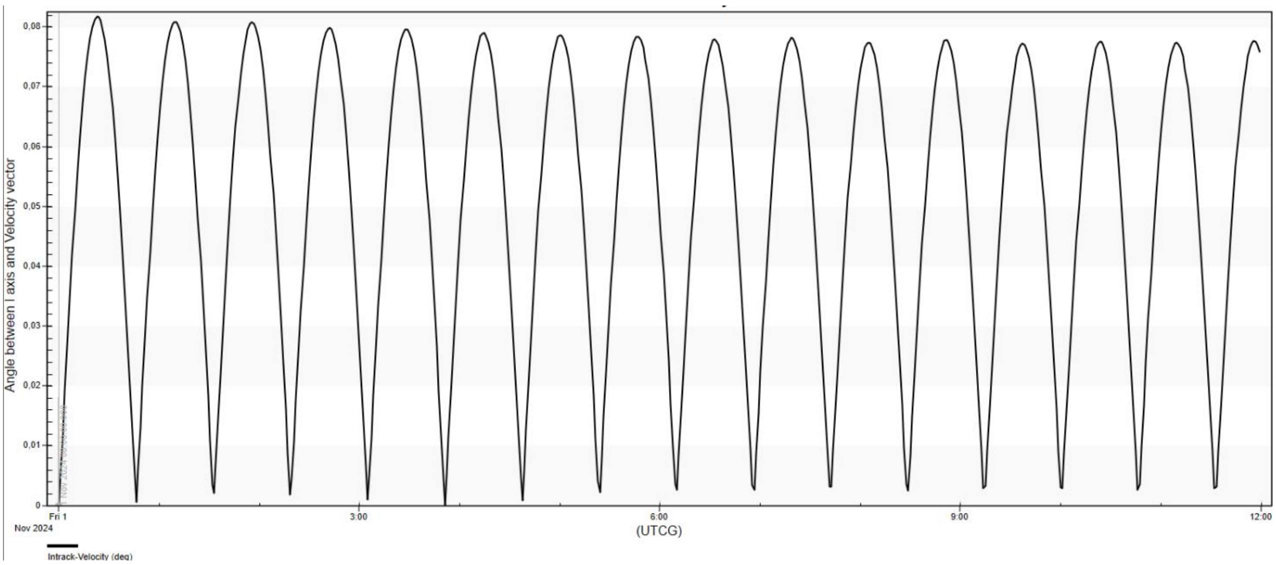


Figure 3.2: Angle between the InTrack axis of the RIC coordinate system and the velocity vector for the first 12 hours of the simulation

Instead, if a propagator which does not consider any of the effects of the non-sphericity of the Earth is used, the angle between this vector is negligible (Figure 3.3). Both this graph and the previous one were obtained using STK's analysis workbench tool.

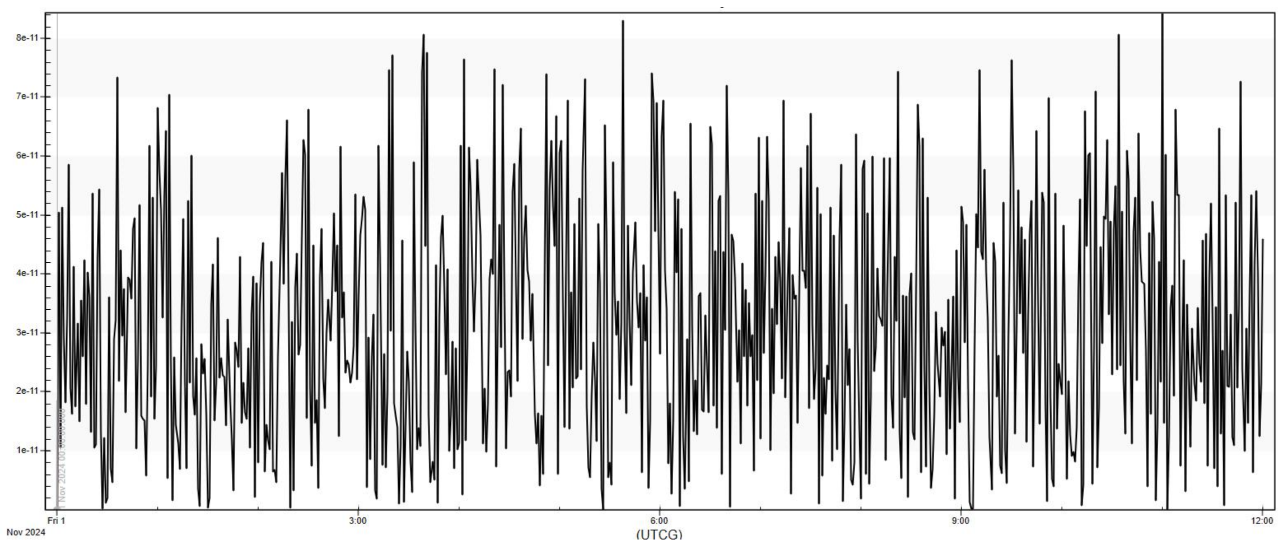


Figure 3.3: Angle between the InTrack axis of the RIC coordinate system and the velocity vector for the first 12 hours of the simulation (no effects of Earth non-sphericity)

The motion of one satellite with respect to another one is described by a system of non-linear differential equations, that, with specific conditions, can be linearized and solved more easily. The simplified Hill-Clohessy-Whiltshire (HCW) equations are obtained by making the following assumptions [11]:

1. Small relative position vector magnitude compared to the chief position vector magnitude
2. Pure Keplerian motion of both the leader and the follower
3. Leader spacecraft is on a circular orbit

The assumption of the circular orbit for the leader spacecraft has already been discussed; the first assumption is respected since, in the nominal scenario, the furthest relative distance is 373 km, which is one order of magnitude less than Space Rider position vector magnitude (6778.1 km). The pure Keplerian motion assumption is a rough approximation since the STK scenario considers the effects of external forces such as the atmospheric drag and the solar radiation pressure. Moreover, even the effects of a continuous thrust cannot be evaluated under this assumption, since F_{thrust} must be null. However, the effect of an impulsive manoeuvre can still be assessed, just by using its resulting velocity as the initial condition to restart the analysis. Under this assumption the HCW equations are homogeneous:

$$\ddot{x} + 2n\dot{z} = 0$$
$$\ddot{y} + n^2 y = 0$$
$$\ddot{z} - 2n\dot{x} - 3n^2 z = 0$$

Where $n = \frac{2*\pi}{T}$ and T is the orbital period of the leader satellite. The solutions of this equation are the following:

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

Where $x(0), y(0), z(0), \dot{x}(0), \dot{y}(0)$ and $\dot{z}(0)$ are respectively the positions and velocities along the Radial, InTrack and CrossTrack directions. Although the inconsistency with the assumptions at the base of the simplified Hill-Clohessy-Whiltshire equations, this model was still used to perform a small calculation, that is evaluate the initial conditions to perform the WSE, since it could still give a solid guess of SROC motion during this sub-phase.

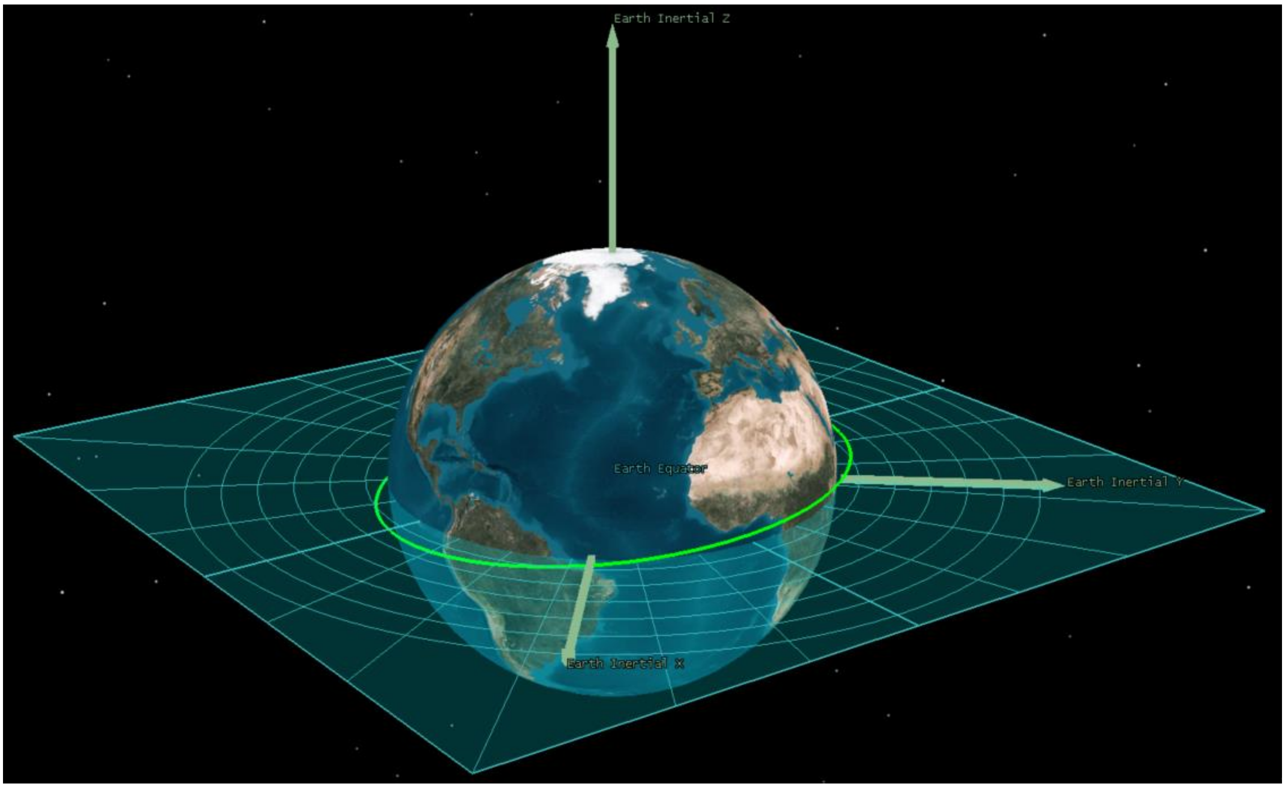


Figure 3.4: Space Rider's orbit and reference system

While SROC is defined according to the RIC reference system, Space Rider's reference orbit is J2000, one of the most used Earth Centred Inertial reference system. Its axes (shown in Figure 3.4) are defined as follows:

- X axis: it points from the centre of the Earth to the vernal equinox;
- Y axis: it is defined by the cross-product between Z and X;

- Z axis: is normal to the mean equator of date at epoch J2000 (1 January 2000 at 12:00 Universal Time), which is approximately Earth's spin axis orientation at that epoch;

Another Satellite Coordinate System which has been used during the analysis is the VNC (Velocity Normal Co-Normal) reference system [13]. It is centred in the spacecraft's centre of mass, and the axes point in the following direction:

- V: along the Velocity vector
- N: along the orbit normal
- C: completes the orthogonal triad ($\hat{Z} = \hat{V} \times \hat{N}$)

This system is used to define the Thrust Vector components in STK.

3.2 STK Scenario

Now that the coordinate reference system and the analysis process have been defined, this section focuses on the STK scenario, its settings, and its mission control sequence. Regarding the mission control sequence, for now, only the nominal Observe & Retrieve and Observe scenarios will be considered, while the variant analysis will be presented in Chapter 6.

3.2.1 Scenario Settings

The following Space Rider orbit for the Baseline scenario was assumed:

Table 3.1: Space Rider Orbital Parameters for the Baseline Scenario

Orbital Parameter	Value
Apoapsis Altitude	400 km
Eccentricity	0
Inclination	6.2 deg
Right Ascension of the Ascending node (RAAN)	0 deg
Angle of Perigee	0 deg
True Anomaly	0 deg

Since the launch date of Space Rider maiden flight is still not precisely defined, but just refers to a generic Q4 2024, it was assumed the beginning of the SROC mission on 01 November 2024 (during the middle month of the fourth quarter). The radius of the KOZ was also updated from 150 m to 200 m (Figure 3.5) to comply with the new minimum distance required by the Space Rider project.

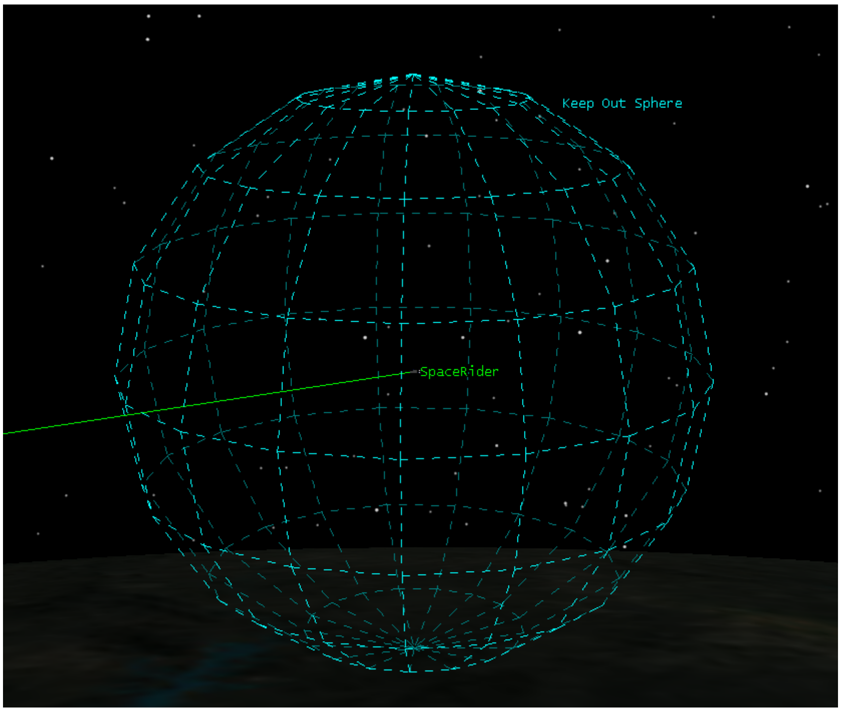


Figure 3.5: The perimeter of Space Rider's Keep Out Zone in STK

Space Rider's orbit and its properties, as well as SROC's, were defined using the Astrogator Tool of STK [10]. This capability enables specialized analysis for orbit manoeuvring and trajectory design and calculates the ephemeris of the selected satellite(s) following the Mission Control Sequence (MCS). This sequence is composed of different mission segments, which are dived into two categories: those that generate ephemeris (for example a manoeuvre or a propagation segment) and those that affect the execution of the MCS. Among this last category, there are two fundamental blocks:

- Target sequence: it defines manoeuvres and propagations in terms of the desired goal. What the control sequence does is run the segments nested within it and apply the profiles to the run according to its configuration. In this analysis two types of profiles have been used: search (which defines a goal and changes the selected variables to achieve them) and segment configuration (which is used to change the configuration of a specified segment inside the target sequence). An example of a target sequence is shown in Figure 3.6: the target sequence "IPA Rendezvous" uses a search profile (specifically a differential corrector) that evaluates the necessary value for the control parameter (in this case thrust during the "IPA Rendezvous Man" manoeuvre) to get the equality constraint (7 km along InTrack at the end of propagation segment "PropToSR").

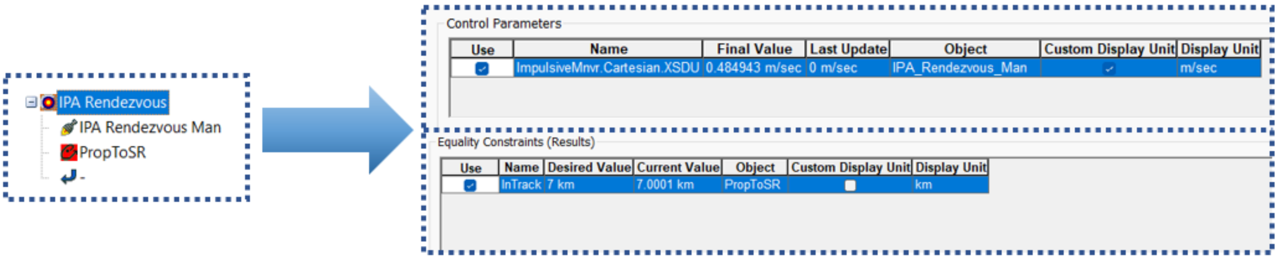


Figure 3.6: Example of a target sequence

- Sequence: this structural element organizes the segments nested within and defines the nature of the results to pass on to the next segment of the MCS. It also allows to set number of times that the sequence will run.

To define the properties of the spacecraft (e.g.: its mass, drag area) and its initial orbit, the segment Initial State is used. For Space Rider, the following properties were set:

- Dry Mass: 4165 km;
- Attitude fixed with TPS towards nadir direction;
- Propagator: Space Rider motion is assumed to be controlled, therefore only the effect of the gravitation force is considered (JGM2 model with maximum order and degree equal to 4). The Joint Gravity Model (JGM) version 2 is a model that describes Earth's gravity field up to degree and order 70. It was developed by Goddard Space Flight Centre in cooperation with American universities and private companies [14];

For SROC the following properties were set:

- Dry mass: 24 km
- Drag coefficient: 2.2
- Drag area: 0.06 m^2
- Solar Radiation Pressure coefficient: 1.3
- Solar Radiation Pressure area: 0.06 m^2
- Propagator: it uses the following disturbances:
    - Gravitational Force: JGM2 with maximum degree and order equal to 4.
    - Lunar Third Body Force
    - Solar Third Body Force
    - Drag Model: drag with MSISE 1990 Atmospheric Density Model. It uses fixed values for the solar flux and geomagnetic effects: Daily $F_{10.7}$ = 150, Average $F_{10.7}$=150 and K_p=3 (they are STK's default values for the model). The solar radio flux $F_{10.7}$ is an indicator of the solar activity which correlates well with the number of sunspots and UV and visible solar irradiance records. When this activity changes, the thermospheric density changes too, thus varying the atmospheric drag: the higher the solar activity, the higher the atmospheric drag [15]. The K_p index is used to characterize the magnitude of geomagnetic storms and disturbances in Earth's magnetic field; geomagnetic storms can produce large short-term increases in upper atmosphere temperature and density, increasing drag on satellites and changing their orbits.

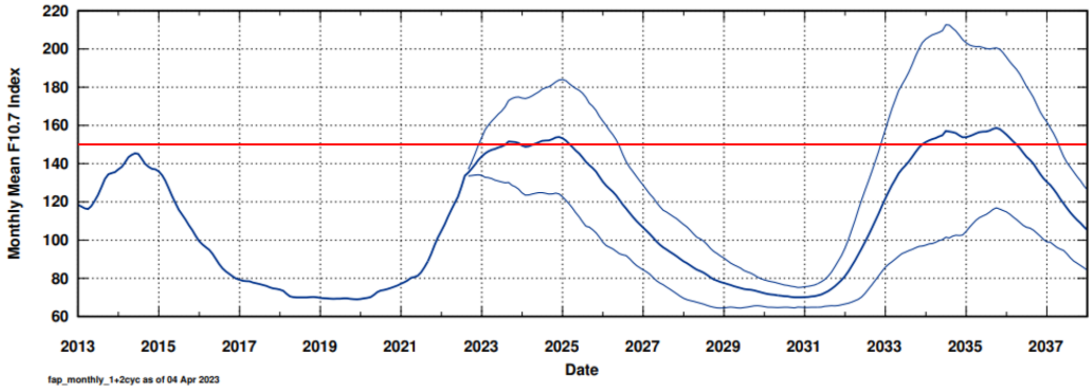


Figure 3.7: ESA prediction for the monthly mean F10.7 index [16]

Figure 3.7 and Figure 3.8 show the prediction by ESA for the monthly mean $F_{10.7}$ index and the A_p index (it is another index comparable to K_p but can be easily converted to it using an online converter [17]). The middle and darker line in both graphs represents the 50 percentile of the prediction and it was used to verify the reliability of STK's default values. Using $F_{10.7} = 150$ for both the daily and average values is very consistent with the prediction by ESA, while the value for the K_p is a bit higher than the average predicted by ESA (3 instead of 2.75), however, it was still considered reliable enough.

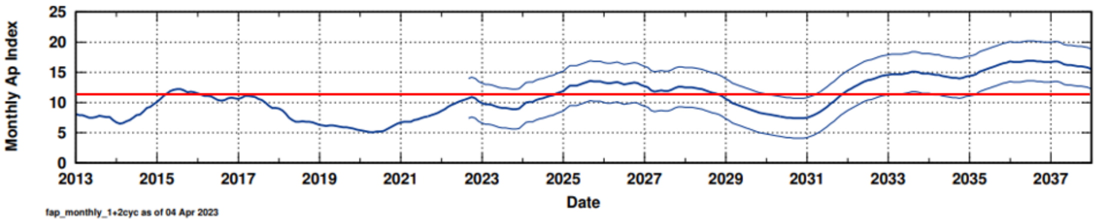


Figure 3.8: ESA prediction for the Monthly AP Index [16]

- Spherical Solar Radiation Pressure: it uses the Dual Cone shadow model, which uses the actual size and distance of the Sun to model regions of full, partial (penumbra), and zero (umbra) sunlight.

For the drag/solar radiation pressure area of SROC, it was used the area of the +Z surface of a 12U CubeSat, while for the propagator it was decided to consider more disturbances than Space Rider. The reason behind this choice is that Space Rider was assumed to be following a controlled orbit, where only the effects of gravity are considered. These external forces, especially the atmospheric drag, change the orbital parameters of SROC during the mission and can affect, some in a bigger magnitude than the others, the required deltaV or duration of each manoeuvre.

3.2.2 Mission Control Sequence

This paragraph describes all the mission segments, in which all the different phases and subphases of the mission have been divided to compose the following Mission Control Sequence (MCS).

PreDeployment: this phase is used to define all the properties of SROC, which have been discussed in the previous section.

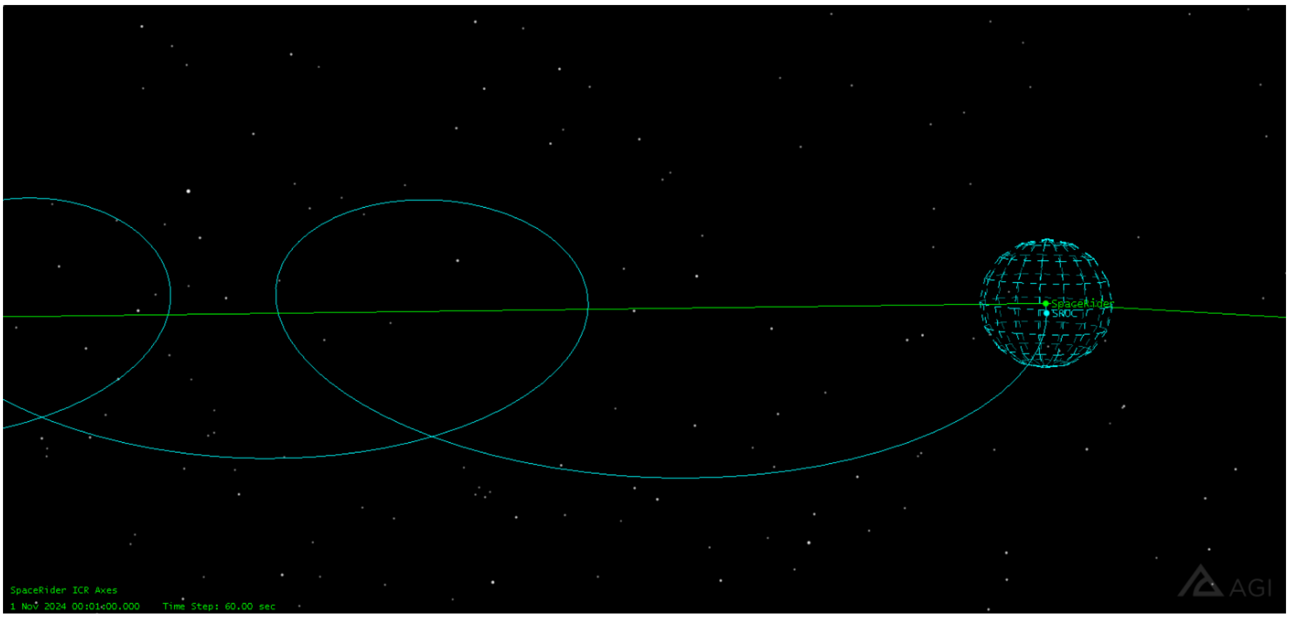


Figure 3.9: SROC initial trajectory after the deployment

Deployment: this impulsive manoeuvre was made to simulate the deployment of SROC from Space Rider's MPCB; for this reason, since it will not be performed in the real mission, the deltaV accounted for this manoeuvre will not be considered in the total deltaV evaluation. A previous study defined this manoeuvre to avoid any possible collision/conjunction with Space Rider:

- Azimuth: 180 deg
- Elevation: -80 deg
- Magnitude: 0.5 m/sec

Figure 3.9 and Figure 3.10 show the deployment direction and the initial trajectory of SROC after the deployment.

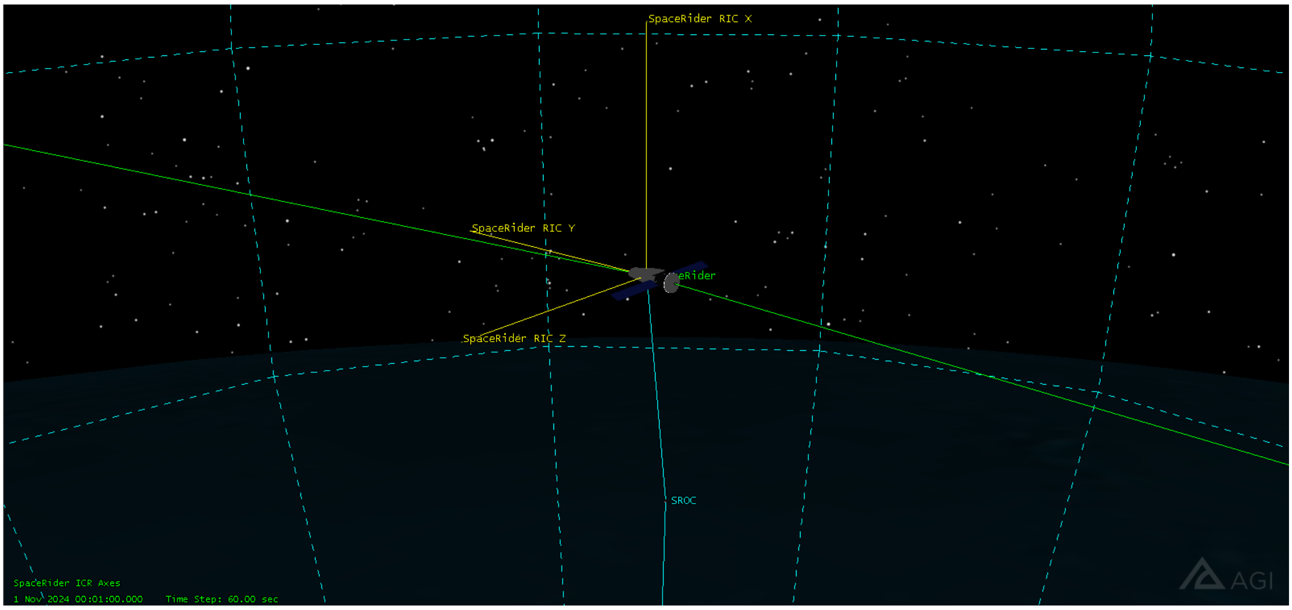


Figure 3.10: SROC deployment overview

Commissioning: this segment is just a propagation one which simulates the free flight during SROC's commissioning sub-phase. Its stopping condition is the duration: after 5 days the commissioning ends. At the end of this mission segment, the final SROC position is:

- Radial: -10.5 km
- InTrack: 372.9 km
- CrossTrack: -0.007 km

What happens during this propagation is that, because of the drag force, SROC decreases its semi-major axis (Figure 3.12) and increases its relative speed, especially along the InTrack direction (as shown in Figure 3.11, where the InTrack position increases exponentially). This behaviour would be seen also if the effects of the drag force were considered for Space Rider's propagator since it has a bigger ballistic coefficient.

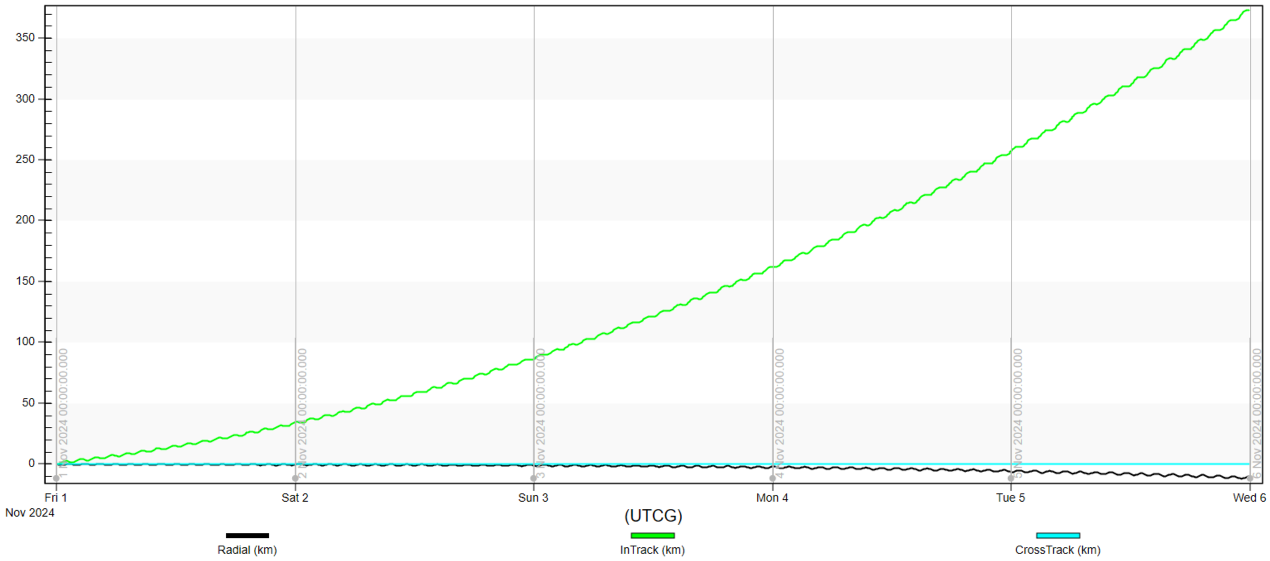


Figure 3.11: RIC components during the commissioning

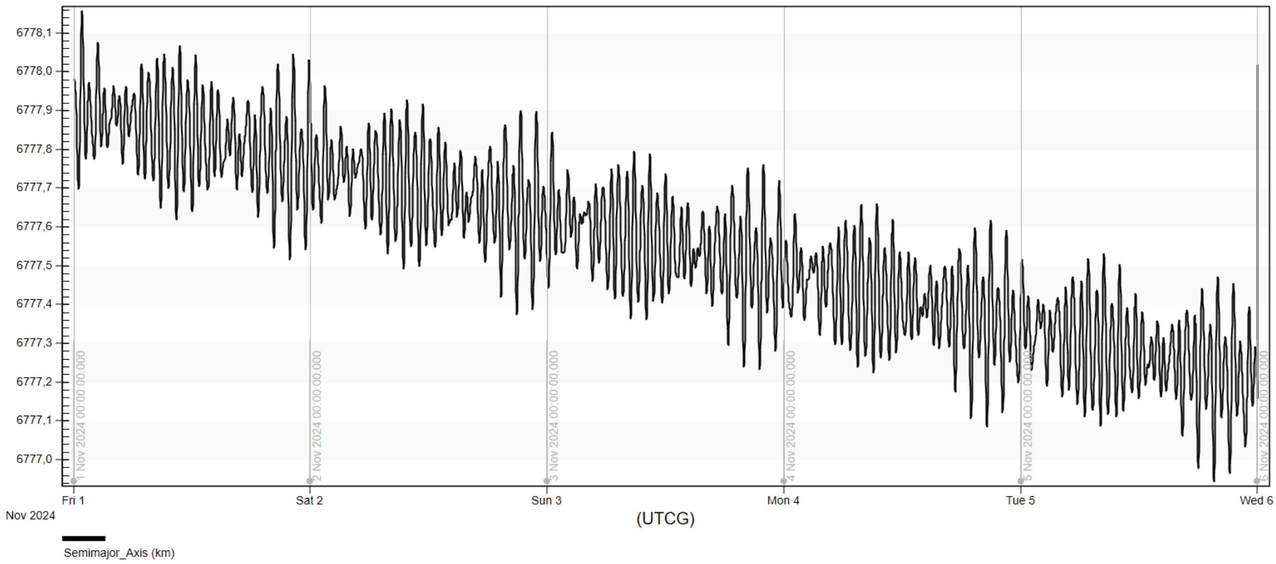


Figure 3.12: SROC semi-major axis during commissioning

HP1: this target sequence simulates the verification sub-phase. As explained in Sub-section 2.3.1.1, since the main manoeuvres to be performed in this sub-phase are yet to be decided, this mission segment was only modelled as a manoeuvre ("Enter HP") and a propagation segment ("Hold Point)", with the differential corrector set to ensure that the semi-major axis of SROC at the end of the sequence will be the same as Space Rider's. As it can be seen from Figure 3.13, the segment shown here is not a proper hold point, since the relative RIC components vary by a few km during it, but it is more a manoeuvre to slow down SROC's drift from Space Rider. However, this nomenclature was still kept in order to be coherent with the Mission Analysis Report [19]. Finally, the duration has been set temporarily set to 4.5 hours, which was the value used for the previous studies. When the Verification sub-phase will be better defined, it is probable that the manoeuvres performed for this segment will change, thus changing its duration and deltaV required.

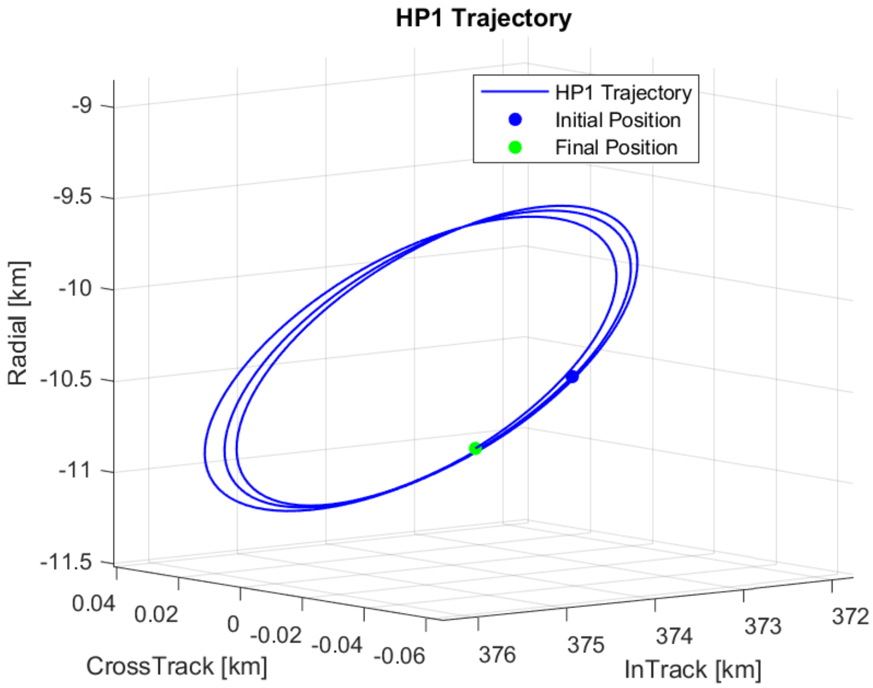


Figure 3.13: HP1 Trajectory

IPA Rendezvous: this target sequence simulates the In-Plane Approach rendezvous. Its duration is set to 5.76 days and the InTrack target for the differential corrector is 7 km; these two values are the result of an optimization, whose main constraint and objectives are described in Section 4.2. This target sequence comprises an impulsive manoeuvre segment ("IPA Rendezvous Man") followed by a propagation one ("PropToSR"). The control parameter for the differential corrector is the thrust along the V axis of SROC's VNC coordinate reference system.

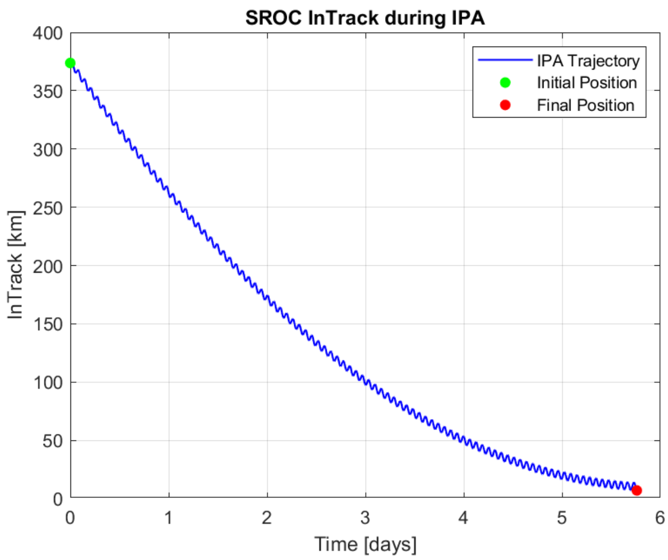


Figure 3.14: SROC InTrack during IPA

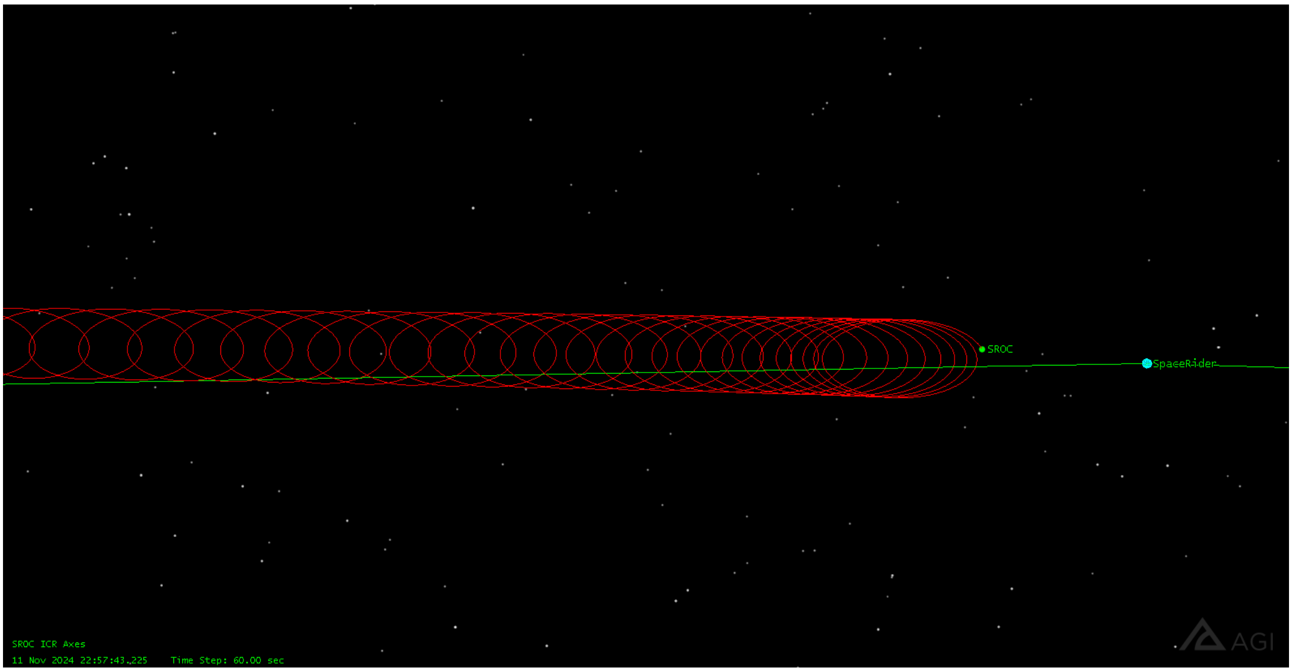


Figure 3.15: SROC's relative final motion at the end of the IPA

HP2 Insertion: this target sequence, is again composed of an impulsive manoeuvre segment (“HPInsertion Man”) and a propagation segment (“PropToHP”); its duration (2 hours) was evaluated using the same optimization process used for the IPA Rendezvous, while the InTrack Target (2 km) was chosen to get SROC as close as possible to Space Rider, while also respecting the observation requirements. Figure 3.16 shows the passage from the last moments of the IPA to the OPA. The desired results of the differential corrector are all the relative position vectorial components on the RIC axes (0 km along CrossTrack, 2 km along InTrack and 0 km along Radial) at the end of the propagation segment. The control parameters are the thrust vectors along all three axes of the VNC reference system.

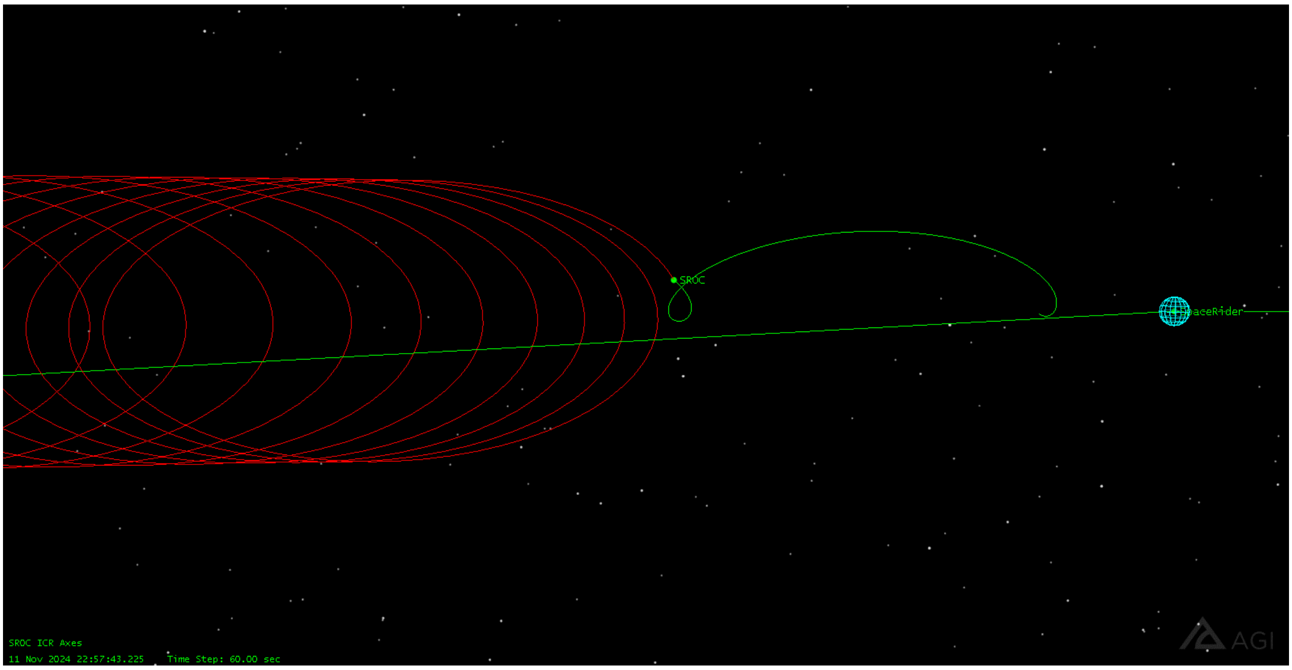


Figure 3.16: Last orbits for IPA (red) and the OPA (green)

ZeroRelVel2: the HP2 insertion manoeuvre is completed with this target sequence, which has only an impulsive manoeuvre and it is set at zero the relative velocity between SROC and Space Rider. By doing so, at the end of this segment, the satellite has null relative velocity and has a relative position of 2 km along the InTrack axis and 0 km for both the Radial and CrossTrack axes. The control parameters are again the thrust vectors along all three axes of the VNC reference system.

HP2 Sequence: this segment was not part of the first version of the code but was added during the development of this thesis to provide an HP2 more similar to the actual manoeuvre, where the position is continuously controlled to guarantee an almost constant relative position with respect to the target. The manoeuvre was set to last 4.5 hours. The details about how it works and how it has been defined are reported in Section 4.3. The sequence is divided into nested sequences and each of them is composed by the following segments:

- HP2 target sequence: it targets the desired relative position, and it is in turn composed of a finite manoeuvre and a propagation segment;
- ZeroRelVel target sequence: it sets to zero the relative velocity of SROC;
- Propagation segment: SROC freely propagates until its relative position exceeds the maximum error on the relative position.

Inspection: this is another sequence composed of the following segments:

- OPA Rendezvous: it is a target sequence that simulates the Out-Of-Plane Rendezvous from the HP2 to the insertion to the WSE. It contains one impulsive manoeuvre segment (called "PositionMan") and one propagation segment (called "PropToWSE"). The differential corrector is set to reach the following position:
    - Radial: 0.216 km
    - InTrack: -0.047 km
    - CrossTrack: -0.129 km

  This is the starting point for the WSE, and it is evaluated using a Matlab function described in Section 5.3. Figure 3.17 and Figure 3.18 show SROC's relative trajectory respectively on the InTrack-Radial and CrossTrack-Radial planes.

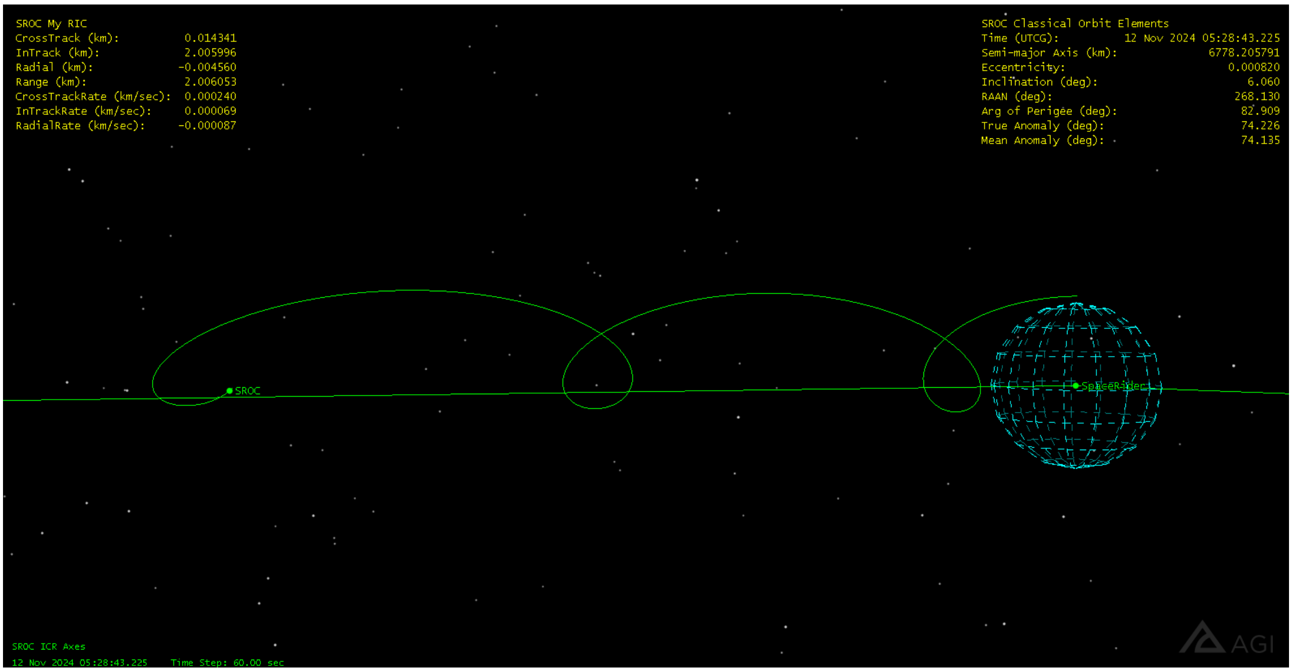


Figure 3.17: SROC's trajectory during the OPA – InTrack-Radial plane view

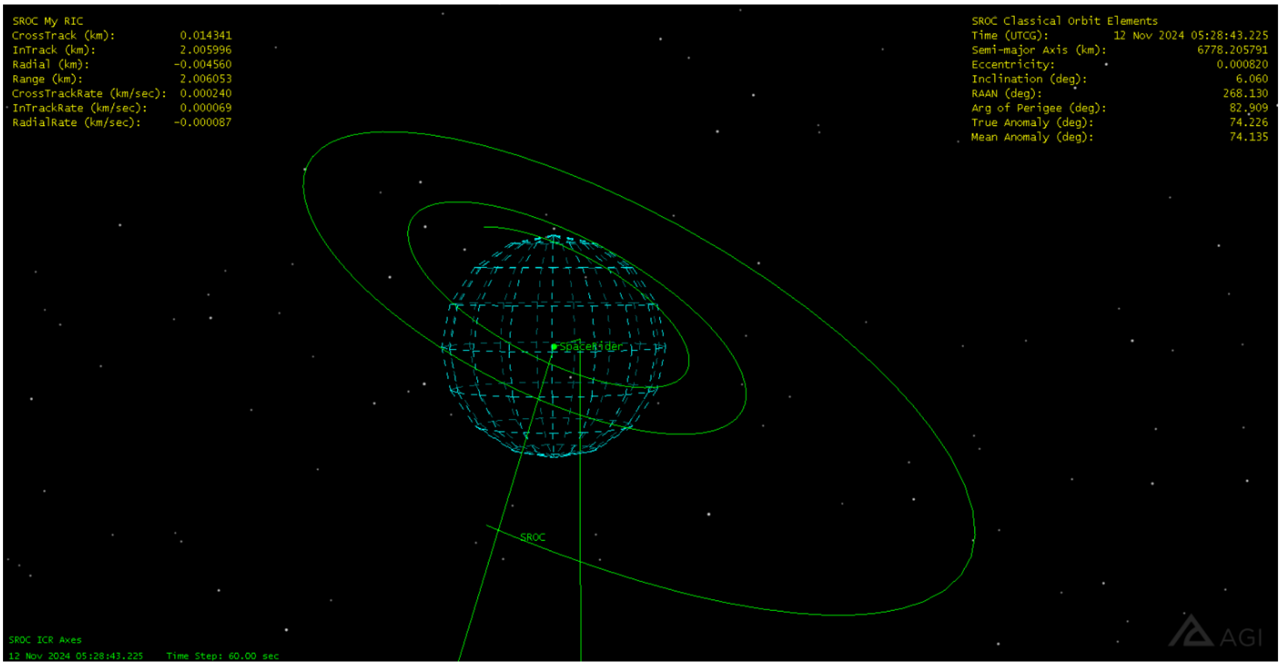


Figure 3.18: SROC's trajectory during the OPA – CrossTrack-Radial plane view

- WSE Insertion: this target sequence is only composed of an impulsive manoeuvre, called “VelocityMan”. The target profile sets the desired values for InTrack, Radial and CrossTrack rates, defined by the same Matlab function used to define the WSE insertion point.
- Inspection: this segment is just a propagation one lasting 8 hours. It simulates the observation phase as one uncontrolled propagation. The impulsive manoeuvre boosts the WSE along negative InTrack, then, because of the effect of the drag force, the WSE starts moving along positive InTrack. By doing so the duration of the observation phase increases. Figure 3.19 shows SROC’s relative trajectory during the Inspection.

Although the nominal scenario considers only one observation cycle, by considering all these mission segments inside a single sequence, it is easier to add more inspection by simply copying and pasting the external one.

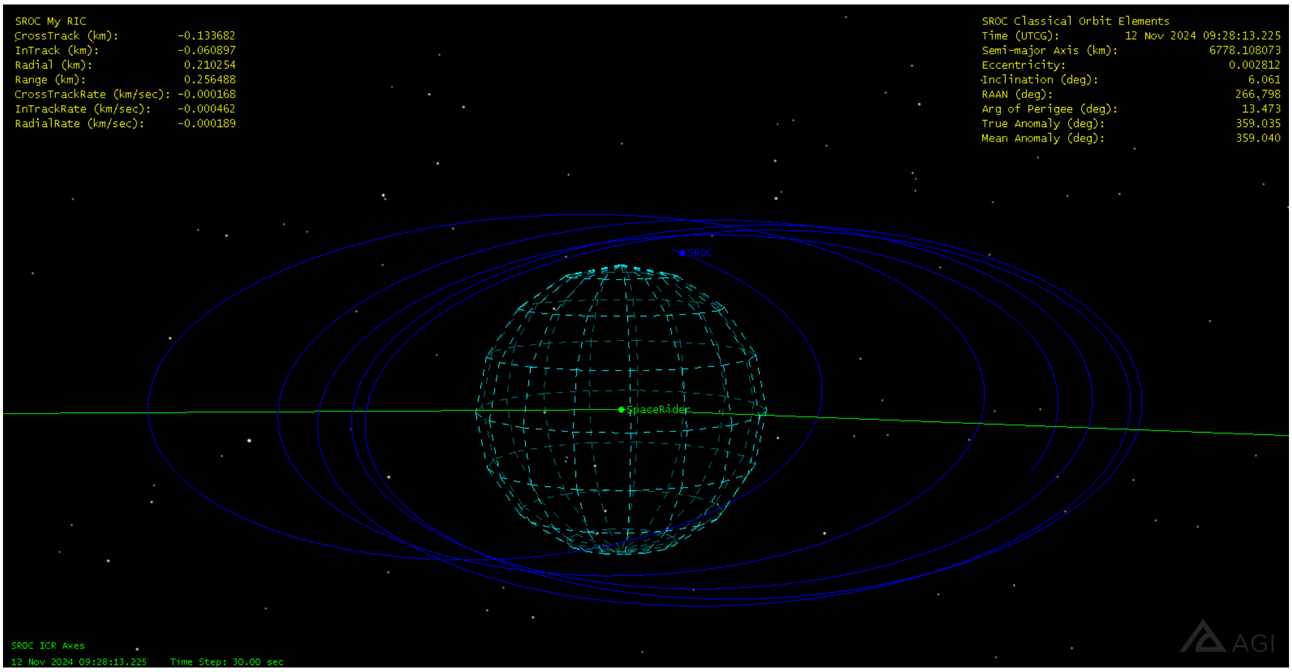


Figure 3.19: Last part of the OPA rendezvous (green) and WSE (blue)

Free Flight: this propagation segment represents the free flight after the inspection, during which the satellite sends to the Ground the mission data. It presents two stopping conditions and only one of them is required to stop the propagation: either the duration exceeds 16 hours (which should be more than enough to downlink the data) or the relative range exceed 2 km. In the nominal case, the condition which actually stops the propagation is the second one, thus causing the duration to last 8.06 hr. Figure 3.20 shows, besides the position and speed in the RIC reference frame, that the range at the end of the free flight is 1.99 km. This segment and the WSE could be represented by only one propagation segment since there are no manoeuvres between them. The division between these two is the result of a trade-off between how much time after the WSE insertion can be deputed to the observation and how much is required to send data to the Ground. For the Observe scenario, this is the last segment considered, while for the Observe&Retrieve, the free flight is followed by the HP3 insertion.

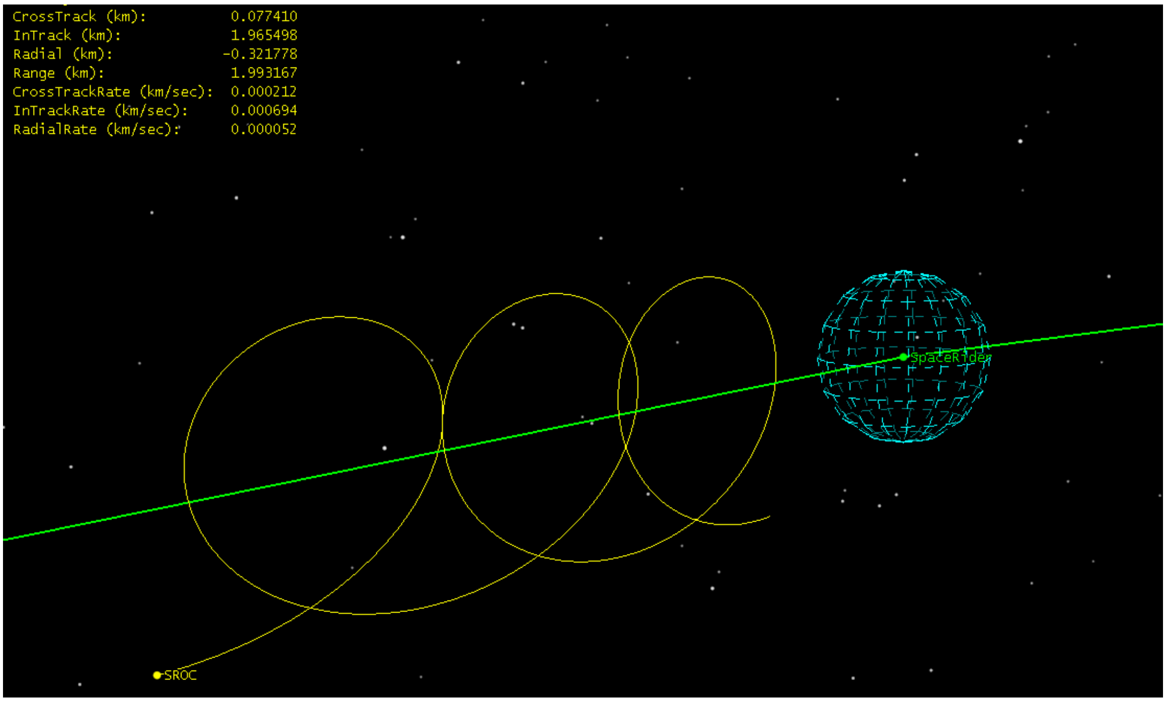


Figure 3.20: Last relative orbits of the free flight propagation

HP3 Insertion: this target sequence has the same structure as the one used for the HP2 insertion; it is composed of an impulsive manoeuvre segment ("HPInsertion Man") and a propagation segment ("PropToHP"); its duration (2.7 hours) was evaluated using the same optimization process used for the IPA Rendezvous, while the InTrack Target (0.2 km) was chosen to get SROC just at the limit of Space Rider's KOZ. The desired results of the differential corrector are all the relative position vectorial components on the RIC axes (0 km along CrossTrack, 0.2 km along InTrack and 0 km along Radial) at the end of the propagation segment. The control parameters are the thrust vectors along all three axes of the VNC reference system. Figure 3.21 shows SROC's relative trajectory during this segment, while Figure 3.22 highlights that the final position is at the perimeter of the KOZ.

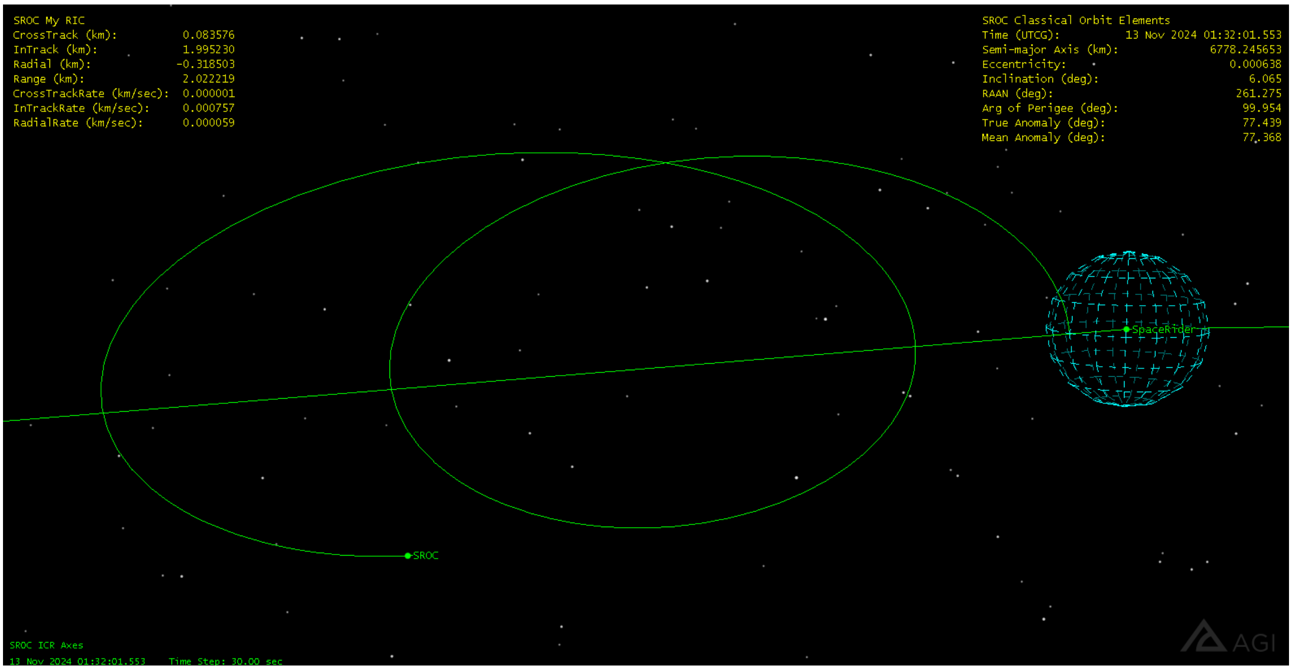


Figure 3.21: SROC's reltavi trajectory during the HP3 insertion

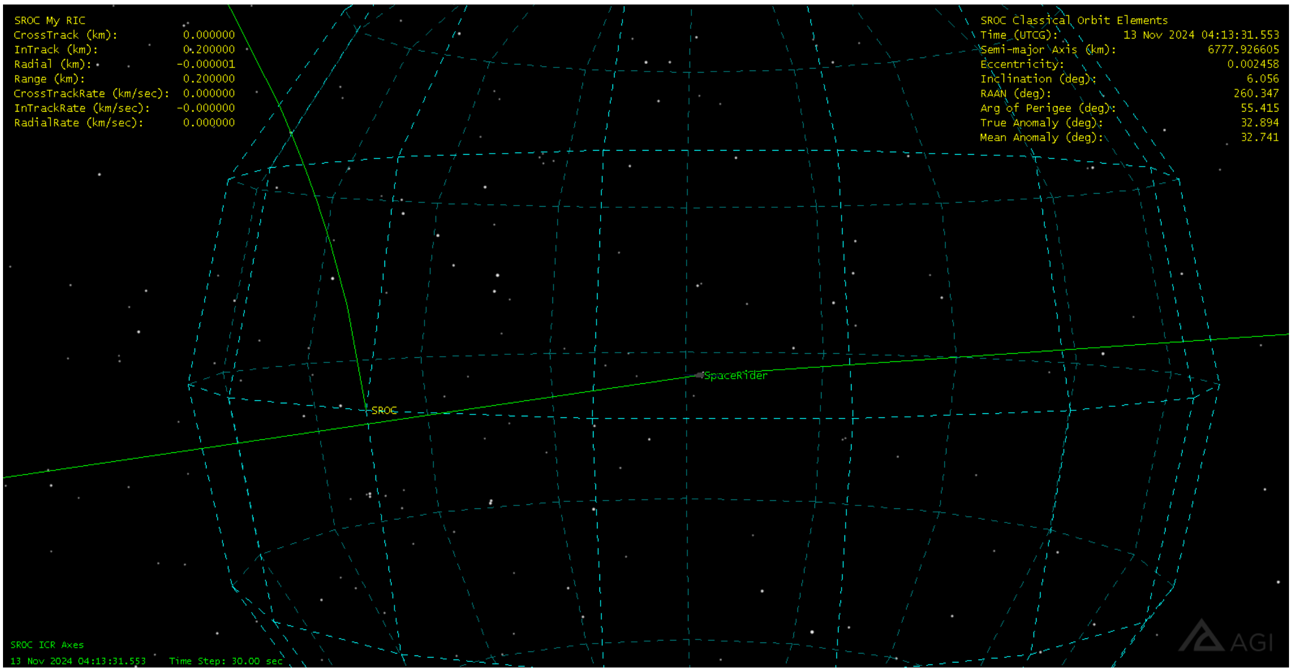


Figure 3.22: SROC's final relative position at the end of the HP3 insertion

ZeroRelVel3: this target sequence, its segments, its control parameters, and its desired results are the same as for ZeroRel2.

HP3 Sequence: this sequence is similar to the HP2 sequence, with the only difference being the desired relative position, which is now 0 km along CrossTrack, 0.2 km along InTrack and 0 km along Radial.

4 Updated Matlab Functions

The Matlab functions are a crucial part of the analysis process performed for this study. They work as an interface between the user and the STK, automating actions that would be tedious and repetitive to perform and which would greatly increase the analysis time. The main tasks of these functions usually are:

- Setting the mission segments (e.g.: defining the duration and the stopping condition of a propagation segment, the target results and the control parameters of the differential corrector);
- Manage the MCS, by adding or removing segments;
- Run the STK scenario;
- Post-process the data from STK (e.g.: defining the optimal manoeuvre or producing plots);

The Matlab code has been organized in the following way: a main function sets the interface with STK and the scenario, then calls specific functions to define or analyse each mission segment. This software structure was already defined before starting this thesis, however, before actually using or expanding it, it was reorganized and updated. This process was necessary since until phase B1 several mains and functions were produced to analyse mission segments or scenarios which are now discarded or significantly different. Since describing exactly every minor change would be unnecessarily long and not particularly important to understand the scope of this thesis, the main features of this first task work can be summarized as follows:

- When there was one or more variation of the same function, they were condensed into a single function; of course, keeping only the useful features;
- Small errors were identified and corrected;
- All the variables' names were updated to be consistent with the nomenclature used in the Mission Analysis Report [19]; this change was also applied to the JSON files and the STK scenario;

The only major changes which will be further discussed are:

- Improvement of the performances of both the IPA optimization (Section 4.2) and HP definition (Section 4.3) functions;
- Addition of several flags to avoid entering the KOZ during the Observation Phase (Section 5.3)

4.1 Analysis Process Overview

Figure 4.1 illustrates the workflow which was followed every time an analysis was performed:

- The Matlab function is started;
- The Matlab function retrieves all the information required to set the STK scenario, which usually are the properties of each mission segment, such as its duration, the propagator used during the propagation phase or the desired target for a target sequence. These data are saved in different JSON files;
- Using the STK object model [8][9] the Matlab function connects to STK scenario and sets the Astrogator propagator for SROC;
- The STK scenario performs the orbital propagation and evaluates the thrust magnitude and orientation required to get the desired result(s) for the target sequences;
- The STK object model is used by the Matlab function to retrieve the output of the STK simulation, to produce tables and graphs. If its relative control flag is true, STK can also overwrite each of the JSON files according to the results of the analysis;

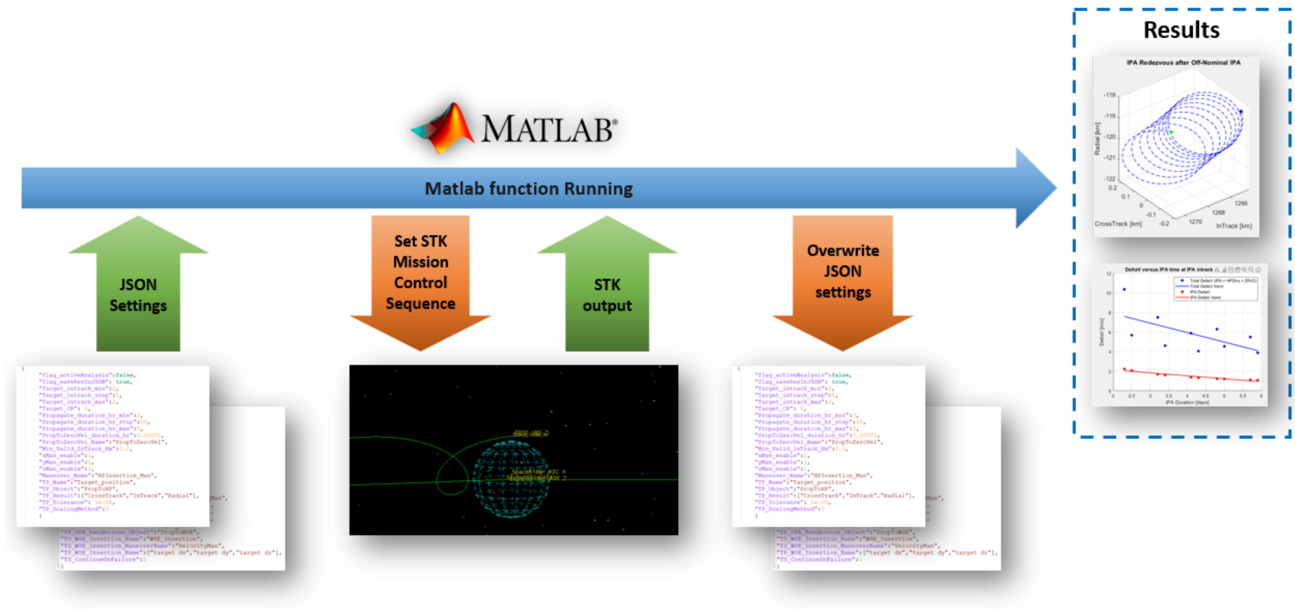


Figure 4.1: Analysis Process Overview

The STK Object model is an object-oriented interface to STK, built on Microsoft Component Object Model (COM) technology and, among the different environments to which is compatible, it can be used in Matlab. This Object model is a collection of different COM libraries containing type, interface, events, and classes representing the many aspects of the STK application structure; for this thesis, it was mostly used the STK Astrogator COM library, since the purpose of the Matlab function is to model and analyse the MCS in Astrogator.

4.2 IPA Optimization

The definition of the optimal IPA manoeuvre is one of the most complex and long analyses performed by the Matlab and STK functions. First, it is important to define what makes an IPA the optimal one: the minimization of the total deltaV cost required to perform the IPA, the HP2 insertion, and the zeroing of the relative velocity (called ZeroRelVel2 in the MCS). The reason why these three deltaVs are considered together is that the constraints imposed on the IPA, which are the target position and the duration of the segment itself, determine both the final relative position and velocity of SROC, thus also affecting the successive mission segments. The effect on the mission segments after the HP2 is considered negligible since the HP2 always starts at a specific relative position (0 km Radial, 2 km InTrack and 0 km CrossTrack) and relative velocity (null).

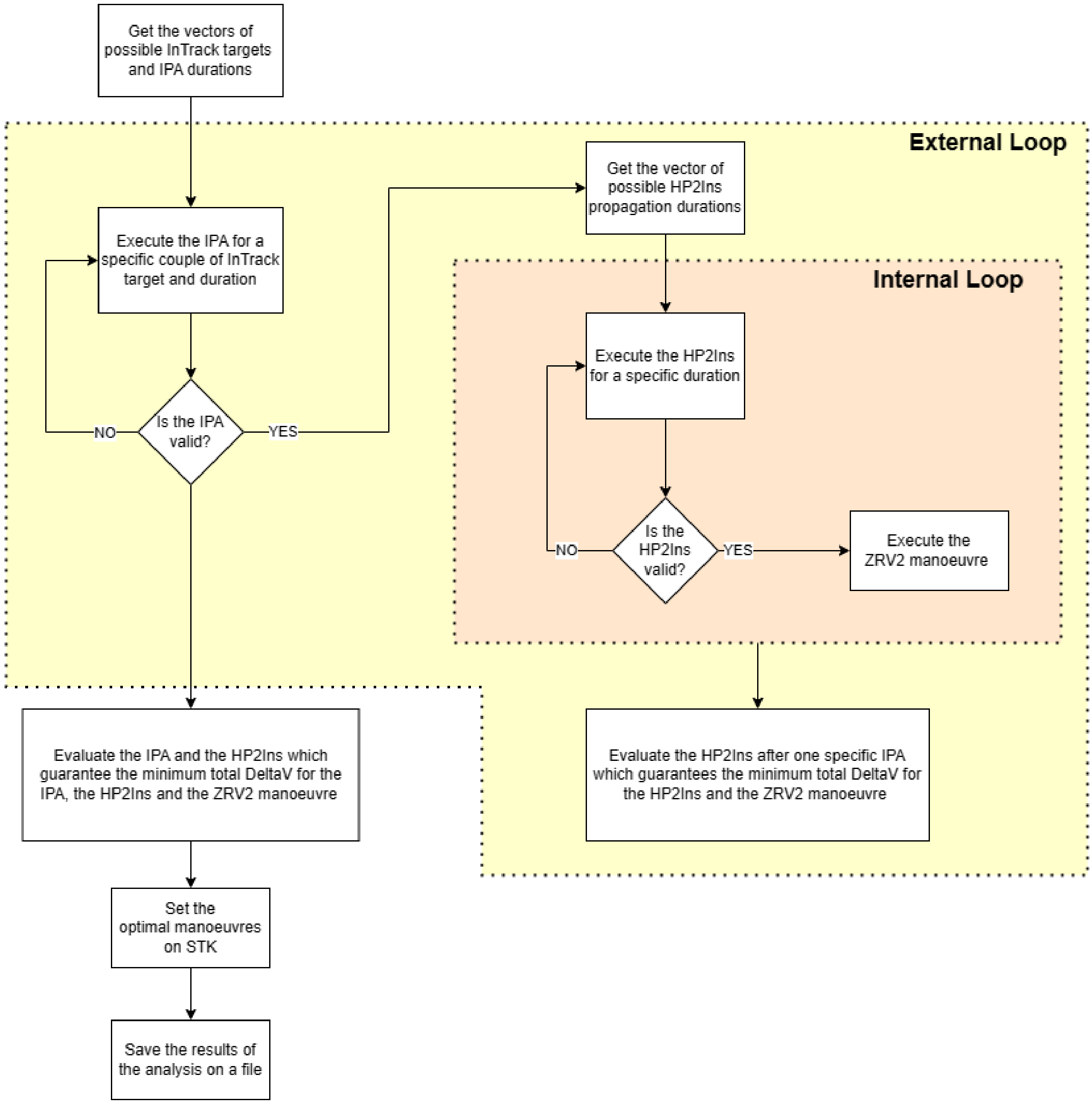


Figure 4.2: Diagram showing the functioning of the IPA optimization

Before the update, the IPA optimization function evaluated only the deltaV of the IPA itself and the HP2 insertion; now it also evaluates the effect on the ZeroRelVel2 manoeuvre. Moreover, the function used to evaluate the optimal IPA more times than necessary, thus increasing the run time.

Figure 4.2 shows how the updated version of the function works. The possible IPA manoeuvres are evaluated by considering every combination between the elements of a vector composed of target InTrack values with the elements of a vector composed of IPA duration values (External Loop). For each combination, the STK scenario is run, and its results are analysed to determine if the solution is valid, which means that the IPA must fulfil the following constraints:

- The IPA final InTrack position obtained in STK must not differ by more than 0.5 km from the desired final InTrack position;
- SROC must not cross below 200 m along the InTrack axis during a 24-hour propagation after the IPA completion. The Matlab interface with STK is used to add a 24-hour propagation segment, then, at the end of the optimal IPA definition, this segment is eliminated since it does not really occur in the

mission ConOps. This condition was added to assess the safety of this manoeuvre in case an off-nominal condition prevented SROC to perform the successive manoeuvres for 24 hours; this period of time was chosen to simulate the time required to assess the occurrence of a fault and to make SR perform a Collision Avoidance Manoeuvre from SROC. Figure 4.3 shows that the 24-hour propagation (in green) after the IPA (in red) does not cause SROC to decrease its InTrack distance below 200 m. In fact, the minimum InTrack distance is approximately 5 km away from Space Rider. At the beginning of the propagation, SROC is moving at a low speed and it is progressively slowed down by the atmospheric drag until its relative speed changes its direction from toward SR to the opposite direction. This means that even if this propagation lasted more, it would not change the minimum relative distance, but it would only cause SROC to move away even more from SR, thus making an SR collision avoidance manoeuvre useless.

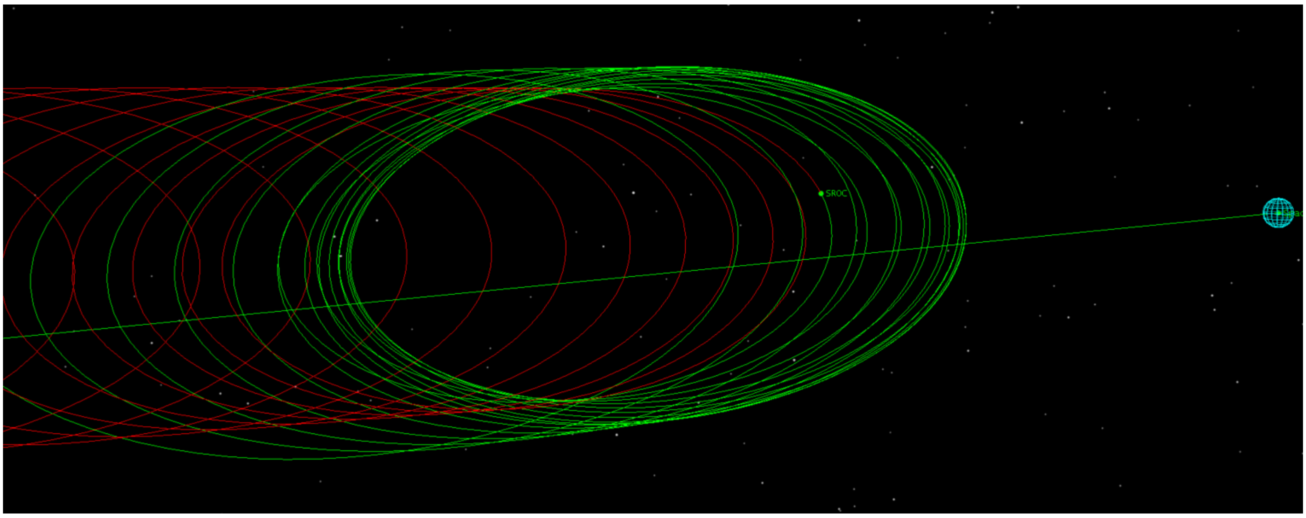


Figure 4.3: trajectory during the 24 hours propagation (green) after the IPA (red)

If the IPA iteration is valid, the Inner Loop is started: all the possible HP2 insertions are evaluated iterating on the duration of the propagation segment during the insertion. The final position is fixed at 0 km along CrossTrack, 2 km along InTrack and 0 km along Radial. An HP2 insertion is considered valid if:

- The IPA final InTrack position obtained in STK must not differ by more than 0.5 km from the desired final InTrack position;
- SROC must not cross below 200 m along the InTrack axis during the propagation to the insertion point. This constraint was set for safety purposes to avoid SROC from passing through the KOZ or flying "behind", that is in the negative InTrack, SR. Figure 4.4 shows an example of a not-valid HP2 insertion manoeuvre: although the final position is the one desired, SROC reaches it by passing behind SR;

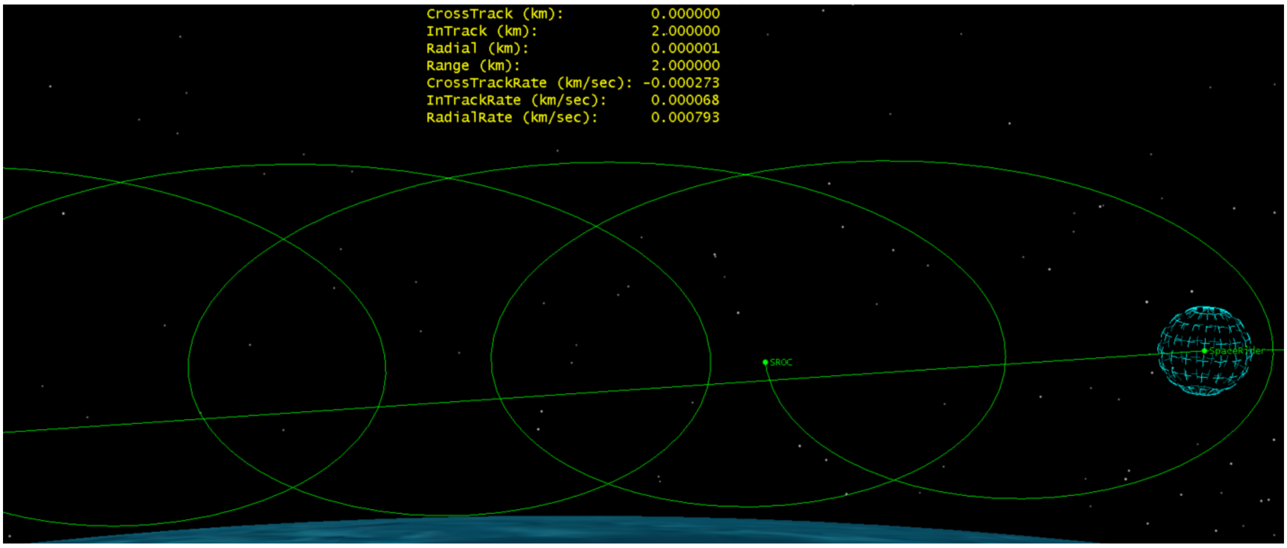


Figure 4.4: Final position of a not valid HP2 insertion

When an HP2 insertion is valid, the successive ZeroRelVel manoeuvre is evaluated and its deltaV cost is saved. After iterating on all the possible HP2 insertions and simulating their relative ZeroRelVel manoeuvres, the optimal insertion, in terms of minimum total deltaV for both the insertion and the ZeroRelVel manoeuvre, is evaluated. It is noted that this is not the absolute optimal result, but only the optimal result for a specific IPA. When the External Loop finishes, which means all the possible IPAs have been evaluated, the optimal IPA + HP2 insertion + ZeroRelVel manoeuvre is determined. Finally, the selected sequence is set on STK, and the scenario is run to save these changes.

Table 2.1 shows the improvement in the total deltaV cost with respect to the previous version of the code. While the cost for the IPA is almost the same, the updated version of the code sets an insertion that requires a higher deltaV but guarantees a lower deltaV for the ZeroRelVel manoeuvre. The previous version of the code, instead, includes a much less deltaV-consuming insertion, but a much higher ZeroRelVel manoeuvre. This is because it considers only the insertion in the optimization process, so it sets the less deltaV-consuming inspection, with no regard for the cost of the successive manoeuvre.

Table 4.1: deltaV budget for IPA + HP2 insertion + ZeroRelVel2

Mission segment	Previous Code	Updated Code
IPA Rendezvous deltaV [m/s]	0.486	0.485
HP2 Insertion deltaV [m/s]	0.188	1.280
ZeroRelVel#2 deltaV [m/s]	1.877	0.423
Total Sequence deltaV [m/s]	2.551	2.188

Figure 4.5 shows the evolution of the RIC rate during the last hours of the IPA and the whole HP2 insertion. As shown in the lower image, the relative velocity of SROC is almost null for every RIC component: the kinetic energy variation required to nullify the relative speed is very low, thus demanding a low deltaV impulse during the ZeroRelVel manoeuvre.

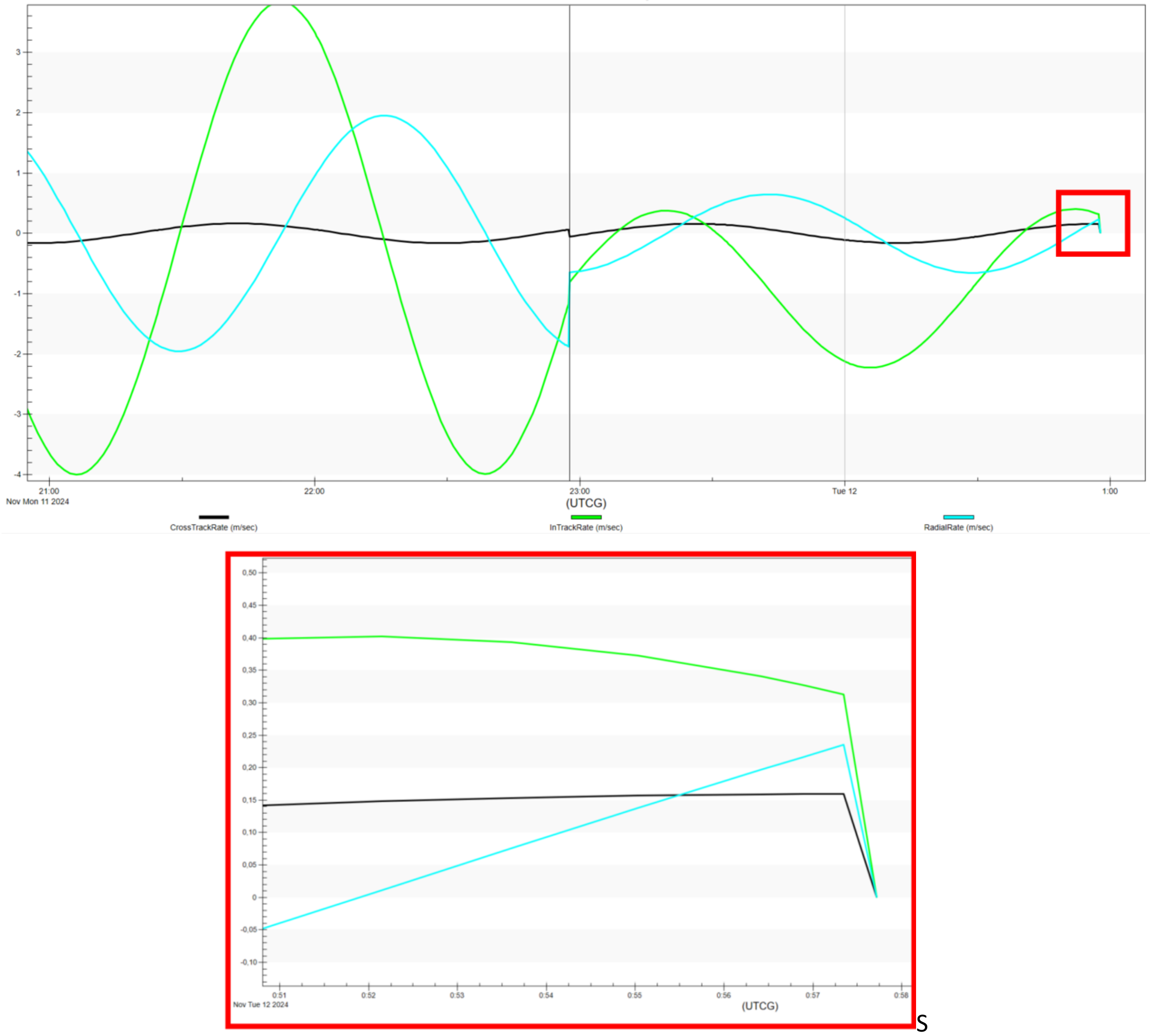


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Figure 4.5: RIC Rate at the end of the IPA and during the HP2 insertion (up); zoom on the RIC rate after the ZRV2 manoeuvre (down)

Figure 4.6 shows that in case any faults prevented the execution of the ZeroRelVel manoeuvre, SROC would not enter SR KOZ; the minimum range, in this case, would be 1.119 km.

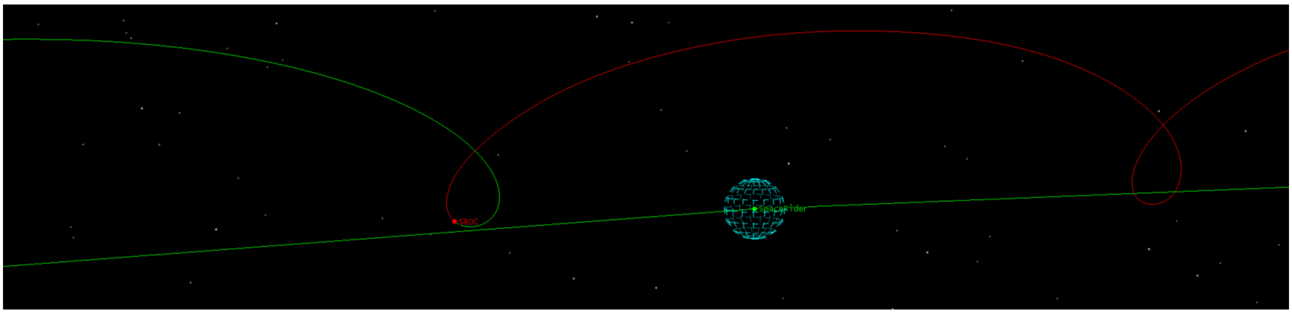

Figure 4.6:Trajectory of a propagation segment (red) after the nominal HP2 insertion (green)

The discussion so far focused on the analysis process performed by the Matlab code but not on which functions were used and how they communicate between them. Figure 4.7 schematizes the features of these functions and how they interact: each coloured blocks represent a Matlab function, whose name is placed on the top of the block itself. The functions have also been divided between analysis functions, which actually interface with STK, and utility functions, which process the data obtained from them.

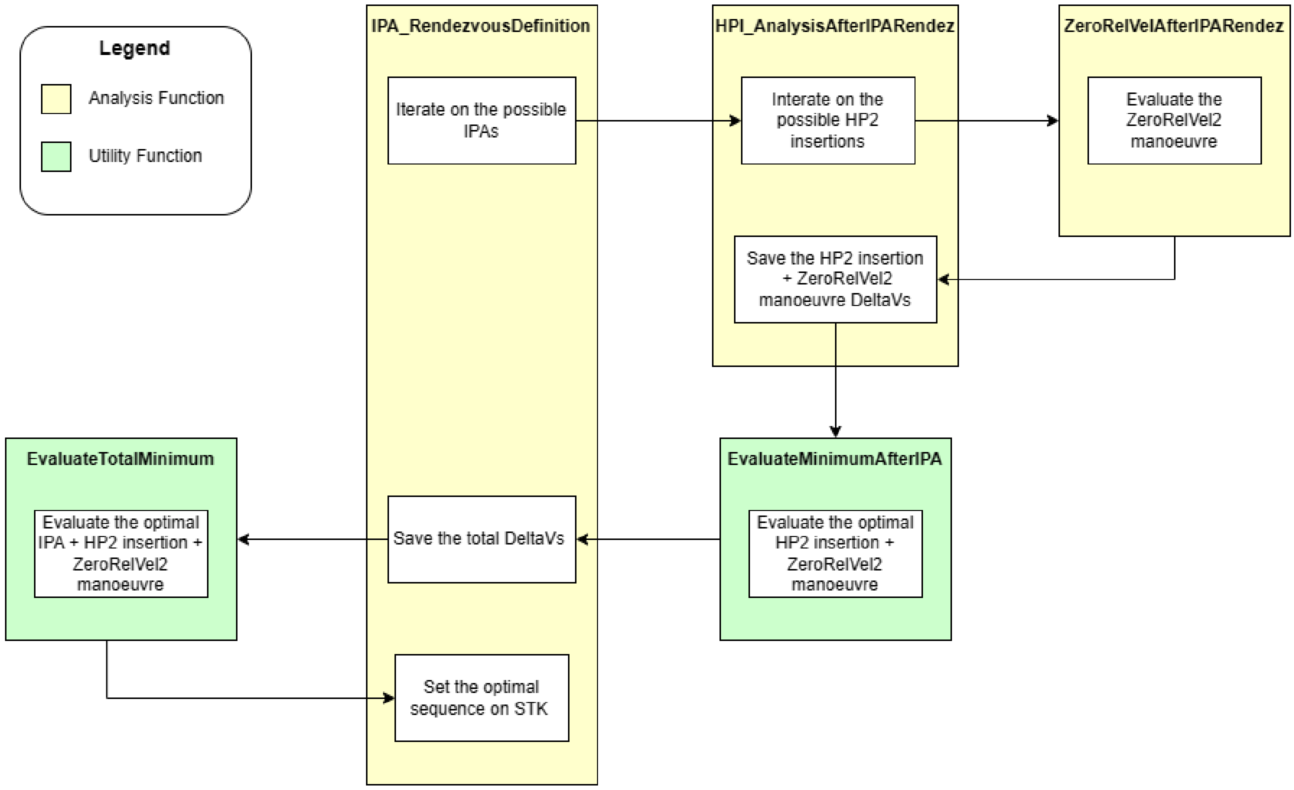

Figure 4.7: Matlab functions flowchart

4.3 HP Sequence

The Hold Points 2 and 3 were defined using a new Matlab function, called "HoldPointTrue_Sequence". This update was made to add the following features to the HP segment and the analysis function:

- Set a minimum and maximum relative distance during the HP
- Model the HP as a sequence of multiple finite burns, instead of a single one

By doing so it will be possible to change the main properties of the segment if or when more precise constraints will be available. Moreover, this modelling of the HP better reflects what the real segment could be, thus giving a more faithful estimate of the deltaV which is crucial to the scope of this thesis.

Figure 4.8 shows the segments composing the HP sequence:

- **First Propagation**: this propagation segment takes place after the Hold Point insertion and the ZeroRelVel manoeuvre. Initially, the satellite is exactly at the desired Hold Point relative position (0 km along CrossTrack, 2 km along InTrack and 0 km along Radial) with an almost null relative velocity, but because of the effect of the external disturbances it starts accelerating and moves from the desired position. SROC keeps drifting until it reaches either the maximum or the minimum acceptable range.
- **HP Burn #n**: this sequence is composed of three sub-segments:
  - **HP target sequence**: the desired result of its differential corrector is the final relative position (0 km along CrossTrack, 2 km along InTrack and 0 km along Radial) and the control parameter is the thrust along all the 3 VNC axes of SROC and the duration of the propagation segment ("To Target"). It is composed of a finite manoeuvre segment and a propagation segment. The target sequence uses three different profiles: the first one is a differential corrector targeting the aforementioned results considering an impulsive manoeuvre. Then, a second profile, called "Change Maneuver Type" changes the manoeuvre from impulsive to finite. The third profile is a differential corrector targeting the same results but considering a finite manoeuvre. The reason why three profiles were used

is that a differential corrector targeting a finite manoeuvre usually requires a representative guess for the thrust vectors: by first running the target sequence with an impulsive manoeuvre, its results can be used as the first guess values. Another peculiarity of this target sequence is that the duration of the propagation segment, called "To Target", after the manoeuvre is not known in advance: this is why "To Target" 's duration is one of the control parameters of the differential corrector. However, to converge on a solution, it is required to start with an accurate first guess of the actual final values. For this reason, the Matlab function iterates on a vector of possible durations (from longest to shortest) and selects the first one which enables the profiles to converge.

- **ZeroRelVel**: this target sequence is identical to the ZeroRelVel segment described in Sub-section 3.2.2: its desired result are null relative velocities along all the RIC axes and the control parameters are the thrust vectors along the three SROC's VNC axes.
- **free flight:** this propagation segment is similar to the First Propagation one: because of the external disturbances SROC starts drifting from the desired position until either the range-stopping conditions are met or the Hold Point duration is reached.

The HP Sequence can be composed of a different number of HP Burn segments, depending on the total duration of the HP. If the free flight stops because of the range constraints and the HP is not over, the Matlab function adds another HP Burn sequence. Its target sequences are reset and recalculated, as well as the duration of the "To Target" propagation segment. This process is repeated until the HP lasts for the desired duration. Since the desired HP duration may vary between different analyses, before performing any action on the whole sequence, the Matlab function erases all the HP Burn sequences except for the first one, thus avoiding scenarios where the HP sequence at the beginning of the analysis is already longer than the desired HP.

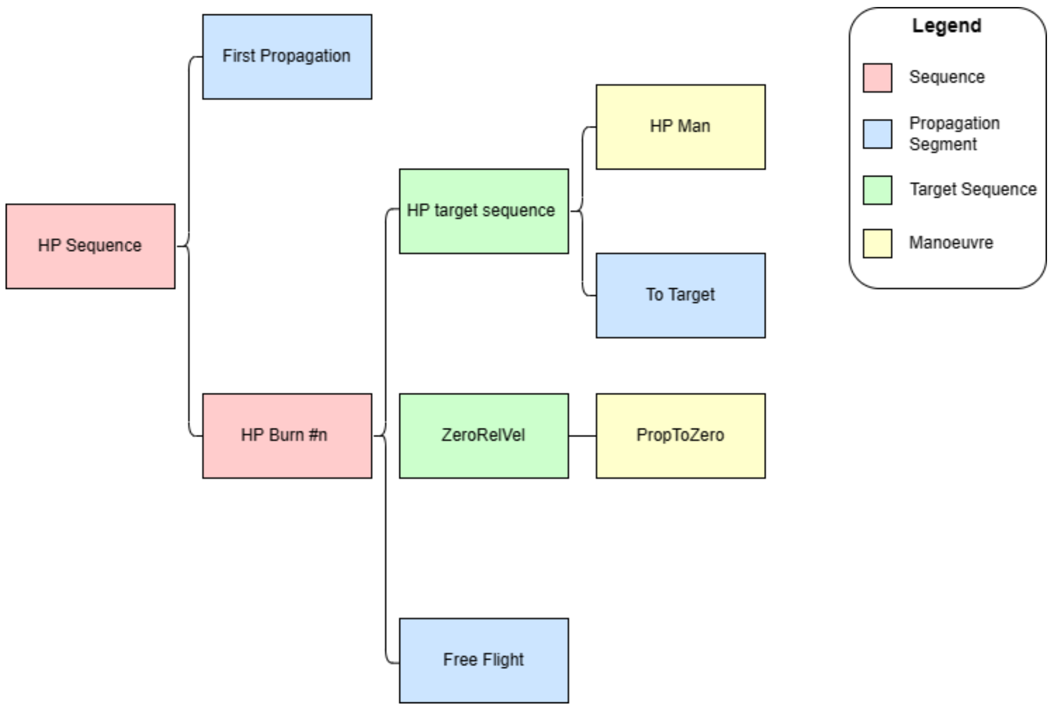

Figure 4.8: HP sequence segments

Figure 4.9 show the variation of the range during HP2 (left) and HP3 (right). For the first one, the maximum range error is 11 m, while for the second one is 13 m. These maximum errors show the improvement from the outdated version of the code which had a maximum error of 20 m. Figure 4.10 shows SROC's trajectory in RIC components for HP2 and HP3. Generally, during HP2 SROC tends to oscillate both between a higher and lower InTrack with respect to the desired position, while during HP3 it mostly moves to higher InTrack values. For both HPs, the displacement along the CrossTrack axis is almost negligible. This is probably due to the fact that the biggest disturbance, that is the atmospheric drag, mostly acts on the InTrack and Radial positions: by decreasing the spacecraft speed, it changes its semimajor axis, thus varying the InTrack and Radial coordinates.

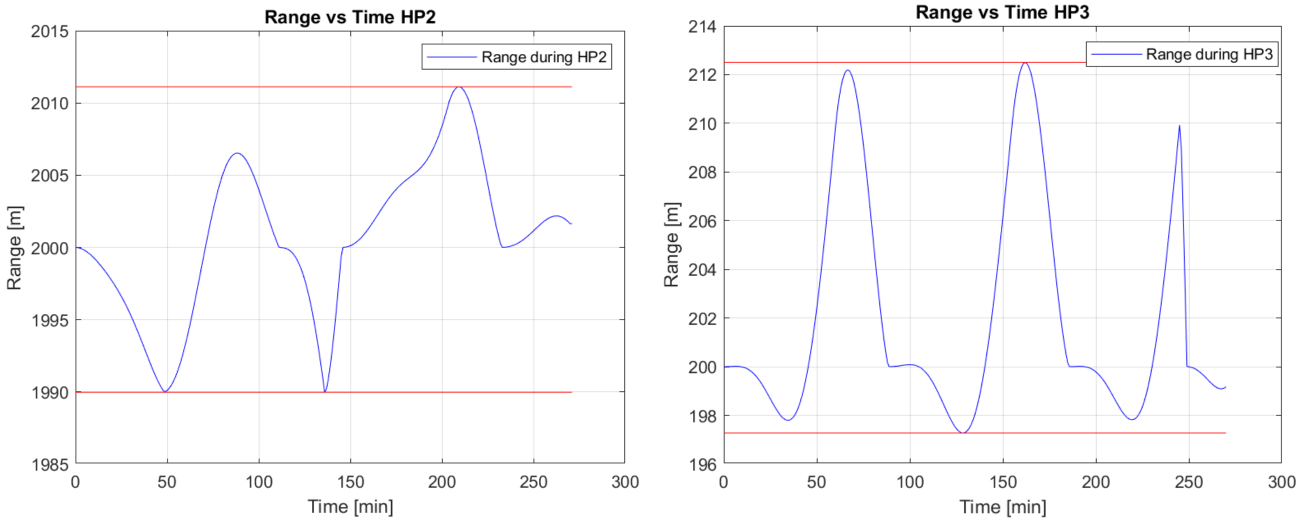

Figure 4.9: SROC Range as function of the time from the beginning of the HP; HP2 is on the left and HP3 on the right

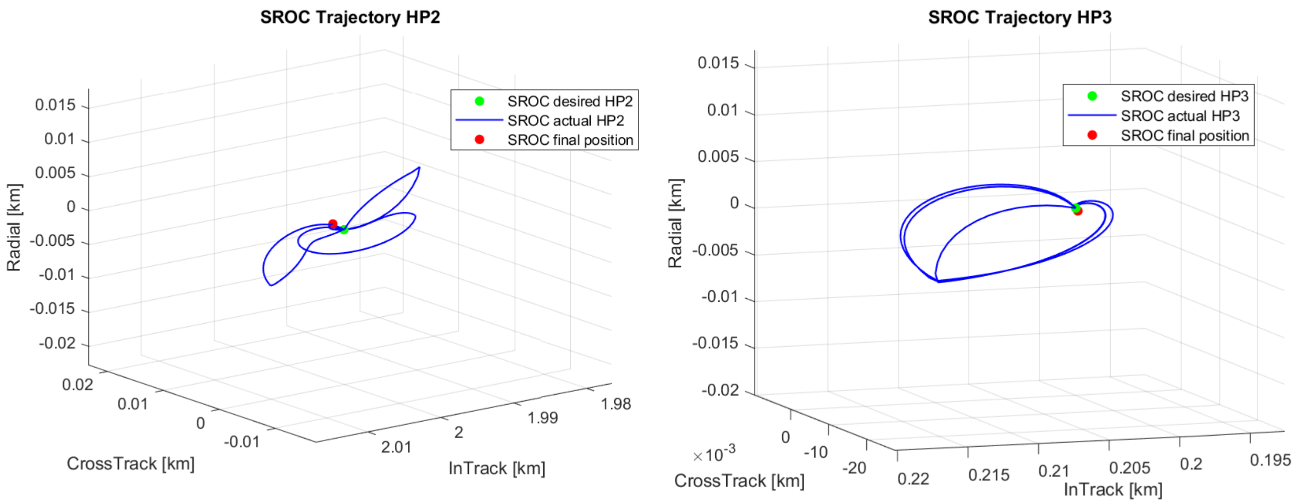

Figure 4.10: SROC trajectory as function of the time from the beginning of the HP; HP2 is on the left and HP3 on the right

Table 4.2 and Table 4.3 list the duration and the deltaV cost of every segment of respectively HP2 and HP3. Especially for HP2, the deltaV cost of the manoeuvres and the duration of the successive propagation segments noticeably vary between the different burns. The different behaviour between the two HPs may be due to the fact that the ZeroRelVel manoeuvres, although highly reducing the relative speed, still leave SROC with a small relative velocity with respect to SR., which then influences the successive propagation segments. Moreover, a more stable behaviour could be achieved by integrating the actual Simulink model of SROC's propulsion system and GNC algorithms with STK, or at least by mimicking its behaviour with a simpler closed-loop controller. In conclusion, the quality and results of this analysis were still considered more than adequate to evaluate the deltaV required to perform the manoeuvre; in fact, the relative position achieved at the end of both HPs has a relative range error of less than 0.1% and, as seen, before, the maximum absolute range error during the HPs is 13 m for HP3.

Table 4.2: deltaV and duration of all the HP2 segments

Segment		deltaV [m/s]	Duration [sec]
First Propagation		-	2908
Burn1	HP Man	$3.464 \cdot 10^{-3}$	1.960
	To Target	-	3756
	ZeroRelVel	$5.952 \cdot 10^{-3}$	-
	free flight	-	1493
Burn2	HP_Man	$41.05 \cdot 10^{-3}$	23.22
	To Target	-	544.2
	ZeroRelVel	$24.12 \cdot 10^{-3}$	-
	free flight	-	3499
Burn3	HP_Man	$11.63 \cdot 10^{-3}$	6.577
	To Target	-	1722
	ZeroRelVel	$9.820 \cdot 10^{-3}$	-
	free flight	-	2246
Total		**0.096**	**16200**

Table 4.3: deltaV and duration of all the HP2 segments

Segment		deltaV [m/s]	Duration [sec]
First Propagation		-	3615
Burn1	HP Man	$13.22 \cdot 10^{-3}$	7.476
	To Target	-	1681
	ZeroRelVel	$12.02 \cdot 10^{-3}$	-
	free flight	-	3999
Burn2	HP_Man	$12.90 \cdot 10^{-3}$	7.299
	To Target	-	1812
	ZeroRelVel	$10.87 \cdot 10^{-3}$	-
	free flight	-	3583
Burn3	HP_Man	$26.61 \cdot 10^{-3}$	15.05
	To Target	-	818.8
	ZeroRelVel	$17.91 \cdot 10^{-3}$	-
	free flight	-	661.4
Total		**0.094**	**16200**

5 Nominal Scenarios Analysis

In the previous Chapter, the updates to the Matlab and the STK scenario were described. After performing these modifications and enhancements, a few aspects of the Nominal Scenario were re-defined. The fact that SR's orbit changed between Phase B1 and Phase B2, made it necessary to perform the following tasks:

- Evaluate the Ground Stations (GS) visibility during the mission;
- Evaluate the illumination conditions and the GS coverage for the Final Approach;
- Define the optimal WSE;
- Estimate the required deltaV and duration of the whole mission;

5.1 Ground Station Visibility Analysis

The analysis of the ground stations has been carried out considering the following assumptions:

- The ground station network in the simulation is composed of ESTRACK stations, a set of commercial stations including some run by Tyvak and the PoliTo CubeSat Control Centre (C3). The complete list of the ground stations used is presented in Table 5.2;
- It is required a minimum elevation angle of 10 degrees;
- AzElMask was applied for all ground stations: this mask evaluates the terrain-based visibility restrictions by extending constant azimuth arrays outwards the point indicated. With this process, obstruction information is evaluated, and it is used to account for obscuration of the line of sight when computing the access;
- A minimum access duration of 3 minutes was set to consider the margin of time needed for tracking the signal and establishing a stable link with SROC;
- This simulation was carried out considering a 1-month long scenario, from the 1st of November 2024 to the 1st of December 2024;

The analysis was carried out considering the MCS of the nominal scenario; since it ends on the 13th of November, from that moment onward the state of the satellite was blocked using the Hold segment. This segment blocks the satellite in the same relative position with respect to SR it has at the end of its previous segment (in that case the HP3) until the end of the analysis. Table 5.1 reports the number of access for the whole month, the daily number of access, the average and maximum duration and the number of access lasting more than 5 minutes for all the ground stations covered by SROC.

Table 5.1: Ground Station visibility analysis

Location	Access [#/month]	Access >5 minutes [#/month]	Access [#/day]	Maximum Duration [min]	Average Duration [min]
Kourou_Station	363	277	12	6.624	5.693
Malindi_station_STDN_KENS	438	339	14	6.638	5.778
South_suwalesi_LAPAN	438	236	14	5.888	4.833
SriLanka_Leasfpace	308	234	10	6.611	5.651

Table 5.2: Ground Stations list

Location	ESTRACK	OWNER	FREQUENCY
Turin	No	Polito	S, UHF
AbuDhabi_Tyvak	No	Tyvak	S
Awaruna_LeafSpace	No	Leafspace	S, UHF
Bardufoss_Tyvak	No	Tyvak	UHF
Cebreros_DSA_2	Yes	ESA	Ka, K, X
Dongara_Station_AUWA01_STDN_USPS	No	Universal Space Network	S, Ku, X, Ku
DSS_26_Goldstone_STDN_D26D	No	NASA	
Esrange_Station_ESTC_STDN_KU2S	No	SSC	S, X (UHF downlink)
Esrange_Station_SSC-CNES	No	SSC	S, X, (UHF downlink)
ESRIN	No	ESA	
Kerguelen_Island_STDN_KGLQ			
Kourou_Station	Yes	ESA	
Malargue_DSA_3	Yes	ESA	Ka, K, X
Malindi_Station_STDN_KENS	Yes	ESA	X
Masuda_USB_F2			
New_Norcia_DSA_1	Yes	ESA	S, X
Orbcomm_Hartebeesthoek_A	No	SANSA	S, C, Ext C, X, Ku, DBS, Ka
Petaluma_Tyvak	No	Tyvak	S
Peterborough_Tyvak	No	Tyvak	S
Poker_Flat_Station_PF1_STDN_DX2S	No	NASA	S, C
Redu_Station	Yes	ESA	L, X X Ku, Ka
RiodeJaneiro_Telespazio	No	Telespazio	L, S, C, Ku, Ka
SanDiego_Tyvak	No	Tyvak	UHF
Santa_Maria_Station	Yes	ESA/leafspace	S ,X
Santiago_Leolut	No	Ssc	S, C, Ka
Shetland_Islands_LeafSpace	No	Leafspace	S, X, UHF
South_Point_Station_USHI01_STDN_USHS	No	Ssc	S, X, Ku
south_sulawesi__LAPAN	No	lapan	S
SriLanka_LeafSpace	No	Leafspace	S ,X
Svalbard_STDN_S22S	No	Kongsberg Satellite Services	C, L,S,X and
TrollSat_Ground_Station	No	Kongsberg Satellite Services	S, X, C (uplink)
Usuda	No	JAXA	S, X
Villafranca_VIL-4	No	ESA	S, C
SMILE Lab	Yes	ESA	S, UHF

The new baseline orbit presents a total number of access and average durations slightly better than the baseline orbit of phase B1, however, as shown in Figure 5.1, most of the time SROC cannot communicate with the Ground and the visibility interval with the longest duration is only 6.638 minutes.

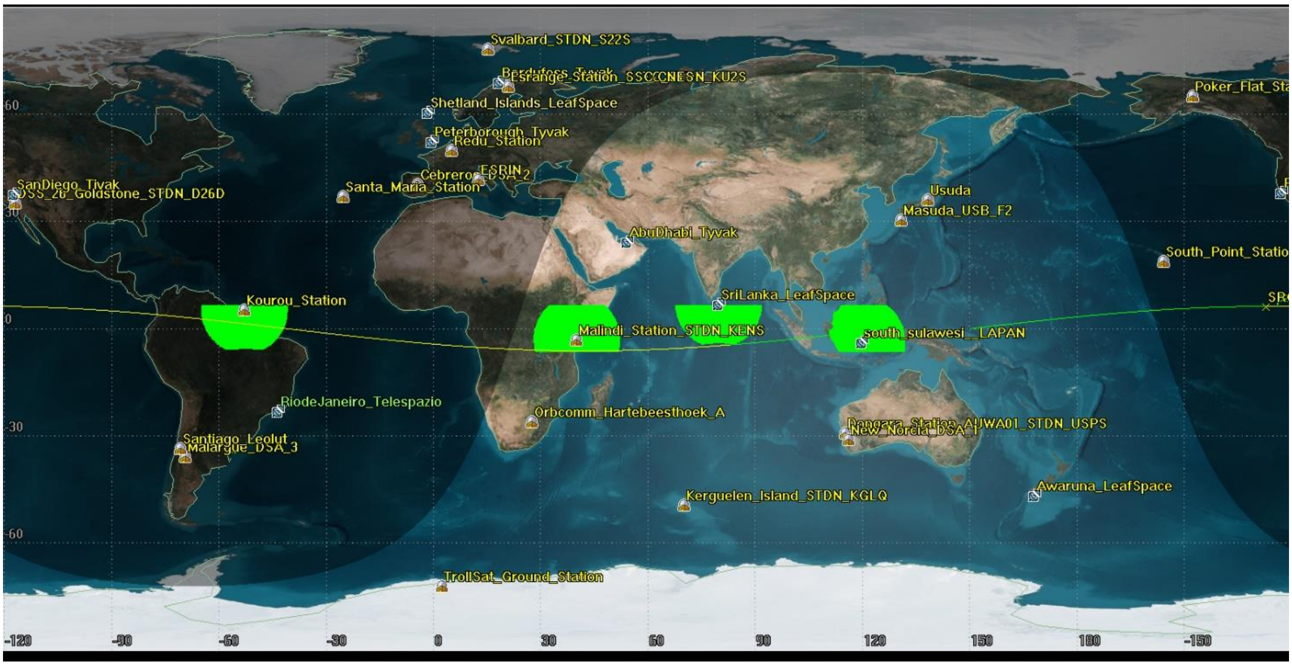

Figure 5.1: GS take over

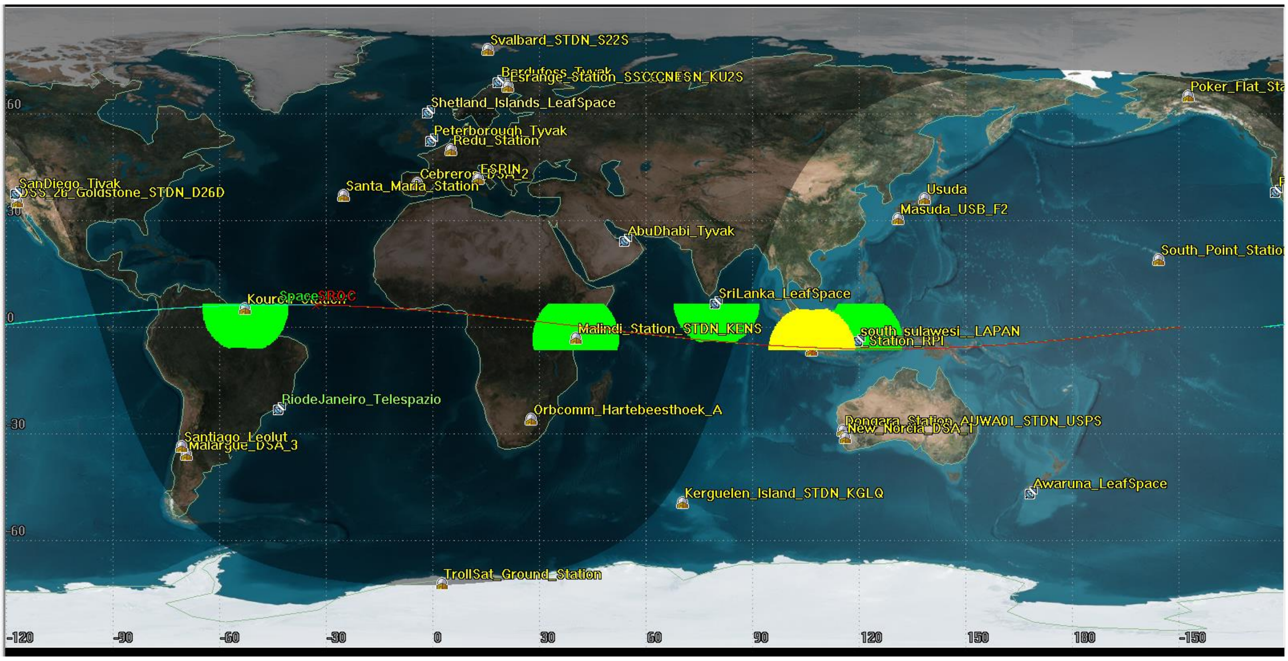

Figure 5.2: GS take over with an additional ground station

This global coverage could not be adequate for the mission since it may not have communication windows long enough. This property is not particularly crucial for the downlink of mission data during the free flight, but it could be fundamental during the commissioning or the final approach, where a combination of proper illumination conditions and ground station coverage is required (see Section 5.2 for more information). The duration of the longest access window could be increased by considering more already existing ground stations or by creating ad-hoc ground stations. Figure 5.2 shows the GS coverage if another ground station (LAPAN's Rumpin Ground Station) is added. By doing so, the longest access changes from 6.611 minutes to 10.309 minutes thanks to the uninterrupted passage from the additional ground station to the south_sulawesi_LAPAN ground station. However, the possibility to create a significantly longer uninterrupted coverage window could not be feasible, since a considerable portion of SROC's ground track is above the sea and not land. Moreover, considering or even building new ground stations would increase the cost and the complexity of the project.

An alternative solution could be to use a GEO satellite constellation to perform data relay of SROC's data to the Ground. In this scenario, SROC would need a transponder that uses the GEO satellites connectivity; there are already TRL9 COTS available for this application, such as AddValue's IDRS system [33], which relies on Inmarsat GEO satellites. Its mass (1 kg) and volume (125x96x70 mm^3) are compatible with SROC's remaining mass and volume margins [20]. The real-time connection provided by this service presents the following properties:

- Network availability higher than 99.5%;
- Link budget availability higher than 99%;
- IP session continuity during rapid GEO satellite spot beam handovers;
- Latency: 0.5 – 1.5 seconds end to end;
- Capability of supporting data rates in excess of 200 Kbps for SROC's orbit;

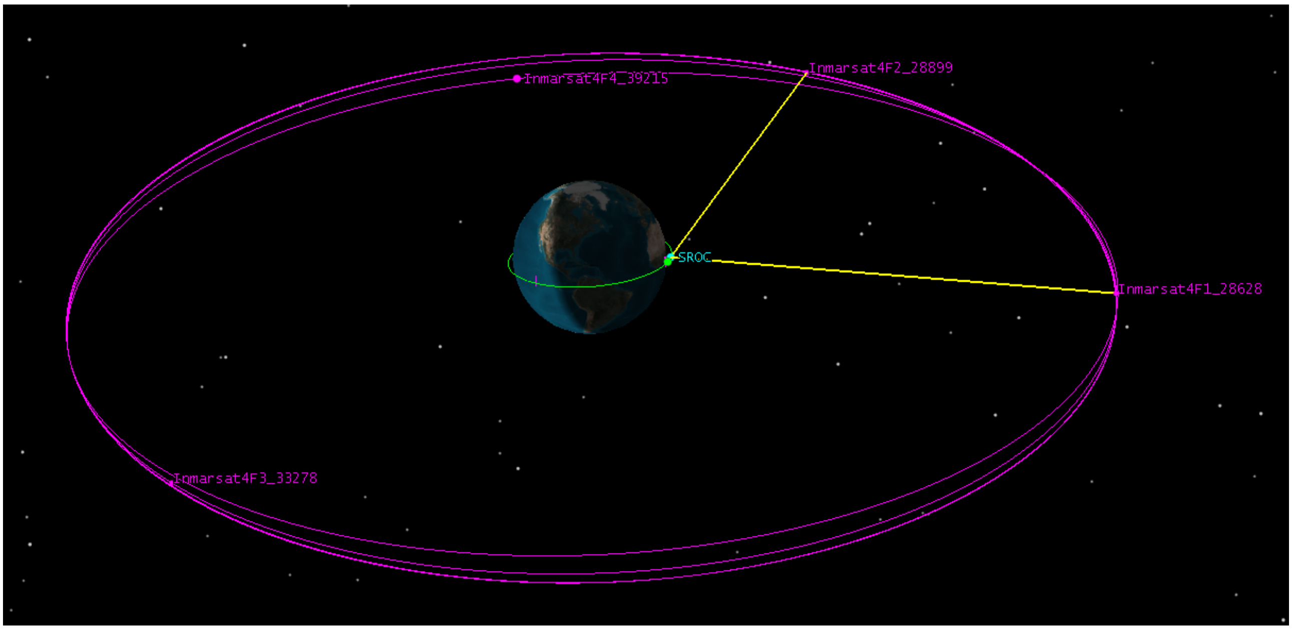

Figure 5.3: INMARSAT -4 GEO constellation

Figure 5.3 shows the INMARSAT-4 GEO constellation that is the one used by IDRS. A 1-month access analysis between SROC and the constellation was performed and showed that the satellite is always in line of sight with at least one element of the constellation. In conclusion, is this solution was confirmed to be feasible also from other points of view such as the cost, it would be the best way to guarantee an uninterrupted communication window with SROC.

5.2 Final Approach Analysis

Although the Final Approach and Docking are not evaluated in the STK scenario, the conditions to ensure their successful outcome have been evaluated in SKT. As stated by the requirement SROC-MIS-111: "The angle between the Sun Vector and the docking axis shall be less than 60 (TBC) deg for the final approach and docking". This angle, also called Line of Sight (LOS) angle in STK, was evaluated from the end of the HP3 to the end of the analysis time (1st Dec 2024). Figure 5.4 shows the LOS as a function of the time for the first 24 hours after the end of HP3 while Figure 5.5 zooms on one of the many suitable illumination intervals when the LOS constrain is respected; specifically, the interval in the image lasts 35 minutes.

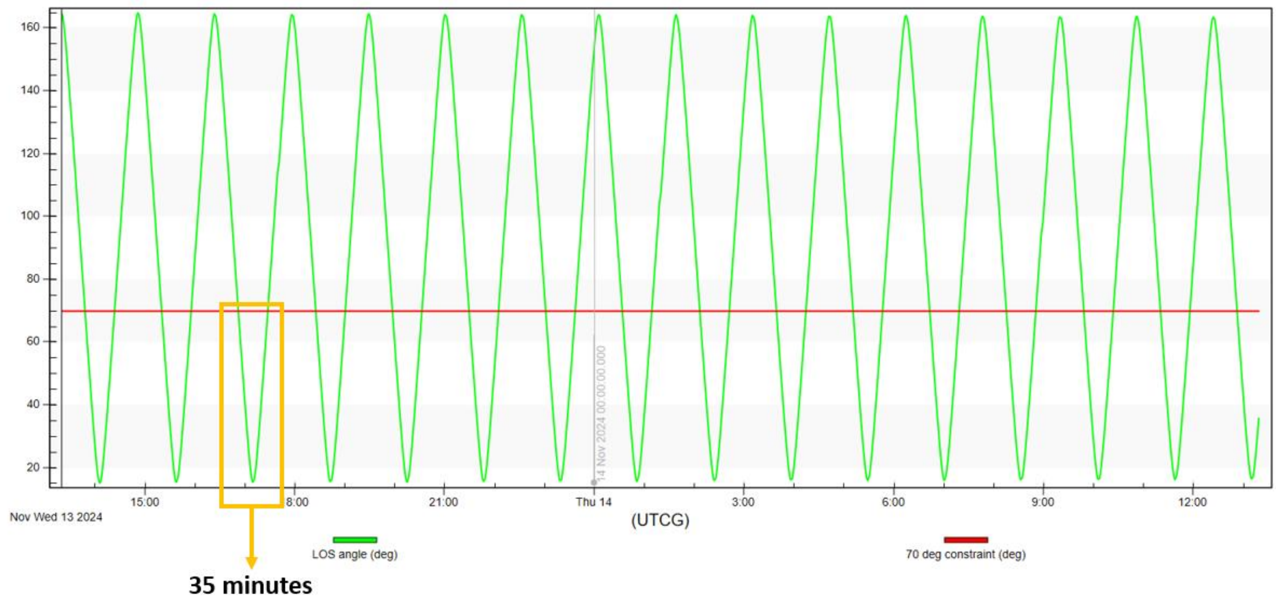

Figure 5.4: LOS angle during the first 24 after the HP3 end

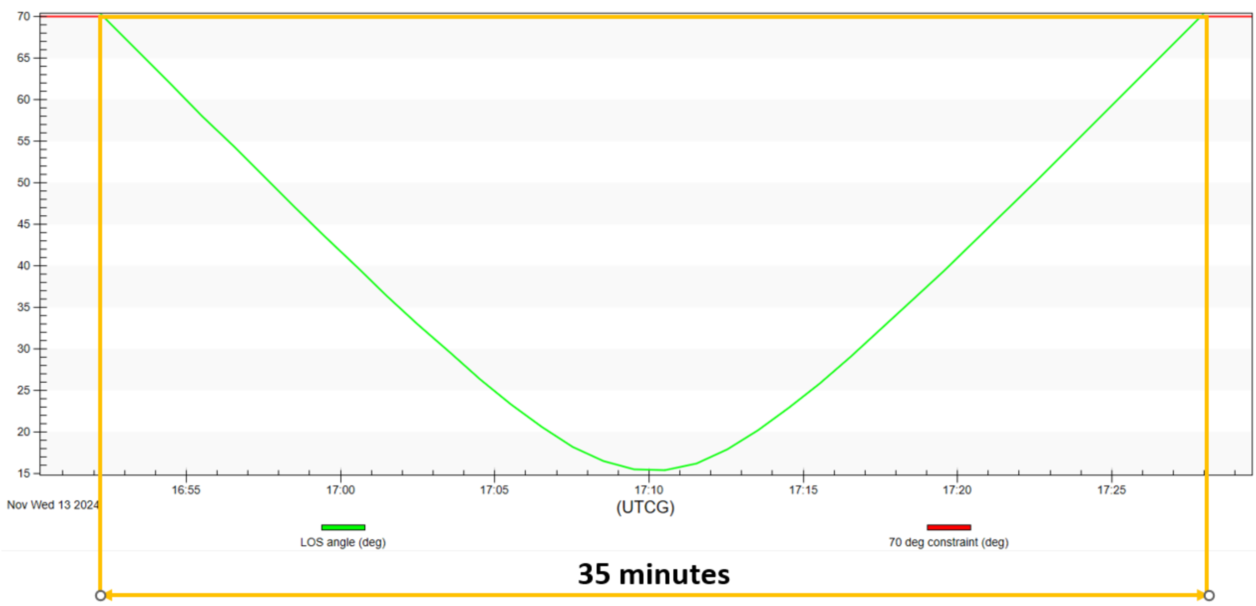

Figure 5.5: Zoom on one acceptable interval

The evolution of the LOS angle for the next 29 days is the same as the one reported in Figure 5.4, so from the end of the HP3 onwards, there are many windows with an acceptable illumination (approximately 15 per day). The next crucial step is to synchronize the start of the Final Approach with a good illumination and ground station visibility window. Figure 5.6 shows, from top to bottom: the single ground stations visible from SROC, all the intervals when at least one of them is visible (the brown line referred to as "SROC"), the intervals with good illumination, and the windows with both good illumination and GS visibility. If the windows shorter than 3 minutes are discarded from this last set of intervals, the following results are obtained:

- Min Duration: 201.7 seconds;
- Max Duration: 398.3 seconds;
- Mean Duration: 310 seconds;
- Number of Intervals: 13;

In conclusion, the 35 minutes window is reduced to an approximately 6.6-minutes window.

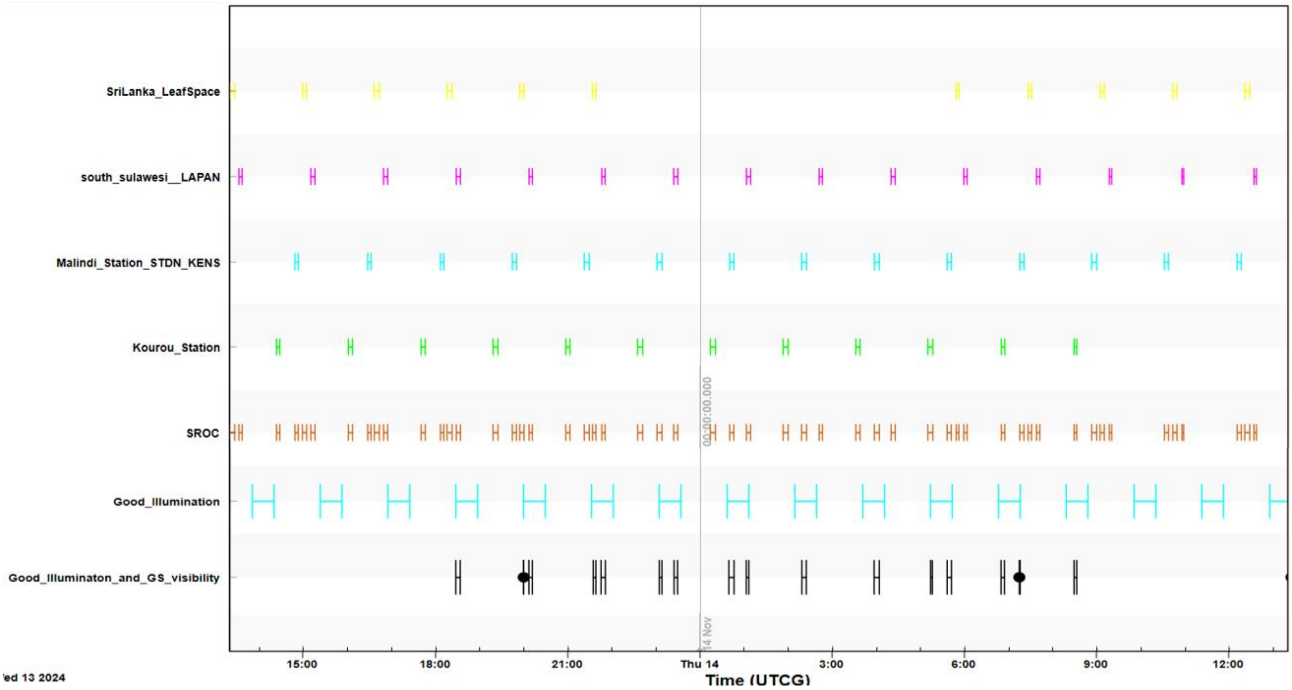

Figure 5.6: Illumination and GS visibility analysis

Of course, in case the data relay using GEO satellites was considered instead of the direct communication with the ground station, the suitable intervals to perform the Final Approach would coincide with the ones with an acceptable LOS angle, since it would always be possible to communicate with SROC through a GEO satellites constellation.

5.3 WSE Design of Experiment

As mentioned before, during the observation phase SROC will perform observation of SR in its proximity. During this subphase, SROC will fly in a passively safe trajectory called Walking Safety Ellipse (WSE), whose geometry depends on the insertion's relative position and velocity. A Matlab function evaluates these parameters to generate a WSE which satisfies a set of user-defined constraints. Once the WSE insertion position has been defined, it is possible to set the OPA target sequence to get there from the HP2, while the desired insertion velocity becomes the desired result of the WSE insertion target sequence.

5.3.1 Ideal Safety Ellipse

Before showing the results of the WSE DoE, the geometry of the ideal Safety Ellipse is described, to give some context behind the set of constraints used by the Matlab function to define the WSE. A Safety Ellipse is an out-of-plane elliptical period relative trajectory around the target spacecraft such that the chaser (SROC) never crosses the primary spacecraft (SR) velocity vector. Since the drift of the two spacecraft would not result in a collision, the trajectory is considered passively safe. Figure 5.7 shows several geometrical features of the Safety Ellipse:

- The X_E and Y_E axes lay on the Safety Ellipse Plane. The first axis is parallel to the major axis of the ellipse and points towards negative CrossTrack; Y_E is perpendicular to X_E and it points toward the positive Radial direction. SR's centre coincides with the centre of the ellipse;
- χ (polar angle) is the angle between SROC distance from the ellipse's origin and the X_E axis; it is equal to zero at the insertion with the $Y_{RIC}Z_{RIC}$ plane and it is positive counter-clockwise.
- a_{SE} and b_{SE} are respectively the semi-major and semi-minor axes of the ellipse;

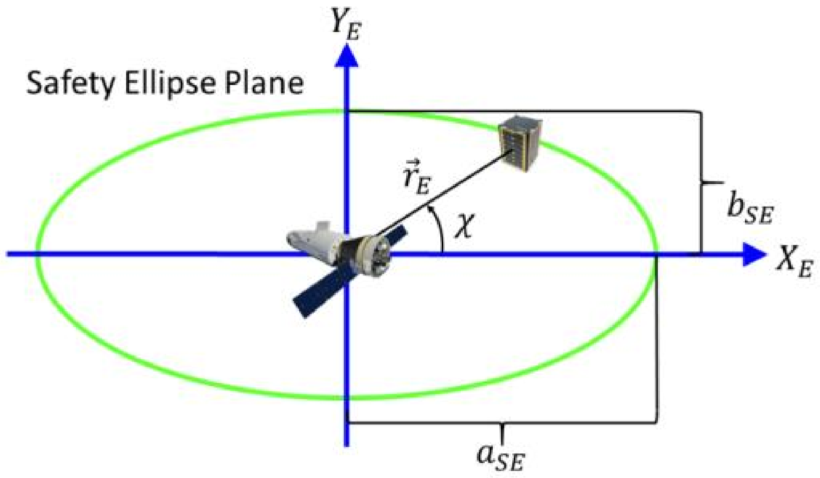

Figure 5.7: Safety Ellipse Plane

It is possible to describe SROC's position in the Safety Ellipse reference frame as a function of the polar angle:

$$\begin{bmatrix} a_{SE}\cos(\chi) \\ b_{SE}\cos(\chi) \\ 0 \end{bmatrix}$$

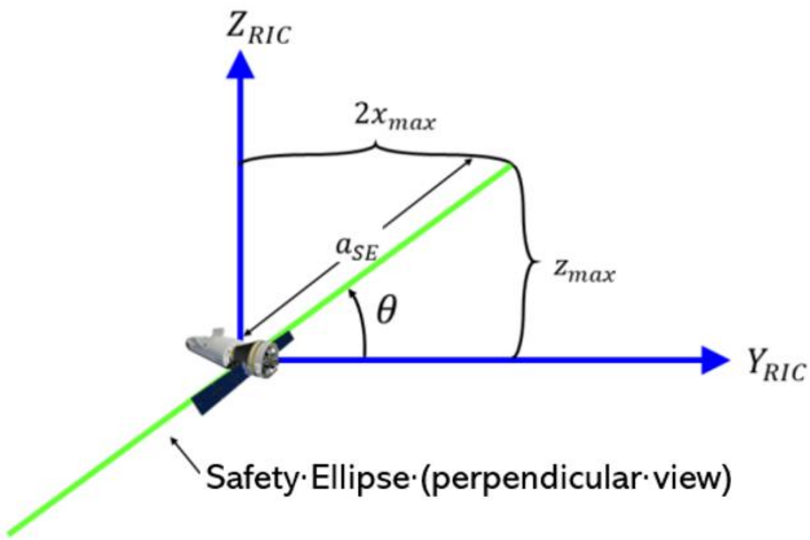

Figure 5.8: View perpendicular to the Safety Ellipse

Figure 5.8 shows another fundamental geometrical parameter: the inclination angle θ between the ellipse plane and the $X_{RIC}Y_{RIC}$ plane. Moreover, it also shows the maximum radial distance (z_{max}) and the maximum CrossTrack distance ($2x_{max}$). These two values can be evaluated using the following equations:

$$2x_{max} = a_{SE} \cdot \cos(\theta)$$
$$z_{max} = a_{SE} \cdot \sin(\theta)$$

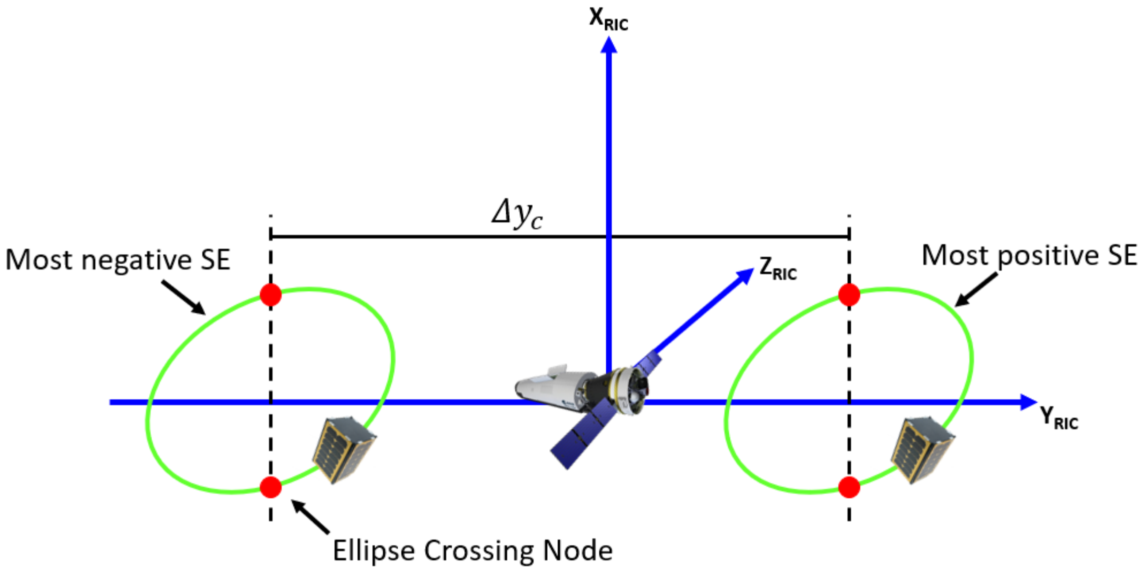

Figure 5.9: Walking Safety Ellipse offset

This safety ellipse, however, does not correspond to the actual trajectory of SROC: since the satellite undergoes the effects of the external disturbances and it does not perform any manoeuvre after the WSE to contrast them, its trajectory is modified. The most evident effect is the motion along the positive InTrack axis due to the atmospheric drag. For this reason, it is obtained a Walking Safety Ellipse (where "walking" refers to the translation along the InTrack axis), which is characterized by the InTrack offset Δy_c. This parameter is the distance between the crossing nodes of two ellipses which are the points of the ellipse with a null CrossTrack (Figure 5.9). The two reference ellipses that define Δy_c are the most positive one (which is the one with the SE centre with the highest InTrack value) and the most negative one (which is the one with the SE centre with the lowest InTrack value).

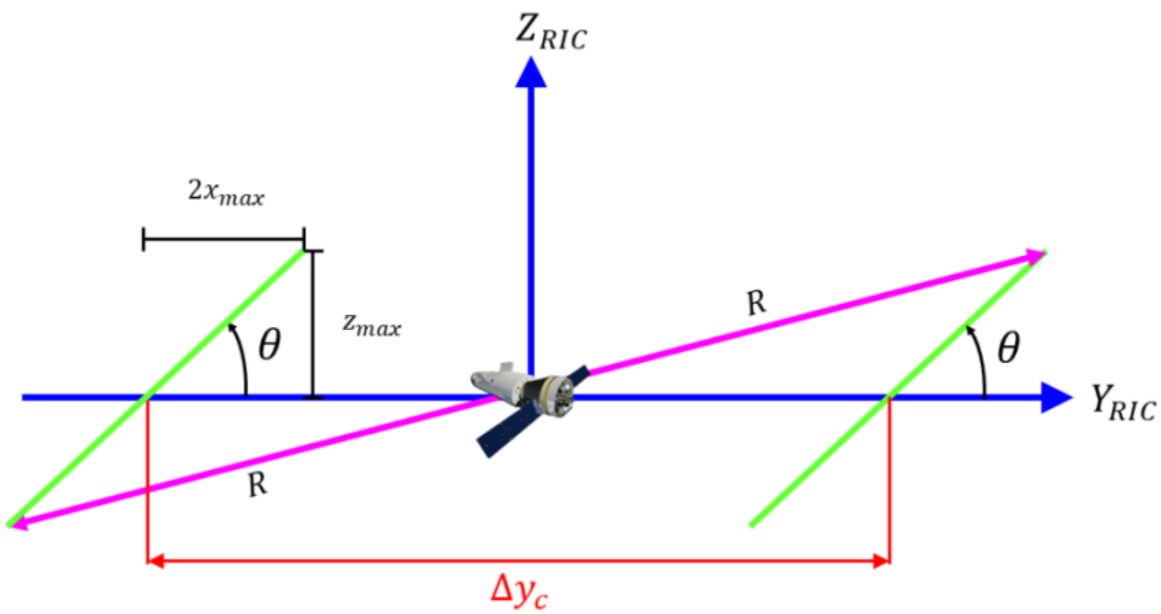

Figure 5.10: Walking Safety Ellipse geometry

Figure 5.10 shows the relationship between the safety ellipse offset Δy_c and R, which is the maximum range between SROC and SR. These two parameters are related by the following equation:

$$R = \frac{\Delta y_c}{2} + 2x_{max}$$

The equations used to approximate in the RIC reference frame the SROC motion along the WSE are shown below:

$$x(\chi) = x_{max}\, sin(\chi) - \frac{2\dot{y}_c}{3n}$$
$$y(\chi) = 2x_{max}\, cos(\chi) + \frac{\dot{y}_c\left(\chi - \frac{\pi}{2}\right)}{n} + y_c$$
$$z(\chi) = z_{max}\, cos(\chi)$$
$$\dot{x}(\chi) = x_{max} n\, cos(\chi)$$
$$\dot{y}(\chi) = -2x_{max} n\, sin(\chi) - \dot{y}_c$$
$$\dot{z}(\chi) = -z_{max} n\, sin(\chi)$$

Where n is the mean motion of the primary spacecraft. These equations show that SROC motion depends on x_{max}, z_{max}, χ and two additional parameters:

- y_c: it is the InTrack distance of the crossing nodes of the initial SE;
- $\dot{y}_c$: it is the initial velocity of the SE along the InTrack direction;

The final parameter on which the WSE depends is the desired duration of the inspection. The way that the Matlab function defines the WSE is the following:

- x_{max} and z_{max} are defined by the user; they must be an adequate compromise between the maximum payload range and the constraint to not enter SR's KOZ;
- The duration of the inspection and Δy_c are also user-defined;
- y_c and $\dot{y}_c$ are evaluated by the Matlab function through an iterative process until a valid WSE is founded. To be considered valid, a WSE must respect the following constraints:
    - $y_{c,max} < \frac{\Delta y_c}{2}$ and $y_{c,min} > -\frac{\Delta y_c}{2}$;
    - SROC trajectory never enters the KOZ during the OPA rendezvous, the observation and the free flight;
- Once y_c and $\dot{y}_c$ have been calculated, it is possible to define the insertion point of the WSE: this position is set as the desired result of the OPA Rendezvous target sequence. The desired velocity in RIC components is also evaluated and set as the desired result for the WSE Insertion target sequence;

All the above demonstration is just an approximation for the design of a WSE useful for the SR observation, but it was necessary due to the high complexity of the motion and the disturbances. Further studies and improvements shall be implemented to increase accuracy and evaluate the effects of the disturbances on the WSE. This could be done by analytically evaluating the acceleration caused by the disturbances and therefore calculating the actual trajectory of SROC during the motion. Another option may be using STK's pre-built proximity operations manoeuvres. The analysis presented in the next sub-section was still useful to define a reasonable deltaV guess for both the OPA and the WSE Insertion, as well as giving a first approximation of the illumination condition of SR during the observation and a solid analysis of the ground station visibility during the free flight phase after the observation.

5.3.2 DoE Results

A DoE was conducted to select an optimal WSE. But to decide which WSE is the best, it was necessary to define a set of constraints or parameters to minimize/maximize:

- **Payload maximum range**: this value was temporarily set to 200 m in the previous study. However, since then SR's KOZ has been updated to 200 m, thus making it impossible to respect both constraints. Since the work performed for this thesis concerns Phase B2, an updated value for the payload maximum range was not available. For this reason, the intervals during which SR is visible by the payload have been calculated considering three possible maximum ranges: 250-300-350 m (although the values most close to the actual payload requirements should be the first one);

- **Minimum actual observation time**: how much time during the observation SR in the payload range; since a minimum value had not been defined, it was selected the WSE with the highest actual observation time;
- **Minimum duration of the single observation**: although the total time may be enough, it could be obtained by considering periods too short to produce useful data. However, from the analysis of the WSEs from the DoE, it was noticed that the shortest interval was lasting 165 seconds, which was considered more than sufficient for the payload to take pictures of SR;

As mentioned before, the Matlab function requires the user to define the following parameters: duration of the observation, Δy_c, x_{max} and z_{max}. Several combinations of sets of these values were tested, and their results were evaluated in terms of: deltaV required for both the OPA rendezvous and the WSE Insertion, duration of the FreeFlight and duration of the actual observation (reported both in hours and percentage of the whole observation segment).

Every set of values for each variable was selected considering the ones used for the WSE DoE in phase B1, which selected a WSE with:

- $\Delta y_c = 400\ m$;
- $Duration = 6\ hr$;
- $x_{max} = 150\ m$;
- $z_{max} = 150\ m$;

For this analysis, higher x_{max} and z_{max} were considered because the radio of KOZ was increased to 200 m; since the phase B1 analysis stated that only for a small percentage of the observation SR was in the payload range, a higher duration was considered to increase the total actual observation time.

Table 5.3: WSE DoE DeltaVs

		x_{max}, z_{max} [m]		
Δy_c [m]	**Duration [hr]**	**250-200**	**250-250**	**DeltaV [m/s]**
300	6	0.485	0.559	
	8	0.485	0.558	
400	6	0.496	0.5689	
	8	0.491	0.549	
600	6	0.459	0.515	
	8	0.459	0.515	

Table 5.4: WSE DoE FreeFlight Duration

		x_{max}, z_{max} [m]		
Δy_c [m]	**Duration [hr]**	**250-200**	**250-250**	**FreeFlight Duration [hr]**
300	6	8.746	8.745	
	8	6.777	6.776	
400	6	8.902	8.898	
	8	6.754	4.96	
600	6	10.062	10.061	
	8	8.062	8.061	

Table 5.3 shows the deltaV of each WSE evaluated and highlights the two solutions with the lowest deltaV in green. The only difference between the two solutions is the duration of the observation phase, while the geometrical parameters of the WSEs are the same. The reason why they have the same deltaV is that they

are the same WSE: they start at the same relative position from SR and with the same initial relative velocity, but the first one just ends the observation phase two hours before. This is also confirmed by the duration of their free flight segments (Table 5.4): since they are the same WSE, they take the same total time to perform the observation and propagate during the free flight to the maximum range stopping condition. In fact, the total duration of both solutions is 18.062 hours, the only thing that separates them is the decision to stop performing observations and start the downlink of the data.

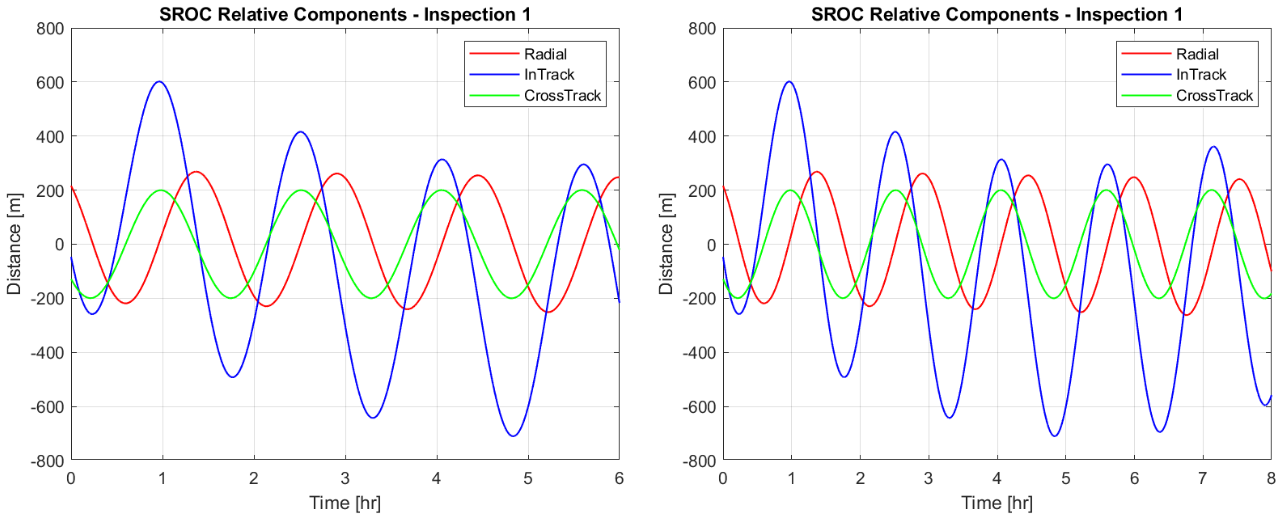

Figure 5.11:RIC components as functions of the time for the two highlighted WSE (6 hr on the left and 8 hours on the right)

Table 5.5: Total duration of the actual observation [hr]

			x_{max}, z_{max} [m]		
Δy_c [m]	**Max Range [m]**	**Duration [hr]**	**250-200**	**250-250**	**Actual Observation Duration [hr]**
300	250	6	0.3792	0.2622	
		8	0.6176	0.4696	
	300	6	1.5036	1.377	
		8	2.1088	1.9352	
	350	6	2.2974	2.1144	
		8	3.2096	2.9472	
400	250	6	0.4584	0.3138	
		8	0.5136	0.408	
	300	6	1.641	1.431	
		8	2.0344	1.9464	
	350	6	2.664	2.289	
		8	3.0944	2.7936	
600	250	6	0.555	0.3552	
		8	0.6472	0.4384	
	300	6	1.7712	1.5312	
		8	2.1424	1.8616	
	350	6	2.7888	2.4306	
		8	3.3752	2.9656	

Table 5.6: Total duration of the actual observation [%]

Δy_c [m]	Max Range [m]	Duration [hr]	x_{max}, z_{max} [m] 250-200	x_{max}, z_{max} [m] 250-250	
300	250	6	6.32	4.37	Actual Observation Duration [%]
		8	7.72	5.87	
	300	6	25.06	22.95	
		8	26.36	24.19	
	350	6	38.29	35.24	
		8	40.12	36.84	
400	250	6	7.64	5.23	
		8	6.42	5.1	
	300	6	27.35	23.85	
		8	25.43	24.33	
	350	6	44.4	38.15	
		8	38.68	34.92	
600	250	6	9.25	5.92	
		8	8.09	5.48	
	300	6	29.52	25.52	
		8	26.78	23.27	
	350	6	46.48	40.51	
		8	42.19	37.07	

For each solution the evolution of the RIC components and the range as a function of the time were saved and graphed. Figure 5.11 shows the RIC components of the two highlighted WSE: it is possible to see that for the first 6 hours, they have the same components. Table 5.5 and Table 5.6 show the actual observation duration for each of the three maximum payload ranges considered and highlight the highest. As expected, the actual duration of the observation is higher for the WSE lasting 8 hours than for the ones lasting 6 hours; however, the percentage of the actual observation duration with respect to the total duration is often lower for the WSE lasting 8 hours. This can be explained by looking at the trajectory of SROC during these WSEs: for example, Figure 5.11 shows that SROC starts the observation with a slightly negative InTrack, then the trajectory moves to the more negative InTrack until, because of the effect of the atmospheric drag, SROC starts moving towards positive InTrack. For a portion of the 8-hour case, SROC is in the most negative SE, therefore it has a very short interval during which it can observe SR.

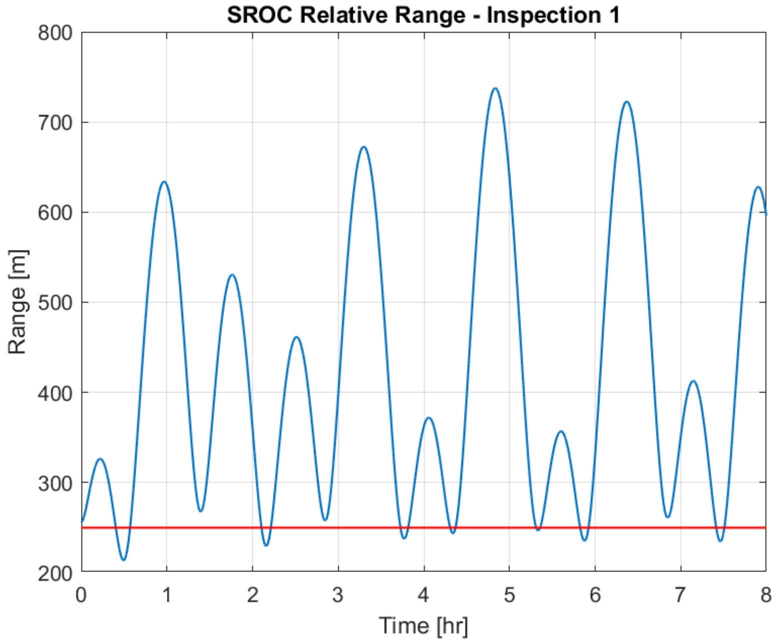

Figure 5.12: Range as function of the time for the WSE with the longest actual observation duration

Figure 5.12 shows the range as a function of the time for the solution highlighted in green in Table 5.5. With this WSE, SROC can take pictures of SR during seven intervals, with the shortest one lasting 162 seconds; it is noticed that, in accordance with what has been said in the last paragraph, by increasing the length of the observation from 6 hr to 8 hr, only a small interval (approximately 6 minutes) is added to the actual observation duration.

5.3.3 Nominal Observation Cycle

From the results presented in the last sub-section, the WSE with the lowest deltaV and also the highest actual observation duration is obtained by giving the following inputs to the Matlab function:

- $\Delta y_c = 600\ m$;
- $x_{max} = 250\ m$;
- $z_{max} = 200\ m$;
- $Duration = 8\ hours$;

The visibility of SR during the observation was analysed using SRK's tools Analysis Workbench and Access. A sensor object was attached to SROC, and it was set to always points toward SR, thus simulating the camera(s) pointing to SR. The visibility, which STK evaluates as access, of the spacecraft from the sensor was constrained as follows:

- Maximum range between SROC and SR: 250 m;
- LOS illumination angle less than 60 deg;
- SR is in sunlight;

Figure 5.13 shows the intervals when each of these constraints is satisfied. The intervals where the range is less than 250 m (in blue) are the same shown in Figure 5.12, while SR is in sunlight for intervals (in green) lasting 56 minutes divided by 36 minutes long umbra periods. The LOS illumination angle condition is always respected (in red): as shown in Figure 5.14 during the observation the maximum angle reached is 54.29 degrees.

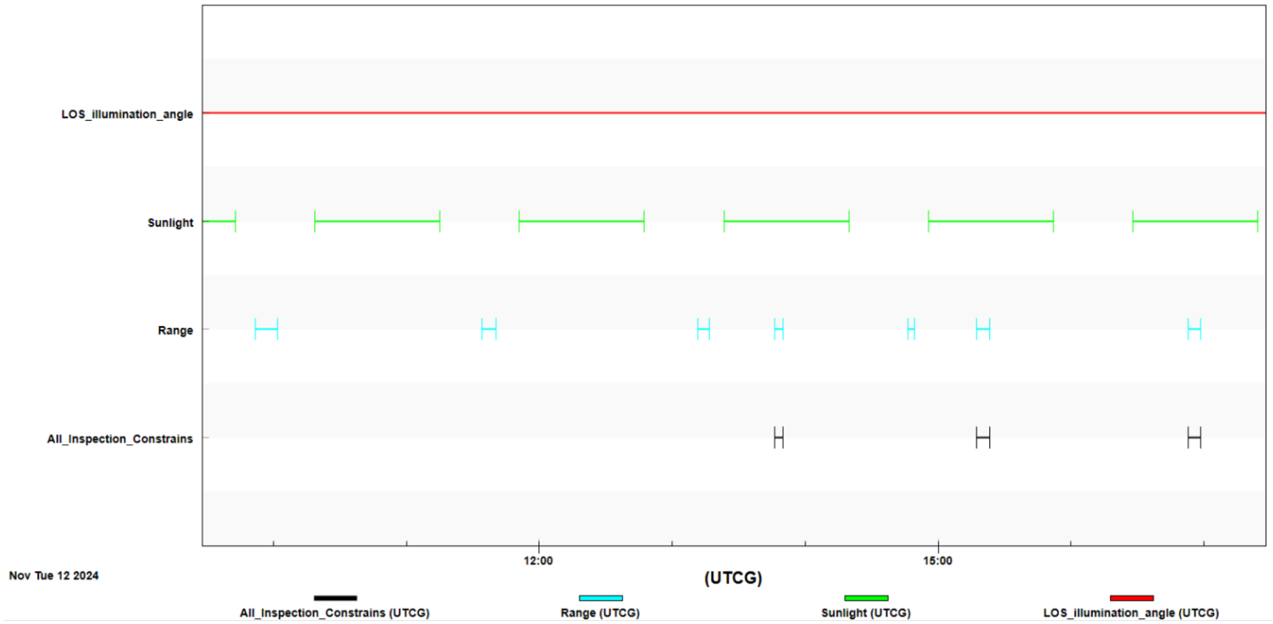

Figure 5.13: Satisfaction interval for every constraint

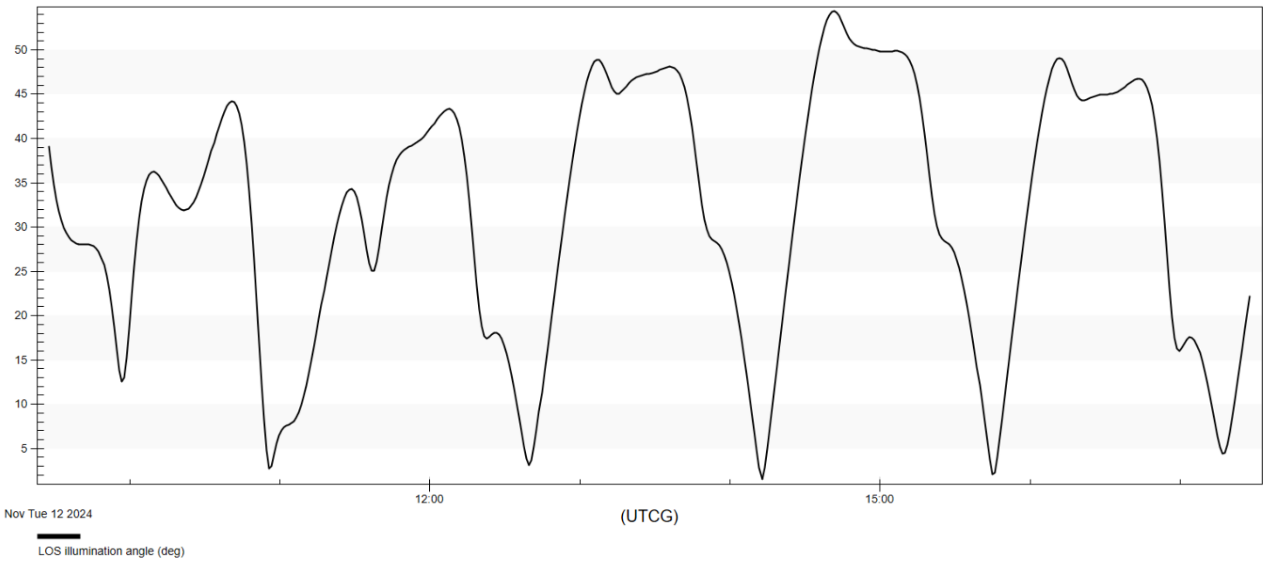

Figure 5.14: LOS illumination angle during the observation phase

By applying all the constraints, the intervals suitable to perform observation of SR are only three and they last for a total of 915 seconds (Table 5.7). Although the length of each interval should be enough to take pictures of SR, the total duration of the actual observation may not be enough to perform a satisfying observation of SR. This problem could be solved by setting the duration of the HP2 in a way that makes the moments when SR is in the payload range with the sunlight interval. For example, Figure 5.15 and Table 5.8 shows how the total observation time increases when the HP is performed after 4.3 hours; however, due to the complex nature of the motion it is difficult to predict how performing the observation at a different moment will affect the LOS illumination angle. As said at the beginning the DoE description, for further analyses it will be necessary to increase the accuracy of the WSE definition by evaluating the effects of the disturbances on the WSE. By doing so, it should be obtained a more "stable" WSE, such that it varies very little from the input geometrical parameters. Moreover, with a more stable function for the WSE definition, it could be possible to reduce the x_{max} and z_{max} without risking the intersection of the WSE with the KOZ.

Table 5.7: Suitable observation intervals

Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
1	12 Nov 2024 13:46:10.830	12 Nov 2024 13:49:54.735	223.905
2	12 Nov 2024 15:17:18.207	12 Nov 2024 15:23:09.336	351.128
3	12 Nov 2024 16:52:39.080	12 Nov 2024 16:58:18.928	339.848
Mean Duration	-	-	304.960
Total Duration	-	-	914.881

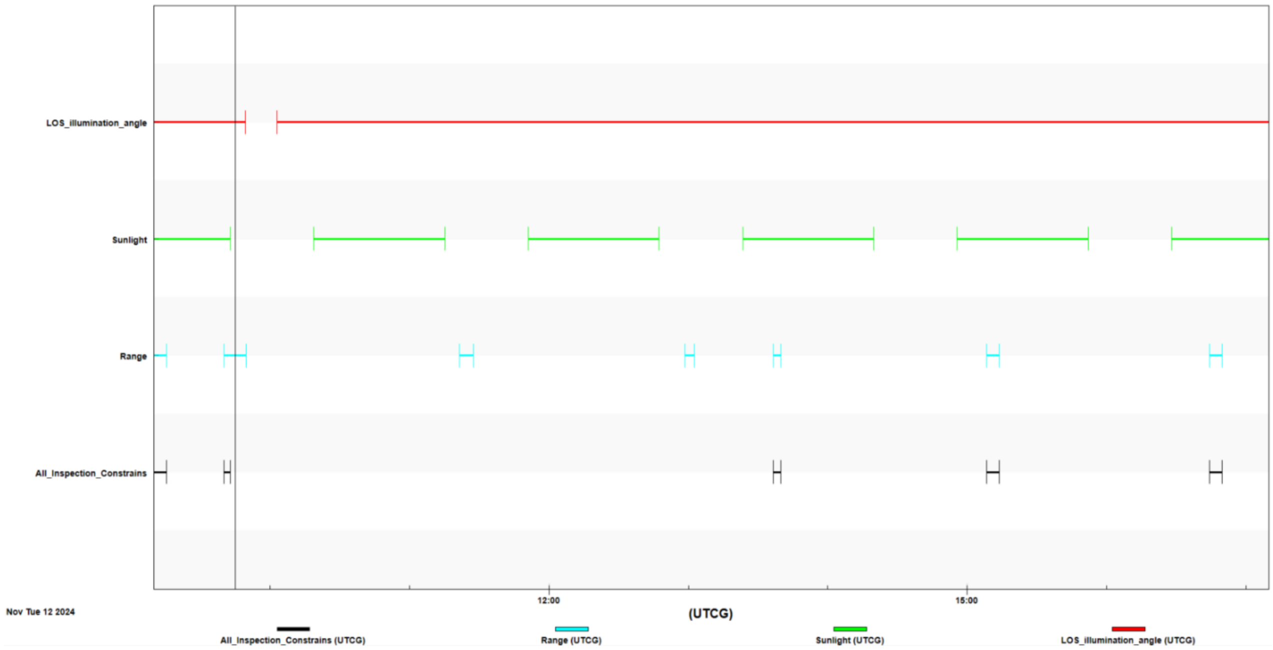

Figure 5.15: Satisfaction interval for every constraint with a different HP2 duration

Table 5.8: Suitable observation intervals with a different HP2 duration

Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
1	12 Nov 2024 09:09:43.225	12 Nov 2024 09:15:12.745	329.520
2	12 Nov 2024 09:40:03.117	12 Nov 2024 09:42:45.466	162.349
3	12 Nov 2024 13:36:29.278	12 Nov 2024 13:39:48.232	198.955
4	12 Nov 2024 15:08:27.998	12 Nov 2024 15:13:44.707	316.709
5	12 Nov 2024 16:44:11.199	12 Nov 2024 16:49:49.577	338.378
Mean Duration	-	-	269.182
Total Duration	-	-	1345.911

Finally, the GS coverage during the free flight was evaluated, to estimate how long SROC has access to the ground station network to perform the Downlink of the mission data. Figure 5.16 shows the access during this phase, while Table 5.9 resume summarizes the results of the analysis. A total time of 1.774 hours should be enough to downlink the mission data to the ground stations.

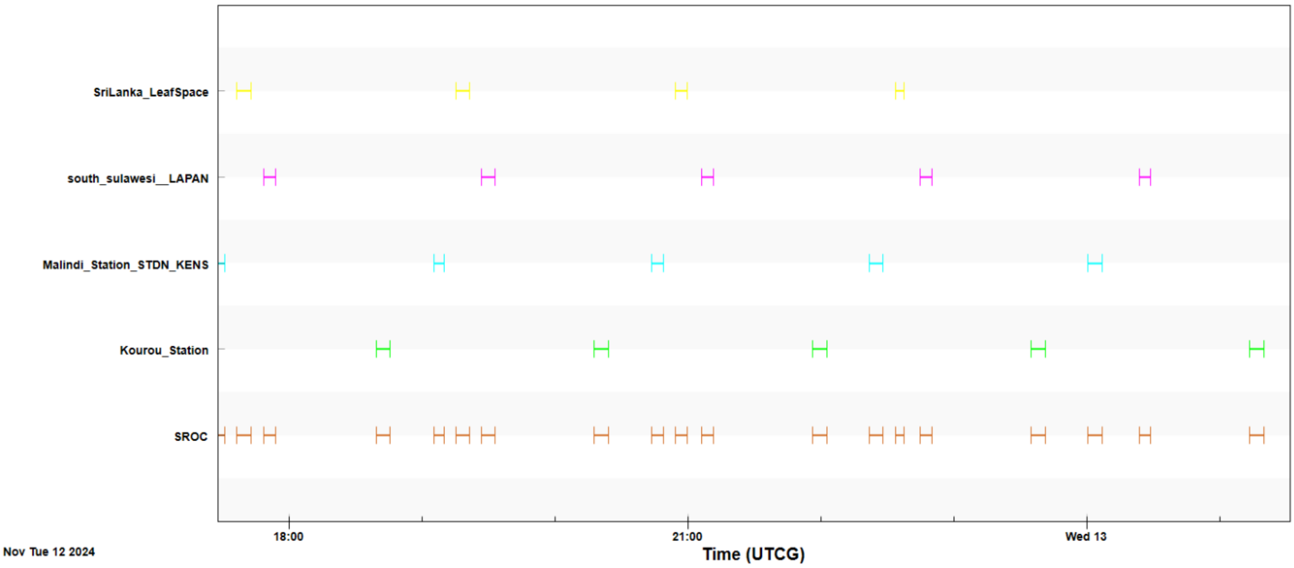

Figure 5.16: Ground station access during the free flight (the SROC line refers to the total access)

Table 5.9: Summary of the access analysis during the free flight

Results	Value
Minimum Duration	196 sec
Maximum Duration	394 sec
Mean Duration	336 sec
Total Duration	6388 sec
Percentage of the free flight	29.28%

Since the ground station coverage seems more than enough to downlink the mission data, another option is to increase the duration of the inspection phase in spite of the duration of the free flight. In fact, there is a portion at the beginning of the free flight when SROC still respects all the observation constraints: to consider this interval the observation phase was increased from 28800 seconds to 39737 seconds. Figure 5.17 and Table 5.10 show a great increase in the total suitable observation interval, while Table 5.11 shows that there is still a considerable amount of time to downlink the mission data (1.036 hours).

Table 5.10: Suitable observation intervals with a longer observation phase

Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
1	12 Nov 2024 13:46:10.830	12 Nov 2024 13:49:54.735	223.905
2	12 Nov 2024 15:17:18.208	12 Nov 2024 15:23:09.336	351.128
3	12 Nov 2024 16:52:39.081	12 Nov 2024 16:58:18.929	339.848
4	12 Nov 2024 18:30:54.677	12 Nov 2024 18:36:12.639	317.961
5	12 Nov 2024 20:12:34.937	12 Nov 2024 20:28:56.278	981.342
Mean Duration	-	-	442.837
Total Duration	-	-	2214.184

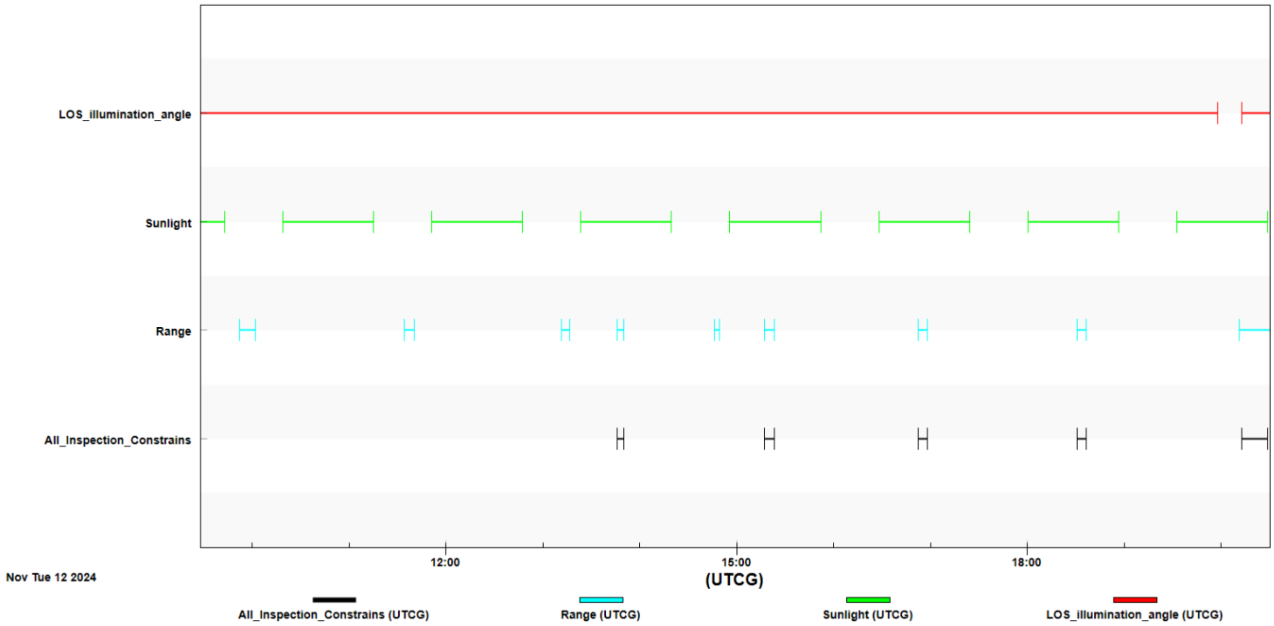

Figure 5.17: Satisfaction interval for every constraint considering a longer observation sub-phase

Table 5.11: Summary of the access analysis during the free flight considering a longer observation sub-phase

Results	Value
Minimum Duration	233 sec
Maximum Duration	394 sec
Mean Duration	339 sec
Total Duration	3731 sec
Percentage of the free flight	20.63%

Since this is a Phase B2 study, this work was carried out before the other mission actors perform new iterations on their respective work. For example, the payload was initially studied to take pictures from a maximum range of 200 m: since the KOZ was updated from 150 to 200m, a new study is required to assess the capability of the camera at higher ranges. For this reason, assessing if the actual observation intervals are enough to take a sufficient number of useful pictures is not possible at the moment; however this study constitutes a solid base to help understand the different constraints during the observation sub-phase and, in case it is changed during future project iterations, the constraints and analysis tools defined in STK for this DoE will still be useful to rapidly assess the feasibility of the new design.

In conclusion, the WSE presented at the beginning of the sub-section was selected for the nominal scenario. The two alternative options to increase the actual observation time have not been considered since, as explained before, is not possible to define minimum observation requirements, so it is not possible to select one option instead of the other. Moreover, picking one of the other two solutions would not significantly affect the results of this thesis, which are the deltaV budgets of the nominal and variant scenarios. In fact, changing the duration of the HP2 would slightly modify an already low deltaV contribution to the total deltaV budget, while the second option would not even modify it since it just postpones the switch from the observation to the free flight. To consider the many uncertainties linked to the analysis of the WSE, the margin on its deltaV was increased and the use of a second observation cycle was considered in the variant analysis (Chapter 6).

5.4 Nominal Scenarios DeltaV Budget

After updating the code and analysing some fundamental aspects of the mission, it was possible to run the complete simulation of the nominal scenarios. The results of these simulations were used to define the deltaV budget and the time budget. The DeltaV budget is fundamental to evaluate if the mission is feasible or if some of its aspect need to be modified to be less deltaV-consuming. As stated in requirement SROC-MIS-060: “The ΔV for all SROC manoeuvres shall be less than 20 (TBC) m/s including margins”, so it is vital for the mission to stay below the 20 m/s threshold. The time budget, instead, was not compiled to verify the compliance with a specific requirement, since there is none; indicatively, it was decided to set a maximum total duration of 30 days since the total duration of SR’s mission is two months. This information will be useful in the future phases of the design when it will be clearer at which moment of its orbital operations SR will deploy SROC and it will be necessary to coordinate SROC’s operations with SR’s.

Table 5.12 shows the deltaV budget for the nominal Observe scenario. The two Virtual CAMs reported in the table refer to the manoeuvres which could be performed around a virtual point during HP1. Both, as well as the SR Collision Avoidance Manoeuvre (CAM) were evaluated using different software and processes, therefore they are not part of this thesis. The Debris CAM (D CAM) was evaluated using the software DRAMA, whose analysis and results are described in Chapter 7. It is noted that the margin philosophy used in this study for the deltaV is the one recommended by ESA and reported in the ECSS [27].

Table 5.12: DeltaV budget for the nominal observe scenario

OBSERVE Nominal Scenario			
Manoeuvre	**ΔV [m/s]**	**Margin**	**ΔV [m/s]**
HP1	0.489	5%	0.513
Virtual CAM + HP1 bis	1.040	100%	2.080
Virtual CAM + HP1 ter	0.500	100%	1.000
IPA	0.485	5%	0.509
HP2Ins	1.280	5%	1.344
ZRV2	0.423	5%	0.444
HP2	0.096	5%	0.101
OPA - Cycle 1	0.266	100%	0.532
WSE Insertion - Cycle 1	0.192	100%	0.385
OPA - Cycle 2	0.000	100%	0.000
WSE Insertion - Cycle 2	0.000	100%	0.000
D CAM	0.068	100%	0.136
SR CAM	0.600	5%	0.630
ΔV TOT [m/s]	**5.439**	**ΔV TOT with margins [m/s]**	**7.674**

Table 5.13 shows the time budget for the nominal Observe Scenario. It considered all the mission phases until the end of the POP, since the successive phase (EMP) does not require any coordination with SR’s mission. It is noted that the only part of the Verification sub-phase that has been considered is the HP1 reported in the STK scenario; so the total duration and deltaV could greatly increase when the nominal Verification sub-phase will be baselined.

Table 5.13: Time budget for the nominal Observe scenario

OBSERVE Nominal Scenario			
Manoeuvre	**Duration [day]**	**Margin**	**Duration [day]**
Commissioning	5.000	5%	5.250
HP1	0.188	5%	0.197
IPA	5.760	5%	6.048
HP2Ins	0.083	5%	0.088
HP2	0.188	5%	0.197
OPA - Cycle 1	0.167	5%	0.175
Observation + FreeFlight - Cycle 1	0.669	5%	0.703
OPA - Cycle 2	0.000	5%	0.000
Observation + FreeFlight - Cycle 2	0.000	5%	0.000
Duration TOT [day]	**12.054**	**Duration TOT with margins [day]**	**12.657**

Table 5.14 and Table 5.15 show respectively the deltaV and the time budget for the nominal Observe&Retrieve scenario. Although this scenario will not be applied for SROC's first mission, it is useful to study it for successive missions, when SROC docking capabilities will be tested. The deltaV for the Final Approach Phase (called Docking) was evaluated outside of this thesis.

Table 5.14: DeltaV budget for the nominal Observe&Retrieve scenario

OBSERVE & RETRIEVE Nominal Scenario			
Manoeuvre	**ΔV [m/s]**	**Margin**	**ΔV [m/s]**
HP1	0.489	5%	0.513
Virtual CAM + HP1 bis	1.040	100%	2.080
Virtual CAM + HP1 ter	0.500	100%	1.000
IPA	0.485	5%	0.509
HP2Ins	1.280	5%	1.344
ZRV2	0.423	5%	0.444
HP2	0.096	5%	0.101
OPA - Cycle 1	0.266	100%	0.532
WSE Insertion - Cycle 1	0.192	100%	0.385
OPA - Cycle 2	0.000	100%	0.000
WSE Insertion - Cycle 2	0.000	100%	0.000
HP3Ins	0.221	5%	0.232
ZRV3	0.438	5%	0.459
HP3	0.094	5%	0.099
Docking	0.900	5%	0.945
D CAM	0.068	100%	0.136
SR CAM	0.600	5%	0.630
ΔV TOT [m/s]	**7.092**	**ΔV TOT with margins [m/s]**	**9.409**

Table 5.15: Time budget for the nominal Observe&Retrieve scenario

OBSERVE & RETRIEVE Nominal Scenario			
Manoeuvre	**Duration [day]**	**Margin**	**Duration [day]**
Commissioning	5.000	5%	5.250
HP1	0.188	5%	0.197
IPA	5.760	5%	6.048
HP2Ins	0.083	5%	0.088
HP2	0.188	5%	0.197
OPA - Cycle 1	0.167	5%	0.175
Observation + FreeFlight - Cycle 1	0.669	5%	0.703
OPA - Cycle 2	0.000	5%	0.000
Observation + FreeFlight - Cycle 2	0.000	5%	0.000
HP3Ins	0.113	5%	0.118
HP3	0.188	5%	0.197
Final Approach	0.007	5%	0.007
Duration TOT [day]	**12.361**	**Duration TOT with margins [day]**	**12.979**

6 Variant Scenarios Analysis

A crucial point for the development of SROC's Phase B2 project is the analysis of the variant scenarios: each mission phase was analysed to assess if and how a deviation from its nominal condition could affect the whole mission. The variant events considered in this study can be divided into two categories:

- Programmatic: they take into consideration that some design features of several mission phases are yet to be confirmed and that they could change in future design iterations. For example, as mentioned in Sub-ection 5.3.2, it may be considered necessary to perform two inspections to successfully observe SR;
- Operative: during SROC's operations, variant events could modify the execution of one or more phases. For example, a fault of the RF link establishment during the Commissioning Phase may increase its duration. Other variant scenarios caused by a thruster error in the direction or the magnitude of the thrust, have not been evaluated, since they were already considered in a Phase B1 study;

HP2 was considered as a discontinuity point, after which no previous variant events affect the successive ones. For this reason, the analysis, and this document too, has been divided as follows:

- Variant events before HP2 (Section 6.1);
- Variant events from HP2 onwards (Section 6.2); they also include a variant EMP: as it is explained in Sub-section 6.2.2, according to an STK simulation, SROC will not approach SR in its proximity after the POP. However, a list of possible manoeuvres to avoid an eventual encounter with SR has been proposed and studied;

Section 6.3 analyses the results of this variants analysis, while also providing deltaV and time budgets of two variant scenarios for both the Observe and the Observe&Retrieve scenarios. Moreover, all the variant scenarios obtained are analysed to see which options are viable and which constitute an off-nominal scenario.

6.1 Variant Events Before HP2

The following variant events were considered:

- Longer Commissioning Phase: it was considered a duration of 10 days instead of 5;
- Longer Verification Phase: it was considered a longer duration (13.5 hours instead of 4.5); although the Verification Phase is yet to be defined completely, it is still useful to understand how a different duration may affect the mission;

Figure 6.1 shows the different possible scenarios which can be obtained by combining the nominal and variant segments of the commissioning and the HP1. The Matlab functions, after analysing a segment containing a manoeuvre, obtain a set of possible solutions to reach the desired results, and, for the nominal scenario, they set as nominal the manoeuvre which minimizes the deltaV. For this reason, the nominal scenarios only have one solution, which is the one minimizing the deltaV, which is referred to as the deltaV-down solution. However, during the analysis of the variant scenario, it came clear that it could have been useful, for the successive design iteration, to also have a set of time-down solutions, which aim at decreasing the duration of the mission, while also maintaining an acceptable deltaV budget.

This goal was particularly difficult for the variant scenario with a longer commissioning, where three alternative solutions to the standard time-down were considered (see Sub-section 6.1.2)

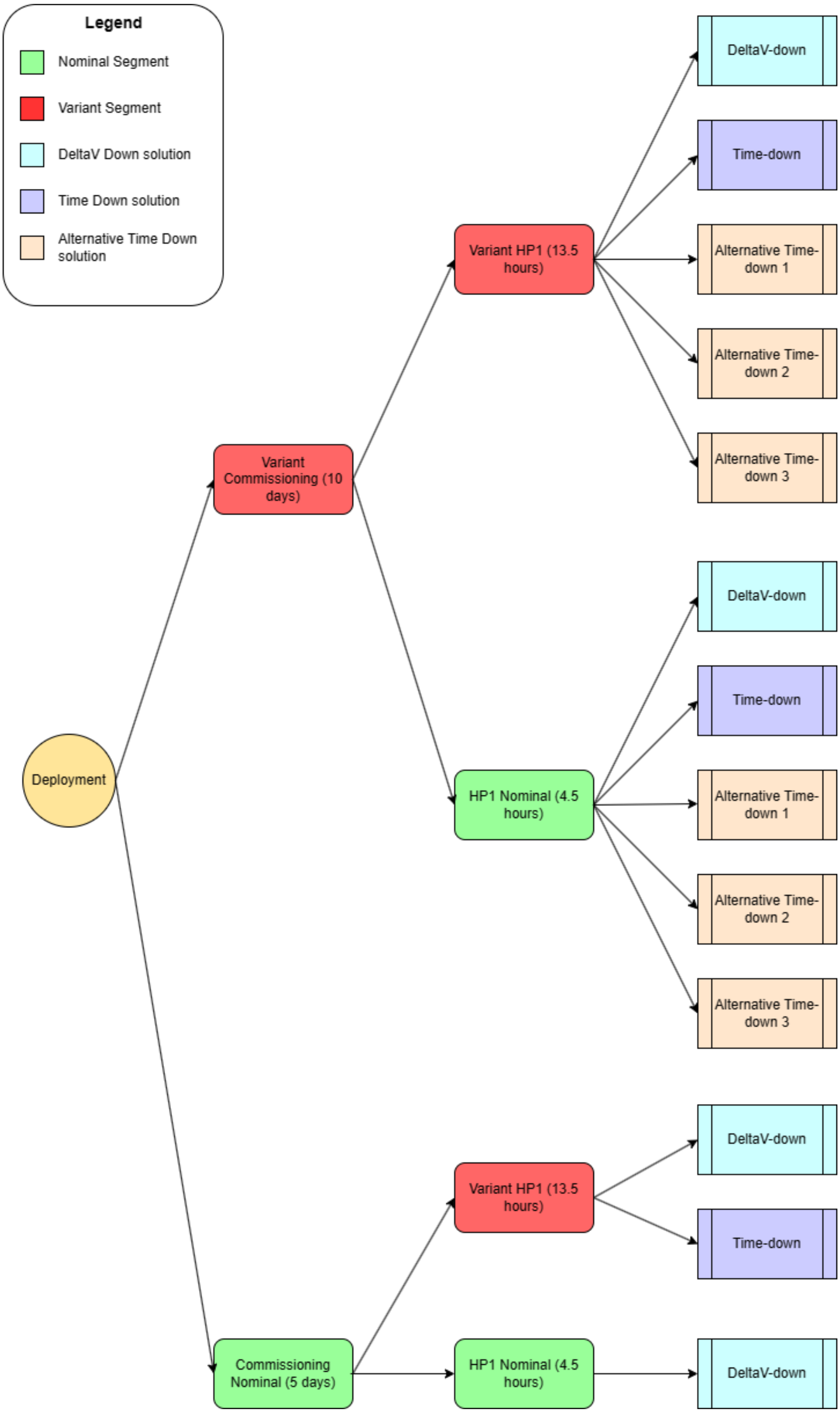

Figure 6.1: Pre-HP2 variant scenarios

6.1.1 Longer HP1

For this variant scenario, it was considered a longer HP1, from 4.5 hours to 13.5 hours. From the results reported in Table 6.1, the deltaV required to perform the HP1 does not change. This is due to how the HP1 manoeuvre has been defined: as mentioned before the target sequence of HP1 does not target a specific relative position, rather it sets SROC's semi-major axis to be the same as SR's. This desired value is reached with the same manoeuvre as the nominal scenario, the only difference is that SROC propagates for a longer time. As shown in Figure 6.2, this causes SROC to reach a further final relative position: -11.3 km along the Radial direction (instead of -10.9 km) and 377 km along the InTrack direction (instead of 373 km).

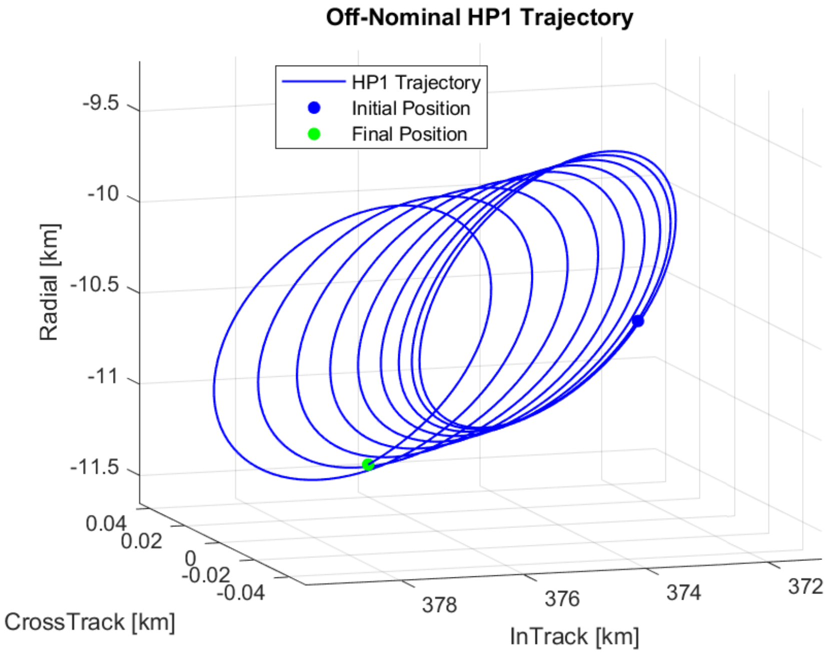

Figure 6.2: HP1 trajectory in RIC coordinates for a longer commissioning

The different relative position reached by SROC affects the successive segments: as shown in Table 6.1, the deltaV-down solution requires a higher deltaV with respect to the nominal scenario, while the duration of the segments after the HP1 does not change. Although a 9-hour delay does not have a huge influence on the total duration of the mission, a time-down solution was still analysed to recover part of the 9 hours lost in HP1. This was achieved by decreasing the duration of the IPA by 7 hours and 41 minutes, but at the cost of a higher deltaV. Table 6.2 summarizes the properties of this time-down solution and compares them to the ones of the nominal Observe&Retrieve scenario. This comparison table, as well as the successive ones, does not include safety margins; they are considered in the summary in Section 6.3.

Table 6.1: DeltaV and duration comparison between the nominal and the LongHP1 - DeltaV-Down (DD) scenarios

Nominal Scenario - LongHP1 (DD) - Duration				
Mission Segment	**Nominal [day]**		**LongHP1 (DD) [day]**	
Comm	5.000		5.000	
HP1	0.188		0.563	
IPA	5.760	5.843	5.760	5.843
HP2Ins	0.083		0.083	
ZRV2	-		-	
Tot PreHP2	11.031		11.406	
Tot Mission	12.361		12.736	

Nominal Scenario -LongHP1 (DD) - DeltaV				
Mission Segment	**Nominal [m/s]**		**LongHP1 (DD) [m/s]**	
Comm	-		-	
HP1	0.489		0.489	
IPA	0.485	2.188	0.518	2.405
HP2Ins	1.280		1.495	
ZRV2	0.423		0.392	
Tot PreHP2	2.677		2.894	
Tot Mission	7.092		7.309	

Table 6.2: DeltaV and duration comparison between the nominal and the LongHP1 - Time-down (TD) scenarios

<table>
<tr><th colspan="5">Nominal Scenario - LongHP1 (TD) - Duration</th></tr>
<tr><th>Mission Segment</th><th colspan="2">Nominal [day]</th><th colspan="2">LongHP1 (TD) [day]</th></tr>
<tr><td>Comm</td><td colspan="2">5.000</td><td colspan="2">5.000</td></tr>
<tr><td>HP1</td><td colspan="2">0.188</td><td colspan="2">0.563</td></tr>
<tr><td>IPA</td><td>5.760</td><td rowspan="3">5.843</td><td>5.440</td><td rowspan="3">5.523</td></tr>
<tr><td>HP2Ins</td><td>0.083</td><td>0.083</td></tr>
<tr><td>ZRV2</td><td>-</td><td>-</td></tr>
<tr><td>Total PreHP2</td><td colspan="2">11.031</td><td colspan="2">11.086</td></tr>
<tr><td>Total Mission</td><td colspan="2">12.361</td><td colspan="2">12.416</td></tr>
</table>

<table>
<tr><th colspan="5">Nominal Scenario -LongHP1 (TD) - DeltaV</th></tr>
<tr><th>Mission Segment</th><th colspan="2">Nominal [m/s]</th><th colspan="2">LongHP1 (TD) [m/s]</th></tr>
<tr><td>Comm</td><td colspan="2">-</td><td colspan="2">-</td></tr>
<tr><td>HP1</td><td colspan="2">0.489</td><td colspan="2">0.489</td></tr>
<tr><td>IPA</td><td>0.485</td><td rowspan="3">2.188</td><td>0.521</td><td rowspan="3">2.480</td></tr>
<tr><td>HP2Ins</td><td>1.280</td><td>1.617</td></tr>
<tr><td>ZRV2</td><td>0.423</td><td>0.343</td></tr>
<tr><td>Total PreHP2</td><td colspan="2">2.677</td><td colspan="2">2.969</td></tr>
<tr><td>Total Mission</td><td colspan="2">7.092</td><td colspan="2">7.383</td></tr>
</table>

6.1.2 Longer Commissioning Phase

The second variant scenario considers a longer HP1: from 5 days to 10 days. Increasing the duration of the commissioning poses two main problems:

- The final relative position of SROC at the end of this phase greatly increases because of the atmospheric drag, which acts for a longer time, further distancing the satellite from SR. As shown in Figure 6.3, the final relative position of the satellite is:
    - Radial: -118.44 km
    - InTrack: 1264.49 km
    - CrossTrack: -0.13 km

  It is noted that the RIC reference system starts losing its meaning at such a high relative position. In fact, it is not respected one of the assumptions at the base of this relative reference system, which is that the relative position vector magnitude must be small if compared to the chief position vector magnitude. A good parameter to compare this final relative position with the one obtained with the nominal duration of the commissioning is the range: about 1270 km for the first case and 373 km for the latter.
- The total duration of the mission almost doubles, not only because of the additional five days required to complete the commissioning phase, but also because the IPA manoeuvre requires more time to reach the desired target position.

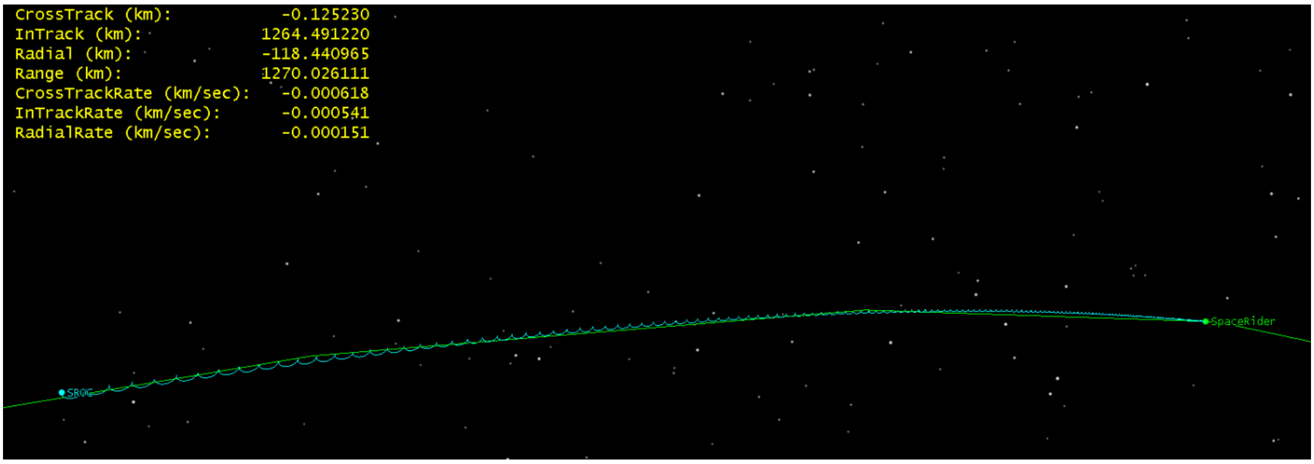

Figure 6.3: Propagation of a 10-days long commissioning

6.1.2.1 DeltaV-Down Solution

The first solution analysed was the deltaV-down one. The InTrack position as a function of the time during the optimal IPA is shown in Figure 6.4: as mentioned earlier, the IPA is longer than the nominal case. So, not only SROC loses 5 days because of the different commissioning, but also requires approximately 6 days more to execute the IPA.

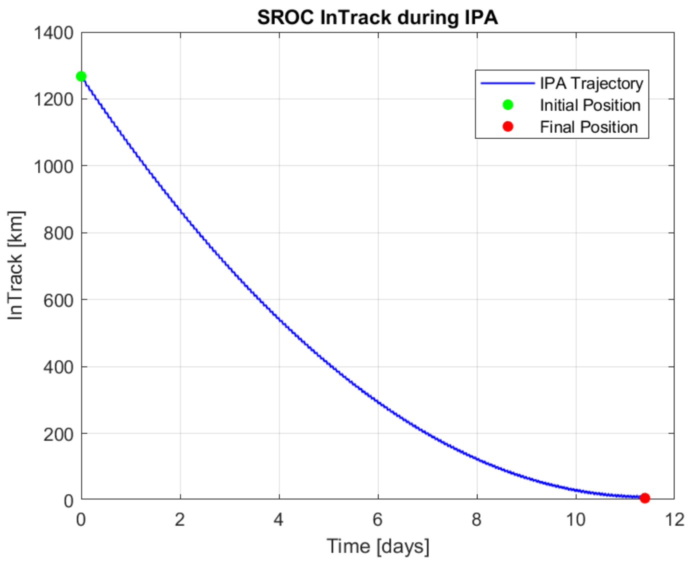

Figure 6.4: InTrack position as function of the time during the IPA - LongComm (DD)

Table 6.3 shows the results of the analysis in terms of duration and deltaV. As expected, the total deltaV increases, although only by approximately 1.2 m/s, since a deltaV solution was applied. This increase is mainly due to the higher deltaV required to perform the IPA: the required value is 81% higher than the nominal one. Another interesting observation is that the deltaV of the HP1 changes with respect to the nominal scenario, although its duration is the nominal one (4.5 hours). This is caused by the decrease of SROC's semi-major axis: since it drifts for more days, the drag decreases its height more than the nominal case. So, it is required a bigger impulse to guarantee a bigger increase in the semi-major axis. The shape of SROC's trajectory during the HP1 changes too, but as shown in Figure 6.5, SROC still rotates around a fictitious point without varying its InTrack position of more than 4 km.

Table 6.3: DeltaV and duration comparison between the nominal and the LongComm – DeltaV-Down (DD) scenarios

<table>
<tr><th colspan="5">Nominal Scenario - LongComm (DD) - Duration</th></tr>
<tr><th>Mission Segment</th><th colspan="2">Nominal [day]</th><th colspan="2">LongHP1 (DD) [day]</th></tr>
<tr><td>Comm</td><td colspan="2">5.000</td><td colspan="2">10.000</td></tr>
<tr><td>HP1</td><td colspan="2">0.188</td><td colspan="2">0.188</td></tr>
<tr><td>IPA</td><td>5.760</td><td rowspan="3">5.843</td><td>11.400</td><td rowspan="3">11.483</td></tr>
<tr><td>HP2Ins</td><td>0.083</td><td>0.083</td></tr>
<tr><td>ZRV2</td><td>-</td><td>-</td></tr>
<tr><td>Total PreHP2</td><td colspan="2">11.031</td><td colspan="2">21.671</td></tr>
<tr><td>Total Mission</td><td colspan="2">12.361</td><td colspan="2">23.001</td></tr>
</table>

<table>
<tr><th colspan="5">Nominal Scenario -LongComm (DD) - DeltaV</th></tr>
<tr><th>Mission Segment</th><th colspan="2">Nominal [m/s]</th><th colspan="2">LongHP1 (DD) [m/s]</th></tr>
<tr><td>Comm</td><td colspan="2">-</td><td colspan="2">-</td></tr>
<tr><td>HP1</td><td colspan="2">0.489</td><td colspan="2">0.887</td></tr>
<tr><td>IPA</td><td>0.485</td><td rowspan="3">2.188</td><td>0.879</td><td rowspan="3">2.943</td></tr>
<tr><td>HP2Ins</td><td>1.280</td><td>1.474</td></tr>
<tr><td>ZRV2</td><td>0.423</td><td>0.590</td></tr>
<tr><td>Total PreHP2</td><td colspan="2">2.677</td><td colspan="2">3.830</td></tr>
<tr><td>Total Mission</td><td colspan="2">7.092</td><td colspan="2">8.245</td></tr>
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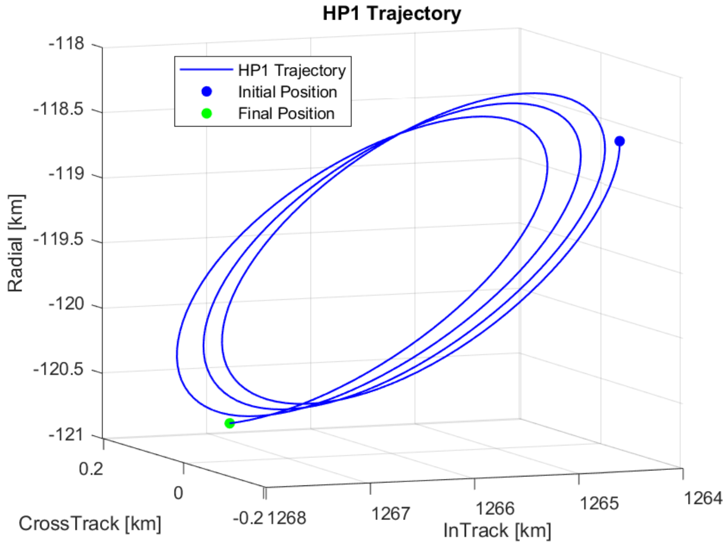


Figure 6.5: SROC Trajectory during HP1 - LongComm (DD)

6.1.2.2 Time-Down Solution

The worrying increase of the total duration of the mission made necessary the study of a time-down solution.

Table 6.4: DeltaV and duration comparison between the nominal and the LongComm - Time-Down (TD) scenarios

Nominal Scenario - LongComm (TD) - Duration				
Mission Segment	**Nominal [day]**		**LongHP1 (TD) [day]**	
Comm	5.000		10.000	
HP1	0.188		0.188	
IPA	5.760		10.820	
HP2Ins	0.083	5.843	0.083	10.903
ZRV2	-		-	
Total PreHP2	11.031		21.091	
Total Mission	12.361		22.421	

Nominal Scenario -LongComm (TD) - DeltaV				
Mission Segment	**Nominal [m/s]**		**LongHP1 (DD) [m/s]**	
Comm	-		-	
HP1	0.489		0.887	
IPA	0.485		0.881	
HP2Ins	1.280	2.188	1.679	3.107
ZRV2	0.423		0.547	
Total PreHP2	2.677		3.994	
Total Mission	7.092		8.409	

The total duration obtained with this solution was not as low as hoped: as shown in Table 6.4, the total duration obtained is just 14 hours less than the deltaV one. As explained in Section 4.2, at the end of the IPA the trajectory of SROC is propagated for 24 hours to assess the risk to SR in case no manoeuvre is performed in the successive 24 hours. The IPA optimization functions only considered an IPA valid if it does not cross 200 m along the InTrack direction during this propagation. The solutions which would decrease the duration of the IPA are also the ones that would result in a higher relative velocity at the end of the IPA, which would cause them to cross the 200 m InTrack limit in the successive 24-hour propagations.

For this reason, alternative solutions to reduce the time were considered:

- Alternative Time-down 1 (ATD-1): run the same Matlab function, but without considering the 24-hours propagation after the IPA;
- Divide the IPA into two parts: during the first one, it is performed the TD1. However, instead of propagating until the desired relative position is reached, the satellite performs a second manoeuvre during the propagation. This additional manoeuvre aims at respecting the 24-hours

propagation constraint by reducing the relative velocity of SROC. Two possible starting points for the IPA Brake were considered:

- Alternative Time-down 2 (ATD-2): the braking manoeuvre is performed 24 hours before the end of the TD1;
- Alternative Time-down 3 (ATD-3) the braking manoeuvre is performed 12 hours before the end of the TD1;

By doing so, it may be recovered some time by the first part of the IPA, which is faster, while the second part should guarantee a manoeuvre safe enough.

Figure 6.6 shows how the ATD-2 and the ATD-3 are defined (lower lines) from the ATD-1 (higher lines). For the ATD-2 and ATD-3, it is important to not confuse the interval of time which is subtracted to the ATD-1 (that is 24 or 12 hours) with the actual duration of the IPA after the brake, which is longer since the relative motion towards SR has been decreased.

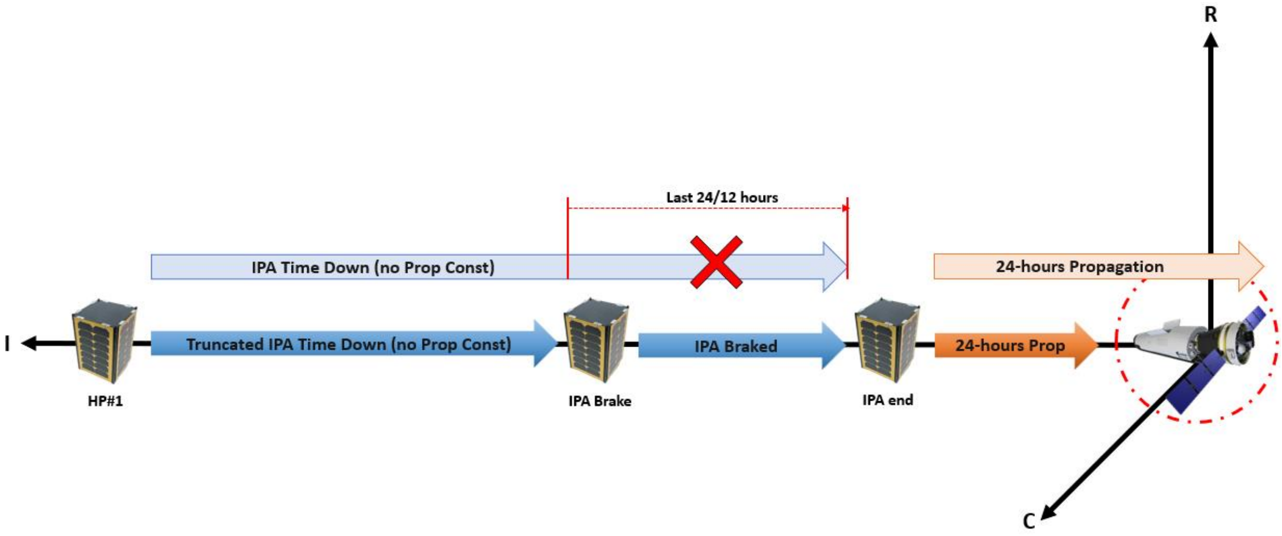


Figure 6.6: Alternative time-down solutions comparison and definition

6.1.2.3 Alternative Time-Down Solutions – ATD-1

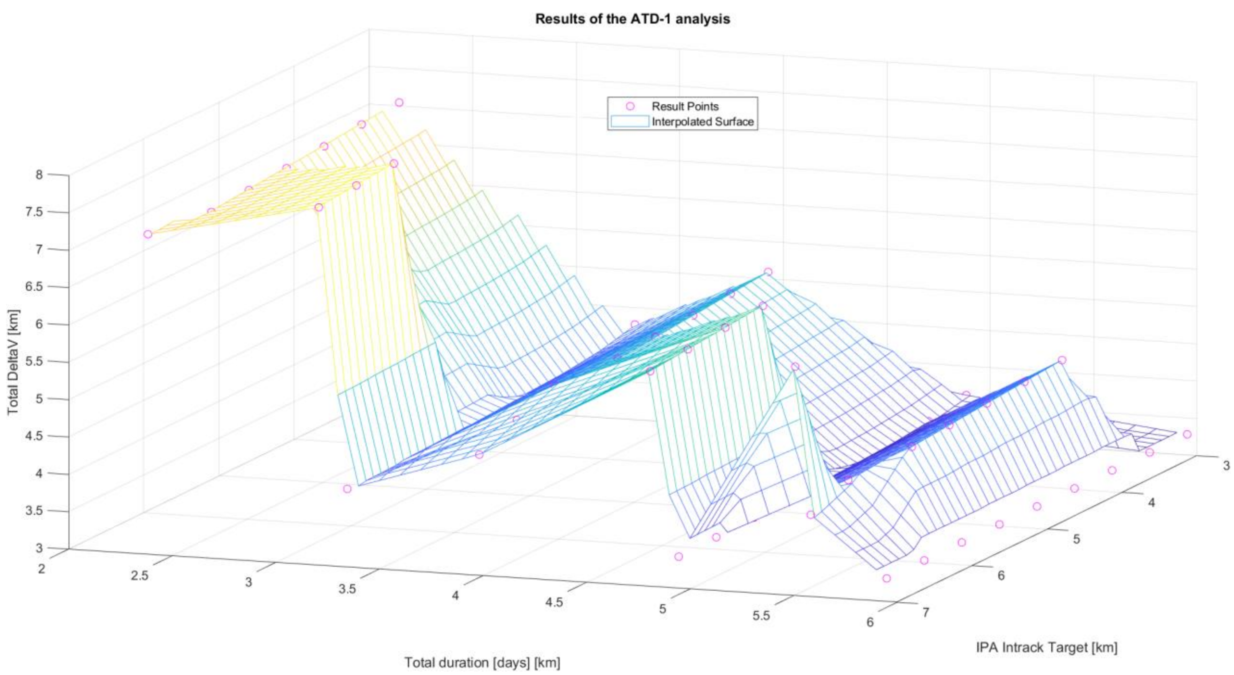


Figure 6.7: 3D plot of the total deltaV as function of the total duration and the IPA InTrack target – ATD-1

Figure 6.7 shows a plot with all the possible IPA + HP2 insertion + ZRV2 sequences for the ATD-1 solution (all the circles refer to one analysed sequence). It is noted that the adjective “total” is used to refer to the entirety of the IPA + HP2 insertion + ZRV2 sequences. These results were obtained by iterating on the following values:

- IPA InTrack target: [3:0.5: 7] km;
- IPA Duration: [2:0.1:6] days;

Although the aim of the analysis is to define a time-down solution, as shown in Figure 6.8, the solutions guaranteeing the lowest total duration are too much expensive deltaV-wise. For this reason, the final solution has been selected among the ones in the red box.

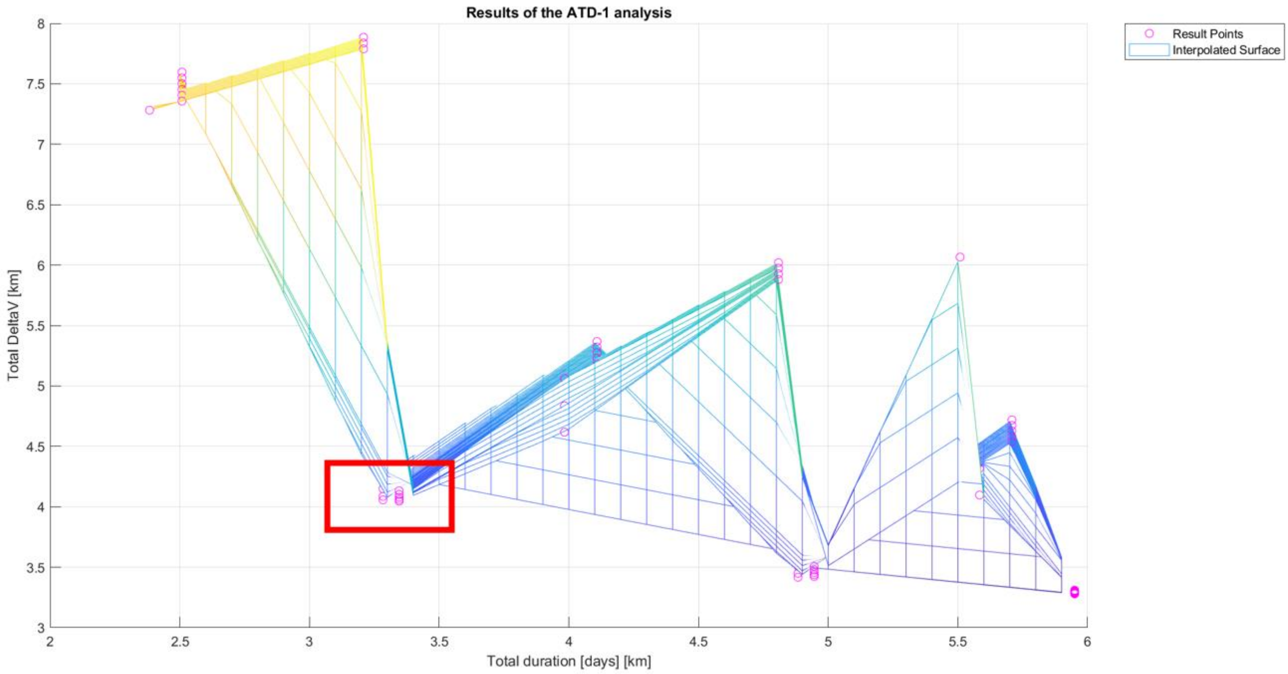


Figure 6.8: Total DeltaV as function of the total Duration – ATD-1

Figure 6.9 shows the trends of both the total deltaV (blue line) and the IPA’s deltaV (red line) as a function of the duration of the IPA. As expected, the IPA’s deltaV trend decreases with its duration: this can be explained by considering that the fastest the IPA is, the higher variation in the kinetic energy is required. The behaviour for the total deltaV, instead, is a bit different: although the trend of the solutions decreases with the duration of the IPA, the single solutions do not. This discrepancy with the IPA’s deltaV is caused by the deltaV contribution of the HP2 insertion and ZRV2 manoeuvres.

Different combinations of durations and InTrack targets define a different set of relative positions and velocities at the end of the IPA, thus influencing the required deltaV for the successive manoeuvres. This means that small differences in the selection of the moment to end the IPA and start the HP2 insertion (in the order of tens of minutes) can change the cost of the HP2 insertion and subsequent ZRV2 manoeuvre. Of course, to consider this factor, it should be used smaller steps for the IPA InTrack target and IPA duration than the ones considered for this analysis. However, this change was not applied because it would have increased considerably the analysis time required by the Matlab functions. Moreover, this approach gives a more conservative estimate of the required deltaV, since, in case an ulterior optimization was required, the allocated total deltaV could only decrease with respect to the current results. Finally, to consider the optimal moment to end the IPA with precision in the order of minutes it would be necessary to assess if the actual manoeuvre could be performed with the same precision during the operative phase. In fact, in case this condition could not be met by the mission, the allocated deltaV would be lower than the actual one.

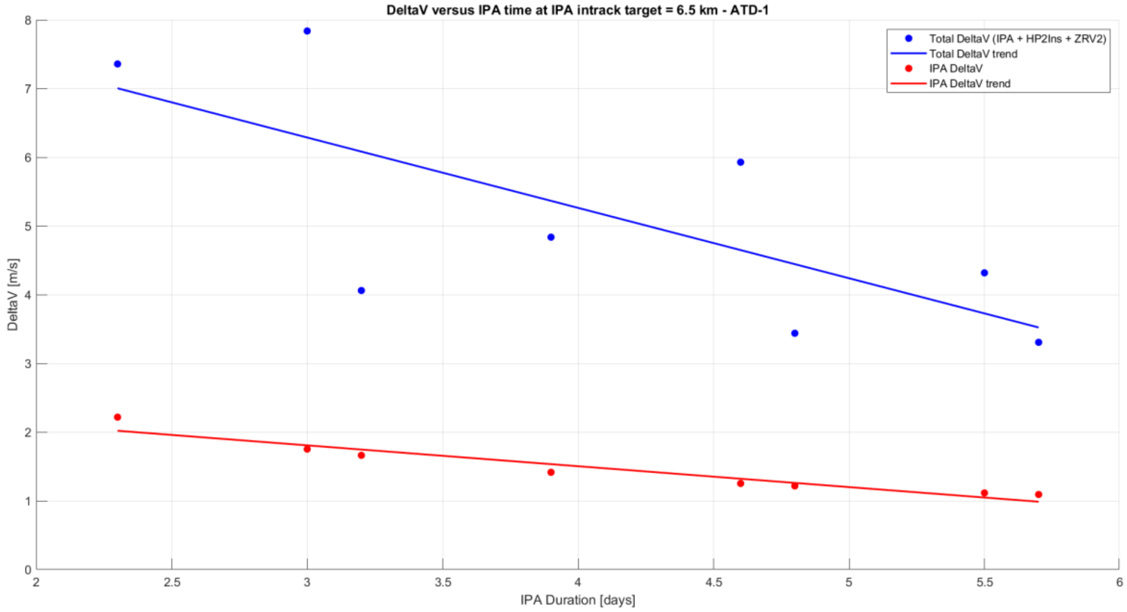


Figure 6.9: DeltaV trend according to the IPA duration - ATD-1

Table 6.5 shows the duration and the deltaV required by the selected ATD-1 and the nominal scenario. Most of the time lost during the longer commissioning is retrieved and the deltaV required, although higher of 2.268 m/s is still acceptable. This increase is mainly caused by the IPA manoeuvre (which costs 1.664 m/s instead of 0.489 m/s as in the nominal scenario), although also the other pre-HP2 manoeuvres require a higher deltaV than in the nominal scenario.

Table 6.5: DeltaV and duration comparison between the nominal and the LongComm – ATD-1 scenarios

Nominal Scenario - LongComm (ATD-1) - Duration				
Mission Segment	**Nominal [day]**		**LongComm (ATD-1) [day]**	
Comm	5.000		10.000	
HP1	0.188		0.188	
IPA	5.760	5.843	3.200	3.283
HP2Ins	0.083		0.083	
ZRV2	-		-	
Total PreHP2	11.031		13.471	
Total Mission	12.361		14.802	

Nominal Scenario -LongComm (ATD-1) - DeltaV				
Mission Segment	**Nominal [m/s]**		**LongComm (ATD-1) [m/s]**	
Comm	-		-	
HP1	0.489		0.887	
IPA	0.485	2.188	1.664	4.058
HP2Ins	1.280		2.030	
ZRV2	0.423		0.364	
Total PreHP2	2.677		4.945	
Total Mission	7.092		9.360	

Table 6.6 reports in each row the following properties of every valid solution: InTrack target, duration and deltaV of the IPA, duration and deltaV of the HP2 insertion, deltaV for the ZRV2 manoeuvre, total deltaV and total duration of the IPA + HP2 insertion + ZRV2 sequence. Finally, it also shows, in the last column, the minutes which would take SROC to cross the 200 m InTrack position if its orbit was propagated after the IPA instead of performing the HP2 insertion. From the safety point of view, these results are concerning since they show that for every solution the KOZ is entered in less than 1 hour. For this reason, this ATD-1 solution may be discarded as off-nominal. The row highlighted in orange is the solution with the lowest total duration, while the one in yellow is the solution considered for the ATD-1.

Table 6.6: Detailed results properties - ATD-1

IPA Intrack [km]	IPA Duration [days]	IPA DeltaV [m/s]	HP2Ins Duration [hr]	HP2Ins DeltaV	ZRV2 DeltaV	Total DeltaV	Total time	Time to 200 m [min]
4	2.3	2.221	5	3.955	1.420	7.597	2.508	45
4.5	2.3	2.220	5	3.930	1.398	7.549	2.508	46
5	2.3	2.220	5	3.905	1.376	7.501	2.508	47
5.5	2.3	2.219	5	3.880	1.354	7.453	2.508	47
6	2.3	2.218	5	3.854	1.332	7.405	2.508	48
6.5	2.3	2.217	5	3.829	1.311	7.357	2.508	49
7	2.3	2.216	2	3.944	1.121	7.282	2.383	49
6	3	1.752	5	3.687	2.445	7.885	3.208	58
6.5	3	1.752	5	3.662	2.423	7.837	3.208	59
7	3	1.751	5	3.638	2.400	7.789	3.208	59
3	3.2	1.666	2	2.384	0.225	4.274	3.283	43
3.5	3.2	1.665	2	2.265	0.219	4.150	3.283	44
4	3.2	1.664	2	2.147	0.275	4.086	3.283	45
4.5	3.2	1.664	2	2.030	0.364	4.057	3.283	46
5	3.2	1.663	3.5	2.257	0.213	4.133	3.346	47
5.5	3.2	1.663	3.5	2.214	0.225	4.102	3.346	48
6	3.2	1.662	3.5	2.171	0.245	4.078	3.346	49
6.5	3.2	1.661	3.5	2.128	0.270	4.060	3.346	50
7	3.2	1.661	3.5	2.086	0.299	4.046	3.346	50
3.5	3.9	1.418	5	2.864	1.087	5.368	4.108	55
4	3.9	1.417	5	2.840	1.065	5.321	4.108	56
4.5	3.9	1.417	5	2.815	1.043	5.275	4.108	57
5	3.9	1.416	5	2.791	1.021	5.228	4.108	58
5.5	3.9	1.416	5	2.766	1.000	5.182	4.108	59
6	3.9	1.415	2	2.824	0.824	5.062	3.983	60
6.5	3.9	1.415	2	2.701	0.721	4.836	3.983	61
7	3.9	1.414	2	2.578	0.625	4.617	3.983	62
5.5	4.6	1.253	5	2.710	2.057	6.020	4.808	68
6	4.6	1.252	5	2.686	2.035	5.974	4.808	70
6.5	4.6	1.252	5	2.662	2.013	5.927	4.808	71
7	4.6	1.252	5	2.639	1.991	5.881	4.808	72
3	4.8	1.219	2	2.136	0.258	3.613	4.883	55
3.5	4.8	1.219	2	2.013	0.276	3.508	4.883	56
4	4.8	1.218	2	1.891	0.340	3.450	4.883	57
4.5	4.8	1.218	2	1.768	0.430	3.417	4.883	58
5	4.8	1.218	3.5	2.026	0.262	3.506	4.946	59
5.5	4.8	1.217	3.5	1.983	0.278	3.478	4.946	60
6	4.8	1.217	3.5	1.940	0.299	3.456	4.946	61
6.5	4.8	1.216	3.5	1.896	0.325	3.438	4.946	63
7	4.8	1.216	3.5	1.853	0.354	3.423	4.946	64
7	5.3	1.140	5	2.036	2.890	6.066	5.508	79
4	5.5	1.116	5	2.478	1.125	4.719	5.708	68
4.5	5.5	1.115	5	2.455	1.103	4.673	5.708	69
5	5.5	1.115	5	2.431	1.082	4.628	5.708	70
5.5	5.5	1.115	5	2.408	1.060	4.583	5.708	71
6	5.5	1.114	5	2.384	1.039	4.538	5.708	73
6.5	5.5	1.114	2	2.397	0.806	4.317	5.583	74
7	5.5	1.114	2	2.276	0.708	4.097	5.583	75
3	5.7	1.094	6	1.182	1.003	3.278	5.950	55
3.5	5.7	1.094	6	1.180	1.008	3.282	5.950	56
4	5.7	1.093	6	1.179	1.013	3.286	5.950	58
4.5	5.7	1.093	6	1.178	1.019	3.289	5.950	59
5	5.7	1.093	6	1.177	1.024	3.293	5.950	60
5.5	5.7	1.092	6	1.176	1.030	3.297	5.950	61
6	5.7	1.092	6	1.175	1.035	3.302	5.950	63
6.5	5.7	1.092	6	1.174	1.041	3.306	5.950	64

6.1.2.4 Alternative Time-Down Solutions – ATD-2

Since the solutions proposed for the ATD-1 may be considered off-nominal for not being safe enough, the ATD-2 and ATD-3 solutions were proposed to guarantee compliance with the 200 m InTrack limit for a hypothetical 24-hours propagation at the end of the IPA. A new Matlab function, called "IPA_Brake" was created to evaluate the ATD-2 and the ATD-3 solutions. To define the ATD-1 solution the same Matlab function described in Section 4.2 is used, with the only difference being that the hypothetical 24-hour propagation at the end of the IPA and the relative InTrack check are not performed. At the end of this analysis, if a specific flag defined by the user in the JSON of the IPA optimization function is true, the "IPA_Brake" function is called. In short, this is the analysis process performed by this new function:

- Change the duration of the IPA propagation: as mentioned before, the braking manoeuvre is defined starting from the point of the IPA propagation at 24 (for the ATD-2) and 12 (for the ATD-3) hours from the end of the propagation itself; the MCS on STK is than run to apply these changes;
- Add to STK the IPA_Brake target sequence, which is composed of the same segments as the IPA Rendezvous target sequence:
    - Manoeuvre segment to perform the braking manoeuvre;
    - Propagation segment to propagate SROC to the desired InTrack position;

  This sequence also has the same desired result (the InTrack position at the end of the propagation) and the same control parameter (the thrust vector along the V axis of SROC's VNC reference system);
- Define the optimal brake manoeuvre in terms of the total deltaV required by the IPA brake + HP2 insertion + ZRV2 manoeuvres. This process is performed exactly as the one for the nominal scenario:
    - Different solutions obtained by iterating on the duration and the InTrack target are analysed. In this case, the IPA duration does not refer to the whole IPA, but only to the braked part, that is the propagate segment after the IPA brake;
    - If they are valid, the successive HP2 insertion and ZRV2 manoeuvres are evaluated. The constraints used to define the validity of the braked section of the IPA are the same used for the nominal scenario analysis. Of course, since the aim of the ATD-2 and ATD-3 solutions is to provide a safer solution than the ATD-1, they include the 200 m InTrack limit after a 24-hours propagation at the end of the IPA;
    - All the valid solutions are saved to be post-processed;

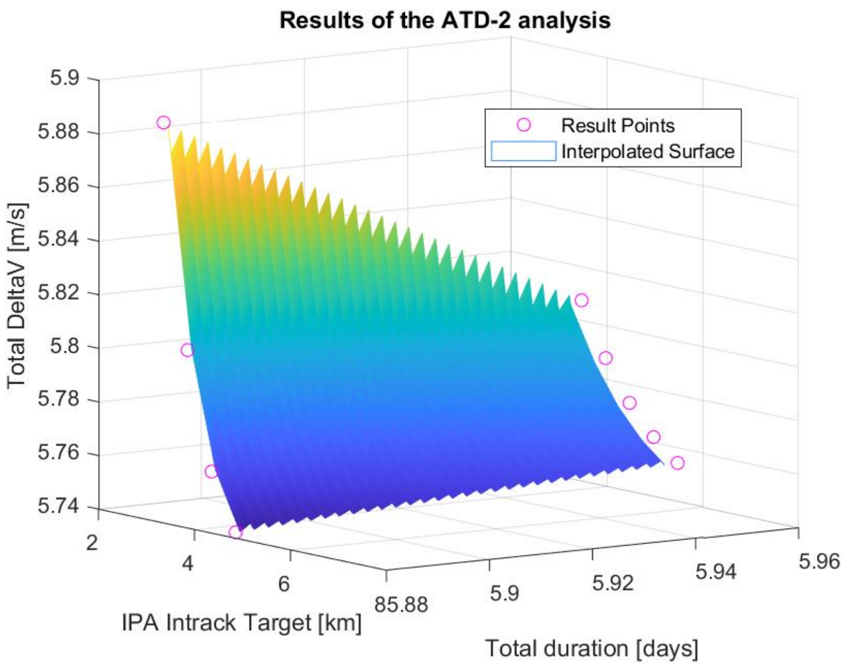


Figure 6.10: 3D plot of the total deltaV as function of the total duration and the IPA InTrack target – ATD-2

Figure 6.10 shows a 3D plot with all the possible IPA brake + HP2 insertion + ZRV2 sequences for the ATD-2 solution; the "total" adjective in the legend refers to the IPA brake + HP2 insertion and ZRV2 manoeuvres. These results were obtained by iterating on the following values for the IPA_Brake target sequence:

- InTrack target: [3:0.5: 7] km;
- Duration: [2:0.1:6] days;

The number of valid results is decisively less than the one for the ATD-1 analysis. This is due to the fact that many solutions were discarded because they could not respect either the desired target values or the compliance with the 200 m InTrack limit for the hypothetical 24-hour propagation at the end of the IPA.

Table 6.7 reports in each row the following properties of every valid solution: InTrack target, duration and deltaV of the IPA Brake, duration and deltaV of the HP2 insertion, deltaV for the ZRV2 manoeuvre, total deltaV and total duration. In this case, the adjective "total" refers to the IPA + IPA brake + HP2 insertion + ZRV2 sequence. In fact, to evaluate the total duration, 2.2 days were added: the IPA with no brake would last 3.2 days, but since the brake is performed 1 day before its theoretical end, the duration of this propagation segment is just 2.2 days. To evaluate the total deltaV, 1.664 m/s were added to consider the deltaV required to perform the first part of the IPA.

Table 6.7: Detailed results properties - ATD-2

IPA Intrack [km]	Braked IPA Duration [days]	IPA DeltaV [m/s]	HP2Ins Duration [hr]	HP2Ins DeltaV	ZRV2 DeltaV	Total DeltaV	Total Time
3	5.8	1.011	2	4.593	0.283	7.551	8.083
3.5	5.8	1.011	2	4.480	0.313	7.468	8.083
4	5.8	1.012	2	4.367	0.382	7.425	8.083
4.5	5.8	1.012	2	4.255	0.473	7.404	8.083
5	5.8	1.012	3.5	4.513	0.290	7.480	8.146
5.5	5.8	1.013	3.5	4.475	0.309	7.461	8.146
6	5.8	1.013	3.5	4.437	0.331	7.446	8.146
6.5	5.8	1.013	3.5	4.399	0.358	7.435	8.146
7	5.8	1.014	3.5	4.361	0.388	7.427	8.146

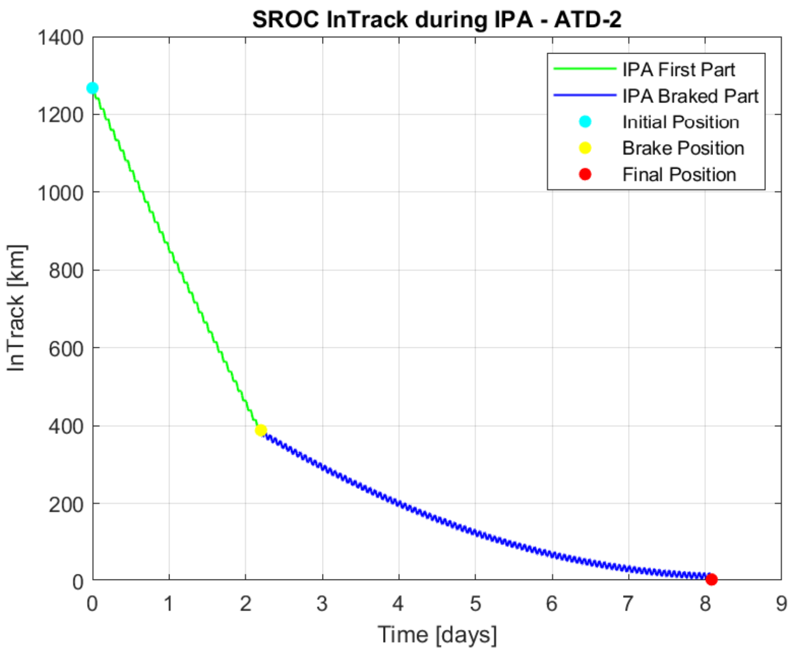


Figure 6.11: InTrack as a function of the time - ATD-2

The row highlighted in yellow in Table 6.7 was selected for the ATD-2 solution since it is both the faster and the less-expensive deltaV-wise. Because of the braking manoeuvre, the total duration increases significantly: Figure 6.11 shows that to cover the last 400 km along the InTrack direction SROC takes 5.8 days, while for the ATD-1 it would have taken only 1 day. Table 6.8 shows the deltaV and duration of each segment before the HP2 (the IPA rows consider both the first and the braked part). In conclusion, the total duration decreases by 2.819 days with respect to the standard time-down solution, which requires a total of 22.421 days. On the other hand, the total deltaV increases by 79.16% from the nominal scenario.

Table 6.8: DeltaV and duration comparison between the nominal and the LongComm – ATD-2 scenarios

<table>
<tr><th colspan="5">Nominal Scenario - LongComm (ATD-2) - Duration</th></tr>
<tr><th>Mission Segment</th><th colspan="2">Nominal [day]</th><th colspan="2">Long LongComm (ATD-2) [day]</th></tr>
<tr><td>Comm</td><td colspan="2">5.000</td><td colspan="2">10.000</td></tr>
<tr><td>HP1</td><td colspan="2">0.188</td><td colspan="2">0.188</td></tr>
<tr><td>IPA</td><td>5.760</td><td rowspan="3">5.843</td><td>8.000</td><td rowspan="3">8.083</td></tr>
<tr><td>HP2Ins</td><td>0.083</td><td>0.083</td></tr>
<tr><td>ZRV2</td><td>-</td><td>-</td></tr>
<tr><td>Total PreHP2</td><td colspan="2">11.031</td><td colspan="2">18.271</td></tr>
<tr><td>Total Mission</td><td colspan="2">12.361</td><td colspan="2">19.602</td></tr>
</table>

<table>
<tr><th colspan="5">Nominal Scenario -LongComm (ATD-2) - DeltaV</th></tr>
<tr><th>Mission Segment</th><th colspan="2">Nominal [m/s]</th><th colspan="2">Long LongComm (ATD-2) [m/s]</th></tr>
<tr><td>Comm</td><td colspan="2">-</td><td colspan="2">-</td></tr>
<tr><td>HP1</td><td colspan="2">0.489</td><td colspan="2">0.887</td></tr>
<tr><td>IPA</td><td>0.485</td><td rowspan="3">2.188</td><td>2.676</td><td rowspan="3">7.404</td></tr>
<tr><td>HP2Ins</td><td>1.280</td><td>4.255</td></tr>
<tr><td>ZRV2</td><td>0.423</td><td>0.473</td></tr>
<tr><td>Total PreHP2</td><td colspan="2">2.677</td><td colspan="2">8.291</td></tr>
<tr><td>Total Mission</td><td colspan="2">7.092</td><td colspan="2">12.706</td></tr>
</table>

6.1.2.5 Alternative Time-Down Solutions – ATD-3

The ATD-3 solution uses the same process and Matlab function used for the ATD-2, with the only difference being the time before the IPA end at which the brake starts. It was selected a lower value (12 hours) with the intention of reducing the total duration of the manoeuvre. Figure 6.12 shows the 3D plot with the total duration, total deltaV and InTrack target of every valid result; for this graph, the adjective "total" refers to the braked section of the IPA, HP2 insertion and ZRV2 manoeuvres. Generally, every solution presents a similar deltaV (between 4.3 and 3.5 m/s) and a similar total duration (between approximately 4.2 and 4.6 days).

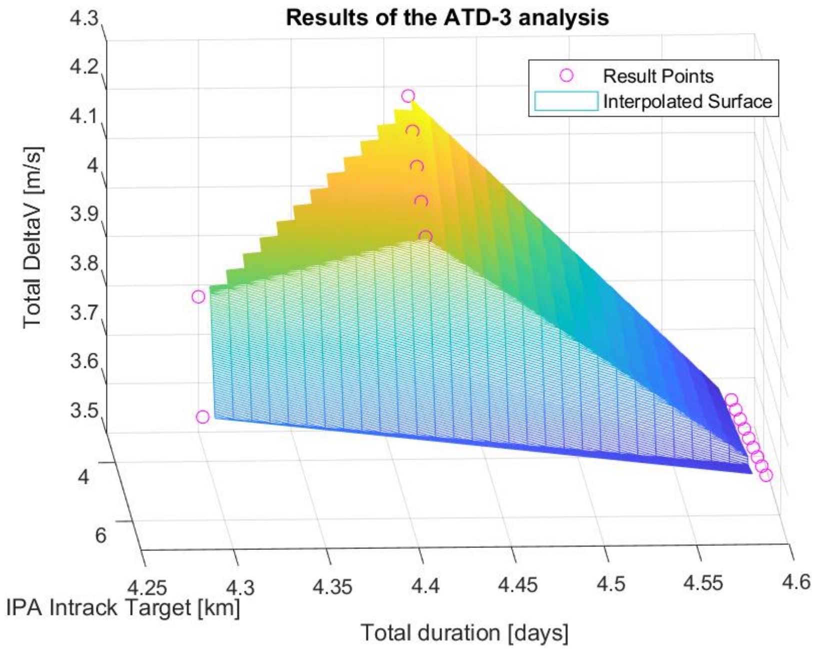


Figure 6.12: 3D plot of the total deltaV as function of the total duration and the IPA InTrack target – ATD-3

Defining which sequence to use for the ATD-3 was simple: as shown in Figure 6.13, the result which minimizes the duration also presents an acceptably low deltaV, which is only 0.216 m/s higher than the minimum total deltaV.

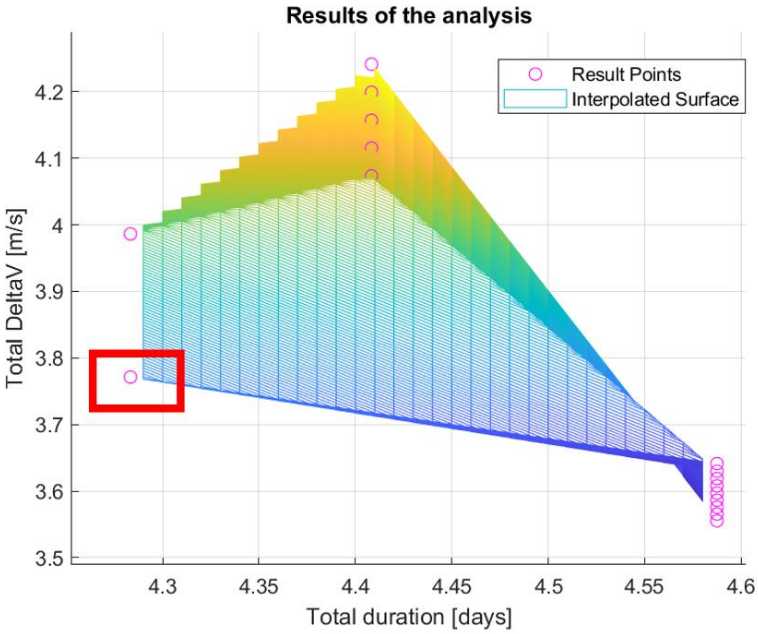


Figure 6.13: Selected solution for the ATD-3

Table 6.9 Detailed results properties - ATD-3

IPA Intrack [km]	Braked IPA duration [days]	IPA DeltaV [m/s]	HP2Ins Duration [hr]	HP2Ins DeltaV	ZRV2 DeltaV	Total DeltaV	Total time
4	4.2	1.110	5	1.922	1.209	5.905	7.108
4.5	4.2	1.111	5	1.902	1.186	5.863	7.108
5	4.2	1.111	5	1.882	1.164	5.821	7.108
5.5	4.2	1.112	5	1.862	1.142	5.779	7.108
6	4.2	1.112	5	1.842	1.119	5.738	7.108
6.5	4.2	1.113	2	1.960	0.914	5.650	6.983
7	4.2	1.113	2	1.851	0.807	5.435	6.983
3	4.4	1.110	4.5	1.464	0.981	5.219	7.288
3.5	4.4	1.111	4.5	1.466	0.989	5.229	7.288
4	4.4	1.111	4.5	1.467	0.997	5.240	7.288
4.5	4.4	1.112	4.5	1.469	1.005	5.250	7.288
5	4.4	1.112	4.5	1.471	1.014	5.261	7.288
5.5	4.4	1.112	4.5	1.473	1.022	5.272	7.288
6	4.4	1.113	4.5	1.475	1.031	5.283	7.288
6.5	4.4	1.113	4.5	1.477	1.040	5.294	7.288

Table 6.9 reports in each row the following properties of every valid solution: InTrack target, duration and deltaV of the IPA Brake, duration and deltaV of the HP2 insertion, deltaV for the ZRV2 manoeuvre, total deltaV and total duration. In this case, the adjective "total" refers to the IPA + IPA brake + HP2 insertion + ZRV2 sequence. In fact, to evaluate the total duration, 2.7 days were added: the IPA with no brake would last 3.2 days, but since the brake is performed 12 hours before its theoretical end, the duration of this propagation segment is just 2.7 days. To evaluate the total deltaV, 1.664 m/s were added to consider the deltaV required to perform the first part of the IPA. It is interesting to notice that the solutions are valid only

for a duration of the braked IPA equal to 4.2 or 4.4 days, although the analysis was performed iterating on the following values for the IPA_Brake target sequence:

- InTrack target: [3:0.5: 7] km;
- Duration: [2:0.1:6] days;

As for the ATD-2, lower durations were not accepted since they produced faster propagation segments which did not respect the InTrack limit on the 24-hours propagation after the IPA, while higher duration did not provide a solution giving the desired InTrack target.

Figure 6.14 shows that delaying the start of the braking manoeuvre of 12 hours guarantees a lower duration of the braking IPA segment since it starts with an InTrack value of approximately 200 km instead of the ATD-2 which performed the same manoeuvre at approximately 400 km along InTrack.

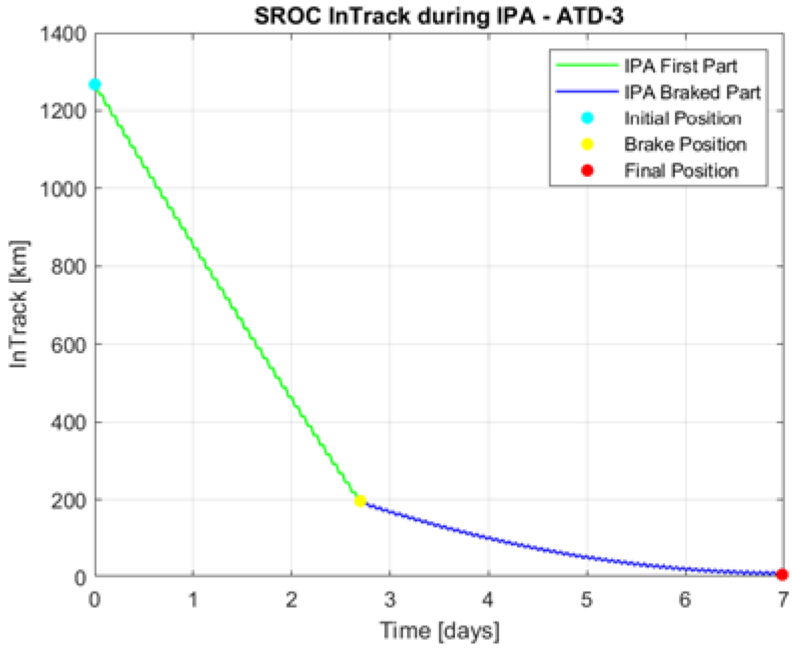


Figure 6.14: InTrack as a function of the time - ATD-3

Table 6.10 offers a comparison between the durations of the nominal scenario and all the time-down solutions in case of longer commissioning. Of course, the ATD-1 solution is the fastest one and guarantees only a delay of 2.441 days with respect to the nominal scenario, but it is not considered safe for SR. The ATD-3 recovers a few days, with a total delay of 6.14 days.

Table 6.10: Duration comparison between the nominal and all the time-down solutions in case of a longer commissioning

Nominal Scenario - LongComm Duration										
Mission Segment	**Nominal [day]**		**LongHP1 (TD) [day]**		**LongHP1 (ATD-1) [day]**		**LongHP1 (ATD-2) [day]**		**LongHP1 (ATD-3) [day]**	
Comm	5.000		10.000		10.000		10.000		10.000	
HP1	0.188		0.188		0.188		0.188		0.188	
IPA	5.760		10.820		3.200		8.000		6.900	
HP2Ins	0.083	5.843	0.083	10.903	0.083	3.283	0.083	8.083	0.083	6.983
ZRV2	-		-		-		-		-	
Total PreHP2	11.031		21.091		13.471		18.271		17.171	
Total Mission	12.361		22.421		14.802		19.602		18.501	

Table 6.11 shows the comparison between the deltaVs of the nominal scenario and of all the time-down solutions in case of a longer commissioning. The ATD-2 And ATD-3 require a higher deltaV since they both include an additional manoeuvre to brake the IPA. Although the ATD-3 may seem like a good compromise between the required deltaV and total duration, it may be classified as off-nominal for not being sufficiently safe since it starts the brake manoeuvre 12 hours before the end of the unbraked IPA segment. This means that in case no manoeuvre was performed SROC would cross the 200 km InTrack limit in less than 13 hours. In conclusion, considering the safety constraint, the required deltaV and the duration, the best solution may be the ATD-2 solution.

Table 6.11: DeltaV comparison between the nominal and all the time-down solutions in case of a longer commissioning

<table>
<tr><th colspan="11">Nominal Scenario - LongComm DeltaV</th></tr>
<tr><th>Mission Segment</th><th colspan="2">Nominal [m/s]</th><th colspan="2">LongHP1 (TD) [m/s]</th><th colspan="2">LongHP1 (ATD-1) [m/s]</th><th colspan="2">LongHP1 (ATD-2) [m/s]</th><th colspan="2">LongHP1 (ATD-3) [m/s]</th></tr>
<tr><td>Comm</td><td colspan="2">-</td><td colspan="2">-</td><td colspan="2">-</td><td colspan="2">-</td><td colspan="2">-</td></tr>
<tr><td>HP1</td><td colspan="2">0.489</td><td colspan="2">0.887</td><td colspan="2">0.887</td><td colspan="2">0.887</td><td colspan="2">0.887</td></tr>
<tr><td>IPA</td><td>0.485</td><td rowspan="3">2.188</td><td>0.881</td><td rowspan="3">3.107</td><td>1.664</td><td rowspan="3">4.058</td><td>2.676</td><td rowspan="3">5.843</td><td>2.777</td><td rowspan="3">5.435</td></tr>
<tr><td>HP2Ins</td><td>1.280</td><td>1.679</td><td>2.030</td><td>4.255</td><td>1.851</td></tr>
<tr><td>ZRV2</td><td>0.423</td><td>0.547</td><td>0.364</td><td>0.473</td><td>0.807</td></tr>
<tr><td>Total PreHP2</td><td colspan="2">2.667</td><td colspan="2">3.994</td><td colspan="2">4.945</td><td colspan="2">8.291</td><td colspan="2">6.322</td></tr>
<tr><td>Total Mission</td><td colspan="2">7.092</td><td colspan="2">8.409</td><td colspan="2">9.360</td><td colspan="2">12.706</td><td colspan="2">10.737</td></tr>
</table>

6.1.3 Longer Commissioning and HP1

The third possible deviation from the nominal scenario involves both a longer HP1 (4.5 hours) and commissioning (10 days). The trajectory during HP1 differs from the ones in the previous case, but the required deltaV to perform the HP1 insertion manoeuvre is the same for the longer commissioning phase. This is the equivalent of what has been said for the longer HP1 deviation: the target of the manoeuvre is to get SR's semi-major axis at the end of the HP1 and this objective is met with the same manoeuvre for a duration of both 4.5 and 13.5 hours. The semi-major axis is increased by an impulsive manoeuvre and stays almost constant for the whole HP1 since its duration is not enough for the external disturbances, particularly the atmospheric drag, to change it. Figure 6.15 shows the SROC's trajectory during HP1.

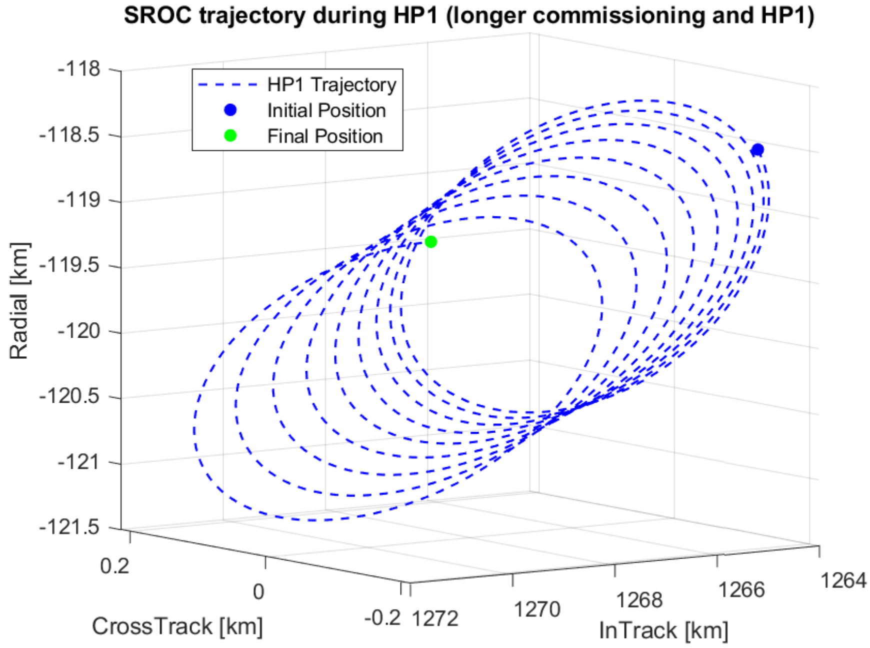


Figure 6.15: SROC's trajectoty during HP1 for a longer commissioning and HP1

The principal problems which arise from this scenario are the same discussed in the previous sub-section. For this reason, the solution investigated are also the same:

- **DeltaV-down** IPA + HP2 insertion + ZRV2 sequence which minimizes the deltaV;
- **Time Down**: IPA + HP2 insertion + ZRV2 sequence which minimizes the duration without breaking the 200 km InTrack constraint on a hypothetical 24-hours propagation after the IPA;
- **Alternative time-down**:
  - **ATD-1**: no InTrack constraint on a hypothetical 24-hours propagation after the IPA;
  - **ATD-2**: solution composed of two parts: during the first one, the same IPA selected for ATD-1 is performed. The second part is constituted by a braking manoeuvre to slow down SROC enough to respect the 200 km InTrack constraint on a hypothetical 24-hours propagation after the IPA. This brake manoeuvre is performed 24 hours before the end of the ATD-1;
  - **ATD-3**: it only differs from the ATD-2 for the time at which the brake is performed (12 hours before the ATD-1 end instead of 24 hours);

6.1.3.1 DeltaV-Down Solution

Figure 6.16 shows the InTrack as a function of the time from the IPA start for the selected solution. The same considerations made for the DD solution in case of longer commissioning can be applied here. Actually, SROC starts the IPA at an even higher relative distance from Space Rider, since the longer HP1 causes SROC to drift a few kilometres more. Since SROC starts the IPA at a higher relative position, it also requires a higher time to perform it in a deltaV-efficient way.

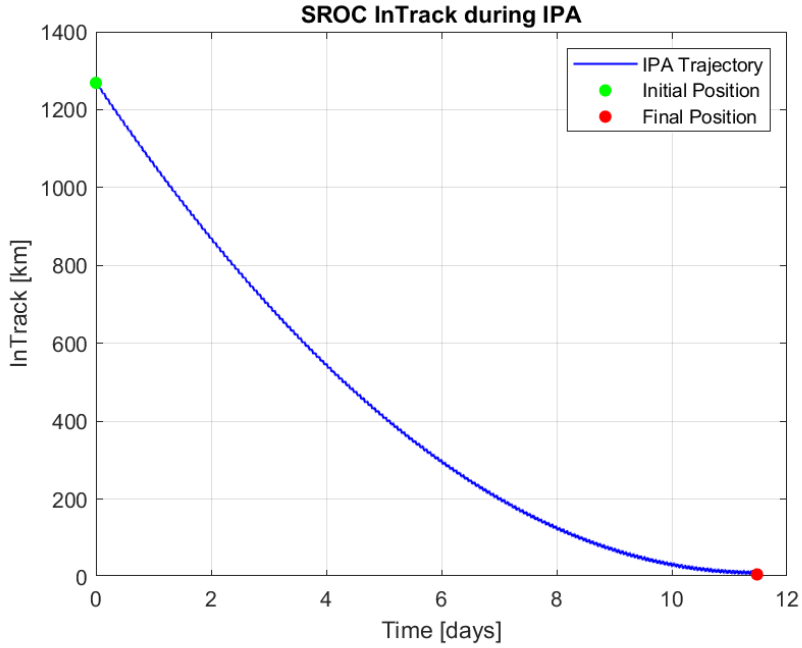


Figure 6.16: IPA InTrack as a function of the time - LongComm&HP1 (DD)

Table 6.12 shows that, because of the longer commissioning, HP1 and IPA, the duration of the mission is almost doubled from 12.3 days to 23.5 days. The deltaV, instead, only increases by 21.33% with respect to the nominal case. Because of the longer HP1, the total duration and deltaV are even higher than the ones for the longer commissioning case, whose deltaV-down solution lasts for 23 days and required a total deltaV of 8.245 m/s.

Table 6.12: DeltaV and duration comparison between the nominal and the LongComm&HP1 (DeltaV-down) scenarios

Nominal Scenario - LongComm&HP1 (DD) - Duration				
Mission Segment	**Nominal [day]**		**LongComm&HP1 (DD) [day]**	
Comm	5.000		10.000	
HP1	0.188		0.563	
IPA	5.760	5.843	11.480	11.563
HP2Ins	0.083		0.083	
ZRV2	-		-	
Total PreHP2	11.031		22.126	
Total Mission	12.361		23.456	

Nominal Scenario -LongComm&HP1 (DD) - DeltaV				
Mission Segment	**Nominal [m/s]**		**LongComm&HP1 (DD) [m/s]**	
Comm	-		-	
HP1	0.489		0.887	
IPA	0.485	2.188	0.906	3.303
HP2Ins	1.280		2.117	
ZRV2	0.423		0.280	
Total PreHP2	2.677		4.190	
Total Mission	7.092		8.605	

6.1.3.2 Time-Down Solution

The time-down solution presents the same problems as the longer commissioning time-down solution: to respect the 200 km InTrack limit on the 24-hour propagation after the IPA, the relative velocity during the IPA cannot be too high, thus limiting the minimum duration of the segment. Table 6.12 reports the total deltaV and duration of this solution and the ones of the nominal scenario. This solution recovers 15 hours and 20 minutes from the deltaV-down solution, while it requires 23.75% more deltaV than the nominal scenario. In case this recovery in time was not considered satisfying enough, three alternative time-down solutions were analysed.

Table 6.13: DeltaV and duration comparison between the nominal and the LongComm&HP1 (Time-down) scenarios

Nominal Scenario - LongComm&HP1 (TD) - Duration				
Mission Segment	**Nominal [day]**		**LongComm&HP1 (TD) [day]**	
Comm	5.000		10.000	
HP1	0.188		0.563	
IPA	5.760	5.843	10.840	10.923
HP2Ins	0.083		0.083	
ZRV2	-		-	
Total PreHP2	11.031		21.486	
Total Mission	12.361		22.816	

Nominal Scenario -LongComm&HP1 (TD) – DeltaV				
Mission Segment	**Nominal [m/s]**		**LongComm&HP1 (TD) [m/s]**	
Comm	-		-	
HP1	0.489		0.887	
IPA	0.485	2.188	0.908	3.474
HP2Ins	1.280		2.279	
ZRV2	0.423		0.287	
Total PreHP2	2.677		4.361	
Total Mission	7.092		8.776	

6.1.3.3 Alternative Time-Down Solutions – ATD-1

Figure 6.17 shows the results of the analysis (purple circles) and a surface interpolating them; here, the total deltaV is shown as a function of the IPA InTrack target and the total duration. As for the previous ATD-1 analysed, the adjective "total" refers to the combination of the IPA + HP2 insertion and ZRV2 manoeuvre. These results were obtained by iterating on the following parameters:

- InTrack target: [2:0.5: 7] km;
- Duration: [2:0.1:6] days;

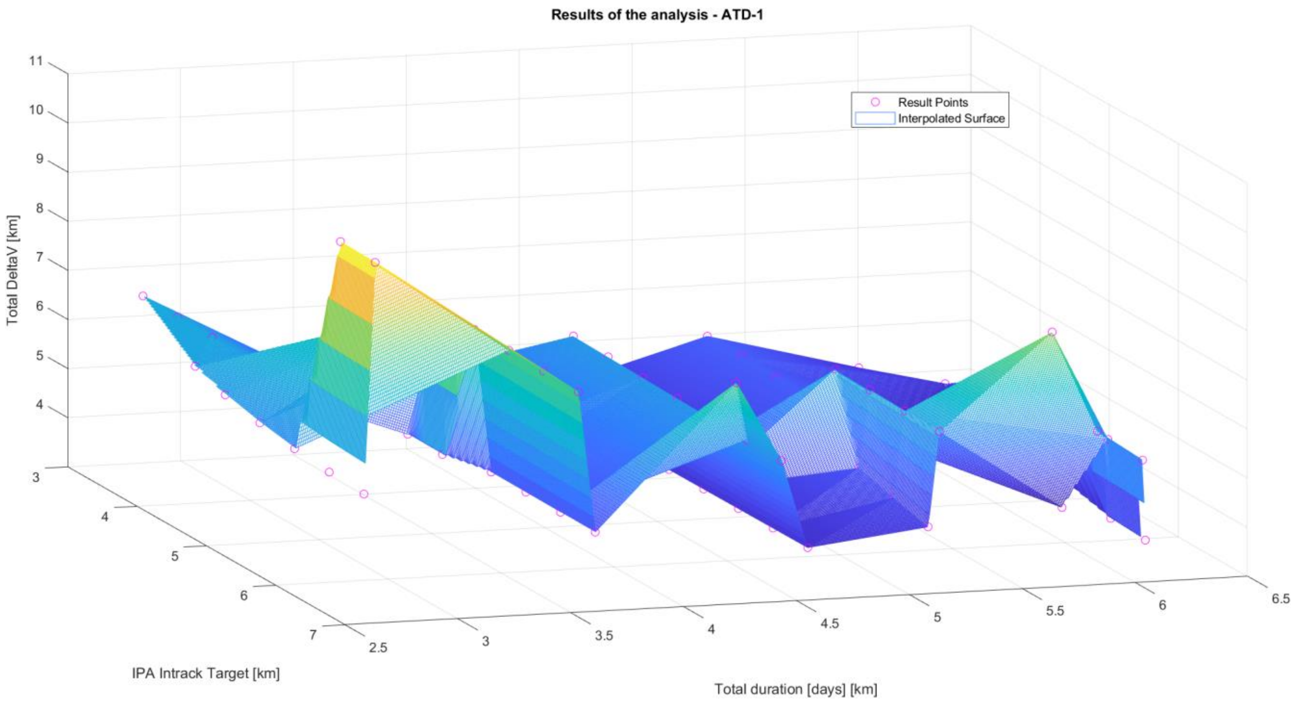


Figure 6.17: 3d plot of the results of the ATD-1 analysis

The optimal solution was selected to reduce the total duration while also avoiding increasing too much the deltaV. The red box in Figure 6.18 highlights the set of solutions that were considered for ATD-1: they were selected because, although they are approximately 1 day longer than the shortest solutions, they require about 1 m/s less.

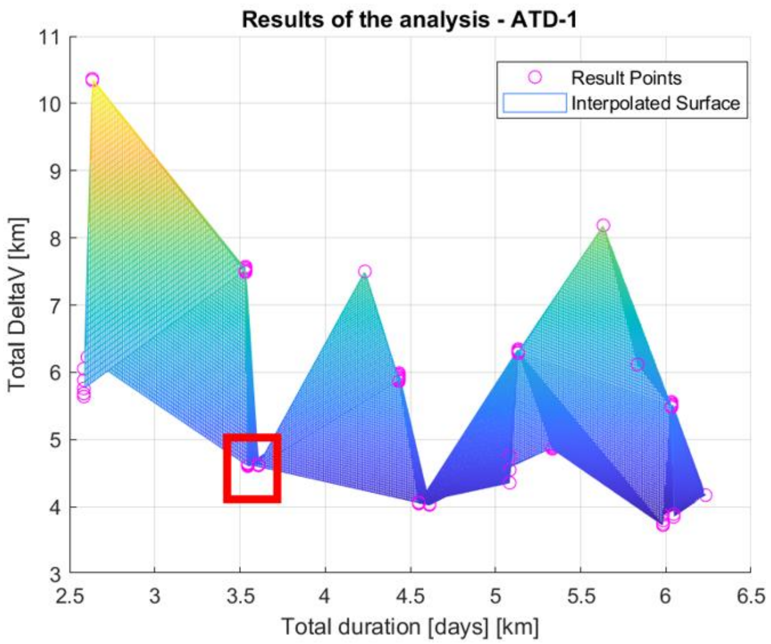


Figure 6.18: Optimal solutions – ATD-1

Figure 6.19 shows the trend of the IPA and total duration as a function of the IPA duration for a fixed InTrack target. As expected, the deltaV cost of the IPA decrease with the increase of its total duration since it requires a longer IPA, thus a lower change of the kinetic energy. The total deltaV cost, although has a decreasing trend, does not show this behaviour for every solution. This is caused by the HP2 insertion and ZRV2 manoeuvres, whose contribution to the total deltaV has already been analysed in Sub-section 6.1.2.3. It is probable that with a finer step on the duration of both the IPA and the HP2 insertion and of the InTrack target of the IPA, these deltaV differences could decrease, with the total deltaV progressively decreasing with the IPA duration for all the solutions.

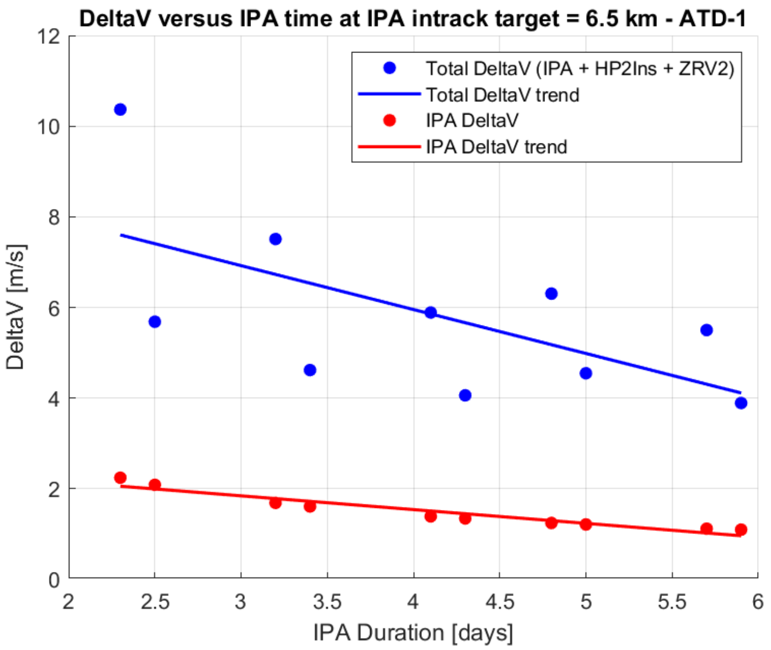


Figure 6.19: DeltaV trend - ATD-1

Table 6.14 reports in each row the following properties of every valid solution: InTrack target, duration and deltaV of the IPA, duration and deltaV of the HP2 insertion, deltaV for the ZRV2 manoeuvre, total deltaV and total duration of the IPA + HP2 insertion + ZRV2 sequence. Finally, it also shows, in the last column, the minutes which would take SROC to cross the 200 m InTrack position if its orbit was propagated after the IPA instead of performing the HP2 insertion. The row highlighted in orange is the solution with the lowest duration, while the one in yellow is the selected one. As seen for the longer commissioning's ATD-1, SROC crosses the InTrack 200 m limit in less than one hour, thus not achieving the desired safety for Space Rider.

Table 6.14: Detailed results for the ATD-1 solution

IPA Intrack [km]	IPA Duration [days]	IPA DeltaV [m/s]	HP2Ins Duration [hr]	HP2Ins DeltaV	ZRV2 DeltaV	Total DeltaV	Total time	Time to 200 [min]	Time to 1000 [min]
6.5	2.3	2.243	8	4.034	4.090	10.367	2.633	58	57
7	2.3	2.242	8	4.022	4.080	10.344	2.633	59	58
3	2.5	2.091	8	3.683	0.621	6.395	2.833	46	44
3.5	2.5	2.091	8	3.671	0.613	6.375	2.833	46	45
4	2.5	2.090	8	3.660	0.604	6.354	2.833	47	46
4.5	2.5	2.089	2.5	3.751	0.386	6.227	2.604	48	47
5	2.5	2.088	2	3.627	0.337	6.053	2.583	48	47
5.5	2.5	2.088	2	3.507	0.289	5.883	2.583	49	48
6	2.5	2.087	2	3.386	0.287	5.760	2.583	50	49
6.5	2.5	2.086	2	3.266	0.333	5.685	2.583	50	49
7	2.5	2.085	2	3.146	0.410	5.641	2.583	51	50
5	3.2	1.689	8	3.258	2.626	7.573	3.533	60	59
5.5	3.2	1.688	8	3.247	2.616	7.551	3.533	61	60
6	3.2	1.688	8	3.235	2.607	7.530	3.533	62	61
6.5	3.2	1.687	8	3.224	2.598	7.509	3.533	63	62
7	3.2	1.687	8	3.212	2.588	7.487	3.533	63	62
3	3.4	1.612	3.5	2.822	0.218	4.652	3.546	49	47
3.5	3.4	1.611	3.5	2.779	0.244	4.635	3.546	49	48
4	3.4	1.611	3.5	2.736	0.274	4.621	3.546	50	49
4.5	3.4	1.610	3.5	2.694	0.307	4.611	3.546	51	50
5	3.4	1.610	3.5	2.651	0.342	4.602	3.546	52	51
5.5	3.4	1.609	5	2.752	0.269	4.631	3.608	53	51
6	3.4	1.609	5	2.728	0.287	4.624	3.608	53	52
6.5	3.4	1.608	5	2.704	0.306	4.618	3.608	54	53
7	3.4	1.607	5	2.680	0.325	4.613	3.608	55	54
7	3.9	1.443	8	1.776	4.282	7.501	4.233	72	71
4	4.1	1.392	8	2.903	1.694	5.989	4.433	63	61
4.5	4.1	1.391	8	2.892	1.685	5.969	4.433	63	62
5	4.1	1.391	8	2.881	1.676	5.948	4.433	64	63
5.5	4.1	1.391	8	2.870	1.667	5.928	4.433	65	64
6	4.1	1.390	8	2.859	1.658	5.907	4.433	66	65
6.5	4.1	1.390	8	2.848	1.649	5.887	4.433	67	66
7	4.1	1.389	8	2.837	1.640	5.867	4.433	68	66
3	4.3	1.347	7.5	1.914	0.765	4.026	4.613	51	49
3.5	4.3	1.347	7.5	1.914	0.770	4.031	4.613	52	50
4	4.3	1.346	7.5	1.915	0.775	4.037	4.613	53	51
4.5	4.3	1.346	7.5	1.916	0.781	4.043	4.613	54	52
5	4.3	1.345	6	1.773	0.929	4.047	4.550	55	53
5.5	4.3	1.345	6	1.773	0.934	4.052	4.550	56	54
6	4.3	1.344	6	1.773	0.939	4.056	4.550	56	55
6.5	4.3	1.344	6	1.773	0.945	4.061	4.550	57	56
7	4.3	1.344	6	1.772	0.950	4.066	4.550	58	57
5.5	4.8	1.245	8	1.769	3.332	6.346	5.133	74	73
6	4.8	1.245	8	1.758	3.323	6.325	5.133	75	74
6.5	4.8	1.244	8	1.747	3.313	6.304	5.133	76	75
7	4.8	1.244	8	1.736	3.304	6.284	5.133	77	76
3	5	1.214	8	2.742	1.003	4.958	5.333	64	62
3.5	5	1.214	8	2.731	0.994	4.939	5.333	65	63
4	5	1.213	8	2.721	0.985	4.919	5.333	66	64
4.5	5	1.213	8	2.710	0.977	4.900	5.333	67	65
5	5	1.212	8	2.700	0.968	4.880	5.333	68	66
5.5	5	1.212	8	2.690	0.959	4.861	5.333	69	67

6	5	1.212	2	2.902	0.640	4.753	5.083	70	68
6.5	5	1.211	2	2.782	0.554	4.547	5.083	71	69
7	5	1.211	2	2.662	0.484	4.356	5.083	72	70
7	5.3	1.172	8	2.486	4.528	8.187	5.633	15	13
7	5.5	1.144	8	0.754	4.220	6.118	5.833	83	81
5	5.7	1.120	8	1.850	2.591	5.561	6.033	77	76
5.5	5.7	1.120	8	1.839	2.581	5.541	6.033	79	77
6	5.7	1.119	8	1.829	2.572	5.520	6.033	80	78
6.5	5.7	1.119	8	1.818	2.563	5.500	6.033	81	79
7	5.7	1.119	8	1.808	2.554	5.480	6.033	82	80
3	5.9	1.100	2.5	2.655	0.397	4.151	6.004	66	65
3.5	5.9	1.100	8	2.647	0.427	4.173	6.233	68	66
4	5.9	1.099	2	2.626	0.287	4.013	5.983	69	67
4.5	5.9	1.099	2	2.507	0.273	3.879	5.983	70	68
5	5.9	1.099	2	2.388	0.309	3.796	5.983	71	69
5.5	5.9	1.098	2	2.270	0.382	3.750	5.983	72	70
6	5.9	1.098	2	2.152	0.475	3.725	5.983	73	72
6.5	5.9	1.098	3.5	2.520	0.273	3.890	6.046	75	73

Table 6.15 shows a comparison between the ATD-1 solution and the nominal scenario: most of the delay is recovered and the duration of the whole mission is just 3 days higher. Even the total required deltaV, although 2.9 m/s higher than the nominal one, is well below the 20 m/s limit.

Table 6.15: DeltaV and duration comparison between the nominal and the LongComm&HP1 (ATD-1) scenarios

<table>
<tr><th colspan="5">Nominal Scenario - LongComm&HP1 (ATD-1) - Duration</th></tr>
<tr><th>Mission Segment</th><th colspan="2">Nominal [day]</th><th colspan="2">LongComm&HP1 (ATD-1) [day]</th></tr>
<tr><td>Comm</td><td colspan="2">5.000</td><td colspan="2">10.000</td></tr>
<tr><td>HP1</td><td colspan="2">0.188</td><td colspan="2">0.563</td></tr>
<tr><td>IPA</td><td>5.760</td><td rowspan="3">5.843</td><td>3.400</td><td rowspan="3">3.546</td></tr>
<tr><td>HP2Ins</td><td>0.083</td><td>0.146</td></tr>
<tr><td>ZRV2</td><td>-</td><td>-</td></tr>
<tr><td>Total PreHP2</td><td colspan="2">11.031</td><td colspan="2">14.109</td></tr>
<tr><td>Total Mission</td><td colspan="2">12.361</td><td colspan="2">15.439</td></tr>
</table>

<table>
<tr><th colspan="5">Nominal Scenario -LongComm&HP1 (ATD-1) - DeltaV</th></tr>
<tr><th>Mission Segment</th><th colspan="2">Nominal [m/s]</th><th colspan="2">LongComm&HP1 (ATD-1) [m/s]</th></tr>
<tr><td>Comm</td><td colspan="2">-</td><td colspan="2">-</td></tr>
<tr><td>HP1</td><td colspan="2">0.489</td><td colspan="2">0.887</td></tr>
<tr><td>IPA</td><td>0.485</td><td rowspan="3">2.188</td><td>1.610</td><td rowspan="3">4.602</td></tr>
<tr><td>HP2Ins</td><td>1.280</td><td>2.651</td></tr>
<tr><td>ZRV2</td><td>0.423</td><td>0.342</td></tr>
<tr><td>Total PreHP2</td><td colspan="2">2.677</td><td colspan="2">5.489</td></tr>
<tr><td>Total Mission</td><td colspan="2">7.092</td><td colspan="2">9.904</td></tr>
</table>

6.1.3.4 Alternative Time-Down Solutions – ATD-2

Figure 6.20 shows the total deltaV as a function of the total duration and the IPA InTrack target; the term “total deltaV” refers to the deltaV required by the IPA brake + HP2 insertion è ZRV2 manoeuvres, while “total time” refers to the duration of the propagation segments of the braked IPA and the HP2 insertion. These results were obtained by iterating on the following parameters:

- IPA InTrack target: [5, 0.5, 7] km;
- Braked IPA Duration [4.5:0.1:6.5] days;

The range of possible duration of the braked part of the IPA was decreased because, as it has been seen form the longer commissioning’s ATD-2, it is not necessary to evaluate too low duration since they produce a too fast IPA which would not respect the 200 m InTrack limit on a hypothetical successive propagation segment. As expected, all the valid solutions are in a small interval of long total durations, approximately from 5.88 to 6 days.

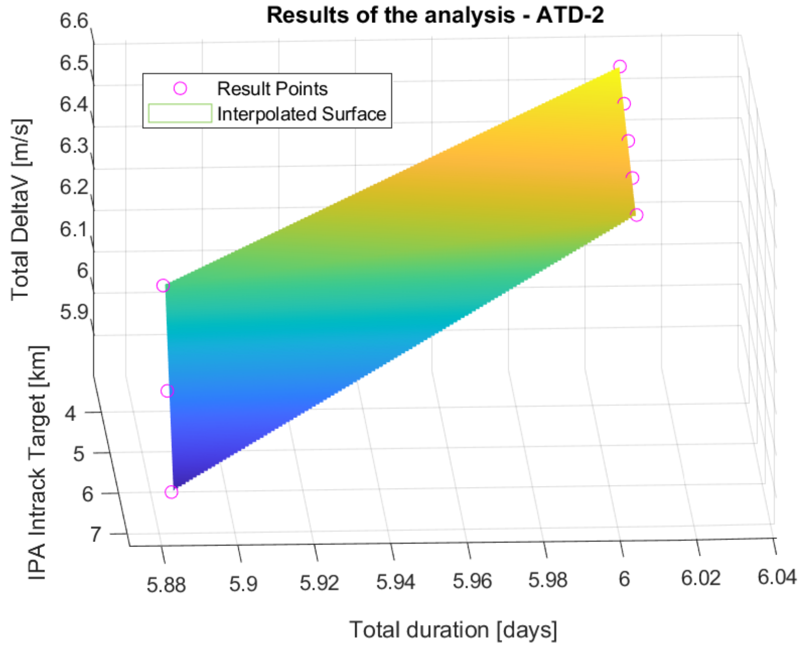


Figure 6.20: 3D plot of the results of the ATD-2 analysis

Figure 6.21 shows the total deltaV as a function of the total time. For this ATD-2, defining the optimal result was a simple task, since the one with the shortest duration is also the one with the lowest deltaV. This may seem counterintuitive since, usually the solutions present a total deltaV trend decreasing with the total duration. As reported in Table 6.16, the acceptable solutions are obtained only for a duration of the braked segment of the IPA of 5.8 days, so the differences in the total durations and deltaV are caused by the different combinations of IPA InTrack targets and HP2 insertion durations. For this reason, the analysis of the deltaV trend, which considers the effect of different IPA duration, cannot be applied here.

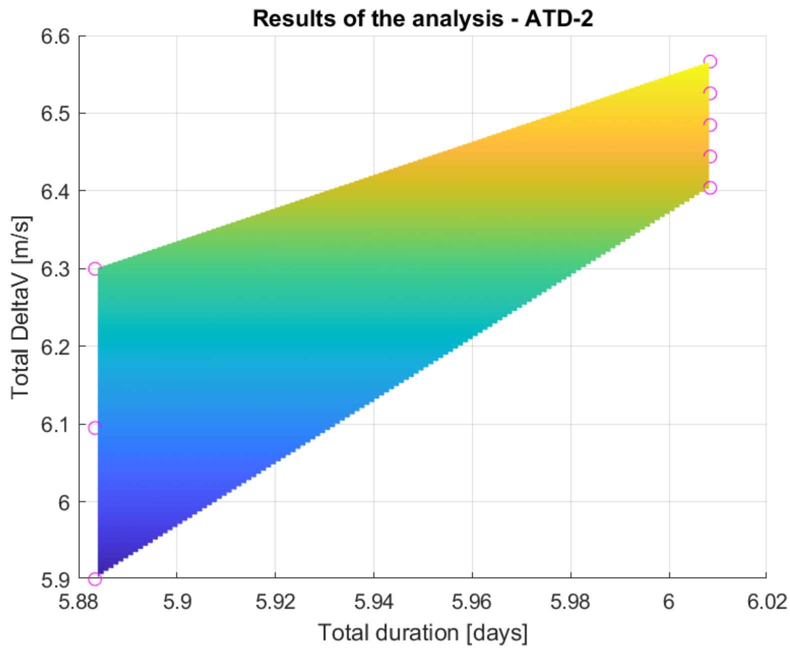


Figure 6.21: Total DeltaV as function of the total duration for the ATD-2 analysis

It is noted that each row in Table 6.16 refers to one valid ATD-2 solution, with the same column already described for the previous ATD-2 solution: InTrack target, duration and deltaV of the IPA Brake, duration and deltaV of the HP2 insertion, deltaV for the ZRV2 manoeuvre, total deltaV and total duration. The adjective "total" refers to the IPA + IPA brake + HP2 insertion + ZRV2 sequence. In fact, to evaluate the total duration, 2.4 days were added: the IPA with no brake would last 3.7 days, but since the brake is performed 24 hours before its theoretical end, the duration of this propagation segment is just 2.4 days. To evaluate the total deltaV, 1.610 m/s were added to consider the deltaV required to perform the first part of the IPA. The selected solution is the row highlighted in yellow.

Table 6.16: Detailed results for the ATD-2 solution

IPA Intrack [km]	Braked IPA Duration [days]	IPA DeltaV [m/s]	HP2Ins Duration [hr]	HP2Ins DeltaV	ZRV2 DeltaV	Total DeltaV	Total time
3.5	5.8	0.931	5	4.533	1.102	8.176	8.408
4	5.8	0.932	5	4.513	1.081	8.135	8.408
4.5	5.8	0.932	5	4.493	1.060	8.094	8.408
5	5.8	0.932	5	4.473	1.039	8.054	8.408
5.5	5.8	0.933	5	4.453	1.018	8.014	8.408
6	5.8	0.933	2	4.530	0.837	7.909	8.283
6.5	5.8	0.933	2	4.418	0.744	7.705	8.283
7	5.8	0.934	2	4.306	0.661	7.510	8.283

Figure 6.22 shows the InTrack during the whole IPA as a function of the time from its start. Most of the relative distance is recovered in 2.4 days during the first part of the IPA, while the remaining 370 km are covered in 5.8 days. It is, of course, a great downgrade with respect to the ATD-1, but it is necessary to increase the safety of the mission to an acceptable level.

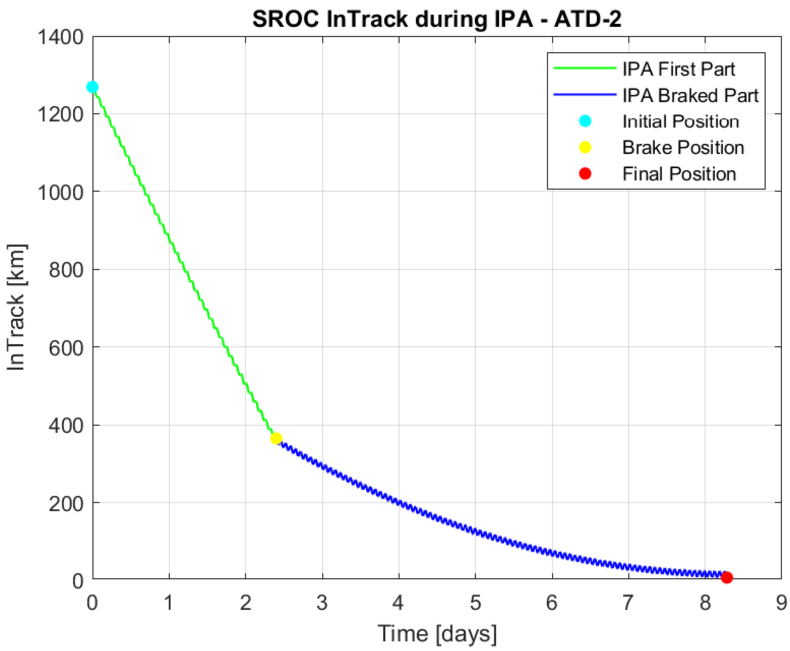


Figure 6.22: InTrack as a function of the time - ATD-2

Finally, Table 6.16 shows the comparison between the durations and the deltaV of the ATD-2 and the nominal scenario. Here, the properties of the IPA refer to it as a whole, including both the unbraked and braked parts.

Table 6.17: DeltaV and duration comparison between the nominal and the LongComm&HP1 (ATD-1) scenarios

Nominal Scenario - LongComm&HP1 (ATD-2) - Duration				
Mission Segment	**Nominal [day]**		**LongComm&HP1 (ATD-2) [day]**	
Comm	5.000		10.000	
HP1	0.188		0.563	
IPA	5.760	5.843	8.200	8.283
HP2Ins	0.083		0.083	
ZRV2	-		-	
Total PreHP2	11.031		18.846	
Total Mission	12.361		20.176	

Nominal Scenario -LongComm&HP1 (ATD-2) - DeltaV				
Mission Segment	**Nominal [m/s]**		**LongComm&HP1 (ATD-2) [m/s]**	
Comm	-		-	
HP1	0.489		0.887	
IPA	0.485	2.188	2.543	7.510
HP2Ins	1.280		4.306	
ZRV2	0.423		0.661	
Tota PreHP2	2.677		8.397	
Total Mission	7.092		12.811	

6.1.3.5 Alternative Time-Down Solutions – ATD-3

Figure 6.23 shows the total deltaV as a function of the IPA InTrack target and the total duration. The meaning of the adjective "total" in both this graph and the following graphs and tables are the same used for the same graphs and tables in the previous sub-section. This analysis was performed considering the following vectors:

- IPA InTrack target: [5, 0.5, 7] km;
- Braked IPA Duration: [3:0.1:6.5] days;

A higher number of valid solutions than for the ATD-2 was obtained, although only for a total number of 3 braked IPA durations.

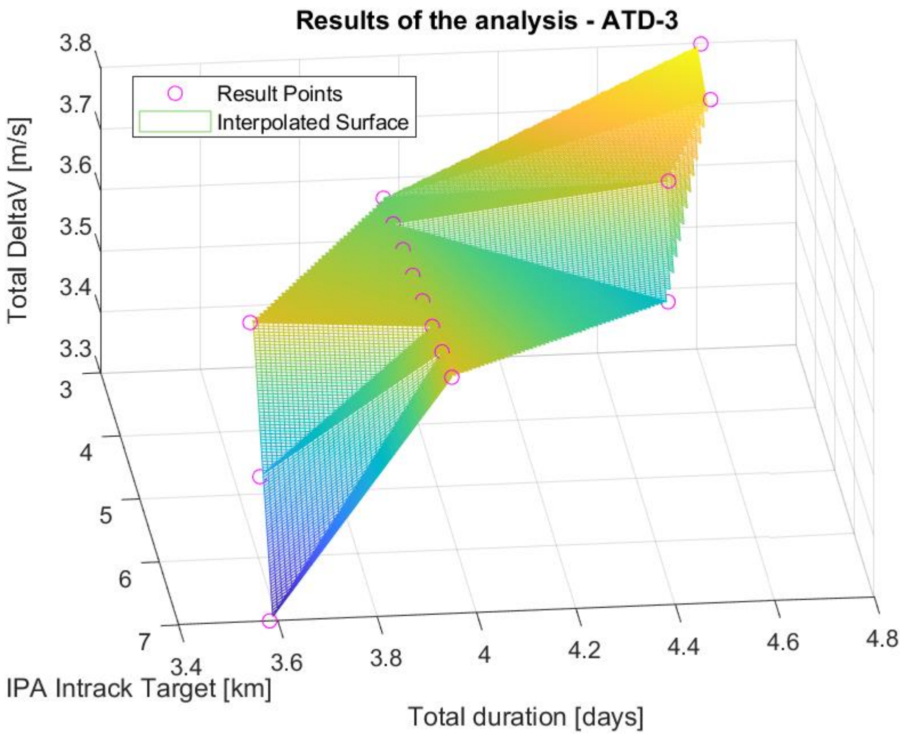


Figure 6.23: 3D plot of the results of the ATD-3 analysis

Figure 6.24 shows the total deltaV as a function of the total duration. The selected solution in the red box both minimizes the deltaV and the duration. Again, the trend of the total duration is different from the expected one, with the total deltaV not decreasing with the total duration.

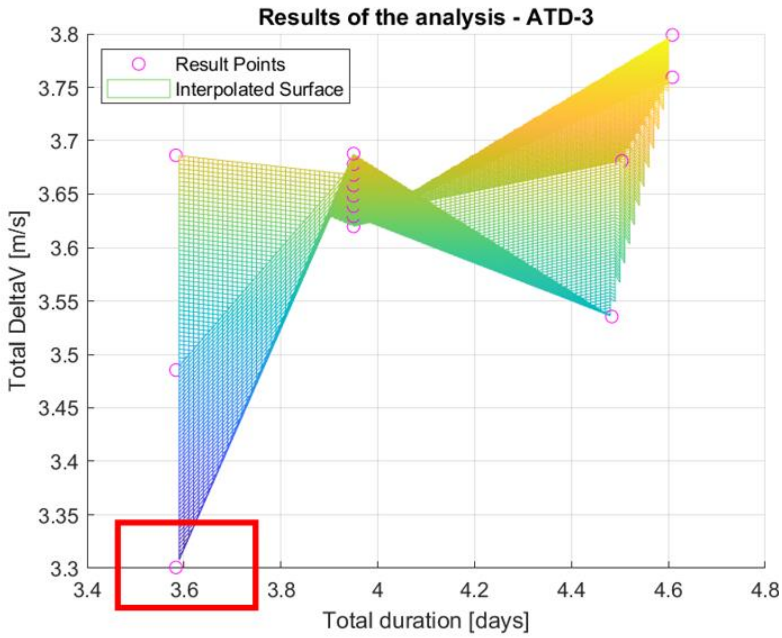


Figure 6.24: Total DeltaV as function of the total duration for the ATD-3 analysis

As shown in Table 6.18, the trend of the IPA brake deltaV is to increase with the duration, with the contributions of the HP2 insertion and ZRV2 varying even more in the final deltaV. The discrepancy between the data reported in the table and the expected behaviour could probably be eliminated by using finer steps for the duration and InTrack target values used for the analysis. However, this would have increased the analysis time too much and it would not have been possible to perform analysis this fine on all the possible variant scenarios. Moreover, this analysis gives more conservative results: with finer steps, the total deltaV and duration should not significantly change, and slowly better results should be obtained.

Table 6.18: Detailed results for the ATD-3 solution

IPA Intrack [km]	IPA Duration [days]	IPA DeltaV [m/s]	HP2Ins Duration [hr]	HP2Ins DeltaV	ZRV2 DeltaV	Total DeltaV	Total time
6	3.5	1.021	2	1.992	0.673	5.296	6.483
6.5	3.5	1.022	2	1.880	0.584	5.095	6.483
7	3.5	1.022	2	1.769	0.510	4.910	6.483
3.5	3.7	1.021	6	1.312	1.287	5.230	6.850
4	3.7	1.022	6	1.313	1.294	5.239	6.850
4.5	3.7	1.022	6	1.315	1.301	5.248	6.850
5	3.7	1.023	6	1.317	1.309	5.258	6.850
5.5	3.7	1.023	6	1.319	1.316	5.268	6.850
6	3.7	1.024	6	1.321	1.323	5.278	6.850
6.5	3.7	1.024	6	1.323	1.331	5.288	6.850
7	3.7	1.025	6	1.325	1.338	5.298	6.850
3	4.4	1.023	5	2.161	0.614	5.409	7.508
3.5	4.4	1.024	5	2.141	0.595	5.369	7.508
4	4.4	1.024	2.5	2.127	0.530	5.291	7.404
4.5	4.4	1.024	2	2.062	0.449	5.145	7.383

Figure 6.25 shows SROC's InTrack as a function of its duration: more than 1200 km are covered during the first part of the IPA, while to cover the remaining relative distance 3.5 days are required.

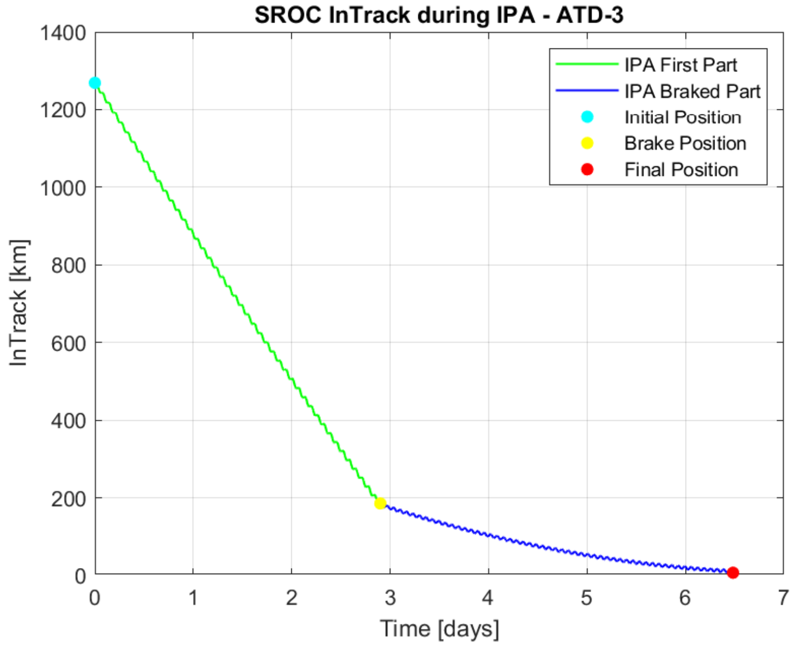


Figure 6.25: InTrack as a function of the time - ATD-3

Table 6.19 and Table 6.20 compare the duration and deltaV of the nominal scenario and all the time-down solutions. Generally, the same conclusion made for the longer commissioning analysis can be applied here: the shortest option is the ATD-1, but it is not safe enough. The TD and ATD-2 solution meet the safety constraints of the project; however, they present a higher total duration (respectively 22.816 and 20.176 days). The ATD-2 is shorter than 2 days and 15 hours and 22 minutes, however, since it includes an additional manoeuvre to slow down the IPA, it costs 12.811 m/s instead of 9.904 m/s. Finally, the ATD-3 could be a good compromise between these two options, since it is even shorter and requires a 10.212 m/s deltaV. However, this manoeuvre is performed 12 hours before the end of the unbraked IPA, which means that if no braking manoeuvre was performed, SROC would cross the 200 m InTrack limit in approximately 13 hours. For this reason, the solution could be considered not safe enough. The final choice between the TD and ATD-2 could be the presence of other variant mission segments: in case any of them increased the total deltaV even more, the additional cost of the ATD-2 could not be worth as much as the time recovered with respect to the TD solution.

Table 6.19:Duration comparison between the nominal and all the time-down solutions in case of a longer commissioning and HP1

<table>
<tr><th colspan="11">Nominal Scenario - LongComm&HP1 Duration</th></tr>
<tr><th>Mission Segment</th><th colspan="2">Nominal [day]</th><th colspan="2">LongComm& HP1 (TD) [day]</th><th colspan="2">LongComm& HP1 (ATD-1) [day]</th><th colspan="2">LongComm& HP1 (ATD-2) [day]</th><th colspan="2">LongComm& HP1 (ATD-3) [day]</th></tr>
<tr><td>Comm</td><td colspan="2">5.000</td><td colspan="2">10.000</td><td colspan="2">10</td><td colspan="2">10</td><td colspan="2">10.000</td></tr>
<tr><td>HP1</td><td colspan="2">0.188</td><td colspan="2">0.563</td><td colspan="2">0.563</td><td colspan="2">0.563</td><td colspan="2">0.563</td></tr>
<tr><td>IPA</td><td>5.760</td><td rowspan="3">5.843</td><td>10.840</td><td rowspan="3">10.923</td><td>3.4</td><td rowspan="3">3.55</td><td>8.2</td><td rowspan="3">8.28</td><td>6.400</td><td rowspan="3">6.483</td></tr>
<tr><td>HP2Ins</td><td>0.083</td><td>0.083</td><td>0.146</td><td>0.083</td><td>0.083</td></tr>
<tr><td>ZRV2</td><td>-</td><td>-</td><td>-</td><td>-</td><td>-</td></tr>
<tr><td>Total PreHP2</td><td colspan="2">11.031</td><td colspan="2">21.486</td><td colspan="2">14.109</td><td colspan="2">18.846</td><td colspan="2">17.046</td></tr>
<tr><td>Total Mission</td><td colspan="2">12.361</td><td colspan="2">22.816</td><td colspan="2">15.439</td><td colspan="2">20.176</td><td colspan="2">18.376</td></tr>
</table>

Table 6.20: DeltaV comparison between the nominal and all the time-down solutions in case of a longer commissioning

Nominal Scenario - LongComm&HP1 DeltaV										
Mission Segment	**Nominal [m/s]**		**LongComm& HP1 (TD) [m/s]**		**LongComm& HP1 (ATD-1) [m/s]**		**LongComm& HP1 (ATD-2) [m/s]**		**LongComm& HP1 (ATD-3) [m/s]**	
Comm	-		-		-		-		-	
HP1	0.489		0.887		0.887		0.887		0.887	
IPA	0.485	2.188	0.908	3.47	1.610	4.6	2.543	7.51	2.632	4.910
HP2Ins	1.280		2.279		2.651		4.306		1.769	
ZRV2	0.423		0.287		0.342		0.661		0.510	
Total PreHP2	2.667		4.361		5.489		8.397		5.797	
Total Mission	7.092		8.776		9.904		12.811		10.212	

6.2 Variant Events After HP2

The main variant segments that take place after the IPA rendezvous are the possibility of a second inspection cycle and the demand to perform an additional manoeuvre during the EMP phase to avoid a potential collision with Space Rider. Longer durations for both HP2 and HP3 have also been considered, although only their results are reported (see Section 6.3), since their analyses only required changing the input duration given to the same Matlab function described in Section 4.3 and they did not affect the previous or the successive segments.

6.2.1 Second Observation cycle

In Section 5.3 it was described the WSE design process. As mentioned at the end, the total actual observation time during which SROC can take pictures of SR is relatively low (about 37 minutes). In case during successive design iterations of the project this value was not considered high enough, a second observation cycle would be necessary. For this reason, a second observation cycle was considered.

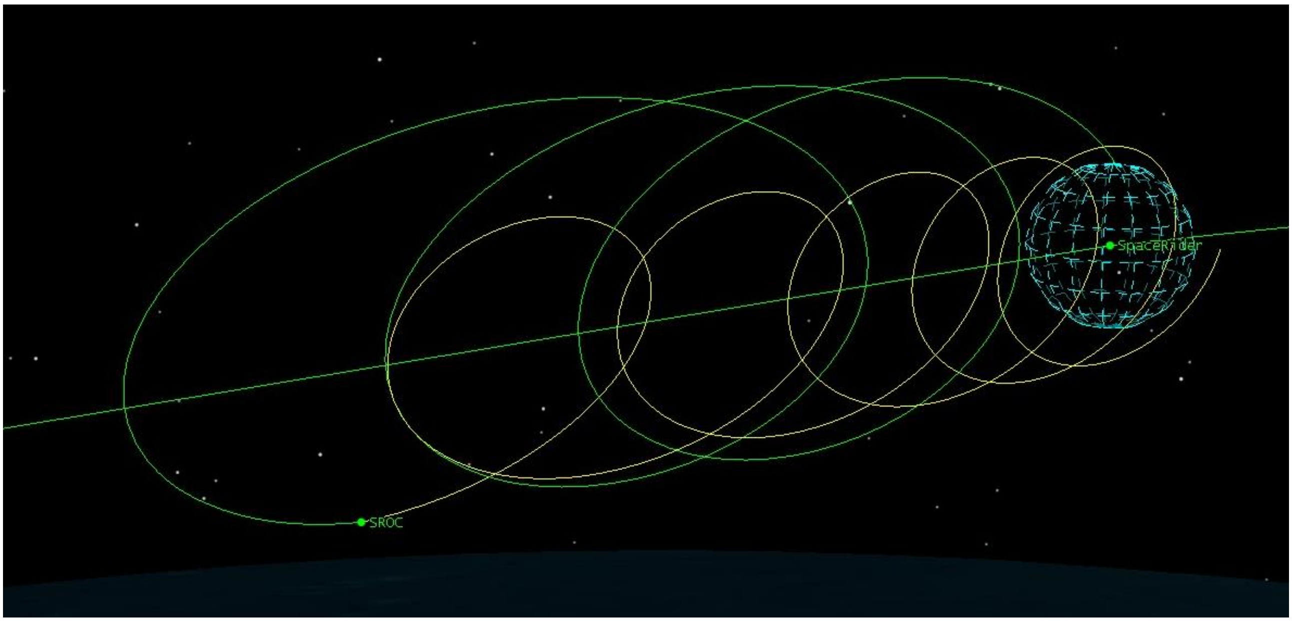


Figure 6.26: Second OPA after the first free flight

Figure 6.26 shows that the second OPA rendezvous (in green) starts after the first free flight (in yellow). After that, a second WSE insertion, a second inspection phase and a second free flight are performed. Since the second OPA starts with a different relative position and relative from SR than the first one, it is necessary to evaluate again both the OPA and WSE insertion manoeuvres. The goal of the analysis was to set a new inspection phase that was as similar as possible to the first one, so it was selected the following inspection:

- Same input geometrical parameters
- Similar actual observation time: the number of intervals is the same and their duration is similar, although the total duration for the second inspection is slightly more (46 minutes in total)
- Almost the same duration for the free flight (8.058 hours instead of 8.060 hours)

Figure 6.27 shows a comparison of the range as a function of the time for both inspections. Although the second one presents a higher maximum range, it can be seen that the number of intervals of actual observation, their duration, and the time at which they take place are almost the same.

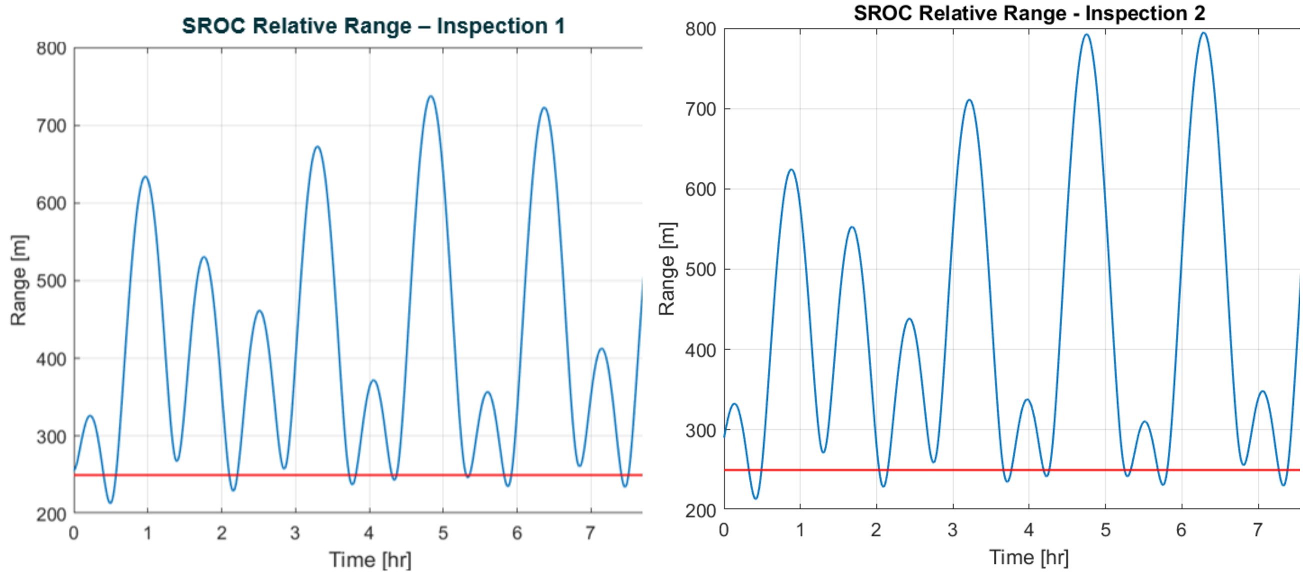


Figure 6.27: Comparison between the range as a function of the time for the first (left) and second(right) inspection

Figure 6.28 compares the RIC components during both inspections: the CrossTrack and Radial components are very close, while the InTrack reaches a lower minimum from the fourth hour of the propagation onward (which explains the higher ranges shown in Figure 6.27).

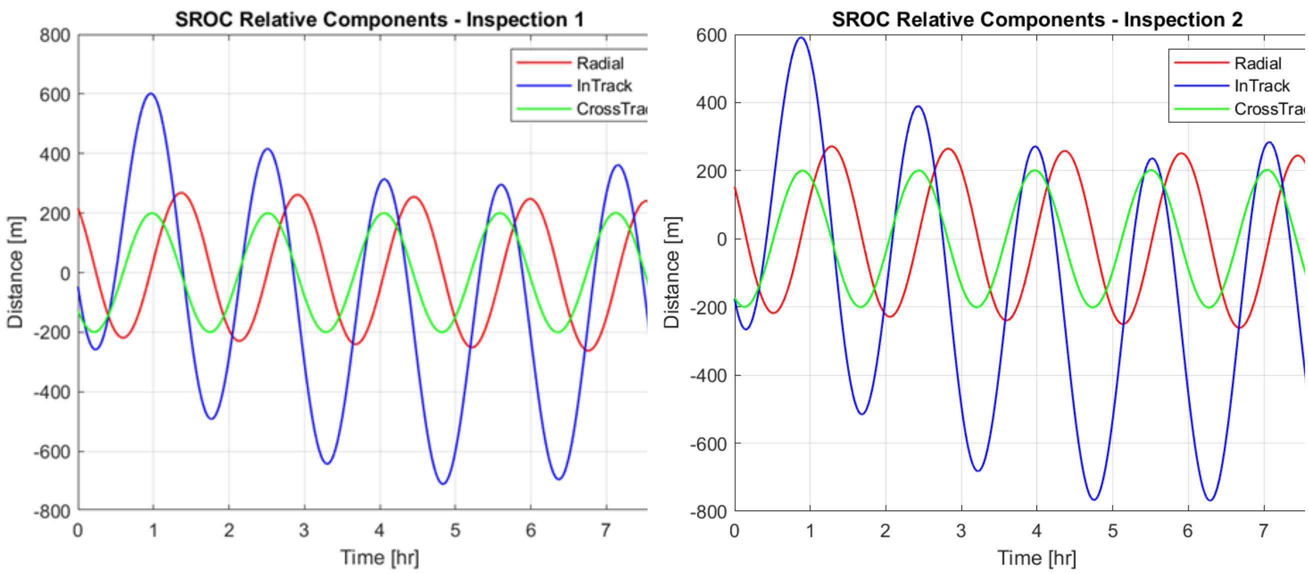


Figure 6.28: Comparison between the range as a function of the time for the first (left) and second(right) inspection

Since at the end of the second inspection sequence SROC's relative position and velocity differ from the first one, the HP3 insertion deltaV and duration may differ from the nominal observe & retrieve scenario. For this reason, the HP3 insertion and ZRV3 were re-evaluated for this different scenario. Figure 6.29 shows SROC's trajectory during the HP3 insertion, while Table 6.21 reports the deltaV and duration of both the nominal and this variant scenario. The deltaV cost of the second inspection differs from the first one: the cost of the WSE insertion is similar, with the second one requiring 0.029 m/s more, while the deltaV of the second OPA manoeuvre is significantly lower (0.095 m/s instead of 0.226 m/s). The manoeuvres after the inspection phase are identical in duration and very similar in the deltaV cost, probably because both the HP3 insertion manoeuvres are performed after similar WSEs and free flight segments.

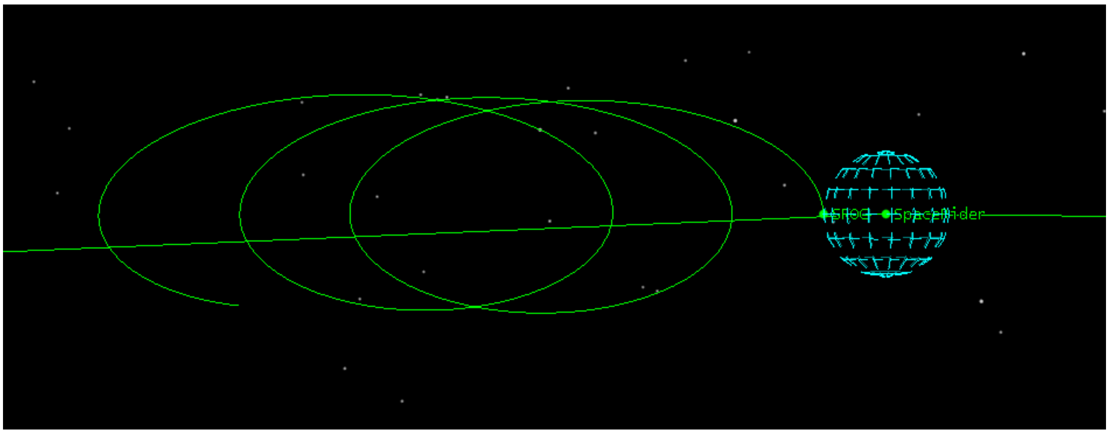


Figure 6.29: HP3 insertion after the second inspection cycle

Table 6.21: DeltaV and duration comparison between the nominal and the 2 Inspections scenarios

Nominal Scenario - 2 Inspection - Duration		
Mission Segment	**Nominal [day]**	**2 Inspections [day]**
OPA - Cycle 1	0.167	0.167
Inspection + Free Flight - Cycle 1	0.669	0.669
OPA - Cycle 2	-	0.171
Inspection + Free Flight - Cycle 2	-	0.669
HP3Ins	0.113	0.113
ZRV3	-	-
Total Mission	12.361	13.201

Nominal Scenario -2 Inspection - DeltaV		
Mission Segment	**Nominal [m/s]**	**2 Inspections [m/s]**
OPA - Cycle 1	0.266	0.266
WSE Insertion - Cycle 1	0.192	0.192
OPA - Cycle 2	-	0.095
WSE Insertion - Cycle 2	-	0.221
HP3Ins	0.221	0.208
ZRV3	0.438	0.421
Total Mission	7.092	7.377

6.2.2 End of Mission Phase Analysis

For the Observe scenario, after the completion of the inspection cycle, SROC is free to drift away from Space Rider. Since its ballistic coefficient is inferior to Space Rider's, SROC's orbit height decreases faster, which means that its orbital period becomes shorter. This may lead to an unsafe situation where SROC approaches SR from behind until a point where SROC InTrack is null; this point will be referred to as Encounter Point (EP) from here on out. Before proceeding with this analysis, it is reminded that in the STK scenario, SR's orbit is controlled, thus the effects of the drag are not considered. At this moment, it is unknown if such a fine orbit control will be applied, although it is more probable that SR will not control its semi-major axis for the whole mission, rather it will perform one or more manoeuvre to increase it and restore it to the initial value. Considering SR continuously controlled is, however, a more conservative approach because it considers the highest difference between SROC and Space Rider's semi-major axis, which means the highest difference in the two orbital periods, thus the fastest time to get to the EP.

In the nominal Observe scenario, the orbit SROC orbit was propagated until the EP. Figure 6.30 reports the evolution of the SROC's range as a function of the UTC time: the first relative maximum takes place after the end of the HP1. After that, the successive relative minimum happens during the inspection cycle; next, SROC's range increases until it reaches a relative maximum approximately on 29 December 2024. This is the moment when angle ϕ, which is the difference between the true anomaly of the two spacecrafts, is 180 degrees. From that point onwards, the range decreases since ϕ increases even more until another relative minimum is reached again. Figure 6.31 zooms on all the relative minimums found by the analysis, which propagated SROC until May 14th. After the first minimum, the other relative minimums progressively increase because SROC's semi-major axis is continuously decreased by the atmospheric drag. The time intervals between each relative minimum decrease with the time because SROC orbital period increases.

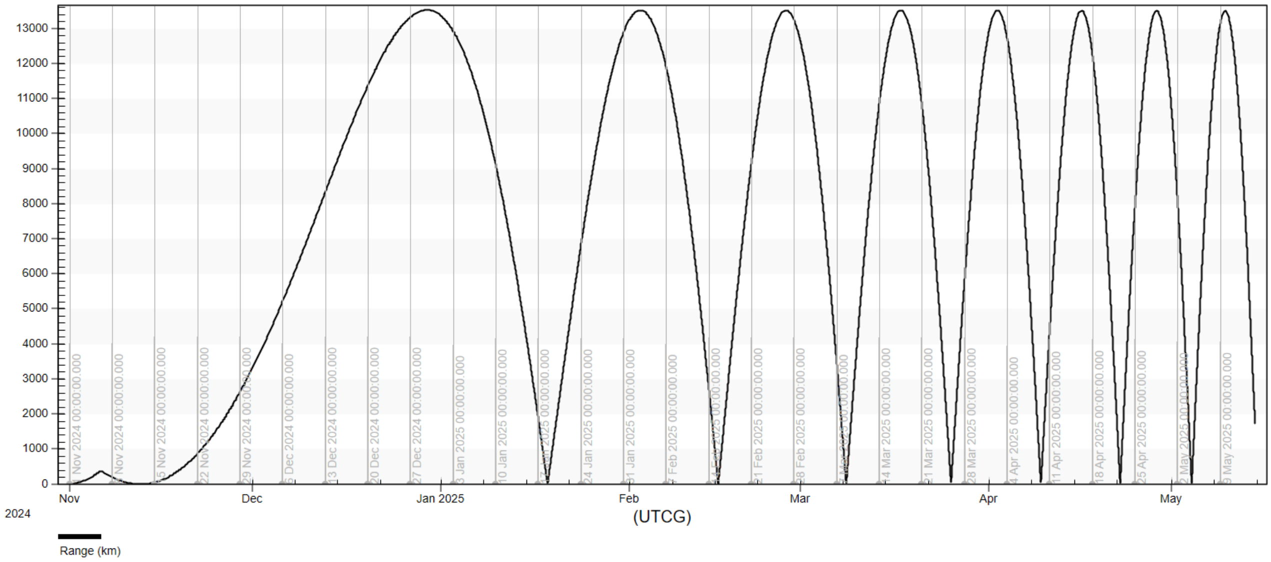


Figure 6.30: SROC's range as a function of the time

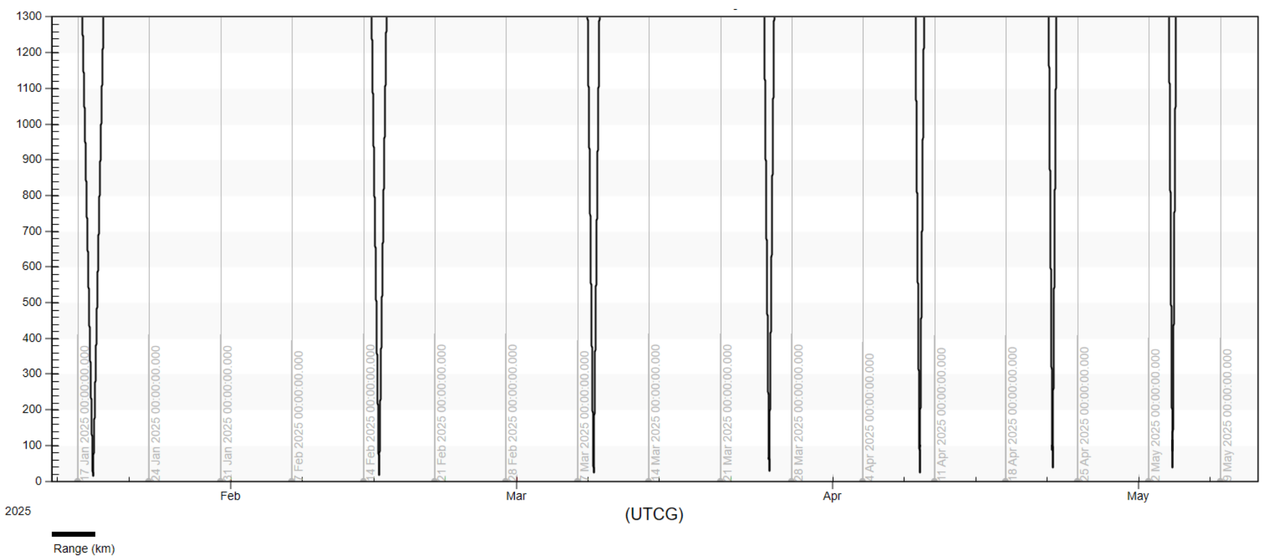


Figure 6.31: SROC's range as a function of the time - zoom on the minimum ranges

The first relative minimum is characterized by a 14.787 km range and only a radial separation of 7.734 km. It is important to notice that the EP takes place on 18 Jan 2025 at 11:20 UTCG which is more than two months after the end of the inspection cycle (13 Nov 2024 at 01:31 UTCG). Considering that the whole SR mission should take a maximum of two months to be completed, at the time of the virtual EP the mission would be completely over. However, three possible solutions to guarantee a minimum radial separation at the EP (20 km) or to further delay it were analysed to define the best option deltaV and safety-wise in case more assurance is required. Since the EP should not take place during a nominal mission, the results of this analysis were not considered in the nominal scenario.

6.2.2.1 Hohmann Manoeuvre

The first option envisages a Hohmann manoeuvre shortly after the end of the observation cycle. Its objective is to guarantee a radial separation of 20 km after its completion. To simulate the manouvre the following sequence was created:

- "ToApogee": a propagation segment that ends when SROC is at its apogee;
- "Change Radius of Periapsis": a target sequence that includes a manoeuvre segment followed by a propagation one which propagates until SROC reaches its perigee. The control parameters of the differential corrector are the thrust vectors of the manoeuvre segment, while the objective is SROC's radius of periapsis at the end of the propagation;
- "Circularize": this target sequence is composed of only a manoeuvre segment which is used to reach the desired result of the differential corrector: a null eccentricity;

After defining the sequence in STK, it is necessary to set the desired value for the radius of periapsis. Figure 6.32 shows how the semi-major axis (black), the radius of periapsis (green), the position vector magnitude (light blue) and eccentricity (purple) vary for SR. The image considers only two hours of the whole propagation time, but the trend is similar for the whole mission. Because of the gravitational field disturbances, the eccentricity of SR varies from 10^{-4} to $3 \cdot 10^{-3}$ every orbit. When the satellite is at the apogee (red box) the eccentricity is almost null, so the actual position vector magnitude is less than 1 km more than the semi-major axis. Instead, when SR is at the perigee (yellow box), the eccentricity is higher, thus obtaining a radius of periapsis equal to 6758.6 km which is sensibly less than the semi-major axis.

The propagation of both SR and SROC orbits cannot be performed with a precision high enough to guarantee that the exact moment at which the EP will actually happen is the same as the simulated one. For this reason, the desired value for SROC's radius of periapsis was set to 20 km less than the minimum SR's radius of periapsis.

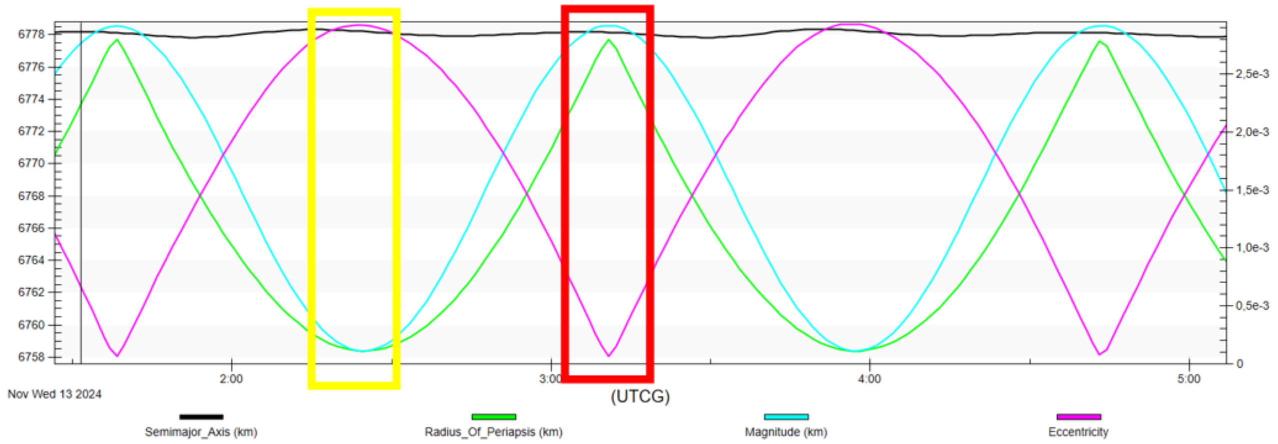


Figure 6.32: several Keplerian elements during approximately two SR's orbits

Figure 6.33 shows how SROC's semi-major axis and range from the Earth's center vary before and after the Hohmann sequence. The semi-major axis decreases in two different moments: at first, after the radius of periapsis reduction, then, after the orbit is circularized. It can also be seen that the position vector magnitude varies similarly to SR.

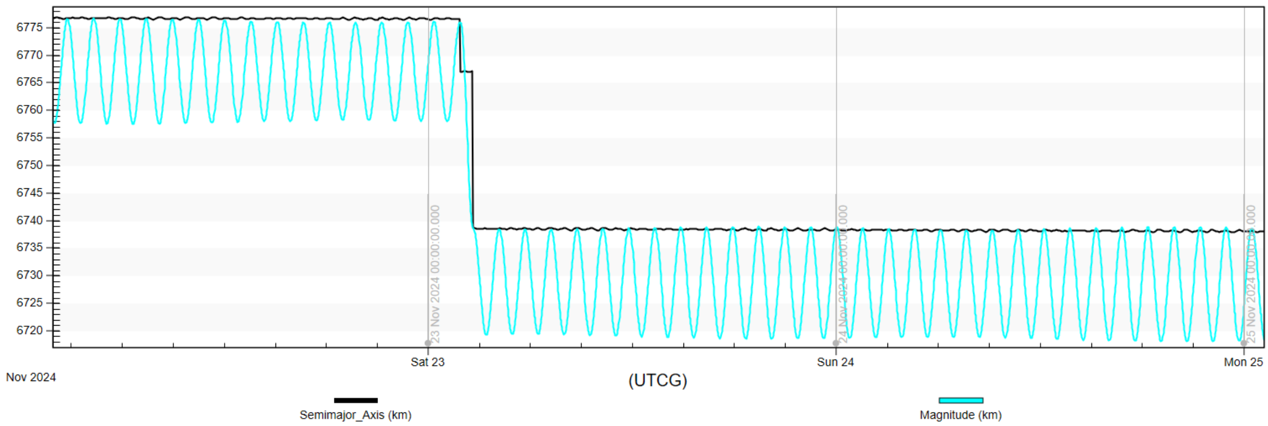


Figure 6.33: SROC semi-major axis and position vector magnitude before and after the manoeuvre

Figure 6.34 shows that SROC's range from SR as a function of the time; the red box highlights how the slope of the curve greatly increases after the completion of the Hohmann manoeuvre since it moves SROC in a lower orbit. This causes the EP to take place in less time, on 30 Nov 2024. As shown in Figure 6.35; the minimum range is reached at the first EP and it is equal to 32.25 km, while the minimum radial separation is 24.24 km and it is reached on 13 Dec 2024.

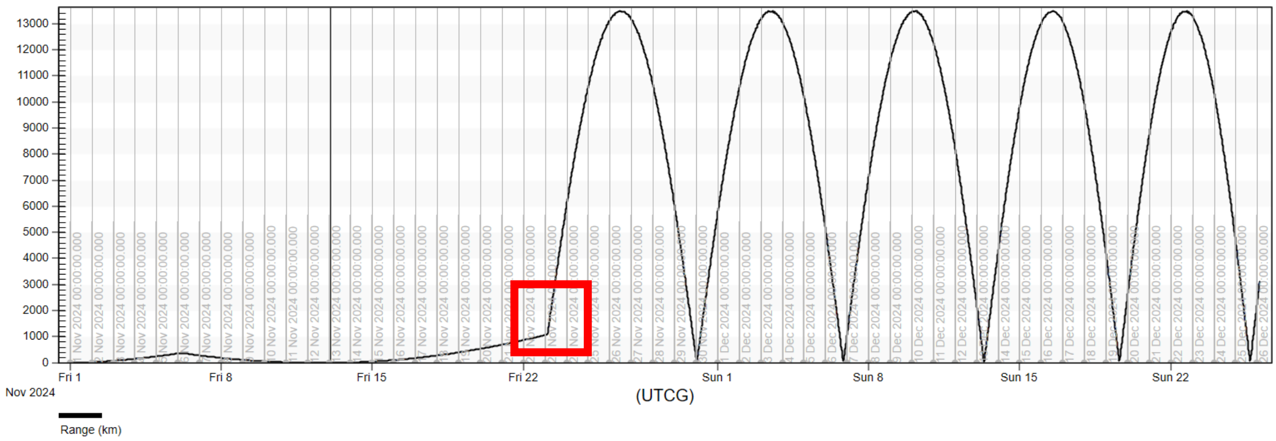


Figure 6.34: SROC's range as a function of the time considering the Hohmann manoeuvre

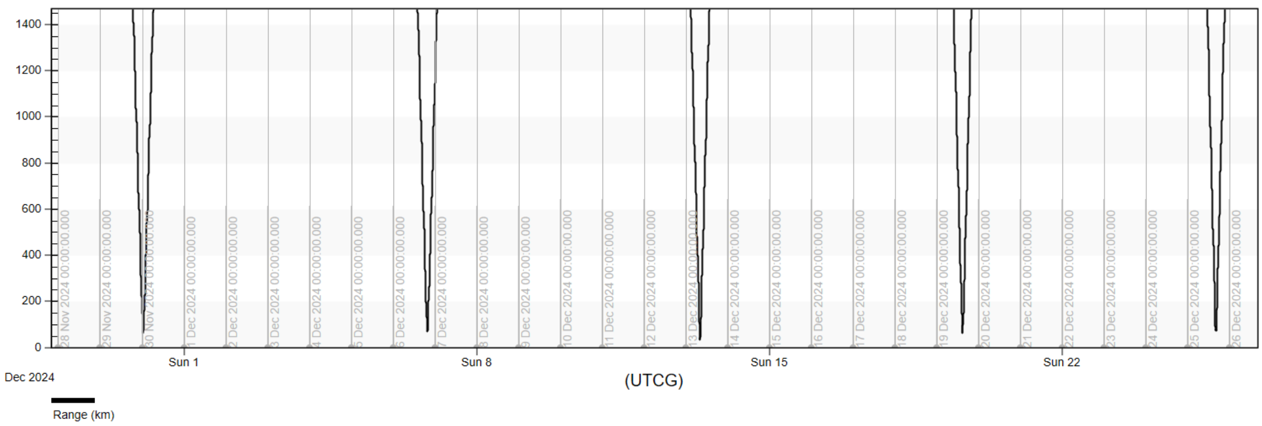


Figure 6.35: SROC's range as a function of the time considering the Hohmann manoeuvre - zoom on the minimum ranges

The costs of the two manoeuvres are the following:

- Radius of apoapsis decrease: 5.446 m/s;
- Circularization: 16.202 m/s

The total deltaV is 21.648 m/s which is more than the total maximum deltaV allocated for the mission. In conclusion, the Hohmann manoeuvre was considered not feasible for its excessive deltaV cost.

6.2.2.2 CAM near the EP

The second option is a CAM performed 1 day before the envisaged EP. The goal of the CAM is to set a minimum 20 km radial separation between SR and SROC. To achieve this goal the following segments were added after the free flight:

- "PropagateToEP": it is a propagation segment which propagates until SROC reaches the EP; this condition is evaluated in STK as the moment when the satellite crosses SR's Radial – CrossTrack plane;
- "Backward 1 Day": the type of this segment is backward sequence. All the segments contained inside of it are propagated backwards: this means that the initial state is actually the last, in time. Inside this sequence there is a propagation segment which stop after one day; by doing so, SROC trajectory is propagated to 1 day before the EP;
- "CAM": this segment is a target sequence, which contains a manoeuvre and a propagation that stops when the EP is reached. The differential corrector is set to achieve a desired Radial value using the manoeuvre thrust as the control parameter.

Since the minimum acceptable radial separation is 20 km, this value was set as the desired result of the CAM target sequence. Figure 6.36 shows that the radial separation is almost 20 km (18.965 km) at the EP.

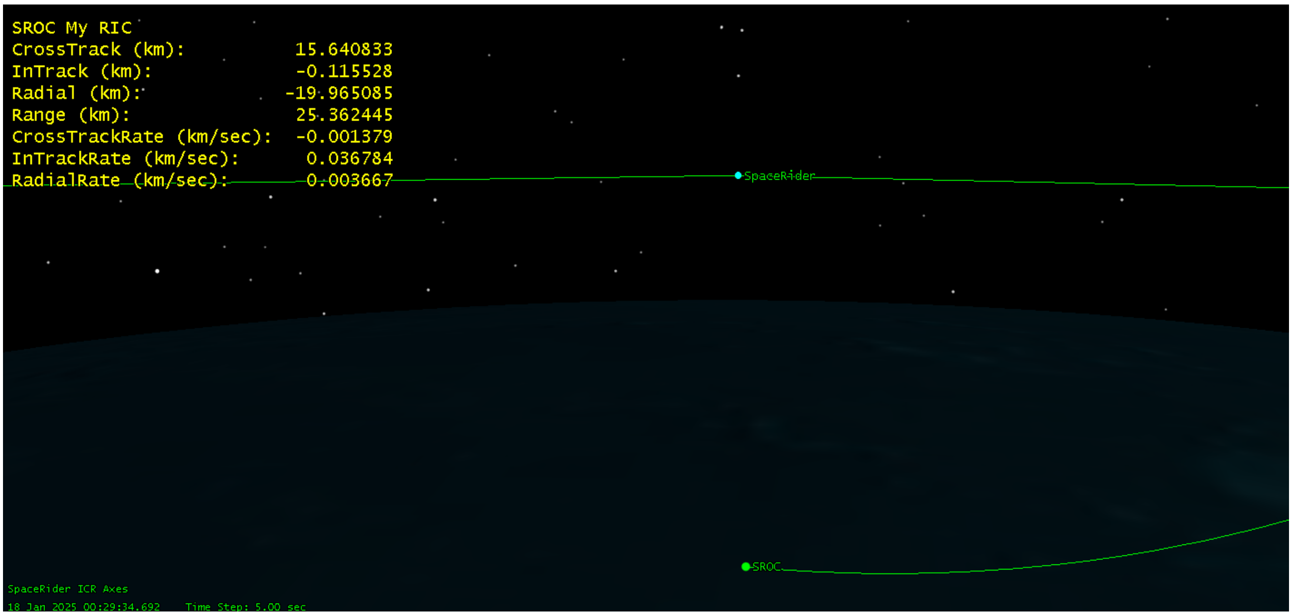


Figure 6.36: Radial separation at the EP

However, it is important to note that the deltaV required to guarantee a 20 km radial separation between SROC and SR may differ from the case analysed, because the disturbances to the eccentricity described in Sub-section 6.2.2.1 may change their heights at the EP. For this reason, several runs where tested to assess the deltaV cost for different radial separations; the results of this analyses are report in Table 6.22. For lower values the deltaV required is compatible with the deltaV available for the mission (3.868 m/s for a 20 km radial separation), but for higher values the deltaV is too high: for example, to achieve a 30 km radial separation a 9.717 m/s deltaV is required. Since it is not possible to assess with total accuracy the state of the satellites at the EP, it would make sense to consider a higher radial separation than 20 km. However, as reported in the table, this causes the deltaV cost to greatly increase to non-acceptable values.

Table 6.22: DeltaV cost for different radial separation values

Radial Separation at the EP [km]	DeltaV cost [m/s]
15	2.265
20	3.868
25	6.305
30	9.717
40	14.656

6.2.2.3 Semi-Major Axis Increase Manoeuvre

The last option evaluated was to perform a semi-major increase manoeuvre to increase the orbital period of SROC, thus delaying the EP. For the sake of this analysis, it is not useful to use the RIC reference system to measure the relative position of SROC, since when the satellite is far away from SR the assumption of small relative position vector magnitude compared to the chief position vector magnitude does not hold up. For this reason, the angle between SROC and SR's position vectors, called ϕ, was used as a reference for the relative position between the two satellites. Figure 6.37 shows the angle between SR and SROC's normal to the orbital plane; since its value is almost close to zero, it can be assumed that the two orbits are co-planar and that the angle ϕ lays on this plane.

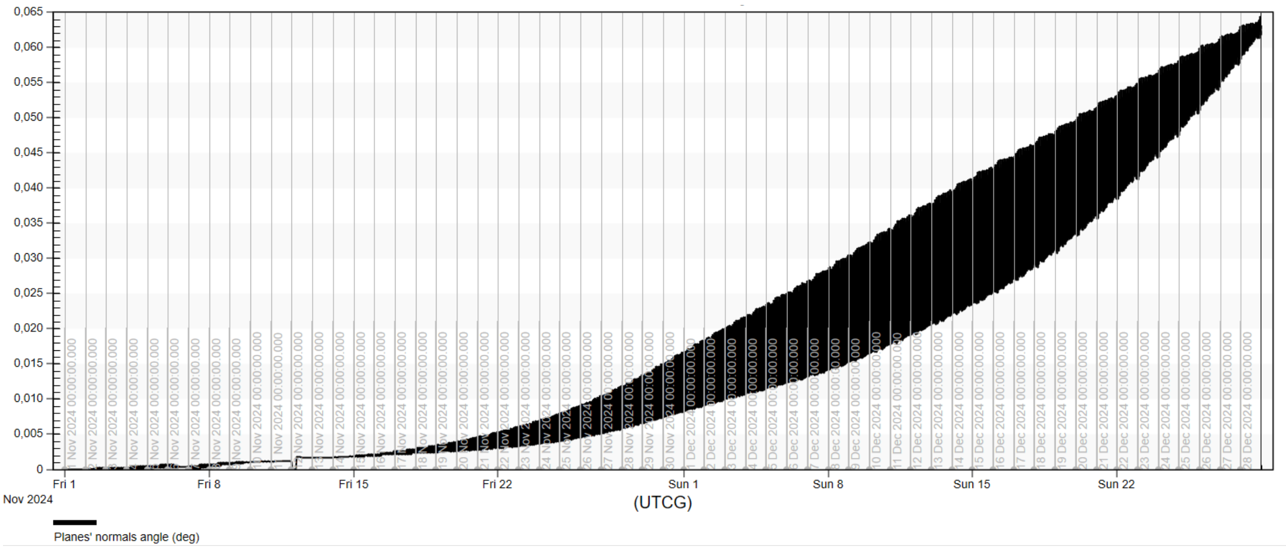


Figure 6.37: angle between the two orbital planes as a function of the time

Figure 6.38 shows the ϕ angle seen in the orbital plane. This angle was created in STK using the analysis workbench tool, which evaluates the angle between 0 and 180 degrees, which means that the angle is not defined by any direction. The following STK segments were added to study this manoeuvre:

- "EMPProp1": it propagates SROC's trajectory until it is met a user-defined values of ϕ, which it was set to 175 degrees for this analysis;
- "SMA Increase": this target sequence includes a propagation and a manoeuvre segment (in this order); the propagation segment stops at the perigee: by performing the successive manoeuvre there, an increase of the apoapsis is obtained. The desired value of the differential corrector is the semi-major axis, and the control parameter is SROC's trust vector along the velocity direction.
- "EMPProp2": this propagation segment propagates until a user-defined epoch. For this analysis, it was set to more than three months after the end of the free flight (13 Feb 2025 00:00);
- "PropToEP": this propagation segment propagates SROC's trajectory until the EP;

These segments are set and run by a Matlab function, which iterates on a vector of user-defined radius of semi-major axis increases and selects the first one valid.; since the vector contains increasing values, the first valid one is also the less deltaV-consuming. To set the desired values for the differential corrector, the analysed semi-major axis increase is added to the semi-major axis at the epoch of the manoeuvre. To be considered valid, every iteration must stay above a threshold ϕ value (for this analysis it was set to 5 degrees) until the end of the "EMPProp2" segment. Finally, the last propagation segment is used to define when evert valid solution reaches the EP. In conclusion, the Matlab code lets the user investigate which apoapsis increase manoeuvre guarantees that SROC will stay above a certain ϕ values until a desired epoch. Another value that the user can define is the ϕ at which the phasing manoeuvre starts; different values could be considered to investigate the best moment to perform a less deltaV-consuming manoeuvre, or to study other variant scenarios which envisage a specific time window to manoeuvre.

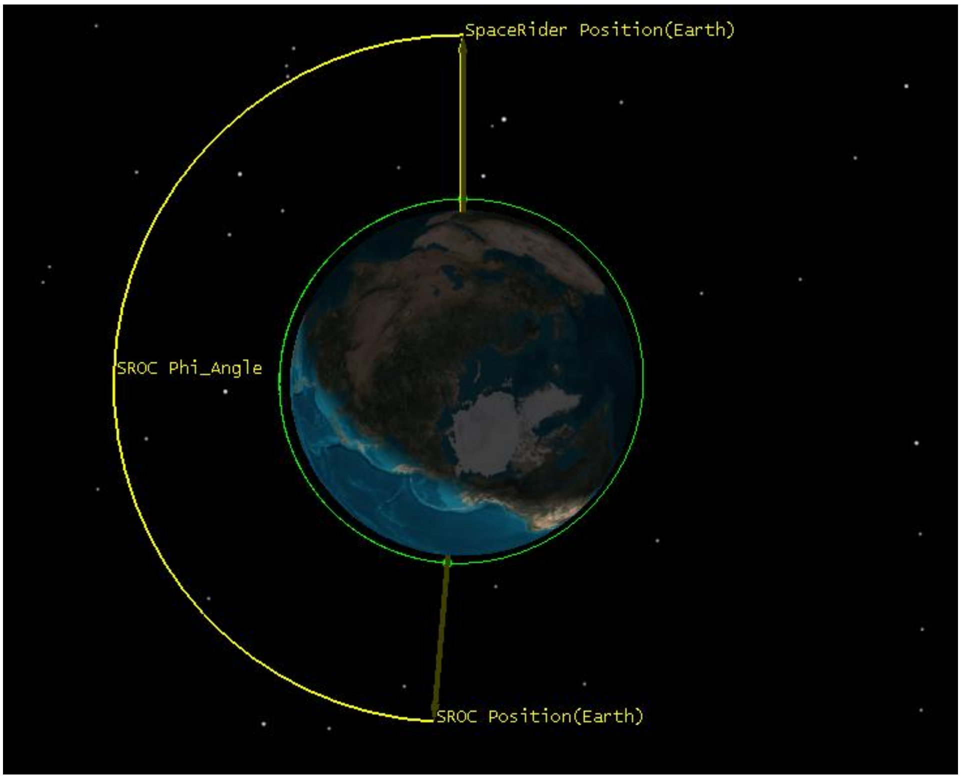


Figure 6.38: ϕ angle

For this analysis, the following vector of apoapsis increase values was used: [5:0.01:8] km. The first solution was found at 5.600 km, which increases the semi-major axis to 6777.77 km for a deltaV cost equal to 3.112 m/s. Figure 6.39 and Figure 6.40 show the evolution of ϕ as a function of the time: in the first picture, it can be seen that the rate at which ϕ increases is noticeably slowed down after the manoeuvre (highlighted in the red box). Instead, the second picture, which does not consider the semi-major axis increase manoeuvre, shows how the ϕ rate increases with the time, until the EP is reached on 18 Jan 2025. Instead, by performing the manoeuvre, the EP does not take place until 13 February 2025 23:44 UTCG. Figure 6.41 shows the range as a function of the time: again, after the semi-major axis increase manoeuvre, the range rate decreases. Figure 6.42 zooms on the last hours before the EP, where the range between SROC and SR would only be 12.700 km.

Since the deltaV required is acceptable and the option does not envisage any additional operations in the proximity of SR, this option was picked for the variant EMP scenario.

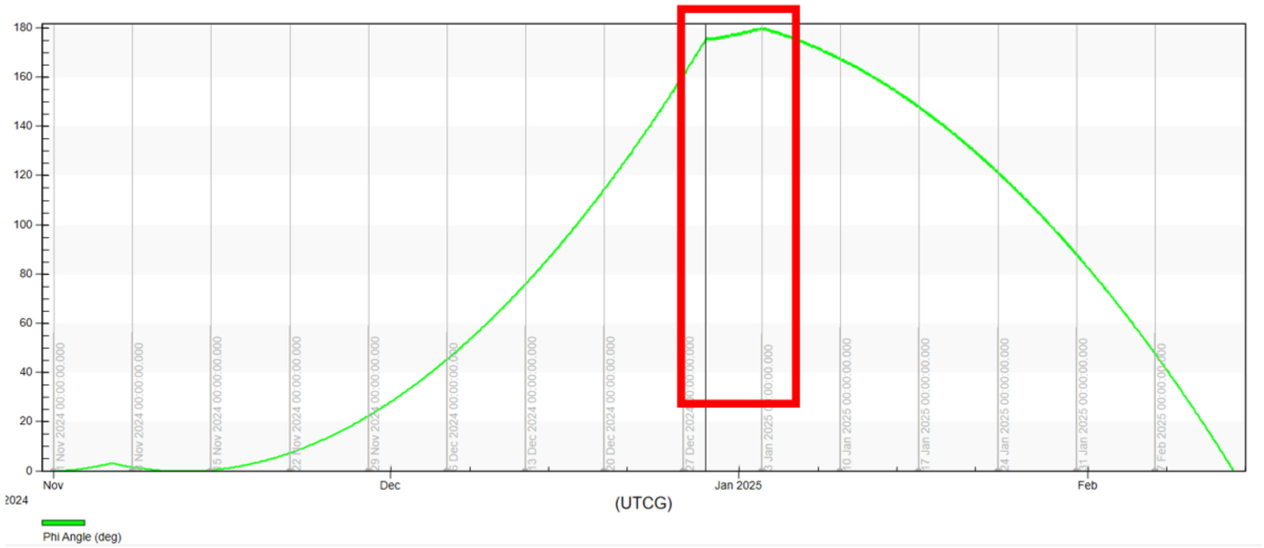


Figure 6.39: ϕ as a function of the time if the manoeuvre is performed

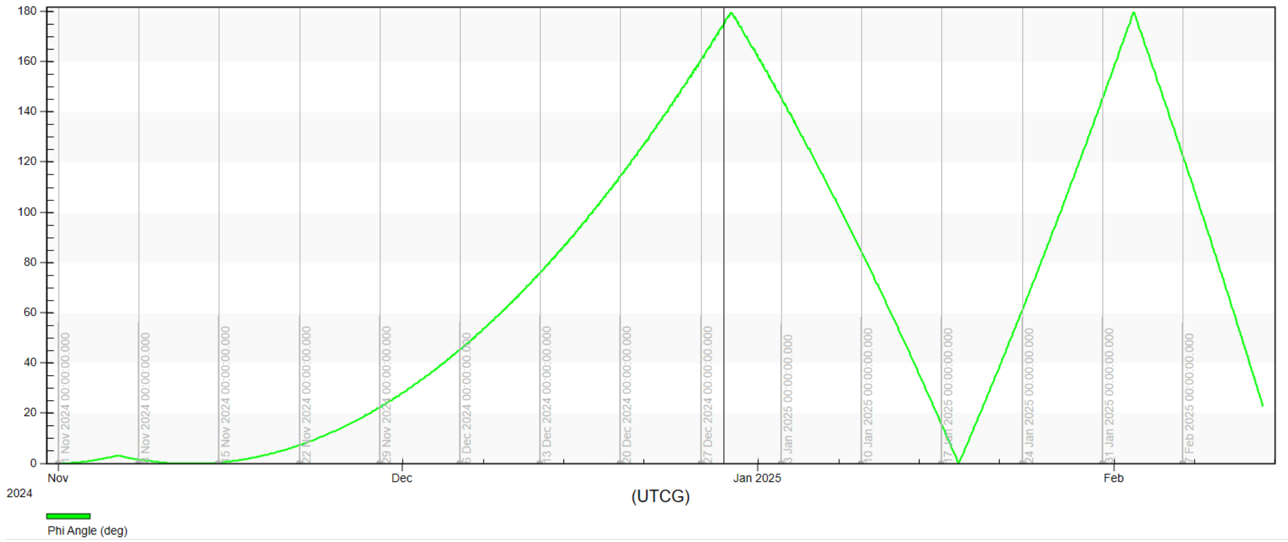


Figure 6.40: ϕ as a function of the time if no manoeuvre is performed

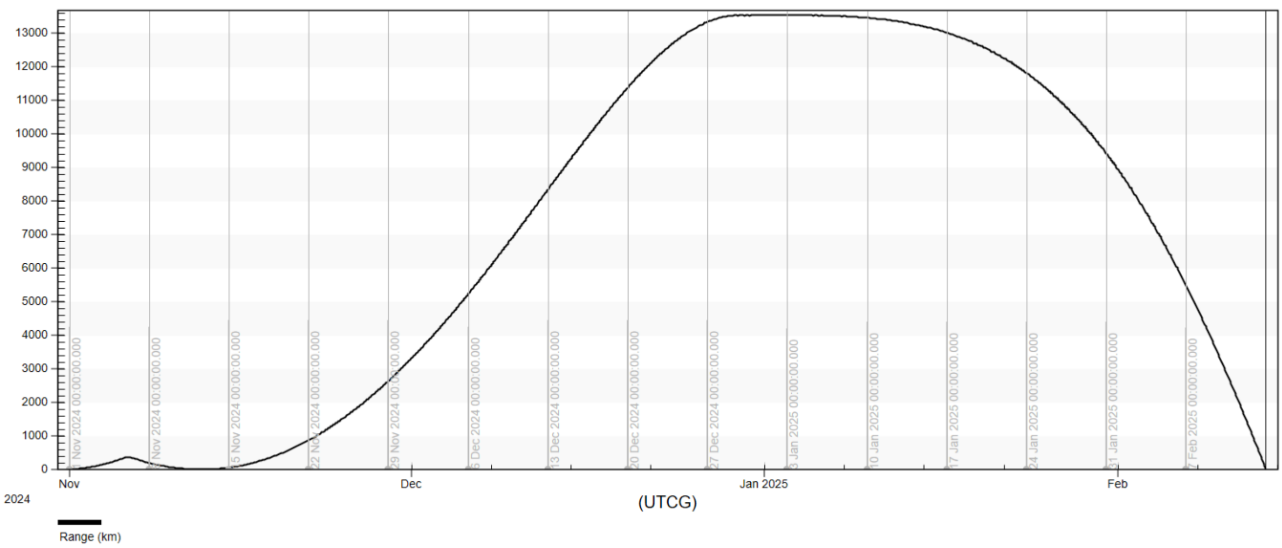


Figure 6.41: Range as a function of the time if the manoeuvre is performed

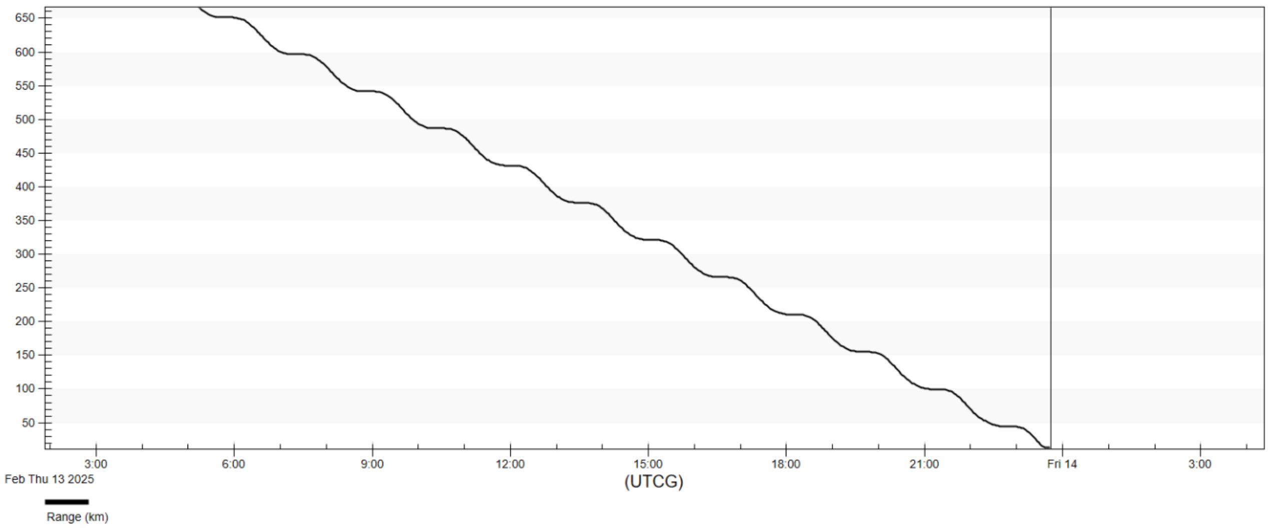


Figure 6.42: Range as a function of the time - zoom on the final hours before the EP

6.3 Variant Scenarios Results Summary

Now that the single variant mission segments have been described, it is possible to study how the occurrence of one or more of them affect the mission. In particular, it is analysed if every variant scenario preserves the following nominal properties:

- Total deltaV less than 20 m/s;
- Total duration less than 30 days;
- IPA safe with respect to SR: this means that after the segment, if no additional manoeuvre is performed, SROC propagates its orbit without getting closer to SR of less than 200 m along the InTrack direction for at least 24 hours;

These three properties have been analysed for every possible MCS; since the total number of all the possibilities considered is very high, all the analysed cases have been divided into four tables to simplify their browsing:

- Table 6.23: refers to all the possible variants for the Observe&Retrieve scenario with a nominal duration for both HP2 and HP3;
- Table 6.24: reports the results of all the possible variants for the Observe scenario considering a nominal duration for the HP2;
- Table 6.25: reports the results of all the possible variants that present a longer HP2 and HP3 (both lasting 13.5 hours instead of 4.5) for the Observe&Retrieve scenario;
- Table 6.26: reports the results of all the possible variants that present a longer HP2 (lasting 13.5 hours) for the Observe scenario;

Each row in these tables refers to a different MCS and it is highlighted in two possible colours: green if all the constraints are satisfied, red if at least one of them is not. It is noted that all the Observe scenarios' variant MCSs consider an additional manoeuvre during the EMP and that the values for the deltaV and the duration include the margins. It is interesting to notice that all the scenarios considered always respect the limitations on the deltaV cost and the duration, thus showing the robustness of the mission to alternative MCSs. Although the ATD-1 and the ATD-3 have been labelled as not safe enough, it is noted that for every possible variant event there is at least one acceptable solution. So, even if the ATD-1 and ATD-3 are not safe enough, their relative variant MCSs can be approached using other solutions such as the ATD-2, the time-down, or the deltaV-down solutions. In conclusion, all the solutions which did not respect the nominal constraints on duration, deltaV cost and safety, have been labelled as off-nominal.

Table 6.23: Overview for the Observe&Retrieve scenarios

Observe and Retrieve – Nominal HP2 & HP3			
Scenario	**DeltaV cost**	**Duration**	**Safe**
Observe&Retrieve LongerHP1 (DD)	9.637	13.373	Yes
Observe&Retrieve LongerHP1 (TD)	9.716	13.037	Yes
Observe&Retrieve LongerHP1 (DD) 2 Insp	10.236	14.076	Yes
Observe&Retrieve LongerHP1 (TD) 2 Insp	11.248	13.740	Yes
Observe&Retrieve LongerComm (DD)	10.620	24.151	Yes
Observe&Retrieve LongerComm (TD)	10.792	23.542	Yes
Observe&Retrieve LongerComm (ATD-1)	11.791	15.542	No
Observe&Retrieve LongerComm (ATD-2)	15.304	20.582	Yes
Observe&Retrieve LongerComm (ATD-3)	13.236	19.426	No
Observe&Retrieve LongerComm (DD) 2 Insp	12.152	24.854	Yes
Observe&Retrieve LongerComm (TD) 2 Insp	11.391	24.245	Yes
Observe&Retrieve LongerComm (ATD-1) 2 Insp	12.390	16.244	No
Observe&Retrieve LongerComm (ATD-2) 2 Insp	15.904	21.284	Yes
Observe&Retrieve LongerComm (ATD-3) 2 Insp	13.836	20.129	No
Observe&Retrieve LongerComm&HP1 (DD)	10.998	24.629	Yes
Observe&Retrieve LongerComm&HP1 (TD)	11.177	23.957	Yes
Observe&Retrieve LongerComm&HP1 (ATD-1)	12.362	16.211	No
Observe&Retrieve LongerComm&HP1 (ATD-2)	15.415	21.185	Yes

Observe&Retrieve LongerComm&HP1 (ATD-3)	12.685	19.295	No
Observe&Retrieve LongerComm&HP1 (DD) 2 Insp	11.597	25.332	Yes
Observe&Retrieve LongerComm&HP1 (TD) 2 Insp	11.777	24.660	Yes
Observe&Retrieve LongerComm&HP1 (ATD-1) 2 Insp	12.962	16.914	No
Observe&Retrieve LongerComm&HP1 (ATD-2) 2 Insp	16.014	21.888	Yes
Observe&Retrieve LongerComm&HP1 (ATD-3) 2 Insp	13.285	19.998	No

Table 6.24: Overview for the Observe scenarios

Observe – Nominal HP2			
Scenario	**DeltaV cost**	**Duration**	**Safety**
Observe LongerHP1 (DD)	11.170	13.051	Yes
Observe LongerHP1 (TD)	10.315	12.715	Yes
Observe LongerHP1 (DD) 2 Insp	11.800	13.753	Yes
Observe LongerHP1 (TD) 2 Insp	11.878	13.417	Yes
Observe LongerComm (DD)	12.152	23.829	Yes
Observe LongerComm (TD)	12.325	22.114	Yes
Observe LongerComm (ATD-1)	13.323	15.219	No
Observe LongerComm (ATD-2)	14.165	20.259	Yes
Observe LongerComm (ATD-3)	14.769	19.104	No
Observe LongerComm (DD) 2 Insp	12.782	24.532	Yes
Observe LongerComm (TD) 2 Insp	12.955	23.923	Yes

Observe LongerComm (ATD-1) 2 Insp	13.395	15.922	No
Observe LongerComm (ATD-2) 2 Insp	17.467	20.962	Yes
Observe LongerComm (ATD-3) 2 Insp	15.399	19.807	No
Observe LongerComm&HP1 (DD)	12.530	24.307	Yes
Observe LongerComm&HP1 (TD)	12.710	23.635	Yes
Observe LongerComm&HP1 (ATD-1)	13.895	15.889	No
Observe LongerComm&HP1 (ATD-2)	16.948	20.863	Yes
Observe LongerComm&HP1 (ATD-3)	14.218	18.973	No
Observe LongerComm&HP1 (DD) 2 Insp	13.160	25.009	Yes
Observe LongerComm&HP1 (TD) 2 Insp	13.340	24.337	Yes
Observe LongerComm&HP1 (ATD-1) 2 Insp	14.525	16.591	No
Observe LongerComm&HP1 (ATD-2) 2 Insp	17.578	21.565	Yes
Observe LongerComm&HP1 (ATD-3) 2 Insp	14.848	19.675	No

Table 6.25: Overview for the Observe&Retrieve scenarios - Longer HP2 and HP3

Observe and Retrieve – Longer HP2 & HP3			
Scenario	**DeltaV cost**	**Duration**	**Safety**
Observe&Retrieve LongerHP1 (DD) LongerHP2&3	10.244	14.160	Yes
Observe&Retrieve LongerHP1 (TD) LongerHP2&3	10.322	13.824	Yes
Observe&Retrieve LongerHP1 (DD) 2 Insp LongerHP2&3	10.843	14.863	Yes
Observe&Retrieve LongerHP1 (TD) 2 Insp LongerHP2&3	10.922	14.527	Yes
Observe&Retrieve LongerComm (DD) LongerHP2&3	11.227	24.939	Yes
Observe&Retrieve LongerComm (TD) LongerHP2&3	11.399	24.330	Yes
Observe&Retrieve LongerComm (ATD-1) LongerHP2&3	12.397	16.329	No
Observe&Retrieve LongerComm (ATD-2) LongerHP2&3	15911.000	21.369	Yes
Observe&Retrieve LongerComm (ATD-3) LongerHP2&3	13.843	20.214	No
Observe&Retrieve LongerComm (DD) 2 Insp LongerHP2&3	11.826	25.641	Yes
Observe&Retrieve LongerComm (TD) 2 Insp LongerHP2&3	11.998	25.032	Yes
Observe&Retrieve LongerComm (ATD-1) 2 Insp LongerHP2&3	12.997	17.032	No
Observe&Retrieve LongerComm (ATD-2) 2 Insp LongerHP2&3	16.510	22.072	Yes
Observe&Retrieve LongerComm (ATD-3) 2 Insp LongerHP2&3	14.443	20.916	No

Observe&Retrieve LongerComm&HP1 (DD) LongerHP2&3	11.605	25.416	Yes
Observe&Retrieve LongerComm&HP1 (TD) LongerHP2&3	11.784	24.744	Yes
Observe&Retrieve LongerComm&HP1 (ATD-1) LongerHP2&3	12.969	16.999	No
Observe&Retrieve LongerComm&HP1 (ATD-2) LongerHP2&3	16.022	21.972	Yes
Observe&Retrieve LongerComm&HP1 (ATD-3) LongerHP2&3	13.292	20.082	No
Observe&Retrieve LongerComm&HP1 (DD) 2 Insp LongerHP2&3	12.204	26.119	Yes
Observe&Retrieve LongerComm&HP1 (TD) 2 Insp LongerHP2&3	12.384	25.447	Yes
Observe&Retrieve LongerComm&HP1 (ATD-1) 2 Insp LongerHP2&3	13.569	17.701	No
Observe&Retrieve LongerComm&HP1 (ATD-2) 2 Insp LongerHP2&3	16.621	22.675	Yes
Observe&Retrieve LongerComm&HP1 (ATD-3) 2 Insp LongerHP2&3	13.892	20.785	No

Table 6.26:Overview for the Observe scenarios - Longer HP2

Observe – Longer HP2			
Scenario	**DeltaV cost**	**Duration**	**Safety**
Observe LongerHP1 (DD) LongerHP2	11.579	13.444	Yes
Observe LongerHP1 (TD) LongerHP2	11.658	13.108	Yes
Observe LongerHP1 (DD) 2 Insp LongerHP2	12.209	14.147	Yes
Observe LongerHP1 (TD) 2 Insp LongerHP2	12.288	13.811	Yes
Observe LongerComm (DD) LongerHP2	12.562	24.223	Yes
Observe LongerComm (TD) LongerHP2	12.734	23.614	Yes
Observe LongerComm (ATD-1) LongerHP2	13.733	15.613	No
Observe LongerComm (ATD-2) LongerHP2	17.246	20.653	Yes
Observe LongerComm (ATD-3) LongerHP2	15.178	19.498	No
Observe LongerComm (DD) 2 Insp LongerHP2	13.192	24.925	Yes
Observe LongerComm (TD) 2 Insp LongerHP2	13.364	24.316	Yes
Observe LongerComm (ATD-1) 2 Insp LongerHP2	14.363	16.316	No
Observe LongerComm (ATD-2) 2 Insp LongerHP2	17.876	21.356	Yes
Observe LongerComm (ATD-3) 2 Insp LongerHP2	15.808	20.200	No

Observe LongerComm&HP1 (DD) LongerHP2	12.940	24.700	Yes
Observe LongerComm&HP1 (TD) LongerHP2	13.119	24.028	Yes
Observe LongerComm&HP1 (ATD-1) LongerHP2	14.304	16.283	No
Observe LongerComm&HP1 (ATD-2) LongerHP2	17.357	21.256	Yes
Observe LongerComm&HP1 (ATD-3) LongerHP2	14.628	19.366	No
Observe LongerComm&HP1 (DD) 2 Insp LongerHP2	13.570	25.403	Yes
Observe LongerComm&HP1 (TD) 2 Insp LongerHP2	13.749	24.731	Yes
Observe LongerComm&HP1 (ATD-1) 2 Insp LongerHP2	14.934	16.985	No
Observe LongerComm&HP1 (ATD-2) 2 Insp LongerHP2	17.987	21.959	Yes
Observe LongerComm&HP1 (ATD-3) 2 Insp LongerHP2	15.258	20.069	No

To simply browse all the different solutions, the duration, and the cost of each of their mission segments, it was created a database on an Excel document. Then a graphic interface was added to let an external user select a desired variant for every segment which presents one or more of them. Finally, another Excel sheet reports the deltaV budget and the duration budget of the selected MCS for both the Observe and the Observe&Retrieve scenarios, while also reporting the nominal ones for comparison purposes. By doing so, any user can easily observe the properties of a desired MCS and can confront it to another variant scenario or to the nominal one. As an example, two couples of tables are here reported:

- Table 6.27 and Table 6.28 respectively show the deltaV and time budget for the variant MCS with the highest deltaV required, that is the Observe scenario with longer Commissioning, HP1 and HP2, using an ATD-2 solution for the IPA rendezvous, two observation cycles, and an additional manoeuvre during the EMP;
- Table 6.29 and Table 6.30 respectively show the deltaV and time budget for the variant MCS with the longest duration, that is the Observe&Retrieve scenario with longer Commissioning, HP1, HP2 and HP3, using a deltaV-down solution for the IPA rendezvous and two observation cycles;

Table 6.27: DeltaV budget for the variant scenario MCS with the highest deltaV cost

OBSERVE Off- Nominal Scenario			
Manoeuvre	**ΔV [m/s]**	**Margin**	**ΔV [m/s]**
HP1	0.887	5%	0.931
Virtual CAM + HP1 bis	1.040	100%	2.080
Virtual CAM + HP1 ter	0.500	100%	1.000
IPA	2.543	5%	2.670
HP2Ins	4.306	5%	4.521
ZRV2	0.661	5%	0.694
HP2	0.486	5%	0.510
OPA - Cycle 1	0.266	100%	0.532
WSE Insertion - Cycle 1	0.192	100%	0.385
OPA - Cycle 2	0.095	100%	0.189
WSE Insertion - Cycle 2	0.221	100%	0.441
D CAM	0.068	100%	0.136
SR CAM	0.600	5%	0.630
EMP Manoeuvre	3.112	5%	3.267
ΔV TOT [m/s]	**14.976**	**ΔV TOT with margins [m/s]**	**17.987**

Table 6.28: Time budget for the variant scenario MCS with the highest deltaV cost

OBSERVE Off - Nominal Scenario			
Manoeuvre	**Duration [day]**	**Margin**	**Duration [day]**
Commissioning	10.000	5%	10.500
HP1	0.563	5%	0.591
IPA	8.200	5%	8.610
HP2Ins	0.083	5%	0.088
HP2	0.563	5%	0.591
OPA - Cycle 1	0.167	5%	0.175
Observation + FreeFlight - Cycle 1	0.669	5%	0.703
OPA - Cycle 2	0.171	5%	0.179
Observation + FreeFlight - Cycle 2	0.669	5%	0.703
Duration TOT [day]	**21.084**	**Duration TOT with margins [day]**	**22.138**

Table 6.29: DeltaV budget for the variant scenario MCS with the highest duration

OBSERVE & RETRIEVE Variant Scenario			
Manoeuvre	**ΔV [m/s]**	**Margin**	**ΔV [m/s]**
HP1	0.887	5%	0.931
Virtual CAM + HP1 bis	1.040	100%	2.080
Virtual CAM + HP1 ter	0.500	100%	1.000
IPA	0.906	5%	0.951
HP2Ins	2.117	5%	2.223
ZRV2	0.280	5%	0.294
HP2	0.486	5%	0.510
OPA - Cycle 1	0.266	100%	0.532
WSE Insertion - Cycle 1	0.192	100%	0.385
OPA - Cycle 2	0.095	100%	0.189
WSE Insertion - Cycle 2	0.221	100%	0.441
HP3Ins	0.208	5%	0.218
ZRV3	0.421	5%	0.442
HP3	0.282	5%	0.296
Docking	0.900	5%	0.945
D CAM	0.068	100%	0.136
SR CAM	0.600	5%	0.630
ΔV TOT [m/s]	**9.468**	**ΔV TOT with margins [m/s]**	**12.204**

Table 6.30: Time budget for the variant scenario MCS with the highest duration

OBSERVE & RETRIEVE Off - Nominal Scenario			
Manoeuvre	**Duration [day]**	**Margin**	**Duration [day]**
Commissioning	10.000	5%	10.500
HP1	0.563	5%	0.591
IPA	11.480	5%	12.054
HP2Ins	0.083	5%	0.088
HP2	0.563	5%	0.591
OPA - Cycle 1	0.167	5%	0.175
Observation + FreeFlight - Cycle 1	0.669	5%	0.703
OPA - Cycle 2	0.171	5%	0.179
Observation + FreeFlight - Cycle 2	0.669	5%	0.703
HP3Ins	0.113	5%	0.118
HP3	0.563	5%	0.591
Final Approach	0.007	5%	0.007
Duration TOT [day]	**25.046**	**Duration TOT with margins [day]**	**26.298**

7 DRAMA Analysis

The ESA software DRAMA (Debris Risk Assessment and Mitigation Analysis) was used to perform several analyses in order to be compliant with the Space Debris Mitigation [7] and, more in general, to the Statement of Work for the Phases B2/C/D of SROC [18]. In particular, the following topics were analysed [21]:

- Computation of the geometric cross-section (with the CROC tool);
- Collision avoidance manoeuvre frequencies to avoid debris and/or meteoroids along the trajectory (with the ARES tool);
- Natural decay of the satellite after the proximity operations phase (with the OSCAR tool);
- Re-entry survival prediction for SROC and its main components and the associated risk on ground for any object surviving the re-entry phase (with the SARA tool);

The last two points apply only to the Observe scenario.

7.1 CROC tool

The analysis with CROC was the first one performed to evaluate the average cross-section of the satellite. SROC is modelled as a simple box, with width = 0.226 m, height = 0.34 m and depth = 0.226 m as reported in SROC System Design Definition File [20]. The attitude of the satellite is set to randomly tumbling, as it happens in many phases during the mission; even during the ones with a controlled attitude, such as the observation one, the relative orientation of the body axes varies with respect to the RIC axes.

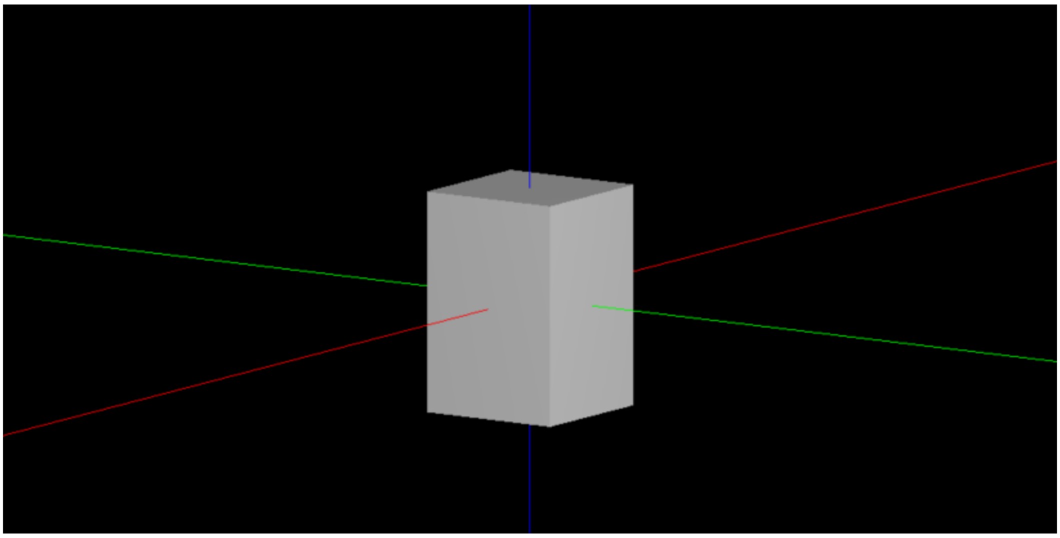


Figure 7.1: SROC body in CROC

The output of the results is an average cross-section of 0.1014 m^2.

7.2 Collision Avoidance Manoeuvre Evaluation

The space debris density in the space environment can be evaluated using the ARES tool. This software combines the orbit information with the accuracy of space surveillance systems to evaluate the statistical number of collision avoidance manoeuvres as a function of the acceptable risk levels [22]. In particular, the decision to perform a CAM is related to the risk associated with a near-miss event, which in turn depends on the geometry of the of the encounter, the collision cross-section, and the uncertainties in the state vector of both objects. Since in LEO there are many poorly tracked objects with a location uncertainty in the order of kilometres (assumed to be a Gaussian distribution) and a part of them cannot even be tracked, the risk threshold is best defined in terms of risk reduction with respect to the unavoidable background population. This value is defined as a function of the ACPL (Accepted Collision Probability Level), which most space missions set at 10^{-4} [24].

To start the analysis, the start epoch was set to 2024/11/01, which is when the satellite is deployed in the STK simulation. Since ARES considers SROC with a spherical shape, the spacecraft radium was set equal to the distance from the centre of SROC to its furthest point, which is half of the diagonal (0.233 m). The orbit used for this analysis is the same used in STK and it is defined in Table 7.1.

Table 7.1: Orbit definition in DRAMA

Orbital Parameter	Value
Semi-major axis	6771 km
Eccentricity	0
Inclination	6.2 deg
Right Ascension of the Ascending node (RAAN)	0 deg
Angle of Perigee	0 deg
True Anomaly	0 deg

7.2.1 MASTER Analysis

Before analysing the ACPL it is useful to evaluate the space debris flux in the target orbit in order to understand the space debris situation that the satellite will encounter. Useful information can be extrapolated from this analysis, such as the direction from which it can be expected to receive most of the conjunctions (if there are any). Moreover, this additional analysis, although it is not strictly required to assess the CAM required, is still recommended in ARES guidelines document [22]. To perform this study an additional ESA software is used: MASTER (Meteoroid and Space Debris Terrestrial Environment Reference) [25].

The analysis was conducted on the same orbit described in Table 7.1 and for debris with the same size as the ones analysed in ARES, which are from 1 cm to 100 m of diameter. Technically, MASTER can provide fluxes of impact object size down to 1 micrometre, but ARES only considers only an impact object size down to 1 cm, since it is the best resolution achievable by state-of-the-art ground surveillance systems. The most relevant results are the 2D flux distribution of the debris according to the azimuth and elevation angles. From the first one (Figure 7.2) it is possible to see that the flux is distributed between all the possible azimuth values, which means that there is the possibility of a reverse conjunction (that is debris approaching SROC from the back). The probability of this type of collision, however, is lower with respect to one coming from the front, especially if the range from -80 deg to -40 deg is considered.

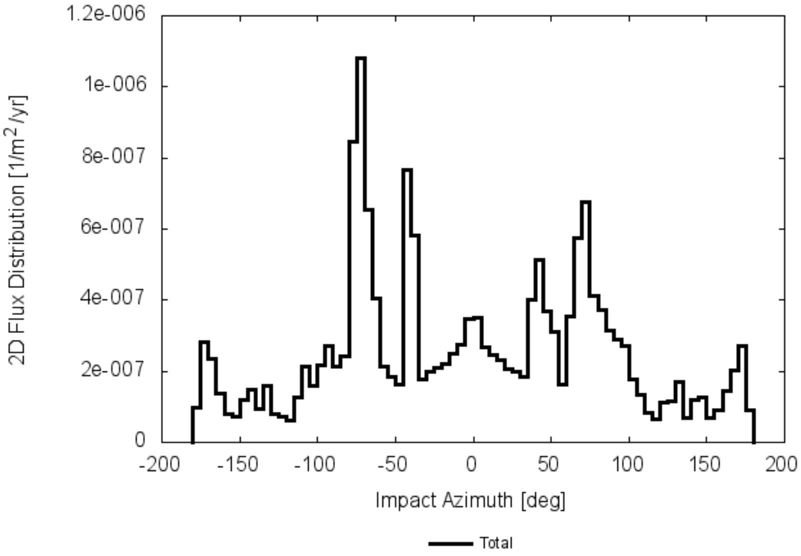


Figure 7.2:Flux Distribution according to the Impact Azimuth

Figure 7.3 represents the flux distribution according to the impact elevation: it shows that most of the debris are encountered along null or very low elevation angles, with a flux distribution almost constantly decreasing with the increasing of the impact elevation angles.

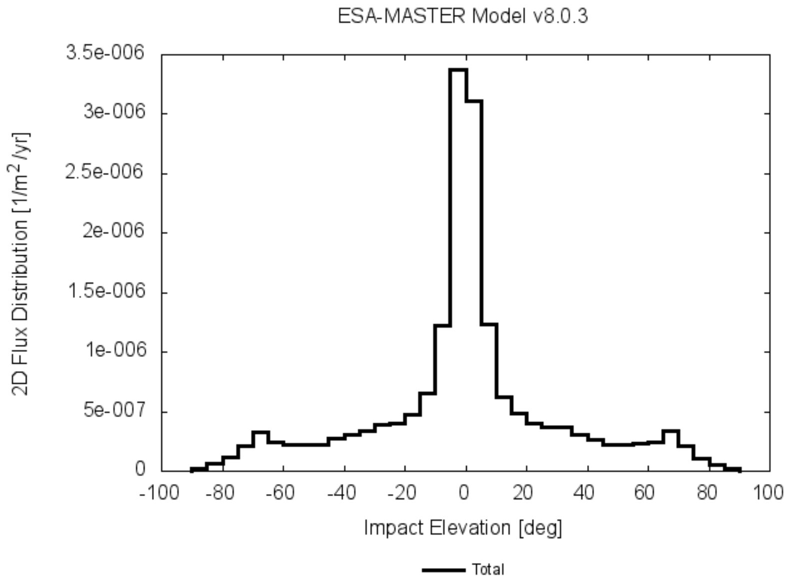


Figure 7.3: Flux Distribution according to the Impact Elevation

Figure 7.4 shows the Flux Distribution versus the impact velocity expressed in km/s. The flux distribution is homogeneously distributed across the impact velocity range, which varies from 1 km/s to 28 km/s but presents the highest values for lower impact velocities.

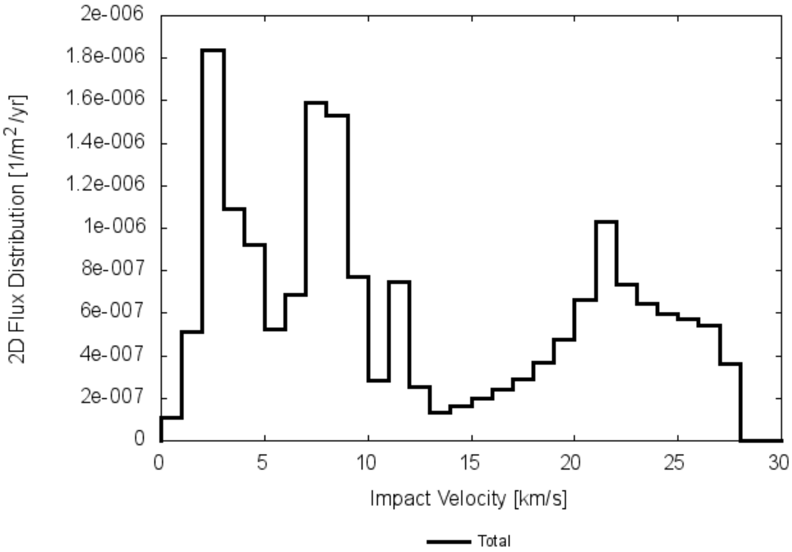


Figure 7.4: Flux Distribution according to the Impact Velocity

Figure 7.5 shows the 2D flux distribution with different object diameters; as it may be expected, for the minimum diameters there are the maximum values. As mentioned before, the flux distribution is evaluated considering objects with a diameter set between a minimum and a maximum value. The values selected for both ARES and Master are the default ones, which in the DRAMA user manual are said to be sufficient for an adequate risk analysis.

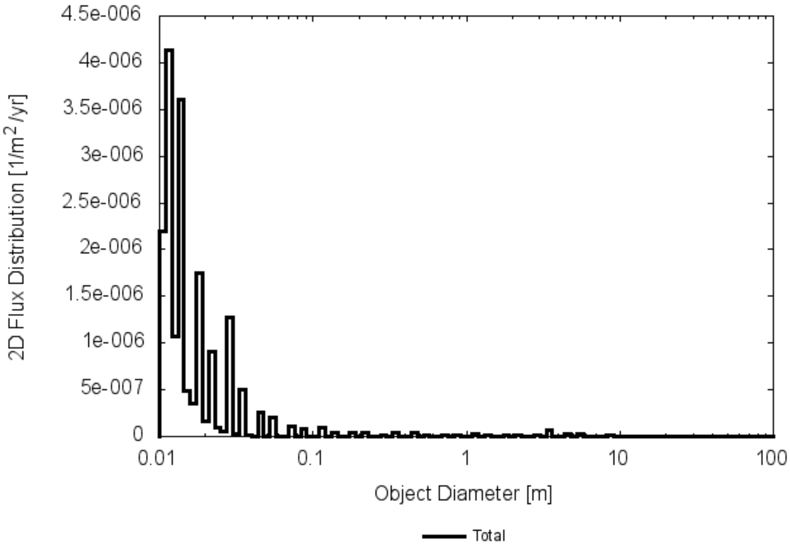


Figure 7.5: Flux Distribution according to the Object Diameter

7.2.2 ARES set-up

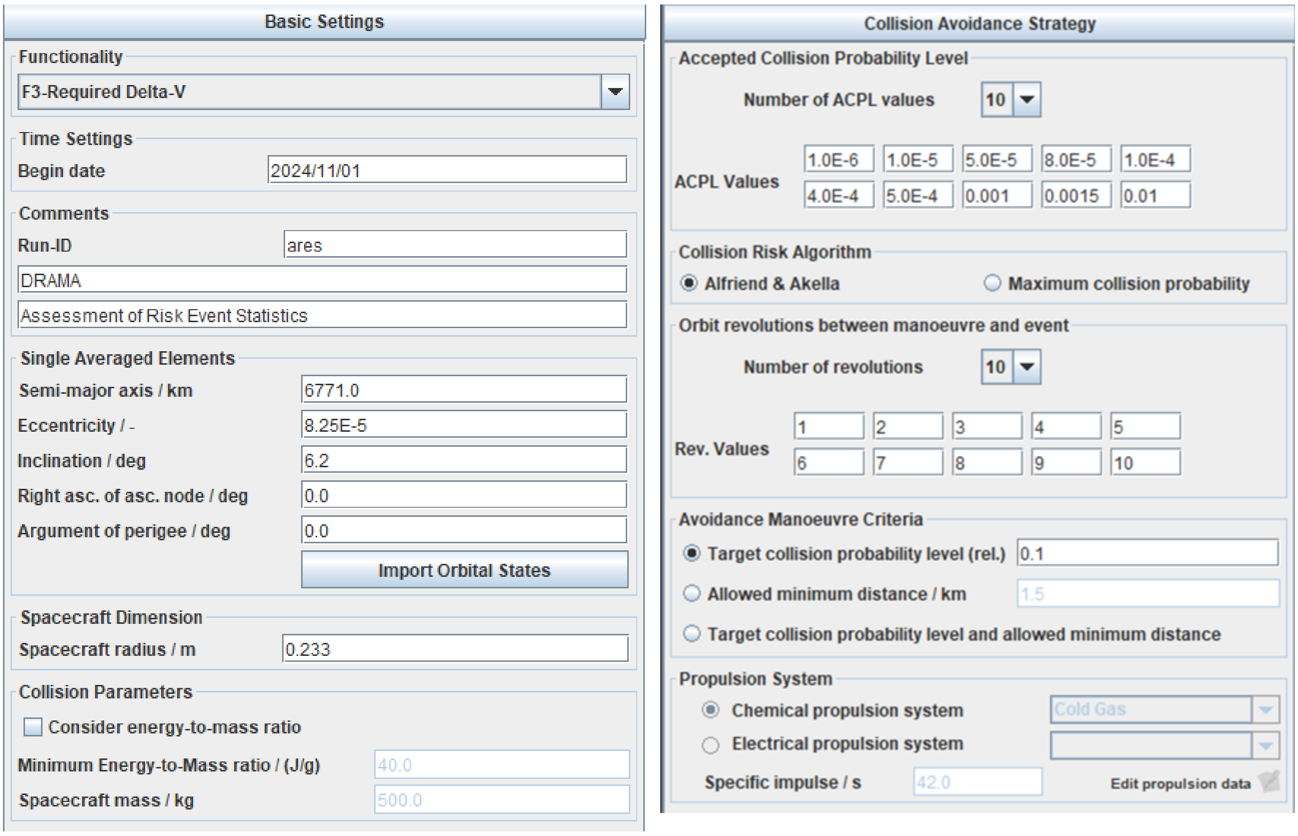
After setting the spacecraft orbit and radius, and the range size for the debris, the radar equation [21] is configured to estimate the cataloguing performance:

$$D_{min}(h) = D_{ref} \cdot \left(\frac{h}{h_{ref}}\right)^{exp}$$

Where D_{min} is the minimum detectable diameter, D_{ref} is a reference diameter, h is the orbit altitude and h_{ref} is a reference altitude. To account for different determination processes for LEO and MEO/GEO objects, two branches for this equation are provided; ARES evaluates both branches, then chooses the one which guarantees the minimum detectable diameter. The inputs for both branches are the default ones.

The reason why this equation is used is that ARES's scope is to simulate the CAM in the entirety of the actions which would take place in the real operative scenario, to the extent that every supporting space surveillance network has a certain minimum detectable radius. For this reason, even the time between the prediction of a collision and the actual occurrence of it is considered; this parameter was set to 1 day. There is also the possibility to set a global scaling factor to correct the population covariances; to set a neutral scaling factor, this value was set to 1.

Finally, the collision avoidance strategy was defined. Ten different values for the ACPL were considered as well as ten values for the orbit revolutions between manoeuvre and event. ARES always considers a short term (half an orbit) along-track manoeuvre, while these values are used to consider one-day-increasing durations for the orbit revolutions. The target collision probability level is a scaling factor for the ACPL which defines the target collision probability for a collision avoidance manoeuvre. It is set to 0.1, which means that, if the ACPL is 10^{-5}, then ARES triggers a manoeuvre for each event with a collision probability level above $0.1\text{x}10^{-5}$. The propulsion system category (cold gas) and specific impulse (42 s) were defined in accordance with SROC system design document [20]. Figure 7.6 shows the complete setup for this analysis.



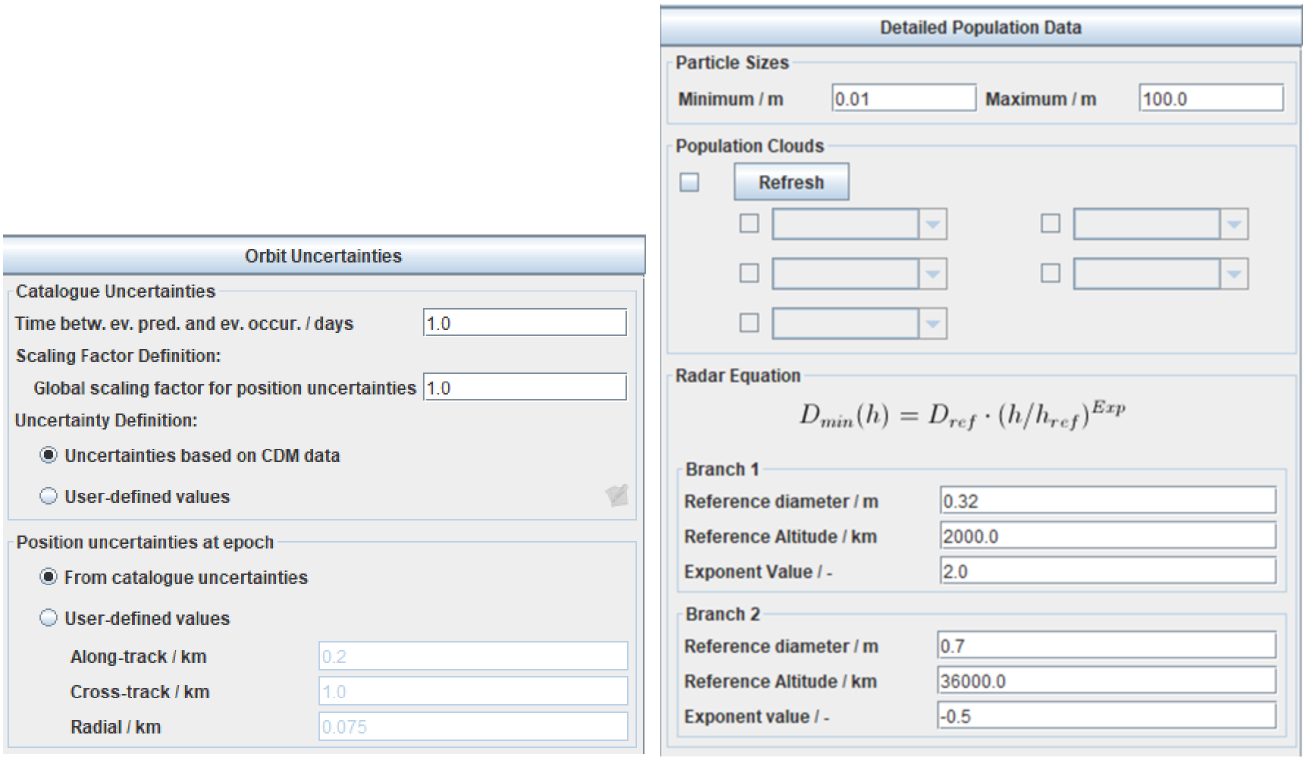


Figure 7.6: ARES settings

7.2.3 Ares results

Figure 7.7 shows the risk reduction and the residual risk as a function of the ACPL. For ACPL values around 5x10^{-5} the risk is reduced by 50%, while with 10^{-6} the risk is reduced by nearly 90%. Typically, what is aimed at is a risk reduction of around 90% and a ACPL of at least 10^{-4}: in this case, these recommendations are meth with an ACPL of 10^{-6}. The residual risk is calculated considering all the risks with a collision probability inferior to the defined threshold; in turn, the risk reduction is the accumulated collision probability of the events above the decision threshold. Besides these two parameters, there is a third one called remaining risk, which is the risk due to non-trackable objects, whose contribution is usually not shown since nothing can be done about them. Figure 7.8 illustrates that the mean number of avoidance manoeuvres increases almost logarithmically with the decrease of the ACPL, with a maximum value of 0.3 manoeuvres to guarantee an ACPL of 10^{-6}. This number refers to a mean number evaluated for a year of propagation.

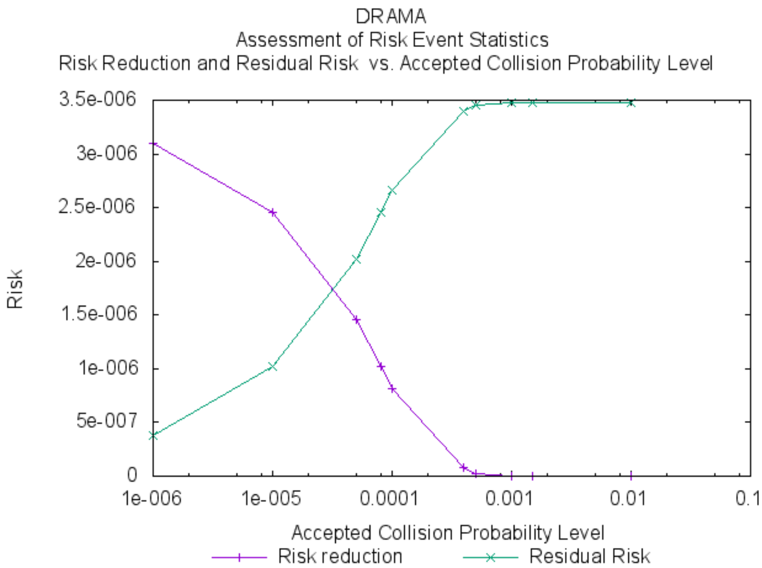


Figure 7.7: Risk reduction and residual risk as function of the ACPL

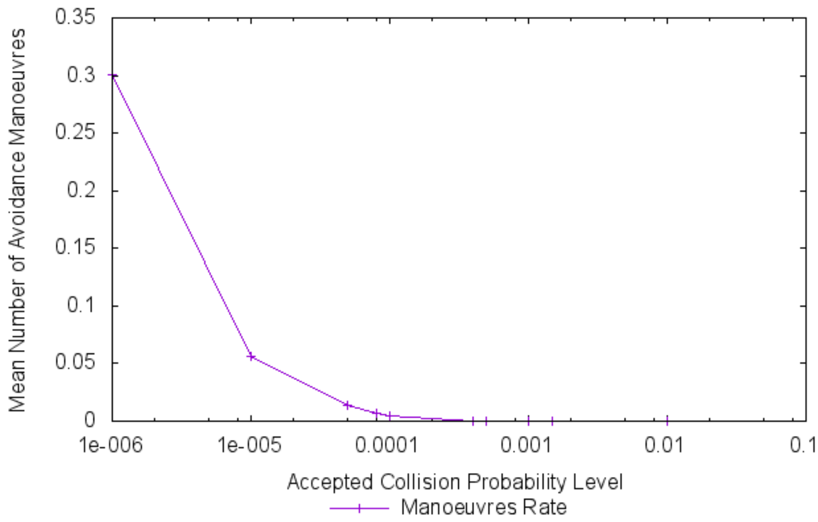


Figure 7.8: Manoeuvres frequency as function of the ACPL

Before defining a final value for the ACPL and showing other useful results, a robustness analysis is recommended [22]. The following cases were considered:

- -30% of the time between the event detection and the event occurrence (0.7 days); for a shorter interval, DRAMA does not guarantee the capability to avoid the manoeuvre;
- Double the time between the event detection and the event occurrence (2 days);
- Double the scale factor on covariance (2);

For the first variation, the results (Figure 7.9 and Figure 7.10) show that there is an increase in the risk reduction (around 91% for an ACPL of 10^{-6}), but also a decrease in the mean number of avoidance manoeuvres (almost 0.26 per year for an ACPL of 10^{-6}). In the second case (Figure 7.11 and Figure 7.12), for an ACPL of 10^{-6}, the risk reduction is only 82% and the required number of avoidance manoeuvres is 0.34 (a bit more than the nominal case). The third and final case, as expected, shows worse results (Figure 7.13 and Figure 7.14) than the nominal analysis: for an ACPL of 10^{-6}, the risk reduction is approximately 82% and the required number of manoeuvres is 0.4. Moreover, the risk reduction decreases to 50% for much lower values (around 10^{-5}) with respect to the nominal analysis. In conclusion, although noticeable changes happen between the different scenarios, the robustness of the analysis and the solution is consistent with the results provided in the verification guidelines of ARES [22].

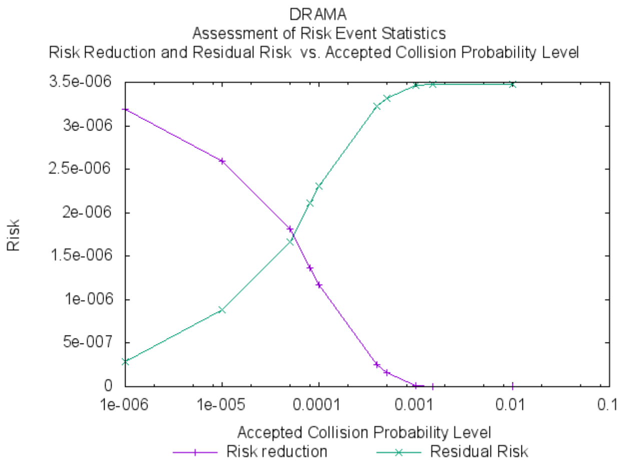


Figure 7.9: First case - Risk reduction and residual risk as function of the ACPL

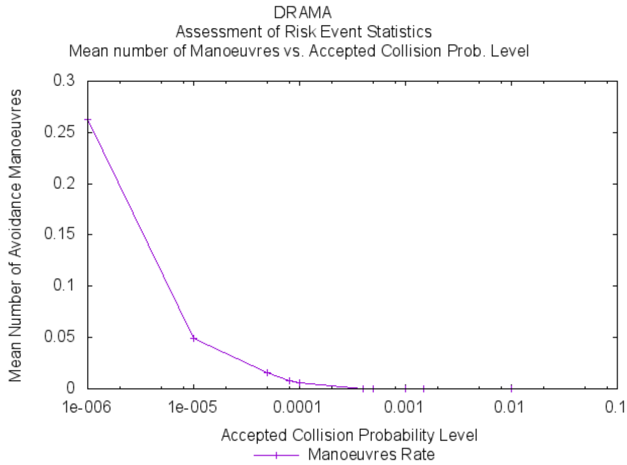


Figure 7.10: First case - Manoeuvres frequency as function of the ACPL

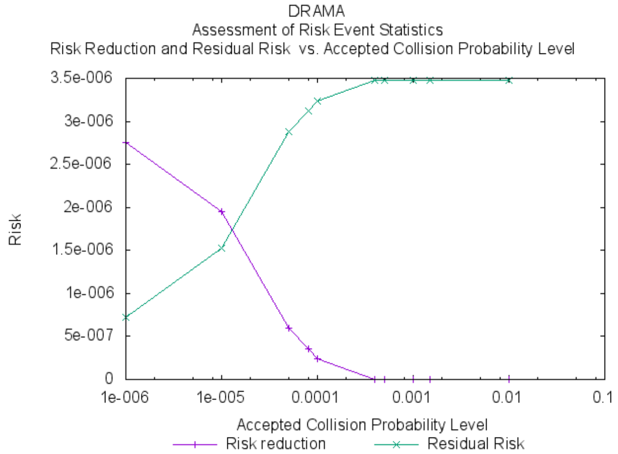


Figure 7.11: Second case - Risk reduction and residual risk as function of the ACPL

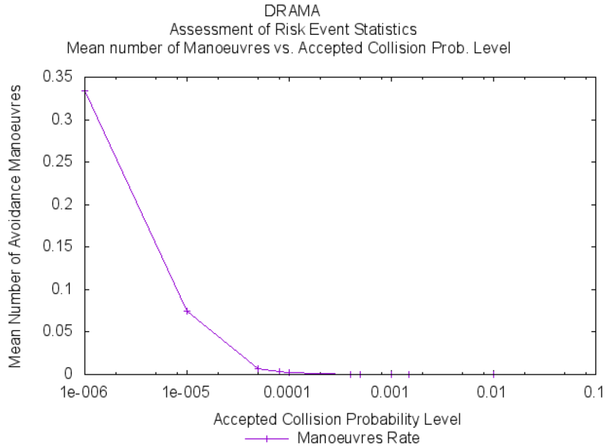


Figure 7.12: Second case - Manoeuvres frequency as function of the ACPL

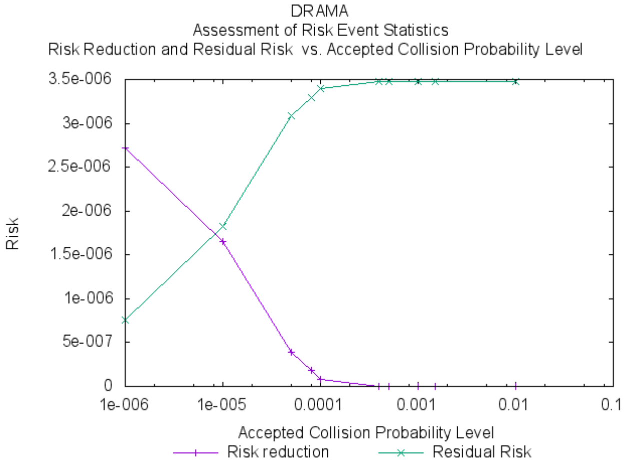


Figure 7.13: Third case - Risk reduction and residual risk as function of the ACPL

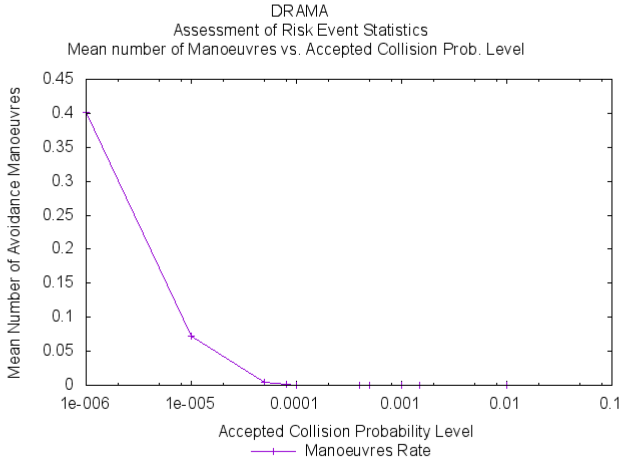


Figure 7.14: Third case - Manoeuvres frequency as function of the ACPL

To achieve a much higher risk reduction, the ACPL could be lowered even more. As shown in Figure 7.15 and Figure 7.16, if an ACPL equal to 10^{-7} is considered, it is obtained a basically null residual risk, but at the cost of a mean number of avoidance manoeuvre equal to 2.25 (7.5 times more than the nominal case). For this reason, this option was discarded.

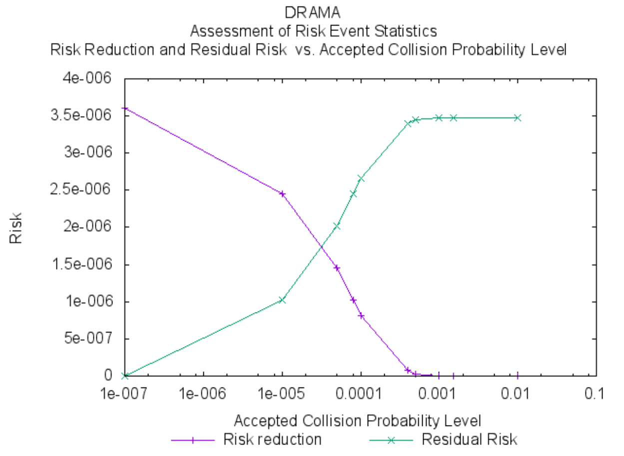


Figure 7.15: Minimum ACPL=10^{-7} - Risk reduction and residual risk as function of the ACPL

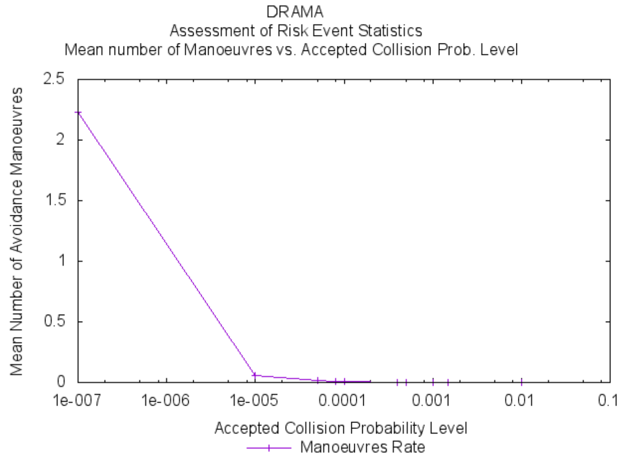


Figure 7.16: Minimum ACPL=10^{-7} - Manoeuvres frequency as function of the ACPL

Figure 7.17 shows for the nominal case the required deltaV for one year across the orbit. It is no surprise that for lower ACPL the deltaV is lower since an inferior manoeuvre frequency is required. Two different types of strategies are implemented by ARES [23]:

- Short–term strategy: it is the one evaluated for a number of revolutions = 0. It targets additional radial separation between the two objects at the TCA (Time of Closest Approach).
- Long-term strategy: different manoeuvres are evaluated for different multiples of one revolution. They target a different phasing and thus a larger along- and/or cross-track separation at the TCA. The required deltaV to perform these manoeuvres decreases with the increase of the number of revolutions before the TCA.

For the selected ACPL level (the purple line) the maximum deltaV (0.123 m/s) is required for a long-term strategy starting one revolution before the TCA. The short-term strategy requires a smaller deltaV ($0.680 \cdot 10^{-1}$ m/s).

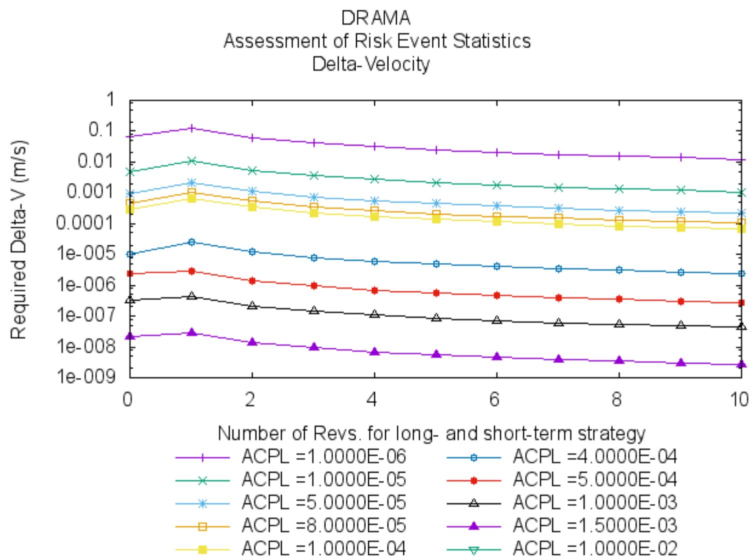


Figure 7.17: Required deltaV as function of the number of revolutions for long and short term strategy

Figure 7.18 shows interesting information about the risk category analysed with this study. Besides the already defined risk reduction and residual risk, it is also presented the remaining risk, which is, as explained before, not avoidable. This fixed value increases of $0.1767 \cdot 10^{-5}$ the residual risk; for the ACPL considered the remaining risk is equal to $0.2166 \cdot 10^{-5}$.

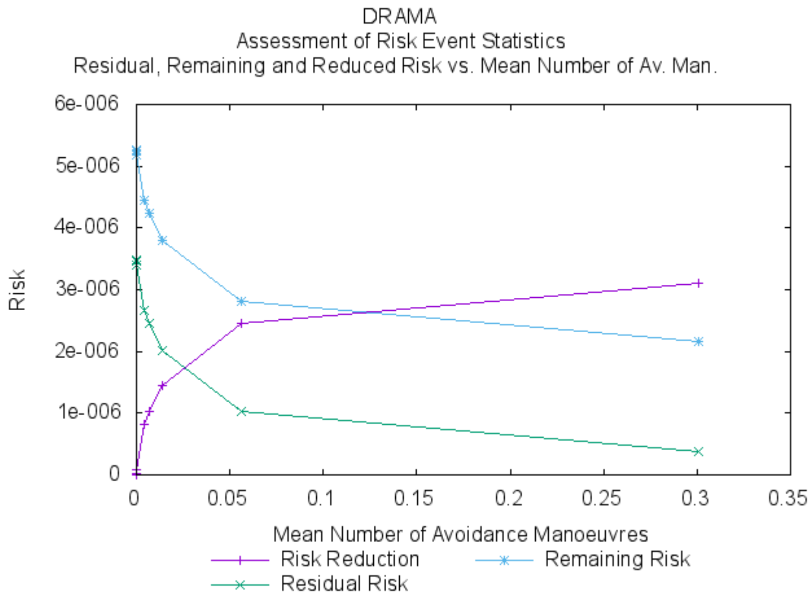


Figure 7.18: Risk reduction, residual risk and remaining risk in function of the mean number of avoidance manoeuvres

The final step of this analysis was selecting a deltaV for the Debris CAM (called D CAM in the deltaV budget). Just using one of the deltaVs plotted in Figure 7.17 would not make sense, since they refer to an average of 0.3 CAM per year. These values would be useful to allocate the deltaV budget for missions staying in the same orbit for several years, which is not the case for SROC. For this reason, the selected deltaV was divided by 0.3 to assess the deltaV required to perform a single manoeuvre. Considering a CAM performed 6 revolutions before the EP, it is obtained a deltaV equal to 0.0683 m/s. Table 7.2 shows the deltaV required to perform one CAM and how many hours before the EP the manoeuvre must be performed. The highlighted solution was considered a good compromise between the preparation time required and the deltaV.

Table 7.2: CAM deltaV summary

Rev before EP [#]	Time before EP [hr]	DeltaV for 0.3 CAM a year [m/s]	DeltaV for one CAM [m/s]
0.000	-	0.068	0.227
1.000	1.543	0.123	0.411
2.000	3.086	0.062	0.205
3.000	4.628	0.041	0.137
4.000	6.171	0.031	0.103
5.000	7.714	0.025	0.082
6.000	9.257	0.020	0.068
7.000	10.800	0.018	0.059
8.000	12.342	0.015	0.051
9.000	13.885	0.014	0.046
10.000	15.428	0.012	0.041

7.3 OSCAR tool

OSCAR is used to the de-orbit of a satellite after its nominal end-of-life, to verify the compliance of the SROC mission with the Space Debris Mitigations for Agency projects [7]. The spacecraft parameters were defined as follows:

- Cross-section area [m^2]: 0.1014, which is the value computed by CROC.
- Mass [kg]: 21 kg
- Drag coefficient: 2.2
- Reflectivity coefficient: 1.3

The initial orbit is the same described in Table 7.1, but the begin date (YYYY-MM-DD) is 2021-11-15 at 00:00, which is approximately when the proximity operation phase should end. OSCAR requires to define the disposal option, which in this case is none. The orbit prediction is dependent on the prediction for the solar and geomagnetic activities used, since, as it has already been explained in Sub-section 3.2.1, they were used to model the atmospheric drag. To account for these differences, three analyses were performed, which all confirm that SROC deorbits less than 13 months, thus respecting the Space Debris Mitigations for Agency projects [7].

- **ECSS sample solar cycle**: final date (YYYY-MM-DD) 2025-04-22
  This analysis is recommended by the ECSS standard [26] and uses the solar and geomagnetic parameters of the solar cycle 23 for the complete propagation time span.

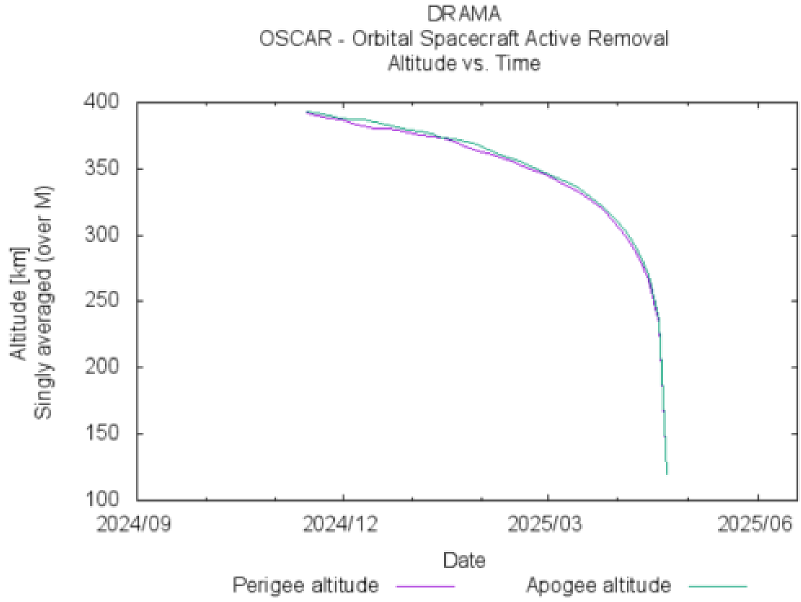


Figure 7.19: ECSS Sample - SROC altitude vs time

- **Latest predictions**: final date (YYYY-MM-DD) 2025-06-05
  This model uses the available up-to-date prediction on solar and geomagnetic activity provided by ESA. At the time of the analysis, the data were updated on 2023-05-27.

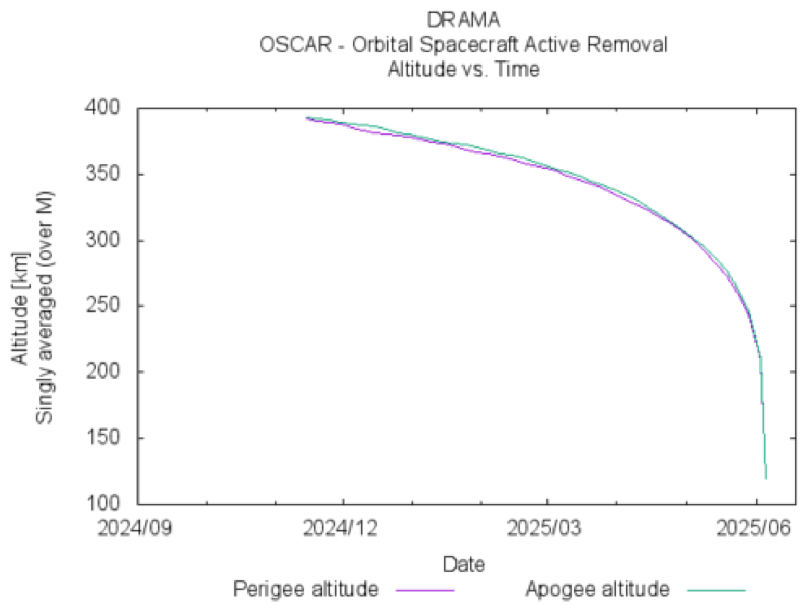


Figure 7.20: Latest prediction - SROC altitude vs time

Another analysis based on the latest prediction is the worst case/best case. Specifically, it was carried out a worst-case analysis with a confidence interval of 95%, which means that the worst-case results in solar and geomagnetic activity resembling historical activity data which is about 47.5% lower than the mean cycle but not higher than the cycle from the latest prediction. The predicted final date becomes (YYYY-MM-DD) 2025-11-16, as shown in Figure 7.21. Figure 7.22 confirms what has previously been presented regarding the dependence of the orbital lifetime on the solar activity: the worst case has a lower activity (light blue line), thus the drag is minor and the orbital lifetime is higher.

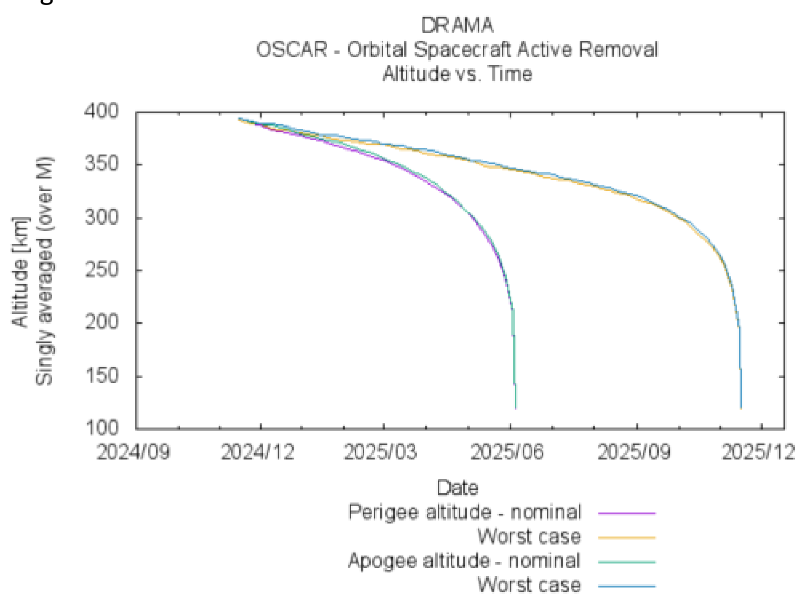


Figure 7.21: Lifetime comparison between the worst case and nominal case

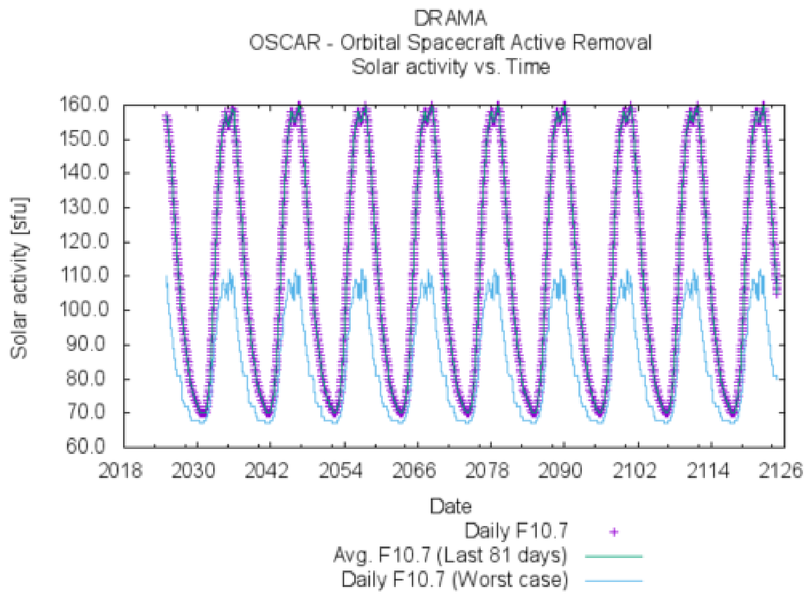


Figure 7.22: Solar activity comparison between worst case and nominal case

- **Monte Carlo Analysis**: final date (YYYY-MM-DD) 2025-04-09
  This analysis is one of the ISO-recommended methods [28] and requires the random selection for each day within the propagation time span of a solar and geomagnetic activity data triplet (daily and mean F10.7 as well as daily planetary amplitude Ap) from a specified number of solar cycles, which can vary between 1 and 6. The results of this analysis, which considers the last five solar cycles (from 19 to 23), are shown in Figure 7.23.

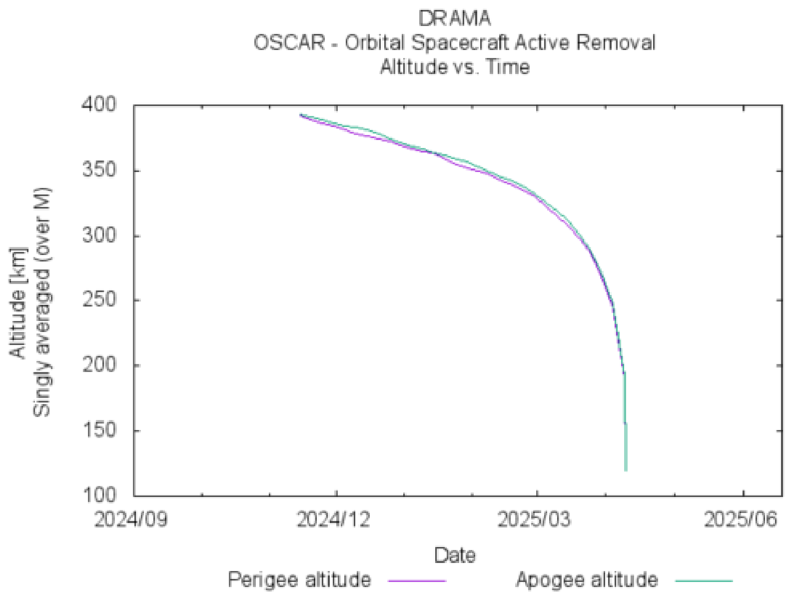


Figure 7.23: Monte Carlo sampling - SROC altitude vs time

7.4 SARA tool

Since the inherited propagation uncertainties do not allow to know in advance if or when a satellite (or one of its components) will hit Earth's surface, the risk of a specific re-entry is evaluated and then confronted with an accepted casualty risk threshold of 10-4 [30]. This assessment is performed using the tool SARA, whose main settings and satellite model definition are reported in this section.

7.4.1 SARA settings and SROC model definition

First, it is necessary to define the basic settings of the analysis. The software is run in expert mode to correctly define several parameters which could not be defined in basic mode. The initial orbit is the same used for ARES and OSCAR (Table 2.1), as well as the beginning date. The inputs for the propagation setting were defined in accordance with the setting for OSCAR, since SARA uses OSCAR to propagate the state of the satellite until it reaches 140 km in altitude. From here down, the propagator used is that of SARA, which also automatically evaluated the cross-section and the drag coefficient. Therefore, the following propagation settings only define OSCAR's propagation:

- Reflectivity coefficient: 1.3
- Cross-section: 0.101 m^3
- Drag coefficient: 2.2

The initial attitude was set to tumbling, while the attitude of the fragments was set to "inherited", which means that the attitude is inherited from the satellite. Since the SROC is randomly tumbling, the fragments will be randomly tumbling too. An important parameter is the Voxelator resolution length, which determines the size of the voxels used when estimating the aerothermodynamic properties of the compound object. A voxel is a 3D cube located on a three-dimensional grid used to create 3D models [31]: if its resolution length is low, it increases the faithfulness of the results, but it will also increase the run-time. To select a proper value, both the dimension of SROC components and SARA's limitation on the maximum number of voxels were considered, obtaining a Voxelator resolution length of 2 mm.

The environment is defined considering a dynamic atmospheric model, which includes a solar and geomagnetic activity database based on ESA's latest prediction's (the same used in OSCAR) and an atmospheric wind model. Finally, the on-ground risk is defined considering a casualty threshold of 15 J. This is the lowest kinetic energy to be considered for the on-ground risk assessment; 15J is the default values in SARA. The re-entry is modelled considering a non-controlled type from a circular orbit with an inclination of 6.2 deg. Figure 7.24 is a screenshot from SARA showing all the basic settings.

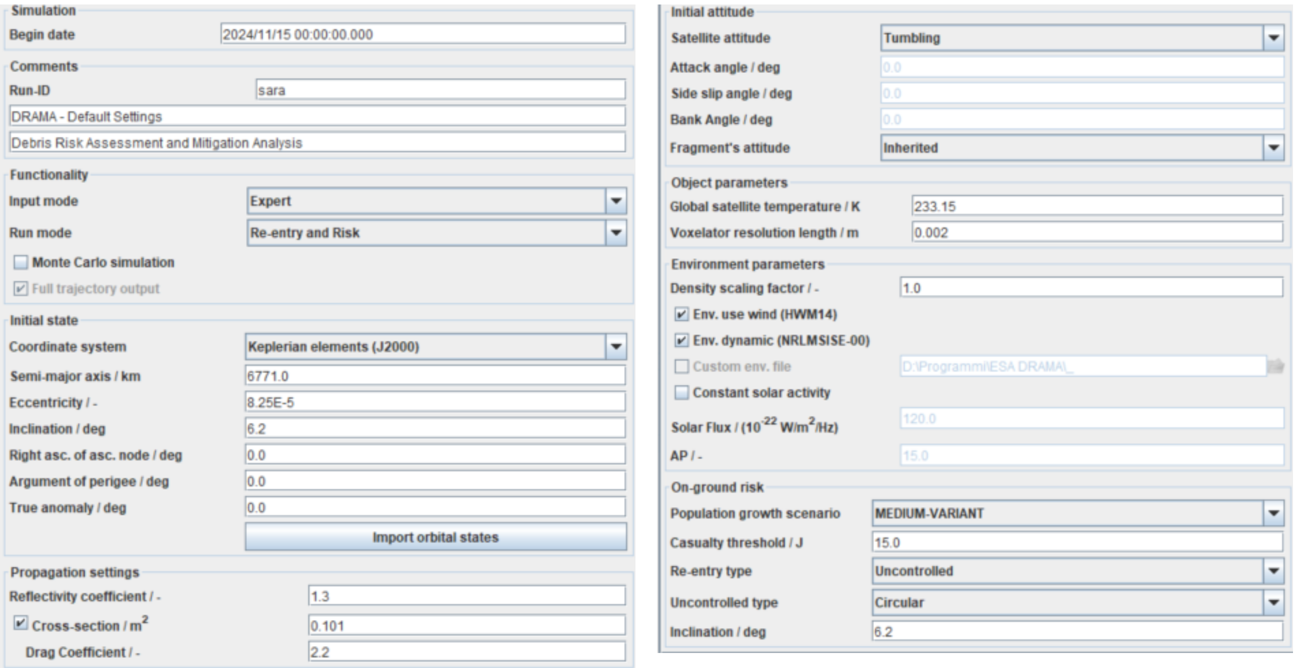


Figure 7.24: SARA Basic Settings

The SROC model was defined considering the recommendations from SARA's user manual [30]. To produce an accurate assessment of the re-entry risk, the model was built considering SROC's components with the highest mass and volume. Their virtual counterparts are modelled to resemble their mass, volume and shape as close as possible. Moreover, SARA lets the user select the material of the component from a material list, whose element either represent a physical material (such as the AA7075) or the properties of a piece of equipment. One example of this last category is the material El-Mat which is the equivalent material to model electronic components [29]. By, selecting the material, the following properties are set: density, melting temperature, specific heat at 300 K, heat of melting, conductivity and emissivity.

Each object added to the model is ordered with a parent-children method: components on the same level of the hierarchy can be connected one to another; moreover, it is possible to define the connection area between them and one or more “Dissolution Triggers”, such as the temperature or the altitude. When the conditions specified on the “Dissolution Trigger” window are met, the components separate from each other. For each parent component, it is possible to define a “Chid Release Trigger”: when one of the selected conditions is met the children objects contained inside the parent will be released. It is important to notice that when the “Chid Release Trigger” is triggered, the parent object disappears from the simulation even if it has not demised yet. For this reason, no “Child Release Trigger” is active during the analysis: the objects contained inside another one are released only when the latter is demised.

The higher hierarchical level of the model is composed by of following elements:

- Structure
- Solar Panel x (the solar panel on the positive x face of SROC)
- Solar Panel y (the solar panel on the negative y face of SROC)
- UHF Antenna

All the last three objects are connected to the structure, with a connection area equal to the contact area they have with the structure (0.0174 m^2 for the solar panels and 0.00746 m^2 for the antennas). The dissolution trigger for all these connections is when the altitude decreases below 103 km, as suggested by the SARA user manual [30]. Table 7.3 contains a list of all the components added to the model, as well as their mass, volume, shape and material used. The mass and volume of the components reported in the SROC system definition file [20] are added to compare the differences with the objects in the SARA model.

It is noticeable that for a few components, the relative error with respect to the system defined in phase B1 is high. This is due to the fact that for some elements it was not possible to equally represent both the volume and the mass of the objects. For the Load Controller Module (LCM) and the RWA it was preferred to use a bigger volume to maintain a representative mass of the component. The Torque rod, instead, is maintained a lower mass since the absolute difference between the actual component is only 58 g which does not affect significantly the mass budget of the model.

The following materials were used to define the structural and thermal properties of the equipment composing the satellite [29]:

- Drama-AA7075: this class is used for aluminum alloys as the baseline. It is used for most of the components of the model; as suggested in the SARA user manual, if an object:
    - is constructed primarily of aluminum or magnesium;
    - has a mass under 5 kg;
    - contains no contiguous parts of a higher demise temeperature material which are over 50 kg, and the accurate properties of such components are not available, it is allowed to use the Drama-AA7075 material.
- Drama-A316: class example for steel alloy baseline.
- Drama-El-Mat: this material is used to model electronic components, including boards and wiring thereon, but not the casing that includes them. For this reason, it was used only for the boards and the wiring.

Table 7.3: Components of the SARA model

Component	Mass [kg]	Mass Relative Error	Volume [U]	Volume Relative Error	Shape	Material
Structure	5.400	0%	12U	0%	Box	Drama-AA7075
Solar Panel (x2)	0.540	0%	0.138	-0.7%	Box	Drama-AA7075
Antenna UHF	0.105	0%	0.077	0%	Box	Drama-AA7075
Thruster Module	4.026	0%	3.703	0%	Box	Drama-AA7075
RWA	1.059	-11.5%	0.092	+ 95.7%	Box	Drama-A316
Battery Module Assemlby	0.473	0%	0.123	0%	Box	Drama-AA7075
Endeavour Avionics Module	0.588	0%	0.265	0%	Box	Drama-AA7075
Backplane PCBA	0.48	0%	0.445	0%	Box	Drama-El-Mat
DOCKS-A	0.272	0%	0.352	-24.6%	Cone	Drama-AA7075
Payload Interface Board	0.300	0%	0.124	0%	Box	Drama-AA7075
NFOV Camera	0.108	0%	0.081	-1.2%	Cylinder	Drama-AA7075
LIDAR	0.036	0%	0.0259	0%	Box	Drama-A316
Payload	0.061	0%	0.034	-2.9%	Cylinder	Drama-AA7075
IR Camera	0.145	0%	0.102	-1.4%	Cylinder	Drama-AA7075
Housekeeping Board	0.092	-12.6%	0.003	-25%	Box	Drama-El-Mat
Torque rod (x3)	0.017	-78.4%	0.004	0%	Cylinder	Drama-AA7075
LCM	0.125	-0.7%	0.031	+ 82.3%	Box	Drama-AA7075
Harness (x2)	0.148	-6%	0.034	-	Box	Drama-El-Mat

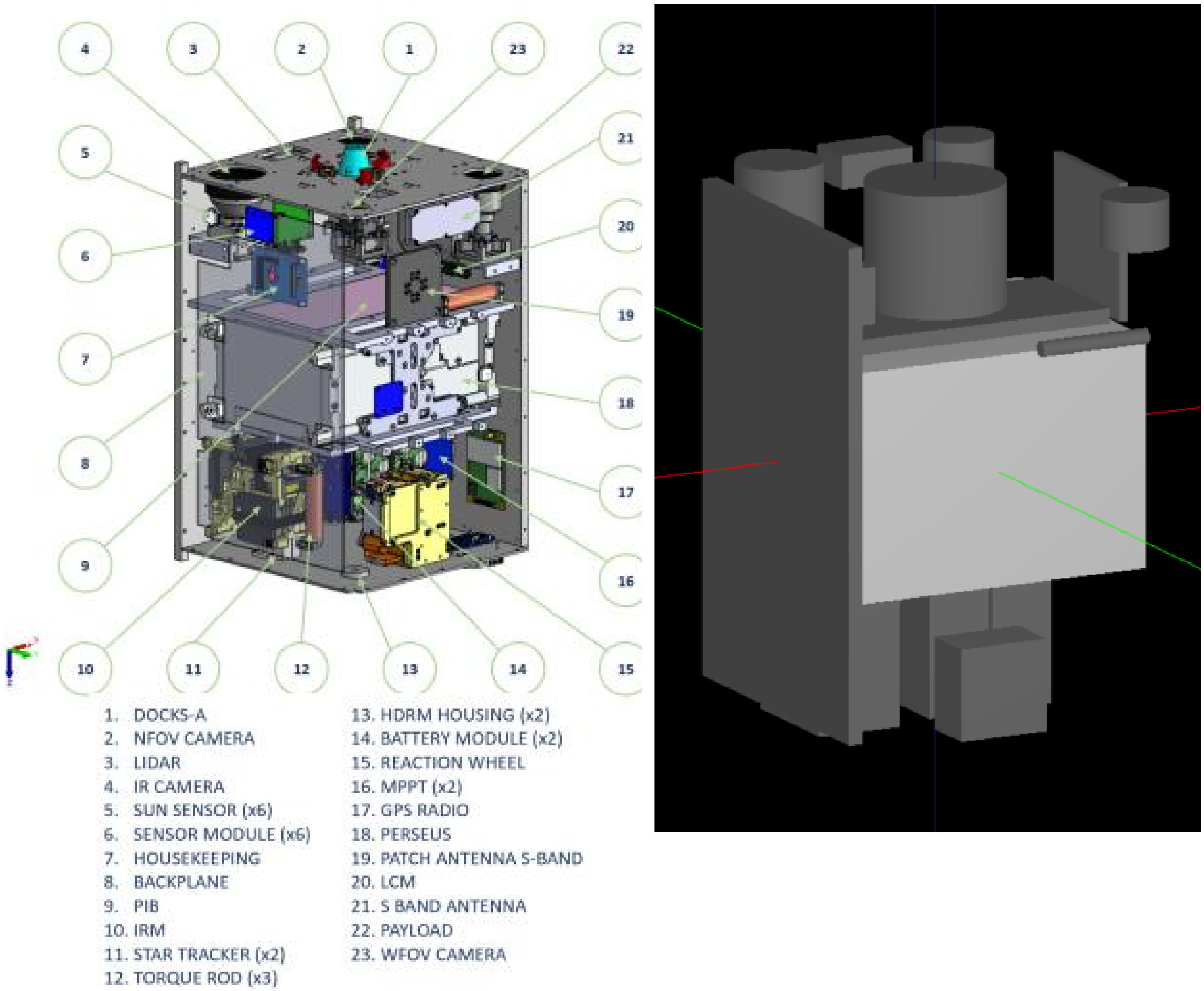


Figure 7.25: Comparison between SROC internal view (left) with the SARA model (right)

SARA user manual [30] recommends creating a model with at least 75% of the spacecraft's mass. This recommendation is respected since the total mass of the model is 13.945 kg, while the total mass reported in the SROC system definition file [20] is 16.062 kg, which means that 84.82% of SROC mass is considered. It is noted that the reference values used to build the model include the margins on the single component, while the total system margin is not considered. A comparison between the CAD of the internal components of SROC and the components of the SARA model is presented in Figure 7.25.

7.4.2 SARA results

The most important result is the assessment of the total re-entry risk:

- Total casualty area: 0
- Total casualty probability: 0
- Total fatality probability: 0

So, according to SARA's estimate, the satellite does not constitute a potential threat to on-ground safety since it is completely demised during the re-entry. Before starting the analysis of the re-entry of the spacecraft and its fragments, it is necessary to define when, according to SARA, an object is demised [32]:

- Complete mass loss (on both Figure 7.26 and Figure 7.27 these points are marked as "Demise points");

- The kinetic energy of the object drops below the 15 J threshold (on the graphs these points are marked as “Uncritical points”);
- Ballooning, where an object is not allowed to become unphysically thin (on the graphs these points are marked as “Ballooning points”);Figure 7.26:

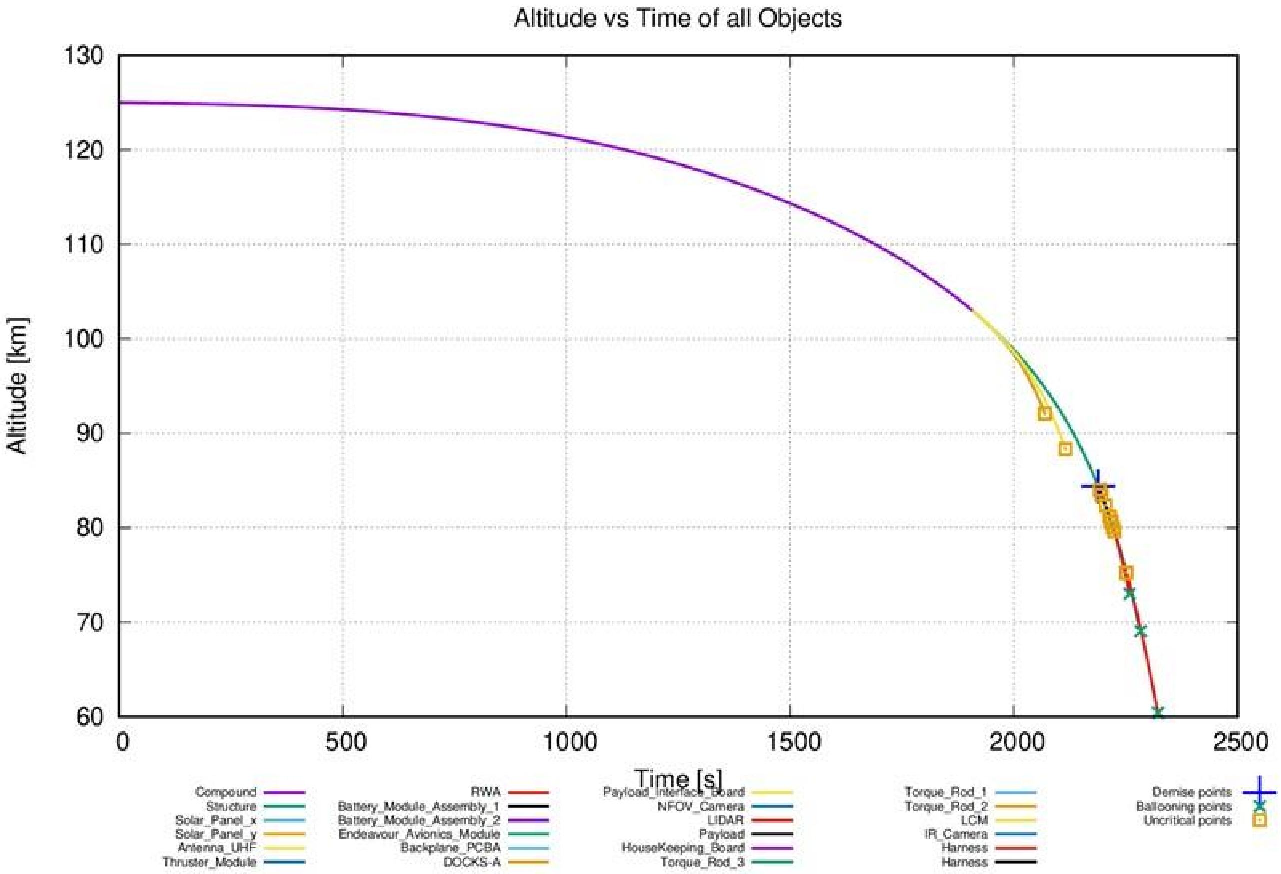


Figure 7.26: Altitude as function of the Time of all Objects

Figure 7.26 shows that all the objects demise before impacting with the ground. Approximately for the first 1800 seconds from the start of the re-entry, the compound is still whole, but at 103 km, as imposed by the dissolution trigger, the solar panels and the antenna separate from the structure and are then demised respectively around 92 km and 88 km. The structure keeps its re-entry until it demises at approximately 85 km (this point is highlighted by the blue cross in the graph) at 2200 seconds. After that, all the internal components inside the structure are released and most of them reach the uncritical point in the successive minutes (red squares). The last component to demise is the RWA which reached the ballooning point at 60 km after approximately 2300 seconds from the start of the re-entry. The reason why this component demises after all the others is probably due to its material (Drama-A316) which is more resistant than the Drama-AA7075 to the high temperature faced during the re-entry.

Figure 7.27 shows the downrange of all the objects: all of them demise between a downrange of 15000 km and 16500 km, with the longest distance obtained by the RWA assembly.

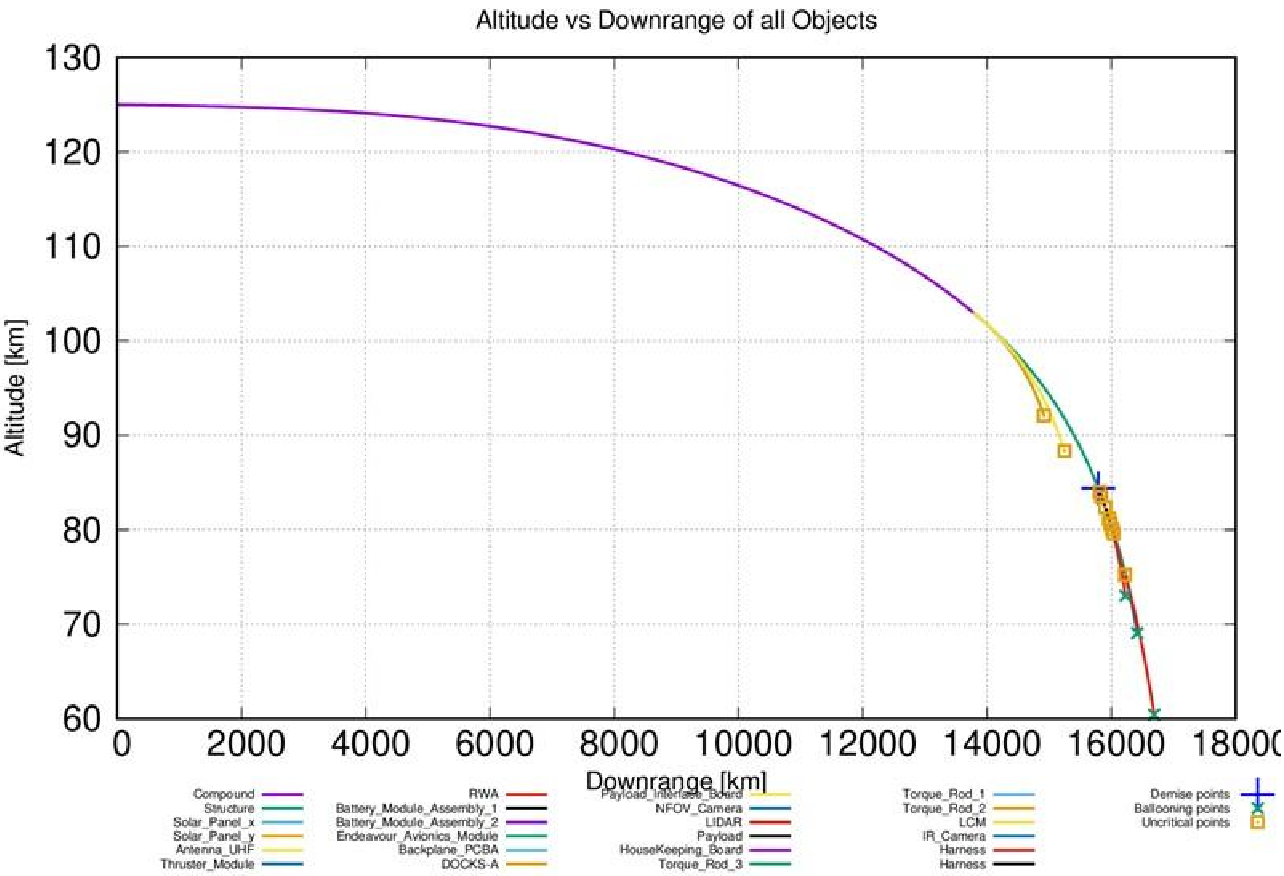


Figure 7.27: Altitude as function of the Downrange of all Objects

8 Conclusions

This thesis was carried out using three different software (Matlab, STK, and DRAMA) to perform several tasks related to the mission analysis and trajectory optimization for Phase B2 of the SROC project, a 12U CubeSat mission. At the beginning of this document, a brief review of the mission was presented to give some context about the mission objectives, its requirements, and the two Concept of Operations.

All the STK scenarios, Matlab function, and files JSON were reorganized to delete the unnecessary ones or to unify similar functions into only one; moreover, all the variables used by this software were renamed to be coherent with the names reported in the MAR produced at the end of the phase B1. Finally, the Matlab function which optimizes the IPA rendezvous and the one which defines HP2 and HP3 were modified to improve respectively the optimization of the IPA and the performance during the HPs. After these changes, the nominal scenario was updated to a new orbit and used to study several aspects of the mission: the WSE of the observation phase, the ground station coverage, and the optimal time windows to perform the Final Approach. It was noticed that the ground station coverage may not be enough to guarantee a sufficiently long communication window, so two possible solutions were presented. Using a GEO satellite constellation could be the best option since it guarantees an uninterrupted link with SROC.

It was proven that the synergic use of Matlab and STK can greatly decrease the duration of an iterative analysis by automatizing the set-up, run, and post-processing of every iteration. This property was particularly useful during the variant scenarios analysis, where STK's object model interface was used to connect the STK scenario to the Matlab functions and to evaluate the nominal or variant segments of the MCS. At first, all the possible variant events were considered isolated, which means that every one of them was evaluated inside an MCS where no other variant events took place. Thanks to this process it was possible to assess how every variant event changes the total duration and deltaV cost, and if and how they influence the successive segments of the MCS. For some variants, different solutions were considered to either minimize the deltaV cost or the duration of the mission. After that, all the possible combinations of variant events were analysed for both the Observe and the Observe&Retrieve scenarios. All their results were saved in an Excel database with an interactive interface that lets an external user define the desired MCS and then shows the relative DeltaV and duration budgets. All the variant scenarios were analysed to define which were off-nominal because of a higher deltaV cost, higher duration, or inadequate safety relative to Space Rider. For every variant MCS it was found at least one valid solution, thus proving the robustness of the mission.

The ESA software DRAMA was used to study other aspects of the mission, all required by SROC's Statement of Work for Phases B2/C/D. The number of CAM manoeuvre and the relative deltaV cost was assessed as a function of the Accepted Collision Probability Level; since this analysis gave a mean number of annual manoeuvres inferior to one, the annual deltaV suggested by DRAMA was rescaled to the cost of one manoeuvre. The OSCAR tool was used to verify the compliance of the SROC mission with the Space Debris Mitigations for Agency projects and, finally, the tool SARA was used to assess the total re-entry risk. To perform this last task, a simplified digital twin of SROC was generated using SARA. The results of the analysis with OSCAR confirmed the compliance with the Space Debris Mitigations for Agency projects, while the SARA analysis assessed that the mission does not constitute any on-ground risk, since the satellite demises before an altitude equal to 60 km.

The next step for Phase B2 could involve a new definition for the WSE: as mentioned before, the current Matlab functions produce an acceptable approximation if used to evaluate the deltaV cost and the duration of the observation phase, but the fact that it is not possible to define the exact geometrical features of the

WSE does not make it suitable to evaluate other aspects of this phase, such as the interval at which the range and illuminations constraints are met. This improvement could be pursued through a new Matlab function that analytically evaluates the desired WSE or using STK's default segments for rendezvous and proximity operations. Another interesting update could be adding a thruster set module which is representative of the properties of SROC's thruster module. Finally, regarding the analysis of the communication windows with SROC, it will be necessary to assess if a GEO link is feasible from different points of view, such as the cost and the system ones.

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