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Design of natural collision-free trajectories for the mission extension phase of a remote sensing formation flying mission

Francesca Scala[a]*, Camilla Colombo[b], Manuel Martìn-Neira[c]

[a] *Department of Aerospace Science and Technology, Politecnico di Milano, Via La Masa, 34, 20156 Milano, Italy*, francesca1.scala@polimi.it
[b] *Department of Aerospace Science and Technology, Politecnico di Milano, Via La Masa, 34, 20156 Milano, Italy*, camilla.colombo@polimi.it
[c] *ESA-ESTEC, Keplerlaan 1, Postbus 299, 2200-AG Noordwijk, The Netherlands*, Manuel.Martin-Neira@esa.int
* Corresponding Author

Abstract

The safety concept is of paramount importance in the trajectory design of formation flying missions. The selection of natural collision-free trajectories is based on the analysis of natural relative motion among the satellites. This is beneficial for the mission both in terms of propellant consumption and control effort, allowing a naturally safe environment. This paper focuses on a formation flying for remote sensing missions in low Earth orbit, in the direction of future Earth observation missions. We consider as a baseline the Formation Flying L-band Aperture Synthesis mission concept, proposed by the European Space Agency. This work proposes natural collision-free trajectories to extend the scientific campaign at the end of the nominal operative life of the FFLAS mission. The possibility to extend the scientific operations, before the final atmospheric re-entry phase, could provide a significant amount of data to improve meteorological and climate prediction. The scenario selected is based on a close formation flying, with a nominal inter-satellite distance in the order of tens of meters. This allows the satellites to behave as a node of a distributed payload for Earth observation, increasing the accuracy in the measurements, thanks to the increase of the radiometer's aperture size. We present two possible strategies to design the extension phase of this mission. First, a single satellite science mission is envisioned, moving the satellites on the same reference orbit, with a certain separation angle. In this new configuration, each satellite operates singularly on different orbits, as a single satellite science mission. Second, we consider the possibility to increase the relative distance among the satellites, maintaining the possibility to do Earth observation with distributed payloads. We perform some analyses to select the augmented geometry, with a bigger formation baseline in the order of tens to hundreds of meters. The analyses are driven by the need to use natural collision-free relative trajectories since at the end-of-life the low thrust control is limited. The final decay of the satellites is provided via a deorbiting low-thrust manoeuvre, to comply with the 25-year mitigation rule. The main idea is to propose disposal in less than three months. The results of the analyses are obtained including the effects of the orbital perturbations, thanks to a high-fidelity relative motion propagator. The relative orbital elements environment is introduced to assess the formation safety more straightforwardly.

Keywords: Formation flying, Dynamics, Control, FFLAS

1. Introduction

In the last 40 years, scientific space missions increase their importance. They provide vital information for everyday life, improving the knowledge of weather forecasts, climate change, agricultural practices, and ocean wealth. ESA and NASA are increasing the number of satellites dedicated to observing Earth from space, developing new scientific instruments and new dedicated mission designs. Every mission is typically designed with a primary or nominal phase, which comprises the scientific phase, where the science data are collected. Most of the missions are tested and designed considering a certain lifetime with a margin. For this reason, it is quite common that a mission close to the end of the nominal phase has still the capability from the system point of view of continuing the operations. An extension of the mission lifetime can be implemented whenever this new phase could improve the validity of the collected data and the understanding of the scientific phenomena. As from a NASA report in 2016 [1], approximately three-quarters of the space science missions are in their extended operations phase, as the GRACE mission, which operated for about 15 years, with respect to its 5 years of nominal lifetime. Another example is the extension of the ESA's Integral mission in 2004. Integral is expected to continue its operations till the end of 2021 [2]. This paper presents the analyses performed to study the possible mission extension phase of the Formation Flying L-band Aperture Synthesis (FFLAS) mission concept. FFLAS is a mission study proposed by the European Space Agency and currently studied at Airbus Space and

Defence and Politecnico di Milano [3]. This work proposes a natural collision-free trajectory to extend the scientific campaign at the end of the nominal operative life of the FFLAS mission. The possibility to extend the scientific operations, before the final atmospheric re-entry phase, could provide a significant amount of data to improve meteorological and climate prediction. In this paper, we present two possible strategies to design the extension phase of this mission. First, a single satellite science mission is envisioned, moving the satellites on the same reference orbit, with a certain separation angle. In this new configuration, each satellite operates singularly on different orbits, as a single satellite science mission. Second, we consider the possibility to increase the relative distance among the satellites, maintaining the possibility to do Earth observation with distributed payloads. We perform some analyses to select the augmented geometry, with a bigger formation baseline in the order of tens to hundreds of meters. The analyses are driven by the need to use natural collision-free relative trajectories since at the end-of-life the low thrust control is limited.

Section 2 presents the FFLAS mission study, providing the reference orbit properties and the main mission operations. Sections 3 and 4 present the single satellite and the formation flying mission extension phase, respectively. The design approach and procedure are presented, together with the main results from the simulations. Finally, at the end of Section 4, a consideration of the payload performances is reported for the new formation geometry for the extension phase.

2. Formation Flying L-band Aperture Synthesis

The main idea of the FFLAS mission study comes from the outcomes of the Soil Moisture and Ocean Salinity (SMOS) mission. The formation flying concept is introduced to achieve the concept of distributed payloads as a potential opportunity to increase the virtual aperture size and the spatial resolution. For the case of FFLAS [3], three satellites flying in a triangular formation are selected, carrying a hexagonal antenna array each, as shown in Fig. 1. The antenna array has a diameter of 8 m The distance among the antenna array centres is selected to be about 12 m, to guarantee the best performances for the spatial resolution of the observation. It is expected an equivalent aperture of 21 km with a consequent spatial resolution of 1-10 km [4,5], giving a significant improvement with respect to the 40 km resolution of the SMOS single satellite. This improvement of achieving a larger aperture with spacecraft in formation flight is in the direction of Earth’s observations scientific needs, where most of the hydrological processes occur in the 1-10 km scale. The reference orbit for the FFLAS formation is a Sun-Synchronous Orbit (SSO) and its characteristics are reported in Table 1, together with the delta-v and mass budget for each satellite.

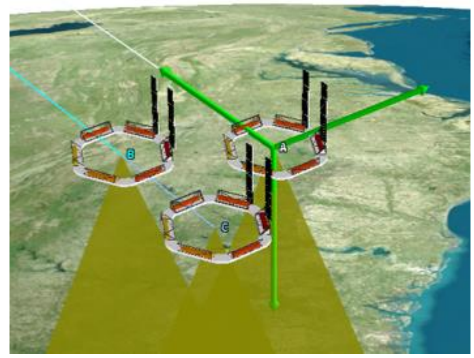

Fig. 1. FFLAS formation flying in the nominal scientific configuration. The yellow cones represent the payload field of view, while the green axes the body frame.

Table 1. FFLAS reference orbit properties

Parameter	Value
Mean altitude	775 km
Eccentricity	0
Inclination	98.78°
Mass budget	~ 1600 kg
Delta-v budget	~ 2300 m/s
Maximum thrust	25 mN (low thrust engine)

2.1 Nominal mission operations

The nominal mission operations for the FFLAS mission are briefly reported in Fig. 2. The initial phase consists of the launch and the satellite's orbit injection. A dedicated mechanism is developed for the insertion and deployment of the satellites in orbit. Then the LEOP and the commissioning is designed, where the in-orbit test and verification of the single satellites functionalities and subsystems are performed. After the achievement of the relative motion and formation reconfiguration [6], the satellites will fly in a triangular formation, with the payload aperture plane in Earth pointing direction. Once per month, the calibration of the scientific instrument of FFLAS is required, thus the transition between the Earth pointing and the Cold Sky Pointing mode is required. In Scala et al. [7] the optimal strategy for the manoeuvre is fully described. This paper aims at providing the strategy for a possible mission extension at the end of the nominal mission operations, just before the disposal.

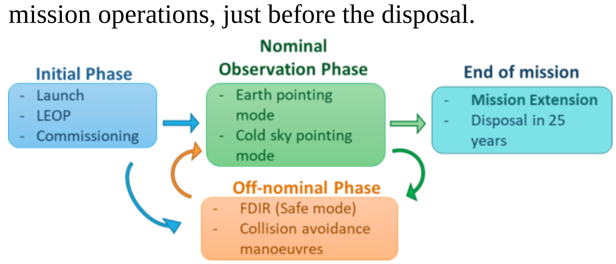

Fig. 2. Nominal FFLAS mission operations

2.2 Mission Extension Phase

For the mission extension phase, two main strategies are identified:

1. A single-satellite mission extension, where no relative motion is required, and the satellites are placed on the same Sun Synchronous Orbit with a certain angular distance.
2. A formation flying mission extension, where the satellites are placed on a closed relative motion with minimum control effort.

The mission extension phase is designed based on the fuel requirement at the end-of-mission when only a minimum part of the onboard propellant is still available. An amount of 1% to 5% of the overall propellant budget could be present at the end of the nominal mission operations. Depending on the analysis of the delta-v budget to maintain the formation, an estimation of the time for the extended mission phase could be provided. Sections 4.1 and 4.2 provides a mathematical background for the relative motion design in presence of orbital perturbation, such as the Earth's oblateness, and the design of the control algorithms.

3. Strategy 1: Single Satellite Mission Extension

This section presents the analyses for the first strategy, to extend the scientific phase of the FFLAS mission with a single/satellite logic. The strategy followed for the preliminary design is based on the following steps:

- At the end of the mission, the satellites are moved on the reference SSO with a certain separation angle along the orbit.
- The manoeuvre is based on a low thrust reconfiguration manoeuvre, which is typically designed by transferring to a lower or nearby orbit, then drift in this orbit for a certain time, then return to the original orbit.

For the high thrust case, this manoeuvre consists of two impulses with Hohmann transfers. On the other hand, for the low thrust case, the continuous thrust is exploited during the transfer legs. The performance parameters for this approach are the delta-v required by the manoeuvre and the selection of $\Delta\theta$ to provide a safe natural evolution among the satellite. This allows a non-propulsive mission extension phase.

3.1 Low thrust phasing design

The phasing manoeuvre is designed considering a continuous thrust in the tangential direction of the radial-transversal-normal frame as a_{T_θ}. Under the hypothesis that the radial acceleration is null, the equations of motion for each satellite are:

$$\begin{aligned} \dot{r} &= v_r \\ \dot{\theta} &= \frac{v_\theta}{r} \\ \dot{v}_r &= \frac{v_\theta^2}{r} - \frac{\mu_\oplus}{r^2} \\ \dot{v}_\theta &= -\frac{v_r v_\theta}{r} + a_{T_\theta} \end{aligned} \tag{1}$$

Where a_{T_θ} is the constant acceleration in the tangential direction, $\dot{r}$ and $\dot{\theta}$ are the radial and the transversal velocity, $\dot{v}_r$ and $\dot{v}_\theta$ are the radial and transversal acceleration.

Solving the system, it is possible to compute the satellite position radius in the Earth frame as a function of the manoeuvre time:

$$r_t = \frac{r_0}{\left(1 - \frac{a_{T_\theta}\, t}{\sqrt{\mu_\oplus / r_0}}\right)^2} \tag{2}$$

For the phasing problem, the final radius after the first leg is computed from the classical Hohmann phasing problem for impulsive systems. For this reason, in our problem, the initial radius r_0 and the radius at the end of the first leg r_t are known. Hence, equation (2) is used to compute the transfer time t.

3.2 Simulation parameters

This paragraph provides the parameters used in the simulation of the mission extension phase. The nominal geometry configuration of the three-satellite formation is characterised by the state vector reported in Table 4. Moreover, the parameter to initialise the phasing analyses are reported, as well as the mean Keplerian elements of the reference orbit. The generic orbital configuration in the EME_J200 frame for the final configuration after the phasing is reported in Fig. 3.

Table 2. Simulation parameters for the mission extension.

Parameter	Value
Initial state in Earth Pointing	
Sat A	$[0, \quad 0, \quad -5.3997, \quad 0, \quad 0, \quad 0]$
Sat B	$[0, \quad 6.2350, \quad 5.3997, \quad 0, \quad 0, \quad 0]$
Sat C	$[0, -6.2350, \quad 5.3997, \quad 0, \quad 0, \quad 0]$
Parameter for the phasing manoeuvre	
$\Delta\theta$	0° to 50°
T_{max}	25 mN
Mean elements of the reference orbit	
Kep_0	$[7.1531 \cdot 1e6, \quad 0, \quad 98.51°, \quad 270°, \quad 0°, \quad 0°]$

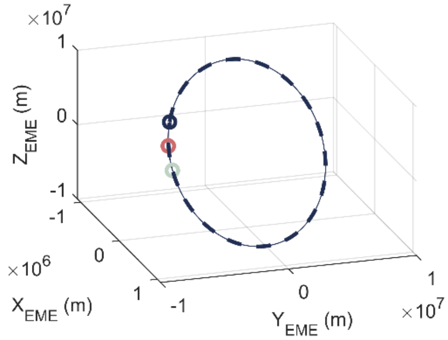

Fig. 3. Orbital configuration in the EME_J2000 with 15° separation among the satellites.

3.3 Simulations results for Strategy 1

This section presents the possibility to move the three satellites from the nominal formation geometry to the same reference SSO with a certain $\Delta\theta$. The performance parameters for this approach are the delta-v required by the manoeuvre and the selection of $\Delta\theta$ to provide a safe natural evolution among the satellite. Moreover, another important aspect is the analysis of the natural evolution of the new satellite configuration, to ensure that no collision risk among the satellites is present.

3.3.1 Delta-v for phasing manoeuvre

To evaluate the performances and the feasibility of this strategy, a first analysis for the delta-v required by the classical Hohmann phasing is computed. A parametric analysis is performed considering different separation angles $\Delta\theta$ and different lengths of the manoeuvre. The latter is expressed in terms of the orbital period of the reference trajectory, called no. of revolutions. The results of the analysis are reported in Fig. 4. It can be seen how the manoeuvre is more expensive with a larger $\Delta\theta$ and with a smaller manoeuvre period.

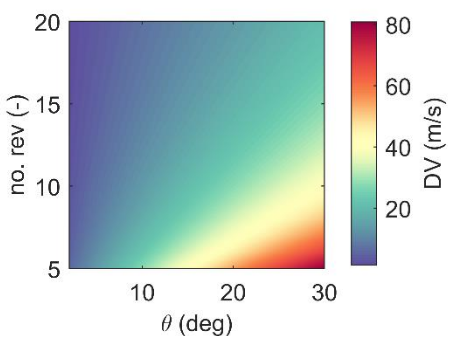

Fig. 4. Delta-v for impulsive phasing manoeuvre.

The second analysis is performed with low thrust technology, exploiting the thrust in the transversal direction. The following approach is implemented:

- For each case, a first low thrust step is computed, where the engines provide the control effort to reach the coasting (or phasing) orbit.
- After the satellites reach the coasting orbit, a second leg is set with no propulsive effort.
- Finally, a third leg is computed with the control effort to reach the final target position on the initial orbit.

The propelled legs are identified by a thrusting time T_{T1} and T_{T2}, respectively. The overall dynamics are solved and optimised to reduce the thrusting time and achieving the commanded phasing angle. In this paper, the case of a 10-degree phasing with 15 orbits revolutions is analysed. Moreover, in view of minimising the delta-v cost, the maximum thrust is limited at 1 mN. The results of this preliminary analysis are shown in Fig. 5, where the variation of the semi-major axis and altitude of the orbit is reported. As expected, to increase the $\Delta\theta$ of 10 degrees requires a phasing orbit with a smaller semi-major axis. Thus, Fig. 6 shows a first decrement and finally an increment of the semi-major axis. The central part, where a remains constant, represents the coasting phase. The delta-v required by the manoeuvre is about 15 m/s, which is less than 1% of the delta-v budget for the FFLAS satellites. This provides the feasibility of the reconfiguration phasing manoeuvre at end of the mission.

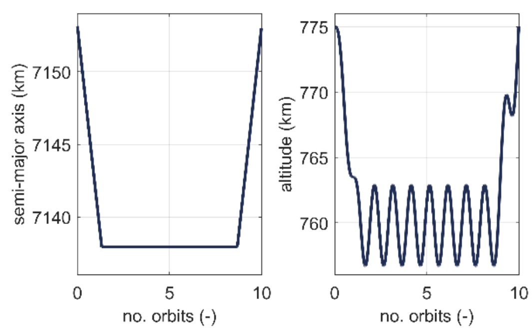

Fig. 5. Variation of the semi-major axis and the orbit altitude during the low thrust phasing manoeuvre.

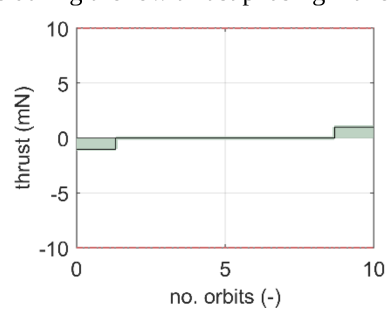

Fig. 6. Thrust profile during the low thrust phasing.

3.3.2 Extension period for different phasing angles

This analysis should relate to the considerations on the natural evolution of the satellites in their final configuration. Some examples of the available mission extension period are reported in for 5 selections of $\Delta\theta$. The collision threshold, in this case, was set equal to 100 m, for a safer extension phase. This strategy could benefit from some control maintenance to remove the natural drift due to external perturbations. This could be investigated in phase B of the mission study.

Table 3. Extension period for different $\Delta\theta$ conditions.

$\Delta\theta$ (deg)	Extension period
2	~ 11 months
5	~ 11 months
10	~ 1 year and 2 months
15	~ 1 year and 6 months
30	~ 1 year and 10 months

4. Strategy 2: Formation Flying Mission Extension

This section focuses on the second strategy and analyses the feasibility of a formation in the projected circular orbit for Earth observation purposes [8]. This strategy allows the possibility of having a projected circle in the plane normal to the radius direction and allowing the continuation of the payload observation as distributed nodes, with degraded performances.

In this scenario, the maintenance of the formation should require null or minimum control effort and the formation should remain safe from an inter-satellite collision point of view with no control effort.

4.1 Close formation flying design under J2 effect

Classically, the relative motion of a satellite with respect to the reference is described in Hill's equations [9], where the only external force is the gravitational field of the central body. A close relative motion can be obtained imposing in the Hill-Clohessy-Wiltshire (HCW) equations, the following condition [8]:

$$\dot{y}_0 = -2\, x_0\, n \tag{3}$$

Where $\dot{y}_0$ is the initial relative velocity in the transversal direction, x_0 is the initial relative position in the radial direction, and $n = \sqrt{\mu/a_0^3}$ is the reference orbit mean motion. For a formation in Low Earth Orbit (LEO), it is important to consider the effect of the Earth's oblateness, in the orbit design. It affects the relative motion, changing the perigee and the node precession rate [8]:

$$\dot{\Omega} = -\frac{3}{2} J_2 \frac{R_e^2}{p^2} \dot{M} \cos i \tag{4}$$

$$\dot{\omega} = \frac{3}{2} J_2 \frac{R_e^2}{p^2} \dot{M} \left(2 - \frac{5}{2} \sin^2 i\right) \tag{5}$$

$$\dot{M}_p = \frac{3}{2} J_2 \frac{R_e^2}{p^2} n \sqrt{1-e^2} \left(1 - \frac{3}{2} \sin^2 i\right) \tag{6}$$

Where J_2 is the Earth oblateness term, R_e is the Earth's radius, p is the reference orbit's semi-latus rectum, and i nd e is the inclination and the eccentricity of the reference orbit, respectively. An immediate strategy to include the Earth's oblateness effect in the relative orbit design is the description of the dynamics in terms of Relative Orbital Elements (ROEs).

For the case under the analysis of a close formation for Earth's observation purposes, a closed relative motion is required to maintain the satellites in the correct relative position for scientific observation. Moreover, the need of minimizing the control effort for the mission extension phase requires the design of a natural bounded motion. It can be shown that modifying the in-plane and out-of-plane phasing, different close relative trajectories can be designed in the radial-transversal-normal frame (x, y, z):

- Projected Circular Orbit (PCO) in the xz plane or the yz plane,
- Projected elliptical orbits in the xz plane (cartwheel and helix) or the yz plane.

In this section, the possibility to use a PCO in the yz plane is investigated. Maintaining a circular projection in the plane orthogonal to the radial direction, allow the satellite to maintain the same footprint for payload interferometry purposes. The PCO is typically used for remote sensing, thanks to its geometry. Finally, the need to reduce at minimum the control effort affects the safety concept of the extension phase. A larger inter-satellite distance is imposed to reduce the collision risk among the satellites. Important consideration must be done concerning the FFLAS payload performances. A detailed analysis should be done to assess whether the L-band interferometer payloads properly work in this new geometry formation, with a larger inter-satellite distance to guarantee a safer formation flying.

4.1.1 Projected Circular Formation

This paragraph presents the design of a PCO, starting from the representation in differential elements from [8]. The periodic solution of the relative motion can be described by four parameters: the amplitude of the motion in the x, y and z direction, as ρ_x, ρ_y and ρ_z, and the angular phase angles, as α_x and α_y.

To obtain a PCO in the yz plane, we impose the following conditions

$$\begin{aligned} \alpha_x &= \alpha_z \\ \rho_z &= 2\,\rho_x \\ \rho_y &= 2\,\rho_x \cos\alpha_x \end{aligned} \tag{7}$$

The condition to initialise the PCO motion in ROEs frame can be computed from the analysis in the differential elements from [8], transforming the differences in orbital elements into ROEs definition [10], with $\delta\boldsymbol{\alpha}_{\boldsymbol{roe}} = [\delta a, \delta\lambda, \delta e_x, \delta\, e_y, \delta i_x, \delta i_y]$:

$$\delta a_0 = 0 \tag{8}$$

$$\delta\lambda_0 = \frac{\rho_y}{a_0}$$

$$\delta e_{x,0} = -\rho_x \sin\alpha_x$$

$$\delta e_{y,0} = -\rho_x \cos\alpha_x$$

$$\delta i_{x,0} = \rho_z \cos\alpha_z$$

$$\delta i_{y,0} = -\rho_z \sin\alpha_z$$

At this point, we must include the effect of the secular J_2, generates a drift among the satellites in the formation, due to different angular rates $\dot{\Omega}$, $\dot{\omega}$, $\dot{M}_p$. This difference in the angular rates causes both across and along-track drifts. First, the effect of J_2 could be included in the differential semi-major axis δa, as a condition to eliminate the along-track drift [8]:

$$\delta a = -0.5\, \mathrm{J}_2 \left(\frac{R_e}{a_0}\right)^2 \left(\frac{3\,\eta_0 + 4}{\eta_0^4}\right) \times \left((1 - 3\cos^2 i_0)\frac{e_x \delta e_x + e_y \delta e_y}{\eta_0^2} + \sin 2i_0 \delta i\right) \tag{9}$$

Where $\eta_0 = \sqrt{1-e^2}$. On the other hand, the condition to remove the cross-track drift is satisfied only for an interval of reference orbit inclination (49.11° to 63.43°). Thus, this condition cannot be applied for the FFLAS situation, having an inclination of 98.51°. Section 4.1.2 presents an ideal control to control the perturbing effects due to the Earth's oblateness.

4.1.2 Ideal control for Projected Circular Orbits

The satellites of the FFLAS are equipped with four low thrust engines, with a maximum capacity of 25 mN. The maintenance during the mission extension phase is designed considering a reduced maximum capability to minimise the delta-v budget for this phase. As a reference continuous control, the ideal control to account for the J_2 the effect is modelled. Specifically, from the modified HCW [8], the control acceleration in the radial-transversal-normal frame can be computed. It depends on the perigee $\dot{\omega}, \dot{M}_p$ and the node precession rate $\dot{\Omega}$ (see Eq. (4), (5) and (6)), which also influences the phase angle rate $\dot{\alpha}$ and the precession rate of the argument of latitude $\dot{\lambda}$:

$$\dot{\alpha} = \frac{1}{17}\left(8\,k \sin^2 i_0 - \dot{\omega}\right) \tag{10}$$

$$\dot{\lambda} = n - n\,J_2 \left(\frac{3\,R_e^2}{4\,a_0^2\,\eta^4}\right)\left(\eta\,(1 - 3\cos^2 i_0) + (1 - \cos^2 i_0)\right) \tag{11}$$

Where n is the mean motion of the reference orbit, $\eta = \sqrt{1-e^2}$, and the parameter $k = -1.5\,n\,J_2 \left(\frac{R_e}{a_0}\right)^2$. The ideal control for the PCO is computed as follow, from the J_2 HCW equations with the reference trajectory:

$$u_{x,c} = 0 \tag{12}$$

$$u_{y,c} = -0.5\,\rho_x\,n(\dot{\omega} + \dot{\alpha}) \times \cos\left(\lambda + \alpha + \left(\dot{\lambda} + \dot{\alpha}\right)t\right) \tag{13}$$

$$u_{z,c} = -2\,n\,\dot{\alpha}\,\rho \sin\left(\lambda + \alpha + \left(\dot{\lambda} + \dot{\alpha}\right)t\right) + 2\,\rho\,k\,n \sin^2 i_0 \cos\alpha \sin\lambda \tag{14}$$

Where the control is expressed in the radial-transversal-normal frame. This ideal control is taken as the reference control for the feedback control law implemented for the satellite control subsystem.

Moreover, it could be implemented in the ROEs framework to describe the relative dynamics, by properly choosing the control matrix B in the dynamic's equation:

$$\delta\dot{\boldsymbol{\alpha}}_{\boldsymbol{roe}} = \boldsymbol{A}\,\delta\boldsymbol{\alpha}_{\boldsymbol{roe}} + \boldsymbol{B}\,\boldsymbol{u_c} \tag{15}$$

Where $\boldsymbol{A}$ is the dynamical matrix from [11] and $\boldsymbol{B}$ is the control matrix for the ROEs representation:

$$B = \frac{1}{n\,a_0}\begin{bmatrix} 0 & 2 & 0 \\ -2 & 0 & 0 \\ \sin u & 2\cos u & 0 \\ -\cos u & 2\sin u & 0 \\ 0 & 0 & \cos u \\ 0 & 0 & \sin u \end{bmatrix} \tag{16}$$

4.2 Guidance and Control

The guidance and control for the mission extension phase of FFLAS are based on the logic developed for the nominal mission operations, as described in [12], where also the GNSS-based navigation system is presented.

The guidance algorithm implements the ideal trajectory for each satellite, based on the solution of Eq. (8). Moreover, it monitors the real-time inter-satellite distance for the collision risk evaluation and the safe flight of the satellites. In case a non-nominal behaviour arises, the satellites should be able to pass in safe mode even during the mission extension phase.

The control system implements a linear quadratic regulator approach [13] to compute the actual feedback control with respect to the ideal one of Eqs. (12), (13), and (14).

$$\boldsymbol{u} = \boldsymbol{u}_c - \boldsymbol{K}\,(\boldsymbol{x} - \boldsymbol{x}_c) \tag{17}$$

Where $\boldsymbol{u_c}$ and $\boldsymbol{x_c}$ are the reference control and trajectory from the guidance algorithm, while $\boldsymbol{K}$ is the control gain matrix computed from the dynamical description of the relative motion including the J_2 effect.

The closed-loop feedback control logic for the FFLAS satellites is shown in Fig. 7. In this paper, the navigation block is considered ideal, with no external uncertainties due to the state reconstruction. Moreover, in the feedback loop, we implemented the model of the low thrust actuators, considering the parameters from the baseline engine (QinetiQ T5 [14]).

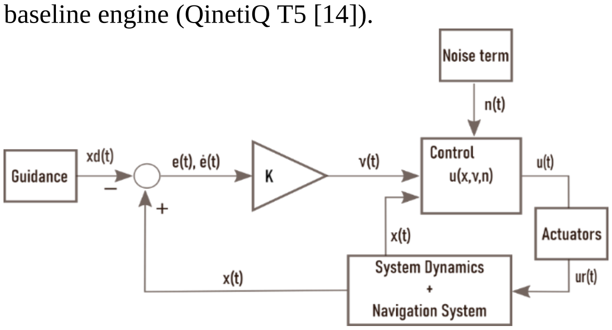

Fig. 7. Guidance and Control system logic for the FFLAS mission study.

4.3 Simulation parameters

This paragraph provides the parameters used in the simulation of the mission extension phase. The nominal geometry configuration of the three-satellite formation is characterised by the state vector reported in Table 4. Moreover, the parameter to initialise the PCO configuration are reported for the three satellites, as well as the mean Keplerian elements of the reference orbit. The PCO configuration is shown in Fig. 8.

Table 4. Simulation parameters for the mission extension.

Parameter	Value
Initial state in Earth Pointing in RTN frame	
Sat A	$[0, \quad 0, \quad -5.3997, \quad 0, \quad 0, \quad 0]$
Sat B	$[0, \quad 6.2350, \quad 5.3997, \quad 0, \quad 0, \quad 0]$
Sat C	$[0, -6.2350, \quad 5.3997, \quad 0, \quad 0, \quad 0]$
Parameter for the PCO orbit	
ρ_x	7.5 m
α_x	$[3\pi/2, \pi/6, 5\pi/6]$
Mean elements of the reference orbit	
Kep_0	$[7.1531 \cdot 1e6, \quad 0, \quad 98.51°, \quad 270°, \quad 0°, \quad 0°]$

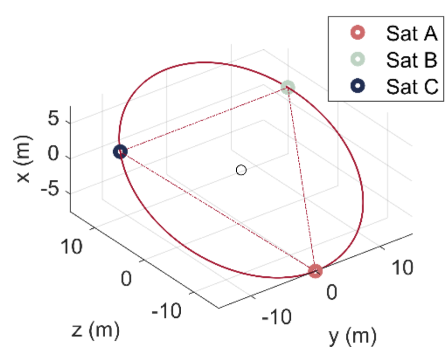

Fig. 8. The three-dimensional distance among the satellites during the formation reconfiguration.

4.4 Simulation results for Strategy 2

This section presents the simulation results for the mission extension phase of the FFLAS study. Initially, section 4.4.1 shows the strategy for the formation reconfiguration from the nominal triangular geometry in Earth pointing mode to the PCO configuration. The time for the manoeuvre was selected to be three-quarters of the orbital period. During the reconfiguration, the real-time inter-satellite distance was monitored, to avoid any unsafe situations. Section 4.4.2 presents the formation maintenance in the PCO configuration for one day of propagation, considering a reduced maximum available thrust level to reduce the fuel request. Finally, section 0 presents the delta-v considerations for the mission extension phase and provides an estimation of the possible extension period depending on the available fuel at the end-of-mission.

4.4.1 Formation reconfiguration

The reconfiguration from Nominal formation geometry in Earth Pointing geometry to the PCO geometry for mission extension is shown in Fig. 9. The delta-v optimal manoeuvre is based on the convex formulation of the problem, considering the limitation in the maximum available thrust from the engines and the minimum distance constraint for collision avoidance [6]. The convex problem is solved via the CVX Matlab® solver with a software package for semidefinite-quadratic-linear programming [15,16].

During the manoeuvre, it is important to monitor the inter-satellite distance among the satellites, which is required to be higher than 10 m. The time evolution during the manoeuvre is shown in Fig. 10, where the three lines indicate the distance among each couple of satellites. Finally, also the delta-v budget for the reconfiguration manoeuvre is presented, in Fig. 11. As the satellites can thrust only in the y and z direction of the body frame, the delta-v in the x_b direction is null. For the three satellites, the delta-v budget is lower than 5 cm/s, which is an acceptable level in comparison with the FFLAS delta-v budget for the nominal mission of 2.3 km/s [3]. In particular, satellite A is the one with the highest delta-v for the manoeuvre.

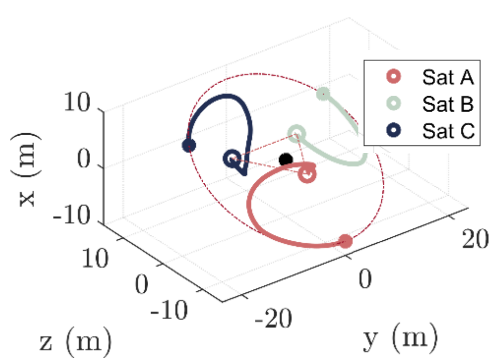

Fig. 9. Formation reconfiguration in low thrust.

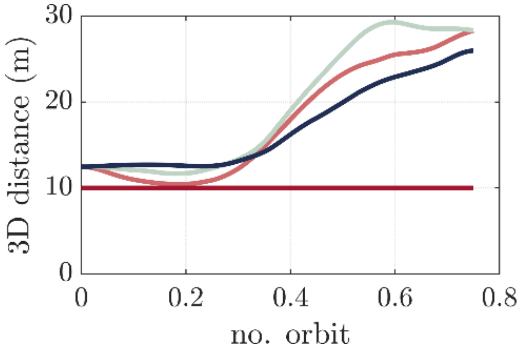

Fig. 10. The three-dimensional distance among the satellites during the formation reconfiguration.

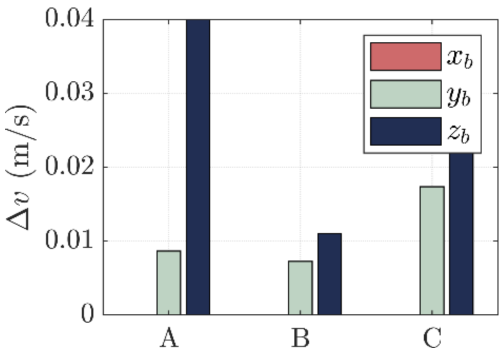

Fig. 11. Delta-v for the formation reconfiguration.

4.4.2 Formation maintenance in PCO configuration

The formation maintenance is designed to have the minimum control effort to follow the ideal PCO trajectory. Moreover, during the whole duration of the simulation, the inter-satellite distance is monitored to detect any possible collision risk.

In this situation, a control term is required to remove the Earth's oblateness disturbance on the PCO. The PCOs are affected by the orbital perturbations, as represented in the top parts of Fig. 12, Fig. 13, and Fig. 14. The Earth's oblateness causes drift and an oscillation around the nominal trajectory, and the magnitude of the oscillation depends on the phasing angle α_x. Consequently, it is more evident for satellites B and C.

At this point, formation maintenance is introduced, to control such drifts and oscillations. This ideal control is taken as the reference control for the feedback control law implemented for the satellite control subsystem. The controlled trajectories in the J_2 perturbed environment is shown in the bottom parts of Fig. 12, Fig. 13, and Fig. 14. Moreover, the monitoring of the distance among the satellites is reported for both the natural evolution and the controlled one in Fig. 15. Even for a non-controlled motion, the satellites remain far apart for 15 orbital periods, without entering the collision area.

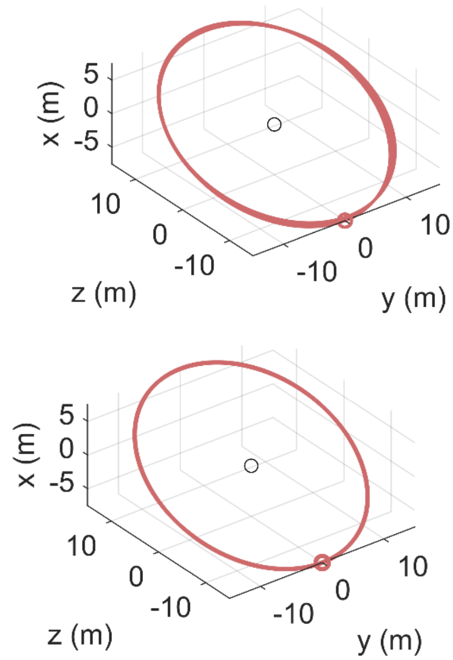

Fig. 12. Natural with J_2 (top) and controlled (bottom) evolution of the PCO for satellite A.

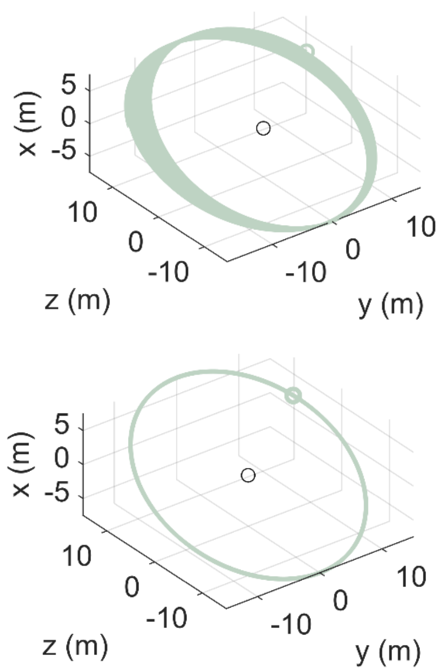

Fig. 13. Natural with J_2 (left) and controlled (right) evolution of the PCO for satellite B.

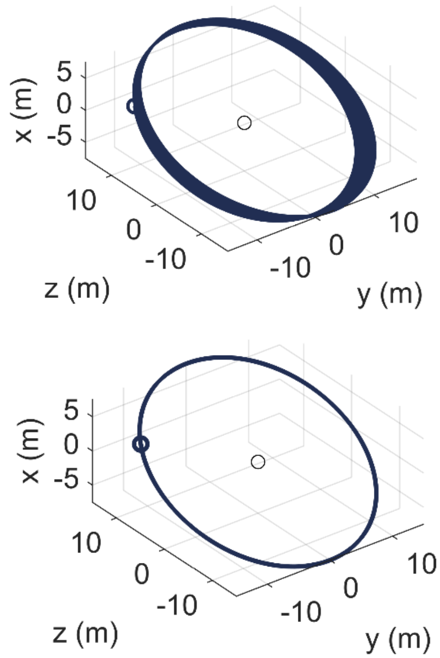

Fig. 14. Natural with J_2 (left) and controlled (right) evolution of the PCO for satellite C.

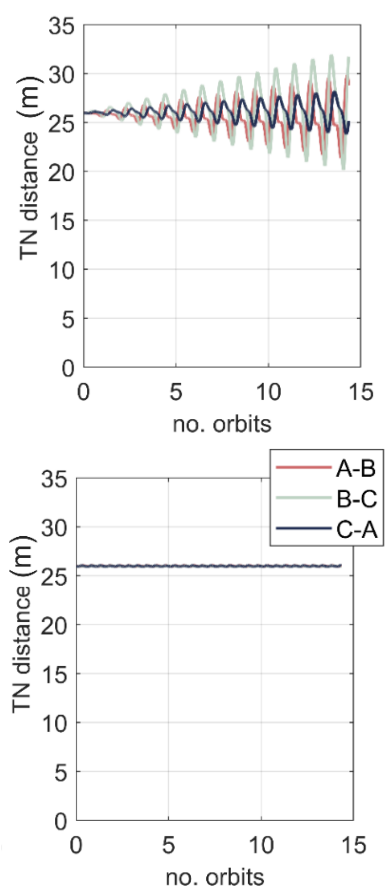

Fig. 15. The two-dimensional distance among the satellites during the formation maintenance.

4.4.3 Delta-v considerations for the Mission Extension Phase

The delta-v budget for the formation maintenance is of primary importance to evaluate the possible extension period for this phase of the mission. The delta-v for the maintenance is about ~ 30 mm/s for one day of simulation for each satellite in the formation.

We analysed the possible extension period based on the percentage of fuel left onboard the satellites. Considering that 1% to 5% of the overall fuel at launch condition is still available at the end-of-life of the nominal mission, different mission extension periods could be identified, as reported in Table 5.

Table 5. Extension period for different fuel conditions at end-of-mission.

% of Available Fuel	Extension period
1%	~ 2 years
2%	~ 4 years
3%	~ 10 years
4%	~ 20 years
5%	~ 25 years

4.4.4 Payload considerations for strategy 2

A consideration of the payload performances should be done. The working principle of the FFLAS payload (an L-band passive interferometer) is related to the correlation between two antenna elements, as a sample of the visibility function [17][5]. The aim of the distribute FFLAS payload is to have an interferometric instrument that samples in an alias freeway the visibility function without any holes. This is possible only with the following assumptions.

- A certain distance among the hexagons is granted (12.47 m nominally).
- The antennas shall lay in the transversal-normal plane.

For this reason, the possibility to perform some interferometry with degraded performance shall be evaluated in a successive study of the mission. In case the interferometry is not feasible in such a configuration, the formation strategy presented in this section is still valid, but it provides a single satellite performance instead of the distributed node observation values.

Nevertheless, this strategy could be of interest for different payload technologies for Earth's observation, which could provide good performances with less stringent requirements.

5. Conclusions

To conclude, this paper presents two possible strategies for the mission extension phase of FFLAS. The approach can be applied to other formation flying space missions for Earth observation. In particular, the first approach provides an effective baseline to extend the science mode as a single satellite mission. In this approach, the payloads will work singularly, but they can still provide an important contribution to scientific knowledge.

In the second approach, a strategy to maintain the close formation among the satellite with a minimum delta-v effort is presented. The aim is to extend the study with the analysis of the payload performances in the new configuration. It is important to evaluate the level of degradation of the combined interferometry, to assess the advantage of selecting such a strategy. The trajectory selected, the PCO, allow for a small delta-v for formation maintenance, granting the possibility to extend the mission for several years even with a small amount of fuel left at the end of the mission.

This study is intended as the final analysis for the FFLAS mission proposal and could serve as a starting point for a successive mission development or for a similar formation study. It is important to consider every aspect during the mission design, including system and payload considerations and

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References

[1] NASA, Extending Science "NASA's Space Science Mission Extensions and the Senior Review Process," National Academies Press, 2016. https://doi.org/10.17226/23624.

[2] C. Colombo, F. Letizia, E.M. Alessi, M. Landgraf, End-of-life Earth re-entry for highly elliptical orbits: The integral mission, Advances in the Astronautical Sciences. 152 (2014) 1771–1791.

[3] F. Scala, G. Gaias, C. Colombo, M. Martìn-Neira, Formation flying l-band aperture synthesis: Design challenges and innovative formation architecture concept, in: Proceedings of the International Astronautical Congress, IAC, 2020.

[4] A.M. Zurita, I. Corbella, M. Martin-Neira, M.A. Plaza, F. Torres, F.J. Benito, Towards a SMOS operational mission: SMOSOps-Hexagonal, IEEE Journal of Selected Topics in Applied Earth Observations and Remote Sensing. 6 (2013) 1769–1780. https://doi.org/10.1109/JSTARS.2013.2265600.

[5] M. Martin-Neira, M. Suess, N. Karafolas, P. Piironen, F. Deborgies, A. Catalan, R. Vilaseca, J. Montero, M. Puertolas, D. Outumuro, I. Corbella, I. Duran, N. Duffo, R. Materni, T. Mengual, M.A. Piqueras, A. Olea, A. Solana, J. Closa, A. Zurita, J.I. Ramirez, O. Breinbjerg, J.M. Bjorstorp, K. Kaslis, S.S. Kristensen, R. Oliva, R. Onrubia, A. Camps, J. Querol, Technology Developments for an Advanced L-Band Radiometer Mission, in: International Geoscience and Remote Sensing Symposium (IGARSS), Institute of Electrical and Electronics Engineers Inc., 2020: pp. 6507–6510. https://doi.org/10.1109/IGARSS39084.2020.9324378.

[6] F. Scala, G. Gaias, C. Colombo, M. Martin-Neira, Three Satellites Formation Flying: Deployment and Formation Acquisition Using Relative Orbital Elements, in: 2020 AAS/AIAA Astrodynamics Specialist Conference, Lake Tahoe, CA, USA, 9-12 Aug., 2020: pp. 1-17 (AAS 20-625).

[7] F. Scala, G. Gaias, C. Colombo, M. Martin-Neira, Design of optimal low-thrust manoeuvres for remote sensing multi-satellite formation flying in low Earth orbit, Advances in Space Reasearch. (2021), https://doi.org/10.1016/j.asr.2021.09.030.

[8] M. D'Errico, G. Fasano, Relative trajectory design, in: Distributed Space Missions for Earth System Monitoring, Springer New York, 2013: pp. 125–162. https://doi.org/10.1007/978-1-4614-4541-8_3.

[9] G.W. Hill, Researches in the Lunar Theory, American Journal of Mathematics. 1 (1878) 5–26.

[10] S. D'Amico, O. Montenbruck, C. Arbinger, H. Fiedler, Formation flying concept for close remote sensing satellites, Advances in the Astronautical Sciences. 120 (2005) 831–848.

[11] G. Gaias, M. Lara, C. Colombo, 18th Australian International Aerospace Congress Accurate Osculating / Mean Orbital Elements Conversions for Spaceborne Formation Flying, (2018) 24–28.

[12] F. Scala, C. Colombo, G. Gaias, M. Martìn-Neira, GNSS-based Navigation for a Remote Sensing Three-Satellite Formation Flying, in: 16th International Conference on Space Operations, Cape Town, South Africa. (SpaceOps-2020,7,X1482) 3-5 May., 2021: pp. 1–18.

https://re.public.polimi.it/handle/11311/1172794 (accessed June 1, 2021).

[13] M.P. Clemente, Optimal Control of Low-Thrust Satellite Formation-Flying Reconfiguration using a LQR, 2020.

[14] C.H. Edwards, N.C. Wallace, C. Tato, P. Van Put, The T5 ion propulsion assembly for drag compensation on GOCE, in: European Space Agency, (Special Publication) ESA SP, 2004: pp. 217–223.

[15] M. Grant, S. Boyd, CVX: Matlab software for disciplined convex programming, version 2.0 beta, (2013). http://cvxr.com/cvx.

[16] M. Grant, S. Boyd, Graph implementations for nonsmooth convex programs, Recent Advances in Learning and Control (a tribute to M. Vidyasagar), in: V. Blondel, S. Boyd, H. Kimuna (Eds.), Springer, 2008: pp. 95–110.

[17] M. Martì-Neira, A. Zurita, M.A. Plaza, J. Benito, SMOS Follow-on Operational Mission Concept (SMOSops-H), in: 2nd SMOS Science Conference, 2015.