POC_ESSAIM: Close-Formation flying demonstration of 3 nanosatellites in LEO

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Abstract – *In recent years, CNES has been involved in the studies of several mission concepts regarding the deployment of multiple satellites to constitute distributed instruments. Two examples are:*

- *ULID: its objective was to demonstrate an Unconnected-L-band Interferometer to measure and moisture ocean salinity. The mission involved 3 satellites and required to maintain a quasi-constant 40m distance between any two satellites.*
- *NOIRE: its objective was to create a low-frequency radio interferometer observatory from space. The concept included up to 50 satellites in a lunar orbit with inter-satellite distances around 10km.*

Due to budget restrictions at the time, both projects were interrupted before phase B.
Fortunately, the multi-satellite demonstration is being reactivated through a low-cost demonstrator focused on the formation-flying technologies to be matured for the future deployment of swarm missions as ULID or NOIRE. This project is called POC_ESSAIM (Proof-Of-Concept). It involves 3 nanosatellites in a LEO orbit and is currently in a phase A study at CNES.

The paper focuses on the challenges of the POC_ESSAIM mission concepts and describes the results of the additional activities conducted to consolidate formation flying strategies. First of all, the mission and the different GNC demonstration scenarios are presented in sections I and II. Then, a focus on two of the critical GNC functionalities is performed: relative navigation (section III) and close-formation flying control (section IV). A short description of the vision-based navigation mission extension is provided (section V). Finally, a conclusion of the current status of the project and the on-coming activities is presented.

I. INTRODUCTION

A. Mission objectives

POC_ESSAIM is a CNES proof-of-concept mission aiming to mature swarm technologies for future missions.

The main technologies targeted in the demonstrations are related to:

- Inter-Satellite Link
- Distributed processing
- Synchronisation
- Operations
- On-board autonomy & Formation flying: relative localisation, close formation acquisition & keeping.

Figure 1 *Main demonstration objectives*

B. Mission description

The mission will consist on three satellites in a LEO orbit. As for ULID mission, the targeted orbit is a 600 km 6h/18h quasi-synchronous orbit but this could change in the on-going phase A due to new end-of-life restrictions that could affect the targeted orbit altitude.

The overall demonstration duration is around 1 year divided in several demonstration phases increasing the on-board autonomy progressively:

- Phase 1: Large formation using GNSS.
- Phase 2: Large formation using inter-satellite distances and angles.
- Phase 3: Close-formation flying
- Possible mission extension: Vision-based navigation demonstration.

These phases and the main GNC functionalities demonstrated in each one will be detailed in chapter II.

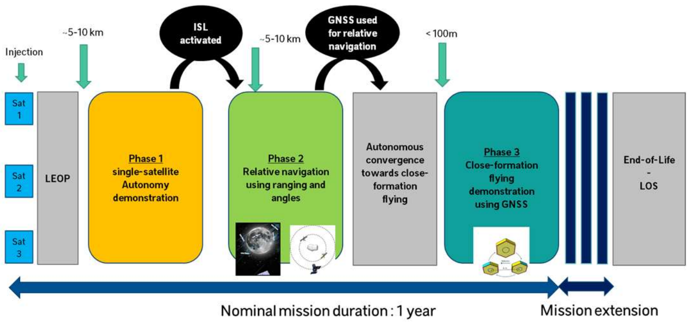

Figure 2 . *Main demonstration phases*

C. Satellites

In order to fulfil the demonstration objectives, the satellite shall carry the following equipment: a GNSS receiver, an electric propulsion system and it shall carry also an ISL(Inter-Satellite-Link) equipment to exchange data with its companions.
The addition of a monocular camera is under analysis; This would improve the relative navigation performances in the large formation phase (adding angles for

navigation) and would allow as well the inclusion of a "vision-based navigation" demonstration in the mission extension phase.

It is expected that in the nominal attitude configuration (station-keeping), the solar panels are parallel to the orbital plane and the Z+ side is Nadir pointed. The thruster is accommodated on the face Y+ which allows performing normal manoeuvres without changing the satellite attitude.

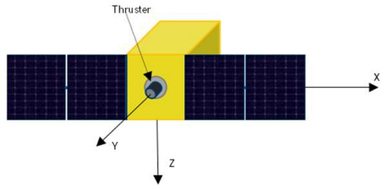

Figure 3 *Satellite Layout*

The satellite sizes are expected to be in the range 8U-12U and their mass is around 20kg.

The electric propulsion system has a low thrusting capacity around 0.4mN. In addition, the propulsion system cannot be used more than 12-19 min per orbit due to the limited power resources. This limitation represents a real constraint in the context of the formation acquisition or reconfiguration for contingency since the magnitude of the necessary manoeuvres is at least several cm/s. Conversely, station-keeping manoeuvres are in the mm/s range and can be easily executed anytime

A detailed analysis of the satellite architecture is to be performed in the on-going phase of the project.

II. GNC Nominal Demonstration Phases

As mentioned before, several demonstrations phases are planned during the 1-year nominal mission increasing the on-board autonomy progressively.

A. Large formation: single-satellite autonomy

After LEOP and commissioning phases, which will be operated from the ground, the first demonstration will target several functionalities focused on single-satellite autonomy.
CNES already has an important background in mono-satellite autonomy and several functionalities are already flight proven. The step forward of this demonstration is to integrate the different CNES functionalities together.

The main GNC functionalities targeted are:
-Absolute State Estimation:
  On-board navigator BOLERO/DIONE based on GNSS measurements.
-Autonomous orbit control (AOC):
  Based on the ASTERIA functionality [4] who has been integrated and tested in OPS-SAT mission from ESA. This functionality also includes autonomous Collision avoidance management.
-Autonomous mission planning:
  The mission planning is adapted on-board taking into account the current energy status and the priority tasks, as for example collision avoidance.

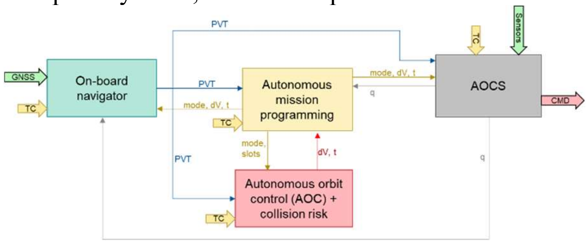

Figure 4 *mono-satellite GNC* architecture

During this phase, the satellites will be placed in the same orbit but with an orbit argument of latitude separation in the 5-10km range. The satellites will be controlled in their control boxes with respect to a reference orbit. The reference orbit may evolve during the demonstration phase due to ground commanding or to collision avoidance manoeuvers.

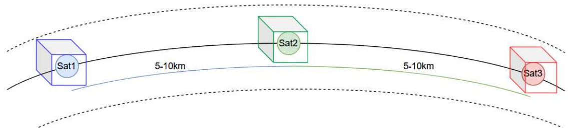

Figure 5 *Large formation configuration (phase 1)*

During this phase the inter-satellite link will be activated and the commissioning of this key element will be performed. The relative navigation using GNSS measurements shared via the ISL will be also commissioned during this phase.

B. Large formation: relative navigation using inter-satellite distances and angles

In this phase, the satellite relative distances remain around 5-10km. The main objective of this phase is to demonstrate relative navigation techniques using inter-satellite distances and angles.
The inter-satellite link is activated and is used to perform synchronisation/ranging. Then, the derived distance information is provided to the navigation filter. Moreover, if a camera is present on board, line-of-sight measurements will be as well provided to the navigation filter to improve the relative configuration observability.

The satellite configuration is close to the one of the previous phase, some relative differences in eccentricity and inclination parameters are to be added in order to improve the observability, as presented in Figure 6

Figure 6 *Large formation configuration (phase 2)*

The main targeted GNC functionalities are:

- Relative State Estimation:
  On-board navigator BOLERO/DIONE based on relative distances and/or angles measurements.
- Object Detection and Tracking:
  GYSELE (GYro-Stellar Enhanced Localization Eyeing in Space) is a gyro-stellar estimation functionality developed by CNES, it allows to accurately estimate the satellite attitude and also to identify and track non-stellar objects providing information about its direction. [6]

In addition to the functionalities described above, the ones already validated in the first phase will still be activated in the GNC loop, as presented in Figure 7.

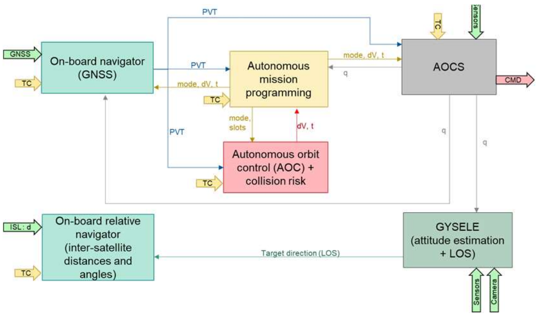

Figure 7 Phase 2 GNC architecture

C. Close formation flying

In this demonstration phase, the satellite configuration is based on the one selected and already studied at CNES for ULID mission. In this mission, in order to reach the expected resolution of the L-band interferometer, the optimal instrument configuration was obtained when the positions of the 3 satellites projected on the TN plane (plane perpendicular to the radial axis) formed a perfect equilateral triangle that rotates at the orbital period.
The distances between the satellites shall remain close to 40m. For station-keeping control, the size of the relative control windows is in the meter-level range.

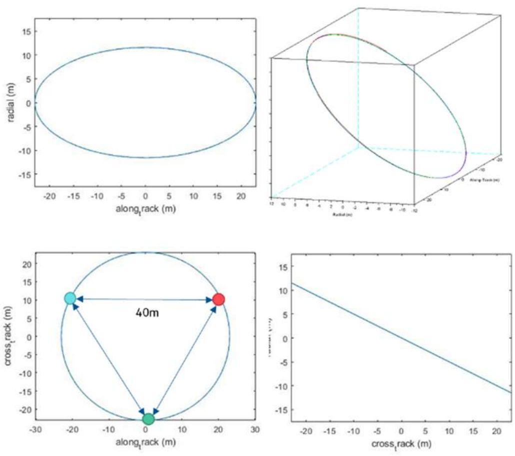

Figure 8 Close formation-flying configuration

Such a configuration is achievable by special settings of the relative orbital elements; this will be detailed in section IV.
The main targeted GNC functionalities are:

- Relative State Estimation:
  On-board navigator BOLERO/DIONE based on GNSS measurements shared through the Inter-Satellite-Link.
- Formation acquisition and keeping:
  This guidance module is built upon the experience gained during the CNES participation to the PRISMA mission and inherits some of the developments.

Given the close formation distances (<50m), the GNC modules have to take special care of the collision risks between the different satellites of the formation.

The GNC functionalities are presented in Figure 9.

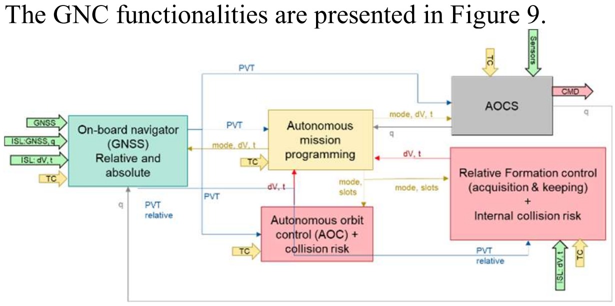

Figure 9 Close-formation GNC architecture

III. RELATIVE NAVIGATION STRATEGY

The relative navigation functionality is based on the flight proven library BOLERO/DIONE, this library has been developed and used by CNES for the past 20 years. BOLERO has been integrated in Thales Alenia Space Topstar GPS receivers, which has already been integrated in many satellite missions. It is also integrated in the Syrlinks N-SPHERE GNSS receiver-navigator that is also flight proven.

BOLERO was initially developed for mono-satellite use, being able to process all GNSS constellation measurements together. Recent missions, as ULID, NOIRE or HERA pushed the navigator to evolve. Due to the relative navigation needs, BOLERO was adapted in order to manage relative navigation and to use new sources of measurements including inter-satellite distances or line-of-sight (LOS) measurements.
The following sections present these evolutions and the expected performances for the POC_ESSAIM scenarios.

A. Relative navigation using GNSS measurements

During Phase-A of the ULID mission, this library was adapted to a multi-satellite configuration.

In the current version of the on-board navigator all the satellite estimated parameters (absolute and relative) are grouped in the same state vector, this is represented in Figure 10.

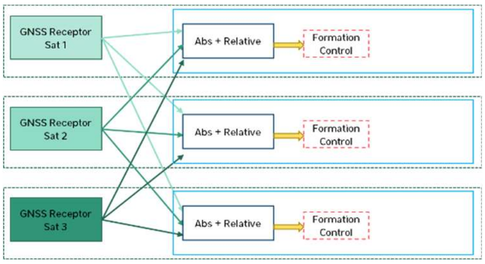

Figure 10 Relative navigator architecture

Since the satellites are close to each other, the common biases (ionospheric perturbations, pole motion, emitter uncertainty on clock and position, etc..) can be perfectly cancelled out through the computation of the relative configuration.

Relative positioning accuracy in the centimetre range can thus be achieved. This was demonstrated simulation, as presented in Table 1, where an accuracy of relative position of about 5cm 3D RMS is reached after a few orbits.

Table 1 : Relative position estimation Error (RMS)

	Radial	Along Track	Normal	3D
Position (mm)	15	45	4	47
Velocity (mm/s)	0.05	0.01	0.005	0.05

For the POC_ESSAIM, the objective is to improve even more the navigator by including single difference measurement of GNSS pseudo range and phase. This combination shall remove the major part of GNSS emitter information uncertainty (position and clock bias), as well as the ionospheric perturbation, as the two receivers are not too far from each other. Thus, it will enable a far better accuracy on relative positioning. This has been already demonstrated on ground using real data for relative orbit estimation (post –processing).

B. Relative navigation using inter-satellites distance and angles measurements

The multi-satellite version of BOLERO was recently adapted to use new measurements in the navigation filter as pseudo-range information generated by the ISL ranging capabilities or angle measurements provided by a camera.
After validation of the integration of the different measurements type, a performance analysis has been done. This analysis targets two situations: when only pseudo-ranges are available and when the pseudo-range information is completed with angle measurements.

The scenario analysed contains only two satellites in a configuration similar to the large formation scenario where intersatellite distances are about 5km. Additionally small eccentricity and inclination variations have been added, this helps to increase the observability of the relative motion.
Similar to ULID mission, the pseudo-range measurements frequency is expected to be close to 1/30Hz. Regarding the angles measurements, two cases have been analysed, one containing an angle measurement every 30s and another one every 5min. The objective was to perform a preliminary analysis of the benefits of adding angular measurements in terms of performances and converging speed.

The following Table 2, summarizes the results of this analysis:

Table 2 : Phase2-Relative position estimation error

	ISL		ISL + OPT (30s)		ISL + OPT (5min)	
	mean	std	mean	std	mean	std
Convergence duration	R/T: 2.5h N: 56min		R/T: 1.75h N: 7min		-	
Radial	0.0	0.18	0.0	0.13	0.01	0.18
Tangential	-0.27	0.45	0.08	0.28	0.02	0.37
Normal	-0.38	4.38	0.00	0.19	0.01	0.7

As expected, the best performances are obtained when the optical measurements (angles) are added at a high frequency. Nevertheless, even for the low frequency the addition of angular measurements allows a reduction of the relative error, mainly along the normal axis.

This analysis will be completed during the on-going phases of the project in order to target more complex scenarios, as for example:

- Reduce the frequency of the angles measurements.
- Analyse different satellite configuration with more complex observability configurations.
- Scenarios design where the relative motion dynamics is poorly known.
- Analyse scenarios including angles-only navigation.

IV. CLOSE SATELLITE FORMATION ACQUISITION AND CONTROL STRATEGY

The close formation control strategy is based on the one already analysed during ULID mission and presented in [1] and [2].

The following sections describe the Relative Orbit Parameters representation selected for the control strategy and the formation configuration. Then, the main control strategies are described and results of simulations are presented.
For the simulations, errors in the relative state estimation and manoeuvring performances are taken into account, as presented in Table 3. It has to be noted that the thrust dispersions (mainly on the direction) represent a major control challenge, as it creates a strong coupling between axes.

Table 3 : Simulation errors (3-sigma)

	Value	units
3D Navigation errors	30	cm
Thrust norm error	15	%
Thrust direction error	5	deg

A. Relative orbit elements (ROE)

As proposed in [1], the relative orbit elements representation selected is based on relative eccentricity and inclination vectors and they are defined as follows:

$\delta a = (a_t - a)/a \qquad \delta\lambda = (u_t - u) + (\Omega_t - \Omega)\cos i$

$\delta e_x = (e_t.\cos\omega_t - e.\cos\omega) \qquad \delta e_y = (e_t.\sin\omega_t - e.\sin\omega$

$\delta i_x = (i_t - i) \qquad \delta i_y = (\Omega_t - \Omega)\sin i$

a,e, i, ω, Ω, and M denote the classical Keplerian elements. u represents the mean argument of latitude ($u = \omega + M$). The subscript t refers to the target/anchor point.

The relative motion is described in the RTN orbital frame centred on the target/anchor point. The linearized Hill-Clohessy-Wiltshire (HCW) equations can be used:

$$\begin{Bmatrix} \delta r_R/a \\ \delta r_T/a \\ \delta r_N/a \\ \delta v_R/v \\ \delta v_T/v \\ \delta v_N/v \end{Bmatrix} = \begin{Bmatrix} \delta a & -\delta e\cos(u-\varphi) \\ \delta\lambda - \frac{3}{2}\delta a(u-u_0) & +2\delta e\sin(u-\varphi) \\ & +\delta i\sin(u-\vartheta) \\ & +\delta e\sin(u-\varphi) \\ -\frac{3}{2}\delta a & +2\delta e\cos(u-\varphi) \\ & +\delta i\cos(u-\vartheta) \end{Bmatrix} \quad (1)$$

The projections of the relative trajectory with respect to the anchor point is presented in Figure 11.

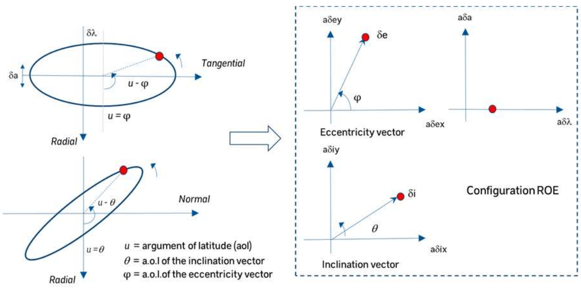

Figure 11 : Relative motion representation Δe / Δi

The following notation is then used:

$$\Delta e = \delta e\left[\begin{pmatrix}\cos\varphi\\ \sin\varphi\end{pmatrix}\right] = \left[\begin{pmatrix}\delta ex\\ \delta ey\end{pmatrix}\right] \qquad \Delta i = \delta i\left[\begin{pmatrix}\cos\vartheta\\ \sin\vartheta\end{pmatrix}\right] = \left[\begin{pmatrix}\delta ix\\ \delta iy\end{pmatrix}\right]$$

This representation is commonly used in formation-flying and rendezvous missions, as presented in [1], [6], [7] and [8].

B. Formation configuration and anchor point selection

As presented in section II.C, the target configuration is chosen such that the relative trajectories of the three satellites projected on a plane perpendicular to the radial axis describe a circle of radius, $R \approx 23.1$. In order to achieve this configuration, the ROE of each satellite with respect to a central point shall be:

$$\delta a = 0 \qquad \delta\lambda = 0$$

$$\Delta e = \left(\frac{R}{2.a}, \varphi\right) \qquad \Delta i = \left(\frac{R}{a}, \vartheta\right) \qquad (2)$$

where:

- ϑ: argument of latitude of the inclination vector
- φ: argument of latitude of the eccentricity vector

Moreover, to the following relation between ϑ and φ shall be maintained: $\vartheta = \varphi + \pi/2$ or $= \varphi + 3\pi/2$.

The anchor point of the formation can be a virtual centred point (centre of the projected circle) or one of the satellites of the formation. As presented in [1], the control strategy based on a central anchor is not optimal from a performance point of view. Therefore, the control strategy selected is the one in which the anchor point corresponds to one of the satellites of the formation, the Δe / Δi vectors configuration for both control strategies are represented in Figure 12:

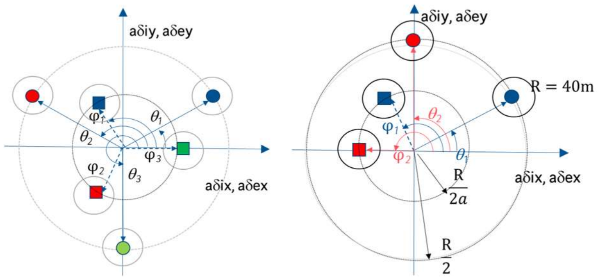

Figure 12 : Anchor point strategy : centred (right) and satellite of the formation (left)

The main perturbation affecting the relative natural motion described in equation (2) is the perturbation due to Earth's oblateness (J2 effect). The main effects of J2 in the ROE are:

- Δe: evolves along a circle (no module change).
- Δi: evolves linearly as a function of time (the slope is proportional to inclination difference δi_x).
- δλ have a secular variation that is proportional to the inclination difference.
- δa is not affected.

For the targeted formation, the order of magnitude of the formation deformation per day due to the drift of the different ROE are:

Table 4 : Formation deformation per day due to J2

	Δe	Δi	δλ
Deformation (m)	0.6	3	2

These effects shall be compensated by the control system to keep the desired formation configuration.

C. Passive safety

It is well established that the inclination/eccentricity vectors constitute a good representation of the passive safety of two satellites flying in close formation.

Safety is maximized when the vectors are parallel/antiparallel (the separation is maximal when one satellite crosses the orbital plane of the anchor) whereas the collision risk reaches its maximum when they are orthogonal. For a demonstration purpose, this is the initial configuration targeted for POC_ESSAIM mission. In order to ensure a minimum distance separation in case of drift, a phase angle can be added modifying the separation between inclination and eccentricity angles ($\Delta\beta = |\varphi - \vartheta| - \pi/2$). The minimum distance is given by the following equation, when $\delta a = 0$:

$$\left(\frac{d_{min}}{a}\right)^2 \geq \frac{2(\Delta e\,.\Delta i)^2}{(\delta e^2+\delta^{\ 2}+|\Delta e+\Delta i|.|\Delta e-\Delta i|} \quad (3)$$

This guarantees a minimum distance in case of drift but will distort the nominal formation configuration as presented in Table 5.

Table 5 : Passive safety

Δβ (°)	Min drift separation (m)	Deformation (m)
0	0	0
10	3.4	+-3.5
20	6.2	+-6.9
30	9.2	+-10.4

D. Formation flying maintenance

The formation maintenance shall mitigate the effects of the major orbital perturbations that degrade the shape formation. Given the relative targeted configuration, the drift of the relative eccentricity vector is smaller than the inclination vector. The control approach for formation the maintenance is therefore the following:

- <u>Eccentricity vector (Δe):</u>
  The eccentricity vector phase is not controlled; it will drift naturally. Only the eccentricity vector magnitude needs to be corrected in the long run to cancel the cumulative effect of the propulsion errors. Some phase corrections are needed in the long run as well to correct the relative drift of the eccentricity vector between the two active satellites.
- <u>Inclination vector (Δi):</u>
  The inclination vector phase is controlled with respect to the eccentricity vector to maintain the targeted Δe/ Δi phase. This correction is conveniently achieved using a single manoeuvre along the normal axis which magnitude and phase is computed using the equations (4), (5) and as defined in Figure 13:

$$\Delta V_N = n\left\|a\delta i^{cur} - a\delta i^{ref}\right\| \quad (4)$$

$$u_m = \tan^{-1}\left((a\delta i_y^{cur} - a\delta i_y^{ref})/(a\delta i_x^{cur} - a\delta i_x^{ref})\right) \quad (5)$$

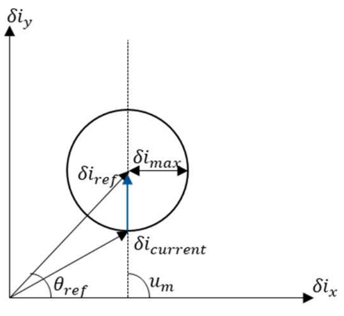

Figure 13 : Inclination control strategy

- <u>Along Track relative position (δλ):</u>
  To control the relative along-track position, a single manoeuvre along the tangential axis is needed (drift control).
  To reduce the drag perturbing effect while mitigating the impact of the propulsion errors, it is proposed to tilt the thruster direction towards the tangential axis by a certain angle while keeping it in the TN plane as illustrated on Figure 14 and detailed in [1].

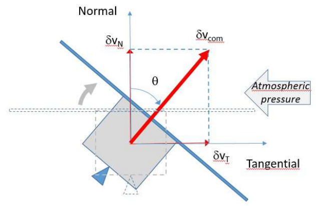

Figure 14 : Tangential control strategy

The control strategy implements a coordination between the along track and the inclination manoeuvres, using coupled manoeuvres in order to optimize positioning accuracy and fuel consumption. This allows use a manoeuvre of one type to reduce as well the error on the other component.

As the target satellite is not manoeuvring, periodic reconfigurations are needed in order to change the anchor satellite and balance the propellant budget. This will be commanded periodically from the ground.

<u>Station Keeping performance analysis:</u>

The analysis is based on a Monte Carlo campaign with 100 simulation runs. The scenario duration is 5 days.

The mean/max values of the along track and inclination errors are presented as well as the overall consumption for the 3 satellites and the total number of manoeuvres.

Table 6 : Overall formation keeping performances statistics for all the scenarios

	Mean	Std Dev (1σ)
DV Budget for all 3 satellites (cm/s)	6.41	0.71
Mean relative along track error (m)	0.03	0.33
Mean relative inclination error (°)	1.22	0.38
Maximum relative along track error (m)	4.67	1.29
Maximum relative inclination error (°)	3.7	1.4
Number of inclination manoeuvres (sum over all satellites)	19.7	1.47
Number of drift manoeuvres (sum aver all satellites)	16.4	1.9

Taking into account that the simulation duration is 5 days, this gives a mean delta-V per satellite of 0.42cm/s per day, which it is well inside the allocated envelope (~1cm/s per day).

An example of control behaviour for one of the satellites during one of the simulations is presented in Figure 15. The first plot describes the along-track error, the second one the error on the inclination vector phase (ϑ). Then, the third plot the error on the inclination and eccentricity vector modules. Finally, the last plot describe which is the current anchor satellite ID.

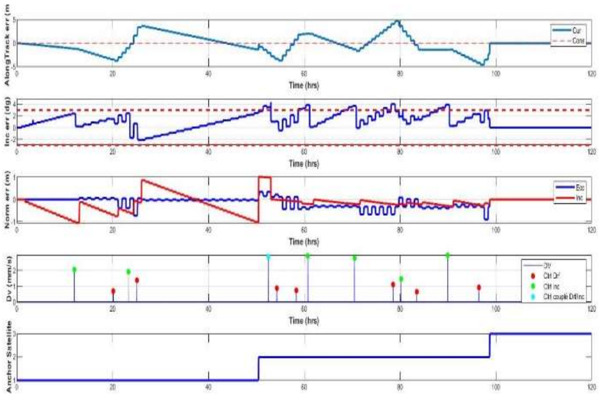

Figure 15 : Control performances for sat #3

Several anchor point changes are applied during simulation period as displayed on the bottom plot and this has no major impact on the station keeping performances. The relative orbits between sat #3 and sat #1 during this simulation is presented in
Figure 16:

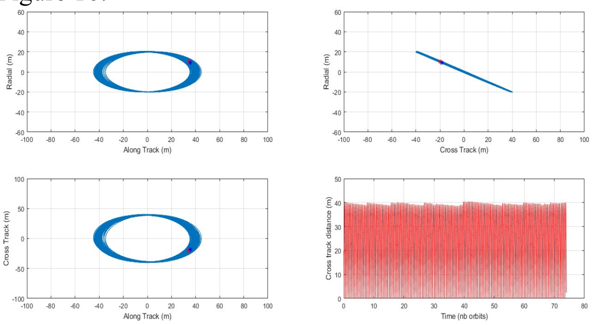

Figure 16 : Relative movement (sat #3 - sat #1)

E. Formation flying acquisition/reconfiguration

In order to change the formation configuration, a different control strategy is needed to include as well eccentricity phase control. This control strategy will be used to initially acquire the nominal formation or in case of anomaly.

The control approach for the formation acquisition and reconfiguration is the following:

- <u>Inclination vector:</u>
  This control differs from what is performed for formation keeping since it considers a given reference inclination vector. Equations (4) and (5) are still valid and the only difference resides in the inclination reference vector computation.
- <u>Eccentricity vector:</u>
  Two strategies are currently implemented to control the eccentricity vector; they both involve three in-plane manoeuvres. Option A applies tangential manoeuvres whereas Option B applies radial manoeuvres.
  The needed delta-V is computed using the equations (6), (7) and (8):

Tangential delta-V (6)	Radial delta-V (7)
$\Delta V_T = na\|\delta e^{cur} - \delta e^{ref}\|$	$\Delta V_R = 2na\|\delta e^{cur} - \delta e^{ref}\|$

$$u_m = \tan^{-1}\left((a\delta e_y^{cur} - a\delta e_y^{ref})/(a\delta e_x^{cur} - a\delta e_x^{ref})\right) \quad (8)$$

Then, the computed delta-V is divided three manoeuvres, as presented in equation (9) and Figure 17. Subscript ΔV_x refers to the ΔV_T or ΔV_R manoeuvres:

$$2.\Delta V_{x1} = 2.\Delta V_{x3} = -\Delta V_{x2}$$
$$u_1 = u_m;\ u_2 = u_m + \pi\ ; u_3 = u_m + 2\pi \quad (9)$$

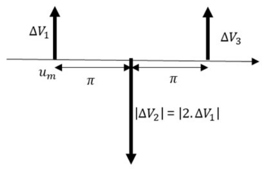

Figure 17 : Eccentricity : three manoeuvres strategy

Applying three manoeuvres allows to reduce each manoeuvre magnitude and to better control the along track direction.

The selection between option A or B, will depend on the satellite final architecture and its capabilities (to be analysed in the ongoing phase). Option B will be more fuel-consuming but it is preferred from a safety point of view as it does not produce any along-track drift in case of manoeuvre execution error.
The current algorithm will be improved during the ongoing phase in order to optimise the possibilities of control of the along-track axis.

- <u>Along Track position:</u>
  The relative along track position is controlled by a purely tangential correction (not tilted). This allows acting on the drift with only one manoeuvre, for example, to correct the errors due to an inclination manoeuvre. The manoeuvre argument of latitude is computed so as to reduce the eccentricity error using equation (8).

In order to guide the ROE to their final values an exponential decay law is implemented, as proposed in [2]. This smooths the relative motion at the final steps of the acquisition where the distances are smaller, improving the robustness of the control strategy.
The interpolation law is defined as follows:

$$X(t) = X(t0) - \left(X(tf) - X(t0)\right)\frac{1 - e^{\frac{t-t0}{\tau}}}{1 - e^{\frac{tf-t0}{\tau}}} \quad (10)$$

Where $X(t)$ in one particular ROE ad τ and tf can be parametrized.
The size of the control boxes is also adapted in order to allow higher deformations for larger satellite inter distances.
<u>Acquisition Performance analysis:</u>

The final part of the acquisition is analysed using Monte Carlo simulations.
The initial relative conditions and the intermediate/final commanded states are defined in the table below. For all cases, $\delta a = 0$:

Table 7 : Acquisition configuration

Sat	Initial	Intermediate	Final state
#1	Anchor satellite		
#2	$\delta\lambda = -200m$ $R = 200m$ $\Delta e = (\frac{R}{2a}, \frac{pi}{3})$ $\Delta i = (\frac{R}{a}, \frac{pi}{3} + pi)$	$\delta\lambda = \mathbf{0}$ $R = \mathbf{40m}$ $\Delta e = (\frac{R}{a}, \frac{pi}{3})$ $\Delta i = (\frac{R}{a}, \frac{pi}{3} + pi)$	$\delta\lambda = 0$ $R = 40m$ $\Delta e = (\frac{R}{2a}, \frac{pi}{3})$ $\Delta i = (\frac{R}{a}, \frac{pi}{3} + \boldsymbol{pi/2})$
#3	$\delta\lambda = 200m$ $R = 200m$ $\Delta e = (\frac{R}{2a}, \frac{2pi}{3})$ $\Delta i = (\frac{R}{a}, \frac{2pi}{3} + pi)$	$\delta\lambda = 0$ $R = \mathbf{40m}$ $\Delta e = (\frac{R}{a}, \frac{2pi}{3})$ $\Delta i = (\frac{R}{a}, \frac{2pi}{3} + pi)$	$\delta\lambda = 0$ $R = 40m$ $\Delta e = (\frac{R}{2a}, \frac{2pi}{3})$ $\Delta i = (\frac{R}{a}, \frac{2pi}{3} + \boldsymbol{pi/2})$

The introduction of an intermediate state allows for a better control the $\Delta e/\Delta i$ phase and ensures passive safety up to the final stages of the acquisition. Moreover, the final manoeuvres will be mainly inclination corrections that have less effect on the along track separation. An example of the commanded relative motion of the $\Delta e/\Delta i$ vectors is presented in Figure 18:

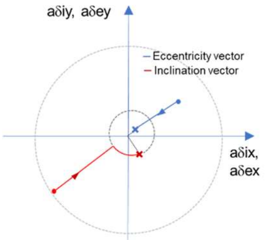

Figure 18 : Relative $\Delta e/\Delta i$ commanded during acquisition

A Monte Carlo analysis has been performed involving 80 cases, with application of radial manoeuvres to control the eccentricity and a duration of 7 days, assuring the convergence towards the final configuration.
Table 8 presents the main performance parameters that consist in minimum distance and the overall delta-V.

Table 8 : Formation acquisition performances statistics

	Mean	Std Dev (1σ)
DV Budget per satellites (m/s)	0.42	0.015
Minimal distance between satellite (m)	34.4	2.8

More detailed analysis will be performed during the on-going phase adding as well more representative perturbation models.

An example of control behaviour for one of the satellites during one of the simulation is presented in

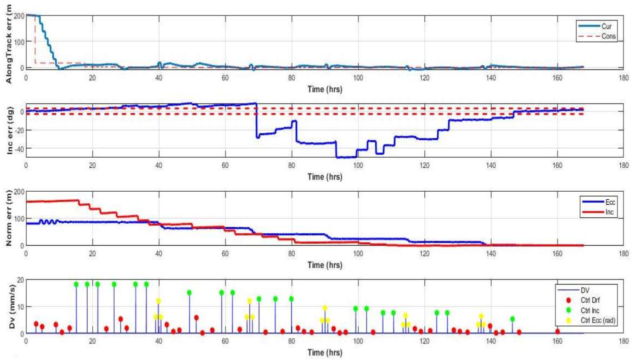

Figure 19 : Control performances for sat #3

The relative orbits between sat #3 and sat #1 are presented in Figure 20:

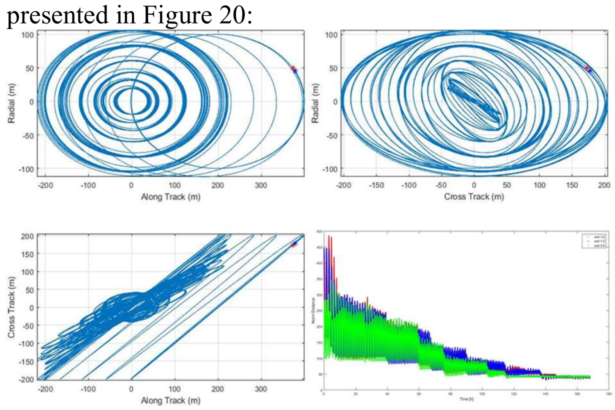

Figure 20 : Relative motion (sat #3 - sat #1)

Reconfiguration Performance analysis:
The Formation Control FDIR will monitor the healthy status of the different satellites. If an anomaly occurs on one of the satellites of the formation (communication loss or anomalous behaviour), it will be set as anchor by the FDIR system. Then, the two active satellites will increase their relative distances and their passive safety with respect to the unhealthy satellite. Once this safe configuration is achieved, the two operating satellites can enter a formation keeping phase while monitoring the state of the problematic satellite until its recovery or until a new mission configuration is commanded by the ground.

A Monte Carlo analysis has been performed including 80 cases. The initial state of the scenario is the nominal formation configuration and the final state will ensure a $\Delta e/\Delta i$ separation of $\pi/4$ and a $R = 80m$.
Table 9 presents the main performance parameters that consist again in the minimum inter satellite distance and the overall delta-V budget.

Table 9: Formation reconfiguration performances

	Mean	Std Dev (1σ)
DV Budget per satellites (m/s)	0.14	0.008
Minimal distance between satellite (m)	36.5	3.6

During the on-going phase, new analysis will be performed including to adapt the reconfiguration strategy to more specific failure scenarios.

The relative orbits between the sat #1 and sat #3 (anomaly) are presented in Figure 21:

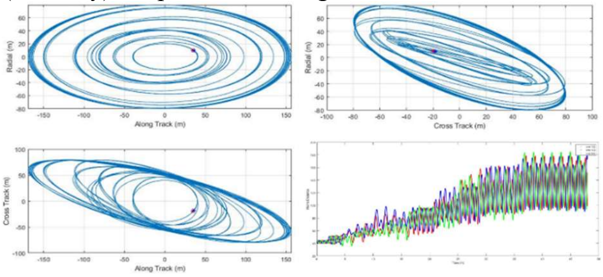

Figure 21 : Relative movement (sat #1- sat #3)

In this scenario, the desired safe configuration corresponds to minimum inter-satellite distances in the 100m range, while ensuring a passive safety distance of 45m.

V. VISION-BASED NAVIGATION EXTENSION

This section presents the mission extension to demonstrate vision-based navigation capabilities in satellite proximity operations.

During the past years, the interest of the space community in space awareness and in-orbit servicing has increased. Vision-based navigation is one of the key elements of these activities. The objective is to benefit from the close-formation flying capabilities of the POC_ESSAIM to mature critical elements of a vision-based navigation system as for example the pose estimation algorithms.

Different technologies will be integrated progressively so as to increase the on-board autonomy capabilities during the different demonstration phases while the relative GNSS navigation will be used in the formation FDIR system to ensure safety.

At first, the satellites will be put in different passive-safe configurations using GNSS-based navigation in order to generate a satellite image data base with different illumination conditions and relative distances.
Then, it is envisioned to test various vision based pose estimation algorithms in open loop mode.
After validation and verification of the real pose estimation performances, it is envisioned to integrate the VBN modules in the GNC loop. Then, it is foreseen to perform different autonomous approaches depending on the overall system performance.

In addition, the 3 satellite configuration will allow to exercise some cooperative localization techniques based on the observation of the same object by two satellites sharing their LOS measurements. These techniques will be evaluated on different scenarios involving the detection and localization of passive targets at long range.

VI. CONCLUSIONS & PERSPECTIVES

This paper has described the main characteristics of the POC_ESSAIM mission in terms of orbit and formation control, its demonstration phases and the main GNC functionalities targeted.

Then, a focus has been done on two of the critical elements, the relative state estimation and the formation acquisition and keeping. The performances of these functionalities have been analysed and no show stoppers have been identified.

Nevertheless, POC_ESSAIM is a challenging mission and critical aspects remain to be analysed during the on-going project phase A, as for example the management of collision risks during the close formation, with debris or other satellites.

Some of the main objectives of the on-going phase A are:

- Detail the CONOPS of different demonstration phases (system analysis).
- Define the satellite architecture.
- Finalise the feasibility and preliminary performances analysis of the main GNC functionalities.

The current phase A will last until beginning of 2025. Next, the possible continuation of the project to a BCDE phase will be analysed by CNES for programmatic decision. The target launch date is planned mid-2027.

VII. REFERENCES

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