DESIGN AND ANALYSIS OF A FORMATION FLYING SYSTEM FOR THE CROSS-SCALE MISSION CONCEPT

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ABSTRACT

The ESA-funded "Cross-Scale Technology Reference Study" has been carried out with the primary aim to identify and analyse a mission concept for the investigation of fundamental space plasma processes that involve dynamical non-linear coupling across multiple length scales. To fulfil this scientific mission goal, a constellation of spacecraft is required, flying in loose formations around the Earth and sampling three characteristic plasma scale distances simultaneously, with at least two satellites per scale: electron kinetic (~10 km), ion kinetic (~100-2000 km), magnetospheric fluid (~3000-15000 km).

The key Cross-Scale mission drivers identified are the number of S/C, the space segment configuration, the reference orbit design, the transfer and deployment strategy, the inter-satellite localization and synchronization process and the mission operations.

This paper presents a comprehensive overview of the mission design and analysis for the Cross-Scale concept and outlines a technically feasible mission architecture for a multi-dimensional investigation of space plasma phenomena. The main effort has been devoted to apply a thorough mission-level trade-off approach and to accomplish an exhaustive analysis, so as to allow the characterization of a wide range of mission requirements and design solutions.

1 INTRODUCTION

The Technology Reference Studies (TRS)[1] have been introduced in the European Space Agency (ESA) Science Directorate to focus on the development of strategically important technologies that are of likely relevance to potential future science missions. The wide spectrum of possible scientific mission scenarios has brought to the fore the need for an assessment of several technologically demanding and scientifically meaningful mission concepts. To address this emerging need, the "**Cross-Scale Technology Reference Study (CS TRS)**" (carried out by DEIMOS Space, Thales-Alenia Space and ONERA in the frame of an ESA contract) is intended to identify the enabling technologies to embark on a demanding plasma physics mission that would improve to a large extent the knowledge of the space plasma processes in the vicinity of the Earth at different relevant spatial and temporal scales.

The three universally dominating fundamental space plasma processes are shocks, reconnection and turbulence (Figure 1). The CS S/C shall visit the relevant regions in near-Earth space where the scientifically most interesting plasma processes occur, i.e. bow shock, magnetosheath, magnetopause and tail current sheet.
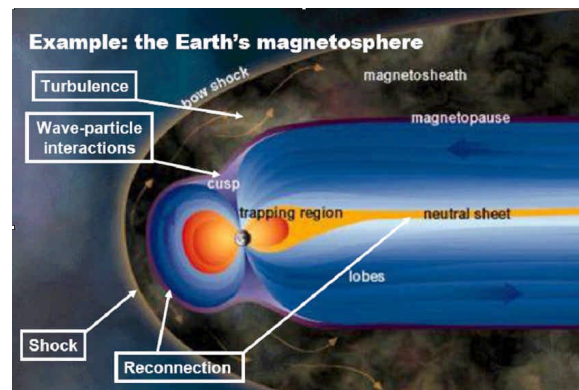

Figure 1: Plasma Processes of Interest and their Locations in Space (courtesy of ESA)

The objective of the CS study is thus to establish a feasible mission profile for a cost-efficient investigation of fundamental space plasma processes that involve coupling across multiple length scales. To fulfil this scientific mission goal, a **constellation of spacecraft** is required, **flying in loose formations** around the Earth and sampling three characteristic plasma scale distances simultaneously, with at least two satellites per scale: **electron kinetic (~10 km), ion kinetic (~100-2000 km) and magnetospheric fluid (~3000-15000 km)**. The multi-satellite measurements generated are sensitive to scales of the order of the S/C separation.

Based on the above considerations, the CS mission concept addresses a space segment including between 8 and 12 S/C relatively located such that instantaneous measurements at three different scales can be obtained. In particular, a constellation configuration

with **10 S/C** has been designed and analysed in details: this configuration comprises a **mother-daughter system on the small (electron) scale located in the centre of two nested tetrahedrons on the medium (ion) and large (fluid) scales** (Figure 2). This multi-satellite system shall be built-up and maintained with respect to a **reference mission orbit**.
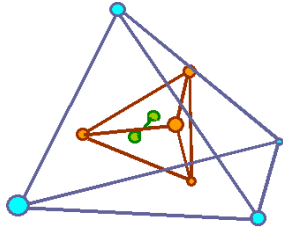

Figure 2: Cross-Scale Constellation Configuration based on 10 S/C on Small Scale, Medium Scale and Large Scale

For cost-efficiency, simple identical spinning S/C have been baselined (apart from slight differences in the dedicated scientific payload and in the inter-S/C synchronisation and ranging equipment for each S/C subset) with a platform dry mass of ~100 kg, which can accommodate 10–40 kg of plasma instruments.

The objective of this paper is to provide a comprehensive overview of the mission design and analysis for the CS TRS. Starting with the reference orbit design trade-offs and the achievable number of S/C, this paper encompasses the transfer and deployment strategy, the formation design, maintenance and reconfigurations, the inter-satellite localization and synchronization process, and the mission operations for the proposed mission concept.

2 DRIVING MISSION REQUIREMENTS

The key requirement related to the CS scientific objectives is that the orbit apogee shall visit the "tailbox" at least once a year during the 3-year nominal and 2-year extended mission lifetime. Science shall be done mainly around apogee, where the overall constellation shall be in an optimised configuration.

The "tailbox" is defined as in Figure 3 [2]:
- Position Q is 10 Earth radii (Re) from the centre of the Earth in anti-sunward direction along the equatorial plane.
- The centre of the tailbox, position P, is at a distance of 30 Re from the Earth's centre, with the line Q-P parallel to the ecliptic plane.
- The tailbox is defined as a rectangular box parallel to the ecliptic plane:
  - 25 Re along the Q-P line, extending 5 Re tailward of the centre P.
  - 4 Re orthogonal to the ecliptic plane (+/- 2 Re from the tailbox centre P).
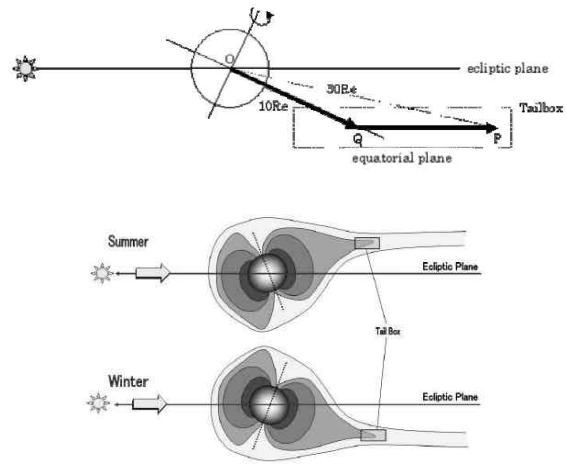 - 10 Re parallel to the dawn-dusk terminator (+/-5 Re from the centre P).

Figure 3: Tailbox Defintion (courtesy of ESA)

The number of S/C is the main driver to optimise the scientific mission return. Other important factors (with less scientific priority, according to ESA) are the payload resources embarked on each S/C (instrument mass and power) and the reference orbit.

The CS launch is intended to take place between 2015 and 2025. The system-level goal to launch all the satellites within one Soyuz/Fregat (together with a Transfer Vehicle) is mainly due to cost limitation and operations simplification and triggers the trade-off of the mission orbit vs. the number of in-orbit S/C.

3 MISSION ORBIT TRADE-OFF

The operational highly elliptical orbit (HEO) for the CS mission concept has been designed taking into account orbit stability with respect to perturbations, eclipse frequency and durations, communications with the ground stations, radiation environment, end-of-life S/C disposal strategy, as well as the main scientific objectives. In fact, the CS reference orbit drives the capability to visit the relevant regions in near-Earth space where the scientifically most interesting plasma processes occur and, in particular, determines the optimum season to cross the tail current sheet, which represents one of the main scientific mission goals. Near-equatorial orbits with apogee of 25 Re and a range of perigees between ~1 Re and 10 Re have been analysed, since they cross the bow-shock region, the magnetosheath and visit the magnetotail each year. An alternative polar orbit, initially considered for preliminary analyses, has been discarded due to the considerable orbit insertion ΔV needed, which does not fulfil the tight requirements in terms of total number of in-orbit S/C.

The key trade-off performed in the frame of the mission analysis activities addresses the selection of the reference mission orbit, mainly in terms of perigee radius, and the assessment of the corresponding impact on the number of satellites that can be inserted into that orbit, depending on the ΔV requirements for

orbit acquisition and mission manoeuvres, and on the launcher injection performances.

The following tables provide a brief comparison of the candidate CS reference orbits, encompassing eclipse and radiation dose assessments, tailbox visiting performance (directly related to the scientific return) and ΔV budget (which drives the number of S/C).

Table 1: Orbit Trade-Off: Eclipse and Radiation Dose Analyses

Orbit	Max Eclipse Duration over 5 years [hrs]	Total Radiation Dose for Solid Sphere [krad_Si]
10 Re x 25 Re	3.37	6 krad
4 Re x 25 Re	4.73	69 krad
3 Re x 25 Re	5.33	90 krad
1.39 Re x 25 Re	7.93	64 krad
1.08 Re x 25 Re	8.50	57 krad

Table 2: Orbit Trade-Off: Tailbox Visiting over 1 Year

Orbit	Visiting Intervals	Tot Visiting [days]	Max Visiting [hr]	Average Visiting [hr]
10 Re x 25 Re	76	29.03	15.50	9.17
4 Re x 25 Re	52	23.44	22.97	10.82
1.08 Re x 25 Re	22	12.81	33.30	13.98

Table 3: Orbit Trade-Off: ΔV Budget for Dispenser and S/C

Orbit Per.	Orbit Insertion ΔV (from GTO) [m/s]	Mission ΔV [m/s]			Disposal ΔV [m/s]	
		Large-Scale S/C	Medium-Scale S/C	Small-Scale S/C	Transfer S/C	S/C
10 Re	1388.5 (1115, lunar resonances)	198.6	50.8	3.8	-	-
4 Re	1005.5	347.0	87.2	5.3	218	185
3 Re	902.1	395.7	101.6	6.0	306	288
2 Re	772.6	444.4	116.2	6.7	177	201
1.4 Re	672.8	540.4	134.4	8.6	77	106

The results obtained point out that the eclipse event with maximum duration occurs during the last year of the mission lifetime and can be assumed as the worst-case scenario for S/C power subsystem sizing. As for the ΔV budget, the CS mission architecture envisages a Dispenser/Transfer Vehicle to perform the orbit insertion manoeuvres and its own disposal, while each S/C shall carry out the deployment, reconfiguration and maintenance manoeuvres and implement a proper disposal strategy at end of life. The largest impact on the mission ΔV is due to the orbit insertion phase, based on a sequence of transfer manoeuvres, the large-scale S/C manoeuvres during the mission lifetime and the disposal strategy.

The main outcome of the orbit trade-off process can be summarised as follows:

❑ A lower perigee decreases the number of tailbox visits and the total time spent in the tailbox.

❑ A lower perigee improves the average constellation shape during tailbox visits (i.e. visits occur close to apogee, the point of highest interest for scientific measurements).

❑ A lower perigee decreases the all-up ΔV requirement (unless using lunar resonances for the transfer phase in the high-perigee case).

❑ A lower perigee increases the number of eclipses per year, and the maximum eclipse duration.

❑ A lower perigee increases the radiation dose affecting the S/C (except between 1.4 and 2 Re).

❑ A lower perigee improves the average data download capability, allowing data delivery closer to the Earth with larger data rates and power.

Orbits with perigee between 2 and 4 Re turn out to be unattractive compared to the 10-Re perigee case and the ~1.4-Re perigee (2500-km perigee altitude), mainly due to the compliance with the guidelines contained in the European Code of Conduct for Space Debris Mitigation [3], which requires that any S/C (or transfer vehicle) shall not enter the "LEO protected zone" and the "GEO protected zone" after mission completion. Consequently, the orbit trade-off has eventually focussed on two orbits:

❑ A high-perigee (> 7 Re) orbit using lunar-resonance transfer (to reduce the transfer ΔV to be provided by the Dispenser/Transfer Vehicle).

❑ A low-perigee (~1.4 Re) orbit with either natural de-orbiting within 25 years or dedicated de-orbiting manoeuvre strategy.

The **low-perigee option** (~1.4 Re) has better performance and allows, with a safe system mass margin, a constellation with **10 S/C** instrumented with the desired payload configuration. The overall mass benefit in propellant allocation in the low-perigee case outweighs the increase in battery mass due to longer eclipse times.

4 MISSION ORBIT DESIGN AND ANALYSIS

Based on the outcome of the orbit trade-off, the low-perigee option has been retained as the baseline for the detailed mission analyses in the frame of the CS study, leading to the selection of the following **reference orbit for the CS TRS**:

❑ Perigee radius: ~1.4 Re (altitude = 2500 km).

❑ Apogee radius: 25 Re.

❑ Inclination: 14°

The orbit **design approach** for the CS mission concept is based on two driving criteria:

❑ **Orbit stability**: the reference CS orbit shall be designed to be stable and minimise orbit evolution. For this type of highly elliptical orbits, the main target is to ensure that the perigee height does not descend into the Earth's atmosphere during the mission lifetime. The dipping motion

of the perigee height is due to the luni-solar perturbations and depends on the orientation of the orbit at launch time (i.e. the initial RAAN, and in particular the initial RAAN with respect to the Moon node position).

❑ **Tailbox visiting performance**, in terms of the following requirements and figures of merit:

• The apogee shall visit the tailbox at least once per year.

• The frequency and duration of the tailbox visiting intervals are the key factors.

The main orbit **design parameters**, which have been varied in the design process to assess their impact on the design criteria, are the following:

❑ Perigee and apogee radii: used for orbit trade-off vs. number of S/C.

❑ Initial Right Ascension of the Ascending Node (RAAN).

❑ Initial inclination.

The initial argument of perigee is selected to achieve optimum coverage for a ground station in the northern hemisphere, which is equivalent to a value of 270°. Thus, it has not been used as design parameter.

For the reference CS orbit selected, the orbit design process deals with the following main analyses:

❑ Analysis of the orbit evolution during the mission lifetime and assessment of the orbit stability.

❑ Eclipse analysis, aimed at evaluating the frequency and duration of the eclipse events.

❑ Tailbox visiting analysis to assess the scientific mission return enabled by the reference orbit.

4.1 Orbit Evolution Analysis

The orbit proposed for the CS concept has a particularly high apogee radius to enable the study of the magnetotail at large distances from the Earth. Such orbit is highly perturbed by the gravity field of the Sun and particularly of the Moon. Therefore, it is to be expected that the absolute orbit orientation changes significantly along the 5-year mission lifetime.

Typically, controlling the natural evolution of HEO orbits is very expensive in terms of ΔV. Thus, it has been assumed that no absolute control of the reference orbit is applied, unless the natural orbit drift causes failure to meet the requirement of yearly tailbox visit. The orbit design is intended to guarantee orbit stability throughout the mission lifetime based on a proper selection of the initial orbital elements.

The selection of the initial orbit RAAN (which depends on the launch time) is mainly driven by the orbit stability requirement, i.e. to guarantee perigee altitude maintenance or an increasing perigee trend during the mission lifetime. For the CS mission concept, the orbital parameters (in particular inclination, argument of perigee and RAAN) are further constrained by the requirement to visit the tailbox once a year. Once the orbit inclination has been selected, the combination of RAAN and argument of perigee determines the period of the year when the tailbox can be visited at apogee.

The perigee height evolution strongly depends on the value of the initial RAAN, and in particular on the value of the initial RAAN with respect to the Moon node, since the lunar perturbation represents the strongest effect acting on the HEO considered. The initial RAAN is actually the main driver for orbital stability; thus, in a first step the baseline orbit evolution analysis has been performed as a function of the initial RAAN, while keeping the argument of perigee and the inclination constant.

Figure 4 and Figure 5 present respectively the evolution of the perigee altitude and of the orbit eccentricity, inclination, RAAN and argument of perigee for the baseline orbit with initial perigee at 2500 km. This parametric analysis is enabled by the degree of freedom allowed by the requirement that the apogee has to be in the tailbox at least once a year, but not in a specific season.

The simulation results obtained point out that low initial RAAN values (close to 0°) provide more stable orbits with respect to luni-solar and Earth gravity perturbations. Stable perigee height and small variations of inclination and argument of perigee lead to less variation of the orbit geometry during the tailbox visiting intervals.

More detailed analyses carried out during the study have assessed the impact of the initial orbit inclination and of the launch date on the orbit evolution. These analyses have demonstrated that the initial orbit inclination and the launch date have a second-order effect on the orbit evolution, thus confirming that the analysis of the orbit stability as a function of the initial RAAN is the leading factor to be taken into account.
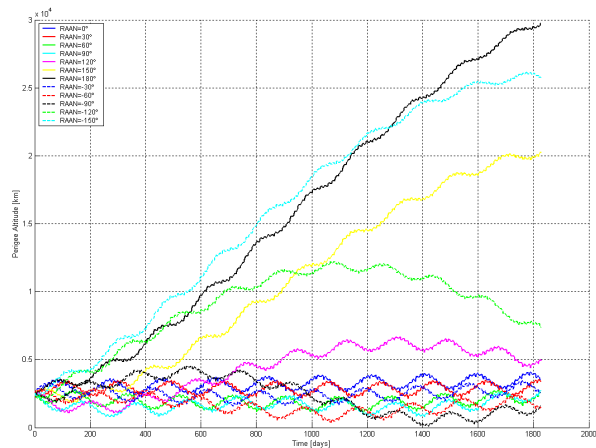

Figure 4: Perigee Altitude Evolution over 5 Years as a function of the Initial RAAN (Initial Inclination = 14°)
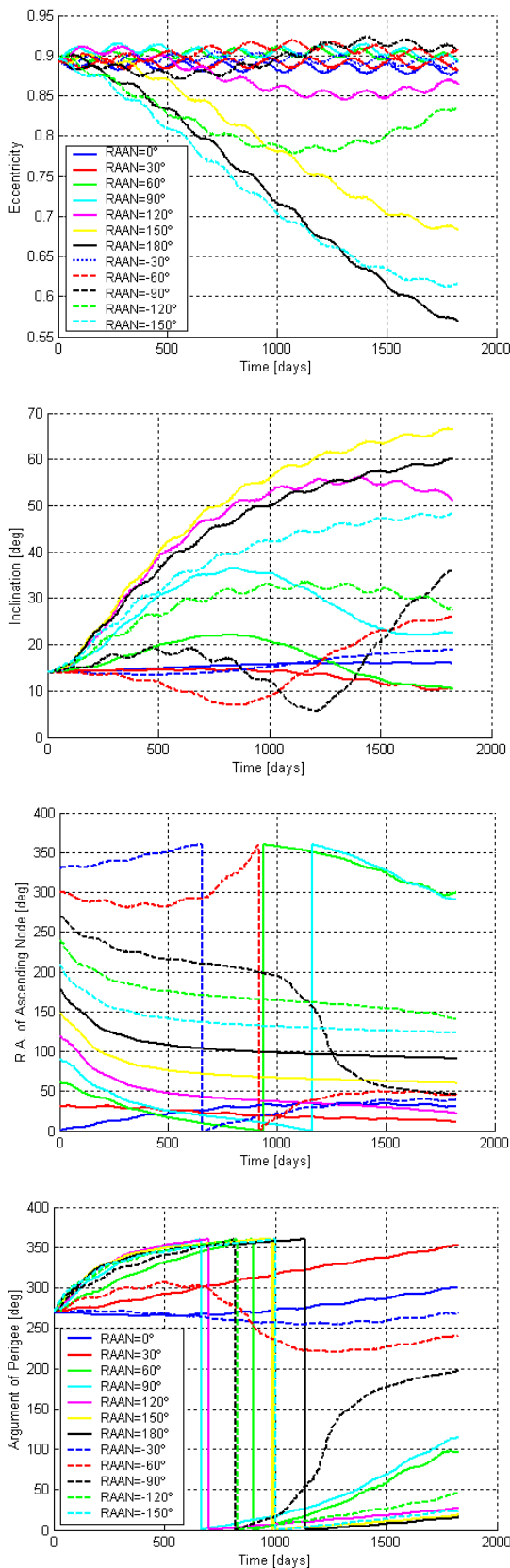
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Figure 5: Orbital Element Evolution over 5 Years as a function of the Initial RAAN (Initial Inclination = 14°)

Based on the outcome of the orbit analysis, an initial RAAN of 0° is selected as the baseline. The launch time shall be selected depending on the day of the year so as to achieve the target initial RAAN. In addition,

for an inclination of 14° and an argument of perigee of 270°, an initial RAAN equal to 0° implies that the apogee is nominally located towards the tailbox at winter solstice (Figure 6).
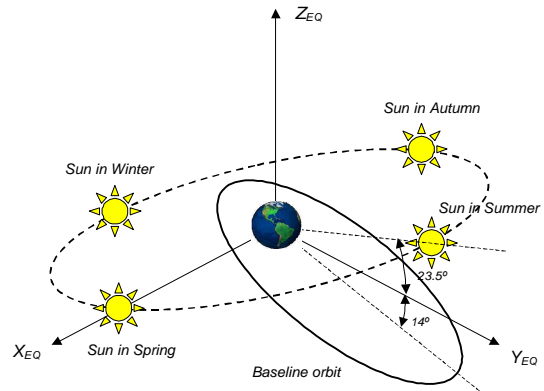

Figure 6: CS Baseline Orbit projected in an Earth Equatorial System, along with the Yearly Orbit of the Sun in J2000.0

4.2 Eclipse Analysis

The eclipse analysis has been performed as a function of different launch dates distributed along the year, in order to assess the impact on the eclipse profile evolution and on the maximum eclipse duration over the mission lifetime. The eclipse events show a shifting pattern as a function of launch date (Figure 7).
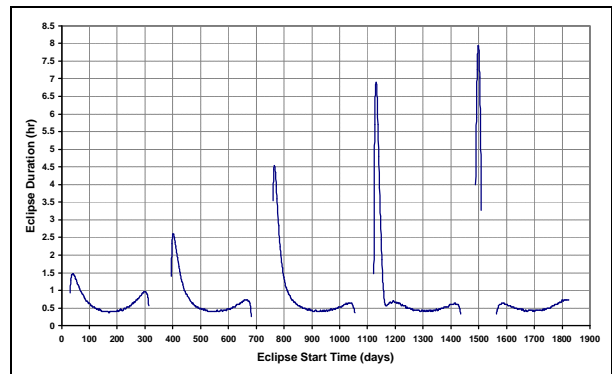

Figure 7: Eclipse Duration as a function of Eclipse Start Time over 5 Years (Launch Date in January)

Eclipse events longer than 4 hrs occur starting from the third year. The maximum eclipse duration occurs during the last year of the mission lifetime and it is between ~7.9 hrs and ~8.4 hrs. Launch dates in winter and summer minimise the maximum eclipse duration and allow a lower total accumulated eclipse time.

For the CS mission orbit the key point is whether the instruments can be operated during the eclipse periods and the corresponding impact on the S/C design. In the case of very long eclipse intervals, an operational scenario where the instruments are operated only during part of the eclipse duration has been envisaged to allow power saving.

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4.3 Tailbox Visiting Analysis

This analysis addresses the frequency, duration and position along the orbit of the tailbox visiting intervals. Preliminary orbit analyses have shown that no relevant tailbox visiting improvement is gained by slightly modifying the inclination with respect to the nominal value of 14°, which is thus retained for the current assessment.

Figure 8 and Figure 9 presents respectively the duration of the tailbox visiting intervals as a function of the visiting start time and the accumulated time in the tailbox along the 5-year mission lifetime. These results point out that the largest tailbox visiting duration is achieved in winter, while long tailbox visiting gaps cover spring, summer and fall. Due to the considerable impact of the luni-solar perturbations on the orbit, an important orbit evolution exists if no orbit control is applied. This leads to a degradation of the tailbox visiting performance over the mission lifetime, such that no tailbox visiting is possible during the last 8 months of the 5-year simulation time.
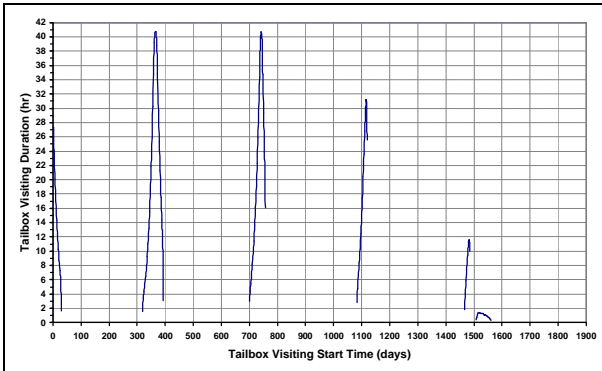

Figure 8: Tailbox Visiting Duration as a function of the Tailbox Visiting Start Time (5-year simulation)
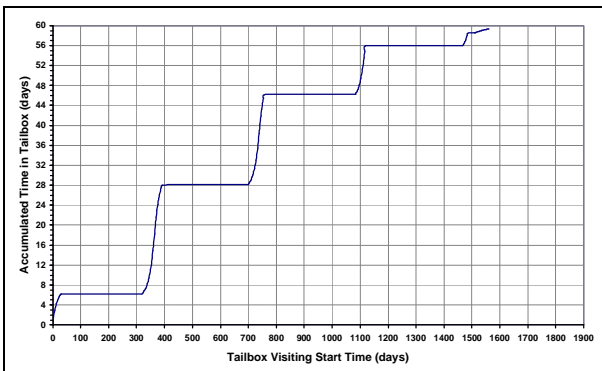

Figure 9: Accumulated Time in the Tailbox as a function of Time (5-year simulation)

The tailbox is visited at true anomaly between 145° and 215° (close enough to apogee), thus ensuring a good average constellation shape during visit. In fact, the apogee identifies the orbit portion where the shape of the constellation is optimised.

Table 4 outlines the evolution of the tailbox visiting performance over the 5-year mission lifetime, encompassing the number of visiting intervals and the associated duration, as well as the accumulated time in the tailbox and the total time of tailbox visiting at apogee (the orbit point of greatest interest). It is important to underline that the tailbox is visited at least once per year at apogee, thus allowing the fulfilment of the driving scientific requirement.

Table 4: Tailbox Visiting Performance over 5 Years

Time	Tailbox Visiting Statistics				
	Intervals	Max Time [hr]	Avg Time [hr]	Tot Time [days]	Tot at Apogee [% Tot]
Year 1	28	40.7	15.6	18.2	25.2%
Year 2	22	40.7	17.4	15.9	50.1%
Year 3	14	40.7	22.9	13.4	65.3%
Year 4	9	31.1	22.4	8.4	31.3%
Year 5	28	11.5	2.9	3.4	11.7%
Summary					
5 Years	101	40.7	14.1	59.3	41.0%

5 LAUNCH AND TRANSFER SCENARIO

5.1 ΔV Budget and Mass Injection Performance

The primary objective of the launch and transfer analysis is to address a scenario based on the use of **one Souyz-Fregat 2-1b launched from Kourou** [4], and to assess the mass injection capability into the CS target orbit, using a dedicated transfer sequence.

The performances of the Soyuz-Fregat 2-1b from Kourou consist of the mass inserted into the launcher injection orbit as a function of the injection orbit apogee radius, covering a range of apogee radii up to the GTO apogee and starting from a mass of 7694 kg injected by Soyuz into a circular parking orbit at 180 km of altitude (Figure 10). In particular, the Soyuz-Fregat injection performance into a GTO inclined of 14° (CS orbit inclination) is possible up to 3026 kg (including the 111-kg adapter). The variation of the injected mass with the apogee of the injection orbit is also a driver because a possible optimisation can be performed with respect to this parameter.
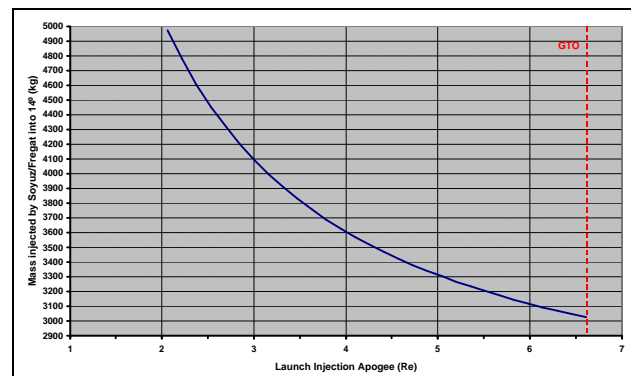

Figure 10: Soyuz/Fregat Injected Mass into Orbit as a function of the Injection Orbit Apogee (Soyuz-Fregat Performance Model from CSG, provided by ESA in February 2007)

The typical transfer phase is based on an injection into an initial orbit by Soyuz/Fregat with a given apogee, followed by a sequence of perigee burns to reach the operational apogee and an apogee burn to reach the target perigee (Figure 11). In the CS mission scenario, the Dispenser carries out the transfer manoeuvres.
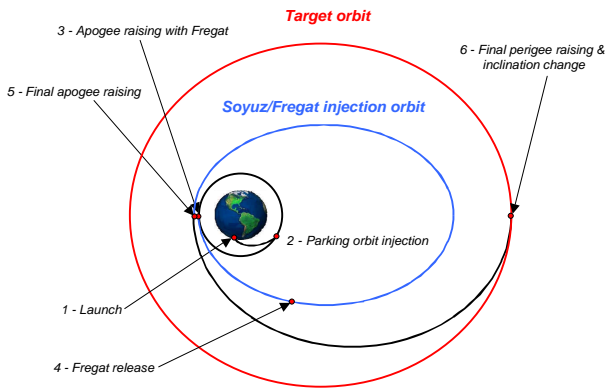

Figure 11: Soyuz-Fregat Lunch Sequence for Injection into the Cross-Scale Target Orbit

An injection apogee below the GTO may bring advantages in terms of mass into the target orbit. A trade-off has been performed between the increase of the mass into the launcher orbit (achieved by decreasing the injection apogee) and the amount of propellant to be carried by the Dispenser, leading to the selection of an injection apogee radius of 20164 km. The corresponding overall ΔV to reach the CS operational orbit, covering the apogee and perigee raising manoeuvres, is 1411 m/s. This solution allows a large mass to be injected into the CS orbit, while fulfilling the structural constraints of the state-of-art Dispenser tanks based on the SpaceBus technology.

The transfer ΔV assessment includes ~15 m/s for launcher injection error correction, gravity losses and an additional 5% margin on the transfer manoeuvres. The gravity losses have been computed for the sequence of perigee manoeuvres devoted to raise the apogee altitude. To enable the use of existing transfer vehicle technology, a sequence of 8 burns with duration of approximately 4.6 days has been selected, leading to gravity losses on the order of 2.6%. The mass margin enabled by lower injection apogees is intended to allow a robust system design that can deal with potential additional mass requirements (Table 5).

Table 5: Gravity Loss, Transfer ΔV and Mass into the CS Orbit as a function of the Injection Apogee (Isp = 325 s, Thrust = 500 N, Fregat adapter = 111 kg)

Rapo (km)	Soyuz Injected Mass (kg)	Gravity Losses (%)	Tot-ΔV (km/s)	Mass in Orbit (kg)
13164	4974	3.20%	2.0353	2569
17164	4320	2.93%	1.6295	2524
20164	**3996**	**2.63%**	**1.4105**	**2496**

5.2 Launch Window

The Soyuz/Fregat vehicle can be launched any day of the year, any time of the day respecting the specified lift-off time. Based on orbit stability considerations, it can be expected that the launch window is open every day of the year for the CS satellites and that the launch time shall be selected depending on the day of the year in order to achieve the target initial RAAN.

Additional constraints on launch seasons and launch time may be driven by:

- Eclipse duration during the transfer phase from the launcher injection orbit to the operational orbit. Eclipse duration has an impact in terms of thermal constraints and battery size for the Dispenser/Transfer Vehicle in charge of performing the transfer to the target orbit.
- Sun illumination during orbit transfer manoeuvres, which has an impact on the Dispenser/Transfer Vehicle power and thermal requirements. The driving constraint applied during the orbit transfer manoeuvres is that the Sun direction should be close to the perpendicular to the thrust direction, so as to avoid direct Sun input into the boost motor.

5.3 Transfer Phase

Starting from the launcher injection orbit, a sequence of manoeuvres is applied around each perigee passage to increase the apogee altitude up to the target value. Then, an apogee manoeuvre is performed to increase the perigee altitude to the target value.

The eclipse duration during the transfer phase has an impact in terms of thermal constraints and battery size for the Dispenser Vehicle. The worst-case eclipse duration (~1.5 hours) occurs for a launch date in fall, as highlighted in Table 6.

Table 6: Eclipse Statistics during the Apogee Raising Phase

Launch Date	Eclipse Events	Max Eclipse (hr)	Average Eclipse (hr)	Total Eclipse (hr)
January	5	1.17	1.10	5.52
March	8	0.88	0.82	6.56
May	9	0.48	0.41	3.72
July	9	0.41	0.37	3.33
September	9	0.55	0.54	4.87
November	9	1.42	1.09	9.84

Figure 12 provides a graphical overview of the orbit spiralling-out phase, covering the evolution of the relevant orbital elements, as well as the evolution of the Dispenser Vehicle mass.
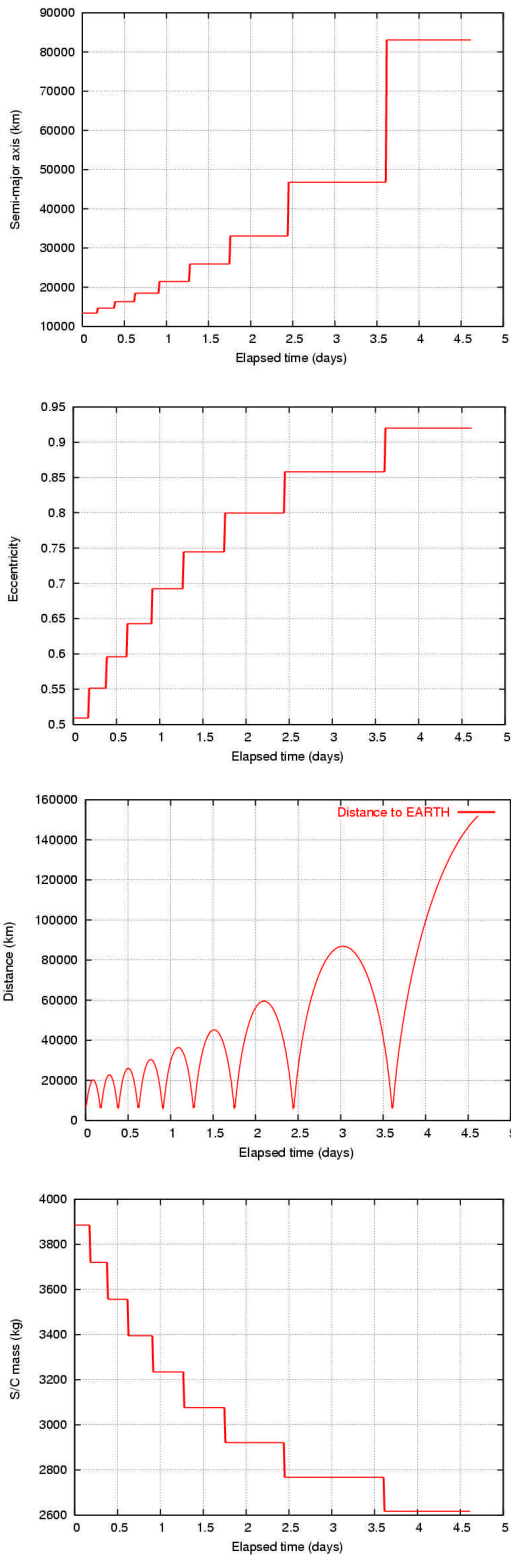
Figure 12: Evolution of the Orbital Parameters during the Apogee Raising Phase

6 FORMATION DESIGN AND CONTROL

In the CS context, a methodological step-wise approach has been applied for formation design, as well as for formation deployment, reconfiguration and maintenance. All the ΔV assessments include 5% margin as required by ESA margin philosophy for conceptual studies.

6.1 Formation Design

The formation[*] design problem consists in defining the orbital parameters for each S/C in the formation in each spatial scale of interest, so that they naturally and repeatedly form the desired space configuration (either a tetrahedron or mother-daughter system) at the apogee, and do not degrade excessively during the rest of the orbit (including perigee), while ensuring collision avoidance throughout the mission lifetime.

This methodology is in line with the requirement that science shall be done mainly around apogee. Therefore, **a regular tetrahedron per scale is imposed for observation phases around apogee**, where the overall constellation shall be in an optimised configuration. The reference orbit is taken as the centre of the tetrahedron at the design point, i.e. at apogee (Figure 13). A virtual master S/C is placed at the geometrical centre of the tetrahedron and moves along the reference orbit, while the four formation satellites are the flyers.
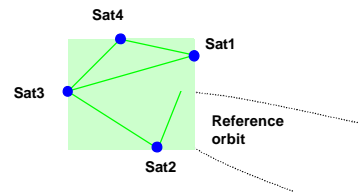

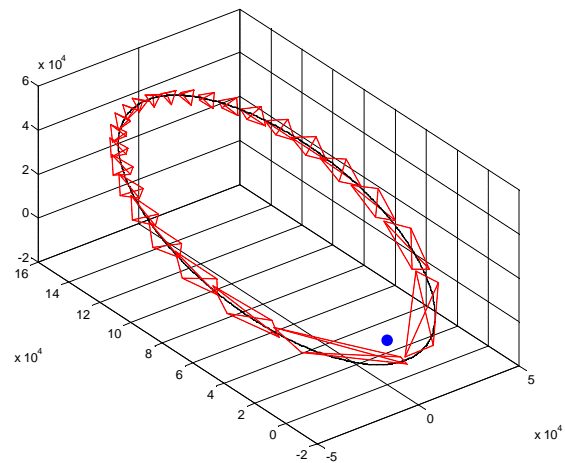**Figure 13: Formation Design at the Optimisation Point**

Figure 14: Evolution of Tetrahedron Shape along One Orbit

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[*] The "Formation Flying" (FF) concept has been applied to the tetrahedrons and to the mother-daughter system to address "loose-formations" where the expected nominal S/C positions are not expressed as reference points with very small control windows, but as volumes where the S/C have to be maintained with rather loose control requirements. The inter-S/C navigation and synchronisation play a key role to provide the scientific measurements, particularly in the small and medium scales, thus yielding a relevant level of interaction between the S/C in these scales.

A stable tetrahedron is designed at the apogee of the reference orbit by imposing null derivatives of inter-S/C distances and angles at that point. The **design scale distances** (distance between two S/C on the same scale) considered to size the formations in the three scales are: 10 km in the small scale, 1000 km in the medium scale and 6000 km in the large scale.

The key design drivers applied to select the formation design solution are the following:

❑ **A minimum allowed distance** is considered for safety reasons. The formations in the three scales have been designed to provide sufficient robustness and avoid collisions between the S/C in the nominal space segment configuration. Those designs providing too small distances between any two satellites of the formation at any point of the orbit have been discarded.

❑ **Minimisation of deployment and reconfiguration DV**. All the possible solutions of the geometrical design problem have been tested in terms of the implied ΔV, leading to the selection of low-ΔV solutions.

In particular, the selection of null deltas in RAAN between the reference CS orbit and the orbits of the flyer S/C has been made (i.e. the S/C composing the formation have the same RAAN as the reference master orbit) in order to save the propellant that would be needed for plane change during the deployment phase. This choice does not jeopardise the generality of the FF design solution, since it takes advantage of the degrees of freedom that exist in the design problem formulation (i.e. in the corresponding equations).

❑ **Analysis of inter-satellite distances, angles and communication geometry**. This analysis is aimed at assessing whether the selected formation design allows easy communications and relative navigation not only between S/C contained in the same scale, but also between S/C belonging to different scales. A reference direction has been assumed for communication purposes: the S/C spin axis, which is normal to the Ecliptic plane.
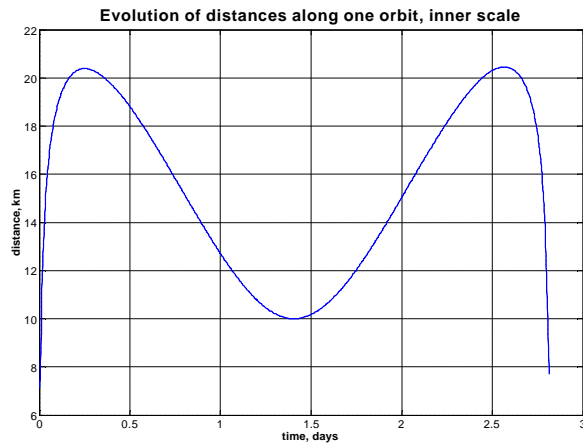

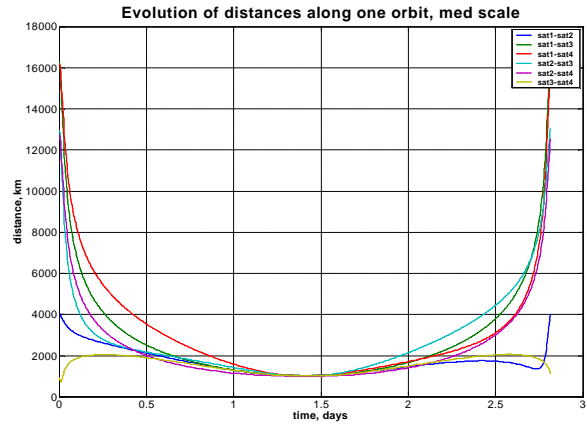**Figure 15: Evolution of Small-Scale S/C Distances over 1 Orbit**

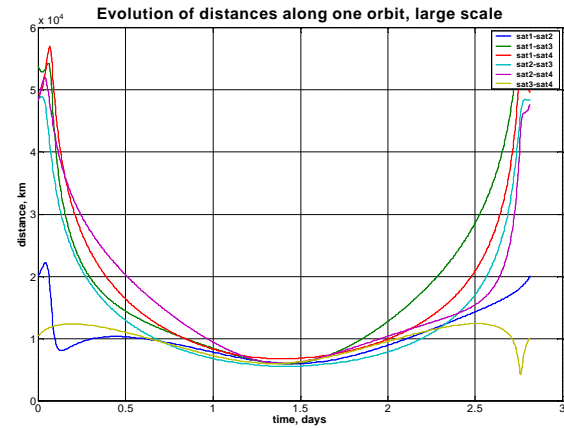**Figure 16: Evolution of Med-Scale S/C Distances over 1 Orbit**

Figure 17: Evolution of Large-Scale S/C Distances over 1 Orbit

6.2 Formation Deployment

The objective of the deployment phase is to distribute the S/C in space so that they enable measurements in three length scales simultaneously, each characterised by a reference scale distance.

The S/C deployment is implemented as a sequence of two manoeuvres (ideally performed in one or two orbits after the accomplishment of the launch and transfer phase), with the objective to change the S/C orbit elements and transfer the S/C from the reference orbit (~1.4 Re x 25 Re x 14°) to their target orbits respectively in the large-scale, medium-scale and small-scale configurations defined by the FF design.

The S/C shall perform the deployment manoeuvres using their on-board propulsion system. The deployment ΔV becomes a driving contribution for the assessment of the fuel budget and, as a consequence, of the S/C mass budget. The most demanding ΔV is associated to the large-scale S/C (108 m/s), since their deployment implies the achievement of the largest S/C separation (i.e. 6000 km) to distribute them in the corresponding tetrahedron. On the other hand, the deployment ΔV for the small-scale and the medium-scale S/C amounts respectively to 0.2 m/s and 18 m/s.

6.3 Formation Reconfigurations

The formation reconfigurations are aimed at modifying the S/C distances in the three scales, so as to tune the characteristic measurement length in each scale, without degrading the orbital properties arranged by design. The reconfigurations are intended to comply with the requirements outlined in Table 7.

Table 7: Formation Reconfigurations [2]

Scale	Number of Changes	Re-Configuration Delta (km)		
		From	To	Delta
Small	5-20	2-10 km	100 km	90 km
Medium	3-10	50 km	2000 km	1950 km
Large	1	6000 km	3000 km	3000 km
	1	3000 km	15000 km	12000 km
	1	15000 km	6000 km	9000 km
			Total delta (km):	24000 km

The timeline for the large-scale reconfigurations has been specified starting from an initial reference S/C separation of 6000 km. During the science acquisition phase, one scan shall be performed from 6000 km (0.5 y + commissioning time) to 3000 km (0.5 y) to 15000 km (1 y) to 6000 km (till the end of the mission).

Reconfiguration manoeuvres to be carried out during the nominal and the extended operational lifetimes provide a driving input for the mission ΔV. The total reconfiguration ΔV for the large-scale S/C is computed considering the overall delta in S/C separations required over the mission lifetime (24000 km), providing an estimation of 432 m/s per S/C.

In the small and medium scales, the number of reconfigurations can be used to perform a parametric analysis and assess different ΔV budgets (Figure 18 and Figure 19). The objective is to identify a feasible number of reconfigurations to comply with the S/C mass constraints. To keep the reconfiguration ΔV for the medium-scale S/C at a reasonable value in terms of impact on the S/C mass, the option with a minimum number of reconfigurations emerges as the most appropriate (104 m/s per S/C). The same criterion can be applied to the small-scale S/C, although in this case the impact of the reconfiguration ΔV is not critical (8 m/s per S/C).
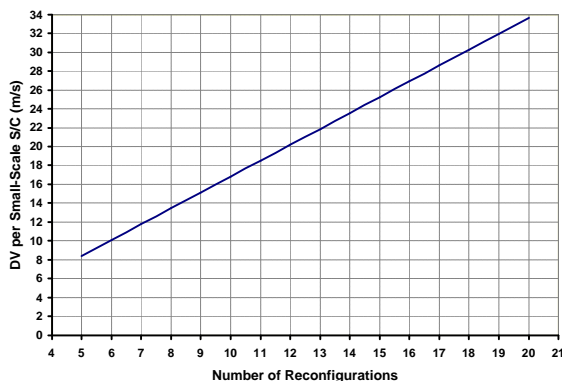

Figure 18: Small-Scale Reconfig. $\mathbf{D}$V vs. Number of Changes
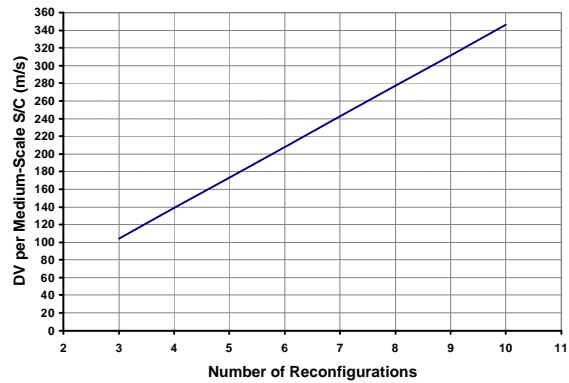

Figure 19: Medium-Scale Reconfig. $\mathbf{D}$V vs. Number of Changes

6.4 Formation Maintenance

Formation maintenance is driven by the following requirements:

❑ **Control of formation centre relative distance**:
   - The distance between the centres of the medium-scale tetrahedron and the mother-daughter system shall be less than 25% of the actual medium-scale distance.
   - The distance between the centres of the large and medium scale tetrahedrons shall be less than 25% of the actual large-scale distance.

❑ **Control of the satellite distance**:
   - Medium scale: the distance between any two S/C shall not differ by more than 10% from the actual average medium-scale distance.
   - Large scale: the distance between any two S/C shall not differ by more than 25% from the actual average large scale S/C separation.

The "actual average" scale distance is defined as the actual separation at the optimisation point averaged over all S/C distances (it differs from the scale distance at the beginning of the mission due to differential drift). The S/C distance requirements are expressed in terms of the deviation from the optimised configuration at the optimisation point, i.e. the orbit point used for formation design. The objective is to impose a constraint on the constellation shape, meaning that the scale distance is allowed to change due to perturbations, but the deviation between the actual inter-S/C distances and the nominal scale distances has to comply with certain limitations.

To assess the need for dedicated maintenance manoeuvres during the mission lifetime, the S/C of the three scales have been propagated in open loop (i.e. without applying any control) and the conditions on the maintenance of the formation centres and on the relative S/C distances have been verified around the reference orbit apogee (design optimisation point).

The distance between FF centres results to be within the specified boundaries; thus, **no dedicated control is required to maintain the FF centres**.

Figure 20 presents the evolution of the large-scale S/C distances around apogee and the corresponding average inter-S/C distance, along with the upper and lower boundaries of the average S/C distance, which define the control window (±25% of the actual average large-scale S/C separation). This figure shows that the control window is no longer fulfilled after 350 days, i.e. ~1 year from the beginning of the mission.
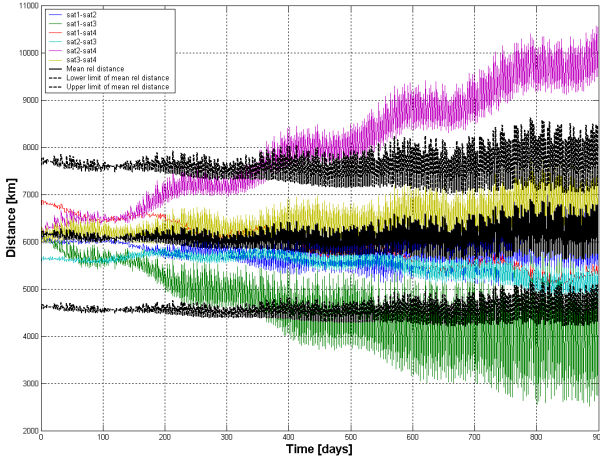

Figure 20: Large-Scale S/C Distance Evolution around Apogee

An optimised strategy to fulfil the formation maintenance requirements, while minimising the number of required manoeuvres and the associated ΔV and operations support needed, is to **optimally combine the reconfiguration manoeuvres and the FF maintenance manoeuvres**. The timeline for the large-scale reconfigurations envisages a first reconfiguration after 0.5 years from the beginning of the mission: this manoeuvre can be used to achieve a scaled configuration with respect to the nominal one (achieved after deployment) and compensate the accumulated formation drift up to that time.

The longest time interval between reconfiguration manoeuvres is 1 year, which is very close to the time interval up to the first large-scale control manoeuvre. Assuming that the large-scale control requirements can be relaxed during the 1-year interval between the second and the third reconfiguration (control window increased to 28.5%), by bringing the third reconfiguration forward of approximately one month, it is possible to **avoid dedicated control manoeuvres to maintain the large-scale S/C relative distances**.

Figure 21 presents the evolution of the medium-scale S/C distances around apogee. The control window (±10% of the actual average medium-scale S/C separation) is no longer fulfilled after 153 days, i.e. 5 months from the beginning of the mission.

Maintenance manoeuvres are needed in the medium scale. A **relative orbit control strategy** has been applied:

❑ The mean orbital elements of each S/C are computed at the manoeuvre time, as well as the average orbit of the four medium-scale S/C.

❑ The design deltas in the orbital elements of each S/C with respect to the master are applied to the average orbit to obtain the target orbital elements of each S/C after the control manoeuvres.

❑ Two manoeuvres are applied to each S/C to enable the change from the mean orbital elements to the target orbital elements.

This control strategy is conducive to minimising the ΔV needed, since it performs control with respect to an average orbit obtained from the positions of the four satellites at manoeuvre time and not with respect to the initial reference orbit. This S/C control strategy does not compensate the reference orbit drift with time: the reference orbit of the four S/C is allowed to change and no absolute control is applied.
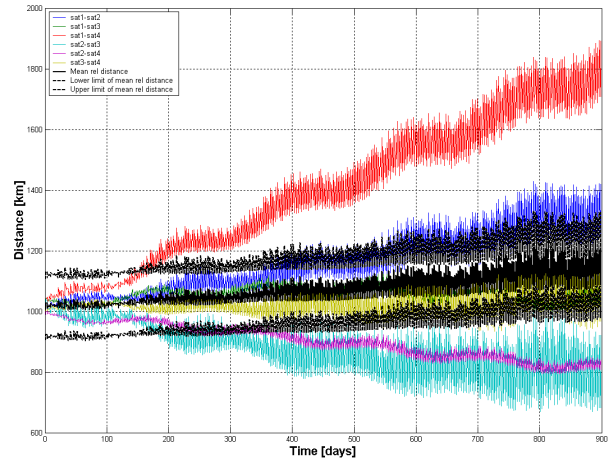

Figure 21: Med-Scale S/C Distance Evolution around Apogee

The average control ΔV per S/C is 1.6 m/s, which is considerably smaller than the ΔV for deployment and reconfiguration manoeuvres. The overall maintenance ΔV depends on the number of control manoeuvres to be implemented along the mission lifetime.

The optimal combination of the reconfiguration manoeuvres (3 to 10 for the medium-scale S/C) and the FF maintenance manoeuvres yields the results displayed in Table 8. If the minimum number of reconfigurations is selected to minimise the associated ΔV, the control ΔV will be the largest, i.e. approximately 13 m/s. In this case, the average ΔV per medium-scale S/C to perform 3 reconfigurations and the control manoeuvres is ~117 m/s.

Table 8: Combination of Reconfiguration Manoeuvres and Maintenance Manoeuvres for the Medium-Scale S/C

Reconfiguration manoeuvres	Reconfiguration frequency (months)	Number of control manoeuvres	Control ΔV per S/C (m/s)
3	**15.0**	**8**	**12.7**
4	12.0	7	11.1
5	10.0	6	9.5
6	8.6	5	7.9
7	7.5	4	6.4
8	6.7	3	4.8
9	6.0	2	3.2
10	5.5	1	1.6

7 END-OF-LIFE SCENARIO

The "European Code of Conduct for Space Debris Mitigation"[3] outlines the guidelines for the disposal of a satellite as a function of the apogee and perigee altitudes at the end of the operational mission lifetime. Any S/C (or transfer vehicle) shall not enter the "LEO protected zone" and the "GEO protected zone" after mission completion. In addition, the S/C shall be passivated at end of life (EOL).

In the case of the CS orbit, the possible S/C disposal strategies encompass:

❑ Direct S/C re-entry (i.e. immediate de-orbiting). Limiting the space system orbital lifetime to less than 25 years after its operational phase might be an alternative.

❑ Disposal orbit above the extended GEO region.

❑ Disposal orbit between the LEO region and the extended GEO region.

For the CS orbit with a low perigee at ~1.4 Re, the cheapest disposal strategy in terms of ΔV is an immediate S/C de-orbiting by implementing one manoeuvre to lower the perigee and allow a controlled S/C re-entry. The corresponding ΔV to be provided by each S/C at EOL is ~106 m/s (including 5% margin).

After performing the transfer phase to the target CS operational orbit and the S/C release, the Dispenser Vehicle shall undergo a disposal strategy. The objective is not only to comply with the Space Debris Mitigation policy, but also to avoid any risk of collision with the CS satellites during their mission lifetime. The ΔV for the Dispenser de-orbiting at beginning of life is ~77 m/s. Hence, appropriate propellant budgets should be added for this provision, with a direct impact on the Dispenser mass.

8 INTER-SATELLITE LOCALISATION AND SYNCHRONISATION

8.1 Requirements and CS Solution

In order to correlate science data from various S/C, good relative timing and localisation accuracies are required. Table 8 summarises the inter-S/C synchronisation and navigation requirements as a function of the characteristic scale [2].

For the CS mission concept, the current requirements should allow for a solution with RF links at small scale, and ground-supported orbit determination for the fluid scale (actually, the large scale has the same requirements as Cluster–II). For the medium scale S/C, a hybrid solution might be envisaged: e.g., for inter-S/C distances below 1000 km, a RF link could be used; while for distances above 1000 km, ground-supported orbit determination could be applied (~2 ms timing accuracy and ~10 km orbit position accuracy).

Table 9: Inter-S/C Localisation/Synchronisation Requirements

Scale	Relative Timing Requirements	Navigation Requirements		
		Relative Distance Req.	Scale Distance	Accuracy
Small	Accuracy ≤ 250 µs between small-scale S/C (goal is 10-100 µs)	Accuracy ≤ 125 m or 1% (most stringent)	2 km – 100 km	20 m – 125 m
Medium	Accuracy ≤ L/500000 between medium-scale S/C and any medium- or small-scale S/C Always ≤ 2 ms	Accuracy ≤ 1%	100 km – 2000 km	1 km – 20 km
Large	Accuracy ≤ 2 ms between a large scale S/C and any other S/C	Accuracy ≤ 1%	3000 km – 15000 km	30 km – 150 km

L = distance between S/C

8.2 RF Navigation System

For the CS mission concept, it is proposed to implement the RF navigation sensor in X-band with the purpose to re-use the on-board TTC omni-directional antenna and TWTA amplifier. This solution is conducive to reducing the mass and the power demand of the RF function.

The maximum operational distance range of the RF equipment is assumed to be ~1000 km, which is mainly driven by power considerations. The shared communications equipment with the downlink function enables the availability of relatively high power levels (~12 W) for the ranging signals.

The satellites exchange messages bearing a coded date of emission, identification of emitter and computed distance to the other satellites. The data exchange between satellites is monitored through a Time Division Multiple Access (TDMA) pattern, where each S/C emits during its dedicated slot.

The inter-S/C distances in the small scale are always compliant with the RF working range (Figure 15), while the inter-S/C distances in the medium scale (Figure 16) and between small and medium scale S/C fulfil the distance constraint around apogee passes. Most probably the S/C will be downloading data to the ground station(s) during the portion of the orbit close to perigee, so as to take advantage of the reduced distance with respect to the Earth, which enables higher data rates and better link budgets. The shared use of the TWTA amplifier and the limitation in terms of available power prevent the S/C from performing inter-satellite ranging and communications with the ground at the same time. In addition, the CS formations in the three scales have been designed to provide optimum configurations close to apogee, where the best navigation accuracy should be achieved using the RF function, so as to guarantee the best timing and inter-S/C distance knowledge for science data collection and processing.

8.3 Performance Analysis

This analysis is aimed at assessing the navigation performance (in terms of inter-S/C distance accuracy) provided by a RF-based relative navigation system and to compare it with the CS requirements (Table 9). The reference CS orbit (~1.4 Re x 25 Re x 14°) has been considered for this analysis.

The core of the analysis consists in propagating the relative position and velocity errors between two S/C starting from the time when the RF relative navigation system is switched off to save power, thus accounting for the **RF duty cycle**. Two main simulation scenarios have been addressed for this **dispersion analysis**, depending on the assumptions in terms of initial relative position and relative velocity uncertainties. The propagation of the initial uncertainties is performed over an interval of one or two hours to assess the impact on the achievable accuracy.

The results obtained in terms of relative distance error at the end of the propagation time are summarised in Table 10. The navigation requirements have been drawn from Table 9, taking into account that the small-scale and the medium-scale formations have been designed for reference inter-S/C distances of respectively 10 km and 1000 km.

Table 10: RF Duty Cycle and Achievable Navigation Accuracy

Initial Uncertainties		Propagation Time [hr]	Max Distance Error [m]	Compliance with Requirements	
Position [m]	Velocity [cm/s]			Small-Scale S/C (Req: 100 m)	Medium-Medium & Medium-Small Scale S/C (Req: 10 km)
5	1	1	94	**YES**	**YES**
5	1	2	225	NO	**YES**
50	50	1	4155	NO	**YES**
50	50	2	10559	NO	NO

Small initial position and velocity uncertainties and a propagation time of 1 hour (corresponding to the frequency of the RF measurements) allow fulfilling both the small-scale and the medium-scale navigation requirements. The RF equipment shall be operated during **3 min every hour** to provide 20 adjacent inter-distance measurements for the 6 S/C that are the potential users of the ranging system, leading to a maximum 5% time loss for the downlink function.

9 GROUND LINKS AND DATA DOWNLOAD

The volume of science data collected and the ground station (GS) delivery intervals are driving factors to determine the on-board mass memory and to define the necessary data transmission rate.

Scenarios with **one or two ground stations** (respectively Maspalomas and Perth) **with 15-m X-band antennas** have been considered to assess the communications and data download performances.

9.1 Ground Station Contact Analysis

The ground station contact analysis is aimed at evaluating the **timeline of GS contacts** and their durations. Maspalomas guarantees approximately one contact per day with the CS constellation, with an average contact duration of ~11 hr and a maximum visibility gap of ~35 hr. If two GS are used, the GS contact time can be improved considerably (~18 hr/day) and the visibility gap is reduced to ~10 hr, thus allowing an increase of the data return. In particular, the selection of two GS located almost at the antipodes (such as Maspalomas and Perth) enables links with the S/C at perigee passes independently of the perigee position with respect to the Earth rotation.

The **TDMA** technique has been selected to implement the communications of the S/C with the GS, meaning that only one frequency is used by all satellites and one S/C at a time is allowed to transmit to the GS. The impact of the TDMA technique consists mainly of the need to allocate a time interval for data download preparation. It has been estimated that ~5 min should be allocated per S/C to set up the link with the GS, summing up to ~50 min for the whole CS constellation per GS contact. The time interval for data download preparation covers:

❑ Time needed to re-point the GS antenna from one satellite to another.

❑ Time needed to lock the receivers on the carrier signal (both on S/C and on ground).

The direct effect of the download set-up time is to shorten the effective GS contact intervals per S/C and for the whole constellation. The application of the set-up constraint reduces the average and the total contact duration, and increases the maximum gap without GS contact, as highlighted in Table 11.

Table 11: Contact Statistics with Maspalomas and Perth

GS	Contacts per orbit	Average Contact [hrs/day]	Max Gap [hrs]
No Comms. Set-up Time			
Maspalomas	2.84	11.44	34.43
Perth	2.85	9.37	36.81
Maspalomas + Perth	3.33	19.10	9.38
Comms. Set-up Time ~50 min (whole constellation)			
Maspalomas	2.77	10.60	35.27
Perth	2.75	8.54	37.63
Maspalomas + Perth	3.16	18.13	10.22

The **S/C-GS link geometry** is characterised by relevant S/C-GS link angles, such as:

❑ Angle between the S/C-GS link and the S/C spin axis perpendicular to the ecliptic plane (Figure 22), which has an important impact on the S/C antenna field of view design.

❑ Angle between the S/C-GS link and GS-Sun direction. This angle conveys the information on the illumination condition with respect to the link,

so as to assess whether the close proximity of the Sun may lead to the loss of part of the contact.

❑ Evolution of the S/C range, elevation and azimuth angles during GS passes (Figure 23), which drive the GS antenna tracking capabilities needed.
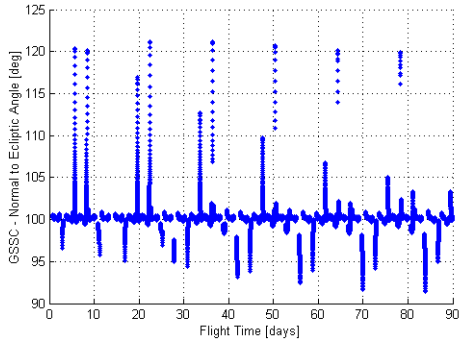

Figure 22: Angle between the S/C-Maspalomas Link and the S/C Spin Axis Normal to the Ecliptic Plane
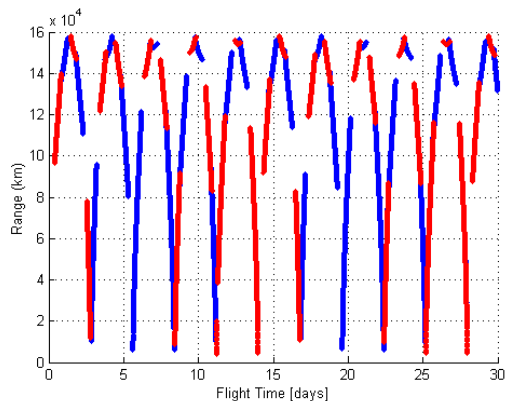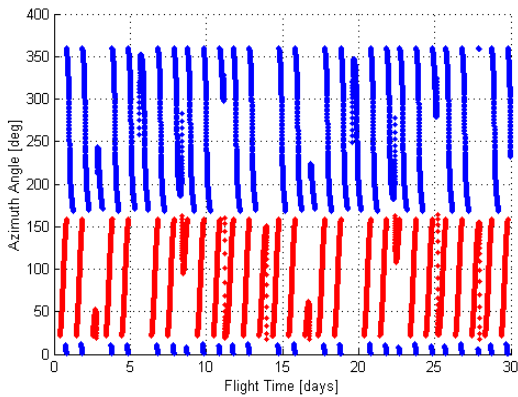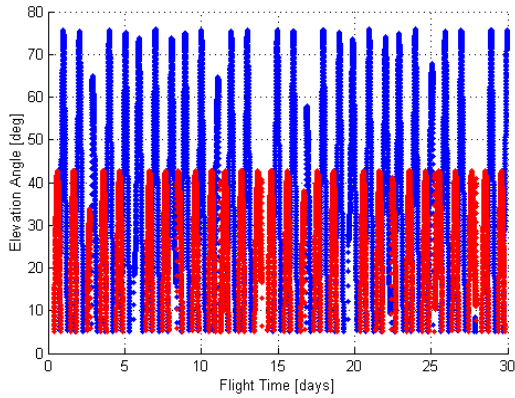

Figure 23: S/C Range, Elevation and Azimuth Angles during Maspalomas (blue) and Perh (red) Passes (min elevation = 5°)

The time of the day during the S/C passes over the GS has been analysed to assess the impact in terms of ground operations. Three time intervals of 8 hrs each are highlighted in Figure 24 to account for a typical 8-hr shift in manpower allocation for S/C operations. The time shift between Maspalomas and Perth contacts, due to the different locations of the two GS, allows coverage at almost all contact times per day.
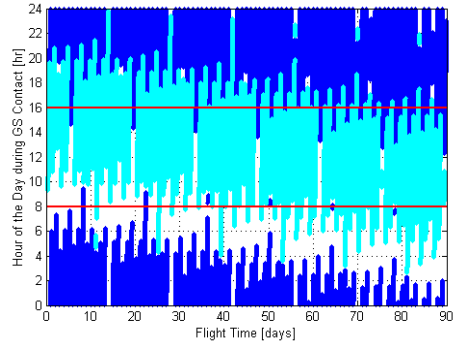

Figure 24: Time of the Day during Maspalomas (blue) and Perth (cyan) Contacts

The outcome of the analysis points out that the time of the day during GS contact undergoes a shift during the year, covering all the possible hours. Thus, the GS should be operated continuously to receive data using all the GS visibility windows, in order to avoid any loss of data. This consideration fosters the **automation of the data reception operations** as an interesting option, since a 24-hr/day manpower allocation for GS operations would result in a too expensive overhead for a mission concept, like the CS one, which aims at a low-cost approach as far as possible. To guarantee the operational feasibility of an automated data download, an enabling technology may be to store data on board for the time needed (on day-shift basis) by the ground segment to check that a bulk of data of the expected size has been received correctly, thus minimising the risk of data loss.

9.2 Data Download Capability

The data generation rates play a key role in the design of the overall CS communications architecture, in particular to size the data download capacity. Table 12 outlines the nominal data generation rate requirements with an embedded 100-200% overhead for high data rate products (i.e. data generated in burst mode at 2.5 Mbps for all S/C when their instruments are operated simultaneously).

Since the orbit of the CS S/C is highly eccentric, a possible solution to increase the data return would be to tune the data download rate as function of the range to the GS, leading to an approach based on a **variable data rate**. The benefits of this technique are drawn from the possibility to take the maximum advantage of the S/C perigee passage, where the relative distance with respect to the GS is minimum and both data download rates and link budgets can be maximised.

Table 12: Required Data Rates for the whole CS Constellation

Science Data Generation		Nominal	Goal
Nominal Data Rate	kbps	400	600
High Data Rate Products	%	200	300
Total Data Rate	kbps	800	1800
Daily Data Volume	Gbit/day	69.12	155.52
Localization/synchronization and Housekeeping Data Generation			
Data Rate	kbps	40	
Daily Data Volume	Gbit/day	3.456	
Total			
Total Daily Data Volume	Gbit/day	72.576	158.976
Average Data Generation	kbps	840	1840
Number of S/C		10	
Average Data generated per S/C	kbps	84	180

The data rate at the apogee altitude has been assumed as the sizing parameter for the data flow analysis, with values between 700 kbps and 900 kbps.

The maximum achievable data download rate is limited by the available bandwidth for each RF channel at the frequency band used. For the X-band from 8450 to 8500 MHz, the available bandwidth for each RF channel is 10 MHz. Within this bandwidth, the proposed communication system design (X-Band, 12-W transmission power and Turbo Coding ½), allows a **maximum data rate of 6 Mbps**. This data rate constraint has a major impact for GS visibility intervals that occur close to perigee, where the variable data download rate tends to be higher due to the reduced distance between the S/C and the GS.

Dedicated simulations have been carried out to assess the impact of the **TDMA** constraint on the download time allocated per S/C: in the TDMA strategy, each S/C can use only 1/10 of each GS contact interval. This constraint reduces the data volume that can be downloaded by each satellite during a GS contact interval, thus increasing the on-board storage capacity required for queuing download data.

Figure 25 presents the results in terms of queuing download data stored per S/C over a representative 1-month simulation period, assuming that only Maspalomas GS is available for data retrieving. The S/C shall have an on-board memory capacity to store housekeeping and scientific downlink data (global data stream) when GS communication is not possible.
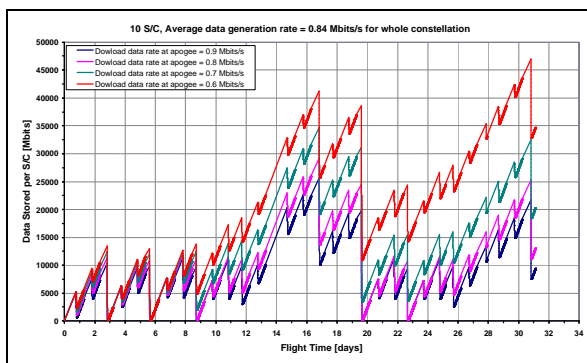

Figure 25: Queuing Download Data Stored as a function of time (1-month simulation with TDMA and max data rates < 6 Mbps)

In addition, the **on-board memory storage requirements for payload data** for the whole constellation of 10 S/C define the upper envelope for sizing purposes [2]. The S/C constellation shall have an onboard data storage capability to store all science data produced during two orbital periods with 50% of time in continuous burst mode (2.5 Mbps). The required storage capability for the 1.4Re x 25Re orbit (2.81-day period) for the whole constellation is 607.5 Gbits (1215 Gbits for a goal of 100% in burst mode over two orbits). This requirement translates into approximately 61 Gbits of storage capacity per S/C.

10   CONCLUSIONS

The Cross-Scale TRS has yielded a comprehensive system-level design of a challenging mission concept for multi-dimensional investigation of space plasma phenomena in the vicinity of the Earth.

To fulfil the scientific mission objectives, a constellation of 10 plasma-instrumented S/C has been proposed, flying in loose formations around a highly elliptical Earth orbit and sampling three characteristic plasma scale distances simultaneously. The driving issues addressed in the study encompass the reference orbit design, the number of S/C, the transfer and deployment strategy, as well the formation design, maintenance and reconfigurations.

Last but not least, operational streamlining and effective communication architecture are paramount to handle such a considerable number of S/C and instruments. In particular, the volume of data collected and the ground delivery intervals are driving factors to determine the S/C on-board mass memory and to define the necessary data transmission rate. Inter-S/C localization/synchronization is another key element to achieve the required timing and distance accuracy, so as to correlate science data from various S/C.

The outcome of the study has outlined a technically feasible mission architecture and has identified the enabling technologies to embark on a demanding plasma physics mission that would improve to a large extent the knowledge of the space plasma processes at different relevant spatial and temporal scales.

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