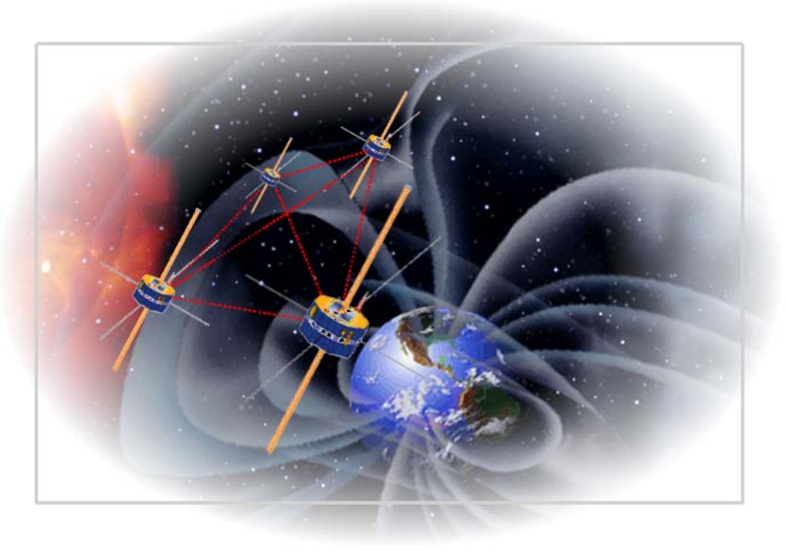
SUMMARY REPORT



Cross-Scale Technology Reference Study

Planetary Exploration Studies Section (SCI-PAP)
Advanced Studies and Technology Preparation Division (SCI-PA), Science Projects Department

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In Memoriam: Marcel van den Berg

This Cross-Scale study summary document is dedicated to Marcel van den Berg, ESA study manager of the Cross-Scale Technology Reference Study, who unexpectedly passed away at the age of 38 on 2.May 2007. Marcel was the driving force behind the Cross-Scale study and largely prepared this summary. Marcel was a rarity in being both a top rate scientist and engineer, with additional personal qualities of being down to Earth and excellent company. He will be deeply missed by all of us.

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Mission Summary

Key scientific objectives	Detailed in situ multi-spacecraft exploration of universal plasma phenomena occurring in near-Earth space, in particular: o Shock processes o Reconnection o Turbulence These processes involve structured, 3 dimensional, time-varying interactions across multiple length scales (electron, ions and magneto-hydrodynamic fluid)
Strawman reference payloads assumed for this study	Combination of the following instruments: DC and AC magnetometers, 2D and 3D electric field instruments (wire booms, axial antennas), electron density sounder, electron and ion electrostatic analyzers, (energetic) ion composition analyzers, high energy particle detectors.
Transfer to operational orbit	• Launch by Soyuz-Fregat 2-1B from Kourou (all days except Feb/Mar/Apr/Aug/Sept/Oct) • 3026 kg into GTO or 3996 kg into 180 km × 13786 km (baseline) • Transfer to operational orbit by dedicated dispenser-like transfer vehicle
Initial operational orbit (no active orbit control)	Perigee altitude: 500 km – 10 R_E (baseline 2500 km (=1.4 R_E)) Apogee altitude: 24 R_E Inclination: 14°
Radiation environment	64 krad (4 mm Al shielding, 1.4 R_E perigee)
Operational lifetime	• 1 year commissioning and early science operations • 2 years science operational phase • 2 years extended science operation

S/C Modules	**Transfer stage**	**Electron-scale**	**Ion-scale**	**Fluid-scale**
Number of S/C	1	2	4	4
Spacecraft separation	N/A	2 – 100 km	50 – 2,000 km	3,000 – 15,000 km
Stabilization	3-axis	15 rpm	15 rpm	15 rpm
S/C Δv requirements (including ~100 m/s for space debris mitigation requirements)	1.49 km/s (injection orbit 180 km × 13,786 km)	114 m/s	223 m/s	538 m/s
Design lifetime	several weeks	5 year	5 year	5 year
Platform dry mass (excl. P/L)	490 kg	115 kg	115 kg	109 kg
Model P/L mass / power	-	42 kg / 34 W	35 kg / 31 W	13 kg / 10 W
Total mass (incl. propellant and 20% system margin)	1992 kg	196 kg	194 kg	181 kg
Maximum power	965 W	170 W	170 W	170 W
Telemetry band	S-band	X-band	X-band	X-band
Continuous compressed science data bit rate per S/C	-	~120 kbps	~100 kbps	~40 kbps

Key mission drivers	• Multiple S/C assembly, integration and verification (AIV) • Multiple S/C mission operations • Requirements on space debris mitigation • Spacecraft / constellation reliability • Payload accommodation/interface requirements • On-board data storage (to minimize data download requirements)
Key technological challenges	• Synchronization / localization • Memory (size & space qualification) • Power control unit • Transponder • Thrusters

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1 INTRODUCTION

1.1 Technology Reference Studies

Technology Reference Studies (TRS) have been introduced by ESA's Science Payload and Advanced Concept Studies (SCI-A) several years before the Cosmic Vision call for proposals to provide a focus on the development of strategic technologies that are of likely relevance to potential future scientific missions (see also http://sci.esa.int/concepts). The focus on technology was achieved through the study of several technologically demanding and scientifically interesting mission concepts, which are not part of the ESA science programme. The TRS subsequently acted as a reference for possible future technology development activities.

1.2 Mission scenario

The Cross-Scale Technology Reference Study (CS TRS) is a mission concept study for the investigation of fundamental space plasma processes that involve non-linear coupling across multiple length scales. Plasma processes play a significant role in many astrophysical objects as well as in solar-terrestrial physics, but most of the important plasma phenomena (magnetic reconnection, non-linear dynamics and turbulence, wave-particle interaction, plasma acceleration and heating) are still poorly understood. These phenomena involve structured, time-varying multiscale interactions, which can be excellently studied in the Earth's bow shock as well as in the magnetospheric tail and tail lobes. Though Cluster II is providing many new insights in space plasma interactions, it was not designed to study interaction on small scales, nor the interaction between different scales (as the four spacecraft cannot perform multipoint investigations at different length scales at the same time).

The key objective of Cross-Scale TRS is to study the coupling processes between electrons, ions and waves in the magnetospheric plasma. In order to achieve this, between 8-12 spacecraft will be required, flying in loose formation around the Earth, to investigate simultaneously plasma processes and interaction in 3-D at three scale distances: electrons (~10 km), ions (~1,000 km), magnetospheric fluid (~6,000 km).

1.3 Cross-Scale TRS summary report

This Cross-Scale TRS report provides a summary of the system design study carried out by industry (Deimos Space S.L. as prime, Thales Alenia Space and ONERA as sub-contractors) under ESA contract between May 2006 until June 2007. The first phase of the system design study consisted of a mission architecture trade. Key trades on the mission architecture included orbits, number of spacecraft, payload suites, deployment strategy, and orbit optimization. The second phase consisted of a assessment of the spacecraft design, ground segment requirements and programmatics.

1.4 Applicable documents

The status report is one of the documents that constitute the complete mission profile for the Cross-Scale TRS. The current list of applicable mission documents is (available from http://sci.esa.int/science-e/www/object/index.cfm?fobjectid=38982):

CS Mission Objectives	SCI-A/2005/072/CS/MvdB	[AD_MOD]
CS Mission Requirements	SCI-A/2005/073/CS/MvdB	[AD_MRD]
CS Payload resources	SCI-A/2005/077/CS/MvdB	[AD_PLR]

2 MISSION OBJECTIVES

The objective of the Cross-Scale Technology Reference Study (CS-TRS) is to:

- ***Establish a feasible mission profile for a cost-efficient investigation in near-Earth space of fundamental plasma processes that involve coupling across multiple length scales***

More specifically, the study profile shall:

- ***Optimize the scientific return of in-situ multi-dimensional space plasma exploration***
- ***Establish the feasibility of a cost-efficient mission concept***
- ***Include a technology development and demonstration plan for critical and enabling technologies***

In the following two sections the mission statement is elaborated into primary objectives and trade-off priorities.

2.1 Primary objectives

1.1 The CS-TRS shall perform an in-situ multidimensional scientific exploration of universal plasma phenomena occurring in near-Earth space
   a) Three length scales shall be explored at the same time
   b) With a constellation of up to twelve S/C
   c) With at least two S/C for the smallest length scale

1.2 The CS-TRS S/C constellation shall visit at least the following relevant regions in near-Earth space where the scientifically most interesting plasma processes occur
   a) Bow shock
   b) Magnetosheath
   c) Magnetopause and tail current sheet (reconnection regions)

1.3 The CS S/C constellation shall be in an optimized spatial configuration to measure multiple scale plasma phenomena when visiting the regions mentioned in 1.2. When multiple regions of interest are visited within one orbit, the spatial configuration shall be optimized for at least one of these regions

1.4 The relative timing of science data between any two S/C shall be retrievable

1.5 The relative position of each S/C in the constellation shall be known

1.6 The acquired science data shall be relayed to Earth

2.2 Trade-off priorities

The trade-off priorities for the mission concept study are:

1: Visit the regions of interest
2: Maximization of number of spacecraft
3: Maximization of payload resources
4: Optimization of operational orbit

The difference between the first and the last item is that the first does not take into account aspects such as constellation configuration or time spent in the regions of interest, spacecraft velocities in the regions of interest, or statistical sampling variation in the regions of interest.

3 MISSION REQUIREMENTS

This section provides a summary of the key requirements that the Cross-Scale TRS should fulfil. A more complete requirement overview is provided in the mission requirements document (SCI-A/2005/073/CS/MvdB). It should be noted that it is not the objective of the Technology Reference Studies to design a mission concept which strictly complies with a predetermined set of given scientific, programmatic and technical requirements, but rather to identify and explore those requirements that drive the mission concept. The driving requirements are subsequently challenged and/or iterated based on the mission concept priorities given in section 2.2. The mission requirements can be roughly classified into four classes: Margin, space segment, ground segment, and operational requirements.

3.1 Margins

For the system design study, the margins listed in Table 1 are used, which are compliant with the SCI-A standard margin philosophy for assessment studies [Margin05]. The nominal mass and power bugets are determined after application of the subsystem margins. All subsystems (e.g. propellant tanks) are sized to accommodate the mass after application of the system level margin. Likewise, the power subsystem is designed and sized to provide the spacecraft required power, including system level power margin.

Table 1: Margin overview

Item		Margin
Delta-v margin		
	Accurately calculated trajectory and orbital manoeuvres (including e.g. gravity losses)	5%
	Estimated orbital maintenance and attitude control manoeuvres	100%
Subsystem mass margin		
	Off-the-shelf equipment	5%
	Off-the-shelf equipment requiring minor modifications	10%
	New designs/major modifications	20%
Power subsystem margin		
	Off-the-shelf equipment	5%
	Off-the-shelf equipment requiring minor modifications	10%
	New designs/major modifications	20%
Data processing		
	On-board memory capacity margin	50%
	Processing peak capacity margin	50%
Communications		
	Communication link	3 dB
	Telecommand and telemetry data rates	3 dB
System level		
	System level mass margin	at least 20%
	System level power margin	at least 20%

3.2 Space segment requirements

3.2.1 LIFETIME

The design lifetime for the Cross-Scale TRS spacecraft shall be at **least five years**:

0.5 – 1 year for spacecraft deployment and commissioning as well as payload commissioning and calibration
2 – 2.5 year for nominal science acquisition
> 2 year for science acquisition extension

The commissioning and in-orbit calibration period is estimated to take between six months and one year. This is strongly dependent on the number and complexity of the spacecraft and instruments as well as the commissioning and calibration strategy (e.g. parallel or sequential, autonomous or ground controlled, accuracy of instrument ground calibration, number of operational modes for spacecraft and instruments).

All spacecraft are designed and sized from launch until the end of the extended operational lifetime (e.g. reliability, power system sizing, radiation tolerance, consumables).

3.2.2 SPACE DEBRIS MITIGATION

In the near future, all missions launched under responsibility of ESA will have to comply with requirements for "Space Debris Mitigation." Though the exact requirements have not yet been established, there already exist guidelines in the "European Code of Conduct for Space Debris Mitigation" [CoC04]. In particular, it is stated that any S/C (or transfer vehicle) shall not enter the "LEO protected zone" and the "GEO protected zone" after mission completion. The LEO and GEO protection regions are graphically shown in Figure 1, where the geostationary altitude Z_{GEO} is equal to 35,786 km (or 5.6 R_E).

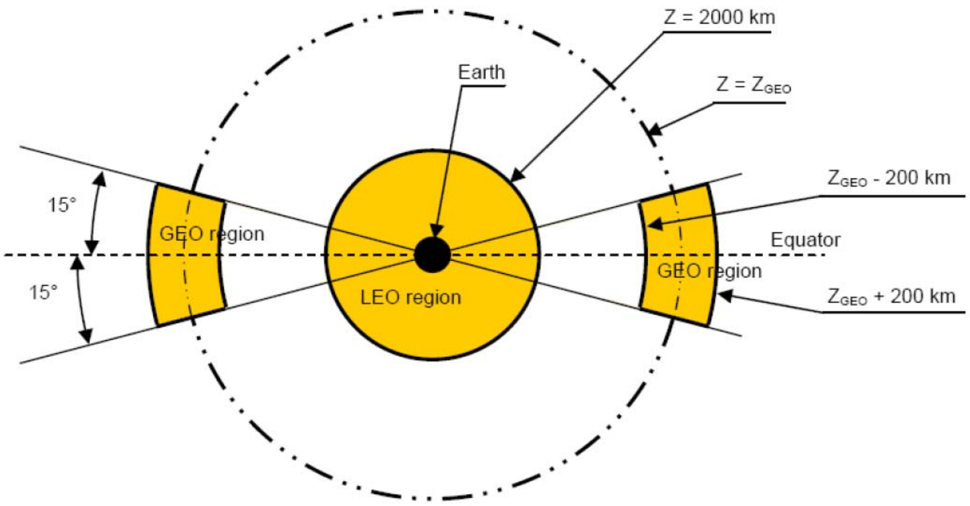

Figure 1: Protected space regions (from [CoC04])

For the Cross-Scale mission concept, this implies that end-of-life measures (re-entry, natural de-orbit, parking orbit) are required if the perigee of the operational orbit is lower than the GEO protected region.

3.2.3 OPERATIONAL ORBIT

Figure 2 shows a picture of the Earth's magnetosphere with the main regions of interest: The bow-shock, the magnetopause, the magnetosheath and the neutral sheet. The apogee is set to 25 R_E to ensure regular seasonal visiting of the magnetotail. As the Earth moves around the Sun, the inertial orbit rotates with respect to the magnetosphere, so that the apogee position essentially crosses all regions of interest in a single year.

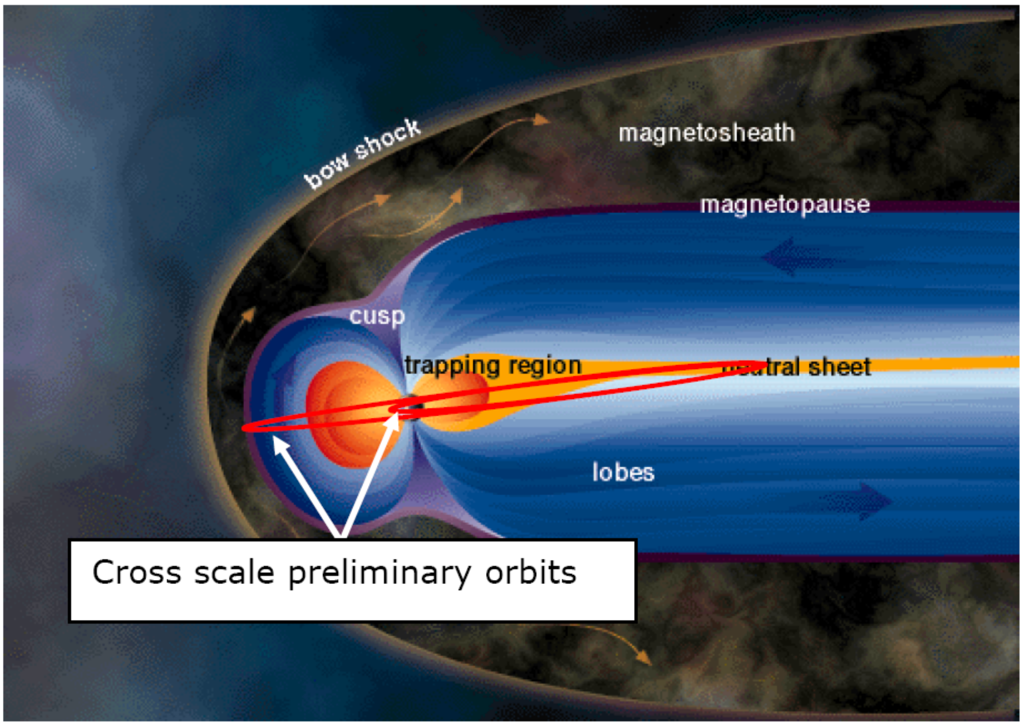

Figure 2: The Earth's magnetosphere with two typical Cross-scale orbits (high and low perigee)

The selection of the perigee can be traded between minimization of the propellant requirements (low perigee altitude) and the possibility to perform magnetopause skimming (~10 R_E). The other orbital parameters (inclination, argument of perigee and Right Ascension of Ascending Node (RAAN)) are determined by the seasonal dependence of the neutral sheet inclination as well as orbital stability considerations.

3.2.4 CONSTELLATION

The Cross-Scale TRS concept comprises a constellation of 8 to 12 spacecraft, flying in loose formation around the Earth. Each plasma length scale (electron, ion, fluid) is sampled by up to four spacecraft that achieves its optimal tetrahedron configuration at true anomalies near apogee. At the optimization points, the centres of each of three spacecraft tetrahedron configurations should approximately coincide. The optimal configuration with 12 spacecraft is shown in Figure 3.

In case the perigee encounters regions of interest as well (i.e. for perigee of 10 R_E), it shall be assessed whether it is possible to optimize the constellation for both perigee and apogee.

During the complete mission lifetime, the spacecraft scale distances shall be sampled over the following distances:

a) 5 to 20 changes in the electron scale distance, establishing S/C distances of 2 to 100 km
b) 3 to 5 changes in the ion scale distance, establishing S/C distances of 50 to 2,000 km
c) The fluid scale configuration shall start at a S/C distance of 6,000 km, then down to 3,000 km, up to 15,000 km and again down to 6,000

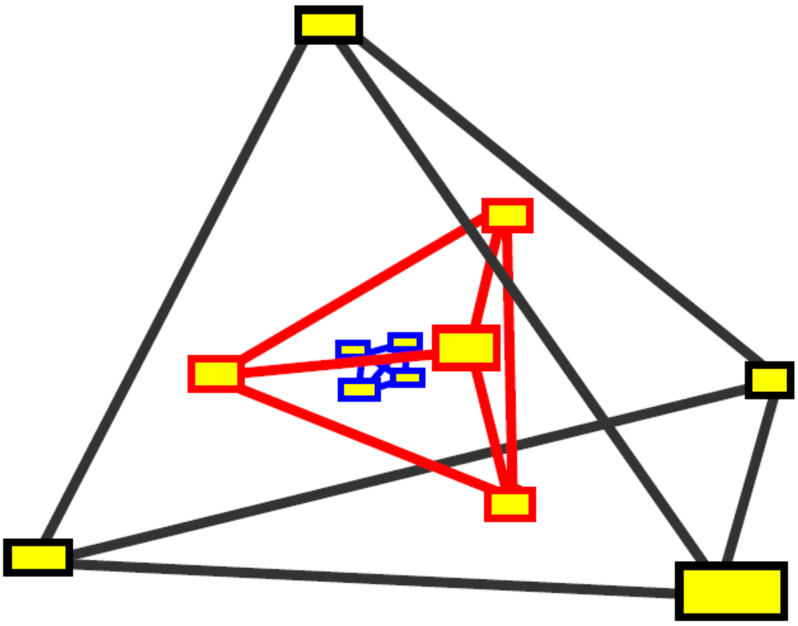

Figure 3: Schematic of optimized spacecraft configuration

The objective is to design a robust spacecraft constellation that requires as little control as possible in order to optimize the science acquisition time (instruments are switched off during thruster operation), to reduce the operational complexity as well as to increase the constellation reliability (if only one or two orbital control manoeuvres per year are necessary, the spacecraft have less risk of e.g. collision when in survival mode or degraded operation).

3.2.5 SPACECRAFT ATTITUDE

For the accommodation and field-of-view requirements of the plasma instruments, the spacecraft shall be spin-stabilized with the spin axis maintained at 4 ± 1° towards the ecliptic north pole. The spinning rate shall be in the range from 15 to 60 rpm.

3.2.6 SPACECRAFT LOCALIZATION AND SYNCHRONIZATION

To investigate complex plasma structures that vary both in time and space, the relative position and time of the spacecraft is required. This particularly applies to the electron scale spacecraft, and to a lesser degree to the ion scale spacecraft. Table 2 shows the requirements assumed for the Cross-Scale TRS system design.

Table 2: Spacecraft localization and synchronization requirements

Scale	Timing accuracy	Localization accuracy
Electron (2 to 100 km)	0.25 ms	125 m
Ion (50 to 2,000 km)	0.25 to 2 ms	1% of distance
Fluid (3,000 to 15,000 km)	2 ms	1% of distance

For the relative localization and synchronization two options exist; determination from ground or a dedicated inter-S/C ranging system. The required timing accuracies can be relatively easy fulfilled by the ground stations and stable on-board clocks. Also for localization accuracies above approximately 5 km, the ground stations can be used. Consequently, only for localization requirements for S/C distances below ~500 km (electron scale spacecraft and ion scale spacecraft when at close separation), a dedicated inter-S/C ranging system is required.

3.2.7 COMMUNICATION

The spacecraft communication subsystems should comply with the customary ESA reliability requirements, such as hot redundancy for the receiver and a cold redundancy for the transmitter. Additionally, all spacecraft shall be capable to receive and acknowledge commands from ground at all times, independent of operational mode, attitude or location on the orbit.

The communication architecture should be sized for down-link capability of an on-orbit average continuously compressed science data rate of at least 800 kbps for the complete constellation. This science data rate comprises nominal science data recorded over a complete orbit as well as burst mode science data in order to analyze in detail interesting events (e.g. shock crossings, neutral sheet crossings, reconnection).

For the selection of interesting events, a straightforward procedure is baselined: The spacecraft should have sufficient memory to store all high-resolution data until the nominal (or a fraction of the nominal) data has been analyzed on-ground. At the next communication window, the selected time slots of the high-resolution data will be down-linked after which the memory is cleared for storing new science data. The requirements on on-board data memory will have to be traded against the possible operational complexity of the introduction of timeliness requirements on the science data. On-board-triggering has been considered, but this would require a data-link between all spacecraft as well as sophisticated triggering algorithms.

3.3 Programmatic requirements

3.3.1 TECHNOLOGY HORIZON

For the Cross-Scale Technology Reference Study, the technology development horizon had been limited to less than 5 years. All selected subsystems should currently already have been validated at component or breadboard level in a laboratory environment (ESA TRL 4, see Table 41).

3.3.2 RELIABILITY

No explicit requirements on the reliability of the complete constellation have been imposed, as this would likely drive the mission concept. Instead, typical (non-driving) spacecraft and ground segment reliability requirements are used.

The individual spacecraft shall be designed to have an average science data return probability of 98% over the mission lifetime, while the ground segment shall guarantee the timeliness delivery of acquired science data with a reliability of 95%.

In order to reduce the operational complexity (see next section), all spacecraft shall be able to survive in safe mode without ground intervention for a duration of at least 30 days.

3.3.3 OPERATIONS

The operational complexity shall be minimized wherever possible. Strategies for operational complexity reductions include: Identical spacecraft, fully automated communication procedure, limited orbit control manoeuvres, robust orbital design, on-board autonomy and autonomous recovery (spacecraft and payload), large on-board memory, effective error detection correction methods, limited operational modes for spacecraft and instruments.

4 MISSION ANALYSIS

This section provides an overview of the mission analysis for several orbits of interest to the Cross-Scale TRS. The mission analysis in this section provides the building blocks for the mission architecture trade in section 6.

4.1 Launch vehicle

A single launch with a Soyuz-Fregat 2-1B from Kourou has been selected as the baseline for the Cross-Scale TRS because it is a cost-efficient and highly reliable launch vehicle. Figure 4 shows the volumetric constraints of the ST-fairing, which will be available from Kourou [Soyuz06]. Multiple (cheaper) launchers have been considered, but this brings up issues of phasing (the spacecraft will have to meet to form the constellation) and additional operational complexity (multiple launch campaigns).

Figure 5 shows the typical launch sequence for launch and transfer to a highly elliptical orbit. After launch of the Soyuz-Fregat into an initial low altitude parking orbit at the required inclination, the Fregat stage is fired to raise the apogee to the launcher injection orbit. At this orbit the spacecraft is separated from the Fregat. The final target orbit is reached after further apogee and perigee raising manoeuvres by the propulsion system of the spacecraft. Though not necessarily mass-efficient, it is generally possible to use the Fregat to achieve the final apogee altitude. However, due to launcher constraints, it is not possible to use the Soyuz-Fregat to perform a perigee raising manoeuvre.

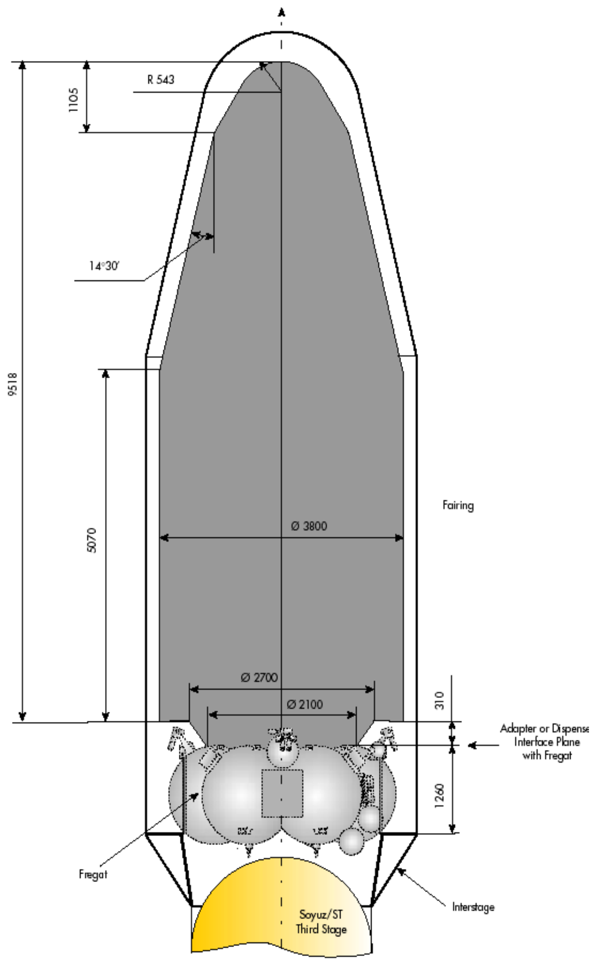

Figure 4: Volumetric Constraints for the Soyuz-Fregat ST-fairing (from [Soyuz01]).

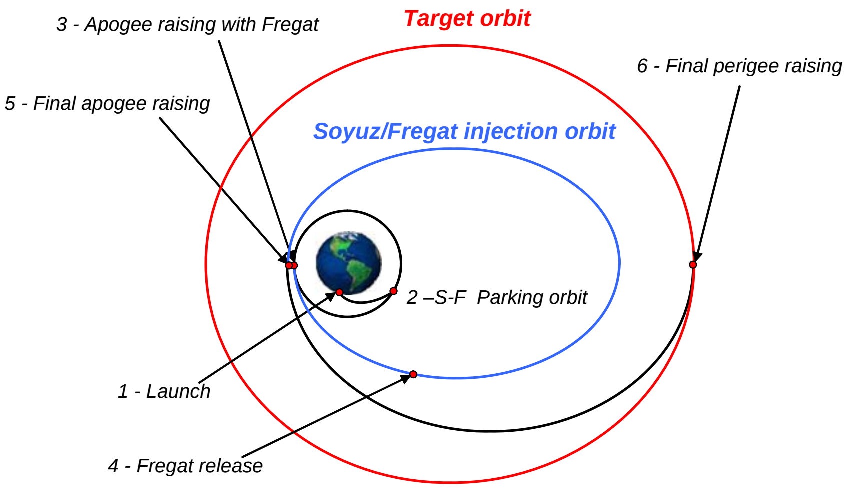

Figure 5: Standard launch and transfer sequence for highly-elliptical Earth orbits.

For the Cross-Scale TRS, a conservative launch performance of 3026 kg (including launch adapter) into GTO (180 km × 35,786 km) has been assumed. The performance estimates for launch into an orbit with apogee lower than GTO are shown in Table 3.

Table 3: Soyuz-Fregat launch performance estimates as a function of apogee (inclination 14°). Note: mass of the launch adapter needs to be subtracted; apogee is given in respect to centre of Earth (and not as surface altitude).

Launch into:				Mass in orbit	comment
apogee radius		apogee altitude (km)			
(km)	(Re)	(km)	(Re)	(kg)	
42164	6.61	35786	5.61	**3026**	**GTO**
40164	6.30	33786	5.30	3070	-
38164	5.98	31786	4.98	3119	-
36164	5.67	29786	4.67	3174	-
34164	5.36	27786	4.36	3234	-
32164	5.04	25786	4.04	3303	-
30164	4.73	23786	3.73	3380	-
28164	4.42	21786	3.42	3467	-
26164	4.10	19786	3.10	3570	-
24164	3.79	17786	2.79	3689	-
22164	3.48	15786	2.48	3829	-
20164	3.16	13786	2.16	**3996**	**selected as baseline**
18164	2.85	11786	1.85	4200	-
16164	2.53	9786	1.53	4453	-
14164	2.22	7786	1.22	4777	-

4.2 Operational orbit

The orbits considered for the Cross-Scale TRS range from a perigee radius of 1.4 R_E (2500 km altitude) up to 10 R_E, and from an apogee radius of 25 R_E up to 50 R_E. Orbits with altitude lower than 2500 km become unstable against atmospheric entry. For the apogee radius a baseline of 25 R_E has been selected in order to minimize lunar perturbations.

4.2.1 TRANSFER

Table 1 shows the Δv requirements and spacecraft in orbit mass for a selected set of target orbits. The quoted Δv values include launcher dispersion correction, gravity losses and margin. For the mass performance, an apogee kick motor with a specific impulse of 325 seconds has been assumed. The initial inclination of 14° with respect to the Earth's equatorial plane is optimized for winter magnetotail visiting (initial argument of perigee of 270° and initial RAAN of 0°).

Table 4: Transfer Δv from GTO for a selected set of target orbits (assuming Isp=325 s and a 110 kg launch adapter mass).

Perigee radius (R_E)	Apogee radius (R_E)	Inclination (°)	Transfer Δv (m/s)	Spacecraft mass in orbit [SF from GTO] (kg)	Remarks
1.4	25	14	675	2360	
2	25	14	775	2285	
4	25	14	1005	2125	
10	25	14	1390	1885	
10	25	14	1215	1990	Lunar resonance transfer. Worst case launches date over the period 2015 – 2020.
10	25	90	1830	1640	Polar orbit

The polar orbit has significant mass penalty compared to near-equatorial orbits, and has therefore been discarded. As expected, near-equatorial low perigee orbits have better mass performance than high perigee orbits. This will however need to be traded against the ΔV requirements for spacecraft disposal, constellation deployment/maintenance/control, eclipse times (spacecraft mass), radiation environment (spacecraft shielding mass, reliability and instrument additional operations) as well as scientific performance (visiting statistics for regions of interest).

An interesting strategy to minimize the Δv requirements of high perigee orbits is to make use of lunar resonances for perigee raising, which for a 10 x 25 R_E orbit yields Δv reductions of 175 – 250 m/s (depending on launch year) compared to a conventional transfer (see Table 5). As the launch year is undefined, the worst case launch date (within 2015 - 2020) has been assumed for this study.

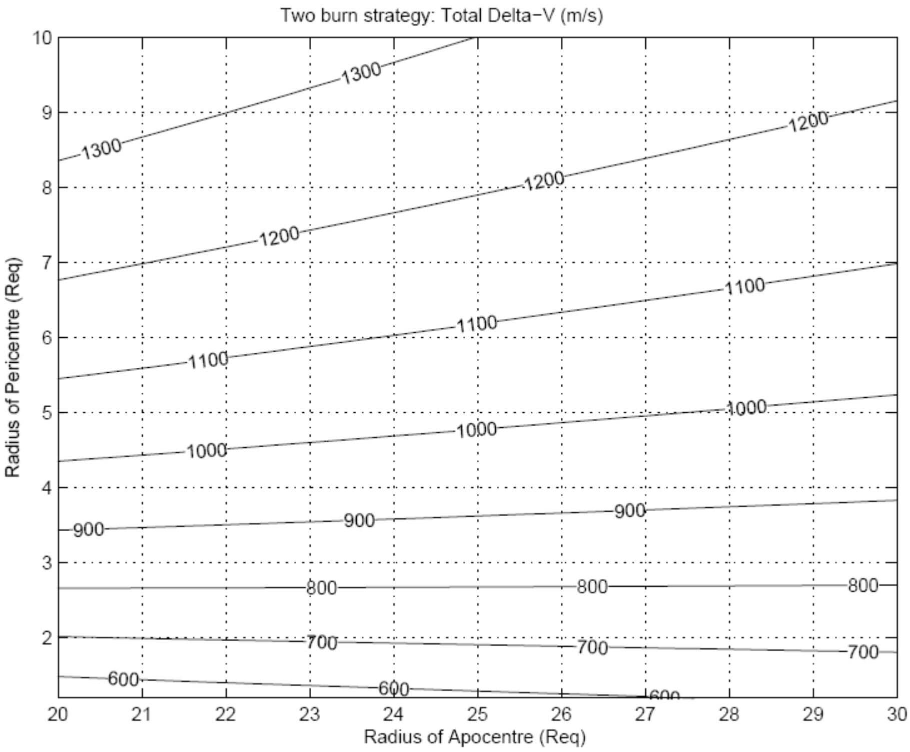

Figure 6: Two burn strategy: total Δv (m/s) for transfers from GTO to final orbits [ESOC-WP511]

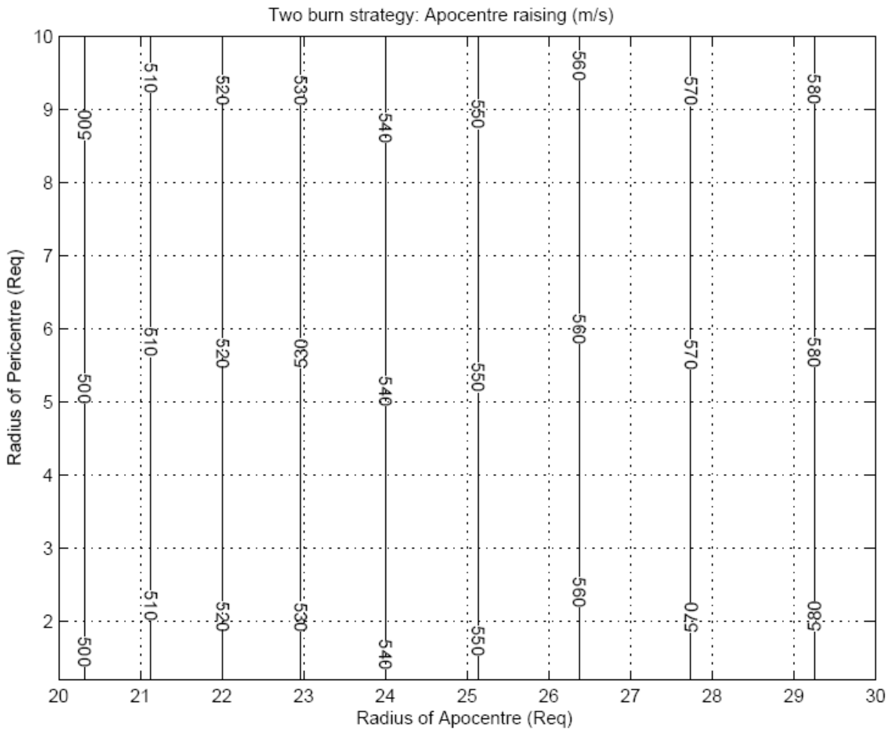

Figure 7: Two burn strategy: Δv for Apocenter raising starting from a GTO launcher insertion orbit [ESOC-WP511]

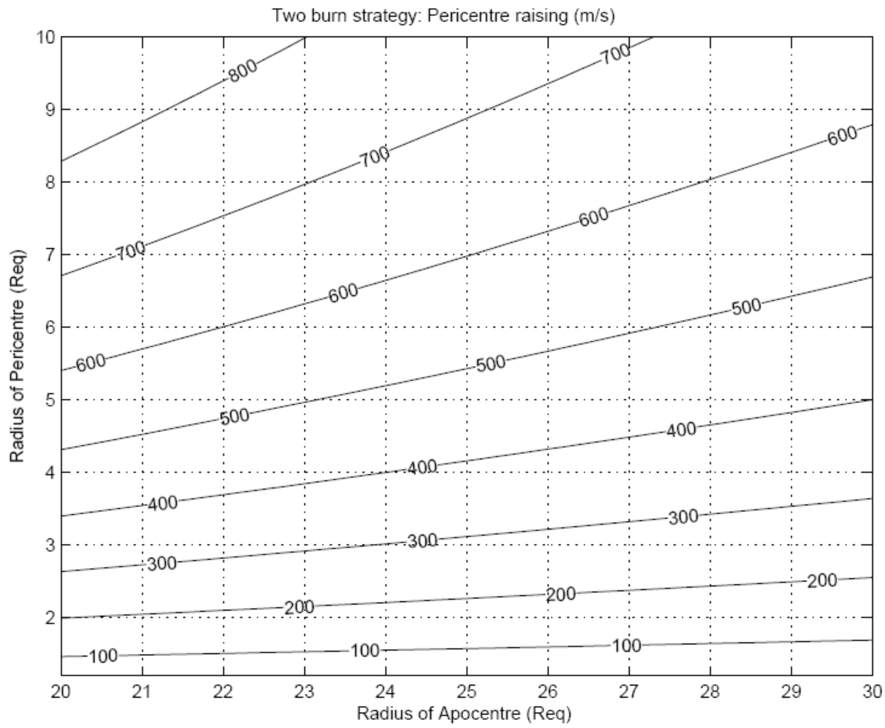

Figure 8: Two burn strategy: Δv for perigee raising (starting from GTO transfer) [ESOC-WP511]

Launch 2015 1st tail cros 2016		Launch 2016 1st tail cros 2017		Launch 2017 1st tail cros 2018		Launch 2018 1st tail cros 2019		Launch 2019 1st tail cros 2020		Final			Tail Cross
Ecl hours	DV m/s	Ecl hours	DV m/s	Ecl hours	DV m/s	Ecl hours	DV m/s	Ecl hours	DV m/s	Rpe Re	Raan deg	Aop deg	Date about
2.1	1031	2.0	1041	2.1	1062	1.9	1108	1.4	1125	10	0	300	Jan
3.0	1002	2.9	1024	2.8	1054	2.5	1101	0.6	1111	10	0	330	Feb
5.1	959	5.5	992	3.9	1024	2.9	1053	3.6	1070	10	0	0	Mar
1.8	987	2.2	947	2.4	979	2.0	1005	6.0	1019	10	0	30	Apr
0.0	968	1.2	952	2.2	968	1.6	959	2.0	958	10	0	60	May
1.8	1027	1.4	1045	0.8	1075	1.6	1093	0.0	1107	10	0	90	Jun
2.2	1040	2.1	1050	2.4	1075	1.9	1099	2.2	1119	10	0	120	Jul
3.6	1026	3.0	1042	3.6	1063	2.5	1081	3.1	1099	10	0	150	Aug
5.6	996	4.1	1022	7.7	1039	3.6	1046	6.0	1055	10	0	180	Sep
2.0	959	2.3	988	21.3	1003	2.3	1000	18.3	993	10	0	210	Oct
0.0	971	1.9	979	2.2	958	1.8	972	2.3	962	10	0	240	Nov
1.7	1022	1.1	1029	1.3	1050	1.5	1075	0.6	1115	10	0	270	Dec

Table 5: Δv and maximum eclipse duration (without eclipse mitigation strategy) for transfer from GTO to 10 R_E x 25 R_E orbit with launch in between 2015 and 2019 (shaded cases: eclipse >3.5 h) [ESOC-WP511]

4.2.2 EVOLUTION

To minimize propellant use, no orbital control is foreseen for the operational orbit. This implies that the operational orbit parameters will evolve with time due to the Earth's oblateness and lunar and solar perturbations. Detailed analysis of the orbital perturbations has shown that the main driving parameter is the initial RAAN (Ω), while the launch date and argument of perigee (ω) have only a secondary effect. An initial RAAN close to 0° yields the most stable orbits.

The orbital evolution after five years for a selected set of orbits is shown in Table 6 for a launch date of 1/1/2015. The perigee and apogee altitudes are reasonably stable for all orbits, though the relative impact of perigee altitude variation becomes larger for lower perigees. The inclination *i*, the RAAN Ω, and the argument of perigee ω generally show an increase with time.

Table 6: Summary of orbital parameter evolution after 5 years for a launch date of 1/1/2015. (*i* = inclination, Ω = Right Ascension of Ascending Node (RAAN), ω = argument of perigee)

Initial Orbital Elements					Orbital Elements after 5 years				
Perigee radius (R_E)	Apogee radius (R_E)	*i* (°)	Ω (°)	ω (°)	Δ Perigee (km)	Δ Apogee (km)	*i* (°)	Ω (°)	ω (°)
10	25	14	0	270	+1220	-1200	18	31	296
4	25	14	0	270	+1440	-1395	20	42	266
2	25	14	0	270	+1025	-1080	19	39	274
1.4	25	14	0	270	+1020	-1230	16	31	300

4.2.3 ECLIPSE ANALYSIS

Not only the Δv requirements, but also the eclipse times are of relevance for the mission architecture trade, since the batteries to power the satellites during eclipses can have a significant impact on the spacecraft mass budget. Table 7 provides an overview of the eclipse times for a selected set of orbits. Lower perigee orbits experience more eclipses per year with a larger maximum eclipse time.

Table 7: Eclipse statistics for selected target orbits assuming five year lifetime.

Initial orbit	Maximum eclipse time (h)	Duration eclipse season (days/year)
10 $R_E \times$ 25 $R_E \times$ 14°	3.4	~80
4 $R_E \times$ 25 $R_E \times$ 14°	4.7	~240
2 $R_E \times$ 25 $R_E \times$ 14°	6.6	~280
1.4 $R_E \times$ 25 $R_E \times$ 14°	8.4	~300

It should be noted that the maximum eclipse time has been calculated over the complete five years mission lifetime assuming an arbitrary launch date. In general, the maximum eclipse time increases with mission duration due to the perturbation of the orbital elements. Figure 9 shows a typical eclipse time evolution for a low-perigee orbit. Since the longest eclipse times occur rather infrequently (typically only a few times during the extended mission lifetime), a dual strategy can be envisaged: For eclipse times shorter than e.g. 4h, all spacecraft should be capable of full operation (including communication and science instruments), while for eclipse times longer than 4h, the spacecraft is allowed to switch off down-link communication and science instruments. This will optimize the power subsystem sizing, while the impact on the data return is minimal.

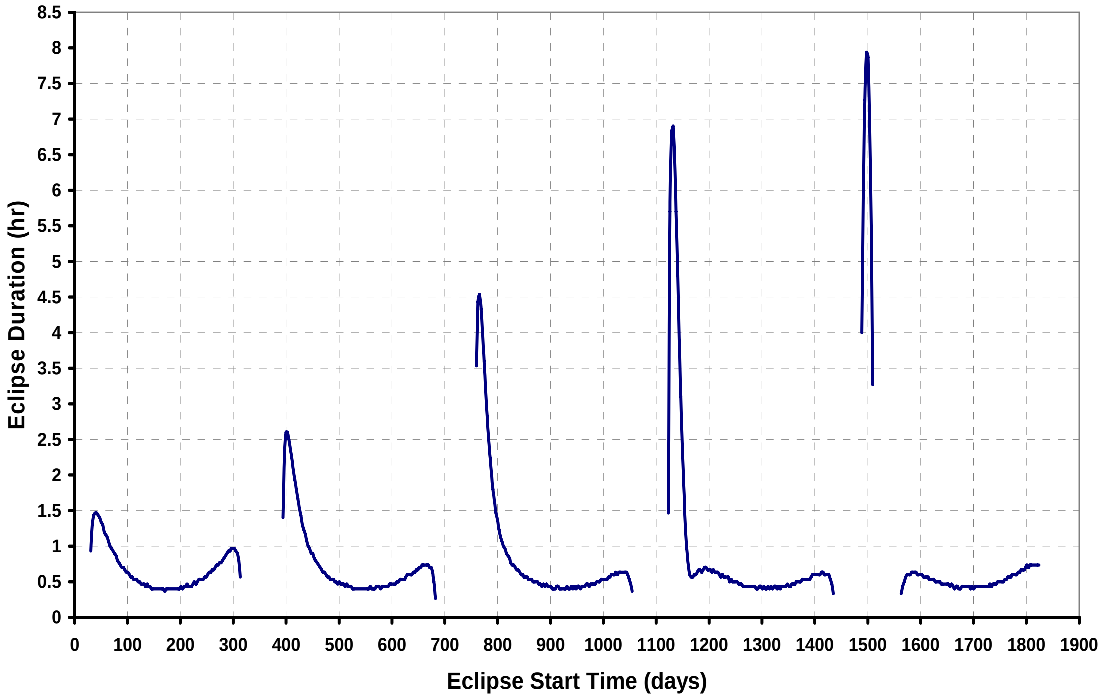

Figure 9: Eclipse duration as a function of mission lifetime for an orbit of 1.4 R_E × 25 R_E × 14° (launch 1/Jan/2015).

4.2.4 VISITING STATISTICS FOR REGIONS OF INTEREST

To assess the relative science return of the different orbits, the time spent in the magnetotail as well as a few other relevant parameters have been assessed. For the magnetotail visiting analysis, a rectangular box-shaped region of interest, the so-called ‘tailbox,’ has been used as defined by Tsuda et al. [Tsuda05]. The definition of the tailbox is schematically depicted in Figure 10 and its seasonal variation in Figure 11.

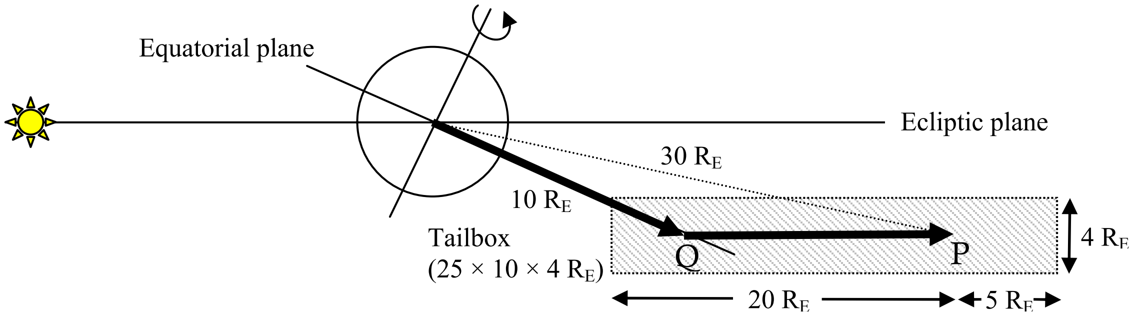

Figure 10: Tailbox definition (from [Tsuda05]).

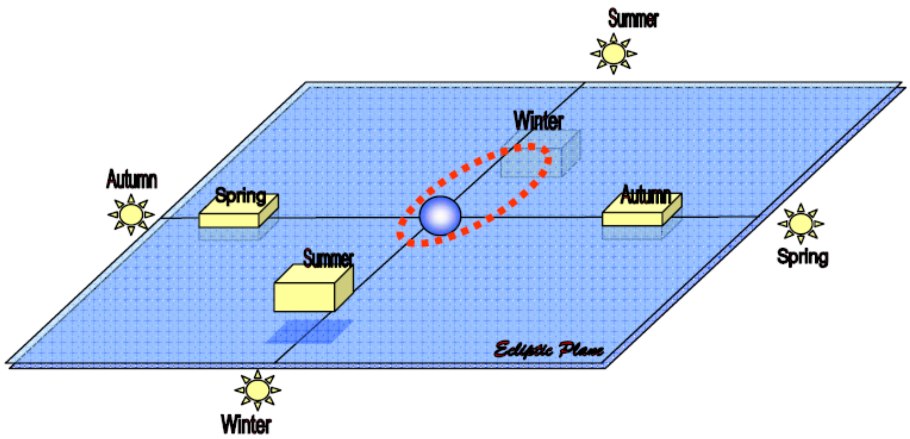

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Figure 11: Seasonal variation of the tailbox (from Tsuda05])

The key parameters of three different orbits with fixed initial 25 R_E apogee, 14° inclination, 270° argument of perigee and 0° RAAN are listed in Table 8 as a function of initial perigee. Several conclusions can be drawn from this table: Magnetopause crossing and tailbox visiting times decrease with perigee. However for magnetopause crossing, also the average spacecraft speed decreases with perigee, allowing more accurate magnetopause investigations. Likewise, also for tailbox visiting other aspects need to be considered as well. Although the time spent in the tailbox decreases with lower perigee, the constellation shape during tailbox visits improves. This is illustrated in Figure 12, which shows the true anomaly of the tailbox visiting point for a low and high perigee orbits. As the perigee increases, the tailbox visits occur increasingly further away from apogee (true anomaly 180°), which is the optimization point for the constellation shape (ref. section 3.2.4).

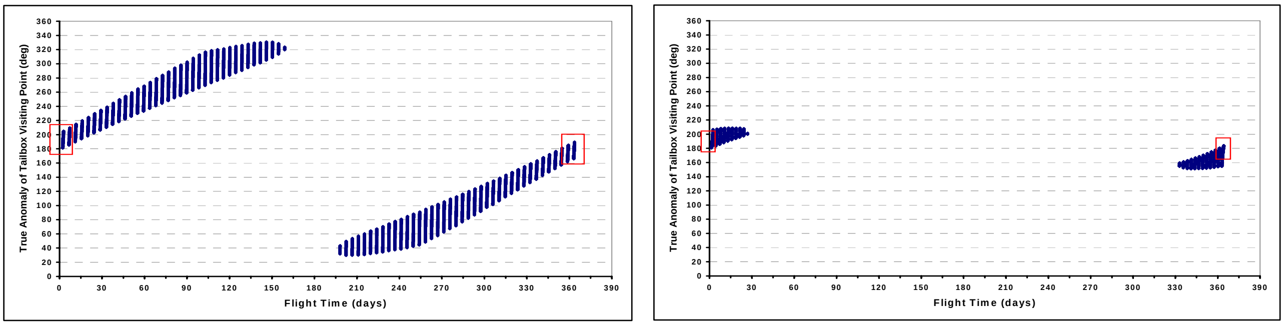

Figure 12: True anomaly during tailbox visits in the first year for an orbit with perigee of 10 R_E (left) and 1.4 R_E (right)

Table 8: Key characteristics of possible Cross-Scale TRS orbits.

Orbit perigee → **Parameter ↓**	**10 R_E**	**4 R_E**	**1.4 R_E**	**Remarks**
Orbital period (days)	4.3	3.2	2.8	
Magnetopause (initial orbit)				
Time spent between 9 – 11 R_E (%)	9.8	4.6	4.2	
Spacecraft velocity 9 – 11 R_E (km/s)	2.9	2.1	1.2	
Tailbox (first year)				
Number of tailbox visits	76	52	28	
Minimum true anomaly during visit (°)	30	110	145	Apogee at 180 degree true anomaly
Maximum true anomaly during visit (°)	330	250	215	
Average visiting time (hr)	9	11	16	
Total visiting time (days)	29	23	18	
Total visiting time (%)	8	6	5	
Tailbox (five years)				
Total tailbox visiting time (days)	123	83	59	
Total tailbox visiting time (%)	6.7	4.5	3.2	

4.3 Constellation design

As outlined in section 3.2.4 , the spacecraft are required to form a regular geometry at certain optimization point(s); around apogee if the perigee is lower than 10 R_E and possibly around apogee and perigee for a baseline orbit with perigee equal to 10 R_E. This formation should be achieved in a natural way by placing each individual spacecraft in its own stable orbit that is slightly different from the baseline reference orbit. During the mission lifetime, the spacecraft separations shall be varied within a certain range.

Table 9: Constellation Δv, maintenance and reconfiguration requirements for 5 year operational lifetime

Orbit perigee → **Parameter ↓**	**10 R_E** **Optimized at apogee and perigee**	**10 R_E** **Optimized at apogee only**	**4 R_E**	**2 R_E**	**1.4 R_E**	**Remarks**
Optimized at apogee	**Y**	Y	Y	Y	Y	
Optimized at perigee	**Y**	N	N	N	N	
Electron scale (2 S/C)						
Deployment Δv (m/s)	~0.1	0.1	0.1	0.2	0.2	Deployment to 10 km
Reconfiguration Δv (m/s)	~4.0	3.7	5.2	6.6	8.4	5 × 90 km [10 km ↔ 100 km]
Maintenance Δv (m/s)	0.1	0.1	0.1	0.1	0.1	
Control frequency (months)	12	12	12	12	12	Reconfiguration and maintenance
Ion scale (4 S/C)						
Deployment Δv (m/s)	~13	7	12	16	18	Deployment to 1,000 km
Reconfiguration Δv (m/s)	~73	39	68	91	104	3 × 1950 km [50 km ↔ 2,000 km]
Maintenance Δv (m/s)	~10	5	8	10	13	
Maintenance frequency	6	6	5	4	4	Reconfiguration and maintenance
Fluid scale (4 S/C)						
Deployment Δv (m/s)	~75	40	70	89	108	Deployment to 6,000 km
Reconfiguration Δv (m/s)	~300	160	278	356	432	6,000 km → 3,000 km → 15,000 km → 6,000 km
Maintenance Δv (m/s)	0	0	0	0	0	
Maintenance frequency	18	18	14	11	10	Reconfiguration and maintenance

For the optimization of the spacecraft orbital parameters, several criteria have been used. First of all, the ideal configuration should be maintained as long as possible around the optimization point(s). Additionally, collision avoidance throughout the mission lifetime as well as minimization of the propellant requirements for deployment, reconfiguration and maintenance has been taken into account. Figure 13 depicts the evolution of a large scale tetrahedron along the orbit for a 2 × 25 R_E orbit. Clearly, between true anomalies of 160° and 200°, the tetrahedron is in a good shape, while further apart it becomes more and more distorted, until it is more or less a flat square around perigee.

Table 9 summarizes the Δv requirements for a constellation of 10 S/C, consisting of concentric regular tetrahedrons on the fluid and ion scale, and a mother daughter system on the electron scale. The first and second column compare the requirements for a constellation that is only optimized at apogee (2nd column) with one that is both optimized at apogee as well as perigee (1st column), where the spacecraft baseline distance of the constellation at perigee is a factor 2.5 less than at apogee. Apparently, to achieve a tetrahedron of both perigee and apogee is quite demanding in terms of ΔV requirements, and therefore this option has been discarded.

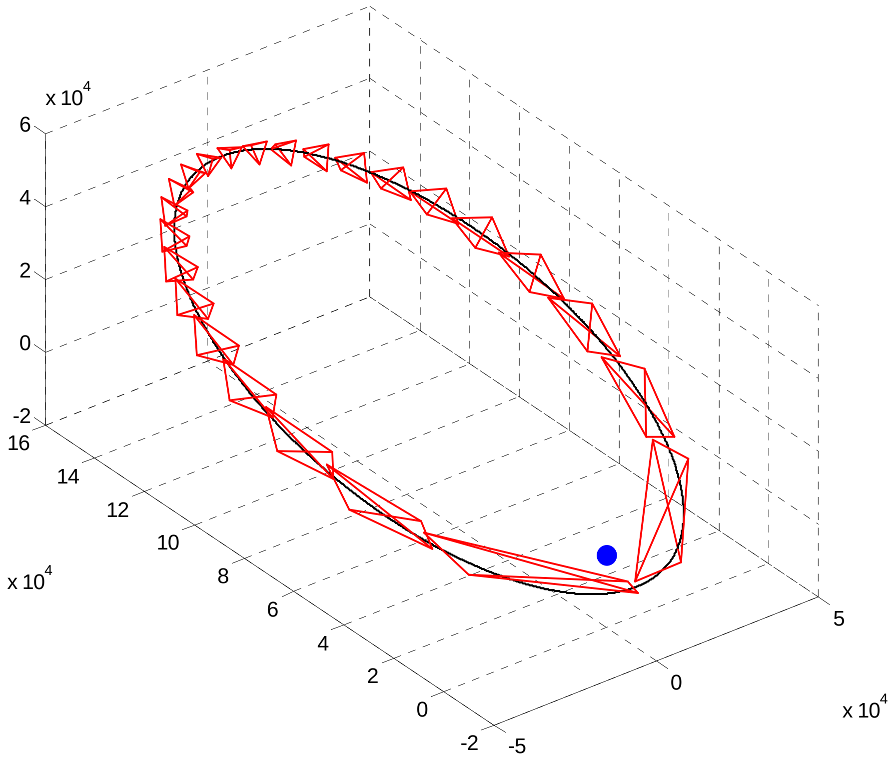

Figure 13: Evolution of a tetrahedron with a spacecraft distance of 6000 km along the 2 × 25 R_E orbit.

The table also clearly shows that, for constellations optimized only at apogee, the Δv requirements as well as the required control frequency increase with lower perigee. This will not only impact the spacecraft propellant budget, but also the effort for spacecraft control.

The control frequency has been determined assuming a 5 year operational lifetime and an optimization between orbit control and reconfiguration manoeuvres (combining both whenever possible). Attitude control manoeuvres have not been taken into account. The following requirements have been found to be driving the control frequency (see also [AD_MRD], R-2.3.2-6/R-2.3.2-7):

- The mission should allow for at least 5 changes in the small scale distance from 2 – 10 km to 100 km
- The distance between any two S/C at the medium scale shall not differ more than 10% from the actual average (medium scale) spacecraft distance

- The large scale separation shall start with a baseline separation of 6,000 km, then move to respectively 3,000 km, 15,000 km and back to 6,000 km separation

As a last note, it is important to remark that the Δv values provided in Table 9 depend on the geometry of the constellation. If the 10 spacecraft are not in two concentric tetrahedrons with a mother-daughter system in the middle, but e.g. in three tetrahedrons with one corner shared, then the Δv requirements for the 'shared corner' spacecraft would be similar to the small S/C requirements, but for the other nine small, medium and three large scale spacecraft the Δv requirements approximately increase with 35%, because in this geometry the corner spacecraft is placed in the reference orbit and therefore does not move.

4.4 Disposal strategy

The "European Code of Conduct for Space Debris Mitigation" [CoC04] has significant implications on the Cross-Scale mission concept design for orbits with a perigee lower than 6.6 R_E, since these orbits cross the GEO protected region (ref. section 3.2.2). Table 10 provides an overview of the recommended mitigation procedures for several candidate orbits for the Cross-Scale TRS.

For orbits with perigee below approximately 2.5 R_E, de-orbiting at end-of-life is the most efficient procedure, while for orbits with a higher perigee, a perigee raising manoeuvre to an altitude above the GEO protected region requires the least Δv. It should be noted that the design probability for successful completion of the mitigation procedures at end-of-life should be at least 90%, which has important implications on the spacecraft design and possibly also on the operational lifetime (to guarantee a successful disposal, a spacecraft with degraded functions might have to be de-orbited before it might not be possible anymore). One solution to circumvent the reliability requirements is to baseline acceptable low-perigee orbits which can be guaranteed to de-orbit naturally within 25 years (or which need a modest perigee raising manoeuvre every couple of years).

Table 10: Space debris mitigation procedures for candidate orbits for the Cross-Scale TRS.

Orbit	Mitigation procedure	Δv	Remarks
$10\ R_E \times 25\ R_E$	Only S/C passivation	0 m/s	Outside protected regions
$4\ R_E \times 25\ R_E$	Increase perigee at end-of-life to above GEO altitude plus S/C passivation	185 m/s	Parking orbit above GEO problematic, as eccentricity of 0.003 is required (!) – otherwise potentially unstable configuration[1]
$3\ R_E \times 25\ R_E$	Increase perigee at end-of-life to above GEO altitude plus S/C passivation	288 m/s	Very expensive. Orbit discarded.
$2\ R_E \times 25\ R_E$	De-orbiting at end-of-life	200 m/s	
$1.4\ R_E \times 25\ R_E$	De-orbiting at end-of-life	106 m/s	Natural re-entry within 25 years might be possible

[1] According to ESOC (CDF session 28-Nov-07)

5 MISSION ENVIRONMENT

The environmental conditions for Earth orbiting missions are already well-documented in e.g. the ECSS standards [ECSS00]. This section therefore focuses only on the environmental parameter which primarily affects the spacecraft and payload design: the induced radiation dose for the orbits of interest to the Cross-Scale TRS.

The total radiation dose for the Cross-Scale TRS spacecraft is a strong function of perigee altitude because several of the candidate orbits cross one or more of the Earth's radiation belts, which extend up to approximately 7 R_E in the equatorial plane. The radiation dose for a five year mission lifetime at solar maximum and a 90% confidence level is shown in Table 11 as a function of shielding thickness (solid aluminium sphere). For all orbits with a perigee below 10 R_E the contribution of the radiation belt crossings to the total radiation dose is significant. The orbits with a 3 R_E and 4 R_E perigee cross approximately through the peak of the electron outer belt and therefore experience the highest radiation, while for lower perigees the total dose decreases again due to the increased spacecraft velocity during radiation belt crossing.

It can be concluded that, apart from the high perigee orbit, well-shielded spacecraft with space qualified radiation hardened components are necessary. The same applies to the payload instruments.

Table 11: Radiation environment for candidate orbits for the Cross-Scale TRS (5 year lifetime).

Orbit	1 mm Al shielding (krad)	2 mm Al shielding (krad)	4 mm Al shielding (krad)	Radiation belts
10 $R_E \times 25 R_E$	140	30	6	Above radiation belts
4 $R_E \times 25 R_E$	1,800	520	70	Through peak of electron outer belt
3 $R_E \times 25 R_E$	1,700	550	90	Through peak of electron outer belt
1.4 $R_E \times 25 R_E$	1,300	370	64	Electron outer belt and proton inner belt
1.08 $R_E \times 25 R_E$	1,200	340	57	Electron outer belt and proton inner belt (at high speed)

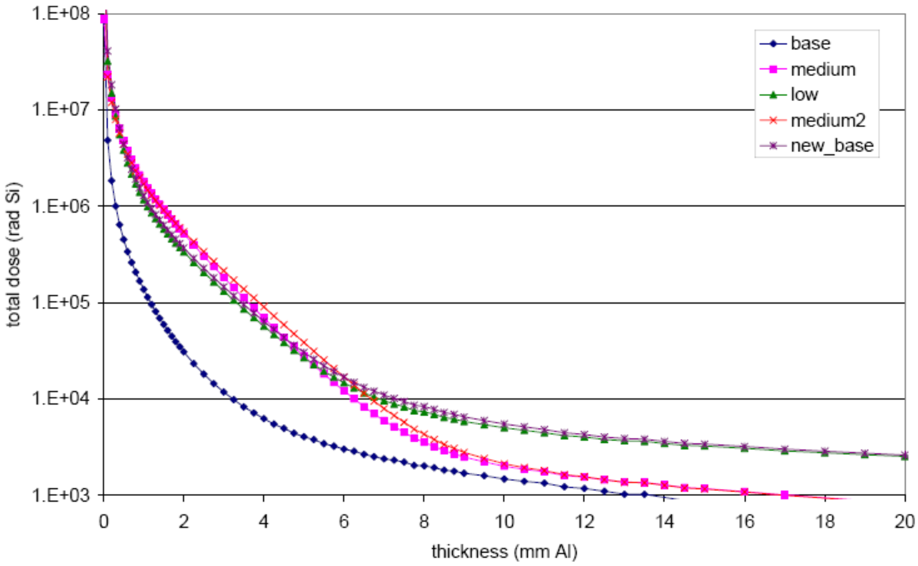

Figure 14: Total doses for solid sphere for the 5 year CS-TRS at the different target orbits.

6 DEFINITION OF MISSION ARCHITECTURE

This section details the mission architecture trade that has been performed in order to determine the most applicable mission architecture that best fulfils the mission study objectives and requirements, which were outlined in section 2 and 3.

6.1 Key system drivers

The key system drivers for the Cross-Scale TRS architecture are:

- The number of S/C (ref. section 2.2)
- The Δv requirements for transfer to the operational orbit (section 4.2.1)
- The Δv requirements for reconfiguration (section 4.3)
- The Δv requirements for compliance with space debris mitigation rules (section 4.4)
- Mission operations (spacecraft/constellation reliability) (section 3.3.3)

The objective of the mission architecture trade is essentially to maximize the number of spacecraft (at least 10) to be launched with a single Soyuz-Fregat while minimizing the projected mission cost. Secondary objectives are the maximization of spacecraft payload resources, and optimization of orbit (science return, environmental conditions).

6.2 Major trade elements

The major trade elements and their impact are listed in Table 12. Clearly, the architecture trade is not a trivial task, because several trade options have a positive impact on one system driver, while at the same time a negative impact on another system driver.

Table 12: Cross-Scale mission architecture trade elements

Mission element	Impact on	Remarks
Launch vehicle	Cost, mass	Single Soyuz-Fregat (section 4.1)
Transfer to operational orbit	Cost, mass	
Operational orbit	Mass	Also impacts space debris mitigation requirements, communication architecture, S/C design (radiation environment), ground segment
Spacecraft design philosophy	Cost, mass	For cost optimization, fully identical spacecraft busses preferred. Mass considerations would require opposite. Interacts with trade on transfer to operational orbit and payload design philosophy.
Payload design philosophy	(Cost), mass	Aim for nearly identical design, i.e. only switch between instruments with identical interfaces allowed.
Spacecraft reliability	Cost, mass	There exists a trade between S/C redundancy (spare policy) and subsystem redundancy (S/C reliability). Also impacts ground segment and operations
Communication architecture	Cost, mass	Trade between relay satellite and individual download.
AIV philosophy	Cost	Identical S/C is the key for cost minimization.
Ground segment and operations	Cost	Primarily driven by S/C design philosophy, S/C autonomy and reliability as well as communication architecture. Number of ground stations and performance of ground stations is less critical. Data return reliability and timeliness requirements can be important drivers as well.

6.3 Mission architecture trade

6.3.1 TRANSFER TO OPERATIONAL ORBIT

Table 13 lists a number of options for transfer of the constellation of spacecraft to the operational orbit together with a discussion of the advantages and disadvantages. The first option, transfer with a Soyuz-Fregat to the operational orbit is not feasible as the Fregat stage cannot perform a perigee raising manoeuvre. The second option, in which after launcher separation, all spacecraft are individually transferred to the operational orbit, is not attractive due to the heavy operational requirements (when spacecraft are transferred sequentially also phasing issues need to be addressed). Constraints on the launch vehicle accommodation are an issue as well, particularly the centre of gravity (if all spacecraft are stacked on top of each other) or the distribution of launch loads (if several stacks). Using a dispenser (option 3) issues with the launch vehicle constraints can be solved, though the mission operations complexity still makes it unattractive.

Table 13: Options and discussion for transfer to the operational orbit

	Option	Pro	Con
1	Transfer by SF2-1b	Simple	• Not mass-efficient for orbits above GTO. • Additionally Fregat restart sequence does not allow increasing perigee.
2	SF2-1b to GTO/HEO. Launch without dispenser. Each S/C performs own transfer.	No need for dispenser (mass saving)	• Mission operations is complex. • Each S/C needs main engine. Main engine conflicts with foreseen payload. • Launch vehicle accommodation issues (more than 10 stacked spacecraft)
3	SF2-1b to GTO/HEO. Launch with dispenser. Each S/C performs own transfer	• Simple dispenser. • Dispenser mass does not need to be transferred further than GTO/HEO (mass saving)	• Mission operations is complex • Each S/C needs main engine. Main engine accommodation conflicts with foreseen payload (axial booms). • Possibly launch vehicle accommodation issues due to larger S/C
4	SF2-1b to GTO/HEO. Groups of S/C perform transfer (mother/daughter system).	No need for dispenser	• Non-identical S/C • Mother S/C (likely) need 3-axis stabilization • Main engine on mother S/C conflicts with foreseen P/L. • Likely LV accommodation issues.
5	SF2-1b to GTO/HEO. A dedicated dispenser-like transfer vehicle.	• Launcher accommodation similar as with dispenser for small satellites (3-axis stabilized transfer vehicle which distributes launch loads) • Single main engine • Relatively simple mission ops (during LEOP)	• Design of unique transfer vehicle • Additional mass needs to be carried towards operational orbit and de-orbiting

The mission operations complexity of option 2 and option 3 can be reduced by grouping the spacecraft during transfer to the operational orbit (option 4). The larger spacecraft (the mother) carries a number of smaller spacecraft (daughters) to the operational orbit. All spacecraft would carry science instruments and become part of the constellation after deployment. For a constellation of more than ten spacecraft, this requires more than one mother spacecraft. The accommodation of several mother/daughter systems in a single launcher has similar accommodation issues as option 2 and has therefore been discarded. The last option, using a dedicated transfer vehicle that during the launch distributes the loads similar to a dispenser is the most attractive solution. There are no significant launcher accommodation issues (apart from volumetric constraints) and only a single spacecraft needs to be controlled during the transfer sequence. An additional advantage of separating the transfer function from the science investigation is that the design of the constellation spacecraft is relatively independent of the operational orbit (the sizing of the constellation spacecraft does not depend critically on the propellant tanks). The main disadvantages are the cost and the mass for the transfer vehicle.

6.3.2 OPERATIONAL ORBIT

The operational orbit needs to be mainly traded against the number of spacecraft and available payload resources. Secondary objectives are the orbit impact on science return and environmental conditions. Table 14 provides a summary of the key Δv requirements for the different options for the operational orbits. It should be noted that these values cannot be simply added to find the useful spacecraft mass in orbit, because the deployment/reconfiguration/maintenance Δv only applies to a sub-set of the constellation (while the transfer vehicle is directly de-orbited). Additional complications for the orbit trade against the number of spacecraft and the available payload resources are the significantly differing eclipse times and radiation environment for the different orbits (see Table 7 and Table 11).

Table 14: Summary of transfer Δv requirements for the different options for the operational orbits

Orbit	Transfer Δv from GTO (m/s)	Large scale deployment / reconfiguration / maintenance Δv (m/s)	Disposal Δv (m/s)	Remarks
$10\ R_E \times 25\ R_E$	1215	200	0	Lunar resonance transfer. Constellation configuration not optimized at perigee
$4\ R_E \times 25\ R_E$	1005	362	185	Disposal to higher perigee
$2\ R_E \times 25\ R_E$	775	445	200	De-orbiting at end of life
$1.4\ R_E \times 25\ R_E$ (2500 km altitude)	675	540	106	De-orbiting at end of life

However, several insightful conclusions can already be drawn by making a few reasonable assumptions. First of all, all spacecraft (including the transfer vehicle) have to be transferred to the operational orbit and properly disposed of at the end of the operational life. Secondly, an upper limit for the constellation spacecraft propellant requirements can be derived by assuming all the available mass in orbit is distributed over ten spacecraft (two at small scale, four at medium scale, four at large scale). Lastly, the impact of eclipses on the useful spacecraft mass can be estimated by assuming a constellation of ten spacecraft with typically 100 W per spacecraft, and 100 Wh/kg batteries.

Table 15: Estimate for useful spacecraft mass in different operational orbits.

Orbit perigee→ / Item ↓	10 R_E	4 R_E	2 R_E	1.4 R_E	Remarks
Available mass in orbit (kg)	1660	1770	1900	1970	Assuming I_{sp} = 325 s., 20% system level margin subtracted.
Propellant for disposal (kg)	0	100	120	70	Assuming I_{sp} = 308 s.
Propellant for deployment and reconfiguration (kg)	~55	~100	~135	~165	Assuming 2 small scale S/C, 4 medium scale S/C, 4 large scale S/C. I_{sp} = 308 s.
Battery mass for eclipses (kg)	~35	~50	~65	~84	10 × 100 W, 100 Wh/kg
Net effective mass (kg) (estimate)	**~1570**	**~1520**	**~1580**	**~1650**	

Table 15 shows the estimated masses using those assumptions. Clearly the orbits with a medium perigee of 2 R_E and 4 R_E are of little interest: The net effective mass is comparable to the orbit with 10 R_E perigee using a lunar resonance transfer, while the 10 R_E perigee orbit has the most benign radiation environment, the most relaxed space debris mitigation requirements, the fewest maintenance manoeuvres, and spends the largest fraction of time in the tailbox. The maximum effective mass is achieved for the orbit with the lowest perigee. If the mission concept is mass constrained, the low perigee orbit will therefore be the best choice.

It should be noted that this rough trade does not take into account the mass for the dispenser vehicle (which would effectively reduce the constellation S/C mass but also the propellant mass for deployment/reconfiguration or the instrument extra shielding mass due to a more severe radiation environment). On the other hand, the available mass in orbit can be further optimized by launching the Soyuz-Fregat into an orbit different from GTO (see Table 3). However, the general conclusion that either a low perigee (1.4 R_E) or a high perigee (10 R_E) orbit are of most interest is not affected.

6.3.3 SPACECRAFT AND PAYLOAD DESIGN PHILOSPHY

As the different constellation spacecraft have slightly different requirements (propulsion, telemetry, synchronization / localization, payload resources), the most mass efficient solution would be to have a different design for each spacecraft scale:

- Small scale:
  - Relatively low propellant requirements,
  - Heavily instrumented,
  - Inter S/C localization/synchronization
- Medium scale:
  - Moderate propellant requirements,
  - Moderately instrumented,
  - Possibly inter S/C localization/synchronization
- Large scale:
  - High propellant requirements,

- Lightly instrumented (therefore also reduced data rates),
- No inter S/C localization/synchronization

However, for cost efficiency, all spacecraft should be completely identical. The impact of a (set of) different spacecraft is very significant and pervades practically all key cost areas:

- Spacecraft design
- Parts procurement
- Ground support equipment
- Spacecraft qualification / calibration (including number of models)
- AIV procedures
- Mission operations (mission preparation, spacecraft simulators, spacecraft commanding)

To achieve the lowest cost, essentially all the spacecraft hardware and software should be completely identical. On the other hand, it might be challenging to implement such a rigid philosophy, because it essentially requires to design all spacecraft to meet the 'most demanding requirement of any spacecraft,' e.g. propellant tanks and propulsion system sized for maximum Δv, power and structure sized for maximum payload configuration.

As an illustrative example, the demanding Δv requirements for the large scale deployment and reconfiguration necessitate the use of a bipropellant thruster system with additional tank configuration and valve system complexity. Such a system is essentially not required for the small scale spacecraft, which could be simpler in this respect. However, a cheaper propulsion system for the small scale spacecraft would result in a different structural design and qualification, different spacecraft software (and validation), different spacecraft simulators, different mission operations manual, etc. The increase in spacecraft cost and complexity by baselining a bi-propellant system for all spacecraft appears to be significantly less than the cost impact of two differently designed spacecraft. Alternatively a mono propellant system could be used also for the large-scale spacecraft at the expense of descoping the number of constellations changes.

The only deviation from the identical spacecraft philosophy that can be introduced without significant cost overheads is to allow reductions in the number of functions / subsystems / instruments for some of the spacecraft (which would be comparable to a non-functioning subsystem). E.g. not all spacecraft need to have a localization/synchronization system, all possible instruments or the same number of memory boards. Additionally, not all propellant tanks will need to be filled up to their maximum capacity. Naturally, one will need to make sure that the mass of all (spinning) spacecraft is still properly balanced.

For cost-efficiency the Cross-Scale TRS has baselined fully identical spacecraft designs, including identical payload interfaces. For mass efficiency, it is not required for each spacecraft to be assembled with all possible subsystems (e.g. no synchronization/localization or less/different instruments).

6.3.4 SPACECRAFT RELIABILITY

The individual Cross-Scale TRS spacecraft will either need to be designed for sufficient reliability to ensure that there is a reasonable probability that the full constellation will be functional during the entire mission lifetime or a spare replacement strategy should be implemented.

For a spare replacement strategy, one or more spare spacecraft would need to be launched with the first or a second Soyuz-Fregat or a new transfer vehicle needs to be designed that is tailored to a small launcher. Provided sufficient mass is available on the first Soyuz-Fregat launch, a single launch with the baseline and spare spacecraft is the only attractive option. However a spare replacement strategy has several important drawbacks:

- Since the different spacecraft in the constellation are differently instrumented, several (differently instrumented) spare spacecraft would be required.
- Depending on the malfunction, non-functioning spacecraft can be a significant hazard for the other spacecraft in the constellation (risk of collision, particularly on the small scale).
- Even if spare spacecraft would be available for replacing non-functioning spacecraft, the individual spacecraft will still need to be designed for sufficient reliability to comply with the space debris mitigation requirements (for orbits with medium and low perigee).

Considering the disadvantages of a spare replacement policy, the spacecraft for the Cross-Scale TRS should be designed for sufficient reliability that no replacement is required. Nevertheless in case of a loss of a spacecraft following strategy could be applied:

- loss of an electron S/C: move the remaining electron S/C close to an ion S/C
- loss of an ion S/C: move the two electron S/C to replace the lost ion S/C
- loss of an fluid S/C: orient the three remaining fluid S/C parallel to region of interest

The impact of individual spacecraft reliability on the constellation is illustrated in Table 16, where it has been assumed that the causes for malfunctioning are fully independent (excluding e.g. major design errors). Spacecraft with fully redundant and space qualified critical subsystems (e.g. power, on-board-data handling, communication) typically have a reliability of around 98% for a 5 year design lifetime. This would imply that there is about 82% probability that all spacecraft are functional, and 17% probability that one spacecraft will not be in operation at the end of the mission lifetime.

Table 16: Reliability for a 10 spacecraft constellation as a function of individual spacecraft reliability (assuming independent failure modes)

Individual spacecraft reliability	Probability for constellation			
	10 S/C [all S/C functional]	9 S/C [one S/C loss]	8 S/C [two S/C loss]	< 8 S/C [> 2 S/C loss]
99%	90.5%	9.1%	0.4%	0.01%
98%	**81.7%**	**16.7%**	**1.5%**	**0.09%**
97%	73.7%	22.8%	3.2%	0.28%
96%	66.5%	27.7%	5.2%	0.62%
95%	59.9%	31.5%	7.5%	1.1%

6.3.5 COMMUNICATION ARCHITECTURE

For down-link of the data to Earth, several options exist. The simplest approach is to have all spacecraft communicating independently to ground. On the other hand the use of inter-satellite communication to collect all data at a single dedicated spacecraft that transfers all the data to ground offers the potential implementation of an on-board triggering algorithm, which could signal interesting constellation events, even without prior ground station involvement. Based on the assumption that all Cross-Scale TRS spacecraft should be designed identical (for cost reduction), only the transfer vehicle could be used as a relay satellite.

The down-link rates that are achievable with a 3-axis stabilized transfer vehicle/relay satellite (using a high gain antenna) are significant. However, the data-rates that are achievable with inter-satellite links with (on one side) a spinning satellite, are very limited, particularly for the medium and large scale spacecraft, which are far removed from the centre of the constellation (particularly when being away from apogee, as shown in Figure 3). A detailed assessment has shown that inter-spacecraft communication would only be efficient for the small scale spacecraft. The additional complexity, risk of single point failure and increased cost to implement a data relay function on the transfer vehicle, which is used only by less than one third of the constellation is not on appealing option. Consequently, the selected baseline for the Cross-Scale TRS is that each spacecraft communicates independently with the ground.

6.3.6 AIV PHILOSOPHY

The key driver for the Assembly/Integration/Verification (AIV) process is the spacecraft design philosophy. Identical spacecraft will allow significant savings in time and cost, as the design qualification has only to be performed on a single spacecraft, while the other spacecraft only need to be verified for workmanship compliance. There is significant experience with multi-spacecraft integration and verification inside and outside Europe, which can potentially be used to optimize the AIV process for the Cross-Scale TRS. Relevant examples are the ESA Cluster mission (4 spacecraft), the NASA Themis mission (5 spacecraft), but also the Globalstar constellation (first generation: 76 spacecraft, second generation: 48). For the first generation Globalstar Low-Earth Orbit satellites (700 kg), a production speed of three spacecraft per month was achieved.

6.3.7 GROUND SEGMENT AND OPERATIONS

The ground segment and operations are primarily driven by the required staffing, both in preparation and during the operational lifetime. Table 17 lists the key mission architecture drivers that impact the ground segment and operations. As already pointed out in section 6.3.3 , the most critical driver is the spacecraft design philosophy. Other key drivers are the transfer to operational orbit, spacecraft reliability, and the communication architecture. Although not listed in the table of mission architecture elements, other relevant mission design trades that drive the ground segment are e.g. spacecraft robustness, spacecraft autonomy, data return reliability and timeliness requirements. These are further discussed in section 7.5.

Table 17: Impact of other mission architecture elements on the ground segment and operations

Mission element	Impact on ground segment and operations	Cross-Scale baseline
Launch vehicle	Single launch vehicle reduces launch campaign and LEOP effort	Single launch vehicle
Transfer to operational orbit	Single transfer vehicle reduces LEOP effort	Single transfer vehicle
Operational orbit	o High perigee (10 R_E): less maintenance manoeuvres, more benign radiation environment (spacecraft anomalies), less demanding space debris mitigation requirements o Low perigee (1.4 R_E): shorter link ranges	*Mission design trade*
Spacecraft design philosophy	Fully identical spacecraft busses are critical: - software (generic simulators and databases, which allow spacecraft specific entries) - documentation (e.g. spacecraft operations manual) - training of staff - single operations team for all spacecraft	Identical spacecraft
Payload design philosophy	Limitation of number of different instruments, limitation of instrument modes, autonomy (radiation belt crossings, anomaly recovery), limitation of or automatic in-orbit calibration.	To be determined
Spacecraft reliability	Spacecraft reliability and also autonomy are key to minimize the operations effort. This needs to be traded against mission requirements and technological readiness.	*Mission design trade*
Communication architecture	The number of ground stations needs to be traded against resource allocation requirements on the space segment.	*Mission design trade*

6.4 Mission architecture baseline

Table 18 summarizes the mission architecture baseline for the Cross-Scale TRS.

Table 18: Cross-Scale TRS mission architecture baseline.

Mission element	Cross-Scale TRS baseline	Remarks
Launch vehicle	Single Soyuz-Fregat 1B from Kourou	
Transfer to operational orbit	Single dispenser-like transfer vehicle	To reduce LEOP operations
Operational orbit	1.4 R_E × 25 R_E × 14°	Lowest Δv for transfer, most mass at target orbit
Spacecraft design philosophy	Identical spacecraft	
Payload design philosophy	Interchangeable if identical spacecraft interface	
Spacecraft reliability	Highly reliable spacecraft	To avoid spare S/C replacement
Communication architecture	Direct and independent communication to ground	Inter S/C communication is a power driver for large scale S/C
Ground segment and operations	Minimization of complexity and requirements	To be traded with space segment design and mission requirements

7 MISSION DESIGN

7.1 System overview

Table 19: System mass summary including a system margin of 20% for a fully loaded Soyuz launch into insertion orbit of 180 km x 20164 km with 14° inclination

Item	Small scale (kg)	Medium scale (kg)	Large scale (kg)	Carrier dispenser (kg)	Remark o Launcher injection 180 x 20164 km o Final Target Orbit 1.4 R_E x 25 R_E
Science instruments	42.0	34.5	13.0	-	
Localization & synchronization	5.0	5.0	-	-	X-band solution (S-band 8 kg)
Communications	9.5	9.5	9.5	6.1	• S/C: 12 W_{RF} X-band, omnidirectional • Dispenser: 2 W_{RF} S-band, omnidirectional
Structure	28	28	28	224.4	Thrust tube concept
Propulsion	8	8	8	125	• Chemical bipropellant system. • Tanks sized for maximum separated mass plus 5% margin.
AOCS	7	7	7	32.9	2 star trackers, x sun sensors, accelerometers
OBDH	10.7	10.7	10.7	10.8	Including data harness and memory
Power	28.2	28.2	28.2	45	Together with power harness
Sttructure link disp	3	3	3		to support deployment from dispenser
Thermal	6	6	6	6.4	
EPS harness	6	6	6		
Nominal dry mass	**157**	**149**	**122**	**490**	
Propellant	6	13	29	1170	Both high and low thrust manoeuvres included
Wet mass	163	162	151	1660	
System level margin (20.3%)	33	32	30	332	
S/C wet mass incl. system level margin	196	194	181	1992	For propellant calculation and propulsion system sizing.
Number of S/C	x 2	x 4	x 4	x 1	
	392	776	724	1992	
	1892			**1992**	
Total wet mass	**3884**				
Launch adapter	110				
Total launch mass	**3994**				Including 20.3% system level margin
Launch vehicle capacity	**3996**				Reference launch date 2/11/2013. Incl. launch window margin. Launch into 180 km x 20164 km

7.1.1.1 Orbital characteristics

Table 20: Delta-v requirements for the two orbits options, using lunar resonance transfer for the 10 R_E option

Item	BASELINE Low perigee 1.4 Re	High perigee 10 Re	Remarks Transfer from GTO
Transfer from launcher insertion to target orbit	675 m/s	1215 m/s	
Disposal of transfer vehicle	77 m/s	~50 m/s	Dispenser needs to be transferred out of constellation orbit
Small scale	7 m/s	4 m/s	Deployment / reconfiguration / maintenance
Medium scale	125 m/s	51 m/s	Deployment / reconfiguration / maintenance
Large scale	475 m/s	200 m/s	Deployment / reconfiguration / maintenance
Disposal of S/C	106 m/s	0 m/s	Space debris mitigation

A detailed assessment of the key parameters of the two main orbit options, based on above Δv requirements has resulted in the following table:

Table 21: Key parameters for the two orbit options

Item	1.4 Re	10 Re	Remarks
Maximum eclipse times	8.4 h	3.4 h	
Radiation dose	63 krad	4 krad	Also impacts S/C SEE (and thus mission ops + instrument shielding)
Orbital stability	Poor	Good	Also impacts mission ops
Ground station visibility	49%	46%	Assuming single 15 m G/S
Maximum gap in contact	34 hr	26 hr	Assuming single 15 m G/S
Average link distance	+	+/-	
Number of S/C in orbit	10	10	Both assume SF launch into orbit lower than GTO. High perigee option is very mass critical and therefore TBC
Total P/L mass into orbit	~315 kg	~240 kg	Mass including localization/synchronization (1.4 Re: small + medium scale, 10 re: only small scale)
Total effective P/L mass into orbit	~280 kg	~228 kg	Mass excluding localization/synchronization
Space debris mitigation	Reliability requirements	No driving requirements	Reliability requirements for de-orbiting might lead to shorter mission operational lifetime

Clearly, the trade between the orbits is not straightforward. The high perigee option has several advantages (higher orbital stability, reduced radiation environment, no significant space debris requirements) at the price of an increased transfer Δv. Looking at the primary objectives (number of S/C and P/L resources), the low perigee option has a significant better mass performance and allows with a comfortable system level mass margin a constellation with 10 S/C instrumented with a meaningful P/L configuration.

The better mass performance for the low perigee option is mainly attributed to a more evenly spreading of the delta-v requirements across the difference mission elements (dispenser and science S/C), so that the mass of the separate stages is better optimized. This applies particularly for the dispenser, which requires a total Δv of ~750 m/s for the low perigee case, while up to 1215 m/s for

the high perigee option (transfer from GTO). The mass benefit due the lower Δv for the dispenser/transfer vehicle becomes even more pronounced for a launch into an HEO orbit below GTO. A final trade comparing the low and high-perigee option has shown that the overall mass benefit in propellant allocation in the low perigee case outweighs the increase in battery mass (due to higher eclipse times) for the low perigee orbit.
Using lunar resonances during the transfer the overall Δv for the high perigee case can be significantly reduced in summary also the high perigee case could become attractive.

Table 22: Main characteristics to compare low and high perigee orbit (with or without lunar resonance transfer)

Option	Relevant characteristics	Conclusion
10 x 25 R_E	• S/C visit magnetopause on dayside • Tailbox is visited 76 times/year (11 months/y). • Average visiting time 9 h. Total visiting time 29d. • Tailbox is visited at almost every true anomaly (average constellation shape during visit is pretty bad) • Two ~50 days eclipse periods with max. duration of 2.5 h **up to 3.3 h** • Low radiation levels: 4 krad (4 mm) for 5 y • Orbit period: 4.3 days **Total-Δv**[(2)]: 1400 m/s (transfer) + 200 m/s (max. reconfiguration) + 0 m/s (de-orbiting) = 1600 m/s [3]	No requirements on space debris mitigation because orbit is above protected regions.
10 x 25 R_E with lunar resonance transfer	• Total-Δv[(2)]: 1115 m/s (transfer) + 200 m/s (max. reconfiguration) + 0 m/s (de-orbiting) = 1315 m/s	Using lunar resonances transfer, the 10 x 25 Re orbit is attractive.
1.4 x 25 R_E	• Tailbox is visited 22 times/y (5.5 months/y). • Average visiting time is 11 h. Total visiting time 23d. • Tailbox is visited at true anomaly between 170° and 210° (average constellation shape during visit is good) • 300 day eclipse period with max eclipse time of 1.5 hr **up to 8.4 hr** • Outer electron belt and proton belt (~60 krad 4 mm 5 y) • Orbit period: 2.8 days • Total-Δv[(2)]: 675 m/s (transfer) + 475 m/s (max. reconfiguration) + 110 m/s (de-orbiting) = 1260 m/s	• Lowest total delta-v • S/C needs de-orbiting at end-of-life • Increased radiation environment introduces some mass penalty

7.2 Dispenser design

The Cross-Scale TRS is based on a dispenser S/C, which is used to:

- Allows for the accommodation of 10 deployable satellites
- Performs the transfer from the launcher insertion orbit to the target orbit
- Provides its own propulsion system for this transfer

[2] Care must be taken with 'total delta-v' here as the individual contributions are done by different modules (e.g. transfer by dispenser + 10 S/C, max. reconfiguration only by ion scale S/C and de-orbiting by all 10 S/C).
[3] Transfer + most demanding deployment/reconfiguration + space debris requirements = maximum all-up delta-v (incl. margins).

- Provides 3-axis stabilization during transfer and S/C release
- Supports the accommodation of the propellant tanks (~1170 kg propellant)[4]
- Allows for the safe release of the 10 satellites (in sequential manner)
- Performs a safe disposal of the dispenser at the end of satellite release

7.2.1 DISPENSER DESIGNS DRIVERS

The main design drivers for the dispenser are given by structural requirements, propulsion system and tank accommodation and the 3-axis stabilisation AOCS.
The 1.17 tons of bi-propellant are accommodated in 2 titanium tanks which are inserted inside of the central carbon tube. The satellites are mounted outside on the S/C interface rings (see Figure 15).

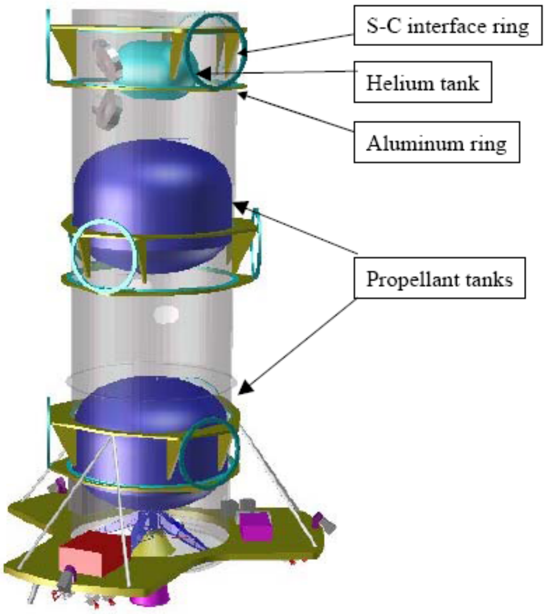

Figure 15: Dispenser design

7.2.2 AOCS

A spin of the composite around the longitudinal axis would not be stable (as this is the axis of minimum inertia) and therefore also a 3-axis stabilisation during the cruise is selected. In cruise phase the dispenser longitudinal axis is oriented towards the Sun, with a fixed solar array lying at the top of the dispenser (see Figure 16). Only in safe mode a low spin rate along the longitudinal axis (Z-axis) is adopted.

7.2.3 ELECTRICAL POWER SUPPLY (EPS)

The Dispenser EPS provides:

- o generation of the electric power
- o energy storage
- o power conditioning and protection
- o power distribution to the 10 Cross-Scale spacecraft

The solar array (SA) is composed of one body-mounted panel placed on the top of the dispenser Tube. Power from the SA is transferred to the Power Control and Distribution Unit (PCDU) and distributed to 8 Cross-Scale satellites via the S/C interface rings. Power during launch and eclipse is provided by the Li-Ion battery.

[4] In case of GTO to 1.4 x 25 R_E orbit.

Power to the 2 satellites mounted at the top of the dispenser is provided by their own solar array since they are facing the Sun directly. For the other satellites the power is provided by the dispenser PCDU.

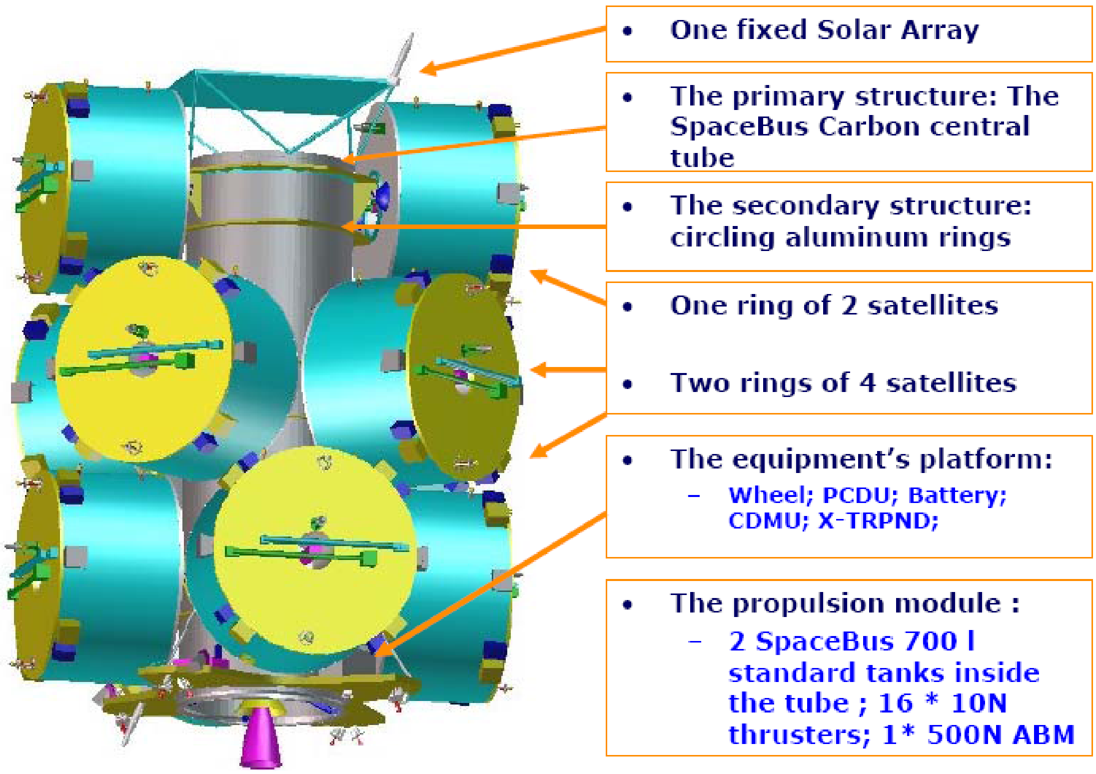

Figure 16: The dispenser carries the 10 S/C and has the propellant tanks accommodated inside the central tube. The solar panel on top of the dispenser is pointing to the sun during the transfer phase.

7.2.3.1 Dispenser Solar Array

The dispenser solar array (3.7 m²) is equipped with GaAs triple junction solar cells (same as for all Cross-Scale TRS satellites). Considering that the two upper satellites are getting power from their own solar panel, the dispenser solar array power requirement is 964 W (see Table 23).
Optical Sun Reflectors are not needed, since the estimated solar array temperature is less than the maximum allowed temperature (~130°C). Considering a solar flux of 1371 W/m², MPPT solar array regulation, a solar cell efficiency of 28%, a filling factor of 85%, tolerance to a one string failure, blocking diode, harness losses and considering the worst-case operative temperature (110°C), the solar array needs to consist of 46 strings with 22 cells per string.
Figure 17 shows the curve of the available power according to its temperature. A top of 110 °C will allow for the required 964 W. The solar array mass budget is given in Table 24.

	Unit Cont.	LAUNCH	Cruise	SUN ACQUISITION	ECLIPSE	Thrust	SAFE
Dispenser Power Budget (per mode)							
Orbit distance			1 AU	1 AU	1AU	1AU	1 AU
AOCS Modes		N/A	Sun Acquisition	Sun Acquisition	Nominal	Nominal	Sun Acquisition
P/L Modes		Survival	Survival	Survival	Science	Survival	Survival
Duration		120 minutes	10 days	TBD minutes	1.5 hours	2 hours	days
PLATFORM							
Avionics : (without margin)		20.0	25.0	25.0	25.0	25.0	25.0
(margin)		2.0	2.5	2.5	2.5	2.5	2.5
Avionics : (with margin)		22.0	27.5	27.5	27.5	27.5	27.5
SMU	10%	22	27.5	27.5	27.5	27.5	27.5
RF System : (without margin)		4.0	18.5	18.5	18.5	18.5	18.5
(margin)		0.4	1.9	1.9	1.9	1.9	1.9
RF System : (with margin)		4.4	20.4	20.4	20.4	20.4	20.4
XPND	10%	4.4	6.05	6.05	6.05	6.05	6.05
SSPA	10%	0	14.3	14.3	14.3	14.3	14.3
AOCS: (without margin)		25.5	30.9	30.9	30.9	30.9	30.9
(margin)		0.0	0.3	0.3	0.3	0.3	0.3
AOCS : (with margin)		25.5	31.2	31.2	31.2	31.2	31.2
Star Tracker	5%	0	5.67	5.67	5.67	5.67	5.67
Wheels	5%	25.2	25.2	25.2	25.2	25.2	25.2
accelerometers	5%	0.315	0.315	0.315	0.315	0.315	0.315
AOCS Actuators : (without margin)		0.0	8.0	30.5	8.0	35.2	23.8
(margin)		0.0	0.4	1.5	0.4	1.8	1.2
AOCS Actuators : (with margin)		0.0	8.4	32.0	8.4	37.0	25.0
Thrusters	5%	0	8.4	32	8.4	37	25
Thermal : (without margin)		10.0	20.0	20.0	20.0	20.0	20.0
(margin)		1.0	2.0	2.0	2.0	2.0	2.0
Thermal (with margin) alloc. apogee		11.0	22.0	22.0	22.0	22.0	22.0
Heaters	10%	11	22	22	22	22	22
Power System : (without margin)		8.3	207.4	41.5	40.4	41.7	41.2
(margin)		0.8	29.6	4.7	4.5	4.7	4.6
Power sub-system (with margin)		9.1	237.0	46.2	45.0	46.4	45.8
PCDU	10%	9.1	33.5	34.7	33.5	34.9	34.3
Battery Charge	15%	0	203.6	11.5	11.5	11.5	11.5
Platform Total Power Consumption. With margin [W]		**72.1**	**346.5**	**179.2**	**154.4**	**184.4**	**171.8**
SATELLITES							
Satellites (with margin)		0.0	440.2	440.2	440.2	440.2	440.2
ESA Margin 20%		**14.8**	**160.6**	**127.3**	**122.2**	**128.4**	**125.8**
Total [W]		**89**	**964**	**764**	**733**	**770**	**755**

Table 23: Dispenser power budget

ITEM	Unit mass	Quantity	Length [m]	TOTAL [kg]	Uncert. %	TOTAL [kg]
Solar Cell [g]	2,60	1012		2,63	2	2,68
Blocking Diode [g]	1	46		0,05	2	0,05
Wiring AWG 20 [g/m]	5,9	24	1,5	0,21	5	0,22
Wiring AWG 24 [g/m]	4,8	8	1,5	0,06	5	0,06
Cover Glass [g/cell]	0,84	1012		0,85	5	0,89
Adehesive [kg/m2]	0,14			0,55	5	0,58
Kapton [kg/m2]	0,08			0,31	5	0,33
Honeycomb panel [kg/m³]	32			0,78	5	0,82
Substrate panel (carbon-carbon fibre) [kg/m2]	1,2			4,59	5	4,82
TOTALMASS without uncertainty				**10,02**		
TOTAL MASS						**10,44**

Table 24: Dispenser solar array - mass breakdown.

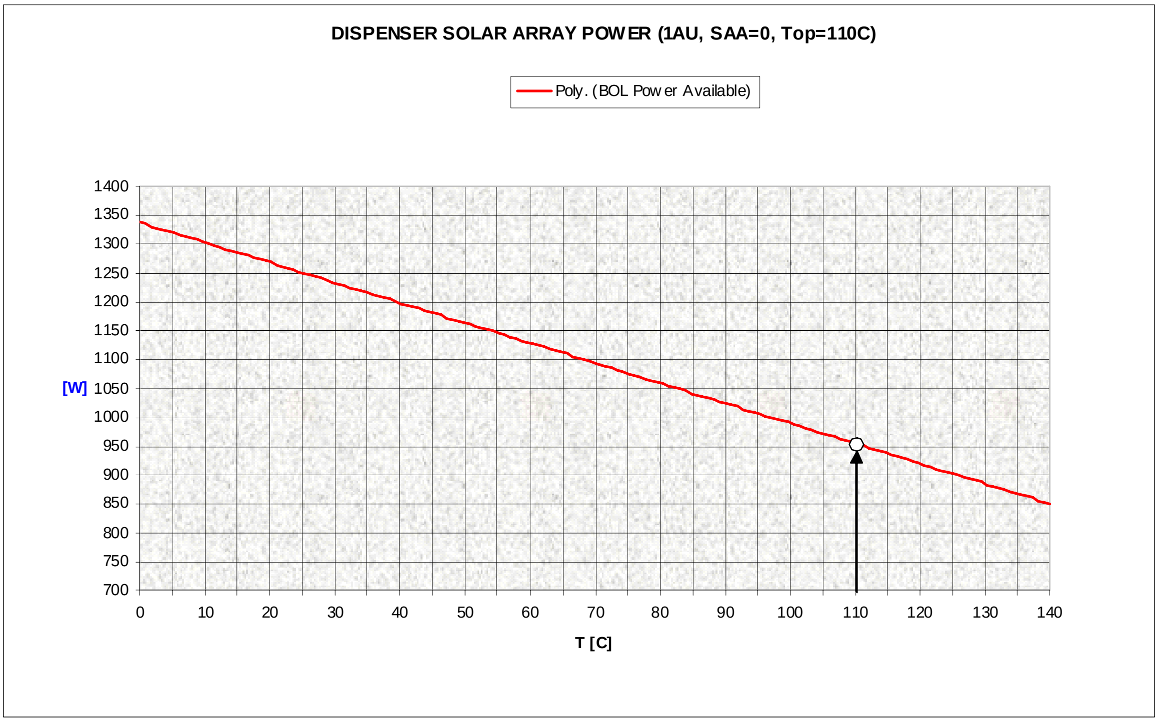

Figure 17: Dispenser solar array available power versus temperature.

7.2.3.2 Dispenser battery

The eclipse phase power consumption is driving the battery sizing (baseline 7.7kg and 6.9dm^3). In order to meet the energy requirement while optimising the battery sizing, the battery charge level as a function of the temperature should be specifically adjusted. Li-Ion technology is used due to its efficiency, simple monitoring, and lack of memory effect. The battery design is based on H/P, Rosetta and Mars Express heritage (SONY 18650HC cells). The SONY Lithium-Ion cell contains built-in safety mechanisms that prevent hazardous results from severe battery abuse, or cell failure.

7.2.4 DISPENSER MECHANICAL DESIGN AND STRUCTURE

The core of the primary structure is a carbon central tube, which holds two large tanks inside, and accommodates the satellites in different structure links to the tube. The tanks are based on existing titanium tanks (SpaceBus) with an additional evolution: the upper and lower tank is linked by a cylindrical part with variable length according to the required propellant mass.
The Cross-Scale TRS satellites are accommodated all around the tube, in 2 stages of 4 satellites, plus an upper stage of 2 satellites. Each stage of satellites is linked to the tube by a set of 2 aluminium rings. The proposed aluminium rings are 7 cm wide and 4.6 mm thick. The load is injected along the circumfence of the tube (proper load case for the cylindrical structure) through the aluminium rings. During launch, the longitudinal accelerations are reacted by the vertical webs, whereas the lateral ones are reacted by the circular panels (see Figure 20). The optimum load

transfer into the shell is achieved by gluing of the aluminium rings onto the tube (standard SpaceBus glue fixation).
Figure 18 shows a lateral view of the total accommodation of the composite for the baseline of 10 satellites. Figure 19 shows the composite within the Soyuz fairing.
The longest available SpaceBus central tube (3977 mm high, 1194 mm in diameter) is suitable for the Cross-Scale TRS in terms of height and mass capacity and is flight proven (TRL 9) for a total load mass of 4.8 tons and qualified up to 5.2 tons. The dimensions of the satellite bodies are 1.0 m in height and 1.5 m in diameter, excluding protruding equipment.
The dispenser subsystem equipment is installed on a CFRP platform at the central tube basis. The solar array is attached on the top of the spacecraft via 6 struts.

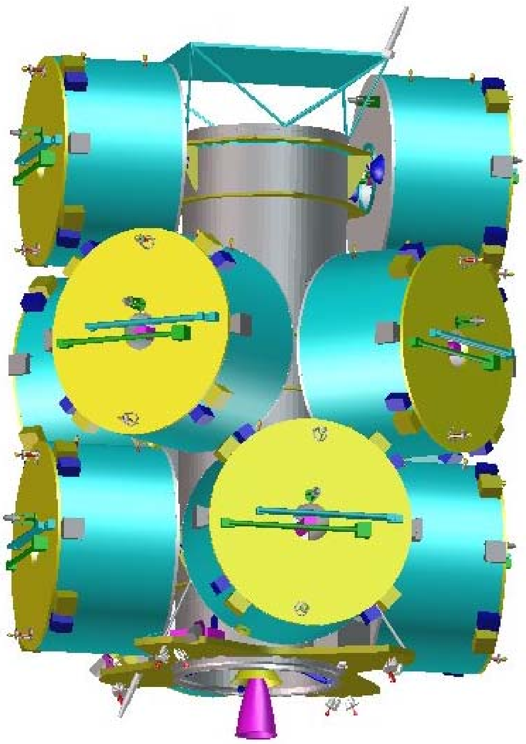

Figure 18: The 10 satellites accommodation. Lateral view.

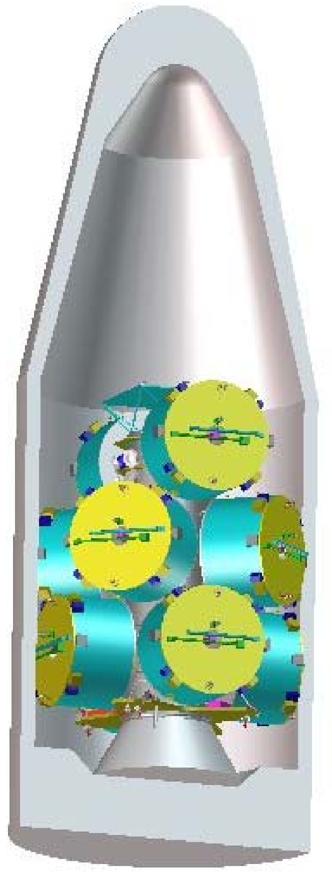

Figure 19: Cross-Scale composite within Soyuz-ST fairing.

7.2.4.1 *Dispenser to satellite interface*

Each satellite is attached to the dispenser in 3 points (see Figure 20). The separation mechanism uses a spring pusher and a pyro release in each point. The spring pushers may be tilted to provide the initial spin during deployment. An allocation of 1 kg per fixation point is estimated (plus 1 kg additional per point in the satellite budget). It is foreseen that the 10 Cross-Scale S/C are released from the dispenser in sequence (one after one).

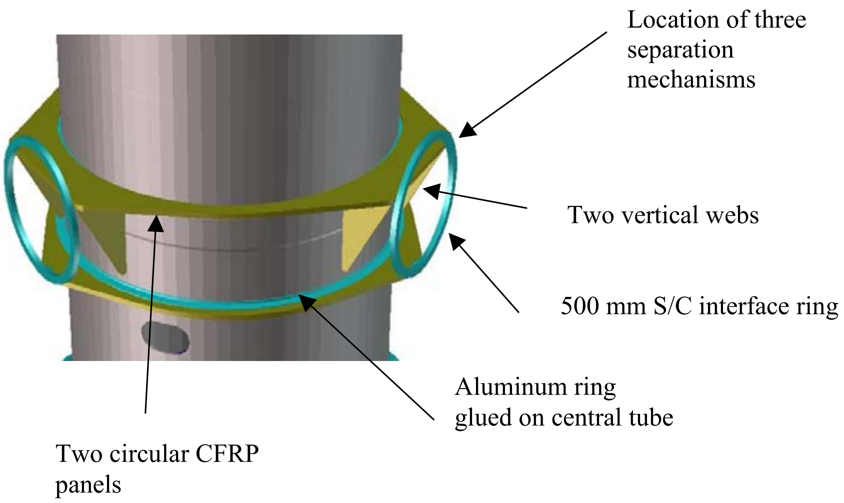

Figure 20: Dispenser to satellite interface.

7.2.5 DISPENSER TT&C AND COMMUNICATION

A solution for the TT&C of the Dispenser could be to use either the same X-band TT&C as the satellites or a standard S-band TT&C sub-system (1.5 to 2 W_{RF} SSPA). To provide an omni-directional link two hemispherical LGAs are used. One is mounted on the platform at the basis of the dispenser tube and the second is accommodated at the top onto the truss structure which supports the dispenser solar array.

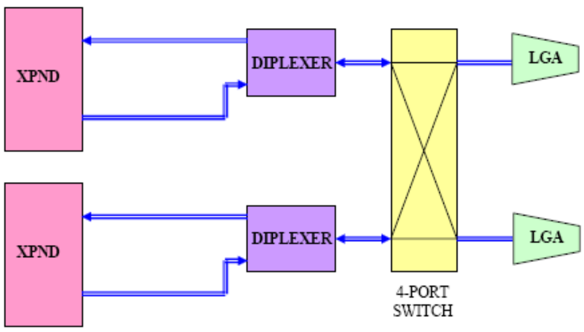

Figure 21: Dispenser TT&C configuration

7.2.6 PROPULSION

The propulsion system is accommodated inside the central tube and on the lower platform (Figure 22). The main engine (enhanced version with 500 N, I_{sp} = 325 s) is accommodated at the base of the tube. A set of 2 x 6 10N thrusters is mounted on the lower platform and additional 2 x 2 10N thrusters are accommodated at the top of the tube.

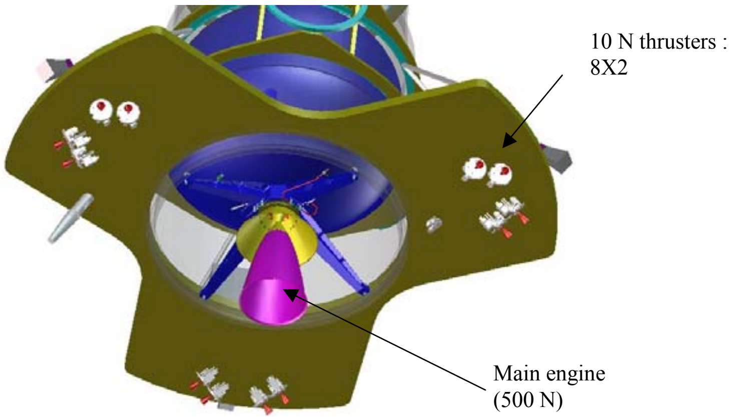

Figure 22: The dispenser thruster and main engine accommodation

7.2.7 DISPENSER MASS BUDGET

FUNCTIONAL SUBSYSTEM	No	Nominal Mass (kg) per Unit	Total Nominal Mass (kg)	Maturity Factor	Maturity Mass Margin	Maximal Mass
		(kg)	(kg)	(%)	(kg)	(kg)
AOCS			31.3	5.0	1.6	32.9
COMMUNICATION			6.1	11.2	0.7	6.8
DATA HANDLING			9.0	20.0	1.8	10.8
MECHANISMS			30.0	20.0	6.0	36.0
POWER			39.3	14.4	5.7	45.0
PROPULSION			121.2	3.1	3.8	125.0
STRUCTURE			195.8	14.6	28.7	224.4
THERMAL CONTROL			5.9	10.0	0.6	6.4
DRY MASS			**438.6**	**11.1**	**48.7**	**487.4**

Table 25: Dispenser mass budget

7.2.8 DISPENSER DISPOSAL STRATEGY

The dispenser needs to be passivated after the release of all 10 S/C by selecting a disposal orbit, controlled re-entry or natural de-orbiting (within less than 25 years). Possible disposal orbits are above the extended GEO region or an orbit between the LEO region and the extended GEO region. For the baseline orbit (1.4 R_E x 25 R_E) a perigee reduction to 150 km by a propulsive manoeuvre in the order of 74 to 77m/s is foreseen followed by natural orbit decay due to the atmospheric drag.

7.3 Spacecraft design

The main requirements to be taken into account in the design of the spacecraft for the Cross Scale TRS are given in Table 26, based on the more detailed requirements given in [AD_MRD]. The constellation baseline includes 10 (most) identical spacecraft.

Parameter	Value
Mission life time	Nominal: 3 years Extended: + 2 years
Number of satellites	8 to 12, baseline 10
Spin rate	15 rpm (orthogonal to the ecliptic)
Max. data rate production	2.5 Mbps for the constellation
Continuous data rate production	400 kbps for the constellation (goal of 800 kbps)
The total down link shall allow to:	down-load 200% of the nominal science data rate, in addition to ancillary data
Configuration of the onboard mass memory	To store data of 1 orbit at 2.5 Mbps Goal: 2 orbits at 2.5 Mbps
Relative timing of science data: • small scale S-C to small-scale S-C • medium-scale S-C to medium-scale S-C • medium-scale S-C to small scale S-C • large-scale S-C to any S-C of the constellation	 • 0.25 ms, goal: 10 to 100 µs • ~distance / 500 000 [/km/sec] , no better than 0.25ms • ~distance / 500 000 [/km/sec] , no better than 0.25ms • 2 ms
Relative distance a posteriori retrieve: • Small-scale to small-scale • Between any S-C of the constellation	 • 125 m • 1% of the distance
• Pointing (2 σ): • APE (absolute) • RPE (reconstructed)	• Axis direction: 1 [deg] • Axis direction: 0.5 [deg] • Phase angle: 0.1 [deg]
Magnetic cleanliness	The Spacecraft shall be compatible with the following electric magnetic cleanliness limits (TBC) a) Magnetostatic cleanliness: 0.5nT and 0.1nT/100s at DC magnetometer location b) AC magnetic field: < 14 dB_{pT} @ 1Hz down to -46 dB_{pT} @ 1kHz and -27 dB_{pT} @ 10kHz at AC magnetometer location. c) Electrostatic cleanliness: < 1 Volt between any two points of the S-C external surface d) Electric field at 1 m distance: <50 $dB_{\mu V}$/MHz above 1 kHz
Radiation environment	64 krad (Si) after 4 mm Al-shielding for $1.4R_E$ * $25R_E$

Table 26: Main spacecraft requirements for the Cross-Scale TRS

7.3.1 GENERAL S/C LAYOUT AND STRUCTURE

The Cross-Scale TRS satellite has the shape of a cylinder with 1.5 m diameter and a height of 1 m. The lower and upper panels are standard honeycomb panels with aluminium skins. These panels are used to support the payload and the servicing equipments. A central carbon tube is used as support between the upper and lower panel. On the lower panel preferable non-active instruments should be mounted, because during cruise configuration the radiative area for this panel is reduced. The upper panel should preferentially accommodate the most thermal dissipative equipment (which needs to be activated during the cruise phase).

At the periphery of the panels a truss of carbon bars links the equipment panels. The solar array panels are fixed to these truss bars. The tanks are accommodated in the lower panel. An aluminium ring, fixed to the lower panel, provides the link to the dispenser S/C interface. An allocation of 3 kg is foreseen for the pyros and the link mechanism. Figure 23 shows the internal accommodation, while Figure 24 shows the external accommodation of the S/C.

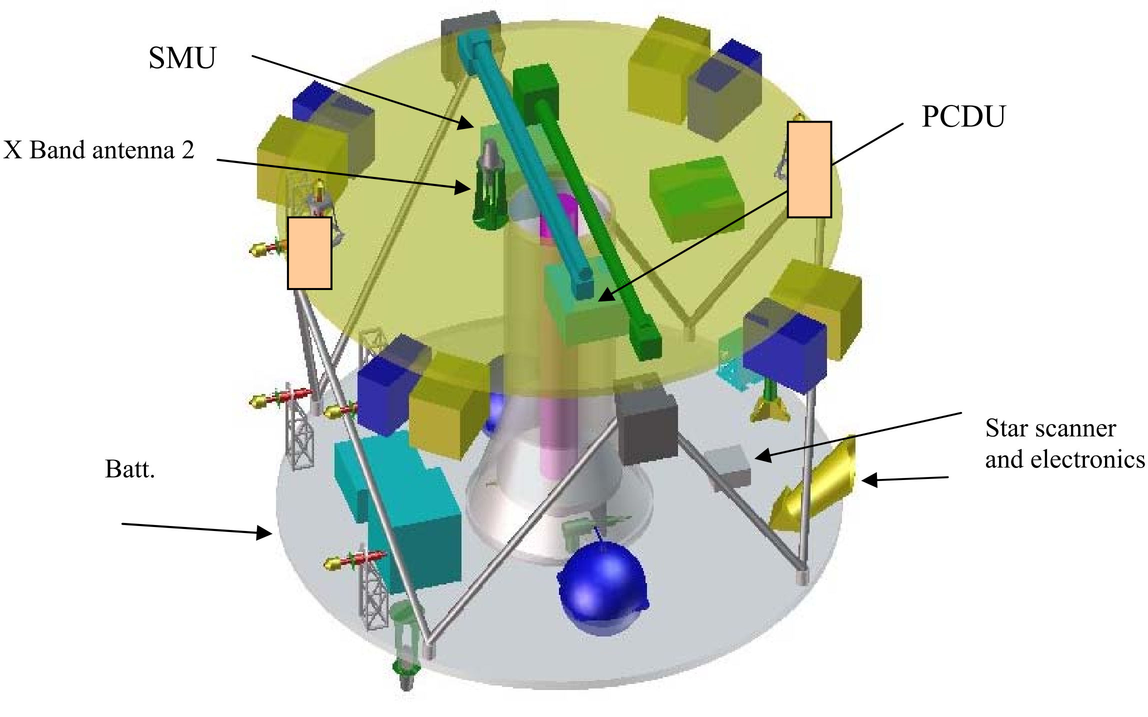

Figure 23: Cross-Scale TRS internal accommodation

The PCDU & SMU are accommodated on the upper panel, because heat needs to be dissipated during the cruise phase. The payload instruments are switched-off during the cruise, as well as the star mapper, which should hence be accommodated on the lower panel. Four thrusters are accommodated on the lateral side and additional two thrusters on the lower panel.

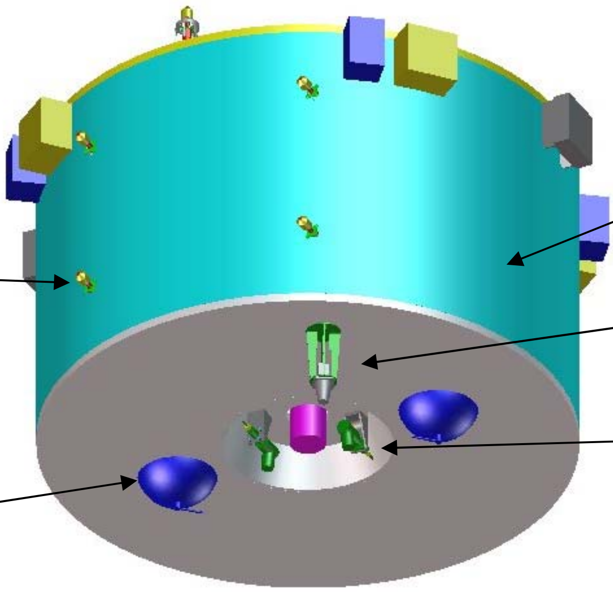

Figure 24: The Cross-Scale TRS external accommodation

7.3.2 PAYLOAD ACCOMMODATION

Table 27 summarizes the Cross-Scale TRS strawman payload for the individual S/C as accommodated on the three scales. Two different configurations are proposed for the electron scale (electron scale e1 and e2) and also on the ion-scale (ion-scale i1 and i2-i4). Only on the large scale all four S/C have an identical instrumentation.

Instrument shortcut	Description	Number of instruments per S/C				
		Electron scale		Ion scale		Large scale
		e1	e2	i1	i2-i4	f1-f4
DCB	Flux-gate magnetometer	1	1	1	1	1
ACB	Search coil magnetometer	1	1	1	1	-
2DE	electric field sensors on 4 radial wire booms	1	1	1	1	1
1DE	AC electric field sensors on 2 axial booms (top/bottom panel)	1	1	-	-	-
LESA	Electrostatic electron analyzer	4 × 2	-	-	-	-
ISA	Electrostatic ion analyzer	-	4 × 2	-	-	-
EISA	Combined electrostatic ion/electron analyzer	4	4	2	4	1
ICA	Ion composition analyzer	-	-	2	-	-
EICA	Energetic ion composition analyzer	-	-	-	2	-
HEP	Energetic particles	-	2	-	2	1

Table 27: Strawman payload for Cross-Scale TRS

The electron scale satellite number 2 (e2) is used to show the accommodation of the payload, as it carries the heaviest payload compliment of all Cross-Scale TRS S/C. The dimensions of the boxes

and fields of view have been modelled, as well as the angular separation of the different instruments on the panel. This accommodation is illustrated in Figure 25 representing only the payload on the upper panel. Due to thermal constraints it might be necessary to use the lower panel too in a final configuration.
The EISA, ISA, HEP instruments field of view for the electron scale e2 accommodation are illustrated in Figure 26.

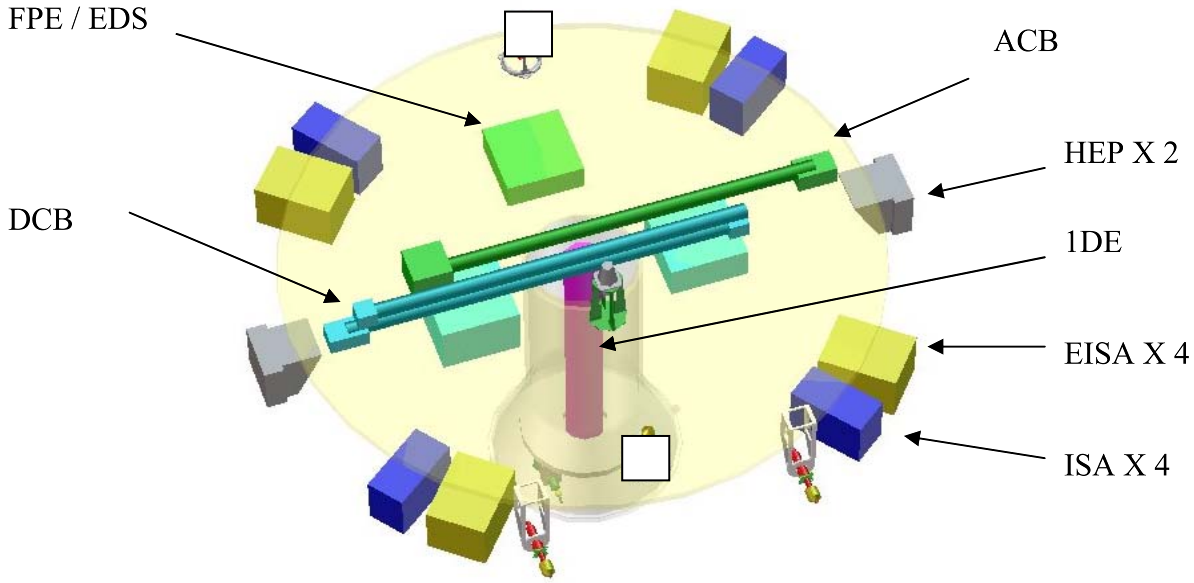

Figure 25: The electron scale e2 satellite payload accommodation.

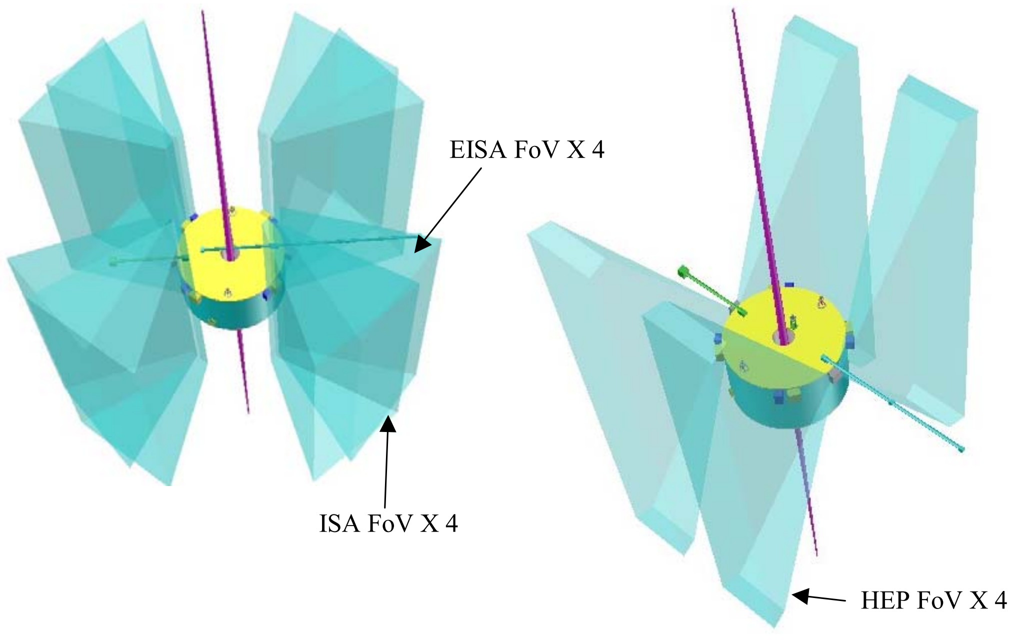

Figure 26: Fields of view (FoV) for the Cross-Scale TRS instruments ISA, EISA and HEP.

The reference plasma physics payload suite assumed for the Cross-Scale TRS consist of established, well-known, plasma field and particle instruments, that have been flown (or are baselined) for numerous plasma physics missions or mission concepts, in particular Cluster II, Geotail, Themis, Magnetospheric Multi-Scale Mission (MMS), and the Japanese SCOPE. More details about the instrumentation of these missions or mission concepts are given in the references in [AD-PLR].

Table 28: Assumed P/L resources for the Cross-Scale TRS

Short-cut	Instrument	Mass [kg]	Mass margin	Mass Total	Power [W]	Remarks
DCB	DC Magnetometer	1.5	10%	1.65	0.5	Incl. ~2 m boom. EMC requirements might drive to longer boom and thus higher instrument mass.
ACB	AC Magnetometer	1.75	20%	2.10	0.1	Incl. ~1 m boom
2DE	E-field 2D	8	10%	8.8	3	4 double probe wire booms (each 30 to 50 m long)
3DE	E-field 3D	12	10%	13.2	5	Dual axial antenna for AC E-field measurements plus 4 wire booms (DC E-field)
EDS	Electron density sounder	0.2	20%	0.24	2.5	Uses E-field instrument. Instrument not operated continuously. Quoted power is peak power.
FPE	AC magnetometer & E-field processor electronics (2D or 3D)	1.5	20%	1.80	2	
LESA	Low energy electron static analyzer	1.5	10%	1.65	1.5	< 40 keV, 3D f(v). Mass TBC as this strongly depends on geometric factor. It should be noted that mass might increase if steerable aperture beam is hard requirement.
ISA	Ion electrostatic analyzer	1.5	15%	1.725	2	20eV – 40 keV: 3D f(v). Up to 100 keV desirable[5] but might require higher mass. It should be noted that mass might increase if steerable aperture beam is hard requirement.
EISA	Combined Electron/Ion analyzer	2.5	20%	3.0	2	Combines above two instruments. Mass strongly depends on geometric factor and energy range. It should be noted that mass might increase if steerable aperture beam is hard requirement.
ICA	Ion composition	5	20%	6.0	6	< 100 keV, 3Df(v) and mass resolution
EICA	Energetic electron and ion composition	1	10%	1.10	0.5	100 keV – 1 MeV, 3Df(v) and limited mass resolution discrimination
HEP	High energy particle detector	1.2	20%	1.44	1	> 30 keV; Electrons and ions (energy resolution only)
CPP	Common payload processor	2	20%	2.40	2	Required for each spacecraft carrying science payload, including harness

A number of critical issues for the development of the payload were identified during the study:

[5] For reconnection and shocks on e-scale S/C.

- Payload provision by institutes will be demanding and collaboration with manufacturing industry might be needed.
- The number of instrument modes need to be limited in order to minimise operations complexity.
- The calibration logistics of the large number of instruments needs to be studied in detail before the start of the payload development as it might have design implications.
- Payload autonomy needs to be implemented. Especially during eclipses and radiation belt crossings the instruments shall not require ground intervention.
- EMC requirements need to be implemented and therefore instruments (as all S/C components) shall be designed such, that the mutual interference is minimal.

7.3.3 PAYLOAD DESIGN PHILOSOPHY

The electrostatic analyzers LESA (1.5 kg) and EISA (2.5 kg) should have identical interfaces to the S/C to allow identical electron scale P/L interfaces on the platform. The same applies to EICA and HEP interfaces on the ion scale S/C. For the implementation of the identical S/C philosophy, each S/C should be able to accommodate the following minimum set of instruments:

- Two magnetometer booms (radial)
- Four wire booms (radial)
- Two axial booms (top and bottom panel)
- 8 slots for electrostatic analyzers (evenly distributed around the S/C):
    - 2 slots with interface to EISA/ICA
    - 2 slots with interface to EISA
    - 4 slots with interface to LESA/ISA/EICA
- 2 energetic particle analyzers

The identical S/C philosophy has an impact on the payload development. Together with the large number of instruments, which require further development, to be designed, built and calibrated, the question arises whether scientific institutes solely will be able of providing the payload to the mission in timely manner. A partnership between scientific institutes and industry capable of handling larger volume manufacturing might be extremely beneficial for the timely delivery of the payload to the prime contractor of the S/C.

7.3.4 AOCS & PROPULSION

All spacecraft of the CS TRS constellation are spin-stabilised with a spin rate of 15 rpm and a spin axis orthogonal to the ecliptic plane. The baseline AOCS sensors are a star mapper unit (SMU) in combination with a sun sensor. An alternative solution would be the replacement of the star mapper with an Earth sensor (potential total mass reduction of ~15 kg). The Earth sensor would need to cope with different sizes of the Earth disk due to the highly elliptical orbit.

Six thrusters (MSG heritage) per S/C are foreseen (including redundancy) to perform all Δv corrections and attitude and spin rate control. Based on inertia of 360 kgm^2, calculations have shown, that thrust levels between 3 to 5 N are required to provide the proper resolution for spin axis orientation and spin rate control. To reduce the nutation of the S/C to a minimum two passive nutation dampers are included in the design. These dissipate energy by viscous friction when S/C oscillations set the damper liquid in motion.

The proposed propulsive element of the S/C is based on a new development in Europe. The MoN-hydrazine hybrid propulsion should be able to deliver an I_{sp} of at least 307 sec, leading to a total mass for the propellant for the 10 S/C of 177 kg, which is reduction of 85 kg compared to a hydrazine propulsion system with an I_{sp} of 215 sec. The baseline propulsion subsystem consists of six identical thrusters delivering thrust levels of 1 to 5 N and having a mass of 0.28 kg each, but this baseline asks for a development programme to design, manufacture and qualify these hybrid thrusters (based on AMPAC 22-N Hydrazine MON-3 DST-11 thruster with a mass of 0.62 kg and I_{sp} of 307 s). An alternative monopropellant system could be considered with a negative impact on the number of possible constellation changes at fluid scale, due to the higher propellant need.

7.3.5 TT&C AND COMMUNICATION

Figure 28 shows the architecture of the CS-TRS communication subsystem. Two low gain X-band omni-directional antennas (LGA) are used to provide a full spherical coverage (Figure 27). Two transponders (XPND) provide the required communication functionalities. RF-power is provided by two travelling wave tube amplifier (TWTA), with 12 W RF-output, around 55% efficiency and a mass of 0.79 kg each (TH4604C tuned at minimum RF power output). Each TWT requires its own electrical power conditioners (EPC).
A dedicated RF-sensor is used to sense the inter-spacecraft distances (at least on e-scale). The RF-sensor signal is coupled via a hybrid coupler and a RF-circulator into the TT&C communication path (see Figure 28), simply to allow the common use of the omni-directional antennas also for the RF-sensor.

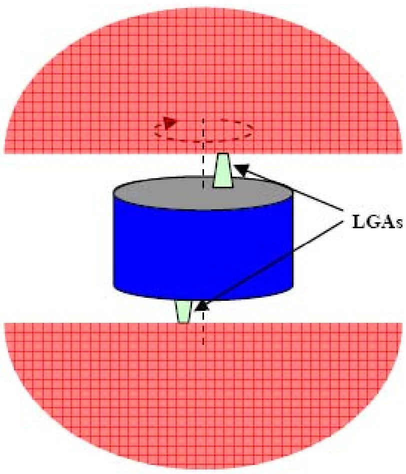

Figure 27: Two hemispherical LGA antennas allow for a full omni-directional communication coverage

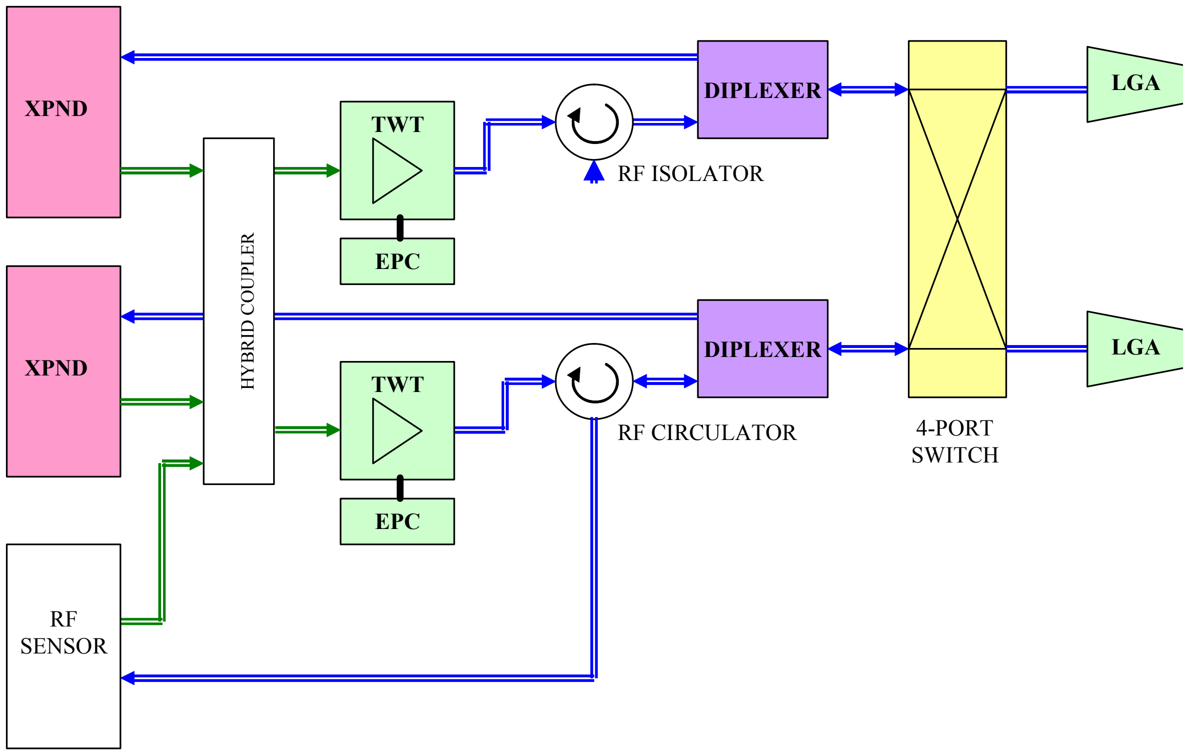

Figure 28: Architecture of the TT&C subsystem on the CS-TRS science satellites

PARAMETER	VALUES	Notes
RANGE [km]	159450	= 25RE
FREQUENCY [MHz]	8450	X-Band - Space-to-Earth Frequency
TX POWER [W]	12	TWTA Thales TH4604C
TX ANTENNA GAIN [dB]	2.5	Helix - Min Gain between 40° and 90°
TX LOSSES [dB]	0.7	2m WG (0.16dB) + 1 SW (0.05dB) + 1 DPLX (0.5dB) = 0.7dB
TX EIRP [dBW]	12.59	Calculated
PATH LOSSES [dB]	215.03	Calculated
ATMOSPHERE LOSS [dB]	0.10	Estimation
RX ANT GAIN [dBi]	59.80	Antenna gain of 15m G/S (e.g. Kourou)
RX POINTING LOSSES [dB]	0.00	TBC (negligible)
RX NOISE TEMP [dBK]	22.30	Required to obtain the specified G/T
RX G/T [dBK]	37.50	RX G/T of 15m G/S (as per ESA indication)
DEMOD. LOSS [dB]	0.00	TBC (negligible)
MOD. LOSS [dB]	0.59	Calculated for a TM mod. index of 1,25
REQIRED Eb/No [dB]	1.20	Turbo Coding 1/2
MINIMUM MARGIN [dB]	3.00	Standard ESA
MAX BIT RATE [kbps]	753	

Table 29: Satellite to ground communication link budget

7.3.6 INTER-S/C LOCALIZATION/SYNCHRONIZATION

The science objectives require good a-posteriori knowledge of inter-satellite distance and time synchronisation of measurements. The requirements are different for the three scales as recalled in Table 30.

Requirement	Value	Ref.
Relative timing of science data: • small scale S-C to small-scale S-C • medium-scale S-C to medium-scale S-C • medium-scale S-C to small scale S-C • large-scale S-C to any S-C of the constellation	• 0.25 ms, goal: 10 to 100 µs • ~distance / 500 000 [/km/sec], no better than 0.25ms • ~distance / 500 000 [/km/sec], no better than 0.25ms • 2 ms	R 2 3 8 1 G 2 3 8 2 R 2 3 8 3 R 2 3 8 4
Relative distance a posteriori retrieve: • Small-scale to small-scale • Between any S-C of the constellation	• 125 m • 1% of the distance	R 2 3 8 6 R 2 3 8 7

Table 30: Requirements for inter S/C localisation determination

The accuracy required for this inter-distance and time synchronisation determination is roughly proportional to the distance between satellites at first order; the most demanding requirements are for the e-scale satellites. Inter-distance measurement and time synchronisation should be performed by on-board equipment for the e-scale satellites, considering that the requirements would be too demanding for ground-based tracking technique only. Also for the medium scale satellites it has been chosen to implement on-board RF-equipment, which will alleviate the ground operations of regular accurate orbit ranging. For the f-scale satellites, no on-board equipment is foreseen, because the satellites are at large distance, which allows for a standard ground-based orbit determination[6].

The satellites exchange messages via the RF-sensor including a coded date and time of emission, identification of emitter and computed distance to the other satellites. The messages are exchanged in TDMA. The RF-sensor developed for the PRISMA demonstrator mission makes use of a high accuracy dual S-band frequency. For the Cross-Scale TRS satellites, a mono frequency RF-signal in X-band is sufficient. By using the X-band for the inter S/C localisation, the TTC on-board equipment can be used (see Figure 28). The use of the TT&C TWTA for the RF-sensor function prevents the simultaneous use for the data download function. The typical duty cycle of the RF inter-distance measurement has been estimated to 3 minutes/hour for those parts of the orbit where the RF-sensor function is used.

[6] For distance monitoring during the deployment of the f-scale satellites from the dispenser dedicated equipment for small range detection might be required (TBC).

7.3.7 ONBOARD DATA HANDLING (OBDH)

More attention needs to be given for the design of the on board data handling and data storage system as this has not been investigated in great detail during the TRS study, as the requirements to the OBDH subsystem where rated to be 'classical' and hence can be handled with standard equipment. For the onboard data storage of 256 Gbit (including 100% margin) a specific memory board is required.

7.3.8 ELECTRICAL POWER SUBSYSTEM (EPS)

The electrical power subsystem consists of the body mounted solar array (SA), a battery and the power distribution and control unit (PDCU). The main constraint from the S/C design is the size of the possible solar array, which can be maximally ~850 x 4620 mm^2 (based on a ~1.5 m S/C diameter). The baseline is a triple junction GaAs cell already qualified for space applications (e.g. Herschel/Planck). The SA design incorporates 6 body-mounted panels placed on the cylindrical face of the S/C. The PCDU distributes regulated power to the S/C bus ($28V_{dc}$) and payload up to 200 W. The total power per S/C needed during the science mode is estimated to 178 W including 20% system margin. During eclipse the maximum needed power is 140 W (including 20% system margin). The proposed battery cell technology is Li-Ion. The Cross-Scale SA sizing takes a number of degradation effects into account to ensure that the EOL performance is 180 W allowing for one potential SA string failure at the worst case operation temperature of 323 K.

Item	Mass [kg]	Dimension [mm]
Solar Array	9.9	6 panels of 850x770
PCDU	6.9	200x150x250
Battery	6.7	300x200x91

Table 31: Electrical power subsystem characteristics

ITEM	Unit mass [g]	Quantity	TOTAL [kg]	Uncert. %	TOTAL [kg]
Solar Array Regulator	500	2	1,00	10	1,10
BCDR	400	2	0,80	10	0,88
Latch Current Limiter	60	12	0,72	10	0,79
Fold-back Current Limiter	60	2	0,12	10	0,13
Heater Switch	50	4	0,20	10	0,22
Capacitor Bank	200	1	0,20	10	0,22
Mother Board	500	1	0,50	10	0,55
Box Scructure, connectors, harness …	2500	1	2,50	20	3,00
TOTAL MASS without uncertanty			**6,04**		
TOTAL MASS					**6,89**

Table 32: PCDU mass budget

ITEM	Unit mass	Quantity	Length [m]	TOTAL [kg]	Uncert. %	TOTAL [kg]
Solar Cell [g]	2,60	816		2,12	2	2,16
Blocking Diode [g]	1	48		0,05	2	0,05
Wiring AWG 20 [g/m]	5,9	24	1,5	0,21	5	0,22
Wiring AWG 24 [g/m]	4,8	8	1,5	0,06	5	0,06
Cover Glass [g/cell]	0,84	816		0,69	5	0,72
Adehesive [kg/m2]	0,14			0,55	5	0,58
Kapton [kg/m2]	0,08			0,31	5	0,33
Honeycomb panel [kg/m³]	32	6		0,80	5	0,84
Substrate panel (carbon-carbon fibre) [kg/m2]	1,2	6		4,71	5	4,95
TOTALMASS without uncertainty				**9,50**		
TOTAL MASS						**9,91**

Table 33: Solar array mass break-down

7.3.9 POWER AND MASS BUDGETS

The S/C power budget is given in Table 34. For all modes of operation including launch, science case and eclipses the power is given for each subsystem including margin. The maximum power figure of 178 W from Table 34 has been used in the dimensioning of the solar array.

	Unit Cont.	SATELLITE MODES: LAUNCH	Cruise stby	SUN ACQUISITION	SCIENCE SUN	ECLIPSE	ECLIPSE	SAFE
Orbit distance			1 AU	1 AU	**1AU**	1AU	1AU	1 AU
AOCS Modes		N/A	Sun Acquisition	Sun Acquisition	**Nominal**	Nominal	Nominal	Sun Acquisition
P/L Modes		Survival	Survival	Survival	**Science**	Science	Survival	Survival
Duration		120 minutes	10 days	TBD minutes	**years**	4 hours	8 hours	days
PLATFORM								
Avionics : (with margin)		37.0	27.5	42.5	**49.7**	49.7	27.5	42.5
RF System : (with margin)		4.4	0.0	6.1	**33.0**	6.1	6.1	33.0
AOCS : (with margin)		0.0	0.0	1.2	**1.2**	1.2	1.2	1.2
AOCS Actuators : (with margin)		0.0	0.0	0.0	**0.0**	0.0	0.0	0.0
Thermal (with margin) alloc. apogee		0.0	10.0	0.0	**0.0**	12.0	24.0	0.0
Power sub-system (with margin)		8.1	40.5	41.1	**27.0**	11.1	8.9	25.0
PCDU	10%	8.1	7.9	8.5	**11.9**	11.1	8.9	9.8
Battery Charge	20%	0	32.6	32.6	**15.12**	0	0	15.12
Platform Total Power Consumption. With margin [W]		**49.5**	**78.0**	**90.8**	**110.8**	**80.0**	**67.6**	**101.6**
INSTRUMENTS								
Payload (with margin)		0.0	0.0	0.0	**33.6**	33.6	0.0	0.0
ESA Margin 20%		**10.1**	**15.8**	**18.5**	**29.6**	**23.3**	**13.9**	**20.8**
Total [W]		**61**	**95**	**111**	**178**	**140**	**83**	**125**

Table 34: S/C power budget for all identified modes of operation

Cross Scale satellites		e-scale	m-scale	f-scale
platform	[kg]	115	115	109
payload	[kg]	**42**	**34**	**13**
dry mass	[kg]	157	149	122
wet mass	[kg]	**163**	**162**	**151**

Table 35: Cross Scale overall satellite mass budget

7.4 Cross Scale System Budgets

The total system mass is given in Table 36. The masses of the e-scale, i-scale and f-scale are separately given. The total mass including 20% margin is within the Soyuz-Fregat performance for the baseline orbit of Cross-Scale. Table 37 summarized the overall mass budget for launcher injection orbit with a reduced altitude of 16164 km, showing a slightly reduced overall performance (0.2 % margin reduction). The final selection of the best orbit strongly depends on the detailed launch performance analysis of Soyuz-Fregat and are here based on 'best knowledge' at available at February 2007.

1.4 Re_25Re 10 sat Soyuz injection: 20164 km		Cross-Scale satellites e-scale	 m-scale	 f-scale	carrier dispenser	
satellites number		2	4	4	1	
delta-V insertion	[m/s]				1411	2.62% gravity loss
delta-V deployment	[m/s]	0.2	18	108		average per sat
delta V maintenance	[m/s]	8	117	432		5 e -maneuvers; 3 m- maneuvers
delta-V disposal	[m/s]	106	106	106	77	
Isp deployment & maintenance	[sec]	308	308	308	325	Sat: Hybrid Mom Hydrazin; Dispenser: 500N
di ergol insert & disposal	[kg]				1167	
(including di ergol disposal)	[kg]				12	
ergol deployment	[kg]	0	1	5		
ergol maintenance & disposal	[kg]	6	11	24		
(including ergol disposal)	[kg]	6	5	4		
Hydrazin ergol spin.	[kg]	0.32	0.32	0.32		
wet mas	[kg]	162	161	151	1657	
dry mass	[kg]	156	148	122	490	
pld mass	[kg]	42	34	13		(mail MdV 12 02 2007)
platform mass	[kg]	114	114	109	490	
structure	[kg]	28	28	28	**490**	total dispenser
structural link / sat	[kg]	3	3	3		part B mecanism
aocs	[kg]	7.0	7.0	7.0		Star Mapper; Accelerometer; SSU, PND
communication	[kg]	9.5	9.5	9.5		2 TWTAs ; 2 EPCs
dhu	[kg]	10.7	10.7	10.7		N & R + CV + 256 GbM
eps harness	[kg]	9	9	9		allocated
power	[kg]	28.2	28.2	28.2		**eclipse 8h @ 5 year**
propulsion	[kg]	8	8	8		6 th ; Tubing & 2 tanks
RF inter sat	[kg]	5	5	0		
thermal	[kg]	6	6	6		paint; mli; radiator; htrs
total mass	[kg]					**3230**
margin	[kg]					**661**
margin	[%]					**20.5**
Soyuz performance into insertion orbi	[kg]					3997 3997.3
including Launcher I-F	[kg]					**110**

Table 36: Total system mass budgets including margin, mass of the e-scale, i-scale (here m-scale) and f-scale satellites (based on a launch into 180 x 20164 km launcher injection orbit).

1.4 Re_25Re 10 sat Soyuz injection: 16164 km		Cross-Scale satellites e-scale	 m-scale	 f-scale	carrier dispenser	
satellites number		2	4	4	1	
delta-V insertion	[m/s]				1715	2.927% gravity loss
delta-V deployment	[m/s]	0.2	18	108		average per sat
delta V maintenance	[m/s]	8	117	432		5 e -maneuvers; 3 m- maneuvers
delta-V disposal	[m/s]	91	91	91	77	for a 500 km deorbiting
Isp deployment & maintenance	[sec]	308	308	308	325	Sat: Hybrid Mom Hydrazin; Dispenser: 500N
di ergol insert & disposal	[kg]				1514	
(including di ergol disposal)	[kg]				12	
ergol deployment	[kg]	0	1	5		
ergol maintenance & disposal	[kg]	5	11	24		
(including ergol disposal)	[kg]	5	5	4		
Hydrazin ergol spin.	[kg]	0.32	0.32	0.32		
wet mas	[kg]	165	164	155	2004	
dry mass	[kg]	160	152	125	490	
pld mass	[kg]	42	34	13		(mail MdV 12 02 2007)
platform mass	[kg]	118	118	113	490	
additional shielding		4	4	4		
total mass	[kg]				3611	
margin	[kg]				732	
margin	[%]				20.3	
Soyuz performance into insertion orbi	[kg]				4453	
including Launcher I-F	[kg]				110	

Table 37: Total system mass budgets including margin, including the masses of the e-scale, i-scale and f-scale satellites into a 180 x 16164 km launcher injection orbit

7.4.1 RADIATION DOSES

The radiation environment for the different orbits has been estimated during the study. For the baseline orbit of 1.4 R_E x 25 R_E, a typical dose of 63 krad (Si) is encountered at the centre of a 4 mm of equivalent aluminium thickness sphere (see also Table 11, page 15). If the average shielding provided by the spacecraft is in the order of 2 mm, an additional payload shielding mass of 7 kg per S/C would be required (based on the assumption of 8 'average' boxes of 200mm x 150mm x 100mm) to stay within the total dose of 63 krad (Si). Spot shielding needs to be applied in certain cases to minimise the overall shielding mass.
For the 4 R_E x 25 R_E orbit the radiation figure is worse (520 krad (Si) after 2mm or 70 krad (Si) after 4mm), where for the 10 R_E x 25R_E orbit the radiation environment is rather relaxed (6 krad (Si) after 4 mm or 30 krad (Si) after 2mm).

7.4.2 ELECTROMAGNETIC CLEANLINESS

The science objective of the Cross-Scale TRS concept includes the measurement of weak magnetic fields. The requirements defined in the payload definition document [AD-PLR] are for magnetic field measurements with the following characteristics:

- 0.5 nT and 0.1 nT over 100 seconds at DC magnetometer location.
- 14 dB_{pT} at 1 Hz down to -46 dB_{pT} at 1 Hz (with a measurement bandwidth of 1 Hz) at AC magnetometer location.

This requires a proper magnetic cleanliness to be defined, following Cluster experience[7]. The same design rules used in the Cluster project are expected to create a design which will meet the specific requirements given. Additional magnetic testing needs to be performed to measure the induced fields at S/C level and eventually to apply magnetic shielding practices and further design improvements to reduce the magnetic field.

7.5 Ground segment

The ground stations selected for the communication analysis are **Maspalomas** in Spain (latitude: +27.76 deg, longitude: -15.63 deg), and **Perth** in Western Australia (latitude: -31.80 deg, longitude: 115.89 deg), which belong to the ESA Core Stations Network.
A multiple-GS network would increase the contact window with the individual CS satellites of the constellation. If two GS are used the contact time can be almost doubled, allowing an increase of the data return or alternatively a reduction of the required power per link. Moving from a scenario with one ground station with a 15-m antenna to a combination of two ground stations with 15-m antennas (e.g. Maspalomas and Perth) at first order allows a reduction of a factor 2 on the required on-board RF power at emission.

In addition, the use of more than one GS could be conducive performing the contact when the S/C are at the best locations with respect to the Earth, i.e. at perigee. 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.
Due to the expected increase of the system noise for smaller GS dishes, a limited number (1-2) of GS endowed with large antennas (15 m) is to be preferred over a system of many small GS. The following Table 38 outlines the advantages and the drawbacks of a multiple-GS network to support the CS satellite mission operations and communications.

Table 38: Advantages and drawbacks of a multiple-GS network

	Advantage	Drawbacks
Multiple GS	- Possibility to increase the data return or to reduce the RF Power on-board - In case of very low RF power, the power amplifier could be included in the transponder saving the mass of the PA box and DC/DC converter - Anomaly handling on instrument / SC easier	- No possibility to reduce the GS antenna diameter, therefore its size shall be the same as in the case of single GS - Higher costs for hiring more than one GS - Needs to handle simultaneous transmission from more than one satellite

Also in the case of two ground stations, the GS contact schedule and S/C to GS link geometry analysis has been performed, taking into account of a 5 min set-up time allocated per S/C for data download preparation to account for the application of the TDMA technique
The effect of the download set-up time is summing up to ~50 min for the whole CS constellation per GS pass, which shortens the effective GS download window.

[7] The use of Li-Ion batteries (which have a significant higher energy density) could become problematic, as Li-Ion batteries have a higher magnetic momentum compared to t he Cluster-II batteries used. Further analysis is required.

7.5.1.1 *Ground station contact for the two baseline ground stations*

The contact intervals between the Cross Scale TRS S/C and Maspalomas and Perth ground stations have been evaluated, results are given in Figure 5 and Table 39. The benefits of using two ground stations located at the antipodes, which naturally improves the average and total GS contact duration and considerably reduces the GS invisibility gap is obvious.

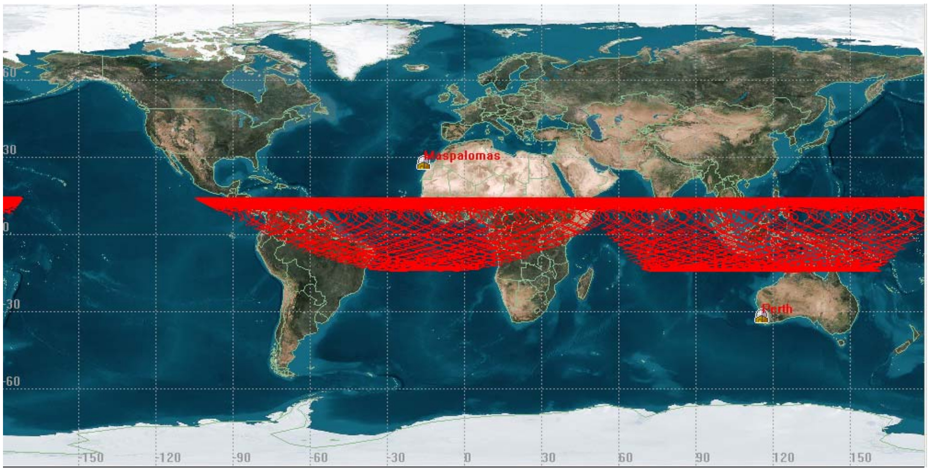

Figure 29: Ground Tracks during the contact intervals with Maspalomas and Perth (min elevation angle = 5°)

GS	Contacts		Min Contact	Max Contact	Average Contact		Tot Contact		Max Gap
	[#/year]	[#/orbit]	[hrs]	[hrs]	[hrs]	[hrs/day]	[days]	[% Time]	[hrs]
No Comms. Set-up Time									
Maspalomas	369	2.84	0.08	15.48	11.31	11.44	173.9155	47.65	34.43
Perth	370	2.85	0.56	13.83	9.25	9.37	142.5759	39.06	36.81
Maspalomas + Perth	432	3.33	0.08	24.44	16.14	19.10	290.4468	79.57	9.38
Comms. Set-up Time ~50 min (whole constellation)									
Maspalomas	359	2.77	0.23	14.67	10.78	10.60	161.2111	44.17	35.27
Perth	357	2.75	0.03	13.00	8.73	8.54	129.9319	35.60	37.63
Maspalomas + Perth	410	3.16	0.03	23.60	16.14	18.13	275.7006	75.53	10.22

Table 39: Contact statistics with Maspalomas and Perth for the CS reference orbit

8 CONCLUSIONS

The Cross Scale TRS has the objective to quantify the coupling in plasmas between different physical scales in order to address fundamental questions such as how shocks accelerate and heat particles, how reconnection converts magnetic energy and how turbulence transport energy.

The described design of the Cross Scale TRS meets the main science requirements of 10 spacecraft to be deployed at different scales at a 1.4 R_E x 25 $R_{E,}$ 14° degree inclined Earth orbit. The total estimated system mass including 20% margin, seems to be compatible[8] with the launch capabilities of Soyuz-Fregat launched from CSG, based on the results of the study with Deimos Space (E), Thales Alenia Space (F) and ONERA (F). A critical assessment of the mass allocations per subsystem is required, as the Cross Scale TRS is rather mass sensitive: every kg mass increase on the single science S/C yields to mass growth with a factor of almost 20 for the wet launch mass[9]. (see also 8.2.1

8.1 Key Technology Developments

A summary of the key technological challenges encountered during the study is provided in Table 40. The definition of Technology Readiness Levels (TRL) is given in Table 41.

Table 40: Critical items and identified technology developments needed for a viable Cross Scale mission

Subsystem	Technology	Advantages	TRL	Remarks
Hydrazine thruster (1 to 5 N)	Increase of the I_{sp} to 308 sec in order to save on the on-board propellant mass	Around 75 kg overall propellant reduction on all 10 satellites	4-5	
Mass memory	256 Gbit mass memory		6	
RF sensor	Reduction of subsystem mass by using one frequency. Reduction of subsystem mass by sharing the frequency with the spacecraft X-band amplifier	Around 3 kg mass decrease (implemented on 6 satellites)	6	
X-band transponder	Reduction of the subsystem mass	Around 2 kg on the transponder	5	US procurement or “higher” mass transponder + SSPA
Dispenser and S/C deployment	Demonstration through analysis and simulation of deployment of 10 S/C without collision.		4-5	

[8] Additional shielding mass, e.g. to gain a 4mm Aluminium equivalent shield around sensitive electronics is not taken into account in the curren overall mass budget. This is required unless equipment can cope with higher radiation levels than 63 krad (for the 1.4 x 25 R_E orbit).

[9] Assuming these increase happens on every science S/C (e.g. subsystem growth), as there are 10 S/C (=factor 10) and the total delta-v requires around the same propellant mass (transfer, reconfiguration and de-orbiting) as the dry mass (= factor ~2) = total factor 20.

System AIV	Assembly , integration and verification of 10 S/C including >100 instruments inline with project schedule			Industry logistics
Instruments	Assembly, integration, verification and calibration of >100 instruments inline with project schedule			Institutes and Industry

Table 41: ESA Technology Reference Level Scale

TRL level	Equivalent Model	Definition
1		Basic principles observed and reported
2	BB	Technology concept and/or application formulated
3	EB	Analytical and experimental critical function and/or characteristic proof-of-concept
4	EM	Component and/or breadboard validation in laboratory environment
5	EQM	Component and/or breadboard validation in relevant environment
6	PFM/FM	(Sub)system model or prototype demonstration in a relevant environment (ground or space)
7		System prototype demonstration in a space environment
8		Actual system completed and “Flight qualified” through test and demonstration (ground or space)
9		Actual system “Flight proven” through successful mission operations

8.2 Project Risks

8.2.1 MASS BUDGET GROWTH

The Cross-Scale TRS mass budget is constrained also due to the fact that the proposed solutions for the sub-systems are based on flight-proven technology where possible (to avoid extra development). Especially with the current state of design it should be possible to perform an early mechanical analysis in particular for the dispenser tube. This activity, which includes the satellite structure definition and modeling, and the dispenser structure definition and modeling, should be run early in the phase-A studies for Cross-Scale. This allows planning of recovery actions when necessary.

Furthermore the impact of a mass increase of one S/C subsystem has a ten-fold increase in the launch mass and additional propellant increase to be taken into account per S/C and the dispenser. A stringent mass and configuration control is hence required.

8.2.2 RADIATION ENVIRONMENT

The radiation environment is relatively high, in particular for low perigee orbits and therefore radiation hardening and shielding design should be taken into account in any subsystems and instrument electronics design.

A large part of radiation damage mitigating can be done by adding shielding material to block high energy proton. The amount of additional shielding mass to ensure safety for the critical components needs to be taken into account in the mass budget (which is not the case in the overall mass budget yet), in particular due to large number of overall subsystems and instruments. By defining the baseline mission profile (orbits and mission duration) a further detailed radiation environment analysis should be done.

8.2.3 INSTRUMENT DEVELOPMENT

The development, design, assembly, integration and verification of more than 100 instruments remain to be a challenge. Most of institutes usually responsible for instrument delivery used to deliver up 4 instruments at most. For Cross-Scale TRS like mission some instruments require up to 18 flight models to be delivered in time. This would need an industrial process and adequate resources to ensure a timely delivery of the instruments to the prime contractor for integration onto the 10 S/C. Delay of instrument deliveries would have a significant negative impact on the overall mission cost.

8.2.4 INSTRUMENT CALIBRATION: IN-FLIGHT AND ON-GROUND

The scientific payload of the 10 spacecraft of Cross Scale TRS need to be calibrated first on-ground, after orbit insertion a in-flight calibration is needed during the commissioning phase (and later as required per instrument). The large amount of instruments can lead to a very long duration for the on-ground and in-flight calibration, which would severely increase the cost of the mission. A streamlined and well monitored approach is needed to ensure a reasonable duration for on ground and in-flight calibration per instrument.

8.2.5 MISSION OPERATIONS

Another critical element in the mission is the missions operations for 10 (+ 1) spacecraft and the science operations for the payload. Full data download of 10 S/C and over 100 instruments require detailed and advanced planning. Fault and anomaly recovery for all spacecraft and instruments could potentially lead to a high ground support cost. With a large number of elements the chances that one element is not working properly is rather high. E.g. based on a reliability of 98% the chance that one of the spacecraft is not working properly during the mission is around 17%. The payload is not taken into account in this assessment and the chances for a payload element breakdown might even be higher. A detailed ground support plan, simulations and training of personnel is needed to ensure a doable way of handling this fleet of spacecraft.

9 LIST OF ABBREVIATIONS

1DE	AC electric field sensors on 2 axial booms
2DE	2D-E field sensors on 4 radial wire booms
ACB	AC-B field search coil instrument
ADCS	Attitude Determination and Control System
BB	Bread Board
COTS	Components of the Shelf
CS	Cross-Scale
CS-TRS	Cross-Scale Technology Reference Study
DCB	DC-B field flux-gate magnetometer instrument
EB	Elegant Bread Board
EICA	Energetic ion composition analyzer
EISA	Combined electrostatic ion/electron analyzer
EM	Engineering Model
EMC	Electromagnetic Compatibility
EQM	Engineering Qualification Model
ESD	Electrostatic Discharge
FM	Flight Model
GTO	Geostationary Transfer Orbit (defined here as 180 km × 35,786 km)
HEO	Highly Elliptical Orbit
HEP	Energetic particles
ICA	Ion composition analyzer
ISA	Electrostatic ion analyzer
LEOP	Launch and Early Orbit Phase
LESA	Electrostatic electron analyzer
LV	Launch vehicle
OBDH	On-Board Data Handling
PCDU	Power conditioning and Distribution Unit
PFM	Proto Flight Model
RAAN	Right Ascension of Ascending Node (Ω)
R_E	Earth's radius (6,378 km)
ROM	Rough order of magnitude
S/C	Spacecraft
SF	Soyuz-Fregat
SMU	Star Mapper Unit
TBC	To be confirmed
TBD	To be determined
TRL	Technology Readiness Level
TRP	Technology Research Programme
TRS	Technology Reference Study
TV	Transfer Vehicle

10 REFERENCES

[CoC04]	"European Code of Conduct for Space Debris Mitigation," issue 1.0, 28 June 2004, available from http://www.stimson.org/wos/pdf/eurocode.pdf
[ECSS00]	"Space engineering: Space environment," European Cooperation for Space Standardization, ECSS-E-10-04A, January 2000.
[ESOC-WP510]	J. Castro, Cross Scale Mission Analysis Global Orbit Properties, issue 1. rev.0 , 21-Jun-2007
[ESOC-WP511]	J.Schoenmaekers, Cross Scale Mission Analysis Transfer Using Moon Resonances, issue 1. rev.0 , 22-Jun-2007
[Margin05]	"Margin philosophy for assessment studies," SCI-A/2003/302/AA, issue 2.0, 7 April 2005.
[Soyuz01]	User manual for Soyuz-Fregat, ST-GTD-SUM-01, issue 3, revision 0, April 2001.
[Soyuz06]	"SOYUZ from the Guiana Space Centre, User's Manual," Issue Draft – January 06 (including annex 1). See www.arianespace.com/site/documents/soyuz_man_csg_index.html
[Tsuda05]	Y. Tsuda, K. Maezawa, H. Kojima et al., "Mission design of SCOPE – small satellites formation flying mission for magnetospheric tail observation," paper presented at AIAA International Conference on Low Cost Planetary Missions, Kyoto, 2005.