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TECHNICAL NOTE

L3 Call For Missions - Technical Annex

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DISTRIBUTION

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1 INTRODUCTION

1.1 Scope of document

This Annex provides technical inputs for the preparation of the proposals in answer to the Cosmic Vision L3 Call. Its main objective is to provide technical information to support the proposers in defining their mission concept to the level required to enable the evaluation of the mission's technical feasibility, as required by the Call for Missions.
Reference information from previous ESA missions, relevant to the preparation of the proposals as technical heritage, can be found at: http://sci.esa.int/home/51459-missions/.

1.2 Reference documents

[RD-01]Ariane 6 User's Manual, issue 0.0, May 2016.
[RD-02] ECSS-U-AS-10C Adoption Notice of ISO 24113: Space systems - Space debris mitigation requirements
[RD-03]ECSS-E-HB-11A DIR1, TRL guidelines, 2016.
[RD-04]ECSS-E-AS-11C, Adoption notice of ISO 16290, 2014.
[RD-05]ESTRACK facilities overview: DHSO-ESTR-OPS-TN-1001-HSO-ONI.
[RD-06] LISA Assessment study report ESA/SRE(2011)3 (available on the sci.esa.int webpages)

1.3 List of acronyms and abbreviations

Acronym	Definition
AU	Astronomical Unit
CaC	Cost at Completion
DV	Delta-V
ESA	European Space Agency
ECSS	European Cooperation for the Standardization of Space
EoL	End of Life
GA	Gravity Assist
GEO	Geostationary Earth Orbit
GTO	Geostationary Transfer Orbit
HGA	High Gain Antenna
ISO	International Standardisation Organisation
Isp	Specific Impulse
LISA	Laser Interferometer Space Antenna
MJD	Modified Julian Date
MOC	Mission Operations Centre
RF	Radio Frequency

S/C	Spacecraft
SOC	Space Operations Centre
SYLDA	Système de Lancement Double Ariane
TBC	To Be Confirmed
TBD	To Be Determined
TRL	Technology Readiness Level

1.4 General Guidelines

The L3 Call is targeting a Cost at Completion (CaC) to ESA of 1050 M€ (e.c. 2016) with a development schedule of ~16-18 years from the Call to launch (2016 until 2034). The purpose of this technical annex is to support the proposers in defining mission profiles that are compatible with these programmatic boundaries.
General guidelines are summarised in the Table 1.

Element	Recommendation	Comment
ESA CaC	≤ 1050 M€	Exclusive of MS participation (e.g. to payload elements) and international collaboration.
TRL	TRL ≥ 6 by mission adoption in 2024	ISO scale, see Appendix A.
Schedule	Launch by 2033-2034	Mission adoption targeted by 2024
International collaboration	Open	ESA-led mission; European back-up solutions shall be identified for mission enabling elements.
Launcher	In case the launcher is provided by ESA, use of ESA new launcher family (VEGA-C or Ariane 62/64)	See Chapter 2 for further details on launcher availability, launcher performance and potential mission profiles. Other launchers depend on international collaboration scenarios.
S/C constraints	No specific constraint beyond programmatic feasibility	See Chapter 2 for system design aspects. Also depends on eventual international collaboration scenario and launcher selection.
Operations cost	< 15% of CaC	Operations costs include spacecraft and Science operations. Nominal operations are not including potential mission extensions, which should be addressed by the proposers for the spacecraft sizing.
Nominal operations	Typically ~3-4 years	

Table 1: L3 Call general guidelines.

1.4.1 Launcher

ESA provision of the launcher is anticipated to be the probable baseline scenario. Based on previous mission concepts (which are TBC), the two launchers that could be envisaged for

L3 are Ariane 6.2 and 6.4. Shared Ariane 6.4 launches are not excluded, but these would carry a number of additional constraints (e.g., the availability of a suitable launch opportunity at the desired time) which need to be clearly addressed.

1.4.2 Space segment constraints

The CaC target implies an ESA industrial contract for the space segment in the range of 600 M€ (e.c. 2016,) including any ESA provision of payload elements. The actual allocation to the space segment depends on the various mission assumptions: launcher, nominal operations, contingency, etc.

Proposers are advised not to make a priori assumptions about possible industrial set-ups for the procurement of the spacecraft in their proposal, e.g., by claiming drastic reductions of spacecraft development costs through heavy reliance on existing developments. While this may appear to minimise industrial costs, experience shows that this is likely to underestimate the actual spacecraft cost for several reasons. These include the fact that 1) recurring costs are valid only if a full and true recurrence is reached for the product, while experience shows that in practice science missions almost invariably require specific adaptations and non-recurring costs, 2) the industrial landscape can evolve over several years and invalidate the assumptions underlying recurring costs, and 3) component obsolescence over the timescale of the implementation of the L3 mission will inevitably impose re-design and nonrecurring costs. Additionally, a pre-defined industrial organisation scheme may not be compatible with the Science Programme constraints at the time of the L3 implementation. While the Agency will explore in due time all means to minimise the spacecraft development costs, including the use of recurring developments, the mission proposals shall be robust against implementation schemes. Therefore, it shall assume a non-recurring space segment development and shall not rely on any specific industrial organisation.

1.4.3 Mission operations

The detailed mission operation costs (Mission Operations Centre and Science Operations Centre) will be mission-dependent. A reasonable cap to the operations cost is 15% of the CaC (for MOC and SOC, including margins). The nominal in-orbit lifetime is suggested to be in the range of 3-4 years, but longer operation phases can be envisaged either by lowering the amount allocated to the space segment or through extensions of the mission beyond the nominal operation phase.

2 MISSION AND SYSTEM CONSIDERATIONS

2.1 General Remarks for Mission Options

This technical annex assumes as reference space segment architecture, a constellation of three spacecraft placed at the vertices of an equilateral triangle in space with a side length of around 2.5 million km ("cartwheel" configuration).

Depending on the measurement concept proposed and on the acceptable variations in S/C relative position and velocity, a large trade-space of orbit options could be envisaged, such as Earth-bound, large amplitude Halo orbits at Sun-Earth L1/L2, orbits at Sun-Earth L4/L5 or orbits at Earth-Moon Lagrangian points.

Where applicable, some basic information for these options is provided. However, we focus our presentation here on previous LISA studies and on the GOAT recommendations.

This implies the centre of the spacecraft triangle configuration is in a heliocentric Earth trailing or leading orbit.

In principle, for this configuration, different values are possible for the trailing or leading angle (the angular distance of the centre of the spacecraft configuration from the Earth along the Earth orbit) and for the spacecraft constellation angle with respect to the Earth orbit.

The reference constellation orbit is shown in Figure 1 below.

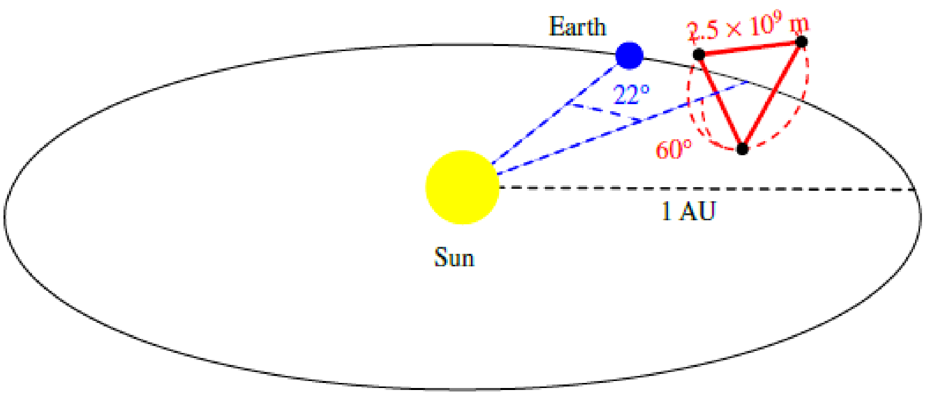


Figure 1: Reference constellation orbit

Example Target Constellation

Based on the GOAT report, an example target constellation has been set-up preliminary, in order to calculate first estimates of transfer durations and delta-V requirements. This target constellation is based on a 2.5 Mkm armlength and is trailing the Earth by about 18 degrees at the start of mission. The evolution of the orbit is provided in Figure **2** for up to 10 years.

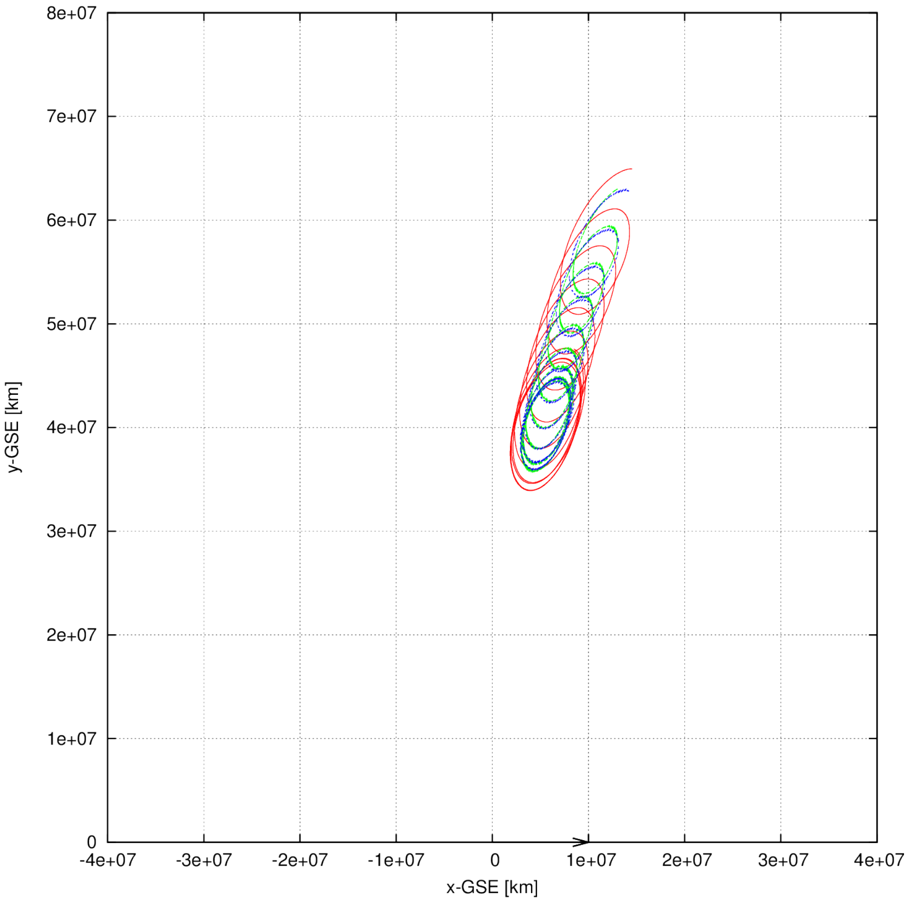


Figure 2: Example constellation dynamics in the Geocentric Solar Ecliptic frame. The Earth is at [0,0]. The constellation starts close to the Earth (18 degree initial trailing angle) and then slowly moves away has been propagated for up to 10 years.

If considering a standard commercial GTO, the launch parameters are more favorable for an Earth heading configuration, i.e. the constellation is placed in front of the Earth on its orbit. For this case, a 15 degree trailing configuration has been derived, also offering suitable evolution over 10 years.

2.2 Launch and Transfer Scenarios

There are also several options to reach the reference orbit in 2.1. They differ by the launch orbit and the means used for transfer from the launch orbit to the final orbit.

The first possibility is to launch the three spacecraft together (e.g. in a single launch in Stack configuration) into escape trajectory with a small hyperbolic velocity.
The S/C are then separated each other and drift into the final orbit. A plane change and stop-drift manoeuvres are required to achieve the final constellation. The actual delta-V of these manoeuvres is slightly different for each S/C and it is performed by the S/C Propulsion Modules, which are jettisioned at arrival.

A second option is to launch into an Earth High-Elliptical Orbit (HEO) for a lunar transfer and subsequently use Moon fly-bys to send the three S/C into final orbit at different times.

This allows saving part of the transfer delta-V (mainly the inclination change required) but may still require delta-V for stop-drift manoeuvres. This option is sensitive to the launch date.

Finally, the launch of the stack of three S/C could be into GTO (with or without another passenger for sharing the cost, depending on the mass of the stack and on the launcher). From GTO, the final orbit can be reached again by a series of propulsive manoeuvres assisted and most likely moon fly-bys.

The following table provides the available launch mass, i.e. the mass of the spacecraft and all adapters, for different launch scenarios that are compatible with reaching an Earth trailing or leading heliocentric orbit. Transfers are split into “direct” and “Moon Gravity Assist” options. A direct transfer implies that all required delta-V to reach the target orbit will be applied by the spacecraft. A Moon Gravity Assist (GA) will enable to use a fly-by manoeuvre at the Moon to lower the delta-V required on the spacecraft, e.g. for an inclination change.

A shared launch on Ariane 6.4 could be envisaged, where L3 would constitute the secondary passenger under the SYLDA adapter, with the main passenger on top of SYLDA.

Launch Into	**Transfer**	**Launch Parameters**	**Launch Mass capability**		
			A6.2	**A6.4 Shared**	**A6.4 Dedicated**
GTO	Direct	Standard GTO Service	5 000 kg	< 7 000 kg, assuming 500 kg for SYLDA adapter and > 3 500 kg for the passenger	11 000kg
	Moon GA	TBD	5 000 kg	< 7 000 kg, assuming 500 kg for SYLDA adapter and > 3 500 kg for the passenger	11 000kg
Escape	Direct	Perigee Alt.: 170 km V_∞: 300 m/s Declination: -0.63 deg AP.: 177.8 deg RAAN: -128.8 deg TA: 17.05 deg	TBD	N/A	7 000 kg

	Moon GA	Perigee Alt.: 180.6 km Apogee Alt.: 150 000 km Inclination: 6.55 deg AP.: 180.0 deg RAAN: -121.1 deg TA: 19.24 deg	2 300 kg	N/A	7 700 kg

2.3 Transfer durations, data rates and power considerations

The following table provides typical transfer durations for various target orbits. These apply regardless of the launcher used.

Orbit	Typical transfer duration
Moon	< 1 week (direct transfer) ~70 – 130 days (low energy transfer)
Sun Earth L1/L2	~1 month
Trailing orbit (18 degree trailing example provided here) via direct transfer	11 – 12 months
Heading orbit via GTO (15 degree heading)	10 months

Table 2: Typical transfer durations

For the specific case of Earth trailing orbits, these orbits are achieved with an initial Earth escape manoeuvre into a hyperbolic trajectory, followed by a final stopping manoeuvre (if no drift is acceptable).
For drifting Earth leading/trailing orbits, i.e. orbits where the spacecraft slowly drifts away from the Earth, there is no, or only a small, arrival manoeuvre required. The only ΔV to consider is the one required to reach Earth escape velocity, with a $C3 \geq 0$ km^2/s^2.

One could imagine optimized scenarios using moon gravity assist manoeuvres to perform the required inclination changes, thus lowering the delta-V requirements by roughly 300 m/s (order of magnitude).

Another option to consider is using electric propulsion to reach the target constellation. The delta-V cost in this case (low-thrust trajectory design) is estimated in the order of 2000 m/s. The advantage with electric propulsion transfers would be the significantly smaller propellant mass to be carried, thus allowing an optimized spacecraft design.

Scenario	Required transfer delta-V	Note:
GTO – direct Transfer into operational orbit (heading)	~2000 m/s	Subject to optimization
Escape – direct Transfer into operational orbit (trailing)	<900 m/s	Subject to optimization, not taking into account moon gravity assists.
Escape – direct transfer into operational orbit (trailing – electric propulsion)	~<2000 m/s	Subject to optimization

Table 3: Typical transfer delta-Vs to a trailing/leading orbit.

2.4 Power, Data transmission and link budget considerations

Power can be generated using solar arrays on the spacecraft. Since the orbit is placed at 1 AU, performance of this solution is comparable to power generation in Earth orbits. A quick overview of the calculation of available power is given in Table 4.

The communication link budget and the achievable data rates are primarily a function of the communication subsystem output power and of the emitting and receiving antennae diameters. For a given receiver noise and coding performance, the data rate scales as:

$$\text{Data Rate} \propto P.(D_t/\lambda)^2.\,(D_r/\lambda)^2.\,(\lambda/d)^2$$

where:

- P is the communication subsystem emitted power
- D_t (resp. D_r) is the diameter of the transmitting (resp. receiving) antenna
- λ is the communication wavelength
- d is the distance between the spacecraft and the ground station

The mission will be classified as Space Research Service (SR) and Category B (i.e. having an altitude above Earth greater than $2*10^6$ km) according to ITU radio regulations.
See ECSS-E-ST-50-3C for relevant bandwidth limitations in the different frequency bands. In particular, note that new assignments in the band 2110-2120 MHz ("S-band uplink for Deep Space") are formally discouraged.

For heliocentric orbits at approximately 20 degree in trailing and/or leading formation, a data rate of >150kbps in Ka-band using the ESTRACK 35m stations as described in Section 2.5 can be assumed.

Orbit	Typical science TM data rates	Power
Moon	X band: ~5-10 Mbps	@ 1 AU Solar radiation: ~1300 W/m2 Cosine loss for 36° off-pointing: 80% Cell efficiency: 28% System losses: 85% Cell packaging ratio: 70% Ageing: 86% (@ 3.75%/year for 4 years) ~150 W/m2 at EoL
Sun Earth L1/L2	X band: 5-10 Mbps K band: 75 Mbps	
Leading/trailing heliocentric orbit	X-band: 20-40 kbps Ka band: ~150 kbps	

Table 4: Typical TM data rates and power generation for potential orbits.

2.5 Ground station characteristics

The reference for ground stations is the ESA ESTRACK network (details in [RD-05]). This network is currently in evolution, with e.g. some 15 m stations being retired from service or handed over to third parties. Considering the M5 timescale, the following stations in the Core Network can be assumed:

Name	Antenna diameter [m]	Frequencies (Tx /Rx)	Note
Cebreros-1	35	X/X Ka	Includes capability in the 25.5-27 GHz band
Malargue-1	35	X/X Ka	
New Norcia-1	35	S X/S X	Complemented by 4.5 m Acquisition Aid Antenna in X-band for LEOP
Kourou-1	15	S X/S X	For LEOP/transfer

Table 5: ESTRACK Core Network ground stations likely available in the L3 timeframe.

Additionally, stations from the Augmented Network consisting of commercially-owned antennas can also be considered for LEOP (Table 6).

Name	Antenna diameter [m]	Frequencies (Tx / Rx)	Note
South Point (Hawaii)	13	S X/S X	
Santiago (Chile)	9	S/S	
Dongara (Australia)	13	S /S X	8000-8500 MHz RX X-band
Svalbard (Norway)	13	S /S X	7500-8500 MHz RX X-band
Troll (Antartica)	7.3	S X/S X	

Table 6: ESTRACK Augmented Network ground stations potentially available in the L3 timeframe.

Finally, stations from the Cooperative Network consisting of antennas owned by Cooperating Space Agencies could also be considered (preferably as back-ups only or during critical operations such as LEOP). Their availability should be explicitly confirmed by the owning entity.

2.6 Space debris regulations

All ESA missions (see reference [6]) have to ensure that no additional orbital debris will contaminate the protected regions (in yellow in Figure 3). The practical consequence is the need to implement a propulsion subsystem, even when using low-Earth orbits, for either moving the S/C into graveyard orbits at its end of life, or to ensure its re-entry in the atmosphere within a specified maximum duration of 25 years.

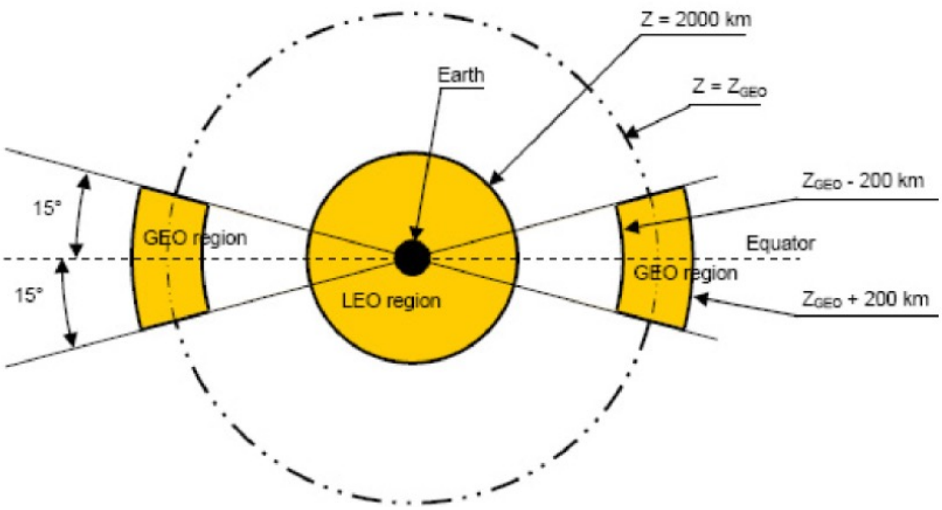


Figure 3: LEO and GEO protected regions [6].

When fragments of the S/C may survive the re-entry (typically for large missions), a controlled re-entry manoeuvre has to be performed to mitigate the risk of ground casualty. For small missions (typically < 1000 kg), an un-controlled re-entry is acceptable, as long as it happens within 25 years.

This requirement applies to the S/C, as well as to any other debris generated by the mission, such as LV upper stages, multi-S/C adapters, ejectable covers etc.
The ΔV required for this manoeuvre will need to be included in the sizing of the propulsion subsystem. As a worst-case estimate, this ΔV can be calculated as follows:

- For an un-controlled re-entry manoeuvre, the perigee of the last orbit should be lowered to an altitude ≤ 60 km. Depending on the initial orbit, more efficient solutions might include placing the S/C into a higher graveyard orbit, or into a very low circular orbit with a Hohmann transfer and let atmospheric drag lower the altitude naturally until re-entry is achieved within 25 years (this depends on the Solar activity, but typically requires lowering the spacecraft altitude to ≤ 550 km by using the on board propulsion system at the end of life).
- For a controlled re-entry manoeuvre, the perigee of the last orbit should be lowered to an altitude of 0 km.

APPENDIX A - TRL DEFINITION (ISO SCALE)

Technology Readiness Level	Milestone achieved for the element	Work achievement (documented)
TRL 1: Basic principles observed and reported	Potential applications are identified following basic observations but element concept not yet formulated.	Expression of the basic principles intended for use. Identification of potential applications.
TRL 2: Technology concept and/or application formulated	Formulation of potential applications and preliminary element concept. No proof of concept yet.	Formulation of potential applications. Preliminary conceptual design of the element, providing understanding of how the basic principles would be used.
TRL 3: Analytical and experimental critical function and/or characteristic proof-of-concept	Element concept is elaborated and expected performance is demonstrated through analytical models supported by experimental data/characteristics.	Preliminary performance requirements (can target several missions) including definition of functional performance requirements. Conceptual design of the element. Experimental data inputs, laboratory-based experiment definition and results. Element analytical models for the proof-of-concept.
TRL 4: Component and/or breadboard functional verification in laboratory environment	Element functional performance is demonstrated by breadboard testing in laboratory environment.	Preliminary performance requirements (can target several missions) with definition of functional performance requirements. Conceptual design of the element. Functional performance test plan. Breadboard definition for the functional performance verification. Breadboard test reports.
TRL 5: Component and/or breadboard critical function verification in a relevant environment	Critical functions of the element are identified and the associated relevant environment is defined. Breadboards not full-scale are built for verifying the performance through testing in the relevant environment, subject to scaling effects.	Preliminary definition of performance requirements and of the relevant environment. Identification and analysis of the element critical functions. Preliminary design of the element, supported by appropriate models for the critical functions verification. Critical function test plan. Analysis of scaling effects. Breadboard definition for the critical function verification. Breadboard test reports.
TRL 6: Model demonstrating the critical functions of the element in a relevant environment	Critical functions of the element are verified, performance is demonstrated in the relevant environment and representative model(s) in form, fit and function.	Definition of performance requirements and of the relevant environment. Identification and analysis of the element critical functions. Design of the element, supported by appropriate models for the critical functions verification. Critical function test plan. Model definition for the critical function

Technology Readiness Level	Milestone achieved for the element	Work achievement (documented)
		verifications. Model test reports.
TRL 7: Model demonstrating the element performance for the operational environment	Performance is demonstrated for the operational environment, on the ground or if necessary in space. A representative model, fully reflecting all aspects of the flight model design, is built and tested with adequate margins for demonstrating the performance in the operational environment.	Definition of performance requirements, including definition of the operational environment. Model definition and realization. Model test plan. Model test results.
TRL 8: Actual system completed and accepted for flight ("flight qualified")	Flight model is qualified and integrated in the final system ready for flight.	Flight model is built and integrated into the final system. Flight acceptance of the final system.
TRL 9: Actual system "flight proven" through successful mission operations	Technology is mature. The element is successfully in service for the assigned mission in the actual operational environment.	Commissioning in early operation phase. In-orbit operation report.

Table 7: Summary definition of the ISO TRL levels, taken from [7] (contains guidelines for the interpretation and implementation of the TRL requirements defined in [10] based on ISO 16290).

APPENDIX B – C3 DEFINITION

In the two-body Newtonian gravitation approximation, the orbital velocity is defined as:

$$V = \sqrt{\mu.\left(\frac{2}{r} - \frac{1}{a}\right)}$$

where:

- V is the orbital velocity
- r is the distance from the centre of the celestial body to the S/C
- μ/r is the gravitation potential
- a is the semi-major axis of the orbit (assumed to be a conic, with the convention a < 0 for the hyperbolic case)

The orbit parameter C3 is defined as:

$$C_3 = -\frac{\mu}{a} = V^2 - \frac{2.\mu}{r}$$

C3/2 is the specific energy of the orbit. C3<0 for elliptical orbits, C3 = 0 for the parabolic orbits and C3>0 for hyperbolic orbits.
For hyperbolic orbits, we also have $C_3 = {V_\infty}^2$, where $V_\infty = \lim_{r\to\infty} V$ is the velocity at infinity, also referred to as the hyperbolic departure or escape velocity ($V_\infty = 0$ for the parabolic limit). Therefore, when applying the above formulas to the two-body system defined by the Earth and the spacecraft, C3 provides the escape velocity in the Earth referential frame. For obtaining the spacecraft velocity in the heliocentric referential frame, the Earth orbital velocity must be added to V_∞.
Exact C3 calculation must take into account the orbit inclinations and the actual arrival date.
Note that transfers may involve gravity assists manoeuvres to reduce the escape velocity required.

APPENDIX C – A6 FAIRING AND ADAPTER

Fairing dimensions for A6 are illustrated in Figure 4 and Figure 5. Note that the launcher is still under development and all dimensions should be taken with margins.

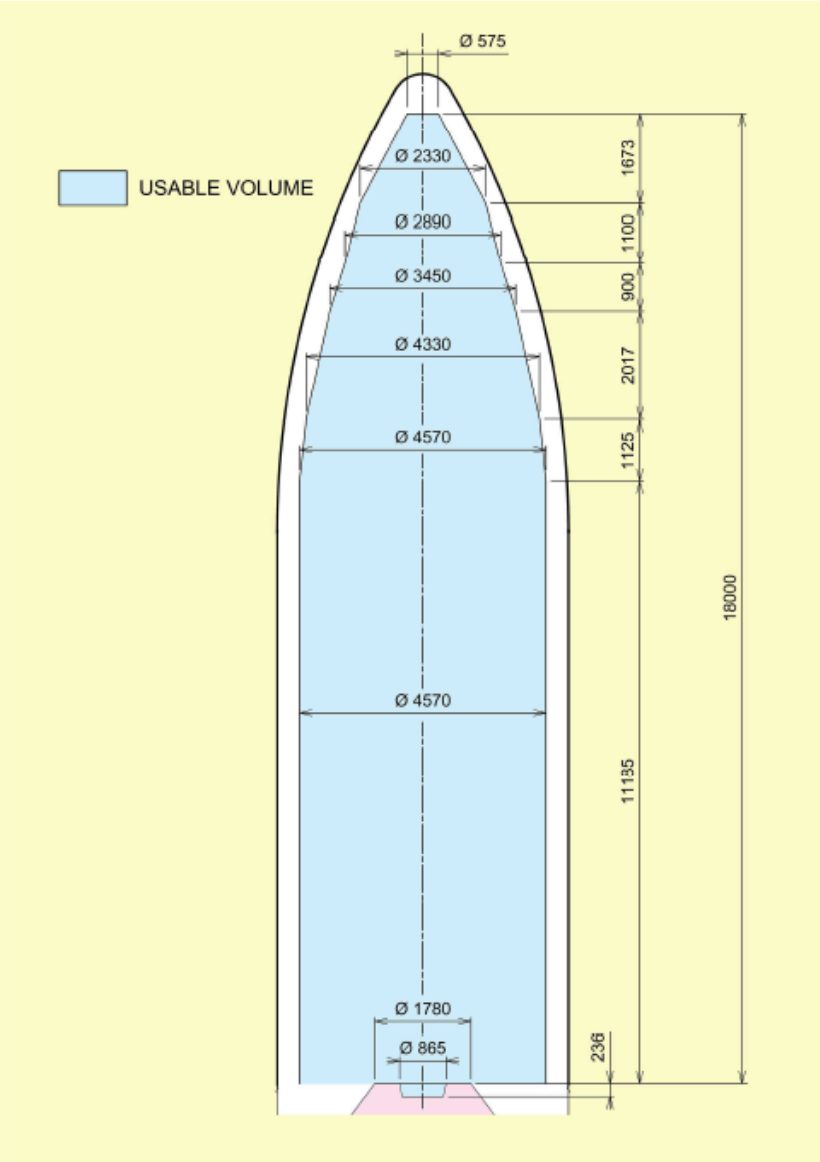


Figure 4: Possible A6 fairings, with long version on the left and short version on the right. A single baseline will be selected within 2016.

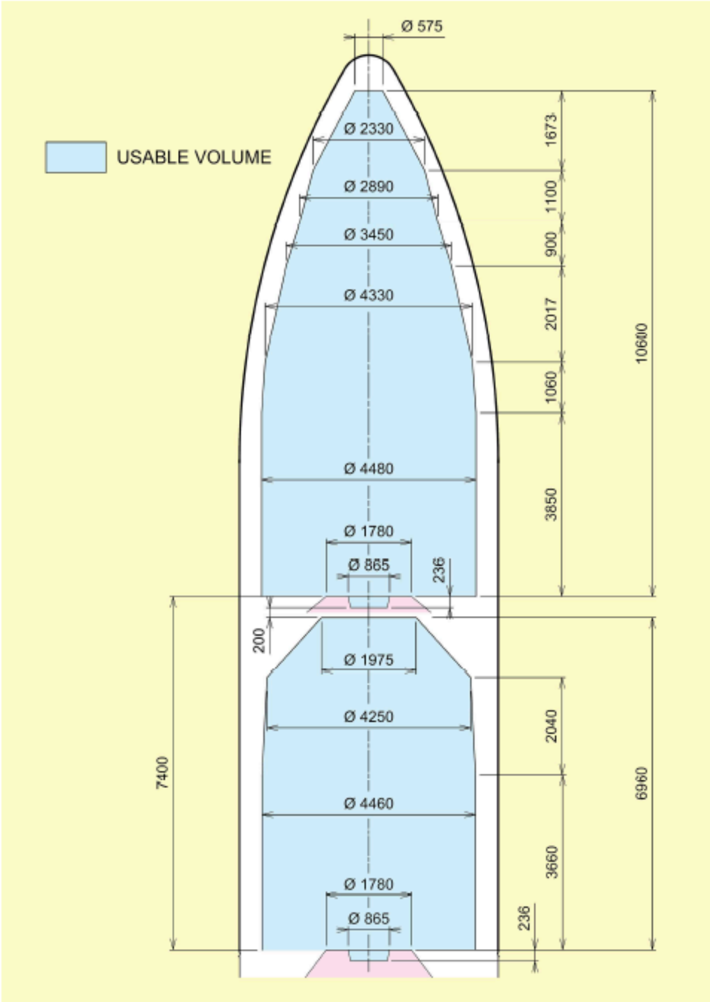


Figure 5: A6 dual launch configuration with SYLDA under long fairing option. The SYLDA adapter has a mass of approx. 500 kg.

Concerning the A6 – S/C launch adapter, a single LV adapter with an upper diameter of 1780 mm is currently baselined. The following adapters are available off-the-shelf from Arianespace:

Table 8: Standard Launch Adapters for Ariane 6

Adapter	Description	Separation system
PAF 937S	Height: 447 mm Max mass: 75 kg Aluminum structure	Clamp band Ø937 with low shock separation system (CBOD)
PAF 937C	Height: 453 mm Max mass: 80 kg Aluminum structure	Clamp band Ø937 with low shock separation system (LPSS*)
PAF 1194VS-VL	Height: 323 mm Max mass: 50 kg Cone and lower ring: monolithic carbon Upper ring: aluminum	Clamp band Ø1,194 with low shock separation system (CBOD)
PAF 1194VS	Height: 317 mm Max mass: 75 kg Aluminum structure	Clamp band Ø1,194 with low shock separation system (CBOD)
PAF 1194C	Height: 321.5 mm Max mass: 90 kg Aluminum structure	Clamp band Ø1,194 with low shock separation system (LPSS*)
PAF 1663	Height: 450 mm Max mass: 95 kg Aluminum structure	4 bolts with pyrotechnic separation nuts (Hi-Shear)
PAF 1666MVS	Height: 450 mm Max mass: 95 kg Aluminum structure	Clamp band Ø1,666 with low shock separation system (CBOD)
PAF 1666S	Height: 450 mm Max mass: 100 kg Aluminum structure	Clamp band Ø1,666 with low shock separation system (CBOD)
Raising cylinder: ACY 1780	Height: adjustable between 70 and 300 mm Max mass: 45 kg Aluminum structure	N/A

APPENDIX D – TECHNOLOGY DEVELOPMENTS IN THE FRAME OF L3

Current technology development activities are listed in the Science Technology Plan, which can be accessed through:
http://sci.esa.int/cosmicvision-tdp

An overview of previous and current developments has been compiled in the frame of the GOAT activities and is contained in the final report, which can be accessed here:
http://sci.esa.int/jump.cfm?oid=57910

APPENDIX E – OVERVIEW OF MISSION ANALYSIS

Launch and Transfer:
We present a first rough calculation of several launch and transfer options, not yet including a lunar flyby scenario.

Launch into escape trajectory and direct transfer
The following assumptions were taken in order to present mass estimates:
- Available launch mass (Ariane 6.4): 7000 kg
- Launch adapter and spacecraft dispenser: 500 kg, reducing the launch mass to 6500 kg
- Thruster performance (specific impulse) of 270 s

For the electric propulsion option, no detailed trajectory calculations have been performed yet. The deliverable mass is estimated in the 1900 kg range for an Ariane 6.4 launch, for an Ariane 6.2 launch, we estimate a deliverable mass around 700-800 kg.

Additional explanations for the table:
- Target initial displacement: angle of constellation on its orbit at 1 AU with respect to the Earth: T Trailing, H Heading.
- Operational Mass: mass that can be delivered to the full constellation
- Mass per spacecraft: available mass per spacecraft (including any propulsion module/system) at arrival at the constellation.
- Time to arrival: Transfer time from launch to arrival at constellation.

Target initial Displacement [deg]	Start Date	S/C #	Escape Velocity [km/s]	Launch mass [kg]	Size of Incl. Man [m/s]	Time of Stop Man. [d from Start]	Time to arrival [d]	Total Imp. Delta-v [m/s]	Delta-v with margin [m/s]	Worst Case Budget [m/s]	Isp [s]	Operational Mass [kg]	Mass per spacecraft [kg]
18T	11 March	1	0.264	6500	264	340	545	809	890	890	270	4645	1548
		2			105	342	538	643	707				
		3			251	330	481	732	805				
18T	21 March	1	0.264	6500	222	335	540	762	838				
		2			98	338	553	651	716				
		3			285	328	475	760	836				
18T	31 March	1	0.256	6500	156	329	529	685	754				
		2			80	335	577	657	723				
		3			331	327	475	806	887				
21T	21 March	1	0.249	6500	193	341	601	794	873	939	270	4559	1520
		2			70	348	656	726	799				
		3			296	338	558	854	939				
15T	21 March	1	0.264	6500	280	326	471	751	826	826	270	4758	1586
		2			109	331	478	587	646				
		3			243	313	415	658	724				
15H	21 March	1	0.981	6500	194	**356**	471	665	732	732	270	4931	1644
		2			85	308	392	477	525				
		3			44	306	566	610	671				

Launch into dedicated GTO, followed by escape manoeuvre and direct transfer

The following assumption were taken in order to present mass estimates:

- Available launch mass (Ariane 6.4): 11000 kg (commercial advertised performance)
- Launch adapter and spacecraft dispenser: 500 kg, reducing the launch mass to 10500 kg
- Thruster performance (specific impulse) of 270 s

A quick analysis of Ariane 6.2 performance into GTO (5000 kg) leads to very small masses per spacecraft at the constellation of 753 kg, including propulsion system and thus is regarded as non compliant.

Additional explanations for the table:

- Target initial displacement: angle of constellation on its orbit at 1 AU with respect to the Earth: T Trailing, H Heading.
- Operational Mass: mass that can be delivered to the full constellation
- Mass per spacecraft: available mass per spacecraft (including any propulsion module/system) at arrival at the constellation.
- Time to arrival: Transfer time from launch to arrival at constellation.

Target initial Displacement [deg]	Start Date	S/C #	Size of 1st Manoeuvre [m/s]	Escape Velocity [km/s]	Launch mass [kg]	Size of Incl. Man. [m/s]	Time to arrival [d]	Size of Stop Man. [m/s]	Total Imp. Delta-v [m/s]	Delta-v with margin [m/s]	Worst case Budget [m/s]	Isp [s]	Operational Mass [kg]	Mass per Spacecraft [kg]
18T	11 March	1	773	0.264	10500	264	340	545	1582	1825				
		2	773	0.264		105	342	538	1416	1643				
		3	773	0.264		251	330	481	1505	1741				
18T	21 March	1	773	0.264	10500	222	335	540	1535	1774				
		2	773	0.264		98	338	553	1424	1651	1825	270	5271	1757
		3	773	0.264		285	328	475	1533	1771				
18T	31 March	1	773	0.256	10500	156	329	529	1458	1689				
		2	773	0.256		80	335	577	1430	1658				
		3	773	0.256		331	327	475	1579	1822				
21T	21 March	1	773	0.251	10500	191	342	601	1565	1807	1875	270	5174	1725
		2	773	0.251		76	348	653	1502	1737				
		3	773	0.251		296	338	558	1627	1875				
15T	21 March	1	773	0.265	10500	321	324	457	1551	1791	1791	270	5340	1780
		2	773	0.265		109	328	498	1380	1603				
		3	783	0.553		192	285	558	1533	1772				
15H	21 March	1	773	0.249	10500	193	378	468	1434	1662	1691	270	5544	1848
		2	803	0.864		58	306	449	1310	1529				
		3	797	0.779		113	314	548	1458	1691				

Please note that while higher masses per satellite are achieved here compared to the direct scenario above, the much higher delta-V cost implies heavier propulsion system mass. This strategy is also significantly more complex from an operational point of view.

Constellation Dynamics

We present here some preliminary results for the assumed baseline constellation, calculated over up to 10 years:

Arm-length variation and rate of change

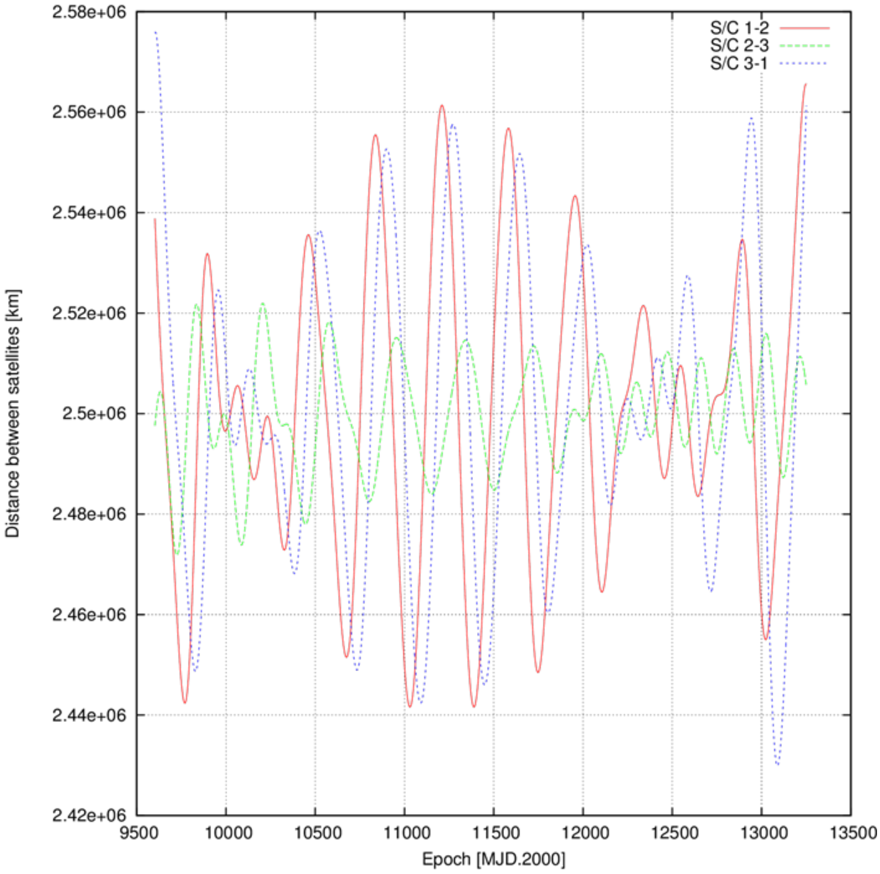


Figure 6: Arm Length variation: distance between spacecraft in km

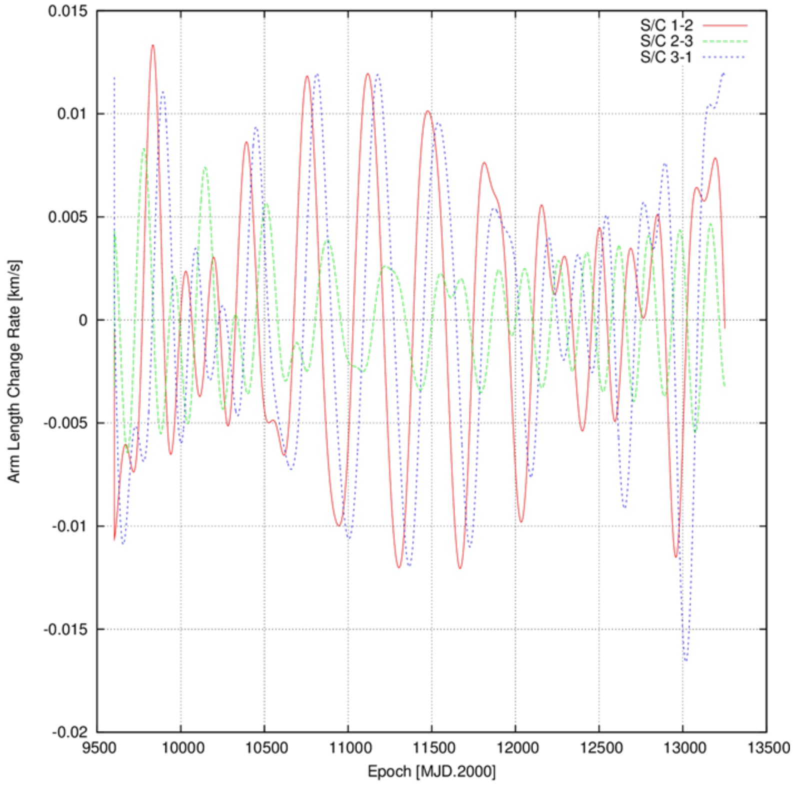
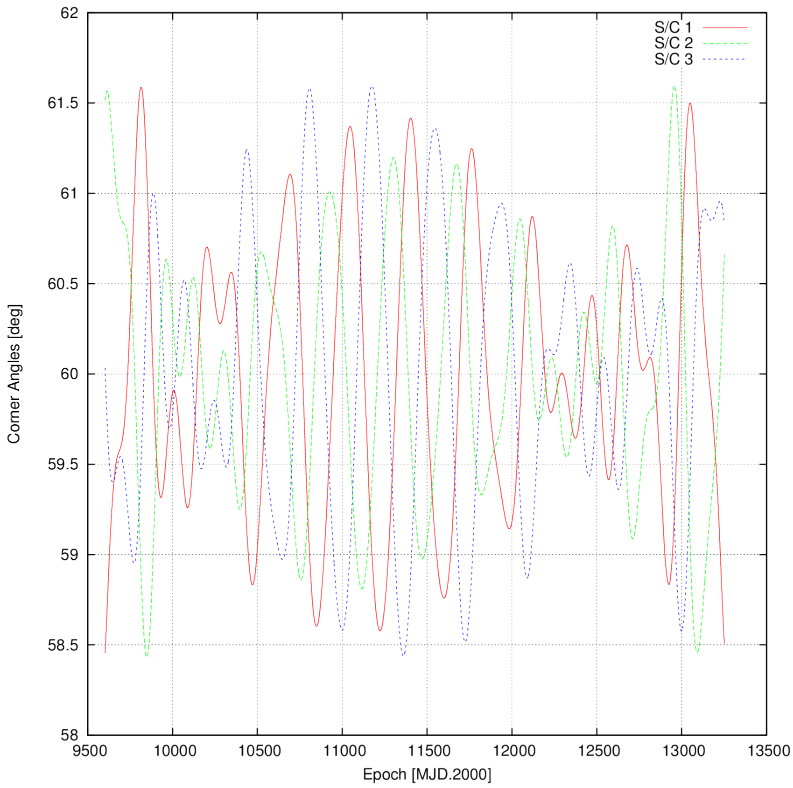


Figure 7: Arm Length variations: rate of change of distance between spacecraft in km/s

Corner Angle Variation



Earth Range

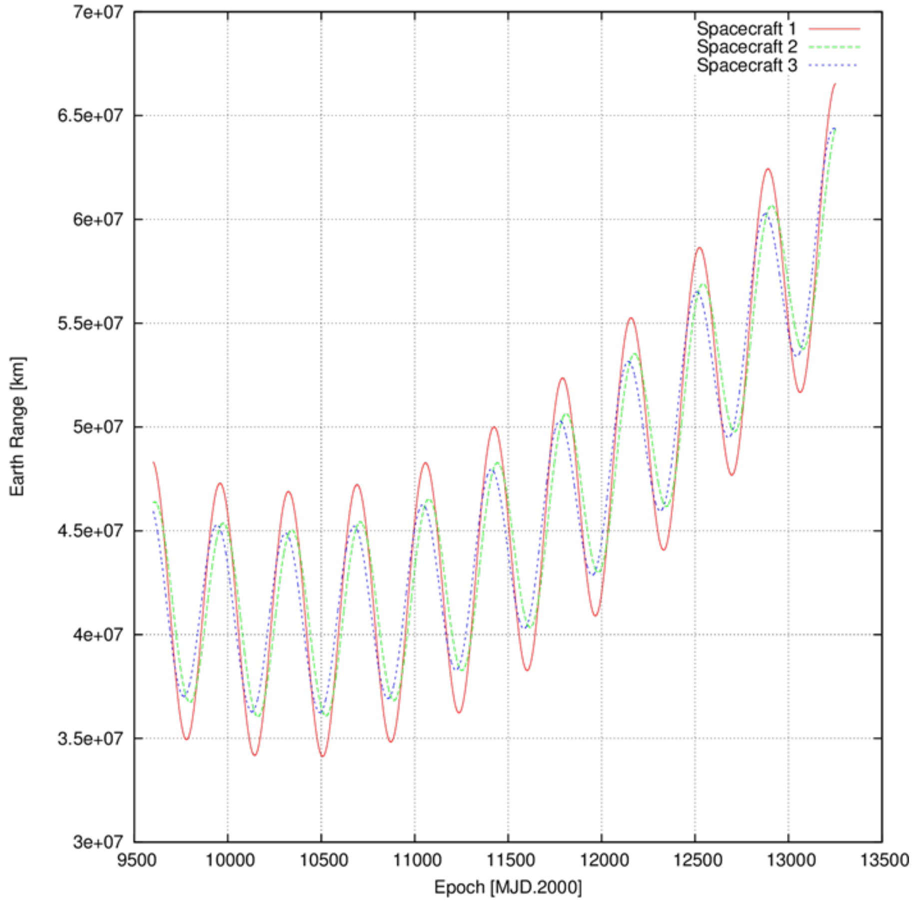


Figure 8: Distance of spacecraft to the Earth in km