ASSESSMENT OF A NEXT GENERATION GRAVITY MISSION FOR MONITORING THE VARIATIONS OF EARTH'S GRAVITY FIELD

TN6: MISSION ARCHITECTURE OUTLINES

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1. INTRODUCTION

1.1 Scope and Purpose

The NGGM study focuses on a future mission dedicated to monitoring the variation of the Earth's gravity field by low-low satellite-to-satellite tracking. The NGGM mission would build on, and extend, the results of GRACE, which forms the backdrop of current satellite missions dedicated to variable gravity (as opposed to GOCE which is the reference for static gravity).

This document is submitted in fulfilment of WP 2410 of the Next Generation Gravity Mission (NGGM) study. Together with [RD-21], it constitutes the output of Task 5 of the NGGM statement of work [see Table 1-1].

The purpose of Task 5 is to produce a set of candidate mission and system architecture concepts answering the set of requirements established in Task 1, consistent with the techniques elaborated in Tasks 2 and 3 and the performance expected from the simulations of Task 4.

According to the SOW, the activities comprised in Task 5 had to proceed through:

- Functional analyses aiming to identify candidate system architecture concepts for sampling the variable Earth's gravity, consistent with the different sets of mission requirements;
- High-level trade-off's as needed to build up the candidate system concepts;
- Shortlisting of a set of candidate concepts, each characterised by its key features: payload, platform(s) and satellite(s), launcher options, operations approach, ground segment and communication architecture, launcher, supported by functional, performance, orbit and operation analysis.
- Elaboration of performance models to support, for each of the selected mission concepts, parametric analyses, assessment of mission performance and error budgets;
- Justification of the selected options in terms of an optimal balance of scientific return with implementation complexity, risk and cost;
- Final identification of proposed mission architecture(s) and justification thereof.

Table 1-1 : NGGM Study Tasks and Work Packages

Task No.	SOW Task	WP No.	Work Package	Contributor
1	***Analysis of the requirements***	1100	Requirements Analysis	ULux
		1200	System Drivers	TAS-I
2	***Observing techniques***	2110	Observing Techniques	IAPG
		2120	Instrument Concepts	TAS-I
		2121	Measurement Technologies	Onera
3	***Mission analysis and attitude and orbit control concepts***	2210	Mission Analysis	Deimos
		2220	Attitude and Orbit Control Concepts	TAS-I
4	***Numerical Simulation***	2310	End-to-End Simulator Design and Implementation	TAS-I
		2320	Variable Gravity Model	IAPG
		2330	Backward Module	DEOS
5	***Mission Architectures***	2410	Architecture Definition and Trade-Off	TAS-I
		2420	Mission Architecture Definition Supervision	GIS
6	***Conclusions and Recommendations***	3100	Mission Architecture Analysis and Conclusions	TAS-I
		3200	Scientific Assessment of the Baseline Mission	ULux

2. DOCUMENTS

2.1 Applicable Documents

[AD-1] Assessment of a Next Generation Gravity Mission to monitor the variations of Earth's gravity field, Statement of Work, EOP-SF/2008-09-1334, Issue 2, 20 November 2008, Appendix 1 to AO/1-5914/09/NL/CT

[AD-2] Special Conditions of Tender, Appendix 3 to AO/1-5914/09/NL/CT

[AD-3] Draft Contract. Appendix 2 to AO/1-5914/09/NL/CT.

2.2 ESA Reference Documents

[RD-1] Rummel et al. (2003), Scientific objectives for Future Geopotential Missions, Technical Note, Version 6 from the ESA contract No: 16668/02/NL/MM "Enabling Observation Techniques for Future Solid Earth Missions"

[RD-2] Koop, R., Rummel, R. (2007), The Future of Satellite Gravimetry, Final Report of the Future Gravity Mission Workshop, 12-13 April 2007 ESA/ESTEC, Noordwiik, Netherlands

[RD-3] Laser Doppler Interferometry Mission for determination of the Earth's Gravity Field, ESTEC Contract 18456/04/NL/CP, Final Report, Issue 1, 19 December 2005

[RD-4] Laser Interferometry High Precision Tracking for LEO, ESA Contract No. 0512/06/NL/IA, Final Report, July 2008

[RD-5] System Support to Laser Interferometry Tracking Technology Development for Gravity Field Monitoring, ESA Contract No. 20846/07/NL/FF, Final report, September 2008

[RD-6] Bender P.L., Wiese D.N., and Nerem R.S., "A Possible Dual-GRACE Mission With 90 Degree And 63 Degree Inclination Orbits", Proceedings of the 3rd International Symposium on Formation Flying, Missions and Technologies, Noordwijk (NL), April 2008

[RD-7] T. van Dam et al., Monitoring and Modelling Individual Sources of Mass Distribution and Transport in the Earth System by Means of Satellites, Final Report, ESA Contract No. 20403, November 2008

[RD-8] Variable Earth Model Description and Product Specification Document, ESA Contract No. 20403, November 2008

[RD-9] Enabling Observation Techniques for Future Solid Earth Missions, ESA Contract No: 16668/02/ NL/MM, Final report, Issue 2, 15 July 2004.A

2.3 NGGM Study Notes

[RD-10] NGGM TN1 "Requirement Analysis", University of Luxembourg, Issue 1, Revision 1, 8 February 2010

[RD-11] NGGM TN2 "System Drivers", Thales Alenia Space, SD-TN-AI-1262, 4 December 2010

[RD-12] NGGM TN3 Part 1 "Observing Techniques and Instrument Concepts", Thales Alenia Space Italy, SD-TN-AI-1289, draft, July 2010

[RD-13] NGGM TN3 Part 2 "Observing Techniques", IAPG, in preparation

[RD-14] NGGM TN3 Part 3 “Instrument Concepts”, ONERA, 1/16598 DMPH, draft, July 2010

[RD-15] NGGM TN4 Part 1 “Mission Analysis and AOCS concepts”, Thales Alenia Space Italy, SD-TN-AI-1290, draft, July 2010

[RD-16] NGGM TN4 Part 2 “Mission Analysis of Candidate Scenarios”, Deimos, in preparation

[RD-17] NGGM TN5 Part 1 “Multi-Satellite Simulation Tool for SST Mission”, Thales Alenia Space Italy, SD-TN-AI-1291, in preparation

[RD-18] NGGM TN5 Part 2 “Scientific Simulation Tool”, DEOS, in preparation

[RD-19] NGGM TN5 Part 3 “Variable Gravity Model”, IAPG, in preparation

[RD-20] NGGM TN6 Part 1 “Mission Architecture Outlines”, Thales Alenia Space Italy, SD-TN-AI-1292 (this note)

[RD-21] NGGM TN6 Part 2 “Scientific Assessment of Mission Architectures”, GIS, in preparation

[RD-22] NGGM TN7 “Conclusions and Recommendations”, Thales Alenia Space Italy, to be initiated

2.4 External Literature

[RD-23] D. N. Wiese et al. (2009) Alternative mission architectures for a gravity recovery satellite mission, J. Geod. (2009) 83:569–581

[RD-24] P.N.A.M. Visser et al. (2009) Space-borne gravimetry and ocean tides: removing aliasing effects, to be published in Geophys. J. Int.

[RD-25] D. Feili, B. Lotz, H.W. Loeb, H. Leiter, M. Boss, R. Braeg, D.M. Di Cara, “Radio Frequency Mini Ion Engines for Fine Attitude Control and Formation Flying Applications”, Proceedings of the 2nd CEAS European Air & Space Conference, 20-26 October 2009, Manchester, UK

3. NGGM REQUIREMENTS

3.1 Science Drivers

The scientific requirements are elaborated in **TN1** [RD-10]. A synthesis is provided below to assist the discussion of mission architectures.

Four types of mass transport processes are identified as the primary focus for an NGGM:

- Ice
- Continental Water
- Ocean Mass
- Solid Earth.

The priorities of the NGGM are summarised in TN1 in terms of accuracy, spatial resolution and coverage, and temporal resolution and coverage. In addition, two further important considerations are made:

- Interpretations are hindered by aliasing caused by inadequate background models and by limited ability to separate the various mass transport signals. Thus, the choice of mission parameters that allow mitigating the nuisance signals caused by poor background models is a priority;
- parameter choices which will allow separating the mass transport signals due to different causes are a priority too.

Spatial resolution

Currently mass transport is observed at spatial scales down to 500 km (optimistically) and ~ 700 km (realistically). Improving the spatial scale down to 200-100 km would lead to major advances.

Temporal resolution

Current information (GRACE) is provided at monthly intervals (integrated image of the mass change over each month). Increased temporal sampling, down to e.g. 8 days, would assist continental water studies and de-aliasing.

Spatial coverage

Many of the signals associated with global climate change (polar ice mass balance, sea ice extent, deep/bottom water warming), are concentrated in the high latitudes, and so is much of the GIA signal. Thus, it is highly desirable for the NGGM to have a near-polar orbit. However, a stated priority of polar observations does not minimise the need for continued observations at the mid- to low-latitudes where continental water storage observations and Earth dynamic processes tend to be concentrated.

Table 3-1: Signal magnitude and associated spatial and temporal resolution for four high-priority geophysical signals [TN3, based on TN1]

Category	Phenomenon	Spatial resolution	Temporal resolution	Signal magnitude [geoid heights]
Ice	Melting of ice sheets (with separation of GIA)	100 – 1000 km	Seasonal – secular	0.01 mm/year (secular)
Continental Water	Ground water (soil moisture and snow) at higher spatial scales	10 – 200 km	Hourly – seasonal – secular	1 cm (seasonal)
Ocean	Non-steric component of sea-level variation at seasonal and shorter time scales	Global to basin level	Interannual – secular	0.1 mm/year (secular)
Solid Earth	Post-seismic deformation	10 – 200 km	Sub-seasonal	1 mm (sub-seasonal)

Temporal coverage

To reliably acquire polar ice mass trends would require at least 10 years of observations. In addition, many climate change trends have cycles on the order of a decade, reinforcing the requirement for long time series.

Accuracy

Table 3-1 shows the required signal magnitude with the associated temporal and spatial resolution for four high-priority geophysical signals, as identified in the science requirements analysis of TN1.

The relationship of gravity potential to geoid heights and EWLT (equivalent water layer thickness) is approximately [TN3]:

$$1\left[\frac{m^2}{s^2}\right] \quad gravity\ potential \cong 0.1[m]\ geoid\ height \cong 1[m]\ EWLT$$

3.2 Mission Requirements

From the prioritization of the science requirements a number of nominal mission profile requirements are derived in a straightforward way:

- Temporal resolution of 1 month for complete cycles; resolving ~ 1 week sub-cycles is of high interest;
- Mission lifetime of 11 years (solar cycle) for long term trends;
- Inclination close to 90 degrees to observe polar areas.

In **TN3**, the science requirements were translated into maximum cumulative geoid errors (CGE) for the nominal mission profile and for monthly gravity field solutions. In this analysis, the magnitude of the secular signals was translated into requirements applicable to monthly gravity field solutions by applying a factor of 10 (assuming a linear trend and 11 years of monthly data). The resulting span of the requirements is shown in Table 3-2.

In a second step, a more realistic envelope of the requirements was defined by limiting the spatial resolution to be reached with a specific CGE. Specifically, a compromise was made whereby only the green bars in the table were taken into account. Thus, the accuracy requirements to be applied for simulations were defined as follows:

Spherical Harmonic degree	150	200	250
Cumulative Geoid Errors CGE [mm]	0.1	1	10

These requirements were further propagated, by means of analytical error models of gravity field retrieval performance, into measurement requirements of a mission including ll-SST and accelerometers (ACC).

Table 3-3 identifies, for SST and for ACC, the maximum noise level allowable in order to meet the requirements above. The noise levels indicated in the table represent the flat noise regions of the assumed error PSD (see **TN 3**):

- for SST, range (in m/ √Hz) above 10 mHz
- for ACC, acceleration (in m/s²/√Hz) between 1 mHz and 100 mHz.

This completes the identification of the set of requirements applicable to the NGGM mission in terms of accuracy, spatial coverage and resolution, and temporal coverage and resolution. These requirements are ideally met by a tandem mission in a polar, circular orbit, at 300 km altitude, with 30-day repeat cycle and 1-week sub-cycle, with satellite separation on the order of 100 km, range accuracy better than $2{\cdot}10^{-13}$ m/√Hz, acceleration measurement accuracy better than 10^{-11} m/s²/√Hz, and lifetime of 10 years.

The de-aliasing requirements are covered by the selection of formation types. These requirements are discussed in §3.3 below.

Table 3-2: Cumulative geoid error requirements for monthly solutions [TN3]

	Wavelength	10000 km	1000 km	200 km	100 km	10 km
	SH degree	2	20	100	200	2000
CGE	10 mm			3		
	1 mm	2			4	
	0.1 mm		1			

Signal 1 = Melting of ice sheets
Signal 2 = Non-steric sea level variations
Signal 3 = Ground water variations
Signal 4 = Post-seismic deformation

Table 3-3: Performance of SST sensor and accelerometer required for meeting the target CGE. wn: white noise, cn: coloured noise. Every mission profile belonging to a grey box will not meet the requirements with the minimum noise levels [TN3].

SST

		Satellite Separation [km]							
		50		100		200		300	
		wn	cn	wn	cn	wn	cn	wn	cn
Altitude [km]	300	5 E-13	4 E-13	4 E-13	1 E-13	4 E-14	4 E-14	3 E-14	3 E-14
	350	8 E-14	7 E-14	6 E-14	5 E-14	<1 E-14	<1 E-14	<1 E-14	<1 E-14
	400	2 E-14	2 E-14	<1 E-14	<1 E-14	<1 E-14	<1 E-14	<1 E-14	<1 E-14
	450	<1 E-14	<1 E-14	<1 E-14	<1 E-14	<1 E-14	<1 E-14	<1 E-14	<1 E-14
	500	<1 E-14	<1 E-14	<1 E-14	<1 E-14	<1 E-14	<1 E-14	<1 E-14	<1 E-14
	550	<1 E-14	<1 E-14	<1 E-14	<1 E-14	<1 E-14	<1 E-14	<1 E-14	<1 E-14
	600	<1 E-14	<1 E-14	<1 E-14	<1 E-14	<1 E-14	<1 E-14	<1 E-14	<1 E-14

Accelerometer

Altitude [km]	300	2 E-11	8 E-12	4 E-11	9 E-12	7 E-11	3 E-11	9 E-11	5 E-11
	350	6 E-12	3 E-12	2 E-11	6 E-12	3 E-11	2 E-11	3 E-11	2 E-11
	400	2 E-12	8 E-13	3 E-12	2 E-12	5 E-12	3 E-12	6 E-12	5 E-12
	450	<5 E-13	<5 E-13	<5 E-13	<5 E-13	8 E-13	6 E-13	1 E-12	9 E-13
	500	<5 E-13	<5 E-13	<5 E-13	<5 E-13	<5 E-13	<5 E-13	<5 E-13	<5 E-13
	550	<5 E-13	<5 E-13	<5 E-13	<5 E-13	<5 E-13	<5 E-13	<5 E-13	<5 E-13
	600	<5 E-13	<5 E-13	<5 E-13	<5 E-13	<5 E-13	<5 E-13	<5 E-13	<5 E-13

3.3 Aliasing and Satellite Formation Requirements

Previous studies [RD-7] [RD-6] [RD-23] [RD-24] have already highlighted a number of facts about the striation/distortion/aliasing problems. They are summarised as follows.

i. The fundamental observable in a leader-follower formation in near-polar orbit is the intersatellite distance (and/or relative velocity). This observable approximates the along-track gravity gradient term Vxx and it is mainly sensitive along the line-of-sight, i.e., in North-South direction (for polar orbits). The directional sensitivity of the observable translates into non-isotropic error behaviour. The distortions are manifested mainly as a North-South striping pattern.

There are various evidences that environmental noise (e.g., from non-resolved atmospheric and ocean tides) is the dominant contributor to the error budget of GRACE data and GRACE-based mass transport models. The estimated atmospheric/ocean tide signal is orders of magnitude larger than instrumental noise at low frequency (Ref. Thompson et al., GRL, 2004

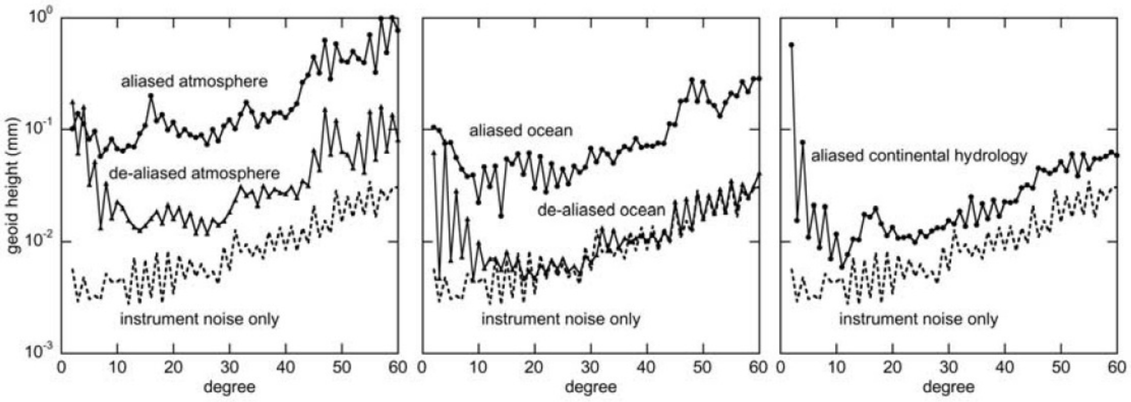
ii. Figure 3-1, Figure 3-2). Based on these results, it can already be concluded that tide model errors need to be improved significantly before full advantage can be taken of improved ll-SST sensors. Also, ocean tide model errors lead to gravity field retrieval errors larger than the gravity variations due to continental hydrology around spherical harmonic degree 25.

iii. Gradiometry of out-of-plane components (Vxy, Vyy, Vyz) can be achieved through non-coplanar satellite configurations. If the relative orbits comprise a cross-track motion, the corresponding observables gain sensitivity in East-West direction. The extra components have been shown capable of eliminating the longitudinal striping seen in the collinear solutions, even in the presence of aliasing.

iv. Extra gravity gradient components may be helpful, too, in de-aliasing signals, but they do not address the fundamental temporal-spatial sampling problem, i.e.: missions with short repeat periods reduce temporal aliasing at the expense of spatial resolution, and missions with good spatial coverage reduce spatial aliasing at the cost of temporal resolution.

v. To overcome aliasing, multiple-formation configurations must be considered. In principle, the spatial resolution can be improved without losing temporal resolution by means of additional satellite tandems flying interleaved ground tracks. The temporal resolution can be enhanced without deteriorating the spatial resolution by adding satellite tandems flying the same ground track with a time shift.

vi. The GRACE satellites fly in non-repeating orbits. These orbits lead to a continuously changing ground track pattern which makes it difficult to identify aliasing periods. Furthermore, the polar orbits have a ground track density that increases towards the poles, introducing a latitudinal change in the resolution of the solutions. These effects can be avoided to a large extent by selecting exact-repeat orbits and by combining satellites that fly in orbits with different inclinations as proposed by e.g. Bender et al. [RD-6]. These designs simultaneously improve both spatial and temporal resolution and also the different aliasing frequencies might be helpful for de-aliasing.



Ref. Thompson et al., GRL, 2004

Figure 3-1: Contribution of rapid mass transport processes to the GRACE error budget

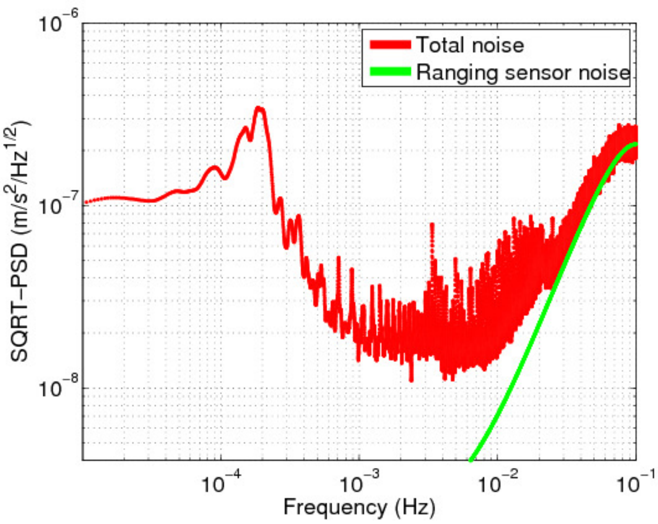


Figure 3-2: Errors in the GRACE inter-satellite accelerations (July 2006)

Based on the above, some guidelines can be drawn, as follows.

1. The NGGM should contribute to solving the aliasing problem as well as the striping problem.
2. One-tandem formations can solve the striping problem if they have an out-of-plane component, such as cartwheels or pendulums.
3. To contribute to the aliasing problem, more than one tandem must be considered. To ease the identification of the aliasing periods, the tandems should fly in exact-repeat orbits.
4. Once multiple tandems are considered, the Bender-type formation (two collinear tandems in orbits with different inclinations, one of them near-polar) has already been shown to have the required characteristics of improving the spatio-temporal sampling, and is considerably easier to implement than the pendulums or cartwheels.

3.4 Implementation Requirements

3.4.1 Payload requirements

The payload requirements are elaborated in **TN3** and summarised below.

Low-low Satellite-to-Satellite Tracking (ll-SST) is the only observing technique potentially capable of detecting the time-variable gravity with the required resolution from the lowest to the highest degrees. Gradiometry could contribute to highest degrees / orders only and no significant contribution is added by the precise orbit determination which can be obtained from a GNSS receiver [RD-13].

In the GRACE-like implementation of ll-SST, the fundamental observable is the distance variation between the centres of mass (COMs) of two satellites, produced by the gravity acceleration, Δd_G, formally obtained from Δd - Δd_D, where Δd is the total distance variation between the COMs, whatever the source, as measured by a distance metrology, and Δd_D is the distance variation produced by non-gravitational (i.e. drag) forces, as measured by accelerometers. This "accelerometer" implementation is preferred to an "inertial sensor" implementation for the reasons outlined in TN3. The required combined performance is shown in Figure 3-3.

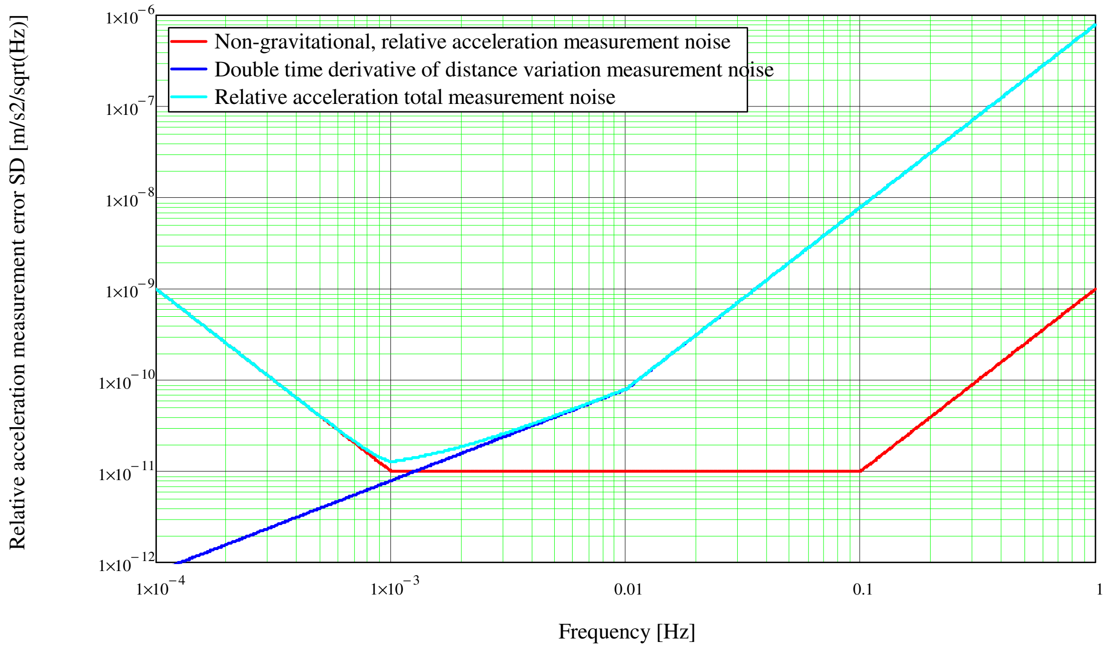
The recommended configuration of the SST laser-interferometer was identified via the following trade-offs:

- Asymmetric vs. symmetric interferometers on Satellite 1 and Satellite 2: symmetric configurations selected;
- Laser Beam Steering Mechanism (BSM) vs. steering by satellite attitude control: steering by satellite attitude control selected;
- Ancillary metrology (angle and lateral displacement metrology) implementation trades: the angle metrology system developed by TAS-I in cooperation with INRIM is considered optimal (see TN3).

For the accelerometers, the following conclusions were reached:

- GOCE-type accelerometers are all that is required for the NGGM;
- A 4-accelerometer set as shown in Figure 3-4 is recommended for reasons of redundancy of measurement of linear and angular accelerations.

A possibility mentioned at the outset of the study is that of having one or two pairs of accelerometers providing a measurement of the cross-track and/or radial components of the gravity gradient i.e., the components that in the more complex formations are provided by the out-of-plane motion of the formation. The conclusion of this study is that the gradiometer measurements are not competitive with the SST out-of-plane measurements. The selected configuration of in Figure 3-4 does provide such gradiometer mode anyway.



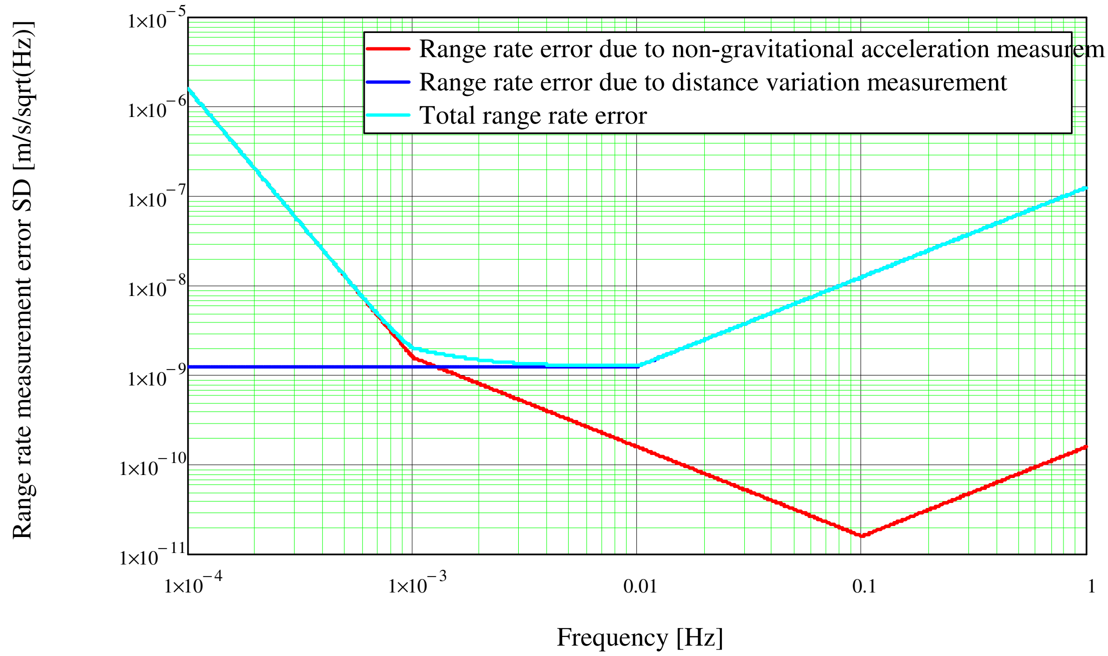


Figure 3-3 : Combination (light blue) of SST (blue) and accelerometry (red) measurement requirements expressed in terms of relative acceleration (top) and range rate (bottom).

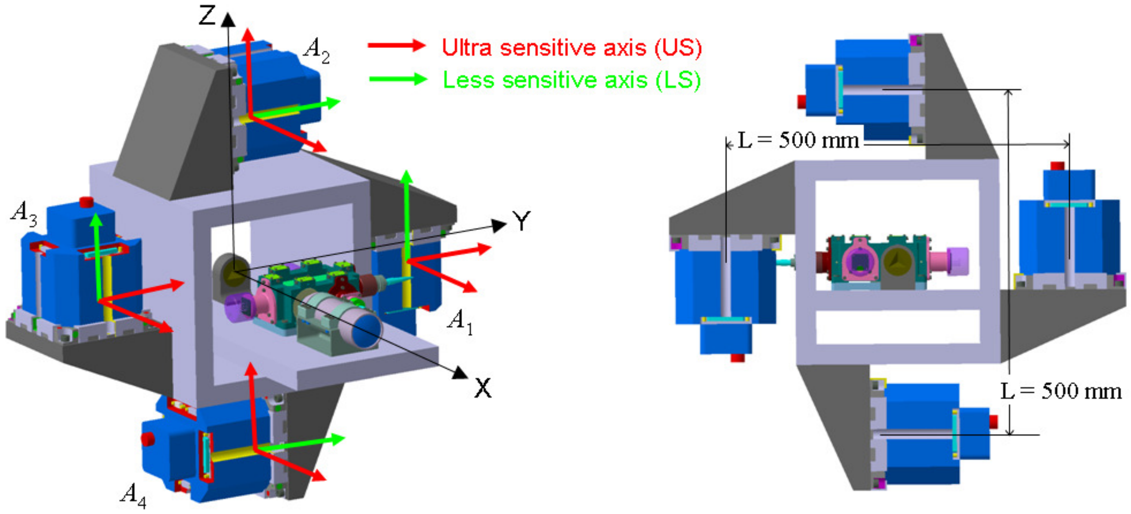


Figure 3-4: Recommended four-accelerometer configuration.

3.4.2 Spacecraft requirements

The main spacecraft implications of the NGGM defined above are on the orbit and attitude control, which are treated in **TN4**. The orbit and attitude control, in turn, will drive the spacecraft resources. The relevant requirements are summarised below.

i. Low altitude (300 km circular) is required by the accuracy criterion. Because of the requirement for exact-repeat orbits, altitude must be maintained within tight limits. Moreover, lifetime exceeding 10 years is demanded. The combination of these three requirements leads to electric propulsion of the GOCE kind as the only practical proposition for orbit maintenance.

ii. Orbit maintenance by electric propulsion has the added bonus that the altitude and repeat cycle can be changed with little effort. This offers the possibility of (a) reducing the orbit maintenance requirements by flying at higher altitude when solar cycle is near maximum, (b) changing the time and space resolution in a controlled way in order to solve for specific aliasing periods.

iii. To achieve full accelerometer performance, the satellite must be kept drag-free in the satellite-to-satellite direction and the angular accelerations must be controlled. This, too, points at electric propulsion as the only practical solution. The requirement on linear acceleration control (drag-free level) is 10^{-8} m/s²/√Hz and the requirement on angular acceleration control is 10^{-8} rad/s²/√Hz; both requirements apply to both satellites in a tandem.

iv. To project the measurements along the sat-to-sat baseline, good pointing knowledge and control are necessary. The laser beam pointing from Sat 1 (beaming) to Sat 2 (reflecting) shall be better than $2{\cdot}10^{-5}$ rad and $2{\cdot}10^{-6}$ rad/√Hz. The reciprocal requirements (attitude of Sat 2 relative to Sat 1) are one order of magnitude less demanding thanks to the “laser retroreflector” concept.

v. The attitude control requires low noise actuators such as microthrusters or magnetic bearing reaction wheels. Among microthrusters, only electric thrusters can provide the required long life at small propellant cost. Since the satellite has to be drag-free (on 1 axis or 3 axes, depending on the type of formation) it is natural to combine the AC and DF requirements in a single system.

vi. Thus one is led to an all-electric attitude, drag and orbit control system. The actuators are GOCE-like ion thusters for main drag control (1- or 3-axis, depending on the type of formation) and smaller ion thrusters for attitude/drag-free control.

vii. Electric propulsion trades propellant mass for electrical power. The propellant requirements for 10-year life are on the order of a few percent of the launch mass if the drag cross section is kept sufficiently low. A fuse-like satellite like GOCE is the optimal solution for the in-line and pendulum formation but not for the cartwheel. In addition, the configuration must provide sufficient solar panel area for feeding the electric propulsion, in all seasons of the year, without the simplifying assumption of a near constant cross section to the sun (as in GOCE in SSO).

viii. Loose formation control (FC) is required, of the order of $\pm 0.05 \cdot d$ in the longitudinal direction and $\pm 5 \cdot 10^{-3} \cdot d$ in the lateral direction. Formation and drag control take place under the constraints that (a) FC shall not interfere with the scientific measurement, (b) FC shall not violate the DF requirements, (c) the drag-free control shall not cancel the formation control action.

ix. Both the SST and the accelerometers require tight temperature control. With respect to the GOCE design, the problem is more difficult since (a) the enclosure containing the laser source is necessarily coupled to the external environment, because of the laser beam aperture, (b) the orbit is not SSO, leading to more coupling with the external environment (radiators will change their aspect to the earth and deep space with seasonal and orbit frequency).

x. Lastly, the spacecraft is subject to stringent mass and volume constraints by the requirement that a low-cost (hence small) launcher is used and one tandem pair is hosted in each launch.

3.4.3 Programmatic requirements

The following programmatic constraints will be taken into account in this exercise:

a) A low cost launcher shall be used (Vega / Rokot class);

b) The total cost to ESA shall be that of an Explorer-class mission;

c) In a multiple-pair constellation, each pair shall be a worthy mission on its own (goal).

4. BASIC CANDIDATE ARCHITECTURE CONCEPTS

In this chapter we begin by reviewing the performance and the implementation aspects of three basic architectures:

- The collinear tandem (chapter 4.1);
- A tandem in "pendulum" configuration (chapter 4.2);
- A tandem in "cartwheel" configuration (chapter 4.3).

In Chapter 4.4 we consider variations on the basic architectures (single or multiple tandems, different orbit parameters, etc.).

4.1 Concept 1: Two collinear tandems in orbits with different inclination

Two tandems must be considered because of the striping problem. One tandem is in polar orbit because of the spatial coverage. The other tandem is in a lower inclination orbit. This architecture type includes the “Bender pair” (312 km mean altitude, 90°/63.7° inclination, 79/360 repeat) proposed in [RD-6].

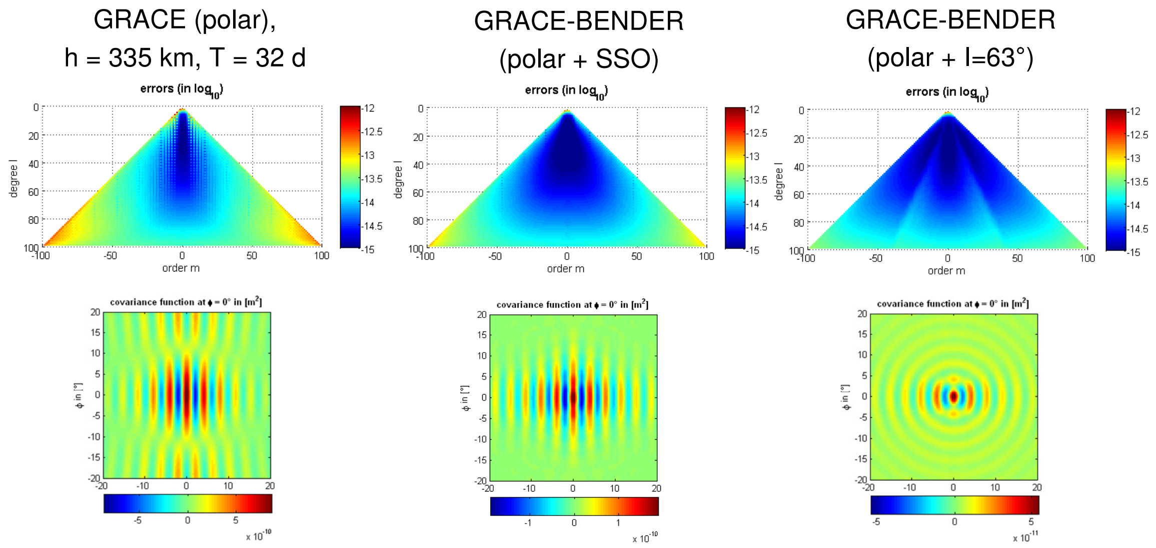
4.1.1 Scientific performance

The performance of the Concept-1 formations is assessed in TN6/Part2 [RD-21], by an analytical quick-look tool, and in TN5/Part2 [RD-18] with a fully-fledged numerical simulation. A thorough discussion is provided therein; here only a short summary is given for completeness.

Figure 4-1 shows a comparison of the performance of a GRACE-type mission (single tandem in polar orbit, with the instrument performance as specified in §3.4.1) with

- Two tandems in polar and 63° inclination orbits
- Two tandems in polar and SSO orbits.

The conclusion is that the highest sensitivity and isotropy is reached for the combination of a polar tandem and a low-inclined (I = 63°) satellite tandem.



ρ = 75 km, new reference noise

Figure 4-1: Performance of Concept-1 formations [RD-21]

4.1.2 Spacecraft implementation

We present below a very preliminary design concept of a spacecraft suitable for Concept 1. The purpose is the trade-off, i.e. a comparative assessment with the other concepts. A real Phase A – level study is outside the scope of this assessment exercise.

The spacecraft design is driven by the following requirements and constraints:

- 2 identical satellites must be accommodated under the Vega fairing;
- each spacecraft must accommodate a pair of GOCE-type electric thrusters (10 mN class) for orbit control in the sat-to-sat direction, coinciding with the direction of the main drag component, and a minimum set of 8 smaller electric thrusters (0.5 mN class), for 6dof attitude / drag free control / angular acceleration control;
- each spacecraft must accommodate a solar array delivering about 1 kW power in all seasons (justification is given further below)
- the configuration must be able to accommodate a payload bay (laser + accelerometer set) in a location close to and symmetric with respect to the c.o.m
- a system must be provided for active centre-of-mass control
- the design must provide a load-carrying external structure and sufficient volume for accommodating all equipment.

The thermal control is not addressed in this preliminary assessment, which focuses on the resources to be provided by the spacecraft to the payload and the associated ACS/DFC system. However the temperature control and stability requirements associated to both the laser metrology system and the accelerometer set will be stringent and a possible mission driver. They are a priority for any follow-up study.

Figure 4-2 and Figure 4-3 show the configuration concept.

Each satellite is a prism with pentagonal cross section. The area projected in the direction of motion is about 1.3 m^2, similar to GOCE. Two identical spacecraft back-to-back fill up the Vega fairing. The internal layout uses four platforms to host the equipment, such as to place the c.o.m. approximately in the geometrical centre, where the payload bay is located. A slightly flared tube crosses the spacecraft from end to end to provide an unobstructed path to the laser beam. All envisaged equipment fit the volume allocated with room to spare so that accommodation does not seem critical.

The top and sides of the prism are covered with solar panels; the sides have 2 panels each, deployed in flight to increase the active area. The panel wings have two positions: (A) perpendicular to Z and (B) perpendicular to Y. The configuration is toggled from A to B four times a year, the transition occurring when the sun is approximately 30° from the orbit plane. The c.o.m. adjustment mechanism must be actuated each time to rebalance the spacecraft.

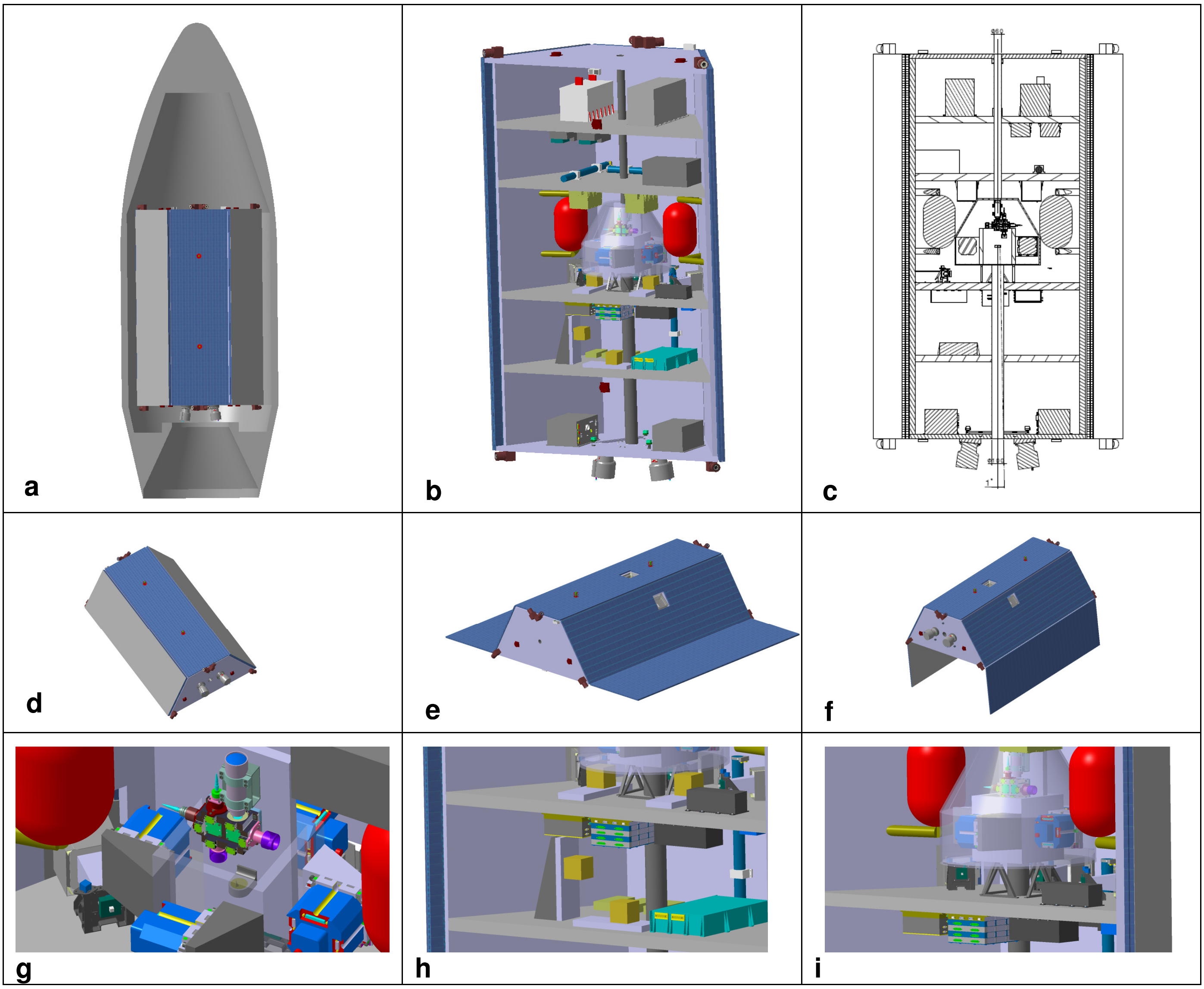


Figure 4-2: Concept 1 spacecraft design.

From top to bottom and from left to right: (a) two identical spacecraft back-to-back under the Vega fairing; (b) 3D view of one spacecraft with solar wings and earth-facing panel removed; (c) cross section showing equipment accommodation; (d) separated spacecraft before solar wing deployment; (e) solar panel position A (sun near orbit plane); (f) solar panel position B (sun near orbit normal); (g) detail of the Laser tracker / Accelerometer accommodation; (h) detail of the c.o.m. control system; (i) sealed payload bay.

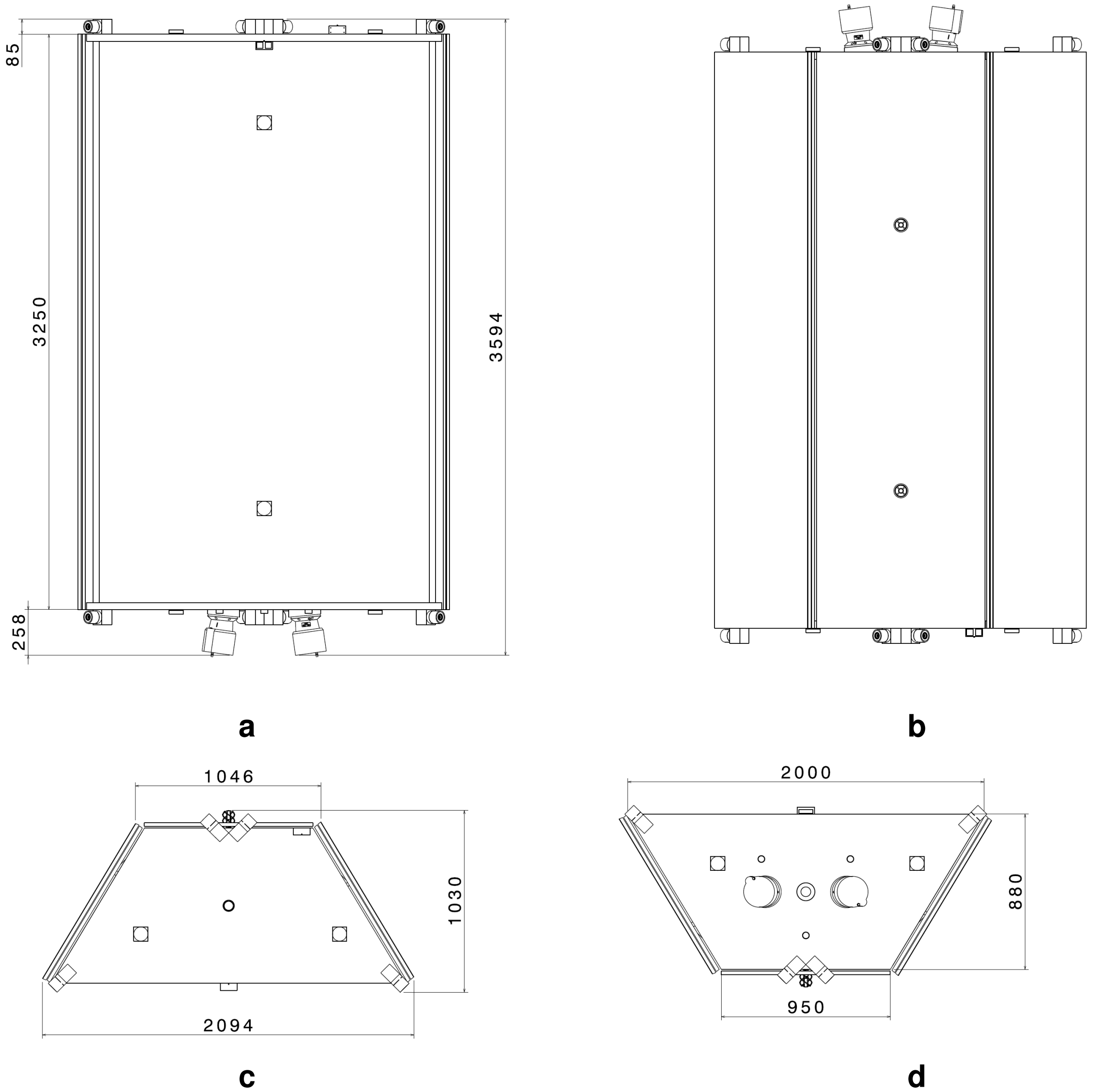


Figure 4-3: Concept 1 spacecraft dimensions

The mass target is about 800 kg per spacecraft (1700kg launched by VEGA in 300-km polar orbit, including the adapter/dispenser, see Figure 4-5). Table 4-1 shows a preliminary mass estimation assuming:

- Equipment mass from GOCE
- Payload mass from TN3 [RD-12];
- Structure, harness and thermal control, scaled from GOCE in proportion to the target mass;
- Solar array proportional to surface area;
- Propellant mass for 11 years assuming specific impulse as for GOCE (X-axis thrusters) and miniRIT [RD-25].

According to this exercise, the design would have no system margin with respect to the target mass. This result is affected by many rough approximations; a proper Phase A design exercise (e.g., structure sizing etc.) should be undertaken to consolidate a mass budget. Note in particular that the propellant mass estimate of the mini-RIT is affected by operation at thrust levels far from optimum (see Figure 4-8 and Figure 4-4); in a proper design, the thrusters would be optimised for the expected thrust range.

However, there is a clear indication that the mass budget is critical for a mission as demanding as the NGGM design considered here, and the next-higher launcher class (Soyuz) may be needed.

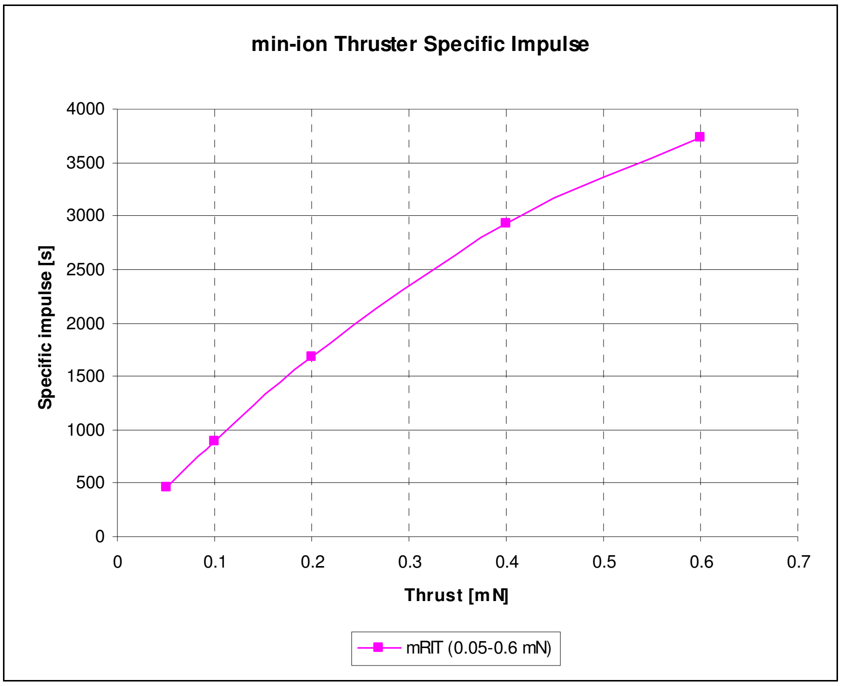


Figure 4-4: Mini-ion thruster specific impulse

Table 4-1: Concept 1 mass budget

	GOCE	NGGM
Years in orbit	*2*	*11*
Structure including balance mass	348	263
EPS	67	67
Solar array	71	52
Harness	77	58
Thermal Control	29	29
Avionics	62	62
TT&C	10	10
Cold gas propulsion	75	
Ion Propulsion Assembly (orbit)	74	74
Mini-Ion Propulsion		24
Total Platform Dry	**812**	**638**
Nitrogen mass	14	
IPA Xenon mass	40	20
mini-IPA Xenon mass		33
Total Propellant	**54**	**52**
Total Platform Wet	**866**	**690**
Gradiometer	181	
Laser SST Assembly		59
Accelerometer Assembly		47
GPS	13	13
Laser Retroreflector	1	1
Total Payload	**195**	**120**
Total Satellite	**1061**	**810**
System Margin		81
Total with margin	**1061**	**891**

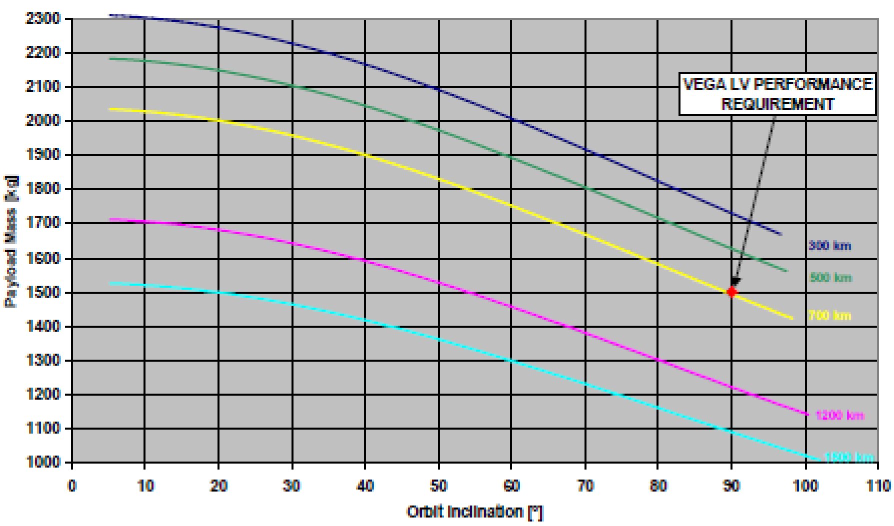


Figure 4-5: VEGA performance for circular orbits

Table 4-2: Concept 1 power budget

NGGM power budget	Power [W]
CDMU [W]	53
PCDU [W]	56
RF Receiver [W]	18
RF Transmitter [W]	13
Magnetometer [W]	2
Star Tracker [W]	21
Magnetotorquers [W]	1
Digital Sun Sensor [W]	2
S/C Heaters [W]	85
Total Platform Power excluding ion propulsion [W]	**251**
Ion Propulsion Assembly (main drag & orbit control)	200
Mini-Ion Propulsion (attitude & transverse drag control)	220
ITA Power [W]	**420**
Laser SST [W]	84
Accelerometers [W]	47
Accelerometer Heaters [W]	35
GPS receiver	30
Total Payload Power [W]	**196**
Total Payload + Platform Power [W]	**866**
Power Bus Harness Losses [W]	26
PCDU Losses [W]	17
System Margin [W]	91
Total Power Demand [W]	**1001**

Table 4-2 shows a preliminary power demand budget. The individual items are taken from similar equipment in GOCE and other spacecraft. The budget is limited by design to 1 kW orbit average by limiting the power allocated to the electric thrusters. For this formation, with the main drag component always on the same side (Figure 4-8), the envisaged configuration has 2 GOCE-type thrusters on the X side, and 8 distributed miniRIT thrusters for attitude and lateral drag control. Using the GOCE power demand curve, we find that the allocated 200 W correspond to 3 mN drag compensation capability (Figure 4-7). The consumption of an 8-microRIT thruster set is estimated from the producer data and a simulation of thruster usage with peak thrust limited to 3 mN (Figure 4-8).

Figure 4-9 shows that the envisaged solar array (with seasonally variable wing positions) can indeed provide about 1 kW orbit-average power whatever the longitude of the sun with respect to the orbit plane.

The 3mN limit on peak drag translates into a constraint on the orbit altitude, given a satellite cross section and a density model (itself depending on the solar activity indices). Figure 4-10 shows the maximum drag per orbit under peak and average solar activity conditions. The figure shows the 3mN limit would allow operation between 325 km and 340 km depending on the phase in the solar cycle.

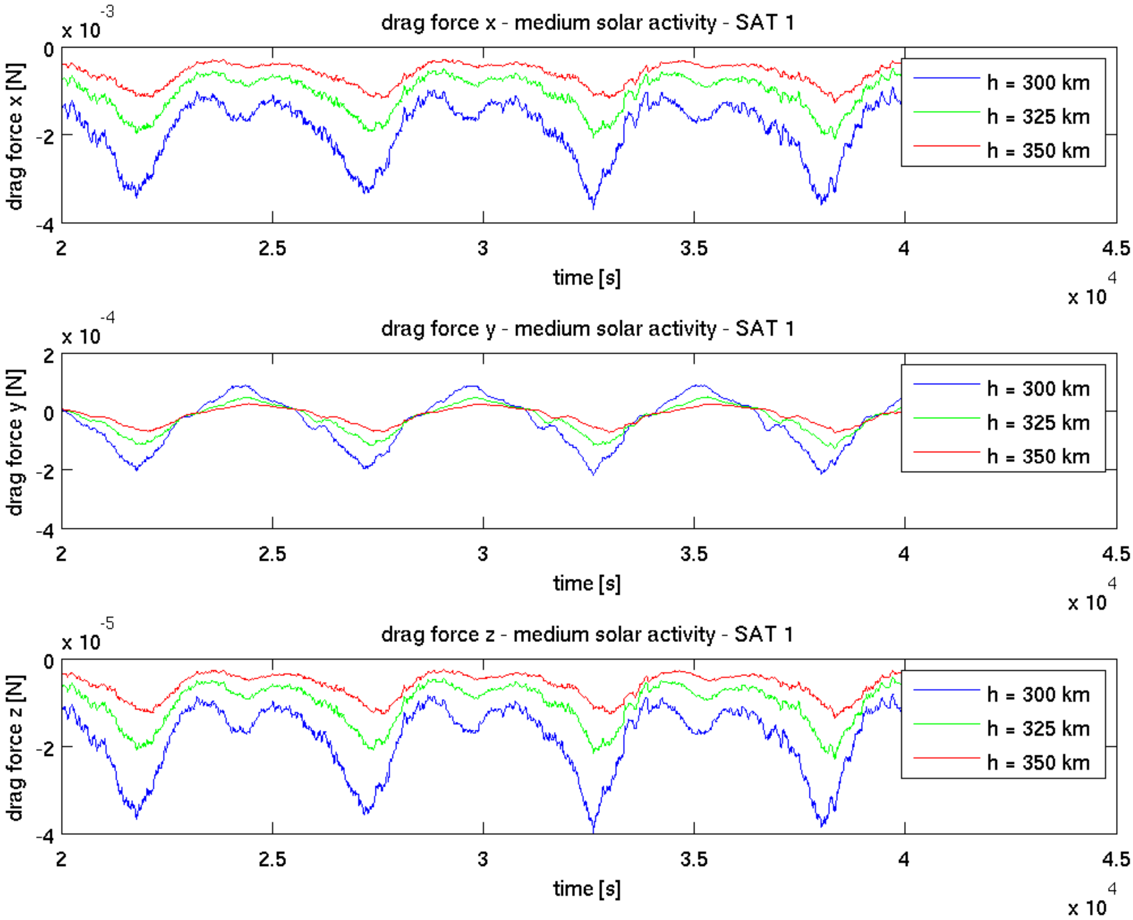


Figure 4-6: Simulation of the drag forces on the envisaged spacecraft in the collinear tandem formation.

The peak Y-axis force is about 6% of the peak X-axis force, and the peak Z-axis force is about 1% of the peak X-axis force. These ratios are conserved whatever the phase of the solar cycle (average solar activity shown).

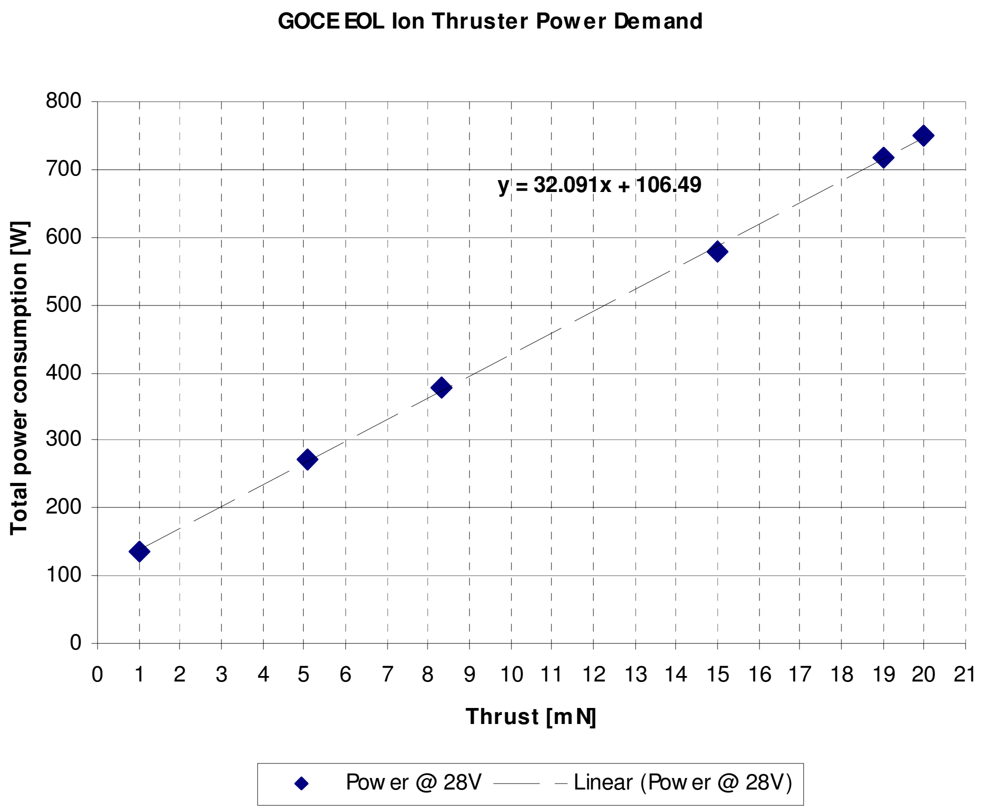


Figure 4-7: GOCE ion thruster power demand

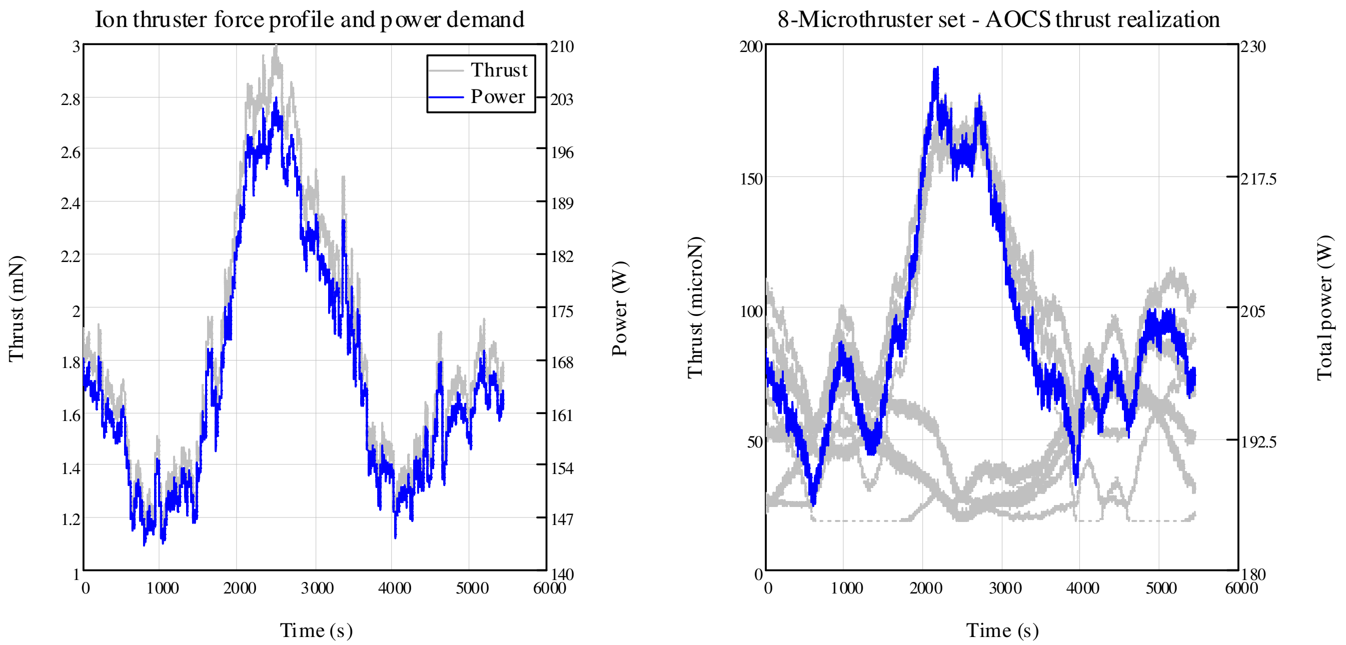


Figure 4-8: Simulation of drag control forces with peak drag component limited to 3 mN

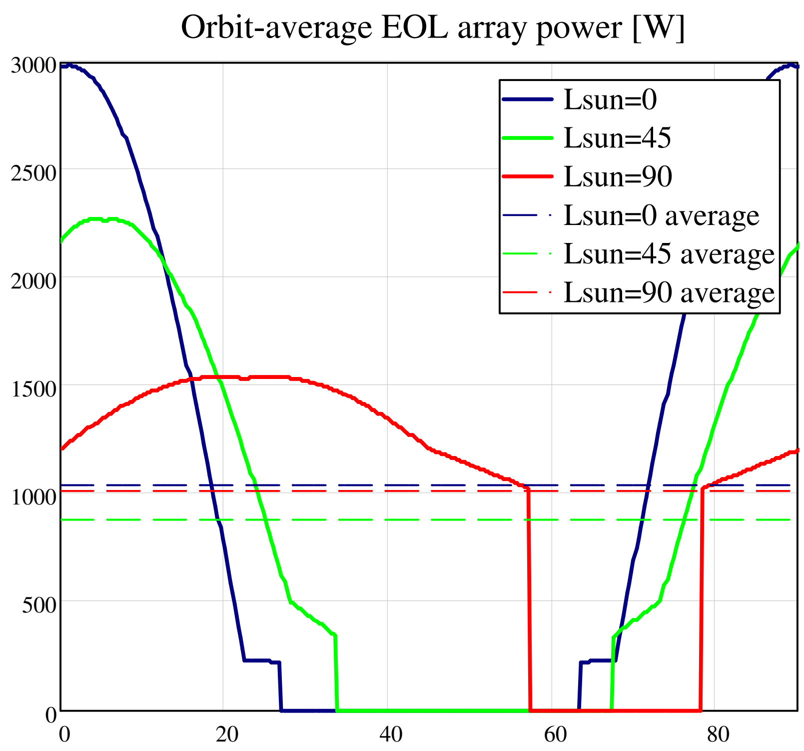


Figure 4-9: Instantaneous and average power provided by the solar array as function of sun longitude w.r.t. the orbit plane (L_{sun}=0 corresponds to sun in orbit plane).

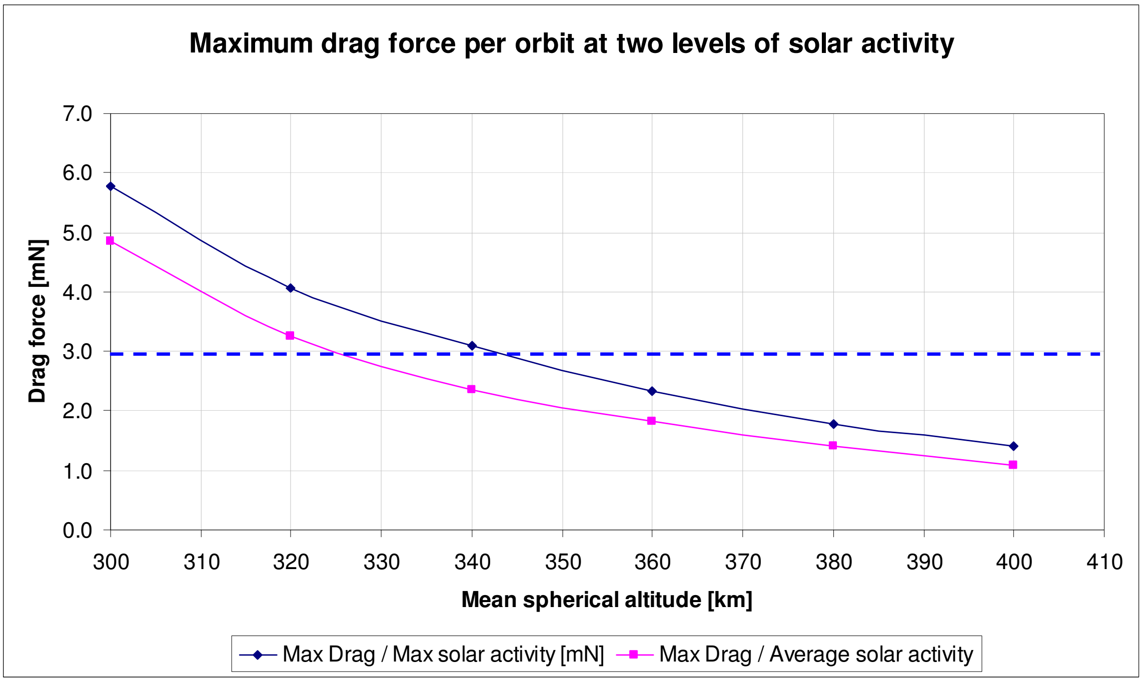


Figure 4-10: Parametric study of the peak X-axis drag force per orbit.

In times of solar maximum, the 3mN upper limit would require flying at altitude above 340 km. In times of average solar activity, the design altitude is 325 km.

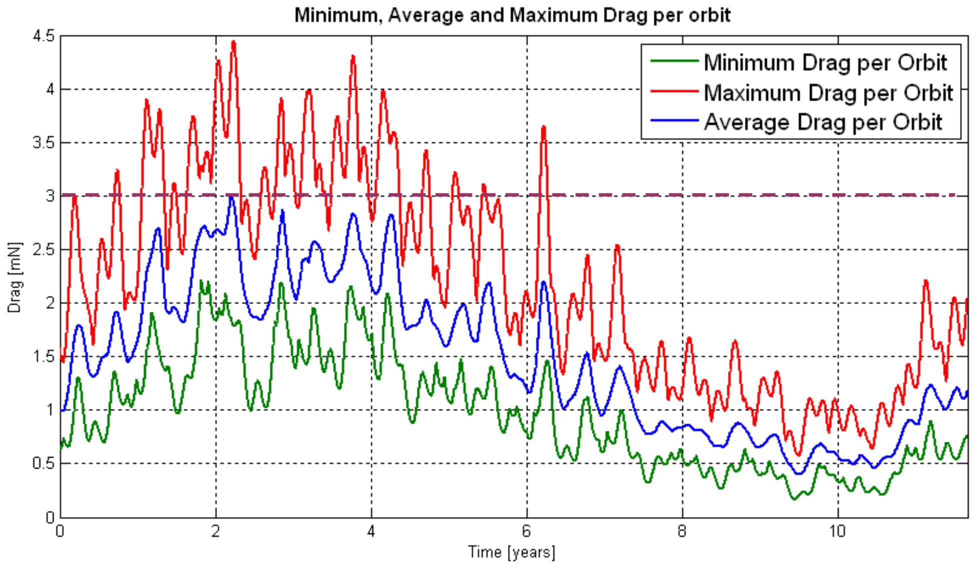


Figure 4-11: Min/average/max drag force per orbit at 300 km altitude under maximum solar/geomagnetic activity conditions. The 3mN limit is exceeded above the black dotted line.

4.2 Concept 2: One pendulum tandem in polar orbit

In a "pendulum" formation the satellites are placed on intersecting orbits, with different inclination or longitude of ascending node. The initial conditions produce a cyclic rotation about yaw, phased with the orbital period, of the line joining the centres of the two satellites, yielding a cross-track (out-of-plane) component of the gravity gradient measured by SST. The satellite-to-satellite distance *d*, too, experiences a cyclic oscillation. In the case of the NGGM, the initial elements are selected such that max(*d*) ≤ 100 km (a requirement of the metrology).

A pendulum formation can solve the striping problem, provided the yaw angle has maximum amplitude of about 45° and the peak cross-track component occurs at the equator (see below). Per se, one pendulum does not solve the spatial/temporal resolution problem, however it could be considered the minimum viable option if the cost constraints forbid more than 1 tandem. Therefore we begin by considering a single-tandem pendulum formation. In order to provide coverage of the whole sphere, a polar orbit is needed.

4.2.1 *Scientific performance*

The performance of the Concept-2 formation is assessed in TN6/Part2 [RD-21], by an analytical quick-look tool, and in TN5/Part2 [RD-18] with a fully-fledged numerical simulation. A thorough discussion is provided therein; here only a short summary is given for completeness. Figure 4-12 shows a comparison of the performance of a pendulum tandem mission (single tandem in 335-km polar orbit, with the instrument performance as specified in the figure) assuming different yaw angle amplitude α. The most accurate and homogeneous results are found for yaw angle α = 45°. Sensitivity and isotropy are improved compared to the collinear formation.

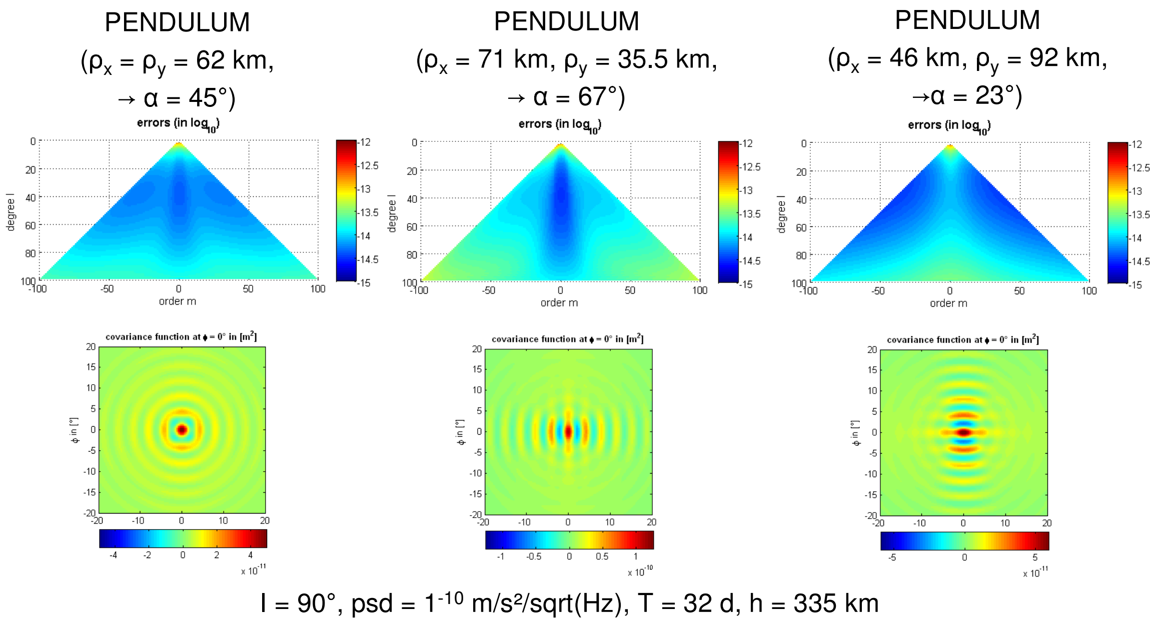


Figure 4-12: Performance of Concept-2 formations [RD-21]

4.2.2 Spacecraft implementation

The pendulum formation reduces to the collinear formation when the yaw angle amplitude α is small, hence the spacecraft formation outlined in §4.1.2 may be expected to work for the pendulum too. In particular, the attitude to the planet is similar (which is not the case for the cartwheel, see further on).

The major difference is the distribution of the drag forces on the spacecraft surfaces. Figure 4-13 shows the distribution of the drag forces assuming the configuration in Figure 4-2, without the deployable solar panel wings. Due to the yaw-steering motion, the direction of the force moves with the orbit period from the X-side (peak force equal to that of the collinear formation) up to 45° from the Y side (because of ratio of areas, the peak force is numerically about twice that on the X side). The force in the Z-direction, too, is numerically large (about equal the X-force) because of the ratio of surface areas.

Because of the behaviour of the drag forces:

- Thrusters of comparable magnitude must be implemented to counteract the large drag force components in the X-Y plane
- The seasonally variable solar array configuration of Figure 4-2e/f is no longer viable (it would multiply by another factor ~2 the drag force acting on the Y-side). Hence there is a shortfall of electrical power when the sun is far from the orbit plane (see Figure 4-14).

The general consequence is that the pendulum formation must fly at higher altitude than the collinear formation, and/or a variable-altitude profile has to be implemented, raising the operating altitude in inverse proportion to the available solar array power. Assuming the 1-kW solar array limit driven by launcher fairing constraints, and the 420W allocation to electric thrusters in Table 4-2, the maximum compensable force component is about 1.4 mN, under somewhat optimistic assumptions about the thrusters' electrical efficiency (see Figure 4-15). Extrapolation from Figure 4-10 shows the 1.4mN limit would allow operation between 380 km and 400 km depending on the phase in the solar cycle. The effect on the mission performance is very significant (Figure 4-16).

As noted above, 1kW would not be available with the sun far from the orbit plane, which would imply even higher altitude in the corresponding seasons. Given the flexibility of the electric thrusters and the adaptability of the configuration, one could envisage a scenario in which the tandem would fly in pendulum mode whenever allowed by the power constraints and in the collinear mode otherwise. The alternative is to fly an SSO with near-constant solar aspect of the solar array.

The mass budget is about 30 kg less than that of Table 4-1 (no solar panel wings and lower propellant mass budget).

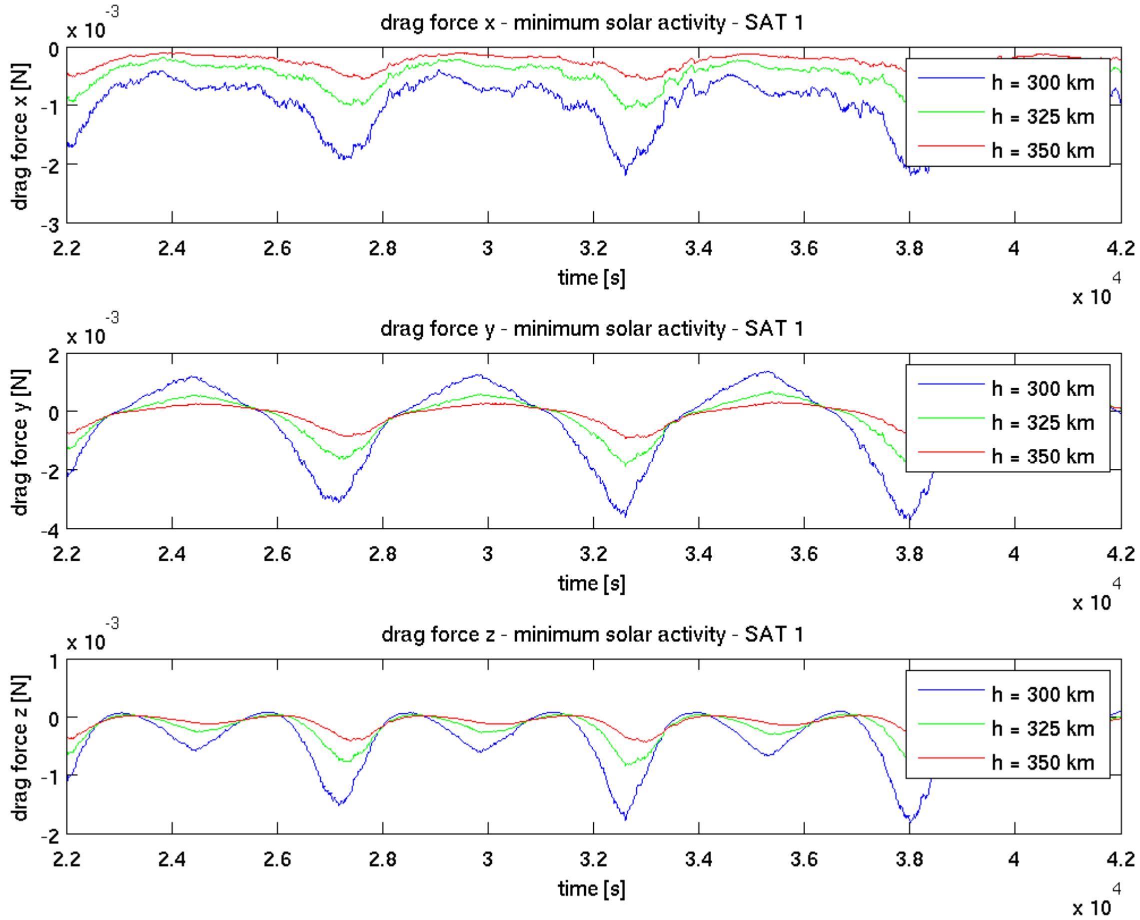


Figure 4-13: Simulation of the drag forces on the envisaged spacecraft in the pendulum tandem formation.

The peak Y-axis force is about twice the peak X-axis force, and the peak Z-axis force is the same magnitude as the peak X-axis force. These ratios are conserved whatever the phase of the solar cycle (minimum solar activity shown).

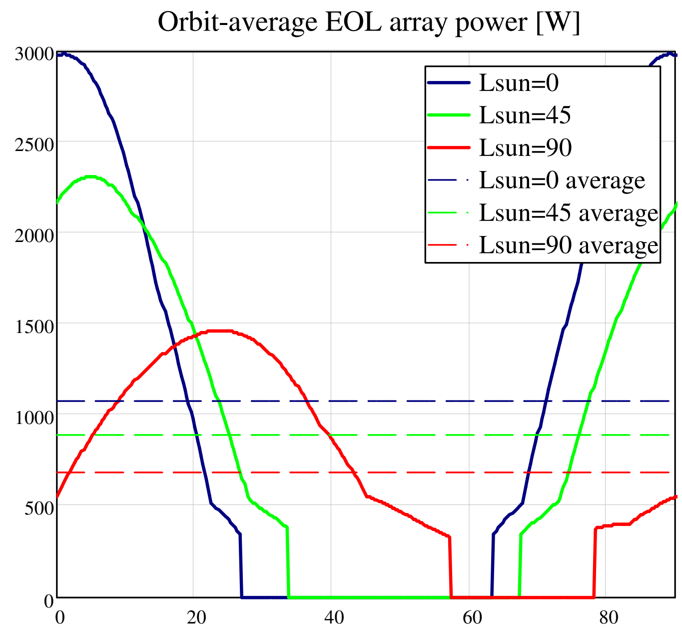
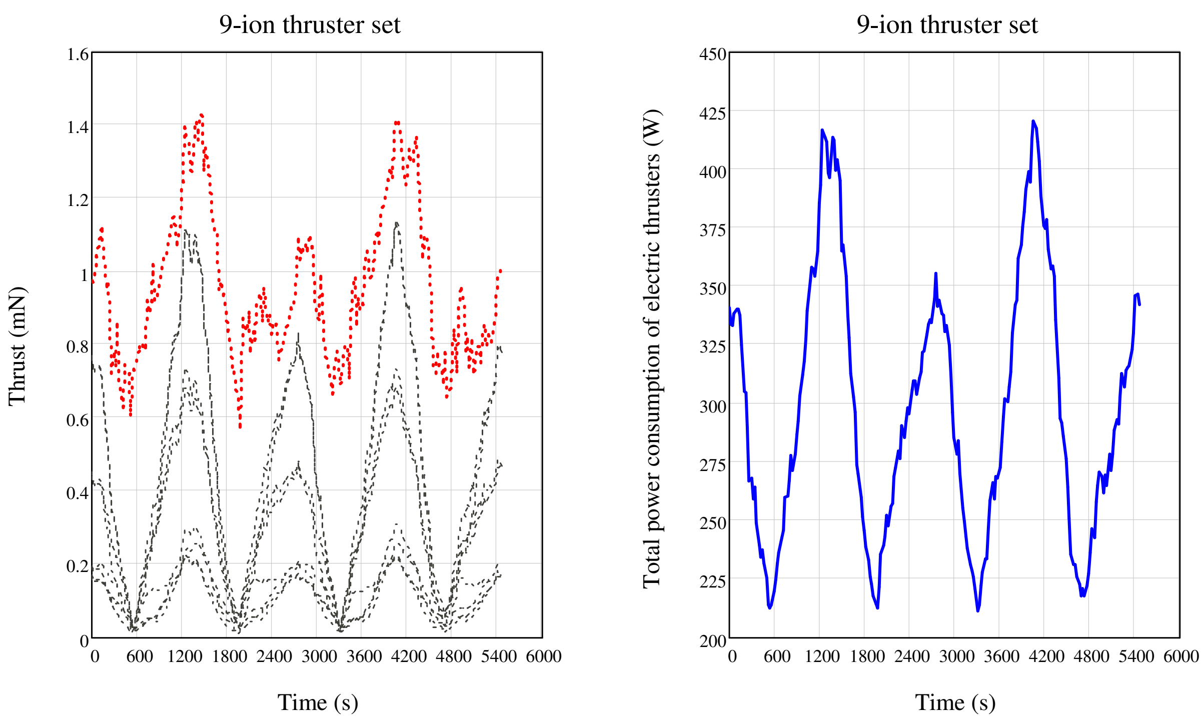


Figure 4-14: Instantaneous and average power provided by the solar array vs. sun longitude in the orbit plane (L_{sun}=0 corresponds to sun in orbit plane). Pendulum formation, α=45°.



red: X-axis component, grey: other components

Figure 4-15: Simulation of thruster usage under a power cap of 420W for the electric thrusters.

The peak power consumption occurs close to the peak of the X-axis force. The maximum compensable X-axis force component is 1.4 mN [Configuration of 9 (active) thrusters; assumed power to thrust ratio of 40 W/mN].

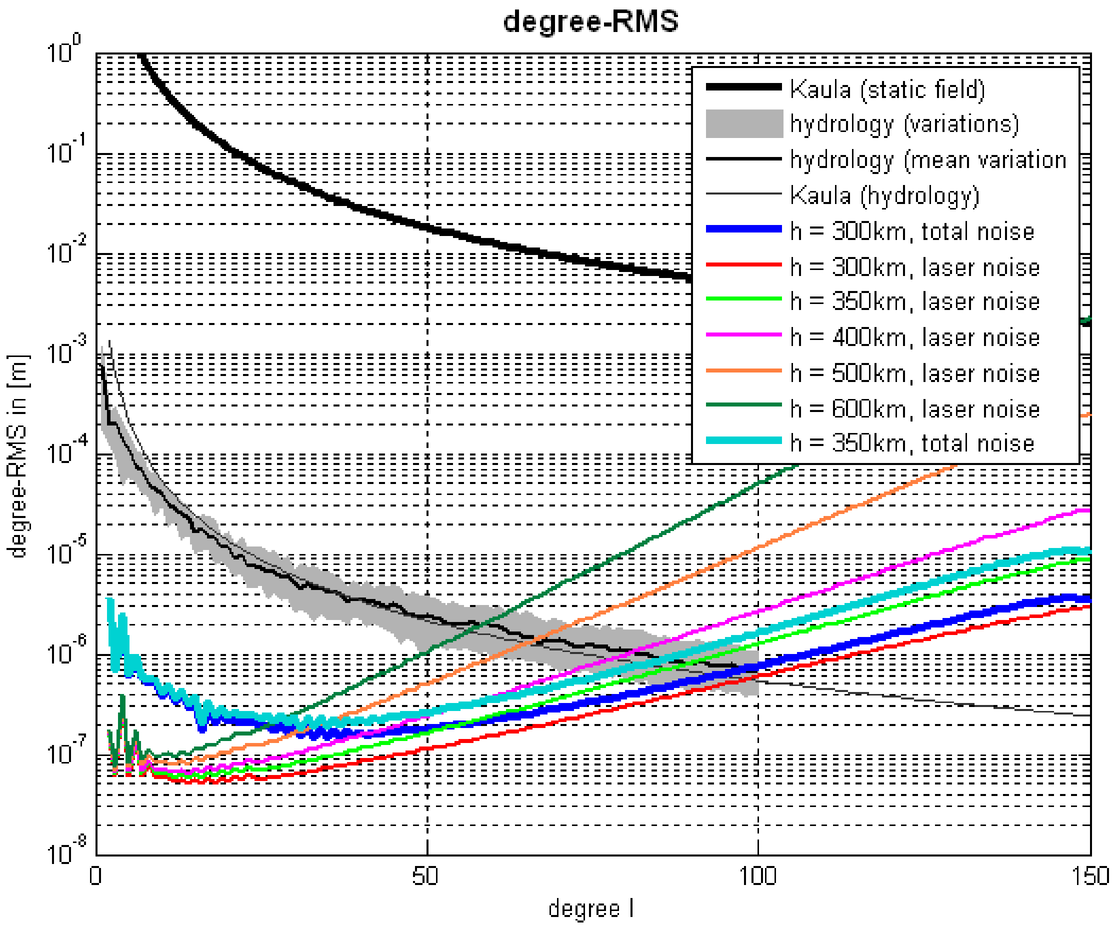


Figure 4-16: Effect of mission altitude on degree resolution (from TN6 Part 2).

[Total noise (laser + accelerometer) vs. laser noise only; optimistic laser PSD , ρ = 100 km, i = 89°, T = 15 d]

A notable feature of the pendulum formation is the generation of harmonic terms, at multiples of the orbit frequency, in the power spectrum of the linear and angular acceleration (see [RD-15] and the example in Figure 4-17). The potential impact on the scientific performance is being assessed (see [RD-18], in preparation).

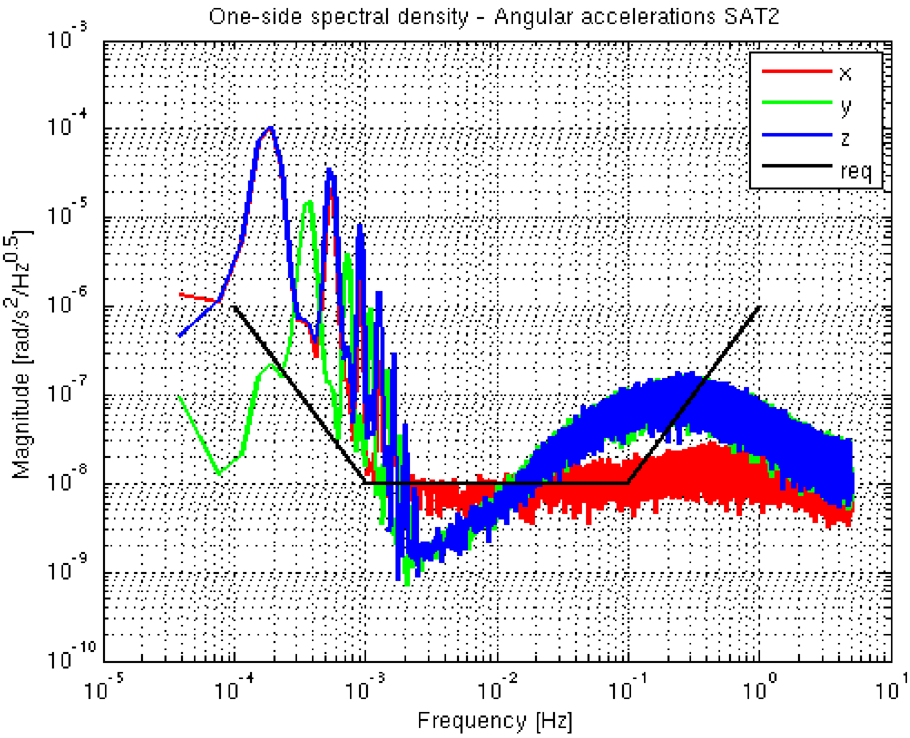


Figure 4-17: Pendulum formation - One-sided spectral density of the SAT2 angular acceleration in SAT2 body reference frame (requirement considered in previous study phase).

4.3 Concept 3: One cartwheel tandem in polar orbit

In a "cartwheel" formation a line joining the satellites' centres of mass oscillates with half of the orbit period around a fixed direction in inertial space. This topology can be obtained by selecting the initial conditions with suitable differences in the eccentricity *e*, argument of perigee ω, and mean anomaly. The satellites are located in the same orbital plane. Though a cartwheel can be obtained for any Δω<180º, a difference of 180º yields the orbits closest to circular. The fixed mean inertial direction can be defined at will.

A cartwheel formation too can solve the striping problem, provided the mean inertial direction is suitably chosen. Here we will address a single-tandem formation in polar orbit or in SSO.

4.3.1 Scientific performance

The performance of the Concept-3 formation is assessed in TN6/Part2 [RD-21], by an analytical quick-look tool, and in TN5/Part2 [RD-18] with a fully-fledged numerical simulation. A thorough discussion is provided therein; here only a short summary is given for completeness. Figure 4-18 shows a comparison of the performance of a cartwheel tandem mission (single tandem in 335-km polar orbit, with the instrument performance as specified in the figure) assuming different orientation of the mean sat-to-sat vector in inertial space. The most accurate and homogeneous results are obtained if the cartwheel is radial at the equator; radial at mid-altitude (φ = ± 45°) yields good results too. The cartwheel, too, displays improved sensitivity and isotropy when compared to collinear-formations. The accuracy is similar to the pendulum.

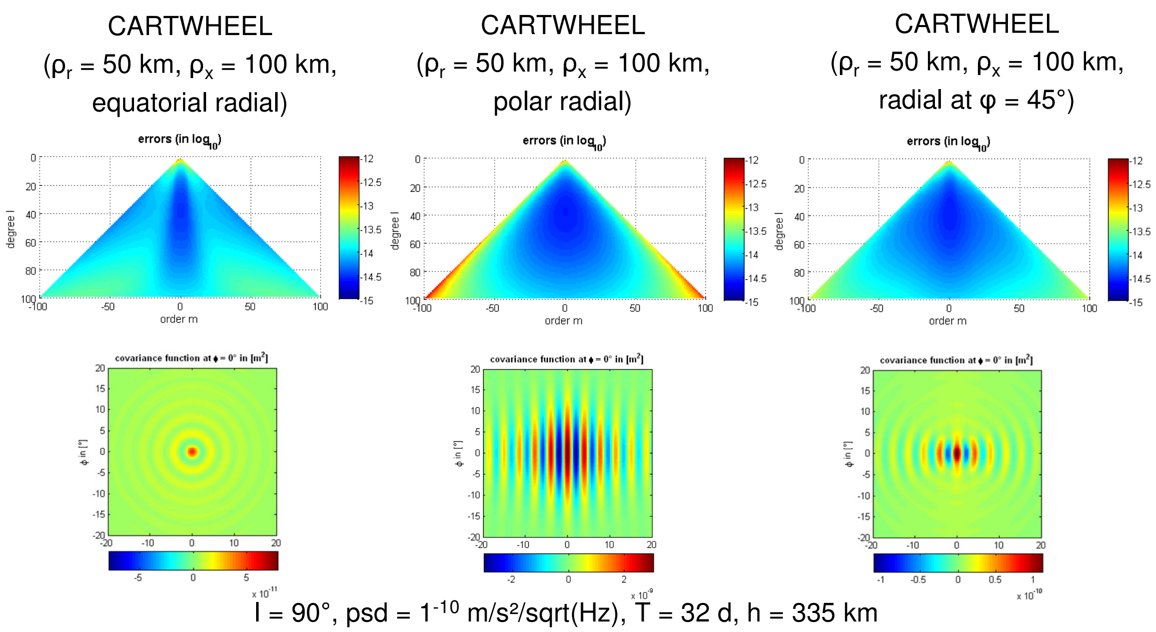


Figure 4-18: Performance of Concept-3 formations [RD-21]

4.3.2 Spacecraft implementation

In a cartwheel formation, the main component of the drag force, aligned with the orbital velocity, turns around the satellite once per orbit. Each satellite must rotate to stay aligned with the inter-satellite line thus keeping the laser beam correctly pointed. As a consequence, the long prism configuration of Figure 4-2 is no longer suitable. A more convenient configuration is approximated by a cylinder with the symmetry axis perpendicular to the orbit normal (Figure 4-19). Each satellite would librate about the symmetry axis to maintain the laser beam pointing. The main thrusters would be distributed all around the cylinder.

Again, in polar orbit one is faced with the problem of generating enough solar power when the sun is in the orbit plane (normal to the cylinder) as well as when it is normal to the orbit plane (normal to the circular top). The ratio of circular cross section to lateral area is ~½π(r/h) and, to get approximately equal power in all seasons, one would require a cylinder height approx. equal to 1.5 times the fairing radius (1.3 m in Vega), leading to a huge drag cross section of 5m². On the other hand, a solar array of 5m² would not be sufficient.

In sun-synchronous orbit, the solar power problem is decoupled from the drag problem. The cylinder could be approximated by a squat prism, and enough solar array area could be installed by deploying solar panels initially stowed against the sides. In a practical design (providing equipment volume about equal to that of Figure 4-2), the drag cross section would be about twice that of Figure 4-3c/d, leading to similar conclusions as the pendulum: the operating altitude would have to be chosen around 400 km.

In the cartwheel, the attitude is no longer earth-referenced but oscillating around an inertial direction, leading to a number of problems that would need dedicated study (sun on the radiators, star tracker blinding by sun and earth). Moreover, the inter-satellite line gets aligned with the satellite-Earth direction twice per orbit, with potential interference (optical and thermal) between the Earth radiation and the laser metrology.

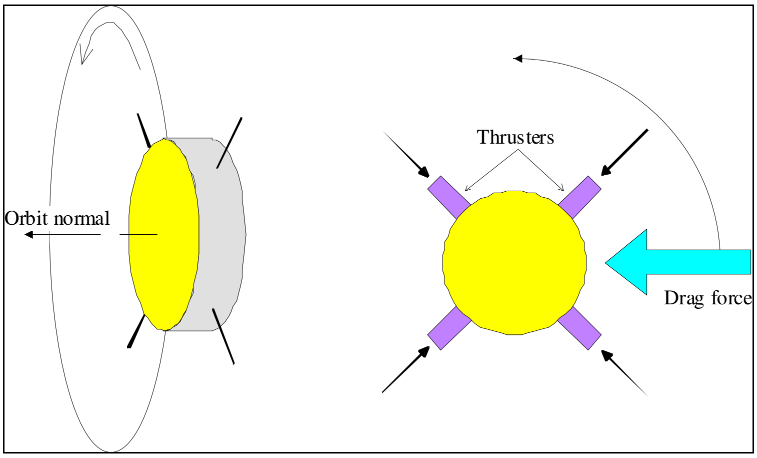


Figure 4-19: In a cartwheel formation, the satellite ought to be a squat cylinder with the symmetry axis aligned to the orbit normal.

As in the pendulum, the satellite-to-satellite distance experiences large oscillations and a significant Doppler shift, with impact in the metrology dynamic range. The cartwheel geometry is obtained from an initially collinear formation by applying forces to each satellite along the orbit radius, in opposite directions. The initial acquisition requires considerable time and propellant consumption. The formation needs to be maintained against differential drift of the line of apsides of the two orbits, with non-negligible cost of formation maintenance.

We conclude that the cartwheel formation is only viable in SSO, at high altitude, and suffers from extra complications w.r.t. a pendulum formation.

4.4 Combinations of the basic architectures

By taking into consideration the 3 basic architectures (collinear, pendulum, cartwheel), and, as parameters, the altitude and inclination, one can produce the following matrix of 18 architecture options:

	a. single tandem polar	b. single tandem SSO	c. 2 tandems, polar	d. 2 tandems SSO	e. 2 tandems, polar & 64°	f. 2 tandems, polar & SSO
1. Collinear					L	L
2. Pendulum	H	H	H			H
3. Cartwheel		H		H		

The collinear formation is the only one that can fly at low altitude (300 km) under the implementation constraints assumed here. It is not very attractive in single tandem form (because of the north-south bias), and 2 tandems either polar or SSO do not solve the problem. The 2-tandem formation has optimal characteristics in the polar+64° form; the variant with polar+SSO tandems could be attractive in a scenario with 2 independent funding agencies (each would provide 1 pair, each pair would be valuable on its own, the best performance would be achieved by combining both data sets). Therefore we keep options 1a/1b (for comparison purposes) and 1e/1f for further consideration as architecture candidates.

The pendulum formation is must fly at higher altitude (400 km) under the implementation constraints assumed here. It is valuable in single tandem form (solves the north-south asymmetry); it has fewer implementation problems in SSO. In a scenario with 2 agencies, each could provide 1 tandem, each valuable on its own, best if both are polar, but one agency could choose SSO and the combined mission would still be near optimal. Therefore we keep options 2a/2b/2c/2f.

Under the implementation constraints assumed here, the cartwheel formation is only viable at high altitude (400 km) <u>and</u> in SSO. It could be valuable in single tandem form (it solves the north-south asymmetry although not so well as the pendulum); 2 tandems would improve the temporal resolution. Therefore we keep options 3b/3d.

In the following chapter we address in more detail the comparative analysis of the 8 remaining options.

5. PRELIMINARY RANKING OF MISSION ARCHITECTURES

5.1 Architecture Selection Criteria

For a preliminary shortlist of mission options, we adopt the following selection criteria:

1. Scientific performance
2. Implementation risk
3. Programmatic impact.

The scientific performance will be evaluated by the degree of compliance with the scientific criteria outlined in Chapter 3.1:

- Spatial coverage
- Temporal coverage
- Spatial resolution
- Temporal resolution
- Accuracy,

as screened by the science team in Chapter 3.2, i.e. in terms of orbit parameters (altitude, inclination) and measurement performance.

Moreover, we shall assess in a qualitative way the ability of the given architecture of coping with the problems of:

- "striping"
- Aliasing.

Under implementation risk, we shall comprise the following items:

- technology readiness
- budget (mass, power) margins
- implementation complexity, as outlined in TN2 [RD-11].

As for programmatic impact, we shall address:

- comparative cost, as outlined in TN2 [RD-11]
- programmatic risk
- programmatic flexibility, i.e., for missions with multiple tandems, the degree to which each tandem would constitute a worthy mission on its own.

In this exercise, these criteria are given weights and scores as shown in the following tables. These numbers are proposed for review together with ESA.

5.2 Comparative Assessment

All selection parameters are given scores from 1 (poor) to 4 (excellent).

5.2.1 Scientific performance

Under this criterion we assess the scientific performance under the given implementation constraints, not the performance per se (e.g., the performance of the pendulum formation is affected by the fact that it has to fly at higher altitude than the collinear formations).

The performance criterion is given 50% weight. All of the 7 individual criteria concurring in the performance evaluation are given the same weight.

Spatial coverage

Spatial coverage defines how much of the terrestrial sphere is covered. Formations including at least 1 polar component cover the whole sphere and are given a score of 4 (excellent). Single-tandem formations in SSO are given a score of 2 (fair).

Temporal coverage

Temporal coverage is defined by the duration of the mission. All architectures are assumed to be designed for 10+ year life (score=4). The consumables might limit the life if they are a large fraction of the spacecraft mass; however the calculated propellant mass ratio is modest (about 5%). The cartwheel is likely to have the largest propellant mass fraction (formation acquisition and orbit maintenance) and the long formation acquisition time may detract from the global lifetime (score=3).

Spatial resolution

For a given instrument performance (assumed the same in all options), spatial resolution (maximum resolved spherical harmonic degree) depends on the altitude (see Figure 4-16). Under the given implementation constraints the pendulums and cartwheels must fly at higher altitude than the collinear formations. Multiple tandem formations improve the spatial resolution.

Temporal resolution

Temporal resolution improves with the number of tandems. Orbits with repeat cycles and sub-cycles may improve the time resolution, but this option is available to all formations.

Accuracy

For a given instrument performance (assumed the same in all options), accuracy depends on the altitude. Under the given implementation constraints the pendulums and cartwheels have to fly at considerably higher altitude than the collinear formations. Therefore they are given a score of 1 (poor).

De-striping

Single collinear tandems are given a score of 1 (poor), and the others a score of 4.

De-aliasing

Single collinear tandems are given a score of 1 (poor), pendulum / cartwheel tandems a score of 3 (good), multiple tandems a score of 4.

5.2.2 Implementation risk

The implementation risk criterion is given 25% weight. All of the 3 individual criteria concurring in the performance evaluation are given the same weight.

Technology risk

All the options share the same laser SST system, which currently has low TRL, but without obvious showstoppers. The miniRIT electric propulsion system, too, needs development and tailoring to the specific mission scenario.

Therefore, under this criterion all options get a score of 3.

Budget margins

As noted in §4.1.2 above, mass is critical for a Vega launch, whatever the option. The SSO options have smaller mass and power budgets (smaller solar array for a given power need; easier thermal configuration). The pendulum design has a mass advantage too on account of the higher altitude required because of thruster power demand.

In the approach taken here, the power demand is met by adapting the operating altitude, hence a power margin is paid for by worse scientific performance. If Soyuz were taken as reference instead of Vega, a larger solar array could be installed on board, and e.g. a pendulum design could be flown at lower altitude. However the comparative difference would remain (at constant altitude, collinear is simpler, has a smaller solar array for given power need and is less massive).

The cartwheel spacecraft design is the most constrained, however since its only viable option is SSO, the criticality is reduced.

Implementation complexity

The collinear formation in SSO is considered the least complex (score=4). The cartwheel formation, also in SSO, comes next (score=3). The collinear formation in polar orbit has complex power system, complex propulsion, and difficult thermal control (score=2); these difficulties are further enhanced in the polar pendulum (score =1).

5.2.3 Programmatic Impact

The programmatic criterion is given 25% weight. Within it, cost and flexibility are given the same weight, 40%, for the following reason: the 2-tandem options have low scores for cost (exceeding the target) but high scores for programmatic flexibility, and, at equal weights, the two criteria more or less balance out. Otherwise the 2-tandem options would be excluded a priori owing to the high cost.

Relative cost

The benchmark is the cost of an Earth Explorer mission. The collinear tandem in SSO is evaluated to be in this cost range (see TN2 [RD-11]). The non-SSO tandems are likely to exceed the cost target. Two tandems will exceed it by a factor of 50% or more.

Programmatic risk

Some development risk is associated with the SST laser system (shared by all) and the miniRIT ion thrusters (shared by all). Schedule risk will be higher for the more complex options.

Programmatic flexibility

Single-tandem options have no flexibility. Of the two-tandem options, the ones in polar orbit have an easy fallback in the corresponding single tandem. Of the two-tandem options with different inclination orbits, the one with polar + SSO is more flexible because each tandem would be acceptable on its own.

5.2.4 Evaluation

For a comparison made in the way illustrated above, the relative weighs given the various indices make all the difference. These weighs, and the rationales for giving scores, have yet to be agreed with the science team and ESA. Therefore this issue of this TN is preliminary and meant above all to propose a method for the comparison of architectures.

Table 5-1 (single tandems) and Table 5-2 (dual tandems) summarise the current evaluation. Among single tandems, the pendulum in SSO scores highest. Note however that the pendulum formation is a good candidate only if the loss of sensitivity due to the high operating altitude is tolerable. This depends on the weight given to accuracy relative to the other performance indices.

Among dual tandems, the Bender pair and its variation with polar + SSO orbits rank highest, with a small preference for the latter owing to its programmatic flexibility.

Table 5-1: Evaluation of architecture options – single tandem formations

		Weight	1a - Single collinear tandem polar		1b - Single collinear tandem SSO		2a - Single tandem polar pendulum		2b - Single tandem SSO pendulum		3b - Single tandem SSO cartwheel	
				Score	Rating	Score	Rating	Score	Rating	Score	Rating	Score
1	SCIENTIFIC PERFORMANCE	50%	2.6	1.29	2.3	1.14	2.7	1.36	2.4	1.21	2.1	1.07
	Spatial coverage	14.3%	4		2		4		2		2	
	Temporal coverage	14.3%	4		4		4		4		3	
	Spatial resolution	14.3%	3		3		2		2		2	
	Temporal resolution	14.3%	1		1		1		1		1	
	Accuracy	14.3%	4		4		1		1		1	
	De-striping	14.3%	1		1		4		4		3	
	De-aliasing	14.3%	1		1		3		3		3	
2	IMPLEMENTATION RISK	25%	2	0.53	3.2	0.81	1.8	0.44	3.2	0.81	2.7	0.67
	Technology readiness	33.3%	3		3		3		3		3	
	Budget margins	33.3%	1		3		1		3		2	
	Implementation complexity	33.4%	2		4		1		4		3	
3	PROGRAMMATIC IMPACT	25%	2.0	0.50	2.8	0.70	1.8	0.45	2.8	0.70	2.6	0.65
	Relative cost	40.0%	3		4		3		4		4	
	Programmatic risk	20.0%	2		4		1		4		3	
	Programmatic flexibility	40.0%	1		1		1		1		1	
				2.31		2.65		2.25		2.72		2.39

Ratings: Excellent = 4 Good = 3 Fair = 2 Poor = 1

Table 5-2: Evaluation of architecture options – dual tandem formations

			1e - Two collinear tandems polar and 63deg		1f - Two collinear tandems polar and SSO		2c - Two tandems pendulum polar		2f - Two tandems pendulum polar+SSO		3d - Two tandems cartwheel SSO	
		Weight	Rating	Score	Rating	Score	Rating	Score	Rating	Score	Rating	Score
1	**SCIENTIFIC PERFORMANCE**	**50%**	**4.0**	**2.00**	**4.0**	**2.00**	**3.4**	**1.71**	**3.4**	**1.71**	**2.9**	**1.43**
	Spatial coverage	14.3%	4		4		4		4		2	
	Temporal coverage	14.3%	4		4		4		4		3	
	Spatial resolution	14.3%	4		4		3		3		3	
	Temporal resolution	14.3%	4		4		4		4		4	
	Accuracy	14.3%	4		4		1		1		1	
	De-striping	14.3%	4		4		4		4		3	
	De-aliasing	14.3%	4		4		4		4		4	
2	**IMPLEMENTATION RISK**	**25%**	**0% 2.1**	**0.53**	**0% 2.3**	**0.58**	**0% 1.8**	**0.44**	**0% 2.0**	**0.50**	**0% 2.6**	**0.65**
	Technology readiness	33.3%	3.0		3.0		3.0		3.0		3.0	
	Budget margins	33.3%	1.3		2.0		1.3		2.0		1.8	
	Implementation complexity	33.4%	2.0		2.0		1.0		1.0		3.0	
3	**PROGRAMMATIC IMPACT**	**25%**	**1.2**	**0.30**	**2.4**	**0.60**	**2.2**	**0.55**	**2.2**	**0.55**	**2.6**	**0.65**
	Relative cost	40.0%	1		1		1		1		1	
	Programmatic risk	20.0%	2		2		1		1		3	
	Programmatic flexibility	40.0%	1		4		4		4		4	
				2.83		**3.18**		**2.71**		**2.76**		**2.73**

Ratings: Excellent = 4 Good = 3 Fair = 2 Poor = 1

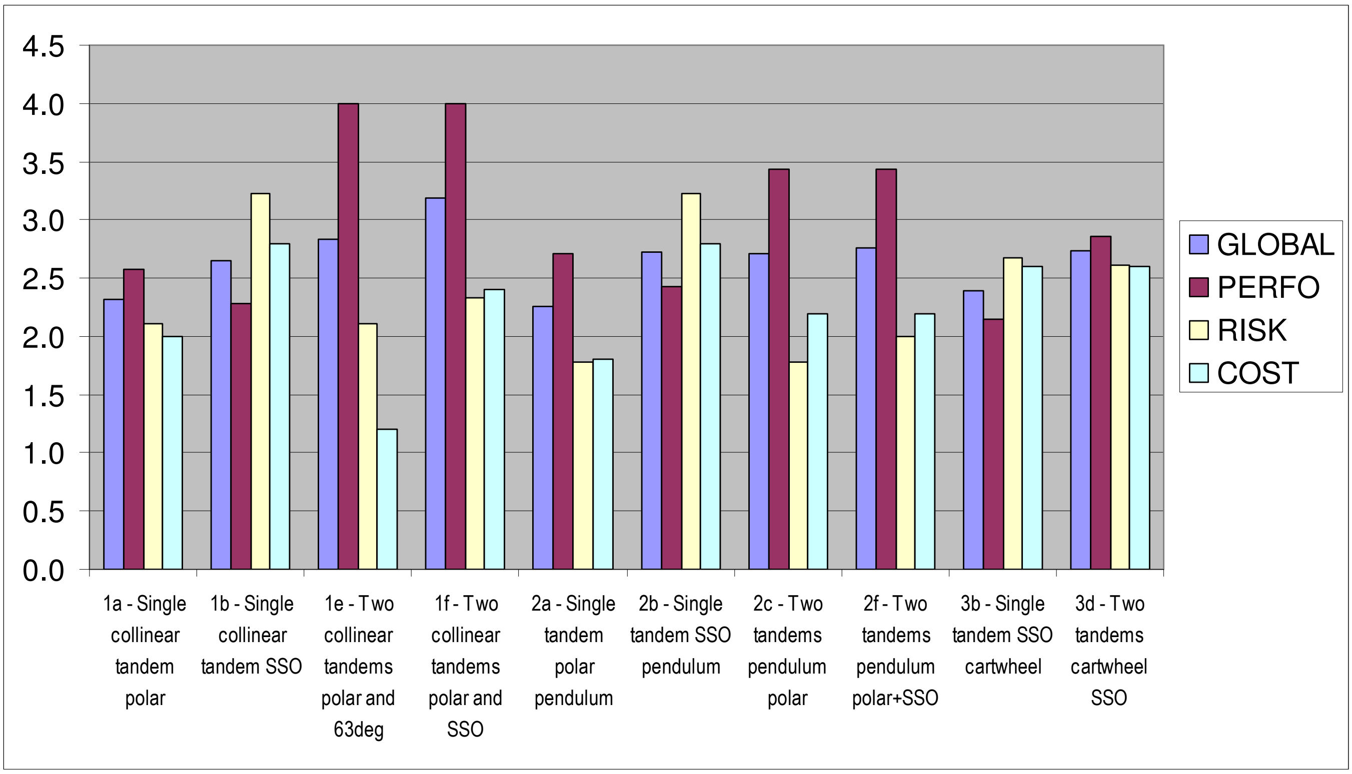


Figure 5-1: Comparative ranking of mission options

6. CONCLUDING REMARKS

Two options seem worth pursuing further:

1. two collinear tandems around 300 km altitude, polar + 64° (or polar + SSO)
2. one or two pendulum tandems, in polar orbit (or SSO), at higher altitude (≅ 400 km).

The mission with two collinear tandems is the only option fully meeting the scientific requirements. Two tandems are needed to solve the striation problem and improve the spatio-temporal coverage. The mass budget margins are critical for a Vega launch. Two Vega launches are required, one tandem each. Two tandems exceed the cost limits of an Explorer mission.

The pendulum is interesting in that it could be viable in the one-tandem configuration too. It is flexible and versatile. Under the given constraints, however, it can only be flown at high altitude, which makes the mission much less attractive scientifically. The mass margins are more relaxed, given the higher altitude.

Each of the options has a variant with SSO, which simplifies the design considerably, at the cost of giving up polar coverage.

Both options require an effort at mass reduction / miniaturisation. As part of this future exercise, a priority is optimised implementation and usage (propellant mass and power consumption) of the ion thrusters.

Issues which are still open at the time of the first draft of this note include:

- scientific value added by 1- or 2-axis gradiometer on board
- value of repeat cycles /subcycles for improving the time coverage.

Preliminary answers are expected to come from the ongoing end-to-end simulations.

7. ACRONYMS AND ABBREVIATIONS

ACC	Accelerometer
ACS	Attitude Control System
AD	Applicable Document
BOL	Beginning of Life
BSM	Beam Steering Mechanism
CDMU	Command and Data Management Unit
CGE	Cumulative Geoid Error
COM	Centre of Mass
DFC	Drag Free Control
E2E	End-to-End
EOL	End of Life
EWLT	Equivalent Water Layer Thickness
FC	Formation Control
GIA	Glacial Isostatic Adjustment
GNSS	Global Navigation Satellite System
GOCE	Gravity field and steady-state Ocean Circulation Explorer
GPS	Global Positioning System
GRACE	Gravity Recovery And Climate Experiment
INRIM	Istituto Nazionale di Ricerca Metrologica
IPA	Ion Propulsion Assembly
ITT	Invitation To Tender
KBR	K-Band Ranging
LEO	Low Earth Orbit
ll-SST	low-low Satellite to Satellite Tracking
LORF	Local Orbital Reference Frame
LRR	Laser Retro Reflector
MBW	Measurement Bandwidth
MST	Mission Simulation Tool
NGGM	Next-Generation Gravity Mission
PCDU	Power Control and Distribution Unit
P/L	Payload
POD	Precise Orbit Determination
PSD	Power Spectral Density
RD	Reference Document
RF	Radio Frequency
RIT	Radiofrequency Ion Thruster
RMS	Root Mean Square
S/C	Spacecraft
SLR	Satellite Laser Ranging
SOW	Statement of Work
SSO	Sun Synchronous Orbit
SST	Satellite to Satellite Tracking
STR	Star Tracker
TAS-I	Thales Alenia Space Italia
TBC	To Be Confirmed
TBD	To Be Defined
TN	Technical Note
TT&C	Tracking Telemetry and Command
WP	Work Package

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