# XEUS Mission Reference Design

Marcos Bavdaz, David H Lumb and Tone Peacock
Science Payloads & Advanced Concepts Office, European Space Agency, ESTEC, Postbus 299,
2200 AG Noordwijk, The Netherlands

**ABSTRACT**

The Xeus mission is designed to explore the X-ray emission from objects in the Universe at high redshifts, and these science requirements necessitate a very large effective area. We describe a completely revised mission scenario that mitigates previous concerns about the deployable mass and use of the ISS. New mirror technology with lightweight optics enables a direct launch to a L2 operational orbit, and we describe the outline of the Mirror and Detector Spacecraft that are deployed in formation flying to achieve the 50m focal distance separation.

**Keywords:** Instrumentation; imagers; spectrometers; X-ray; missions

## 1. INTRODUCTION

The science drivers for the *XEUS* (X-ray Evolving Universe Spectroscopy) mission are described in this volume[1]. The need to investigate the genesis and evolution of massive black holes to the highest red shift requires a combination of large effective area and high angular resolution. This combination naturally facilitates investigations across all other areas of astrophysics. Bavdaz et al[2] describe the mirror design that meets these requirements. One of the key features of that design is a long focal length that allows to maximize reflectivity and to enable the use of a Wolter 1 conical approximation with good resolution. The chosen focal length of 50m immediately implies that the mirror and focal plane must be implemented on separate spacecraft, in a formation-flying configuration.

The large effective area to be deployed is ~100 times greater than the current generation of *XMM-Newton* and *CHANDRA* observatories, and initial designs based on XMM replication technology dictated that a very large mass should be deployed to Low Earth Orbit (LEO). To enable a subsequent growth phase for the mirrors, a scheme was adopted to deliver mirror segments to the International Space Station (ISS) and perform a mirror spacecraft (MSC) docking and a series of robotic assembly stages to upgrade the telescope to the final configuration. Even before the announcement of the phase-out of the ISS, it was realized this entailed a complex and expensive set of multiple launches and spacecraft man-rating qualification. Finally the formation-flying in LEO was found to impose a very large burden of propellant usage on the active detector spacecraft (DSC) in order to maintain its non-Keplerian orbit at the mirror focal point. This scenario was discussed in Parmar et al [3] and Bleeker et al [4].

Clearly, not only must a very much improved mass-efficient technology be developed, but also a more simple deployment scheme is necessary to minimise the number of launches and the operational complexity, and to reduce the costs to match the financial resources available for the next generation of Observatories. With the introduction of the Soyuz-Fregat launch vehicle from the Kourou equatorial site, ESA has access to a capable, dependable and relatively cheap option for large payloads, with a capability to deliver ~2 tonne to the second Langrangian point of the Sun-Earth system (L2). The *Science Payloads and Advanced Concepts Office* has therefore actively studied new mission concepts for XEUS that are compatible with such a launch option, especially considering the advantages of a direct injection to an operational orbit at L2. The activities culminated in a study supported by the ESTEC *Concurrent Design Facility* in Spring 2004, and this paper summarises the main conclusions of these studies.

## 2. MIRROR SPACECRAFT (MSC)

The sub-systems of the MSC serve only to maintain the direction of the telescope axis towards the target location on the celestial sphere, and to preserve the thermal and stray light environment within acceptable limits. In the formation-flying scenario the MSC remains passive, requiring the DSC to follow the MSC focal point. Therefore the MSC becomes a relatively simple spacecraft, where by far the largest portion of the mass and volume comprises the mirror elements themselves. Once the MSC reaches the L2 orbit, the propellant usage is limited to orbit maintenance and momentum wheel dumping, and with relatively small slews around its telescope axis the propellant can be sized for a very long duration. The undemanding requirements of the spacecraft sub-systems also allow for generous margins in power generation capability. These features allow sizing of the spacecraft for 15 years operational duration.

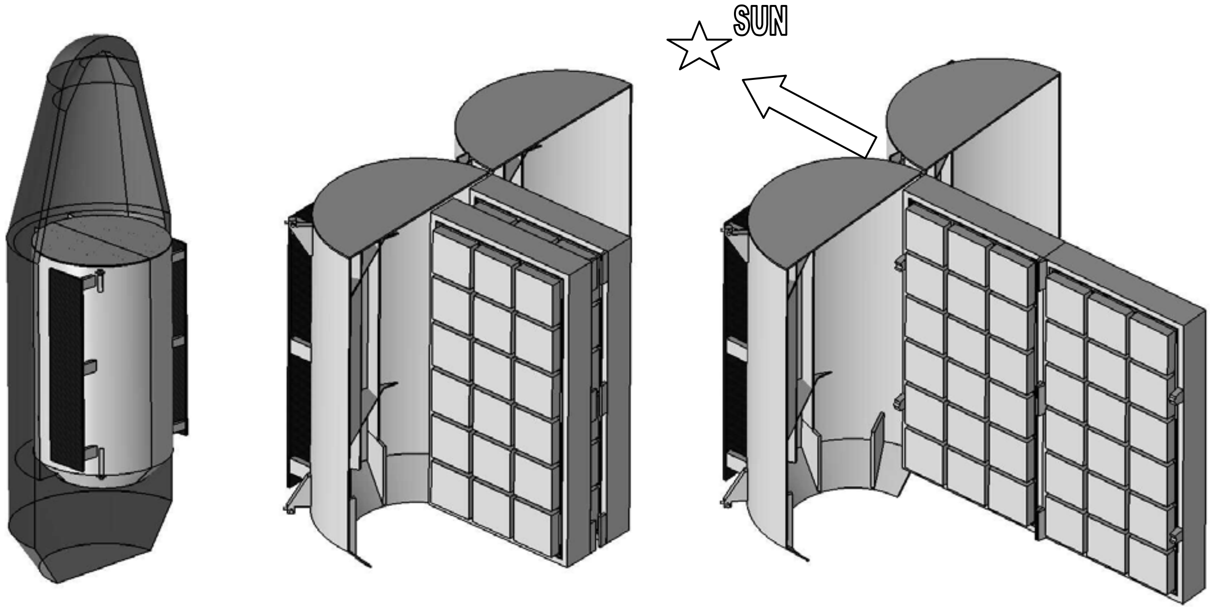


**Figure 1** (a Left): The configuration of the MSC where the full 3.8m diameter of the widest portion of the S-F fairing is occupied by a canister that folds out (b - Middle) to reveal the folded optical bench section which then in turn unfold to deploy the (c – Right) mirror petals in the appropriate location. The dark external panels are the solar arrays. The equipment bays hold electronics boxes on the mirror-facing side of the canister, but will be insulated from the mirror systems to avoid thermal gradients. The canister acts as the sun-shade to maintain the mirrors at a suitable temperature when the sun is maintained within a few degrees of a line parallel with the mirror plane

The most demanding constraint remains the packaging of the mirror "petals" within the volume of the Soyuz-Fregat fairing, using as efficient an optical bench structure as feasible within constraints of providing alignment and thermal control elements. Figure 1(a) shows the concept with the mirrors folded in half and inside a "canister" for the stowed configuration. The fairing allows a diameter of ~3.8m and length of 5 metres at the maximum extent. The set of mirror petals will be individually aligned, and the support structure and external canister designed to optimize thermal performance by minimizing radiative heat loads to the mirror faces, and direct conduction into the mirror mounting. Initial analysis shows that the most critical limit of thermal gradient across the small depth of mirrors plus collimators can be maintained to less than <<5°.

The outer diameter not only provides the launch structural support and covers, but when deployed (Figure 1b), the two halves provide a sun-shading function At the L2 location the Sun, Earth and Moon remain in approximately a collinear direction, and the fixed sun shade allows for very efficient rejection of stray light, and ensures that the mirrors receive a very constant and small thermal load. Detailed analysis shows that a temperature constraint of +/- 1K can be achieved when the telescope remains within +/-15 degrees of the normal to the Sun line. (Figure 1c).

An alignment system has been designed to accommodate possible changes in the coalignment of petals due to launch loads and thermal relaxation on reaching the L2 location. For each petal an array of small actuators can provide a small force necessary to provide the sub-mm travel, and with a total of 108 actuators a mass of ~35kg is required, inclusive of launch locks etc.. A small reflector on the mirror petal is used together with a scanning laser and readout camera on the Detector Spacecraft to make the required adjustment.

Pointing and slewing is maintained by a set of reaction momentum wheels, and the budget allows for momentum off-loading by a cold gas system. It was determined that cold gas is the preferred choice for propulsion, because it minimizes the possibility for contamination on the mirror surfaces once the canister has been deployed. Nevertheless thrusters are arranged on the solar shade shells to make available torques without plume impingement on the mirrors. Hydrazine is used in the early stages before deployment as the propellant for launch dispersion correction.

A relatively low gain X-band antenna suffices for transmission of the low rate of housekeeping data, which can be accommodated with 1 – 2 hours downlink windows per day at a rate of ~9kbs, assuming a mass memory storage of 50Mbits.

Table 1 shows the calculations of the MSC mass budget, including various margins based on sub-system maturity status, and indicates how the 2.05 tonne launcher capability for direct L2 injection is met.

| **Dry Mass System Contribution** | **Without Margin** | **Margin %** | **Total (kg)** |
|---|---|---|---|
| Structure/Optical Bench | 530 | 10 | 583 |
| Thermal Control | 33 | 20 | 40 |
| Mechanisms | 54 | 15 | 62 |
| Pyrotechnics | 5 | 5 | 5 |
| Communications | 17 | 10 | 19 |
| Data handling | 15 | 20 | 18 |
| AOCS | 35 | 5 | 37 |
| Propulsion | 31 | 7 | 33 |
| Power | 40 | 10 | 44 |
| Harness | 63 | 20 | 76 |
| Mirror | 738 | 0 | 738 |
| Metrology | 2 | 5 | 2 |
| **Total Dry (excl adapter)** | **1563** | | **1657** |
| System Margin | | 20 | 330 |
| **Total Dry with margin** | | | **1988** |
| Propellant | 54 | 0 | 54 |
| Separation mechanism | 12 | 0 | 12 |
| **Launch Mass** | | | **2054** |

**Table 1** Summary of the mass budget of the MSC

## 3. DETECTOR SPACECRAFT

### 3.1 DSC Requirements

Elsewhere in this volume[5] we describe the package of science instruments baselined for the XEUS science investigations. In addition to the primary wide- and narrow-field imaging spectrometers, there are ancillary instruments for fast timing, hard X-ray detection, extended field imaging and polarimetry. Apart from the last of these, the ancillary instruments would be packaged with the Wide Field Imager (WFI). The hard X-ray camera can be located directly *behind* the WFI because the long focal length and associated focal depth allow the image resolution to be robust to actual placement. The Fast Timing camera has to be operated at an intrafocal position to avoid pile-up, and therefore the DSC along-axis position needs to be variable up to 150mm, although for stability of response should be maintained (during an observation) to within a few millimeters. Laterally the focal position can wander on timescales long compared with the camera temporal resolution (typically >>milliseconds) as long as this location is accurately known. The range of this movement must be small compared with the point spread distribution on the Narrow Field Imagers (NFIs), to ensure signal is not lost. Therefore the DSC must maintain its location relative to the MSC with a tolerance of

1mm laterally and up to 10mm axially. The means of fulfilling this formation-flying requirement is summarized in Section 4

### 3.2 Propellant Sizing

A concern for the LEO version of XEUS had been the excessive DSC propellant in following the MSC in its non-Keplerian phase. We therefore examined the L2 case in some detail. The starting assumptions include:

– Only the DSC (chaser spacecraft) manoeuvres with respect to the MSC (target spacecraft)
– The DSC moves in inertial space
– The DSC RCS uses nitrogen as the propellant
– The DSC RCS thrusters are proportional, manufactured using MEMS technology
– The DSC thrusters have a maximum thrust of 100mN and a specific impulse of 50s
– The DSC mass is ~ 1500kg
– DSC mission life is sized for 5 years consumables

The propellant sizing has to consider 4 phases of the mission: rendezvous, station-keeping, momentum dumping and slews.

- In order to rendezvous to a final distance between DSC and MSC of 100m, we assume the DSC performs the acceleration and deceleration phases in equal amounts of time with equal forces. The cruise speed (relative velocity) is considered a safe relative velocity for the purpose of preliminary design at 10cm/s. For a total time of the approach manoeuvre (see Section 6) of 30hrs, the nitrogen mass consumed is only ~ 1kg.
- For station-keeping the main perturbation is from solar radiation pressure (SRP) only. We assume that the surface area of DSC is arranged for the same ballistic coefficient as the MSC, but the worst case scenario adopted where the DSC is black and MSC is white (or vice versa). It is found that the nitrogen mass needed to cancel the differential SRP acceleration over 5yrs is ~27kg.
- The SRP torque is compensated for with momentum wheels, and taking a total observation time of no more than 90% of the DSC lifetime, and a arm of the SRP torque ~0.3m, with duration of a momentum dump limited to 240s, then the thrust needed for angular momentum dump is ~70mN. The number of momentum dumps is ~275 (for a total of 18.5hrs allocated to angular momentum dumping) and a nitrogen mass per dump 35g or a total 10kg.
- While the MSC moves about its centre of mass (CoM) only, the DSC moves about its CoM while its CoM translates on a circular arc, whose radius is equal to the focal length of the telescope. The "typical" worst case slew angle is taken to be $90^{o}$ and the n umber of slews per year is ~150, each lasting ~ 1.5hrs. Then we take a focal length of 50m which leads to a nitrogen mass per slew maneuver of 140g, leading to a mass consumption in 5 years of 52kg. This mass scales directly with focal length.

### 3.3 Other Sub-Systems

The Data Handling system (CMDH) for the DSC is a centralized unit responsible for satellite-level tasks, such as data compression or selection, organizing the science data stream between different instruments, re-organising instrument readout schemes to maximize science data content, providing mass memory storage, commanding and receiving AOCS computer data, receiving commands etc.. The unit is not expected to be computing power intensive. For an assumed instrument data rate ~100kbit/s a daily transmission of data would suggest a compression factor ~2 and data storage volume of 4Gbits. With 2-3 hours transmission time per day, the TTC system must be able to accommodate a 1Mb/s link rate at L2. We assumed an X-band system with steerable high gain antenna.

Radiators will be required to keep the payload at the correct temperatures and support the cooling of the detectors. The DSC systems (particularly the batteries) will be thermally controlled using dedicated SC panels. The WFI (together with EFI and HXC if mounted in same unit) will be passively cooled using a multiple radiator on the sun-avoiding side of the DSC with TBC use of a Peltier. For WFI the thermal design is driven less by the DEPFET than by the fast readout electronics. To ease the electrical design, the two sub-systems are best co-located and operated at the same temperature, therefore we assumed that 20W are to be dissipated at a temperature of 210K. This requires a radiator area ~0.27m$^2$ depending on emissivity and leaks. For the NFI1/2, flat panel radiators with heat pipes will dissipate the thermal energy from the mechanical closed cycle coolers used to refrigerate the super-conducting detectors to sub-Kelvin temperatures. All cooling was sized from ambient to 20K via a Stirling cooler. For each cooler we estimated a power requirement

~200W that must be dissipated from radiators at 293K. Again subject to emissivity and leaks this becomes a radiator area of ~0.6 $m^2$ per cryo detector. Assuming all radiators are on the spacecraft side that is permanently anti-sun, then a cuboid spacecraft of dimensions ~ 1 – 2 m is compatible.

In order to baffle stray X-ray and optical stray light, a baffle system forward of the instruments ensures that the instrument field of view does not allow unwanted light to the focal plane. Careful design may allow the baffling structure to be permanent and buried within the instrument nominal envelope. For the WFI, it is calculated that to minimize stray X-ray light from outside the true field of view, due to diffuse X-ray background, then a baffle length ~1.5m is necessary. This can be accommodated as part of the spacecraft structure as long as the WFI is close to the anti-MSC side of the DSC. The NFI need smaller baffle lengths, and it is designed to be provided by the functionality of their thermal baffle shields for the most part.

The power budget is dominated by the science instruments, and assuming thermal cycling is avoided on the NFI's, then a worst case operating mode results in an instrument budget of ~900W, including the coolers operating continuously. The DSC mass budget is summarised in Table 2

| **Item** | **IWG Report Configuration (kg)** |
|---|---|
| Instrument Payload | 465 |
| Monopropellant System | 205 |
| AOCS | 39 |
| Rendezvous & Navigation | 15 |
| Data Handling | 15 |
| Thermal SS | 40 |
| Batteries | 5 |
| Power SS incl Solar Arrays | 95 |
| X-Band | 40 |
| Harness | 55 |
| Structure 15% | 146 |
| Launch Adapter | 75 |
| Margin (20%) | 240 |
| **TOTAL** | **1430 kg** |

**Table 2** Based on the instrument complement described in the XEUS Instrument Working Group Report[6], the masses of various sub-systems have been sized. A baseline system comprising WFI and 2 NFI instruments, or a reduced complement comprising a WFI and only one NFI have also been considered. An instrument mass saving of ~200kg could be considered with additional supporting sub-system reductions. However the mass capability of the Soyuz-Fregat already allows a 60% margin to L2 so has not been discussed here.

## 4. FORMATION FLYING

It has been noted above that the spacecraft must fly in formation while being locked to a ~1mm cube to ensure photons are registered on the focal plane. Compared with interferometric missions, this requirement is rather relaxed, and a simple metrology and attitude control system can be considered to satisfy the requirements.

The MSC is required to locate the celestial target with an acquisition accuracy of ~10 arcseconds, driven mainly by the necessity to ensure that the vignetting variation with field angle is very small over the observation. Then from the formation flying aspects, the MSC remains in a passive role, and the DSC ensures that the focal plane spot is tracked with sufficient accuracy.

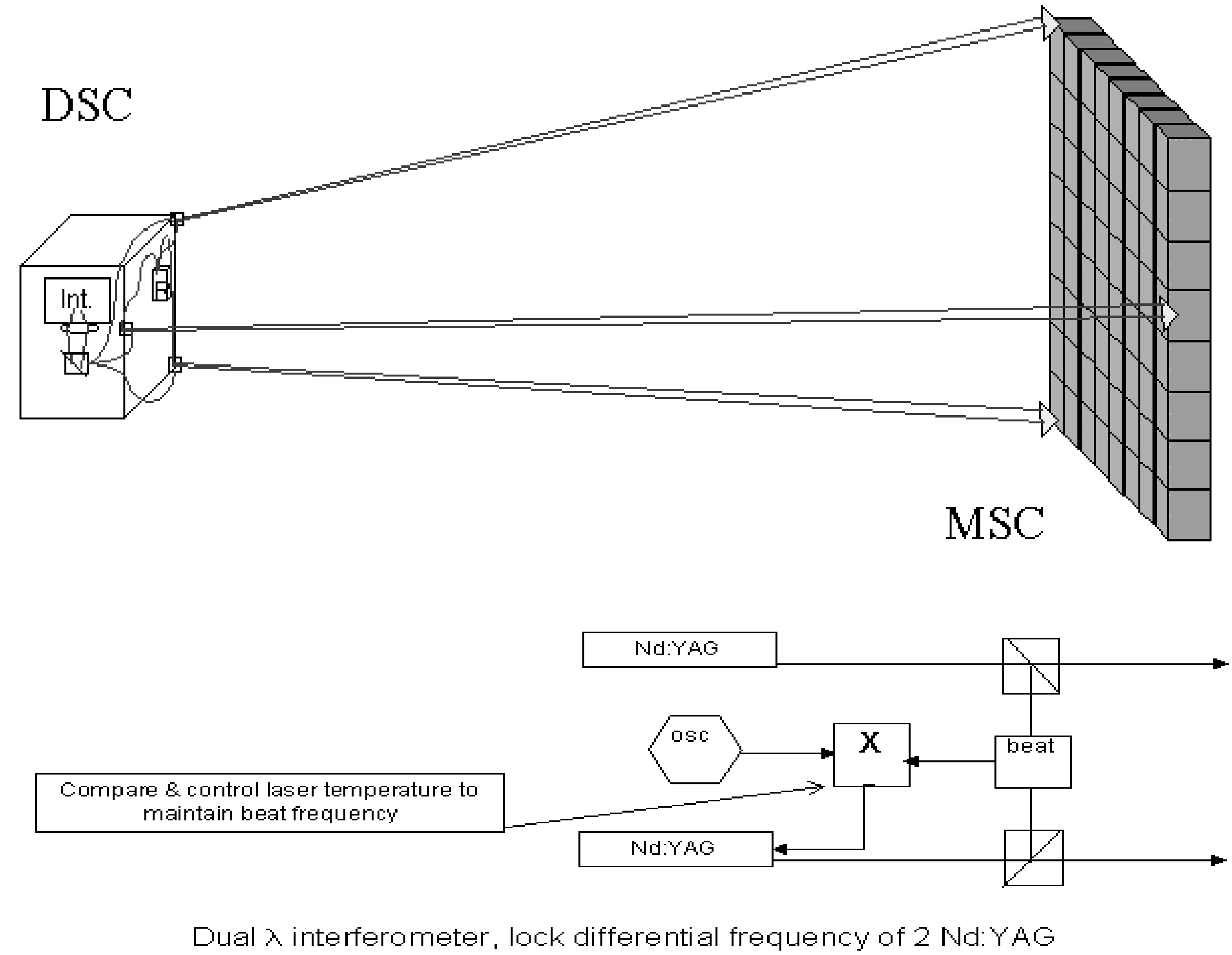


**Figure 2** Concept of the formation-flying metrology

The angular resolution goal of 2 arcseconds HEW demands already that the DSC returns attitude knowledge to 1 arcsecond or better, therefore the DSC as a baseline contains a high quality star tracker. The performance of this unit allows the DSC to be accurately locked to the celestial location, but any relative shift between spacecraft will be measured by the formation flying metrology system.

The system is based on developments for the Darwin project, but already demonstrated as a laboratory breadboard with sufficient accuracy to meet the XEUS requirements. First of all, an *Absolute Distance Measuring (ADM*) laser rangefinder is used to determine the along-axis distance between MSC and DSC. The lab demonstration already offers, at a range of 60m, a resolution of 50 microns. Next a dual wavelength interferometer is used to perform range-finding to 3 corner cubes on the DSC. Two Nd:YAG lasers operating at 3GHz and beat wavelength ~10cm are implemented and the differential frequency lock allows a 3 micron accuracy on lateral shifts given a +/- 2.5 mm unambiguity range from the previous rangefinder absolute measurement. The same optical head is used for both systems and requires modest resources of ~5W and 3kg.

## 5. LAUNCH & RENDEZVOUS SCENARIO

The separation of launchers in principle allows some graceful robustness to failure of one launch subject to a back-up payload being available. Separate launches also allow some decoupling of the development schedules and AIV of the two separate spacecraft. As the more critical item is the MSC, we envisage this is launched first. We have investigated both direct injection and staging at an eccentric High Earth Orbit (HEO) with subsequent L2 insertion with a separate propulsion unit that allows greater mass capability (~300kg). After an exhaustive mass-reduction exercise we believe it is feasible to launch directly to L2, with savings of cost and minimization of risk during early operations. The baseline payload includes a small hydrazine propulsion scheme that accommodates the launcher dispersion and mid-course orbit maneuver, but thereafter all orbit maintenance can be performed with a cold gas propulsion.

The cruise phase to L2 will begin with a short, few weeks, barbecue activity to ensure out-gassing of the spacecraft sub-systems to prevent subsequent contamination of the mirror system. Then the mirror system will unfold and slowly come to thermal equilibrium whilst the spacecraft reaches the final observing orbit. We have selected a halo L2 orbit that allows for mitigating the effects of eclipses that would upset the thermal equilibrium of the telescope system. A small orbit maintenance requirement for $\Delta V$ of ~1m/s per year is required to maintain this orbit.

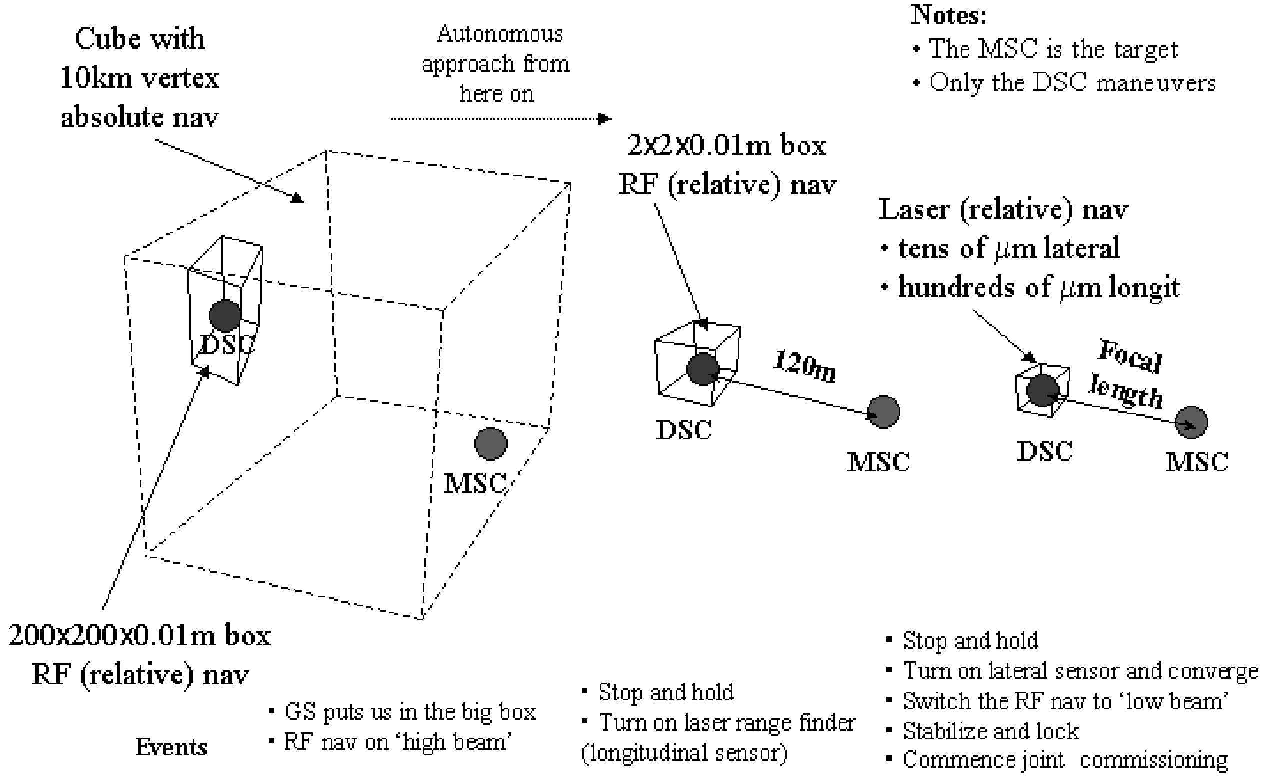


**Figure 3** Concept of the rendezvous strategy. Coarse rendezvous begins (left) with the ground segment locating the two spacecraft within 10km of each other via. Doppler ranging. Autonomous RF ranging between DSC and MSC allows 1 degree later accuracy and centimeter longitudinal accuracy, and allows navigation to less than 100m where the formation-flying subsystems can assume command.

Once it has been confirmed the MSC systems are deployed, the DSC can be launched. The baseline DSC payload has significantly lower mass, and this allows some freedom of launch window in order to match the MSC final orbit. The launch will be also from Kourou on a S-F, followed by correction maneuver and cruise to L2. Activated by normal ground station ranging, the orbits will be adjusted to within a few kilometers. Normal ranging with Doppler measurements at one ground station should allow an accuracy of ~10km at L2. Aspects of the subsequent rendezvous approach are illustrated graphically in Figure 3. At a distance of 10km, the RF system of DSC locates the MSC s with an accuracy ~200 metres. Using cold gas the DSC nears the MSC, with the RF ranging accuracy increasing inversely with the approach distance, where an accuracy of a few centimeters is achieved longitudinally and 1° laterally. when the separation is reduced to ~ 120m. At this stage the laser range finder and interferometer are activated and the final approach to the focal distance operating point is made, with an ultimate accuracy of ~10's μm lateral and 100's μm longitudinal being predicted.

.

In addition to the described baselines we investigated alternative launcher and orbit scenarios. For example the S-F launch from Kourou can be made into an intermediate High Earth Orbit (HEO) where the payload capapbility is ~3tonnes. Part of this payload margin can be used for an additional hydrazine propulsion unit. The $\Delta V$ needed to reach L2 is such that there is an additional spacecraft dry mass payload capacity of ~ 300kg, giving ample margin on sub-systems or (more usefully) additional mirror mass to increase the effective area. However the typical advantage to frontal area or optics mass margin is only ~15%, and the additional complexity of maneuvers does not warrant adoption at this stage.

Until S-F operations begin from Kourou it was felt prudent to examine also the options for a single launch. In this case the MSC and DSC can be mated in (for example) an Ariane V ECA fairing. Although the launch costs are expected to be significantly higher than for the dual S-F launch, there are cost advantages for simpler launch and cruise operations. In addition the mass capability to L2 is significantly higher, and the fairing envelope allows for a larger deployable area. Therefore the unit cost of effective area available at L2 is more comparable than the simple launch cost margins would dictate. Subject to discussions with other agencies or definitions on available budgets in the ESA Cosmic Visions programme, we again do not need to pursue this option yet, as the baseline mission meets our scientific goals at minimum cost.

## 6. OPERATIONS

Science observations occur under the conditions of the formation flying, and the deepest field observations with the WFI could comprise 1Ms pointings. These may be split between observing seasons, partly to satisfy observing constraints but also to allow for target identification and follow-up by NFI. Observations of targets in the local Universe, for example spectroscopy and timing measurements of binaries, bright AGN and clusters of galaxies, will be more typically 100ks in duration. Therefore the spacecraft sub-systems have been sized for slews occurring every 2 days, and with 100% margins for 5 years DSC operations.

Sky coverage is limited by the solar aspect angle and its affect on the thermal stability of the mirrors. Initial estimates of the allowable thermal gradients across the mirror system show a change of <1° is sustained with a change in solar illumination angle of ±15°, and therefore the solar observing constraints are comparable with *XMM-Newton*, and the complications of accommodating earthshine are even ***less*** severe than for *XMM-Newton*, as the Earth is aligned with the Sun and at greater distance of course. In principle a Great Circle 30° wide can be accessed at any time so that continuous viewing of any deep field can be allowed up to the maximum foreseen. Re-pointing within the Great Circle viewing zone has been sized to occur with slews of up to 90° in a duration of 1.5 hours, and this allows for a typical observing efficiency of 95%. The large halo elevation allows XEUS to cross different portions of the deep geotail environment, so that science operations may occasionally be interrupted by solar flare or magnetospheric plasma storms, but in general these will be less frequent than was the case for *CHANDRA* and *XMM-Newton* at lower altitudes

A final operational advantage of the L2 formation-flying scenario is that it is conceivable to replace the DSC. This could occur due to failure of the instruments or spacecraft sub-systems, or more likely as a result of instrumentation developments and/or changes in priorities in scientific investigations that may demand new classes of observations. The MSC consumables have been sized for 15 years, so at the termination of the DSC nominal 5 year life, a second DSC with new instrument package could be launched and the same rendezvous sequence carried out after the DSC1 was commanded to drift away to a safe distance.

## SUMMARY

We have demonstrated a fundamental redesign of the XEUS mission that dramatically simplifies the mission compared with the original ISS implementation. The adoption of new optics technology enables a low mass spacecraft that can be launched directly to a preferred L2 orbit. The proposed formation flying requirements are very modest and can be met with existing lab breadboard technology.

Following ESA's Horizons 2000+ programme, missions to be launched in the period 2015-2025 are being proposed through the Cosmic Visions selection. XEUS will be a very attractive candidate for one of the first missions of this era: it combines world-beating optics technology to implement an ambitious capability matching cosmological investigations in other wavebands. It offers significantly new capability in many other areas of astrophysics that would leverage Europe's strong role in high energy astrophysics, while utilizing the excellent instrument capabilities of the European community. It remains one of few well-studied missions that can be implemented on this timescale with existing technology and with a well-founded detailed cost estimate.

## ACKNOWLEDGEMENTS

The *Concurrent Design Facility* of ESTEC has carried out much of the design activities reported here, and at the time of writing the report is still being finalized. We express our appreciation to all the CDF team members who contributed to the study and we especially like to thank Lothar Gerlach, Thijs van der Laan, Kotska Wallace and Bogdan Udrea for their contributions.

## REFERENCES

1. A Parmar **Proc SPIE** this volume
2. M Bavdaz et al **Proc SPIE** this volume
3. A Parmar, G Hasinger and M Turner **Proc 34th Cospar Assembly**, p2368, 2004
4. J Bleeker, G Hasinger, H Inoue et al. **The Xeus Misison Summary** ESA SP1242, Noordwijk 2000
5. D H Lumb, **Proc SPIE** this volume
6. D Barret et al., **XEUS Instrument Working Group report**, ESA SP-1273, Noordwijk 2003