

Exoplanet explorer: Mechanical and thermal design of a single spacecraft nulling interferometer

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ABSTRACT

The mechanical and thermal design of a single spacecraft nulling interferometer is presented as part of a new study initiated by the European Space Agency to explore the design parameters and performance of an interferometric concept based on a single spacecraft with multiple sub-apertures. Launch constraints impose strict mass and volume limitations and necessitate secondary mirror deployment. Deployment concepts were evaluated through comparison and modal analysis. Due to thermal considerations, a preliminary thermal model was also developed. Results emphasize the necessity for a multi-layer sunshield and highly reflective materials to achieve the required temperature range. Temperature gradients across the payload and their effects on deformations have been identified as key factors for future investigation. This study provides the first integrated thermo-mechanical concept for the interferometer, laying the groundwork for further analysis and trade-off studies in later phases.

Keywords: Mechanical design, Thermal design, Nulling interferometry, Spacecraft, Exoplanets

1. INTRODUCTION

The detection and characterization of temperate rocky exoplanets constitute one of the most compelling scientific challenges in contemporary astrophysics. This has been identified as a top priority in ESA's Voyage 2050 long-term science program¹.

Among the various techniques developed for exoplanet studies, nulling interferometry stands out as a particularly powerful approach for this scientific motivation. It suppresses stellar light at levels of 10^{-5} to 10^{-6} while preserving the faint mid-infrared thermal emission from Earth-like planets in the habitable zones of nearby stars. Operating in the 4-18 μm wavelength range, such an instrument would provide access to key atmospheric constituents, including CO_2 , H_2O , O_3 , and CH_4 , enabling both habitability assessment and biosignature detection.

This space-based nulling interferometer is not a new idea; it dates to 1978, when Bracewell proposed the original idea². From the late 1990s to the mid-2000s, ESA pursued it through the Darwin program and NASA through the Terrestrial Planet Finder Interferometer (TPF-I). Both programs converged on architecture comprising four collector spacecraft feeding a central beam combiner. Despite laboratory demonstrations of 10^{-6} null depths and major advances in formation-flying technology, both programs were discontinued around 2007 due to programmatic constraints rather than technical limitations.

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Since 2017, the Large Interferometer for Exoplanets (LIFE) initiative has revived the concept, combining modern photonic technologies, updated exoplanet demographics, and improved mission-architecture studies^{3,4}. Formation flying, once a major uncertainty, has also progressed significantly, most notably with the launch of PROBA-3 in 2024⁵.

The required accuracy and formation flying configurations, for a nulling interferometer using more than two spacecraft, still remain a major development challenge. It is therefore interesting to also look at alternative concepts that could be feasible on the shorter term.

In late 2022, following the revival of nulling interferometry and the recommendations of the Voyage 2050 report, ESA issued a call for technical proposals titled "Nulling interferometry techniques with a single telescope". The call focuses on the feasibility of nulling interferometry without formation flying, using a single spacecraft that fits within an Ariane 6 fairing. Improved understanding of exoplanet occurrence statistics now allows quantification of the scientific return of such a constrained architecture and determination of whether it is sufficient to justify a dedicated mission.

The consortium (TU Delft, KU Leuven, CSL/ULiège & AMOS) has performed four sequential tasks covering requirements consolidation, technology survey, and architecture trade-offs, preliminary design of selected concepts, and technology road mapping with cost estimation⁶⁻⁸.

Six distinct spacecraft configurations have been analyzed, representing different approaches to achieving the required interferometric baselines:

- Linear configuration configurations (LIN8, LIN16, LIN32): Four telescopes arranged in a quasi-linear array with nulling baselines of 8, 16, and 32 meters, respectively
- X-array configurations (X8, X16, X32): Four telescopes arranged in an X-shaped array with corresponding nulling baselines

Each configuration presents distinct advantages. The trade-off has been made between scientific performance (exoplanet yield, inner working angle, sensitivity), technical complexity (deployability, thermal control, active correction requirements), and feasibility (mass budget, structural stability, integration complexity)⁸.

For most configurations, the large interferometric baselines necessitate deployable structures capable of maintaining precise alignment to achieve interferometric baselines ranging from 8 to 32 meters while respecting the volume constraints of the Ariane 6 long fairing. This includes trade-offs between deployment mechanisms (telescopic booms, articulated systems, deployable trusses), structural dynamics analysis, and thermal stability considerations. This paper presents the resulting mechanical and thermal architecture and highlights the main challenges encountered and those to be faced in the design of such a spacecraft to achieve the previously outlined scientific objectives.

A companion paper is also available at this conference, focusing on the optical design⁹.

2. FIRST APPROACH FOR SYSTEM ENGINEERING AND TECHNICAL BUDGETS

Orbit Selection and impact on mass and volume budget

The best orbit for a nulling interferometer is the Sun-Earth Lagrange Point (SE-L2). It provides a stable thermal environment, with the Sun and Earth on one side and relatively low infrared flux from Earth due to distance. This also maximizes available sky coverage. Attitude disturbances and orbit maintenance requirements are also lower than for orbits closer to Earth. For these reasons, it is also the orbit selected for the James Webb Telescope (JWST) and the Gaia mission. For thermal stability, a halo orbit around L2, like for the JWST, is preferred because it provides the best thermal stability. The launch mass of the Ariane 6, more specifically the Ariane 64 configuration, to a highly elliptical orbit with L2 at its apogee, is 8000 kg¹⁰. This still requires final orbit injection by the spacecraft, which must be included in the delta-V budget.

The long fairing of Ariane 6, as specified in the launch manual, provides the maximum available stowed volume, as shown in Figure 2-1. Any nulling interferometer configuration larger than this stowed volume requires deploying the telescopes, the secondary mirror assembly for each unit, and a sunshield.

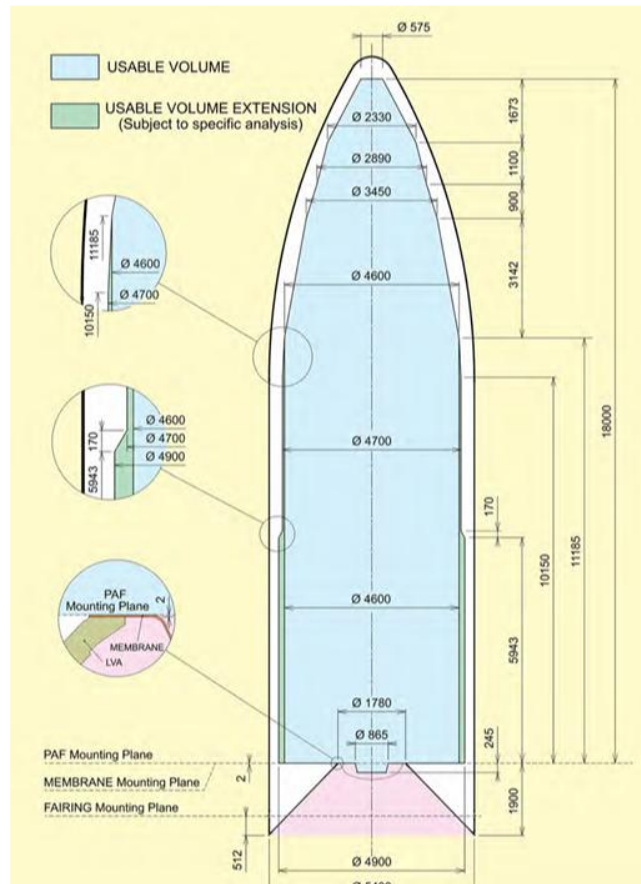


Figure 2-1: Ariane 6 fairing dimensions

Sunshield area requirement

From a mission perspective, full-sky coverage is preferred, even if the zodiacal disk can limit the detectability of exoplanets (absorption and scattering). By nature of the SE-L2 orbit, a 360° rotation in the ecliptic plane is performed within a year. To point towards a given star of interest, it is best to wait until the star approaches its minimum angle from the Sun-Earth vector. For a given observation time to detect or characterize exoplanet(s), the nulling interferometer must be continuously rotated in the ecliptic plane relative to the Sun-Earth vector to maintain pointing. The declination requires a single pitch of the nulling interferometer with respect to the ecliptic plane. For full coverage, the instrument needs to be pointed at $\pm 90^\circ$.

The Sun shield should then be able to cover this without introducing significant thermal gradients or shifts. If this is not possible, a compromise between the feasibility of the (thermal) design and maximum declination, at the expense of lower yield, shall be found. For the first design iteration, full coverage is chosen as the design input to provide a common basis for comparison of the sunshield size relative to the deployed size.

Only volume and mass budgets are currently treated in this section, as they are considered the driving factors for this project. The data produced by the nulling interferometer are relatively limited because they consist of a small number of measurements over time and lack spatial information, compared to, for example, a classical imaging telescope such as the JWST. Also, the electrical power requirements are assumed to be comparable to those of JWST, or even lower, due to technological progress. Therefore, no electrical or data budgets have been provided.

the array is at 90° declination (when the Sun vector is parallel to the array plane). A small savings can be achieved if the maximum declination is reduced to, for example, 70°. However, this only results in a minor reduction in mass and stowed volume and is not considered currently. This Sun Shield design, based on the LIN16 configuration (Figure 4-1) is resized for all configurations to match the physical dimensions of the array, maintaining the same distances relative to the telescopes' sides.

The JWST requires a Δv of 2.4m/s annually over its 10-year mission (see ¹¹) out of a total of 97m/s, using approximately 301 kg of hydrazine fuel and oxidizer, and having a dry propulsion system mass of 74 kg. Adopting a conservative approach (including some dry mass for propellant tank scaling), the JWST station-keeping costs about 9.4 kg of propellant per year, with 280 kg allocated for orbit injections, end-of-life, and contingencies.

Designation (Configuration Tree)	Configuration Item	Mass	Reference or Rationale
PL-Tx-TLA-M1	Primary mirror	4 x 200 kg	Based on Nancy Grace Roman ST Based on scaling w.r.t. JWST
PL-Tx-TLA-M2	Secondary mirror	4 x 20 kg	
PL-Tx-TLA-TSST	Telescope support structure	4 x 50 kg	
PL-Tx-TO	Transfer optics	4 x 50 kg	
PL-Tx-DL	Delay line	4 x 50 kg	
PL-Tx-LOSCA	Line-of-sight-correction-assembly	4 x 50 kg	
PL-MSST-BOOMS	(Deployable) booms	250 kg	
PL-MSST-HDRM	Hold down and release mechanism	100 kg	
PL-CU-DETA	Spectro assembly Beam combiner assembly Spatial filtering assembly Wavelength band separator	1250 kg	JWST integrated science instrument module = 1245 kg
PL-CU-SPA			
PL-CU-BCA			
PL-CU-SPFA			
PL-CU-WLBS			
BUS-TCS-SS	Sun Shield	1500 kg	JWST = 678 kg, needs to be significantly larger depending on array size. JWST = 375 kg, most likely need slightly larger due to increased solar pressure depending on sun shield size. JWST = 122 kg, most likely slightly larger due to increase solar pressure depending on sun shield size Conservative approach based on JWST (1516 kg dry mass + 79 cryocooler - 74 kg dry propulsion -122 ADCS) and increases slightly to account for small scaling effects.
BUS-PRS	Propulsion	400 kg	
BUS-ADCS	Attitude determination and control	150 kg	
BUS-Other	EPS, CDHS, COMMS, TCS (active part), STS and MechS (s/c bus), wiring	1500 kg	
Total allocated		6830 kg	From Ariane 6 Launch capability This is approximately 15% of launch budget,
Launch budget		8000 kg	
Contingency		1170 kg	

Table 2-2: Preliminary mass budget

The delta-V requirement is primarily driven by the sun shield's size due to solar radiation pressure. The station-keeping delta-V is roughly proportional to the ratio of the sun shield area to the total spacecraft mass,

$$\Delta v_{SK} \propto \frac{A_{shield}}{m_{sc}}$$

where:

- Δv_{SK} is the annual delta-V required for station-keeping,
- A_{shield} is the effective cross-sectional area of the sun shield (~300 m² for JWST),
- m_{sc} is the total mass of the spacecraft (~6161 kg for JWST).

Assuming a total mass of 8000 kg for the nulling interferometer spacecraft and a linear relationship between mass ratio and shield area ratio, a first estimate of the required propulsion leads to :

$$m[kg] = \frac{m_{sc}}{m_{JWST}} \cdot (280 + 9.4 \cdot \frac{A_{shield}}{A_{shield,JWST}} \cdot \frac{m_{JWST}}{m_{sc}}) = 364 + 9.4 \cdot \frac{A_{shield}[m^2]}{300}.$$

The worst-case cross-sectional shield area, entering the equation above as A_{shield} , will be the projected area of the Sun shield when the sun vector is perpendicular to the payload array ("top area" from Table 2-1). For the mass of the Sun shield itself, also the sides need to be considered ("total area" from Table 2-1). The sun shield mass is assumed to scale linearly with its size, and the ADCS mass is assumed to scale linearly with the (worst-case) projected area. This leads to mass estimates for the configuration-dependent items provided in Table 2-3.

Configuration Item	Budget	Lin8	Lin16	Lin32	X8	X16	X32
PL-MSST-Booms	250 kg	26 kg	138 kg	930 kg	58 kg	251kg	~1500 kg
BUS-TCS-SS	1500 kg	931 kg	1403 kg	2347 kg	1022 kg	1729 kg	3577 kg
BUS-PRS	400 kg	368 kg	371 kg	377 kg	369 kg	375 kg	393 kg
BUS-ADCS	150 kg	57 kg	93 kg	163 kg	71 kg	146 kg	374 kg
Total of items above	2300 kg	1383 kg	2005 kg	3816 kg	1520 kg	2501 kg	5843 kg
Difference w.r.t. budget	0 kg	-917 kg	-295 kg	+1516 kg	-780 kg	+201 kg	+3543 kg
Total estimated (excl. contingency)	6830 kg	5913 kg	6535 kg	8346 kg	6050 kg	7031 kg	10373 kg

Table 2-3: Current Best Estimate (CBE) for mass of configuration-dependent items

The row with differences in Table 2-3 compared to the general mass budget provided in Table 2-2 gives insight into the feasibility of the design. It is already evident that the X32 configuration largely exceeds the budget, including the contingency.

Other major design driver: ultra stability

The optical alignment of the array with respect to its target star, as well as the internal alignment of the optical paths, provides a high-precision engineering challenge. The optical path differences must remain stable to the nanometer level for effective nulling. Micro-vibrations will cause line-of-sight jitter across the instrument and introduce internal path errors due to deformations of the structural elements. Thermal changes within the telescopes and support structures will also cause deformations, although they are expected to be relatively slow compared to the micro-vibration response. A full structural, thermal, and optical (STOP) model analysis is required to get insight into the exact quantitative errors, their effects, and required mitigation and compensation techniques. This is an extensive future study to be performed. Below is a proposed approach to address the expected needs qualitatively.

- Attitude Determination and Control
 - Achieving an extremely accurate pointing to the target, while achieving the required rotation around the boresight axis (to rotate the modulation map during observation) will limit the need for attitude control during observation of a single star system. The rotation also provides some gyroscopic stiffness against disturbance torques, but the rotational speed will be driven by scientific objectives.
 - Use a spatial (centroiding) sensor within the optical path of the nulling interferometer as the most accurate attitude determination sensing, after having achieved relatively good determination and pointing using star trackers. This would essentially be a very high-accuracy single-star tracker using the full aperture of the primary mirrors of the telescope (which is significantly larger than that of star trackers). This can be done, for example, in the optical domain to avoid interference with observations in the infrared domain. The relative misalignment between the individual telescopes must be actively corrected at the primary or secondary mirror of each telescope.
 - Make use of very precise and stable actuators. Instead of reaction wheels, the eventual trade-off may favor very small thrusters. LISA Pathfinder and GAIA used cold-gas micro-Newton thrusters for ultra-stable drag and attitude control, with in-orbit results of approximately $0.5 \mu N$ (1σ) noise level¹². Field-Emission Electric Propulsion (FEEP) or colloid (electrospray) thrusters have the potential to achieve even lower noise levels. An in-orbit experiment with colloid thrusters on LISA shows four times lower noise levels¹³. An additional advantage of using thrusters over a reaction wheel system (such as used

on JWST) is that the noise is lower at lower thrust levels, where for reaction wheels it depends on the rotation speed, which is based on accumulated momentum and thus not directly controlled. Given the previously described stable pointing observed during the observation, this is an additional trade-off argument for using micro-thrusters over a reaction-wheel-based system.

- Active Optical Path Control and Metrology
 - Use slow-speed large-range actuators (piston, x/y and/or tip/tilt) to correct for initial deployment errors and slow varying OPD drifts (e.g., due to thermal expansion)
 - Use fast-speed short-range actuators (piston, x/y and/or tip/tilt) to correct fast varying OPD errors in real time (e.g., due to vibration).
 - Add a counter mass and actuator to provide equal and opposite forces to the actuated mirrors to suppress vibration (active noise canceling).
 - Implement heterodyne laser metrology, in combination with, for example, retroreflectors at the telescopes, to detect OPDs between telescopes with sub-nanometer precision at high rates.
 - Use a fringe tracker to detect the smallest errors in OPD required for effective nulling.
 - Use high-speed, closed-loop control with a bandwidth of hundreds of Hz.
- Thermal Stability and Management
 - Thermally isolate the whole payload, including the booms, telescopes, and beam combiner, from the spacecraft bus and the radiative load from the Sun.
 - Analyze the thermal gradients and their thermo-mechanical impact, and identify where they need to be actively controlled to stay below required levels.
 - Implement active thermal control where necessary to minimize thermal gradients that cause deformations. The implementation must be vibration-free. If the difference is only a few Kelvin, local heating may be an effective solution.
 - Make use of ultra-low expansion materials to minimize thermal expansion.
 - Include temperature sensors along the booms and optical surfaces, feeding into control algorithms (mechanical and/or thermal).
- Limiting and Isolating Vibration
 - Select spacecraft (bus) components with minimal jitter.
 - Use a two-stage passive damper for isolating main micro-vibration sources in the spacecraft, such as the reaction wheels and a potential active cryo-cooler. The first stage is near the source component, and the second stage is between the spacecraft bus and payload.
 - Use an active isolation platform between the spacecraft bus and payload, if needed, to isolate the spacecraft bus beyond what is passively possible.
 - Use a constant speed of rotation around the line of sight (no rotational accelerations) during the observation period for the rotation of the array, both in-plane for rotating the modulation map, as well as in the ecliptic to stay aligned with the target while moving in orbit.
 - Design telescopic booms with mechanical locks or pre-loaded structures that eliminate micro-slips and hysteresis.
 - Introduce vibration-isolating interfaces at selected structural interfaces to dampen the propagation of small dynamic loads (micro-vibrations).

3. MECHANICAL DEPLOYMENT

Main support structure deployment

Given the limited space available during launch and to achieve sufficient nulling baseline, deployment of the different telescope units first, and within each telescope (secondary mirror deployment) is of particular value here.

Based on the current optical baseline of four telescopes and the size of the primary mirror considered⁹, six different configurations have been proposed for the deployment of the units using a Main Support Structure (MSST). The first three are purely linear, with the four units aligned along a single axis with the beam combiner located in the middle of the spacecraft.

Such large deployable structures for space applications have already been used in past missions, with the most notable being the JWST, a recent example demonstrating that telescopic deployment is flight-proven, although the application to a Sun Shield poses different requirements than for deployment of an optical payload over these lengths.

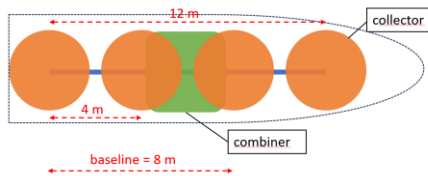


Figure 3-1: Lin8 Configuration - Non-Deployable linear array with 8m nulling baseline

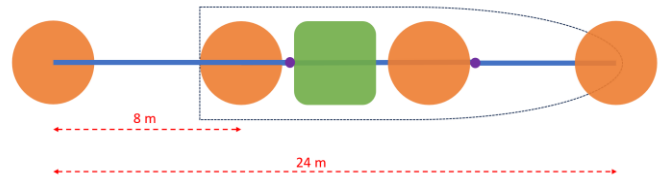


Figure 3-2: Lin16 - Linear array with 16m nulling baseline

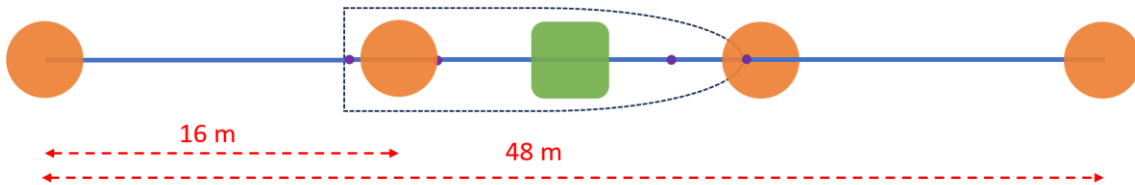


Figure 3-3: Lin 32 - Linear array with 32m nulling baseline

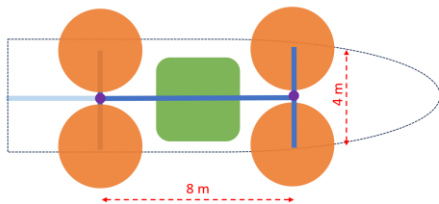


Figure 3-4: X8 - X-Array with 8m nulling baseline

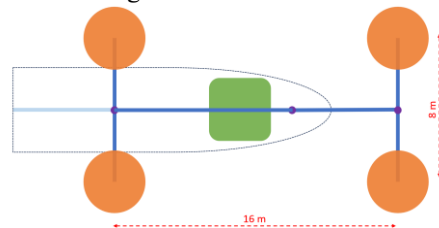


Figure 3-5: X-Array with 16m nulling baseline

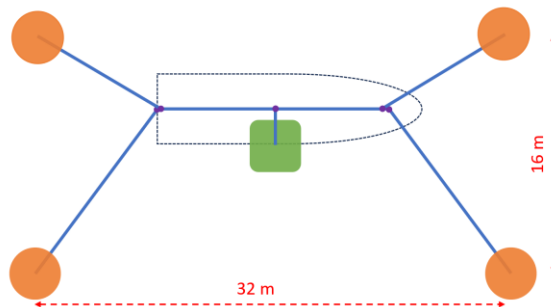


Figure 3-6: X32 - X-Array with 32m baseline

Figure 3-1 to Figure 3-6 show the six different considered configuration dimensions, including a symbolic representation of the four telescopes (orange), articulated arms (blue) and the beam combiner (green) and the Ariane fairing in the background. Note that the baseline is the center-to-center distance between the nulling pairs, which is the long side of the X-array and the first-to-third and second-to-fourth telescope for the linear array. These simple schematics show an articulated structure, but there are several options for deployment of large structures.

Past studies on large deployment structures for astrophysics missions provide a useful overview of the different deployment technologies that could be used for such a spacecraft, and most notably the deployment of the main structure holding the telescope units¹⁴. A trade-off among the available methods was conducted to qualitatively compare them and provide insight into which technologies are worth considering in future developments.

Deployment Structure	Cross-sectional Area w.r.t. Fixed Reference	Deployed-to-Stowed Volume Ratio	Bending Stiffness	Risk
Telescopic booms	Slightly larger	0.1-0.5, depending on number of segments	Approaches rigid structure	High TRL, small deployment risk
Articulated booms	Almost equal	Limited gain	Approaches rigid structure	High TRL, lowest deployment risk
Deployable truss	Relatively high in stowed configuration	0.05-0.3, constraint by number of hinges and clamps	Needs to be significantly larger to approach rigid structure	High TRL, deployment or stiffness risk increases with number of hinges and clamps
Shape memory composite booms	Almost equal	Very high	Poor due to conflicting material properties	Low TRL, risk of inaccurate deployment
Inflatable booms	Almost equal	Very high	Poor due to limited thickness	Low TRL, risk of inaccurate deployment and/or changing stiffness over time
Coilable booms	Almost equal	Very high	Poor due to inherent flexibility	Medium TRL, risk of inaccurate deployment

Table 3-1: Trade-off table for deployment types for the payload main support structure

Of the different technologies proposed, only the telescopic and articulated booms are worth considering, as the other deployment types have significantly lower bending stiffness and a greater number of articulations than these two solutions.

The Lin8 configuration (Figure 3-1) is the only one that fits in the Ariane 6 fairing and does not require a deployable MSST. For the linear configurations Lin16 and Lin32 (Figure 3-2 , Figure 3-3) and for the long axis of the X-array, both linear and rotational deployment are options. For the second dimension of the X-array configurations, rotational capability is required. For purely linear deployment, the significantly better packing ratio is an advantage, and considering good clamping, there should not be a fundamental difference in bending stiffness between articulated and telescopic booms. This is therefore considered as preferred option for the MSST in this study.

Technologies developed by Northrop Grumman's or COMAT-CNES (see ¹⁵ and ¹⁶ respectively) on deployable boom design and tests demonstrate the feasibility of using telescopic structures for high-precision space missions. They offer long deployment length and compact storage, highlighting the potential to achieve long baselines with high packaging efficiency, a critical feature for interferometric configurations. For the nulling interferometer, which requires both tight alignment tolerances and deployment within the constrained volume of the Ariane 6 fairing, these examples show that a telescopic boom can be tailored to the needs.

Structural Dynamics and Stability

To ensure optimal nulling interferometry performance, each spacecraft topology must minimize dynamic disturbances that introduce optical path length errors. As part of the early requirements definition, a target of 2 Hz has been set for the first eigenmode of the main support structure. Also, the frequency response to micro vibrations should be within the limits of the optical error budgets, as even with active correction measures to counteract these vibrations, there will be high-frequency components that cannot be actively compensated.

For the preliminary sizing and modal analysis of the main support structure, two approaches are taken. For each of the six investigated configurations, a simplified design is constructed in which the first eigenfrequency is exactly 2 Hz, a fixed wall thickness of 1 mm is chosen, and the beam diameters are adjusted until the desired stiffness is reached. A second design is made to maximize the first eigenfrequency within the available stowed volume. The latter will thus have the best possible stiffness, despite the higher mass and volume of the MSST. Based on current insights, telescopic booms made of pitch-based CFRP appear to be the best design solution.

Using dimensional results obtained for the individual tube sections, 3D models are created with each tube in the telescopic mechanism being modeled as a hollow cylinder with specific dimensions. For this study, a 1 mm wall thickness is chosen as the baseline, and the outer tube diameter is sized accordingly. The design for the 6 different configurations can be seen in Figure 3-7.

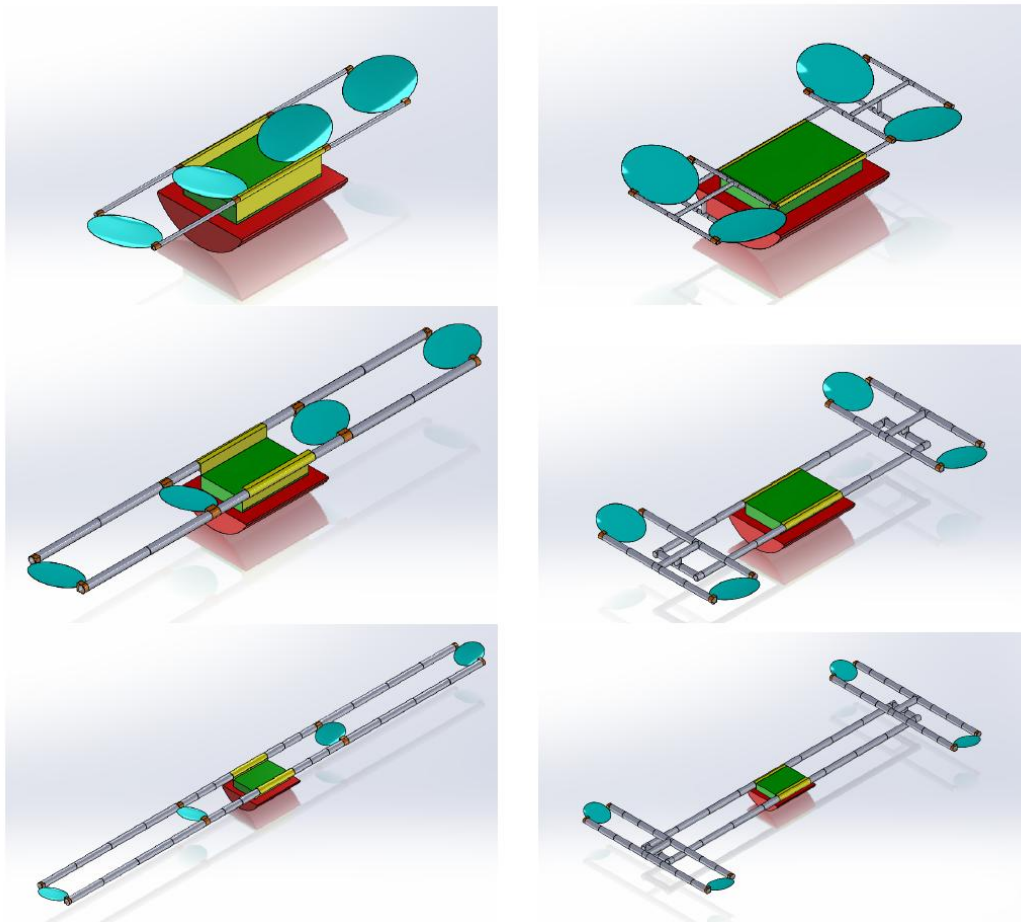


Figure 3-7: The Lin8 and the deployed X8, Lin16, X16, Lin32 and X32 configurations.

In Table 3-2, the key results of the design and analysis of all configurations are provided. The eigenmodes are calculated with a simulated mass of 750 kg per individual telescope and a fixed boundary condition at the interface with the central unit. The stiffness of the telescopes, provided by the telescope structure and clamping, is modeled as approximately 100 times that of the boom material.

This simplification was necessary because no telescope structure or clamping model was available at the time. The focus remains on the MSST itself. The deflection under a hypothetical 1N load at the tips of the array is also provided for all but the modified Lin32 configurations. This does not represent any true load, so it is only used to indicate the order-of-magnitude difference between the various configurations. The mass represents the mass of the tubes, excluding the clamps, deployment mechanisms, and other parts of the main support structure.

For Lin32 and X32, the maximum volume version already had a fundamental frequency below 2 Hz. To fix this issue for the Lin32, the wall thickness is increased, and the booms are made elliptical with the major axis of 862 mm in the plan of the first eigen mode and minor 620 mm, just like the circular maximized volume. This leads to a 2 Hz compliant design at the cost of relatively high mass. For the X32 configuration, a quick evaluation showed that the mass would exceed 1500 kg, and the design would become even more convoluted. Therefore, this configuration was not analyzed further.

Topology	Fundamental frequency [Hz]	Deflection under 1N load [μ m]	Mass [kg]
Lin8 (baseline)	2.1	9.1	26.3
Lin8 (max.vol.)	13.4	0.2	102.2
Lin16 (baseline)	2.1	7.6	137.6
Lin16 (max.vol.)	3.6	2.7	204.5
Lin32 (baseline, modified)	2.0	N/A	929.7
Lin32 (max.vol.)	1.1	24.1	401.9
X8 (baseline)	2.0	4.2	57.6
X8 (max.vol.)	16.0	0.1	326.0
X16 (baseline)	2.0	3.5	250.5
X16 (max.vol.)	4.4	0.7	536.1
X32 (max.vol.)	1.3	7.6	940.9

Table 3-2: Key results for the various MSST designs

The preliminary designs show that all configurations except the X-array of 32 m are feasible. The six configurations studied are discrete examples that can easily be modified to achieve nulling baselines in between those provided. For example, a 20-24 m linear baseline is expected to be feasible within the current mass budget. However, it is recommended to first perform a more detailed system study before consolidating on the exact dimensions.

Secondary mirror deployment

For a nulling interferometer, secondary mirror positioning is essential to reduce optical path length errors and their impact on overall observation performance. In all proposed configurations for the spacecraft, whether the smallest non-deployable linear (Lin8) or the X-array with a 32m baseline (X32), deployment of the secondary mirror is required to ensure the complete spacecraft fits within the Ariane 6 fairing. Deploying the secondary mirror assembly, while necessary for space constraints, can degrade the positioning accuracy of the secondary mirror relative to the primary mirror and the other optical elements.

For these reasons, compensation systems are needed to ensure sufficient positioning accuracy is achieved throughout telescope operations, from initial deployment to normal operations. While such compensation systems exist and are in use in space applications, their capabilities are limited. They indeed require that the initial positioning accuracy be sufficiently good to allow for compensation and that they provide sufficient stability during the satellite operations.

This type of deployment has already been successfully performed in past missions, with the most notorious example being the M2 deployment of the JWST, for which the secondary mirror is held in position by a composite M55J tripod, articulated via four hinges to minimize stowage volume. The use of composite materials is motivated here by the need for a very low, stable CTE material that maintains thermal stability while providing a high stiffness-to-weight ratio.

The tripod design provides reasonable stiffness for the secondary mirror assembly, but due to its size, additional damping systems were required to ensure proper vibration isolation.

In the current baseline for the telescopes, the secondary mirror is located at approximately 4 m from the primary mirror, which might allow for a stiffer assembly. Here, the secondary mirror is not located directly in front of the primary mirror but is placed slightly to the side to avoid obstructing the optical path. Therefore, the deployment system must avoid the full area in front of the mirror.

To handle the secondary mirror deployment, three possible designs are proposed here, based on state-of-the-art and current technologies that could be used in the near future if TRL improves. These designs are briefly described below and in Figure 3-8:

- Telescopic booms deployment: this method involves the use of telescopic booms, as already presented for the MSST.
- Single hinge deployment: The second one is based on a single hinge mechanism where the mirror is held by a single beam and articulated at its base.
- Tripod deployment: The last one consists of three articulated beams with multiple hinges that form a tripod once deployed.

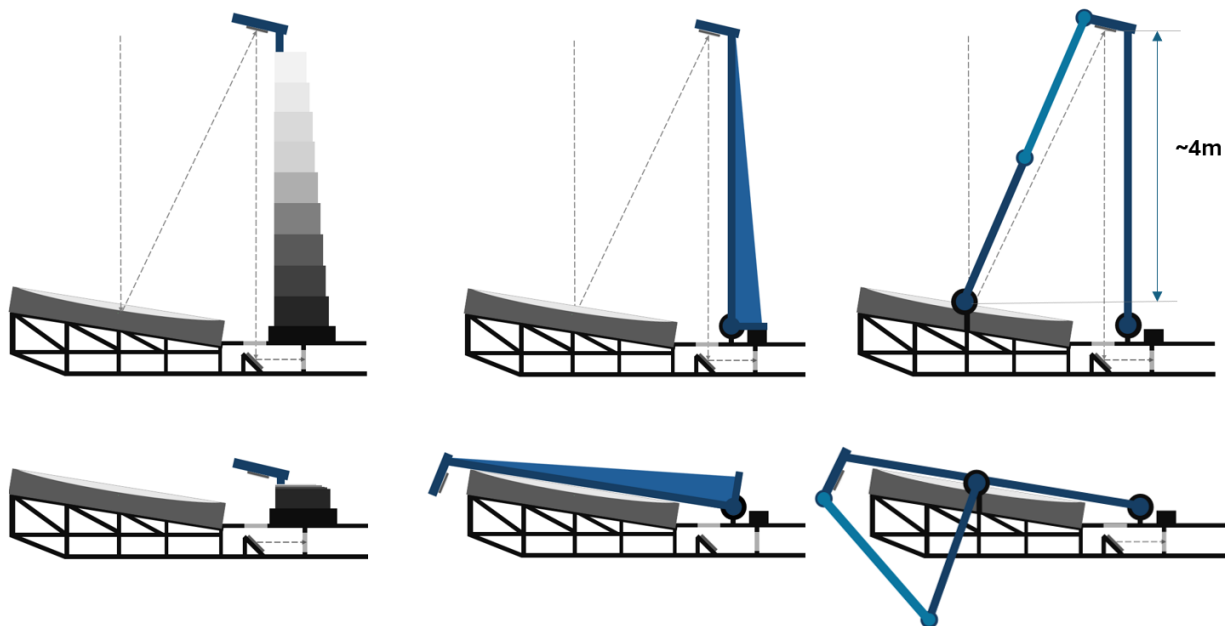


Figure 3-8: Deployable support system alternatives. From left to right: telescopic boom deployment, single hinge design, multiple hinge (or tripod) design.

Structural dynamics and stability

To perform a preliminary analysis of the mechanical behavior of the proposed alternatives, they are simplified to mechanical beam assemblies scaled to the current mirror accommodation baseline. Since the telescopic boom and single-hinge deployments are comparable in their final deployed shapes, they are modeled the same way and designated as the "single beam" configuration. Modal analyses are performed on the two principles to verify that, in both cases, the first eigenfrequencies are above 7 Hz and to compare the relative stiffness of the two structures.

This analysis is not aimed at prediction but rather to provide a proper understanding of each design's capabilities in terms of dynamic stability and overall stiffness relative to the preliminary requirements. A static analysis is also performed under a uniform 1 g acceleration in all three directions to provide a first estimate of secondary movement under load and its susceptibility to uniform accelerations that could arise from the spacecraft's rotating movements.

To make a fair comparison between the two configurations, they are both designed to have the same mass. To do so, the beam thickness and diameter are adapted. While other factors are at play here, such as the presence of hinges, it is assumed to be sufficient for a preliminary analysis.

For the single-beam design, the main beam is modeled as a single 200 mm-diameter hollow tube with a thickness of 2 mm. It is fixed at the bottom, and a lumped mass representing the secondary mirror assembly is defined at the other extremity (20 kg). The mass considered is directly taken from the mass budget in Table 2-2.

The multiple-hinge configuration (simplified as the "tripod" configuration) is modeled with three beams, fixed at the bottom, with a lumped mass of 20 kg at the top. In this case, the beams are modeled as 90 mm-diameter wide hollow tubes with a thickness of 1.5 mm.

In both configurations, the relative position of the secondary mirror assembly with regard to the primary mirror is the same. The attachment point of the single-beam design is chosen at the edge of the primary mirror to avoid having the beam in the optical path. In the tripod design, the beams are connected to the primary mirror assembly at three points around the mirror to ensure that, once deployed, neither the side beams nor the main beam obstructs the optical path, as can be seen in Figure 3-9. The material assumed for the composition of the beams is the same as that used for the main structure assembly, as here, too, there is a need for good thermo-elastic stability and mechanical properties.

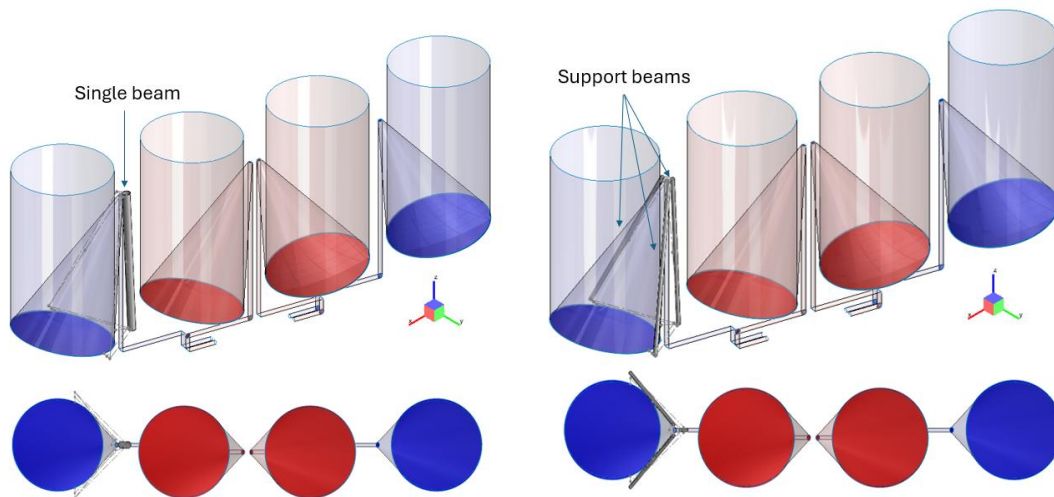


Figure 3-9: Top left : Single beam configuration layout - Bottom Left: Beam position out of the optical path Top Right : Tripod configuration layout - Bottom Right: Beams position out of the optical path

As expected, at an equivalent mass, the single-beam solution is below the specified minimum natural frequency of 7 Hz. Additional optimization efforts could increase the system's stiffness to meet the specification. One such optimization could be the addition of lateral stiffness to cut as much of the first mode as possible. The tripod design is much stiffer than the single-beam design, with its first frequency at about 50 Hz. But it is important to note that this design would require additional elements (hinges, locking at the centers of the beams, etc.), which are difficult to quantify at this early stage of the design.

When exposed to a uniform lateral 1 g acceleration in all three directions, the two configurations yield the absolute displacement of the secondary mirror assembly, as shown in Table 5.4. The single-beam configuration is shown to be much more flexible, with a lateral displacement that is one order of magnitude higher than that of the tripod configuration.

	Displacement under 1G [μm]		
	X	Y	Z
Tripod Configuration	125	205	45
Single Beam Configuration	5662	5656	272

Table 3-3 : Secondary mirror assembly displacement under 1 g load

With its larger footprint, the tripod design is by far the most stable of the two solutions presented here. Indeed, for the same mass, the eigenfrequency is 7 times higher than with the single-beam design. While it is below the 7 Hz preliminary specification, optimization can improve the first lateral mode in the single-beam configuration.

For these reasons, and at this stage of development, both solutions can still be investigated for the secondary mirror deployment system. A trade-off of the different solutions is provided in Table 3-4 to give a clear view of the advantages and drawbacks of each solution presented.

Deployment method	Telescopic Boom	Single hinge	Tripod
Stiffness (Dynamic stability)	Low: at equivalent mass, the first mode frequencies are much lower than tripod configuration. Movement underloads are comparatively high.	Low: at equivalent mass, the first modes frequencies are much lower than tripod configuration. Movement underloads are comparatively high.	High: first mode is far higher than other two solutions. Movement under load is also lower.
Thermal stability	Moderate: the secondary mirror will travel along a predictable path. Use of low CTE material can help reduce greatly this problem.	Moderate: the secondary mirror will travel along a predictable path. Use of low CTE material can help reduce greatly this problem.	Lowest of the three solutions: thermal uniformity degradation can cause tilt in the secondary mirror
Stowed volume	Low	Limited possibility of stowed optimization	Low
Number of actioners	Moderate	Low	Moderate
TRL Level	Medium TRL: while not in use for this exact application, it is in use in space applications	High TRL: widely used in flight conditions	High TRL: widely used in flight conditions
Risk level	Moderate: relative to the other designs proposed	Low: Reduced number of actuators	High: multiple hinges and locking mechanism required. Higher number of potential single points of failures.

Table 3-4: Trade-off table for deployment type for secondary mirror assembly support structure

4. PRELIMINARY THERMAL MODELING

Thermal control of the spacecraft plays a key role in meeting the required performance requirements, as the payload (primarily the telescope units and beam combiner) must be kept at very low and uniform temperatures to perform as intended. Thermoelastic deformations of the structure due to temperature variations can significantly affect measurement quality. At the main-structure level, it can change the relative positions of the different telescopes. The same can be said for the secondary mirror, since, once deployed, changes in its position can cause significant variations in the optical patch if, for example, the structure supporting it is subjected to thermal deformations that are not properly predicted.

To study these phenomena and to obtain a preliminary estimate of the spacecraft's thermal behavior, a thermal model is developed. Early analysis using this model aims to provide a better understanding of the design drivers that will need to be considered for future development activities. Mainly, this analysis will aim to better define the following:

- Temperatures achievable with the current architecture of the spacecraft
- Initial power budget adequacy with regard to targeted thermal performances
- Thermal uniformity of the spacecraft
- Amplitude of movements to be expected due to thermoelastic deformations

Model overview

Because low temperatures are required on the payload side, the satellite platform is placed on the hot side of the sun shield to minimize radiative heating on the cold side. In this preliminary model, as shown in Figure 6.1, the selected configuration is Lin16. As for the M2 deployment system, the tripod solution is represented here. At this stage, it can be assumed that the single-beam deployment behavior can be derived from this configuration, as it is close to a tripod configuration with only the central beam present.

Each subsystem geometry is modeled with simple shapes; the spacecraft platform, beam combiner, and M2 mirror assembly are simplified to boxes. The telescope's main structure and secondary mirror deployment are modeled as cylinders. In contrast, the main mirror and each telescope's structure are modeled as trapezoidal boxes with the primary mirror represented as a flat disc. Due to its large size, the primary mirror is geometrically represented; the other optical elements are not. The solar arrays are not represented either but can be assumed to be placed on the platform in future work or on a dedicated structure. As they will be placed on the hot side of the spacecraft, their impact on the overall thermal behavior of the payload on the cold side is assumed to be negligible at this stage.

The sunshield is designed to enclose the entire payload once deployed and protect it from the sun on all sides. For this preliminary model, the geometry is simplified, with each face modeled as a single surface. The multiple layers required to mitigate heat transfer from the hot to the cold side are accounted for in the mathematical thermal model. Still, they will not appear geometrically in the model.

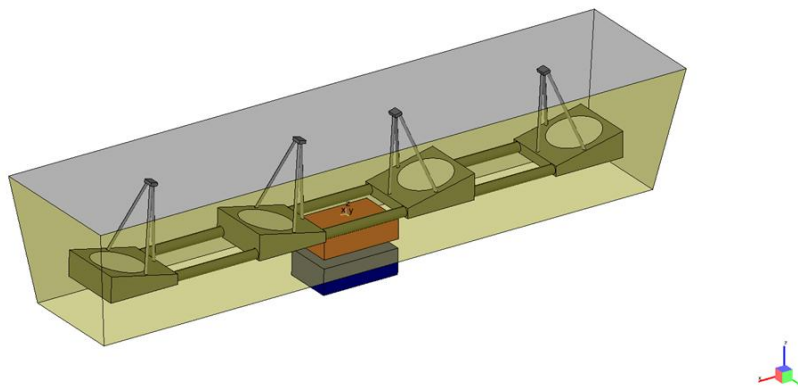


Figure 4-1: Linear configuration thermal model overview

Figure 4-2 provides a breakdown of how the spacecraft is modeled, with the platform and the subsystems on the payload side represented as single nodes, since the detailed listing of each subsystem element is not yet defined. Since the thermal behavior is strongly linked to the illumination direction, the sun shield is modeled with separate nodes for each direction and one node per face to distinguish radiative links from internal faces toward the payload from the solar heat flux on the outer faces.

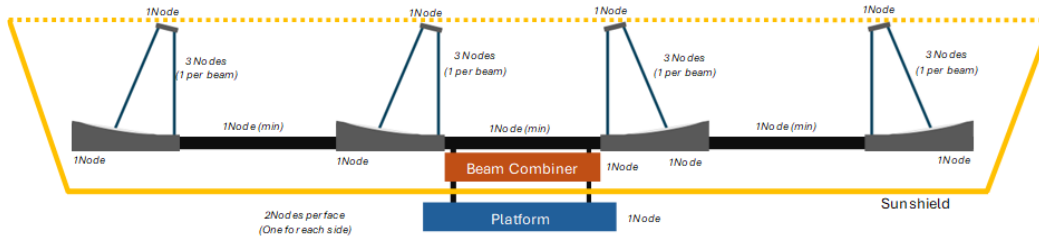


Figure 4-2: Nodal breakdown of the Lin16 thermal model

Preliminary selection of thermo-optical properties

The thermo-optical properties of the spacecraft surfaces play a crucial role in its thermal performance, particularly in ensuring the spacecraft can be sufficiently cooled. As such, the thermal control strategy focuses on dissipating as much heat as possible radiatively on the cold side while reflecting as much heat as possible on the hot side.

The initial choice for the thermo-optical properties of the different subsystems is described hereunder:

- **Sunshield:** on the sun-facing side, a high emissivity-to-absorbance ratio is needed to ensure good radiative dissipation while maintaining a moderate incoming solar heat flux on the hot side. On the cold side, the main objective is to avoid radiative dissipation toward the payload, so a low-emissivity coating is preferred. As such, aluminized Kapton with silicon oxide on the sun-facing side is assumed.
- **Platform:** as it is located on the hot side, the driver for the thermo-optical surface is the same as for the sunshield. As such, the same coating is assumed for now on the platform.
- **Payload (excluding optical elements):** Except for the optical elements, all external surfaces of the payload shall be highly emissive to passively cool the elements down. This encompasses the telescope's main deployable structure, secondary mirror structure, and beam combiner. As a first assumption, the external coating of all payload elements is assumed to be black Kapton for now.
- **Optical elements:** at this stage of modeling, only the primary mirror is represented here. While its material and thermo-optical coating are not yet defined, it is safe to assume it will be coated with a polished metallic finish, resulting in a low emissivity. As it is not exposed to the sun, the surface absorptivity has little impact on preliminary thermal analysis. Silver or gold coating could also be considered here, but their respective infrared emissivity's are close enough to polished aluminum, and such differences will not yield major differences in thermal behavior at this stage of development.

The solar absorptivity and infrared emissivity are displayed in Figure 4-3.

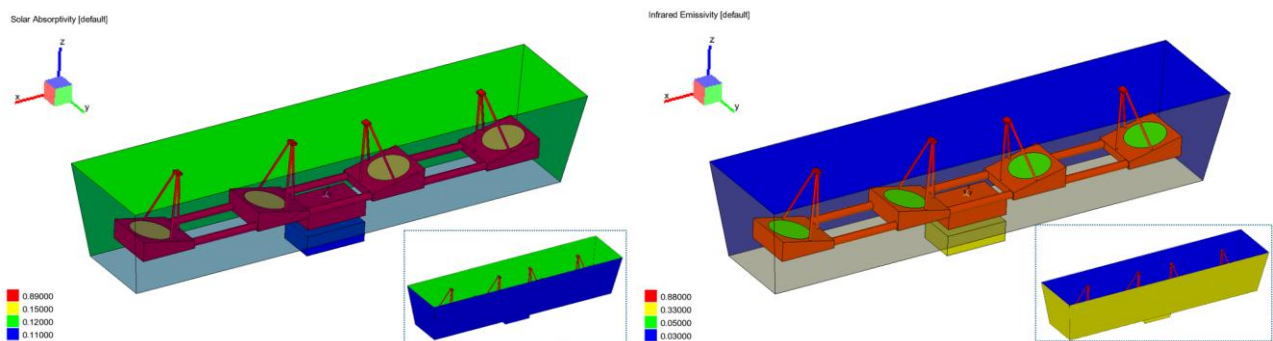


Figure 4-3: Solar absorptivity [Left] and Infrared emissivity [Right] of subsystems external surfaces

Sunshield modelling

The sunshield is geometrically modeled here as a single face. In contrast, it will most likely consist of several layers, arranged to improve insulation between the first and last layers. As these layers will not be in contact with one another, only radiative exchanges can be assumed between them. As such, the sunshield insulation is estimated considering that for N layers of sunshield panel with an emissivity ϵ , the effective emissivity ϵ_{eff} can be expressed as

$$\epsilon_{\text{eff}} = 1/(2/\epsilon - 1) * (1/(N + 1))$$

It is important to note here that, for large insulation systems such as this one, this approach can be considered conservative. More detailed modeling of the sunshield could provide a much better overall insulation, but for the purpose of this preliminary analysis, it is assumed to be sufficient. In the current baseline for the spacecraft design, it is assumed to be composed of five layers.

Thermal analysis performed

During in-flight operations, most sides, except the payload side, are expected to be exposed to direct sunlight at some point. To have a first estimate of the thermal behavior of the spacecraft, three extreme cases are simulated with three different sides fully exposed to the sun (-X, -Y, and -Z directions). Rotations along the Z axis are also expected. Still, they won't be covered at this stage of the analysis as the -X and -y illumination cases represent the two extreme rotation cases along the Z axis.

The heat flux on each sunshield panel is then injected into the thermal model, and a steady-state analysis is performed to determine the temperatures reached passively by the different spacecraft subsystems. Internal dissipation of the spacecraft subsystems is difficult to estimate, given the current design advancement. A preliminary power budget is derived from the JWST power budget and provided in Table 4-1.

Designation (Configuration Tree)	Configuration Item	Power	Reference or rationale
PL-Tx-TLA-M1	Primary mirror	4 x 7 W	Based on scaling w.r.t. JWST optical telescope elements: JWST PM is 6,5m wide for 2,5m here, scaling based on Mirror area would suggest a reduction factor of around 4 for the power budget for the optical elements (M1 + M2). (*) Considered in primary mirror budget
PL-Tx-TLA-M2	Secondary mirror	4 x 1,5 W	
PL-Tx-TLA-TSST	Telescope support structure	0 W *	
PL-Tx-TO	Transfer optics	0 W *	
PL-Tx-DL	Delay line	0 W *	
PL-Tx-LOSCA	Line-of-sight-correction-assembly	0 W *	
PL-MSST-BOOMS	(Deployable) booms	4 x 30W	Based on scaling w.r.t. JWST deployment control subsystems, not accounted in thermal model since not operational during observations
PL-MSST-HDRM	hold down and release mechanism		
PL-CU-DETA	Spectro assembly Beam combiner assembly Spatial filtering assembly Wavelength band separator	300 W	JWST integrated science instrument module = 243W, assuming 25% additional margin
PL-CU-SPA			
PL-CU-BCA			
PL-CU-SPFA			
PL-CU-WLBS			
BUS-TCS-SS	Sun shield	1700W	JWST Spacecraft Bus = 1369 W (including thermal control and cryo-cooler) assuming 25% additional margin
BUS-PRS	Propulsion		
BUS-ADCS	Attitude determination and control		
BUS-Other	EPS, CDHS, COMMS, TCS (active part), STS and MechS (s/c bus), wiring		
Total allocated		2154W	

Table 4-1: Power budget for preliminary thermal modelling

Preliminary thermal model analysis results

The analysis is performed with and without internal dissipation on each subsystem to assess the minimum temperature the spacecraft can reach. The temperature results for the three illumination cases are detailed in Table 4-2. Across the

different illumination cases, the beam combiner's temperature remains relatively stable at -142°C (+/-2°C), while the platform temperature exhibits significant variations. In the X and Y illumination, the platform temperature below -20°C could be too low for the operational requirements of the different subsystems within the platform. In the Z illumination case, its temperature is more consistent with the usual spacecraft platform operating temperature.

Illumination Case		X		Y		Z	
PL-T1	Primary Mirror Assembly	-201°C	-216°C	-199°C	-212°C	-193°C	-202°C
	Center Beam	-209°C	-225°C	-204°C	-215°C	-202°C	-212°C
	Beam -Y	-210°C	-224°C	-204°C	-214°C	-204°C	-213°C
	Beam +Y	-210°C	-224°C	-206°C	-216°C	-204°C	-213°C
	Secondary Mirror Assembly	-187°C	-228°C	-185°C	-220°C	-185°C	-218°C
PL-T2	Primary Mirror Assembly	-189°C	-232°C	-184°C	-211°C	-181°C	-203°C
	Center Beam	-195°C	-235°C	-190°C	-214°C	-190°C	-213°C
	Beam -Y	-199°C	-236°C	-193°C	-214°C	-193°C	-214°C
	Beam +Y	-199°C	-236°C	-195°C	-216°C	-194°C	-214°C
	Secondary Mirror Assembly	-182°C	-237°C	-180°C	-219°C	-180°C	-218°C
PL-T3	Primary Mirror Assembly	-189°C	-238°C	-184°C	-211°C	-181°C	-203°C
	Center Beam	-195°C	-239°C	-190°C	-214°C	-189°C	-213°C
	Beam +Y	-200°C	-240°C	-194°C	-216°C	-193°C	-214°C
	Beam -Y	-200°C	-240°C	-193°C	-214°C	-193°C	-214°C
	Secondary Mirror Assembly	-183°C	-241°C	-180°C	-219°C	-180°C	-218°C
PL-T4	Primary Mirror Assembly	-209°C	-245°C	-199°C	-212°C	-193°C	-202°C
	Center Beam	-214°C	-247°C	-204°C	-215°C	-202°C	-212°C
	Beam +Y	-216°C	-247°C	-206°C	-216°C	-204°C	-213°C
	Beam -Y	-215°C	-247°C	-204°C	-214°C	-203°C	-213°C
	Secondary Mirror Assembly	-189°C	-247°C	-185°C	-219°C	-185°C	-218°C
PL-CU Beam combiner assembly		-144°C	-225°C	-142°C	-207°C	-140°C	-196°C
Platform - BUS		-28°C	-103°C	-20°C	-81°C	32°C	5°C
T1-T2 MSST	- Y	-205°C	-228°C	-197°C	-210°C	-194°C	-204°C
	+ Y	-205°C	-228°C	-199°C	-214°C	-194°C	-205°C
T2-T3 MSST	- Y	-168°C	-227°C	-165°C	-208°C	-164°C	-200°C
	+ Y	-168°C	-227°C	-166°C	-210°C	-164°C	-201°C
T3-T4 MSST	- Y	-208°C	-243°C	-197°C	-210°C	-194°C	-204°C
	+ Y	-208°C	-243°C	-199°C	-214°C	-194°C	-204°C
BUS-TCS-SS Sunshield Outer Side	- Z	-148°C	-171°C	-144°C	-164°C	28°C	27°C
	+ X	24°C	24°C	-254°C	-256°C	-84°C	-84°C
	- X	-257°C	-265°C	-254°C	-256°C	-84°C	-84°C
	+ Y	-253°C	-261°C	24°C	24°C	-84°C	-84°C
	- Y	-253°C	-261°C	-251°C	-256°C	-84°C	-84°C
BUS-TCS-SS Sunshield Inner Side	- Z	-183°C	-214°C	-179°C	-203°C	-110°C	-113°C
	+ X	-115°C	-115°C	-207°C	-215°C	-168°C	-169°C
	- X	-220°C	-247°C	-207°C	-216°C	-168°C	-169°C
	+ Y	-203°C	-233°C	-114°C	-115°C	-165°C	-169°C
	- Y	-203°C	-233°C	-198°C	-216°C	-165°C	-170°C

Table 4-2: Thermal calculation results (Grey number gives the non-operational temperatures)

In all three configurations, a significant issue can be narrowed down to the central telescope units being much hotter than their outer counterparts. Efforts to better insulate the beam combiner from the rest of the units on the payload side and improve its dissipation toward the cold space shall be made. Overall, the payload side would benefit from better insulation from the hot side through the sunshield. Possible trade-offs could be made in the selection of the thermo-optical coating to improve heat rejection from the cold parts of the payload and reduce crosstalk with the hot areas on the payload side.

Thermoelastic stability

Deformations of the spacecraft structure caused by temperature variations play a major role in the overall performance. These deformations, even minimal, can cause significant variations in the optical paths that, in the case of nulling interferometry, can lead to large perturbations in measurements.

Calibration and/or compensation actuators can mitigate some of these deformations. To properly account for such actuators in future spacecraft development, a preliminary estimate of the movements caused by temperature changes in the structure is proposed here based on the provided thermal analysis results. As the 5-layer sunshield configuration is the most probable among those presented, this study is limited to this configuration. The use of very low CTE materials here is also key to achieving thermoelastic stability, and at this early stage of the design, standard CFRP material values must be assumed. Hence, the CTE considered here is $1.5 \mu\text{m/mK}$.

The thermal contraction when cooled down from ambient (20°C) to the calculated temperature is presented in Table 4-3.

Thermal contraction of primary and secondary support structure beams [mm] From ambient [20°C] to calculated temperatures		X Illumination	Y Illumination	Z Illumination
PL-T1	Center Beam	1,31	1,28	1,28
	Beam -Y	1,36	1,33	1,32
	Beam +Y	1,36	1,34	1,33
PL-T2	Center Beam	1,23	1,21	1,20
	Beam -Y	1,30	1,26	1,26
	Beam +Y	1,30	1,27	1,27
PL-T3	Center Beam	1,24	1,21	1,20
	Beam +Y	1,30	1,27	1,26
	Beam -Y	1,30	1,26	1,26
PL-T4	Center Beam	1,34	1,29	1,28
	Beam +Y	1,40	1,34	1,33
	Beam -Y	1,39	1,33	1,32
T1-T2 MSST	- Y	1,49	1,43	1,41
	+ Y	1,49	1,45	1,41
T2-T3 MSST	- Y	1,24	1,22	1,21
	+ Y	1,24	1,23	1,22
T3-T4 MSST	- Y	1,51	1,43	1,41
	+ Y	1,50	1,45	1,41

Table 4-3: Thermal contraction of mirror support structures from 20°C to operational calculated temperatures

As can be observed here, the magnitude of displacement is of the order of the millimeter from ambient to in-flight conditions. Probably far more than can be properly calibrated during on-ground validation. A compensation system would therefore need to handle displacements of this magnitude at the secondary mirror level.

For each telescope unit, the difference in movement between the beams that compose the structure is presented in Table 4-4. It provides an estimate of how the secondary mirror is expected to move. In all cases, the center beam is the hottest one out of the three, hence why the relative contraction of the beam is determined relative to it.

Relative secondary support structure beams retraction		X Illumination	Y Illumination	Z Illumination
PL-T1	Center Beam	0 μ m	0 μ m	0 μ m
	Beam -Y	45 μ m	43 μ m	49 μ m
	Beam +Y	45 μ m	52 μ m	50 μ m
PL-T2	Center Beam	0 μ m	0 μ m	0 μ m
	Beam -Y	63 μ m	56 μ m	60 μ m
	Beam +Y	64 μ m	64 μ m	62 μ m
PL-T3	Center Beam	0 μ m	0 μ m	0 μ m
	Beam +Y	65 μ m	64 μ m	61 μ m
	Beam -Y	66 μ m	58 μ m	60 μ m
PL-T4	Center Beam	0 μ m	0 μ m	0 μ m
	Beam +Y	53 μ m	52 μ m	49 μ m
	Beam -Y	51 μ m	42 μ m	47 μ m

Table 4-4: Relative displacement of secondary support structure beams at operational calculated temperatures [μ m].

Based on the average relative displacement of the side beams, a conservative approach to the M2 angular displacement is to assume it is fixed at the outer diameter of its frame, currently approximated at 300 mm. By retracting, the outer beams tilt the mirror slightly, causing a small angular rotation. The movement is assumed vertical for now, which yields a conservative estimate of the angular movement. This rotation is then used to determine the required linear displacement of the relay optic on the telescope unit, approximately 4 meters from the secondary mirror assembly. Since in most cases, the two main structure beams are at very close temperatures, this calculation is not done for the main structure assembly. Results are presented in Table 4-5.

Secondary mirror angular movement		X Illumination	Y Illumination	Z Illumination
PL-T1	Arcsec ["]	23,4	24,5	25,4
	Degrees [°]	0,0065	0,0068	0,0071
	Relay mirror displacement [mm]	0,45	0,48	0,49
PL-T2	Arcsec ["]	32,8	31,1	31,5
	Degrees [°]	0,0091	0,0087	0,0087
	Relay mirror displacement [mm]	0,64	0,60	0,61
PL-T3	Arcsec ["]	33,9	31,3	31,4
	Degrees [°]	0,0094	0,0087	0,0087
	Relay mirror displacement [mm]	0,66	0,61	0,61
PL-T4	Arcsec ["]	27,0	24,2	24,8
	Degrees [°]	0,0075	0,0067	0,0069
	Relay mirror displacement [mm]	0,52	0,47	0,48

Table 4-5: Secondary mirror angular movements

The preliminary thermal model presented here, and the calculations performed, provides a first estimate of the payload's thermal behavior for the LIN16 configuration. The initial assumption for a 5 Layer sunshield, while providing a good level of insulation, proves insufficient to reach very low temperatures (sub-50 K). Due to the spacecraft's large size and the layout of the different telescope units, another major issue is the lack of thermal homogeneity without active thermal regulation. With current temperature discrepancies across illumination cases, an active thermal control system for the payload might address homogeneity but would require significant heat generation to bring all elements to the same temperature.

Hence, reducing as much as possible the passive cooling of the payload should be the priority of future design work on the thermal architecture of the spacecraft. Such improvement can be made by developing the current thermal model:

- Implement a geometrical representation of the sunshield individual layers.
- Implement passive insulation between telescope units and the beam combiner. Changes in thermos-optical properties, for example, or the addition of insulation blankets to reduce view factors, are possible solutions.
- Perform rotation cases with illumination at an intermediary position between X and Y directions.
- Complement the current analysis with a sensitivity analysis to identify other design drivers (coatings emissivity, conductive links, etc.) At a higher level, efforts should also be made to refine the power budget and, if possible, reduce internal heat generation across the different subsystems.

5. CONCLUSION

Based on optical design for each telescope unit and high-level performance requirements, a first estimate of the mass budget and volume allocation was made, as these two parameters are among the main design drivers for the overall spacecraft architecture.

Several different architectures for the main structure deployment were then proposed, going from linear deployment with an 8m baseline to an X-array configuration with a baseline up to 32m. Different technologies allowing such deployment, and most notably to allow linear extension of the main structure, were discussed. The proposed solutions were compared based on preliminary modal and structural analysis to have a better understanding of their feasibility. Out of the six proposed alternatives, the largest ones (X32 and Lin32) were ruled out due to stiffness and mass limitations. The remaining designs are traded against other criteria, such as key performance indicators such as exoplanet detection and characterization yields, which is further specified in a companion paper⁸.

As a result, it was decided to establish a baseline design for the spacecraft based on the linear configuration with a 16m nulling baseline (Lin16). In addition, several configurations for the secondary mirror deployment were discussed, and for each configuration, preliminary modal analysis was also done to provide a better understanding of each solution's advantages and drawbacks. At this stage, and contrary to the main support deployment, no configuration could be ruled out as none of them showed major drawbacks regarding the mass of stiffness.

Finally, a thermal model was made to have a first assessment of the spacecraft's thermal behavior depending on its orientation toward the sun, since it can be expected that, due to its size, large gradients could be expected across the structure. This model was made assuming a 16m linear architecture with a tripod supporting structure for the secondary mirror and highlighted the need for active thermal control of the structure, as in its current state, passive cooling could not guarantee a stable temperature across the structure. The results obtained also provided insight into the key drivers of the thermal design that will need to be accounted for in further design phases for such a spacecraft.

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