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Planetary sunshade for solar geoengineering: Preliminary design of a precursor system and mission

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ABSTRACT

The increasing urgency of climate change mitigation necessitates innovative solutions beyond terrestrial efforts. Space-based solar geoengineering – particularly a Planetary Sunshade System (PSS) positioned near the photo-gravitational equilibrium point L_1^*, which lies closer to the Sun than the classical L_1 due to the effect of solar radiation pressure – has been proposed as a potential method to reduce incoming solar radiation and stabilize global temperatures.

This paper presents the preliminary design of a precursor mission aimed at demonstrating key technologies essential for the deployment of a full-scale PSS. The proposed mission features a 12U CubeSat equipped with a 400 [m^2] solar sail, which will be used for propulsion, attitude control, and station-keeping at L_1^*. The mission objectives focus on validating the long-term performance of optical shielding materials, demonstrating solar sailing as a sustainable propulsion method, and assessing the feasibility of autonomous orbit and attitude control systems.

The technical and economic feasibility of the precursor mission, with an estimated budget of 10M USD is examined. By addressing key uncertainties in spacecraft formation flying, material degradation, and long-term solar sailing operations, this mission represents a crucial step toward the realization of a scalable PSS for climate intervention.

1. Introduction

Human activities has been influencing the climate, primarily through greenhouse gas emissions that cause global temperatures to rise [1]. According to the World Meteorological Organization (WMO), the 2024 annual average global temperature was 1.55 ± 0.13 °C above pre-industrial levels [2]. This presents a clear risk of surpassing the "safe limit" of 2 °C above pre-industrial levels, as outlined in the Paris Accord and IPCC reports [1,3].

Even with a significant reduction in CO_2 emissions over the coming decades, anthropogenic climate change impact could reach a critical point [4], leading to persistently elevated temperatures. Therefore, it is crucial to consider additional mitigation strategies to counteract the risk of unacceptably high temperatures. One possible mitigation approach is the deployment of sunshades in space, positioned near the Sun–Earth/Moon Center of Mass (CoM) conjunction line, at the photo-gravitational equilibrium point L_1^*, where they could reduce the amount of solar radiation reaching Earth. This point corresponds to the modified equilibrium of the Sun–Earth/Moon CoM Circular Restricted Three-Body Problem (CR3BP), and it is shifted sunward due to the effect of solar radiation pressure (SRP). Investing in such strategies now can be viewed as an insurance policy against potential catastrophic consequences due to climate change in the next few decades [5].

This method is one of several geoengineering options currently under consideration, as briefly described in [6], where an implementation roadmap for a Planetary Sunshade System (PSS) is presented. A PSS might constitute the most promising geoengineering approach with the least negative side effects, specifically when compared to atmospheric-based solutions such as Stratospheric Aerosol Injection (SAI). Indeed,

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most studies on SAI often overlook the practicalities of sustained deployment and the long-term commitment it requires. Moreover, its irreversible nature poses high risk on Earth's fragile atmosphere, with possible both regional and global impacts. Furthermore, the PSS stands out also with respect to other space-based solutions. For instance, Earth-orbiting reflectors would require multiple orbiting satellites which would provide intermittent and localized shading, resulting in non-uniform shading, along with growing concerns about orbital crowding. Another approach involves the use of dust-clouds, but with significant limitations. Near-term deployment would be challenging for the need of space platforms around L_1 or for the required launch infrastructure on the Moon. Additionally, the dynamics of dust particles is highly chaotic, resulting in a non-uniform shading pattern. These particles would also require frequent replenishment, as they are likely to disperse rapidly, making long-term maintenance difficult.

The size of the final PSS will depend on how the climate develops in the coming decades and the actual temperature sensitivity of CO_2 levels in the atmosphere. It would most likely be composed of a large number of smaller spacecraft acting as a swarm, possibly assembled together to larger units.

Deployment of the PSS might be necessary to begin as early as 2040 [7]. To ensure all technology is ready and proven by then, development should commence as soon as possible. Key steps in the roadmap include precursor missions for in-space demonstration and verification [6].

This paper outlines a proposed first precursor mission to test technologies likely to be used in the final design of the PSS. In particular, a 12U Cubesat is considered to reach $L_1{}^*$ using solar sailing and with a budget around \$10M.

In particular, the main original contributions of the present paper are: (1) the identification of the key technological and operational aspects that a precursor mission would require, including the critical systems and capabilities necessary for a successful implementation; (2) an analysis of the most suitable orbit for the deployment and operation of the PSS precursor mission.

The paper is organized as follows: in Section 2 a review of the PSS is provided, including a discussion of the roadmap and the need for a precursor mission. Section 3 outlines the goals and requirements of the precursor mission, including its need and objectives. Section 4 presents details on the precursor mission analysis, focusing on the target orbit identification and injection and transfer phases. Section 5 covers the precursor mission architecture, with a focus on the concept of operation and critical technologies. Section 6 analyze risks and costs of the precursor mission, Section 7 discusses the choices made for the precursor mission, and, finally, a conclusion is provided in Section 8.

2. Background

2.1. The planetary sunshade system

The PSS is a space-based, reversible solar geoengineering infrastructure proposed to reduce the incoming solar radiation by deploying a "solar-light umbrella" between the Sun and Earth, most likely composed of a swarm of solar-sail spacecraft assembled in space.

Typically, research on space-based solar geoengineering assumes a uniform reduction of approximately 1.7% in incoming solar radiation, a value estimated to offset the radiative forcing from a doubling of atmospheric carbon dioxide levels [8]. However, the actual climatic impact and system requirements of such an intervention must be assessed through advanced, high-resolution climate simulations with global circulation models which consider the coupling between the atmosphere and the ocean. So far, only preliminary studies using simplified energy balance models have been conducted [9,10], showing a reduction in Earth's average temperature of about 1–2 °C in a CO_2 680 ppm scenario.

This solution is surveyed in [11], providing top-level comparisons with respect to the non-reversible atmospheric strategies, and in [12], where a conceptual space-climate architecture framework is proposed to drive progress across the entire span of complex issues that are involved.

In 1989 James T. Early proposed a 2000 km-wide glass shield at the Sun–Earth first Lagrange point L_1, to counteract the greenhouse effect on Earth [13]. The first (to the authors' knowledge) to propose a solution with Earth-manufactured shades, was Roger Angel in 2006 [14]. He suggested trillions of flying small space robots, weighing 1 g each, which would be launched with a gigantic electromagnetic gun embedded in a mountain. Generally, the main strategy consists of a swarm of solar-sail spacecraft, capable of active control, which can be either totally or partially assembled to build one or more occulting disks. The swarm astrodynamics, including the proposal of suitable families of photo-gravitational L_1 Halo orbits, has been investigated in [15,16] while orbital transfers strategies are analyzed in [17,18]. Finally, other architectures have been proposed utilizing space resources [19,20], including the Moon [21].

2.2. Roadmap discussion and need of precursor

The precursor mission is designed to test, validate and demonstrate some of the key technologies required for the realization of a future PSS [6], and determine the course of action for its development and deployment. Given the complexity of such a system, achieving the necessary Technology Readiness Levels (TRLs) demands advancements across multiple domains, including orbital robotics, material science, deployment mechanisms, autonomous operations, and system integration.

3. Precursor mission definition

3.1. Needs, statement, objectives

The precursor mission addresses key technological and operational uncertainties associated with the deployment and long-term functionality of the PSS. One of the primary challenges is the validation of solar sailing as a method for long-duration station-keeping. Since the sunshade will need to remain at a stable location within the photo-gravitational environment of L_1^*, demonstrating that solar radiation pressure can effectively be used for maneuvering and maintaining position is crucial. Additionally, the optical shield materials require thorough testing under actual space conditions to assess their durability, thermal stability, and resistance to micrometeoroid impacts. Prolonged exposure to solar radiation and space weather could degrade the material, affecting the overall efficiency of the sunshade.

Another key aspect of the mission is the evaluation of the Attitude and Orbit Control System (AOCS), which is necessary for ensuring precise positioning and orientation of the spacecraft at L_1^*. Given the complex dynamical environment at this location, an advanced control strategy must be implemented and tested to ensure that the sunshade remains optimally positioned. Furthermore, the feasibility of deploying large-scale structures in space must be assessed, particularly in terms of packaging efficiency, deployment reliability, and in-orbit stabilization. The ability to successfully demonstrate these capabilities in a precursor mission will provide essential insights into the scalability of future sunshade designs.

3.1.1. Precursor goals

In order to mitigate the risks associated with new technologies, the precursor mission focuses on those with the highest potential impact and greatest uncertainty. This risk-based approach prioritizes the most critical and unproven aspects of the PSS. Based on our assessment, the long-term performance of the optical shield material and the operational complexities and scalability of the AOCS in the L_1^*

environment emerged as high-risk areas due to their fundamental roles in the mission's success. By addressing these areas through early flight demonstrations, we can accelerate technology maturation and ensure a solid foundation for future system development.

The focused objectives for the precursor mission also consider advancements expected in parallel large-scale systems, such as lunar orbiting stations, space-based solar power, asteroid mining, and interplanetary missions. These initiatives will contribute valuable experience and technology heritage in areas such as large-scale structure assembly, power management, and autonomous operations that can be integrated into the PSS systems. However, while these technologies are advancing independently, specific aspects of the PSS—such as the long-term durability of specialized optical materials and the precise control required for formation flying in the vicinity of L_1^*—require dedicated demonstration. By concentrating on these unique aspects, the precursor mission can complement broader technological advancements and ensure that all critical components are adequately validated prior to the full-scale development of the objective system.

In-space testing allows for the direct observation of material behavior, system dynamics, and environmental interactions in the operational environment. For the optical shield material, this means exposing it to actual space weather effects, radiation, and micrometeoroid impacts at L_1^*, where the operational environment differs significantly from other orbits. Similarly, testing the AOCS in situ provides invaluable data on non-linear dynamics, gravitational perturbations and the challenges of maintaining precise control over large, distributed structures.

Affordability is an important consideration in the development of any space mission, especially when aiming for near-term progress toward an ambitious objective. The precursor mission adopts a focused and cost-effective approach by concentrating on two key demonstration objectives. This targeted strategy minimizes the scope and complexity of the mission, reducing costs while still addressing the most critical technological challenges. By demonstrating the viability of the optical shield material and AOCS in an operational environment, the mission can achieve meaningful progress toward the objective PSS.

3.1.2. Objectives

The precursor mission's targeted objectives are essential steps toward developing a comprehensive PSS. The objectives for the PSS precursor mission include:

(a) Scalable System Operations in the Vicinity of L_1^*.

The precursor mission will focus on gaining operational experience with AOCS in the L_1^* environment. This includes testing the capability to sail and maintain stable orbits, perform station-keeping maneuvers, and manage the orientation of the sunshade elements through feedback control. The mission will explore the challenges of controlling a scalable system, laying the groundwork for the development of guidance, navigation and control (GN&C) algorithms and hardware capable of managing the complexities of a large, distributed sunshade network.

The successful operation of a large-scale PSS will require precise attitude and orbit control. At L_1^*, solar radiation pressure, the centrifugal force and the gravitational ones from the Earth–Moon system and the Sun must be carefully balanced, as even minor deviations can result in significant positional drift. Furthermore, the future sunshade system will consist of multiple large structures, necessitating a sophisticated and reliable AOCS to maintain the desired formation and orientation.

(b) Long-term Performance of Optical Shield Material in the L_1^* Environment.

The precursor mission aims to validate the long-term performance and stability of the chosen optical shield material under the specific environmental conditions at L_1^*. This objective will provide crucial data on the material's degradation over time, its response to space weather, and its mechanical resilience. By accumulating flight heritage, we can refine material choices, coatings, and protective measures for the full-scale system.

Moreover, optical solar sail degradation may influence both the spacecraft's trajectory and its attitude control strategy. A reduction in reflectivity leads to a lower SRP force and, consequently, a shift in the equilibrium position. These effects may require adjustments in the control law to maintain alignment and stability near L_1^* [22].

The effectiveness of a PSS relies heavily on the optical properties and durability of its shield material. The material must withstand the harsh conditions of space, including exposure to solar radiation, temperature fluctuations, and potential impacts from micro-meteoroids. Ensuring the long-term stability and reflectivity of the optical shield material is critical to the system's ability to reduce oncoming solar radiance effectively.

3.2. Mission requirements

The mission aims to develop and demonstrate the critical technologies required for a PSS that can effectively produce a shade toward the Earth. To achieve this, the mission will operate for a specified minimum duration, during which it will gather sufficient data to test all system functionalities and assess the environmental and long-term degradation of materials.

The mission will be conducted in an orbit specifically chosen to demonstrate the capabilities of solar sails within that dynamic regime. This orbit will also simulate the conditions for the future deployment of the PSS. A primary focus will be to prove the feasibility of solar sailing, including precise attitude and orbit control. This will ensure the scalability of these systems in terms of stability, maneuverability, deployment mechanisms, and impact mitigation.

Throughout the mission, data will be collected and transmitted, focusing on the performance of the solar sails, the durability of the materials used, thermal properties, and the overall effectiveness in reducing solar radiation. Additionally, the mission aligns with the United Nations Sustainable Development Goals (SDG), particularly SDG 13, which addresses Climate Action, and SDG 9, focusing on Industry, Innovation, and Infrastructure, by promoting the development of innovative and sustainable space technologies.

4. Precursor mission analysis

4.1. Target orbit identification

The photo-gravitational CR3BP model is employed to study the dynamics of the system. In this model, the two primary bodies are the Sun and the Earth–Moon CoM, modeled as point masses with a fixed distance of 1 astronomical unit (AU) between them. Both bodies are assumed to follow circular orbits around their common CoM [23]. In this work, the values of primaries' masses and orbital parameters are adopted as per NASA's Sun, Earth, Moon Fact Sheet [24], and these are used to non-dimensionalize the photo-gravitational CR3BP equations.

In this model, a solar-sail spacecraft is influenced by the gravitational attraction of both the Sun and the Earth–Moon system, as well as the SRP. SRP is the force exerted by sunlight photons striking the surface of the spacecraft, with the direction of the force assumed to be parallel to the solar sail's surface normal vector, denoted by $\hat{n}$ [25].

To describe the system's dynamics, we adopt a barycentric and synodic (co-rotating) Cartesian Coordinate System (CCS). The origin of the CCS, O_{xyz}, is located at the CoM of the Sun–Earth–Moon system, which remains fixed in an inertial reference frame. The $\hat{x}$-axis lies along the straight line joining the center of the Sun and the CoM of the Earth–Moon system, while the $\hat{z}$-axis is aligned with the angular momentum vector of the Sun–Earth–Moon CoM system. The $\hat{y}$-axis completes the right-handed CCS. In this reference frame, the centers of mass of the Sun and the Earth–Moon system are fixed along the $\hat{x}$-axis, at distances equal to μ and $1-\mu$, respectively, where μ is the mass parameter defined as $\mu = \frac{m_e+m_m}{m_s+m_e+m_m}$, with m_s, m_e, and m_m representing the masses of the Sun, Earth, and Moon, respectively.

The non-dimensional equations of motion governing the orbital dynamics of the solar sail in the rotating Sun–Earth–Moon system are given by:

$$\begin{aligned} \ddot{x} - 2\dot{y} - \frac{\partial U}{\partial x} &= a_x \\ \ddot{y} + 2\dot{x} - \frac{\partial U}{\partial y} &= a_y \\ \ddot{z} - \frac{\partial U}{\partial z} &= a_z \end{aligned} \tag{1}$$

where the potential function U is defined as:

$$U = \frac{1}{2}\left(x^2 + y^2\right) + \frac{1-\mu}{\left\|\underline{r}_1\right\|} + \frac{\mu}{\left\|\underline{r}_2\right\|} \tag{2}$$

and the Sun-sail and Earth–Moon CoM-sail vectors component matrices in CCS are

$$\boldsymbol{r}_1 = [x+\mu, y, z], \quad \boldsymbol{r}_2 = [x-1+\mu, y, z]. \tag{3}$$

The acceleration due to solar radiation pressure for an ideal, perfectly reflecting solar sail, in terms of component matrices in CCS, is $\boldsymbol{a} = \left[a_x, a_y, a_z\right]$, while in his vector form this is as in [25]:

$$\underline{a} = \beta \frac{1-\mu}{\left\|\underline{r}_1\right\|^2}\left(\hat{n}\cdot\hat{r}_1\right)^2 \hat{n}. \tag{4}$$

where $\beta = \sigma^* \frac{QA}{M}$ represents the SRP-to-solar gravity ratio, dependent on the area-to-mass ratio of the sail, with A and M being the area and mass of the sail, respectively, while Q is a function of the optical properties of the solar sail, varying from 1 for perfectly reflecting surface to 0 for fully transparancy. The critical loading parameter for a solar sail is $\sigma^* \sim 1.535$ [g/m^2], which is the mass to area ratio that a sail, oriented perpendicular to the sun line, should have to generate a force equal and opposite to the solar gravitational force.

Furthermore, the orientation of the solar sail is defined by two angles: the cone angle α, which is the angle between the normal vector to the sail's area $\hat{n}$ and the $\hat{x}$-axis, and the clock angle δ, which is the angle between the projection of $\hat{n}$ onto the $\hat{y}-\hat{z}$ plane and the $\hat{y}$-axis. Therefore, the normal vector component matrices in CCS is represented as:

$$\hat{n} = [\cos(\alpha), \sin(\alpha)\cos(\delta), \sin(\alpha)\sin(\delta)]. \tag{5}$$

It is possible to demonstrate that there exists an optimum β, or equivalently a position of the photo-gravitational equilibrium point L_1^*, that minimizes the total mass of the PSS, which is independent of the required solar radiance reduction $\frac{\delta Q}{Q}$ and Q. In particular, the optimal L_1^* is located at $\sim 2.364 \cdot 10^6$ [km] from Earth, with a corresponding optimal $\beta = 0.0349$, resulting in a sunward shift of $\sim 0.868 \cdot 10^6$ [km] with respect to the pure CR3BP L_1 point. For the precursor mission design, the same β value is adopted in order to satisfy the mission objectives of scalable system operations and long-term performance of the optical shield material in the same L_1^* environment as the full-scale PSS.

4.1.1. Orbit design

Since the solar sail operates approximately 2 million [km] away from Earth and the Moon, the CR3BP assumptions are quite accurate, as the combined gravitational influence of Earth and the Moon can be effectively modeled by their CoM. However, to satisfy the shading constraint and ensure uniform shading on Earth, the motion of the Earth–Moon system must be considered. This is because the Sun–Earth conjunction line shifts as the Earth oscillates by approximately ±4640 [km] around the Earth–Moon CoM, i.e. the radius of Earth's circular motion around the Earth–Moon CoM. To maintain continous shading off the Earth, the Sunshade must be positioned along the shifting Sun–Earth conjunction line.

The orbit design for the solar sail relies on the control strategy developed in [15], using SRP to maintain a periodic orbit without the need for propellant, relying solely on the sail's dynamics. This control algorithm requires inputs for the y and z control laws to determine the x motion that guarantee a three-dimensional periodic trajectory for the solar sail.

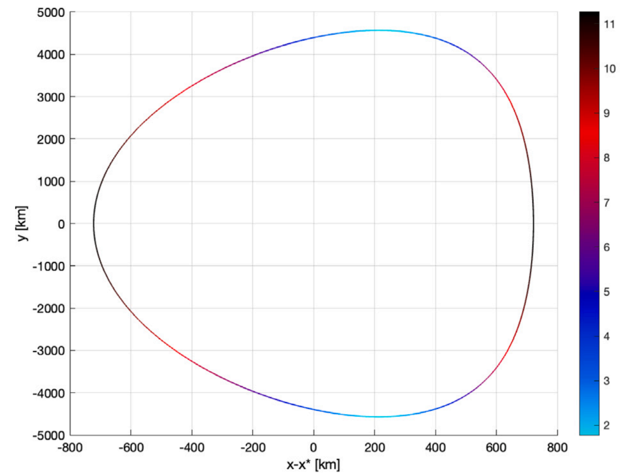

Fig. 1. x-y plane projection of the target orbit, with the velocity along the orbit indicated by color. The plot is expressed in the Co-Rotating CCS, where the x-axis connects the center of the Sun with the CoM of the Earth–Moon system, the z-axis is aligned with the angular momentum vector of the system, and the y-axis completes the right-handed frame. x^* is the operating point located approximately 14000 [km] away from L_1^*. (For interpretation of the references to color in this figure legend, the reader is referred to the web version of this article.)

If the x motion is assumed to be fixed at an operating point, the control laws for the y and z motion are equal to:

$$z = 0, \quad y = \frac{(x^* + \mu) r_3 \sin(\omega t)}{1 + r_3 \cos(\omega t)} \tag{6}$$

where:

- ω is the frequency of the Earth–Moon system,
- x^* is the operating point near the photo-gravitational L_1^* point
- r_3 is the distance between Earth and the Earth–Moon CoM,

However, it is not possible to keep x^* constant, as the SRP control strategy only allows for maintaining the periodic motion in the y and z directions, resulting in a periodic motion around x^*. Despite this, the amplitude of the x motion obtained is such that the maximum misalignment with the Sun–Earth line is limited to just 14.55 [m] along one period. The x-y plane projection of the resulting orbit is depicted in Fig. 1. The maximum value of α is equal to 7.35° and its variation during one period is shown in Fig. 2, while from Eq. (1) it follows that $\delta = [0, \pi]$ if $\ddot{y} + 2\dot{x} - \frac{\partial U}{\partial y} \gtrless 0$.

This will be the target orbit for the precursor mission. However, for the final PSS design other orbits might be desired, e.g. for latitude variations on Earth [10].

4.2. Injection and transfer

This section outlines the trade off between time of flight and propellant consumption for the transfer and injection of solar-sail spacecraft units toward the L_1^*. Starting from the highest transfer period, one possibility concerns an escape trajectory from Earth–Moon gravitational influence and a subsequent transfer toward the equilibrium point. In this case, the solar-sail should gain sufficient energy in order to open the dynamics realms toward the Sun. The trajectory optimization in this case should integrate the Sun–Earth–Moon Bi-Restricted Four Body Problem (BR4BP) with drag, Earth's oblateness (J2 effect), and SRP influence. In principal, the escape could be possible from a minimum altitude around 600 [km] [26], and the propulsion budget should

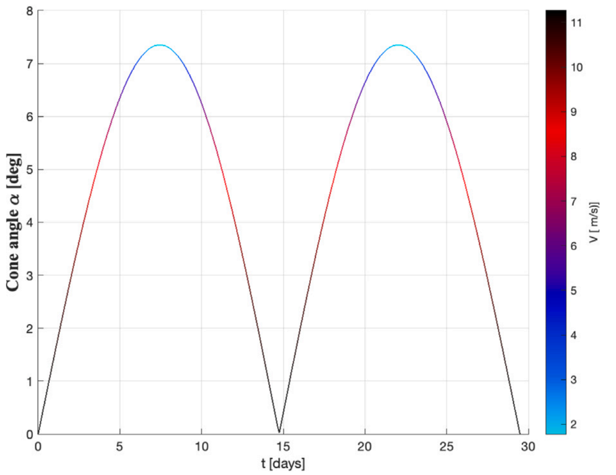

Fig. 2. Cone angle α variation during one period.

accommodate the ascent trajectory with the launch vehicle, and the insertion at escape altitude, followed by the deployment of the solar sail. According to the area to mass ratio, the escape time may reach thousands of days. For the selected β value, it will require almost three years. This would of course scale if the solar sailing starts not at minimum escape altitude but to an higher one. Starting at 2000 [km] altitude a solar sail craft with an areal density of 22 [g/m^2] could escape Earth in 531 days and with an areal density of only 9 [g/m^2] it could be done in just 285 days [17]. However, since a LEO escape by solar sailing puts very high demands on the attitude control of the solar sail an escape from geostationary orbit (GEO) it is considered. Also escaping from GEO it is a challenge, but with even using vanes for attitude control it is possible to escape from Earth with a small solar sail, as shown in [27].

If the solar sail has sufficient energy for escape, a direct transfer with solar sailing would take few months by applying a minimum-time attitude law control [17,18,26]. Instead, if bi-impulsive direct strategy is used for all the phases (launch, transfer and insertion), the period is always in the order of months but higher respect to sailing and less than one year. Anyway, this would require a ΔV (between 4–5 [km/s]) for both escape the Earth–Moon system and stop toward the L_1^* point [18].

Once in the operational orbit, station-keeping is managed by adjusting the solar sail's attitude to counteract perturbations by exploiting the solar radiation pressure to maintain a periodic- or quasi-motion around the photo-gravitational equilibrium.

5. Precursor mission architecture

Building upon the mission objectives, the precursor mission architecture incorporates critical subsystems necessary for validating key technologies essential for the future development and deployment of the PSS. Key challenges for this project include integrating solar sails, selecting materials suitable for space environments, developing deployable structures and mechanisms, and optimizing the AOCS. These challenges are also addressed in [6] and discussed further in this section.

5.1. Concept of operation

The Concept of Operations (ConOps) for the precursor mission outlines the critical phases required to validate key technologies for the development of the PSS. The mission is structured into five distinct phases, as illustrated in Fig. 3, each addressing specific operational objectives and enabling progressive risk reduction for the deployment of a scalable PSS.

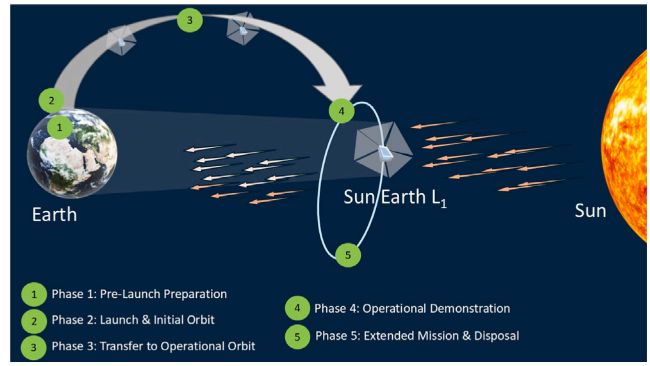

Fig. 3. ConOps for the precursor mission.

5.1.1. Phase 1: Pre-launch preparation

The mission begins with pre-launch activities involving the assembly, integration, and comprehensive testing of the spacecraft systems. These systems include the solar sail, deployable structures, AOCS, and communication subsystems. Rigorous ground testing ensures that all components meet the required performance and reliability standards. This phase concludes with final validation and launch readiness reviews.

5.1.2. Phase 2: Launch & Initial orbit

Considering the results from Section 4.2, the spacecraft will be launched as a rideshare payload into GEO, or possibly just GTO, depending on launch availability. Post-launch, the spacecraft undergoes system activation and a series of health checks to confirm the functionality of all onboard subsystems. Initial orbit adjustments may be conducted to establish optimal conditions for the subsequent transfer phase.

5.1.3. Phase 3: Transfer to operational orbit

Following initial operations in GEO (or GTO), the spacecraft initiates its escape from the Earth gravitational influence. Once free from Earth it transfers to the operational orbit near the photo-gravitational equilibrium point, L_1^*. This transfer employs solar sailing, a propulsion technique that leverages SRP to generate thrust without the need for traditional fuel. The AOCS ensures precise navigation and trajectory corrections during this phase, maintaining stability under the dynamic environmental conditions.

5.1.4. Phase 4: Operational demonstration

Once in the operational orbit, the spacecraft conduct in-situ technology demonstrations. Key objectives include evaluating the performance and durability of the optical shield material under the harsh conditions at L_1^*, such as solar radiation, space weather, and micrometeoroid impacts. The spacecraft also tests advanced AOCS algorithms for maintaining the orientation and position of the solar sail. Data collected during this phase will be transmitted to ground stations for analysis, providing critical insights for the design of the full-scale PSS.

5.1.5. Phase 5: Extended mission & Disposal

The final phase focuses on extended operations, which may include additional technology demonstrations or long-term monitoring of system performance. Depending on the mission's outcomes and resource availability, the spacecraft will either remain in the operational orbit or removed far from the Sun–Earth line. Data gathered throughout the mission will contribute to the refinement of PSS technologies and operational strategies.

5.2. Critical technologies

The PSS precursor mission will use solar sailing as propulsion method, which was first successfully used by JAXA's IKAROS mission in 2010 [28]. Since then, other missions such as NASA's NanoSail-D [29], The Planetary Society's LightSails [30], and most recently NASA's ACS3 [31], have all successfully used solar sails on Earth orbits. However, apart from IKAROS, no other solar sailing mission has traveled outside of Earth's Sphere of Influence (SOI), and solar sailing still has to be further validated for long-distance missions, making further testing crucial for the feasibility of the PSS.

During its mission the precursor spacecraft will be able to validate new technologies specifically developed for this propulsion method, many of which have not yet been tested in space. By advancing these technologies, the precursor mission aims to make significant contributions toward the development of the PSS. These advancements will demonstrate the feasibility of long-distance solar sailing for future missions.

Solar sailing, also known as solar propulsion, will be used as the transfer method to transport the precursor spacecraft from Earth orbit to the vicinity of L_1^*. Additionally, this method will provide continuous control for station-keeping and orbital adjustments at the final destination.

Solar sails use either reflective or transmissive materials, which are crucial to harness SRP to achieve the desired orbital configurations for long-duration missions. Solar sails utilize highly reflective surfaces to maximize the momentum transfer from sunlight, while transmissive variants allow partial light penetration to modulate the forces applied [32]. The efficiency of these sails is directly dependent on their optical and structural properties, which makes material selection a critical aspect of system design. To maximize the effect of SRP, the sail must be extremely lightweight with a high area-to-mass ratio [25]. These characteristics allow the relatively small force exerted by SRP to have a significant impact on the spacecraft's trajectory. These design principles also align well with those of the PSS.

Solar sails will also be able to test many of the necessary technologies for the PSS, such as material durability, deployment mechanisms and attitude control. A solar sail typically consists of two primary components: the membrane and coatings on both sides. The membrane makes up the base of the sail and provides structural integrity, then the coatings are applied to achieve the desired surface properties, particularly high reflectivity. At least one side of the sail must be coated with a reflective material for solar sailing to work. The solar sail will have a square shape which can be divided into four equal isosceles triangles and is ideal for folding, stowage and deployment. The sail is typically supported with lightweight booms, which are discussed further in Section 5.2.2. The materials used for the sail membrane, coatings, and booms are detailed in Section 5.2.1.

5.2.1. Material selection

Material selection is very important to ensure that the precursor mission meets its requirements and is able to demonstrate the necessary capabilities. The focus of material selection was placed on the main elements, including the membrane, coatings, and booms, which were evaluated based on how their respective properties best fit the mission and their long-term durability. Other components, such as adhesives, fasteners, and smaller elements, are still under consideration for further analysis.

Previous missions have utilized three different materials for the membrane: polyimide resin [28], Clear Polymer 1 (CP1) [33], and Polyethylene Naphthalate (PEN) [34]. Among these, CP1 was selected as the most suitable material for the precursor due to its scalability [33], thinner profile compared to polyimide resin, and higher melting point than PEN. CP1 can be scaled efficiently across a wide range of sizes, from 10 to 10 000 [m^2], making it adaptable for various mission requirements. Additionally, it is UV resistant and has a well-established lifetime in geostationary orbit (GEO), making it ideal for space applications [35]. Although new membrane materials were not explored, the precursor mission will test CP1's durability over a longer duration than previously conducted, providing data on its performance in extended space missions.

According to McInnes [25], aluminum is the optimal material for the reflective coating, due to its high reflectivity, high melting point and low density. Aluminum has been consistently used in all previous solar sailing missions, and a 0.1 [μm] thick aluminum coating can achieve a reflectivity of 0.88–0.9. While alternative coatings like lithium or silver are viable options, they are generally less ideal due to lithium's lower melting point and silver's higher density and by considering that it tarnishes easily. Although silver exhibits excellent performance at optical wavelengths, it has a sharp transparency window at UV wavelengths, allowing solar UV radiation to penetrate the coating and degrade the plastic substrate underneath. This makes aluminum a more suitable option for long-term space applications. The choice of plastic substrate, such as CP1, is partly based on its UV resistance; however, this design choice may limit the use of coatings like silver, which could degrade the substrate. Alternatively, other materials could be considered for the substrate if necessary to accommodate different coatings.

For the deployable structure and mechanisms, if the sail employs a spinning design, only tip masses are required, however this introduces an additional layer of complexity required of the AOCS. If booms are used to support the sail, two primary materials have been previously utilized: the stainless steel alloy Elgiloy and Carbon Fiber Reinforced Polymer (CFRP). While Elgiloy was used in the Triangle Rollable And Collapsible (TRAC) booms of early solar sailing missions [30,36,37], CFRP has become the preferred choice due to its lightweight, thermal stability, and lower manufacturing cost. The CFRP material was selected for the PSS precursor mission, considering its successful use in recently developed Collapsible Tubular Mast (CTM) booms made by NASA's Deployable Composite Boom (DCB) project [31,38].

5.2.2. Deployable structure and mechanisms

Two deployment methods were evaluated for the precursor mission: a spinning design and a deployable boom structure. The spinning design uses centrifugal force to extend the solar sail, with tip masses attached to the corners to unfurl it. This system offers reduced mass and easier scalability by eliminating rigid structures like booms. However, it introduces challenges for attitude control, as the sail requires constant rotation to maintain its shape. The flexibility of the membrane makes it sensitive to impulsive torques, from for example thrusters, which could cause unwanted oscillations and affect thrust distribution. Despite this, the spinning motion helps counteract external disturbance torques, an essential feature for ensuring the sail remains aligned for optimal propulsion and navigation [28].

Given the complexity of attitude control, the deployable boom structure was considered a more suitable option. This method provides a simpler and more reliable solution, reducing the demands on attitude control, which is critical for precision in navigation and thrust alignment, key aspects of mission success. While the deployable booms may increase mass compared to the spinning design, the trade-off is deemed acceptable for the precursor mission, given the spacecraft's smaller size. For a larger, nominal spacecraft, the mass difference would need to be reassessed to ensure it remains within acceptable limits.

The solar sail will be supported by four booms and deployed using a composite boom deployer. The solar sail will have a dimension of 20 [m] x 20 [m], resulting in a total surface area of 400 [m^2]. Two types of booms were considered for the precursor mission, the TRAC boom and the CTM boom. After evaluation, the CTM boom, developed under NASA's DCB project and constructed from CFRP was deemed the optimal choice for the mission [39]. These newly developed booms are remarkably lightweight while providing the necessary structural support for the solar sail. The booms have a lenticular-shaped closed cross-section, which provides bistability along with high bending and torsional stiffness [38]. The cross-section can also be pressed flat allowing it to be coiled around a hub for greater volume efficiency. Additionally, the use of CFRP provides thermal stability and significantly reduces both mass and manufacturing cost per boom. The booms weigh only about one-quarter and cost about one-third to produce compared to their metallic counterparts [39].

A complementary composite boom deployer, developed in conjunction with the CTM boom from the DCB project, will also be employed. This deployer features a central hub around which all four booms are coiled for optimal packing efficiency [39]. The system uses a tape-spool mechanism, specifically designed to ensure the smooth and controlled deployment of the booms [31].

5.2.3. Attitude and orbit control system

As the precursor mission is designed to validate critical technologies for the future sunshade, its AOCS must demonstrate the ability to maintain accurate solar sail orientation in this dynamic and unstable environment. Furthermore, the mission will serve as a technology demonstration for the AOCS components, many of which have not flown in operational solar sails missions due to prior cancellations or mission losses. The data gathered will provide critical insights for refining these systems and advancing the development of the full-scale PSS.

Precise attitude control, with accuracy on the order of one degree, will be required to achieve the desired sail orientation for the targeted orbit (Fig. 2). Given the unstable orbital environment and the sensitivity of solar sail dynamics, the AOCS must ensure rapid responses to perturbations – such as variations in solar radiation pressure, gravitational disturbances, and thermal effects – and maintain stability during station-keeping and maneuvering operations. Additionally, resource efficiency is critical to ensuring extended mission operation. Since the spacecraft relies on solar radiation pressure as the primary means of control, minimizing propellant consumption is essential.

Reaction Wheels (RWs) are selected as the primary actuators for fine-pointing adjustments, providing high-precision control while minimizing power consumption and propellant use. However, external disturbances can induce perturbative torques along the yaw, pitch, and roll axes, which must be actively counteracted. To address this, a dedicated momentum management system is implemented, consisting of Active Mass Translators (AMT) to compensate for disturbances in the yaw and pitch axes and Reflective Control Devices (RCD) to regulate roll.

The Reflective Control Devices (RCDs), thin-film devices with Polymer Dispersed Liquid Crystal Diodes (PLCDs), control the reflectivity of the sail's surface [40,41]. By switching the devices ON and OFF, the RCDs generate varying forces. The reflectivity determines the magnitude of the force imparted by solar radiation pressure — the higher the reflectivity, the greater the force. Mounted near the booms at an angle to the sail plane, the RCD blocks generate a differential force by alternately switching devices ON and OFF, producing a roll torque.

The Active Mass Translator (AMT), designed to adjust the CoM relative to the center of pressure (CoP), provides precise control torques in both the pitch and yaw directions [42,43]. Its compact design, utilizes stepper motors, gearboxes, and lead screw drive systems, to achieve fine control, enabling very small and precise movements despite space constraints. The AMT is designed to be placed at the geometric center of the spacecraft, allowing for easy adjustment of the CoM by moving one part of the spacecraft relative to the other. By moving the bus components, it offsets the CoM relative to the CoP, creating control torque.

Both the RCDs and the AMT were specifically developed for solar sailing. RCDs have flown successfully once before [28], however the AMT missions never made it that far. Therefore, the precursor mission serves as a valuable testbed for their in-orbit validation.

Table 1
Subsystem mass estimates.

Subsystem	Mass [kg]
Sail & Boom	6–7
Bus structure	2–4
Thrusters & Propellant	1.5–2.5
Power (Batteries & Panels)	1.5–2.5
Communications	1.5–2.5
Attitude control	1–2
Data handling	0.5–1.5
Payload & Misc.	1–2
Total	15–24

A Reaction Control System (RCS), will be used for initial detumbling, sail deployment stabilization, periodic reaction wheels desaturation, and as a contingency system for recovery in case of anomalies. While not the primary actuator, its inclusion enhances mission reliability.

5.3. Mass, power, and data budget

The exact design of the precursor spacecraft is yet to be finalized, however there are certain key subsystems essential for mission success. This section evaluates the mass, power, and data requirements, largely based on comparable small spacecraft missions and available component data [44]. By using this preliminary assessment, an estimate of the systems required capabilities and constraints can be established, to ensure feasibility within the mission.

5.3.1. Mass budget

A mass budget was made to be able to estimate the final mass of the spacecraft, some key subsystems were chosen, such as structural components, power, communication, attitude control, and payload. The sail and boom masses were approximated based on deployable structure models [31], while other subsystem masses were derived from existing small spacecraft designs and available component data [44].

The subsystem masses were adapted from a similar mission [45], and adjusted for the mission requirements of the precursor, giving an estimated total spacecraft mass of 15–24 [kg], as shown in Table 1. Within this range of mass, it is possible to obtain the exact optimal, or anyway very close, value of $\beta = 0.0349$ as showed in Section 4.1. During the iterative design process, the total mass may vary within the identified range, resulting in different β value that remain close to the optimal one and with a Q varying from 0 to 1. With an assumed sail area of 400 [m^2], the areal density will be 38–60 [g/m^2]. The sail and boom subsystem makes up a lot of the total mass, roughly about 6 [kg]. The sail membrane itself is very lightweight and most of the mass comes from the sail and boom stowage and deployment mechanisms. For the communication system, antennas and transmitters were considered, with a final mass of about 2.5 [kg]. For the attitude control system, reaction wheels, sensors, and torque devices were included and estimated to weigh about 2 [kg]. The remaining mass is taken up by power storage, data handling, payload, and structural components. The preliminary spacecraft is made up of a 12U CubeSat, with a maximum mass of 24 [kg], and it is therefore assumed that the spacecraft mass will remain within the allowed limit.

5.3.2. Power budget

A power budget was also adapted from [45], to make sure the power generation and consumption stayed withing feasible limits. The spacecraft will rely on solar panels for power, which will be able to generate 150–180 [W] under optimal conditions. As variations in solar exposure and panel degradation will occur, the Orbit Average Power (OAP) [46] was calculated to 100–130 [W], giving a more realistic value of the power available.

Table 2
Power budget overview.

Subsystem	Peak power [W]
Communication	30–40
Attitude control	30–40
Data handling	5–10
Payload	5–10
Other systems	2–5
Total	72–105

Table 3
Data budget overview.

Subsystem	Data rate [kbps]
Telemetry	1–5
Science payload	50–200
Housekeeping	1–2
Total	52–207

The spacecraft's power demand have to cover continuous loads, such as data handling and attitude control, as well as communication and payload operations. The peak power requirement is calculated to be 100−110 [W], this value incorporates a 20% safety margin. For energy storage lithium-ion batteries will be used, with a total capacity of 70−80 [Wh], to provide power during eclipse periods (see Table 2).

5.3.3. Data budget

The spacecraft will generate and transmit data from telemetry, payload instruments, and system monitoring, making efficient data storage and downlink management essential. The data generation rate will vary depending on the operational phases of the mission. Telemetry and housekeeping require a low continuous data rate 1−5 [kbps], whereas payload data collection can produce bursts of 50−200 [kbps], depending on resolution and compression efficiency. An onboard storage capacity of 2−5 [GB] will allow for temporary data buffering before transmission.

The spacecraft will utilize an X-band or S-band downlink for data transmission, capable of 100 [kbps]–1 [Mbps], to ensure smooth data transfer during available ground station pass windows (see Table 3).

6. Risks and costs

There are several potential risks for the mission faces that could impact both the cost and feasibility. Financial risks are potential fluctuations in launch prices and component costs, and technical risks can arise when adapting existing technologies to fit the mission. Rideshare programs expose the mission to the possibilities of launch delays, and operational risks, such as subsystem failures, could greatly impact overall mission performance. However, with effective risk mitigation strategies, these risks could be minimized or completely avoided, to ensure a cost-efficient and reliable mission.

Using NASA's cost estimation method for small spacecraft missions [47], the estimated total cost of the precursor spacecraft is about $10 million, with a mass of 15−24 [kg]. The largest cost drivers are the launch and CubeSat platform expenses. The launch cost is estimated to about $1 million for GEO [48], with a rideshare launch being the most cost-effective option. Additionally, a 12U CubeSat platform with integrated hardware is expected to cost $400,000 [49].

7. Discussion

This study highlights the necessity of a planetary sunshade system for solar geoengineering and presents the preliminary design of a precursor mission to demonstrate key enabling technologies. The design process involved several trade-offs, balancing technical feasibility, performance requirements, and mission constraints. While the results provide a foundation for future development, some design choices remain subject to further refinement and validation.

One of the primary considerations in this study is the use of solar sail technology for propulsion, attitude control, and station-keeping at L_1^*. The selection of the reflective coating for the sail was primarily influenced by material degradation under prolonged UV exposure. While the chosen material meets the preliminary mission requirements, further studies are needed to quantify its degradation over time and explore potential mitigation strategies, such as advanced coatings or self-healing materials. Moreover, additional investigation and in-space testing are essential to assess how the degradation of optical properties affects the overall performance of the solar sail—particularly in terms of solar radiation pressure efficiency and control authority. These effects may impact both the trajectory and the stability of the spacecraft during long-duration operations. Additionally, the implications of sail deployment timing on mission dynamics warrant further investigation.

The choice of AOCS architecture played a critical role in the overall system design. The trade-offs made in this study were largely driven by current technological limitations, particularly regarding reaction wheels, which were selected as the primary actuators. However, advancements in alternative control methods, such as miniaturized control moment gyroscopes or novel torque management strategies, may offer enhanced reliability and performance in future iterations of the mission design.

Similarly, the deployment strategy for the solar sail was influenced by its impact on the spacecraft's attitude control. Future improvements in AOCS algorithms and hardware may allow for greater flexibility in deployment scenarios, reducing constraints on mission design. Furthermore, an in-depth assessment of the structural dynamics of flexible components, including the deployment booms and solar sail, with modal analysis will be essential to ensure pointing accuracy under varying environmental conditions.

Another key aspect requiring further investigation is the scalability of the proposed precursor mission. Transitioning from a 12U CubeSat demonstration to full-scale sunshade modules with larger solar sails will introduce new challenges related to formation flying, station-keeping strategies, and large-scale deployment mechanisms. This necessitates additional research to validate the feasibility of scaling up the proposed architecture while maintaining operational stability and effectiveness.

Finally, the trajectory design presented in this work is based on simplified assumptions regarding the transfer from Earth orbit to L_1^*. A more comprehensive mission analysis should explore alternative transfer strategies, considering factors such as deployment timing, propulsion efficiency, and station-keeping requirements in different orbital scenarios.

By addressing these challenges, this study contributes a critical step toward the realization of space-based solar geoengineering as a potential climate intervention strategy. While the proposed precursor mission provides a valuable testbed for key technologies, continued research and iterative design refinements will be essential to advance this concept from theoretical feasibility to practical implementation.

8. Conclusions

This paper has presented the preliminary design of a precursor mission to validate critical technologies for the future PSS. The selection of key demonstration objectives came from the evaluation of potential technology risks, the opportunity of leveraging on other large-scale space projects, and the unique advantages of in-orbit testing.

The mission focuses on evaluating critical technologies, including solar sailing for propulsion and station-keeping, deployable structures and mechanisms, and the AOCS. Additionally, it aims to assess the long-term performance and durability of optical shield materials in the L_1^* environment. These elements are considered high-risk due to their fundamental role in the mission's success. By addressing these challenges through early in-space demonstrations, the mission aims to accelerate technology maturation and establish a solid foundation for the development of the full-scale PSS, designed to reduce oncoming solar radiation and contribute to climate intervention efforts.

A critical aspect of the precursor mission is the selection of the target orbit. Positioned near the photo-gravitational equilibrium point L_1^*, approximately 2.36 million [km] from Earth, this orbit plays a crucial role in optimizing the stability and effectiveness of the future PSS. The control strategy exploits SRP to maintain the required α and δ angles, ensuring continuous alignment with the Sun–Earth conjunction line and guaranteeing consistent shading over time. This approach minimizes propellant consumption while maintaining precise orbital control.

The mission architecture is based on small-satellite technologies integrating advanced systems such as composite deployable booms and high-reflectivity coatings. The spacecraft will be a 12U CubeSat equipped with a 400 [m^2] solar sail and an estimated mass budget of 15–24 [kg]. A feasibility analysis indicates that the total mission cost is projected to be approximately 10M USD, with rideshare launch opportunities enhancing cost-effectiveness and mission viability.

By demonstrating these operational aspects in the L_1^* environment, the mission will provide essential data to refine and confirm the feasibility of the full-scale PSS, thereby contributing to the advancement of space-based climate intervention strategies. Future efforts will need to focus on refining formation-flying strategies, optimizing material durability for extended mission, and assessing the potential for modular and scalable deployments. Further research and development will be essential in advancing the PSS concept from early-stage demonstrations to practical implementation.

CRediT authorship contribution statement

Marina Coco: Writing – review & editing, Writing – original draft, Visualization, Methodology, Investigation, Formal analysis, Conceptualization. **Catello Leonardo Matonti:** Writing – review & editing, Writing – original draft, Visualization, Methodology, Investigation, Formal analysis, Conceptualization. **Chantal Cappelletti:** Writing – review & editing, Writing – original draft, Visualization, Methodology, Investigation, Formal analysis, Conceptualization. **Bruce Chesley:** Writing – review & editing, Writing – original draft, Visualization, Methodology, Investigation, Formal analysis, Conceptualization. **Christer Fuglesang:** Writing – review & editing, Writing – original draft, Visualization, Methodology, Investigation, Formal analysis, Conceptualization. **Giuseppe Governale:** Writing – review & editing, Writing – original draft, Visualization, Methodology, Investigation, Formal analysis, Conceptualization. **Nishanth Pushparaj:** Writing – review & editing, Writing – original draft, Visualization, Methodology, Investigation, Formal analysis, Conceptualization. **Marcello Romano:** Writing – review & editing, Writing – original draft, Visualization, Methodology, Investigation, Formal analysis, Conceptualization. **Gunnar Tibert:** Writing – review & editing, Writing – original draft, Visualization, Methodology, Investigation, Formal analysis, Conceptualization. **Lisa Wilk:** Writing – review & editing, Writing – original draft, Visualization, Methodology, Investigation, Formal analysis, Conceptualization.

Declaration of competing interest

The authors declare that they have no known competing financial interests or personal relationships that could have appeared to influence the work reported in this paper.

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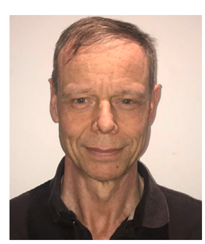
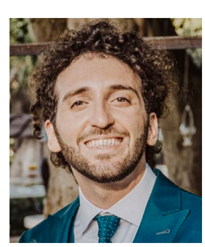
Catello Leonardo Matonti received his bachelor's and master's degree respectively in Aerospace Engineering and Space and Astronautics Engineering from Università di Roma: La Sapienza. Catello Leonardo attended the Sapienza's School for Advanced Studies, obtaining the SSAS's degree in Multidisciplinary Studies. Catello Leonardo was selected for a 6-month GNC Engineering Internship at ISAE-SUPAERO and for a 10-month Data Science Internship at INAF. In March 2022 he started working at Capgemini Engineering as a System Engineering Consultant until November of the same year, when he was selected as Ph.D. candidate in Aerospace Engineering at Politecnico di Torino.

Dr. Chantal Cappelletti leads the Institute of Aerospace Technology Space Group at the University of Nottingham and is a member of the Nottingham Geospatial Institute. Her research focuses on Space Systems Engineering and Small Satellites, with applications in Climate Change, AstroPharmacy, AstroMedicine, GNSS, IoT, and Disaster Monitoring. She was a pioneer on the use of new technologies such as additive manufacturing and femto-satellites. She emphasizes interdisciplinary cooperation, student involvement, and the long-term sustainability of satellite projects. She is a full member of the International Academy of Astronautics.

Bruce Chesley is a Senior Associate with Teaching Science and Technology, Inc (TSTI). He develops and delivers technical short courses and workshops to assist government agencies, start-ups, and established companies develop their workforce for the space domain. His professional career began in the US Air Force. He held assignments at Air Force Space Command, the National Reconnaissance Office, and the Department of Astronautics at the USAF Academy. He left the military in 2000 and spent the next 20 years in industry with Boeing Space and Launch. In addition to engineering and technology development roles, he held executive positions in program management, business development, and strategy.



Christer Fuglesang is an astronaut and professor at KTH Royal Institute of Technology, while also working part time as Space Advisor for Saab. He has a Ph.D. in experimental particle physics from Stockholm University and has worked for CERN and ESA. Fuglesang has made two space shuttle missions to ISS and performed there five EVAs. His research focus has been about particles in space and cosmic rays, and recently studying the possibility of putting sunshades in space to help control global temperature. He teaches a course in Human Spaceflight.



Dr. Giuseppe Governale is a Postdoc at Politecnico di Torino, working on space systems engineering for innovative space concepts since 2018. He earned his Ph.D. in Aerospace Engineering in November 2023 developing cost estimation tools for systems such as human lunar landers and microlaunchers. He contributed to the EFESTO and EFESTO-2 European projects on inflatable heatshield solutions. He completed the international Space Exploration and Development Systems Specializing Master, where he now serves as a professor and academic tutor. In academia, he has taught courses in space management and utilization, space exploration and resources, orbital robotics and systems engineering.

Dr. Nishanth Pushparaj is an Assistant Professor in Aerospace Engineering at the University of Nottingham and a member of the Nottingham Geospatial Institute. His research focuses on Astrodynamics, Spacecraft Mission Design, and Trajectory Optimization, with applications in Interplanetary Multibody Systems, Small Satellites, and Deep-Space Exploration. He has contributed to JAXA missions like MMX and DESTINY+. He is also a Fellow of the UK Higher Education Academy and an associate member of the Royal Aeronautical Society

Prof. Marcello Romano is heading the Institute of Astrodynamics of the Technical University of Munich (TUM), Germany. From 2004 to 2024 he was a professor and founding director of the Spacecraft Robotics Lab at the Naval Postgraduate School, Monterey, California. Dr. Romano obtained his Ph.D. (2000) and M.Sc. (1997) in Aerospace Engineering from Politecnico di Milano, Italy. His research interests are in the dynamics, guidance and control of advanced autonomous space systems, orbital space robotics, and space systems engineering.

Gunnar Tibert is associate professor in space technology and specializing in the mechanics and dynamics of structures. He has designed, analyzed, manufactured and qualified deployable structures for sounding and space missions.

Lisa Wilk is a researcher assistant in aerospace engineering to Christer Fuglesang. She completed her Master's in Aerospace and Mechanical Engineering at KTH Royal Institute of Technology and made her master thesis on a solar sailing precursor spacecraft. She now works as a test engineer for an automotive company in the south of Sweden.