

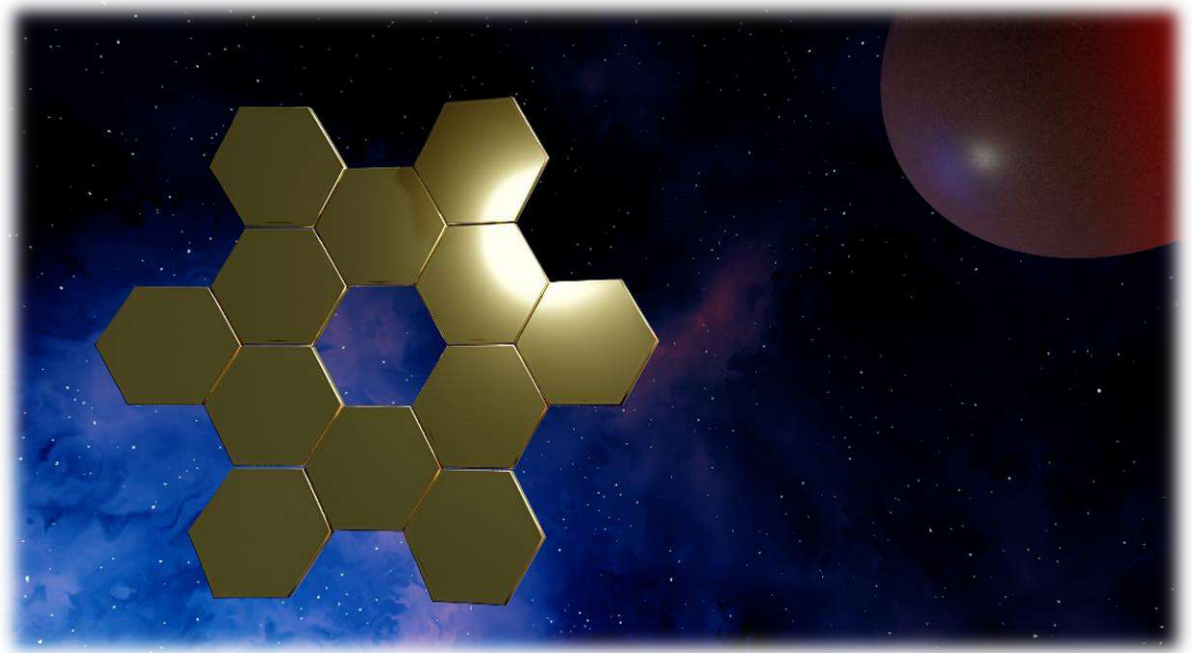
Concept design of a new space telescope with deployable composite booms

Student project

Course: *"Strutture per veicoli spaziali"*

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1 STATE OF THE ART OF TELESCOPES WITH TAPE SPRING-BASED SECONDARY MIRRORS AND ACTIVE CORRECTION CAPABILITIES

1.1 SPACE TELESCOPES

Telescopes have a very long history. Since ancient times, man has observed the sky with curiosity, this has led to the development of techniques and objects that can help him observe all the visible. The first objects built were small in size and were not mainly used for astronomical purposes, but above all as useful tools in the military and naval fields.

The first object used for astronomy came in 1608, when the Dutchman Hans Lippershey built the first model of a refracting telescope, an instrument that the following year was perfected by Galileo Galilei. With this he made numerous astronomical discoveries, including that of the four major satellites of Jupiter. In 1611, the *Accademia dei Lincei* gave the name of *telescopium* to Galilei's telescope, which he called spectacles.

Thanks to new knowledge of optics, lens processing techniques improved; later developments aimed at achieving better brightness, with the construction of larger lenses, and a better angular scale, with increasing focal length.

Already at the end of the 1970s it was clear that visible observation with classical ground-based telescopes had reached its limits. Atmospheric noise prevents ever larger telescopes from producing proportionately more distinct images. This is how the first space telescopes were born, which have the advantage of being able to explore regions of the electromagnetic spectrum otherwise inaccessible from Earth, such as the UV band and the near IR.

Thus, was born the idea of developing satellites or probes to be launched into space with the express purpose of observing planets, stars, galaxies and other celestial objects.

Since 1979, companies and space agencies have been designing, building and launching dozens of space telescopes into space, including in 1990 the Hubble Space Telescope (HST) in a low orbit at 600 km altitude, periodically serviced by astronauts for installation of increasingly advanced tools. The HST is characterized by the high quality of the images, at the theoretical limit of diffraction for the correction of spherical aberration.

In 2021, the launch of HST's successor, the James Webb Space Telescope (JWST), is expected.

The very high cost of space telescopes makes it necessary to use ground-based telescopes for most of the observations, but the discoveries of these instruments in space orbit have contributed enormously to our knowledge of the cosmos.

1.1.1 Space Telescope's Traditional Configuration

A space telescope is a telescope located in outer space to observe distant planets, galaxies and other astronomical objects. Many space telescopes have been launched to increase our knowledge of the universe. The aim is to build a telescope that can produce high resolution images of astronomical objects.

A space telescope is not disturbed by the artificial light sources or by the Earth atmosphere which causes the optical aberration and the distortion of images. In addition, the atmosphere absorbs a large portion of the electromagnetic field, stopping in this way the radiation of many celestial bodies. Due to this fact, lots of observations are impossible from Earth and they have to be made in space, outside the atmosphere. On the other hand, space telescopes are very expensive and, due to their location, extremely difficult to maintain.

Comparing all types of space telescopes, it is possible to portray a traditional configuration and list the typical systems installed:

- *Optical Telescope Element*: every telescope that works in visible spectrum has mirrors, made of special glass, which collect light from the targets and reflect it into the instruments. In front of the optics there is an “aperture door” which can be closed every time the telescope is not working and, if necessary, to prevent Sunlight from entering and potentially damaging the telescope or its instruments.
- *Scientific Instruments*: cameras and spectrographs. Cameras capture framed images and radiations emitted from space bodies. There are lots of types of cameras depending on the radiation that they have to measure. On the other hand, spectrographs break light into colours to analyse its physical features (temperature, mass and chemical composition).
- *Command and Data Handling*: is the telescope’s brain. It devises telemetry and dispose a state solid recorder (SSR) for the observation data. It controls the interaction between the on-board computer, the scientific instruments and the communication system.
- *Communication System*: the digital images and data stored in solid-state recorders are converted into radio waves and then beamed through one of the spacecraft’s high-gain antennas (HGAs) to a communications satellite, which relays them to the ground.
- *Solar panels*: sets of gallium-arsenide photovoltaic cells that produce enough power for all the science instruments to operate simultaneously. The electric system converts the light captured by the solar panels into electric energy.
- *Thermal Shield*: helps the telescope to passively cool down its own temperature to a stable value. This is necessary to allow the correct function of the instruments and the right alignment when the telescope changes its orientation relatively to the Sun. Usually the layers are covered with aluminium and silicon due their optical, conductive and durable proprieties in the space environments. The silicon high emissivity disperses a huge part of the Sun heat and preserves the instruments on board. Moreover, the highly reflected aluminium surface disperses the residual energy to the edge of the thermal shield.
- *FGS (Fine Guidance Sensor)*: is the system used for manoeuvring the telescope. It helps pointing in the right direction the objects in the space during the observation. The Fine Guidance Sensors (FGSs) not only help the telescope stay locked on target but can also be used as science instruments to accurately determine the relative position of stars. They are devices that lock onto guide stars and so keep the telescope pointed in the correct direction. The FGS can determine the positions of stars to a precision 10 times greater than ground-based observatories.
- *Support systems*: Essential support systems such as batteries, gyroscopes, reaction wheels and electronics.

1.1.2 Advantages and Disadvantages of Space Telescopes

Developing, building and using an orbiting telescope involve a wide range of aspect to consider as they have considerable advantages and disadvantages. There are several points to discuss above.

Clearer observations. Orbiting telescopes have no atmosphere to tower over. This is the main reason to use orbiting telescopes: they have a degree of clarity that is still largely unmatched. These telescopes are able to capture detail in a truly impressive fashion, as a matter of fact the best shots of the universe that we have are from orbiting telescopes.

Wider wavelengths of EM spectrum. EM radiation from space is unable to reach the surface of the Earth except at very few wavelengths, such as the visible spectrum, radio frequencies, and some ultraviolet wavelengths: the atmosphere affects incoming radiation so that some phenomena cannot be observed at all from Earth's surface or cannot be observed with enough accuracy. That is one of the reasons to send space telescopes in orbit.

No limitations to observing at night. As on the Earth there is night and day, and the telescopes work only in the dark, there is a huge limit of time, also given by the weather condition. In space there are not this problem, and the observation might be interrupted only by communication between ground and the telescope,

Cost. As space structure, orbiting telescopes are exceptionally expensive to build and position in place. Estimate a cost for each mission is always complicated for the unpredictability of space environment: radiation, stability, temperature, communication are problems only for orbiting telescopes and not for ground telescopes. Each of these aspects means an additional cost for special systems that balance this phenomenon. Furthermore, the costs are increased by the transport of the telescopes as payload: space vectors are the only (expensive) way to the orbits, although in the last period they are cheaper for the privatization of space flight.

Stability. Various phenomena in space disturb a telescope's equilibrium: gyroscopes and motors that move the mirrors into place need updating constantly to maintain the alignment with the mission's subject, and also when the communication with the ground isn't possible, there must be a self-controller on-board to guarantee the stability.

Maintenance and Updating. The special position of space telescopes is also one of their biggest problems. Since they can't be reached normally, the instruments can't update as much as you would like, and the maintenance could be performed only by special expeditions: the famous ones were the expeditions for the Hubble Space Telescope that needed five servicing missions by NASA space shuttles (from 1993 to 2009).

1.1.3 Missions

The mission types of active and in development space telescopes are classified as follows.

Aeronomy. A branch of atmospheric physics. Aeronomy is the meteorological science of the upper region of the Earth's or other planetary atmospheres, which studies the atmospheric motions, its chemical composition and properties, and the reaction to it from the space environment.

Asteroid detection. Mission based on the discovery of asteroids and comets, and on the determination of their orbits.

Astrometry. The branch of astronomy that requires precise measurements of the movements and positions of stars and other celestial bodies. Astrometry studies the kinematics and physical origin of the Solar System and the Milky Way.

Astronomy. A natural science that studies everything that originates outside Earth's atmosphere, in particular celestial objects and phenomena, in order to explain their origin and evolution. Objects of interest include planets, moons, stars, nebulae, galaxies, and comets. Relevant phenomena

include supernova explosions, gamma ray bursts, quasars, blazars, pulsars, and cosmic microwave background radiation. The observatories used for this kind of mission can be distinguished according to their sensitivity to the wavelengths of the electromagnetic spectrum:

- **Multi-wavelength.** The *Hubble Space Telescope* is one of the most versatile space telescopes; its four main instruments observe in the ultraviolet, visible, and near-infrared regions of the electromagnetic spectrum
- **Gamma-ray astronomy.** Gamma-ray astronomy studies astronomical objects and phenomena that emit gamma rays, the most energetic form of electromagnetic radiation with photon energies above 100 keV.
- **Infrared astronomy.** Infrared astronomy studies astronomical objects visible in infrared (IR) radiation, with wavelength from 0.65 to 450 micrometres.
- **Ultraviolet astronomy.** Ultraviolet astronomy is the observation of electromagnetic radiation at ultraviolet wavelengths between approximately 10 and 320 nanometres.
- **Visible-light astronomy.** Visible-light astronomy is restricted to wavelength range from 380 to 750 nanometres.
- **X-ray astronomy.** X-ray astronomy studies celestial objects emitting radiation from 0.008 to 8 nanometers.

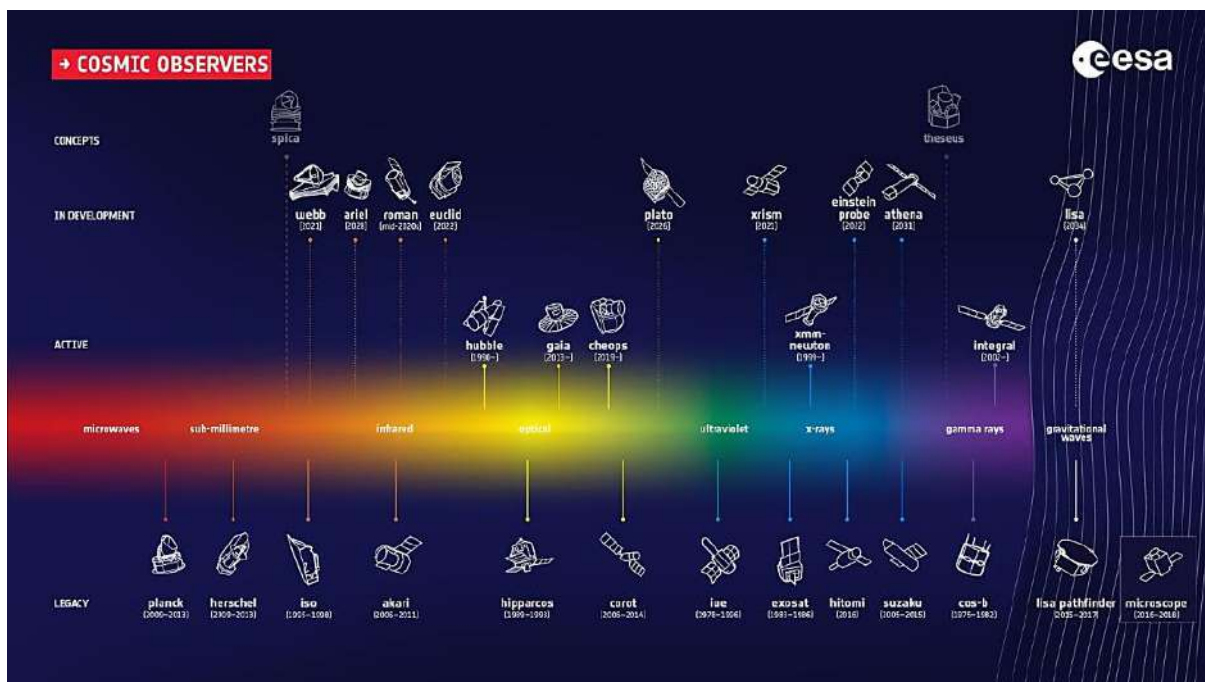


Figure 1.1. Space telescopes and relative missions

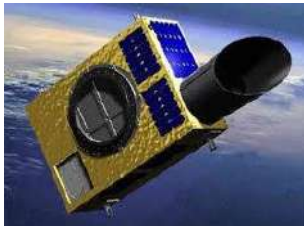
Exoplanetology, or exoplanetary science, is an integrated field of astronomical science dedicated to the search for and study of exoplanets (extrasolar planets).

Heliophysics. The science of the Sun. The physical connections between the Sun and the solar system are studied in this type of mission.

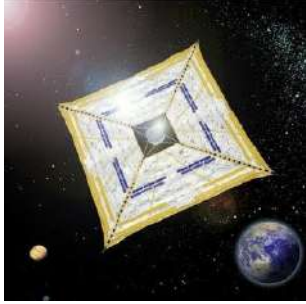
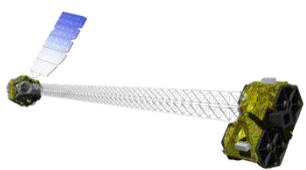
Particle detection (High energy astronomy). Astronomical objects that release electromagnetic radiation of highly energetic wavelengths, including X-ray, gamma-ray, extreme UV, neutrinos and cosmic rays, are studied.

1.1.4 Active Space Telescopes

Some of the currently active telescopes are briefly described in the following table:

Hubble Space Telescope Space Agency: NASA & ESA Launch date: 24 Apr 1990 Location: Earth orbit (586,47-610,44 km) Mission type: Astronomy		It's a long-term, space-based observatory. The observations are carried out in visible, infrared and ultraviolet light. Hubble can detect light with 'eyes' 5 times sharper than the best ground-based telescopes and looks deep into space.
Odin Space Agency: Swedish Space Corporation Launch date: 20 Feb 2001 Location: Earth orbit (622 km) Mission type: Astronomy and aeronomy		The Odin mission objectives are observation of stratospheric ozone chemistry, mesospheric ozone science, summer mesospheric science, coupling of atmospheric regions.
NEOSSat (NEOSSat Near Earth Object Surveillance Satellite) Space Agency: CSA & DRDC Launch date: 25 Feb 2013 Location: Sun-synchronous Earth orbit Mission type: Asteroid detection		It's capable to discover NEOs (Near-Earth Objects), particularly difficult to detect from the surface of the Earth, by searching the sky as close to the Sun.
GAIA Space Agency: ESA Launch date: 19 Dec 2013 Location: Sun-Earth L2 Mission type: Astrometry		Gaia is a mission to chart a three-dimensional map of our Galaxy, the Milky Way, in the process revealing the composition, formation and evolution of the Galaxy.
CHEOPS (CHAracterising ExOPlanets Satellite) Space Agency: ESA Launch date: 18 Dec 2019 Location: Sun-synchronous orbit Mission type: Exoplanetology		It's a European space telescope to determine the size of known extrasolar planets, which will allow the estimation of their mass, density, composition and their formation.

IRIS (Interface Region Imaging Spectrograph) Space Agency: NASA Launch date: 27 Jun 2013 Location: Earth orbit Mission type: Heliophysics		It's a NASA satellite used to investigate the physical conditions of the solar limb, particularly the chromosphere of the Sun. Tracking how material and energy move through this region is a crucial part of understanding the dynamics of the sun.
DAMPE Space Agency: CNSA & CAS Launch date: 17 Dec 2015 Location: Earth orbit (500 km) Mission type: Particle detection		It's used to search and study dark matter particles, the origin, the propagation and acceleration mechanisms of cosmic rays.
INTEGRAL (International Gamma Ray Astrophysics Laboratory) Space Agency: ESA Launch date: 17 Oct 2002 Location: Earth Orbit (639-153,000 km) Mission type: Astronomy		INTEGRAL is detecting some of the most energetic radiation that comes from space. It is design to provide imagine and spectroscopy of cosmic source. In the MeV is the most sensitive gamma ray observatory in space.
Neil Gehrels Swift Observatory Space Agency: NASA Launch date: 20 Nov 2004 Location: Earth Orbit (585-604 km) Mission type: Gamma-ray astronomy		Swift is a NASA mission dedicated to detecting Gamma-ray bursts, the most powerful explosions the Universe has seen since the Big Bang. They occur approximately once per day and are brief, but intense, flashes of gamma radiation.
AGILE Space Agency: ISA Launch date: 23 Apr 2007 Location: Earth Orbit (524-553 km) Mission type: Gamma-ray astronomy		AGILE is an X-ray and Gamma ray astronomical satellite of the Italian Space Agency. Its mission is to observe g-ray sources in the universe. AGILE has the ability to locate g sources with excellent resolution and to analyse data quickly.

FERMI – Gamma-ray Space Telescope Space Agency: NASA Launch date: 11 Jun 2008 Location: Earth Orbit (555 km) Mission type: Gamma-ray astronomy		Fermi is a space observatory that enables scientists to answer persistent questions across a broad range of topics, including supermassive black - hole systems, pulsars, the origin of cosmic rays, and searches for signals of new physics.
GAP – Gamma-ray Burst Polimeter Space Agency: JAXA Launch date: 21 May 2010 Location: Heliocentric orbit Mission type: Gamma-ray astronomy		The Gamma-Ray Burst Polarimeter (GAP) aboard IKAROS is the first polarimeter specifically designed to measure the polarization of Gamma-Ray Bursts (GRBs) from space and will do so in the cruising phase of the IKAROS mission.
Chandra X-ray Observatory Space Agency: NASA Launch date: 20 Nov 2004 Location: Earth orbit (9,942-140 km) Mission type: X-ray astronomy		NASA's Chandra X-ray Observatory is a telescope specially designed to detect X-ray emission from very hot regions of the Universe such as exploded stars, clusters of galaxies, and matter around black holes.
NuSTAR (Nuclear Spectroscopic Telescope Array) Space Agency: NASA Launch date: 13 Jun 2012 Location: Earth orbit (603,5 km) Mission type: X-ray astronomy		NuSTAR is taking a census of collapsed stars and black holes of different sizes, to map recently synthesized material in young supernova remnants and to understand what powers relativistic jets of particles from active galaxies hosting supermassive black holes.
SPRINT-A Hisaki Space Agency: JAXA Launch date: 14 Sep 2013 Location: Earth orbit Mission type: Ultraviolet astronomy		Extreme ultraviolet spectroscope for Exospheric Dynamics and it is the world's first space telescope for remote observation of the planets from the orbit around the earth.

AMS-02 (Alpha Magnetic Spectrometer) Space Agency: NASA Launch date: 16 May 2011 Location: Earth orbit (353 km) Mission type: Cosmic-ray astronomy		Particle physics experiment module that is mounted on the ISS. It is a detector that measures antimatter in cosmic rays; this information is needed to understand the formation of the Universe and search for evidence of dark matter.
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1.2 DEPLOYABLE TELESCOPES

Small satellites offer a good chance for increased instrument presence in space as well as for tactical quick turn missions. However, small satellites fall short in optical performance due to their relatively small aperture size. With small aperture optical sensors in space, diffraction is an important obstacle for imaging resolution.

In addition to the assumption of diffraction limit, there is a problem related to the Signal to Noise Ratio: grainy images and insufficient contrast, caused by a lack of signal or too much noise, which are common manifestations of this phenomena.

The opto-mechanical deployable telescopes can overcome diffraction limits and signal to noise limitations with improved optical performance.

One aspect of the present disclosure relates to an opto-mechanical deployable telescope that includes a hub, at least one deployable multiple petal primary mirror mounted on it, a deployment mechanism, a deployment engine part, and a secondary optical structure. The deployment mechanism is designed to move at least one primary mirror between a stowed position and a deployed one. The deployment engine part is useful to power the deployment mechanism using stored mechanical energy. The secondary optical structure includes a secondary optical element. The secondary optical structure is mounted to the hub and intended to axially position the secondary optical element relative to the primary mirror. The deployment mechanism also includes a kinematic or semi-kinematic interface between the hub and the primary mirror to hold its petals in relative alignment to each other in the deployed position. Each petal of the primary mirror is individually attached to the hub with a hinge connection. In some examples, the secondary optical structure may include a stationary base mounted to the hub, and an extension element mounted to the stationary base, wherein the secondary optical element is placed, by the extension element, axially relative to the primary mirror to optimize optical imaging on a focal plane. The telescope may include a multitude of locating elements placed between the stationary base and the extension element, whereas the variety of locating elements provide translational and rotational alignment between the stationary base and the extension element. The telescope may include at least one magnetic component usable to apply a magnetic connection between the stationary base and the extension element. The telescope may include a guide rod coupled between these two, in which the guide rod provides constrained movement of the extension to the stationary base to relative axial movement.

The secondary optical structure includes a secondary optical element and it is mounted to the hub and operable to axially position the secondary optical member relative to the primary mirror. Each petal of the primary mirror is singly attached to the hub with a hinge connection. The hinge connection

provides radial movement of the plurality of petals when in the deployed position that allows the deployment mechanism rather than the hinge connection to control alignment of the multitude of petals.

The method of deploying opto-mechanical telescopes includes providing a hub, at least one primary mirror mounted to the hub, a deployment mechanism, and a secondary optical structure mounted to the hub and including a secondary optical element, in which the primary mirror includes a multitude of petals. The method includes moving multiple petals from a stowed position to a deployed one with the deployment mechanism, aligning their relative position to each other while moving them into the deployed position. To align the plurality of petals is necessary providing a semi-kinematic interface between them and the hub and operating the secondary optical structure to adjust the axial position of the secondary optical element relative to the primary mirror.

The plurality of alignment members may include a guide cylinder, a guide rod extending into the guide cylinder, a locating surface, a variety of locating elements configured to interface with the locating surface, and at least one magnetic member operable to apply a magnetic force to draw the extension member toward the stationary base.

The engines of the engine part may consist of at least one biasing element, as a compression, extension or coil spring, and they can be actuated automatically by releasing the primary mirror.

Other types of stored energy engines (also referred to as stored energy motors or power devices) may be used to deploy various features of telescope. For example, individual devices driven by actuation of a shape memory material, such as a shape memory alloy, may be used to deploy petals of the primary mirror.

1.2.1 Tape Spring Hexapod for Deployable Telescope Dynamics

A hexapod based telescope concept whose legs are deployable has been investigated in order to stow the secondary mirror during launch and to self-deploy it in orbit. The positioning errors are corrected thanks to the vertical displacement of the hexapod feet and the final optical performance is achieved thanks to adaptive optics.

The main advantages of a deployable and active telescope structure, compared to classical hyper-stable telescope structures, are:

1. A reduced volume during launch allowing large systems design;
2. A very strong reduction of dimensional stability requirements during launch as the geometric instabilities of the structure after deployment will be corrected;
3. A strong mass and inertia reduction which leads to an agility gain of the system.

However, once in orbit, these lightweight structures will have to insure a very high stability of mirrors and this induces new design considerations. Tape-springs have the advantage of being low cost and easy-to-use. But, overall, such steel thin curved strips offer a very simple concept with elastic behaviour and a natural tendency to put themselves in their deployed configuration. Deployment of a rigid-panel, Collapsible Rib-Tensioned Surface and high-resolution deployable telescope are some applications using the folding of tape-springs in order to make simple self-locking hinges. Several joints have also been developed like tape-spring rolling hinges, Aerospatiale “Adele” patented hinge, and Astro / JPL NASA Hinge. Another way to use it is based on the coiling of the strip instead of folding with the same principle as carpenter’s tape-measure like thin-walled tubular booms and deployment cassette for biSTEM. In such mechanisms, the cross-section forms a complete circle with some overlap. In the dealt application, innovation lies in the fact that deployment of the hexapod is provided by using

six specific linear actuators in which tape-springs are coiled around a hub. Such coiling tape-spring mechanisms have already been applied to robotic science.

About the deployment concept, the structure is compacted thanks to 6 legs made with tape-springs. The M2 mirror is stowed by a mechanism which withstands major part of the vibration loads without any strong dimensional requirement. In this configuration, the tape-springs which are coiled around a hub offer a strong volume reduction.

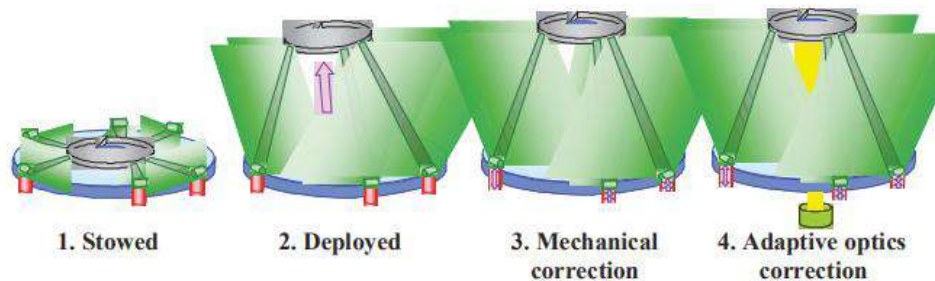


Figure 1.2. Different steps of deployment

Once in operational orbit, the stowing mechanism is released, and the 6 legs autonomously reach their full-length configuration bringing the M2 mirror from the stowing point to a deployed point defined by the final length of the 6 legs and by the structure geometrical and mechanical defects. After deployment, the 6 vertical actuators located under the hexapod feet can be used to compensate for deployment errors that are evaluated thanks to a dedicated measuring system. This stage of mechanical correction of the structure can also be used to correct the long-term instabilities and thermo-elastic deformations. Once corrected, the structure allows the acquisition of images. Even if their quality is reduced, it is sufficient to use adaptive optics which then allows to reach the full performance of the instrument. Tape-spring actuators contain a rotating roll module in which the tape-spring is flattened and coiled. Because of this induced stress, the tape-spring naturally tends to go out to recover its natural curved section. Inside the roll module, a fixed axis is connected to the cradle by a ball bearing. The coiling device has an internal coil-spring which helps the self-deployment when the locking system is released. The active wrist concept used for this hexapod implies the use of one spherical joint at the top and one universal joint at the base of each leg which means that 2 DOF are required at the base of the tape-spring and 3 DOF at its top. For the base junction, a hinge made of two ball bearings and the coiling of the tape-spring have been used as two degrees of freedom in rotation. The top junction is constituted with an axis mounted on two ball bearings, a thin metallic flexural blade and with the torsion of the tape-spring itself. The hexapod is composed of a baseplate on which 6 linear stages are mounted on 90° brackets and a platform representing the M2 mirror held by the 6 tape-springs. The linear stages support the bottom junctions and allow vertical displacements of the coiling mechanisms on a 16 mm stroke with a 20 μm resolution. The intersection of the rotational axes and the hinge axes at the top and base junction have been precisely placed at the positions computed by the optimal design algorithm.

1.2.2 Halca

Halca stands for **Highly Advanced Laboratory for Communications and Astronomy**, but was also known with the name VSOP (VLBI¹ Space Observatory Programme). It was a radio telescope satellite manufactured by JAXA, dedicated only to VLBI missions. It was the first space-borne telescope built for this only purpose.

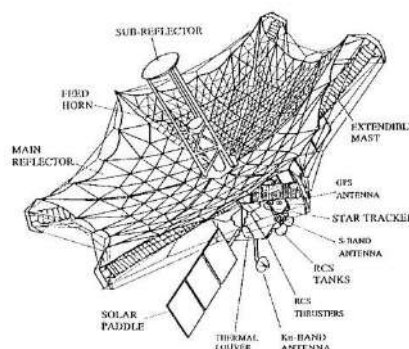


Figure 1.3. HALCA telescope

About the mission:

- *Objective:* to conduct radio-wave observation in orbit in combination with radio telescopes on the ground and various engineering demonstrations such as the deployment technology of a large antenna, highly stable data-transmission technology to compare atomic clock on-board with one on the ground and precise attitude control.
- *Launch date and location:* 13:50, February 12, 1997 (JST); Kagoshima Space Center (Uchinoura)
- *Launch vehicle:* M-V-1 solid fuel rocket
- *Weight:* 830 kg
- *Dimensions:* Rectangular base of 1.5m x 1.5m x 1m and large deployable antenna with maximum diameter 10m (effective 8m) and accuracy of 0.4 mm
- *Orbit altitude:* Perigee 560 km, Apogee 21000 km
- *Orbit inclination, type and period:* 31°, elliptical, approx. 6 h and 2 min.
- *Scientific instruments:* deployable antenna made of a mesh of 6000 cables and a mirror. Radio waves are led into the 2.5 m-long feed horn by two reflection mirrors. Space-qualified low-noise amplifiers in cryogenically cooled refrigerators. GPS/Galileo signal receiver, accelerometers and laser-pointing systems.
- *End of operations:* November 30, 2005
- *Operation:* orbital control performed three times two, four and nine days later the launch, following the establishment of three-axis attitude control. Two weeks from the launch the full deployment of the antenna was completed. Some check-ups were made before the start of VLBI operations, such as the establishment of interactive communication link with the tracking station.
- *Frequency bands:* 1.60/1.73GHz, 4.7/5.0GHz, and 22.0/22.3GHz. The sensitivity of the latter dropped drastically because of launch vibrations.
- *Results:* three years lifetime was estimated, but the satellite kept working for eight years and nine months after the launch. The wide aperture of the virtual telescope obtained thanks to the deployment technique (30000 km) made it possible to observe quasar and celestial bodies with a resolution of 2/10000 arcsec.

The main challenge of this scientific experiment was the antenna, because the high spatial and spectral resolution could be reached only with an antenna of this kind on diameter, but the M-V-1 rocket used for launch wasn't big enough to contain it. That's why engineers came up with the idea of building it with a deployable mechanism in order to fit the launch vehicle and, at the same time, reach the desired dimensions.

The project was such a success that a similar one was proposed, VSOP-2, with the intentions of making a better performing (that implies bigger too) antenna to observe the deep space bodies. For this reason, Astro-G was developed, a telescope very similar to HALCA, confirming the benefit obtainable by a deployable antenna for space purposes.

1.2.3 DISCIT

In Fiscal Year (FY) 2016, Optical Systems and Technology entailed a total of twelve projects, including computational imaging, LADAR technologies, free-space communication technologies and space surveillance capabilities. The portfolio included novel optical technology developments applicable across a broad range of Department of Defense's (DoD) problems. A Digital Coherent Aperture Combiner was developed that performs adaptive optics in the digital domain to synthesize a much larger aperture receiver to reduce the size and cost of communication ground terminals while improving resistance to signal fading.

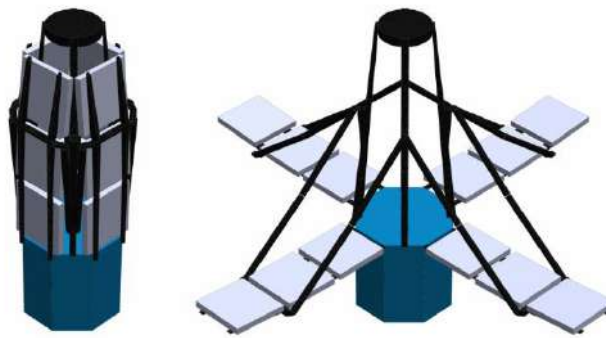


Figure 1.4. Technology developed for DISCIT folded up into the compact shape (left) and expanded into its segmented sparse sub-apertures (right) once deployed.

Moreover, researchers at Lincoln Laboratory, in agreement with MIT, developed technologies for a sparse-aperture telescope, referred to as the Deployable In-Space Coherent Imaging Telescope (DISCIT). DISCIT, if integrated on a small satellite, could provide high-resolution images from a spacecraft that is more affordable to build than one that can carry a large telescope, in order to provide transformational advantages to the DoD. This telescope had the goal of demonstrating a 70 cm telescope with single fold composite hinges and using target in the loop image compensation to achieve image quality comparable to a full-aperture telescope, as well as addressing multiple hinge technology for scaling to larger (multiple meters) apertures. Some of the technologies developed for DISCIT could also be used for creating small, lightweight satellite payloads used in other applications, such as environmental monitoring, communications, or radar. They leveraged recent advances in deployable structures to enable large structures to fold and to be much lighter and smaller than structures designed for earlier satellite payloads. These kinds of structures then deploy to full size after the satellite is in orbit. They used high-strain composite (HSC) materials that allow for structural members to fold and deploy without traditional bearing hinges. Their laboratory demonstrations have shown that DISCIT can be deployed with both precise (micron) positioning and (microradian) angular repeatability, indicating that DISCIT technology may enable the integration of high-performance telescopes on small satellites. Their current work focuses on precision measurements of the deployed shape of larger structures built with many HSC hinges and on the parallel actuation of many actuators for coordinated final shape control. Another key challenge for them in building DISCIT is minimizing the size, weight, and power requirements of the actuators needed for the final fine phasing of the optical components.

1.2.4 Low cost Deployable Telescope for Small Satellite

The emerging demand for low cost, high resolution, regular imagery and the arrival of the '*NewSpace*' or '*Space 2.0*' era has spurred the development of a number of different nano/micro/small satellites, all aiming to quench the thirst for data. There are some companies proposing vast constellations of CubeSats, others aiming at more capable >100kg platforms, and everything in between. The customer base has a strong knowledge and influence on performance and interest, not just in low GSD numbers, but also in high MTF across the range of spatial frequencies. Therefore, the challenge set to the satellite manufacturers, fledgling and experienced, is conceived on quality and quantity. To satisfy the requests of putting in orbit many satellites in one launch, a collapsible Cassegrain-type telescope is designed.

A traditional Cassegrain-type telescope has a large amount of unused volume between the primary mirror and the secondary mirror. The principal aim of this telescope is to reduce this volume during launch.

The secondary mirror can be held in a stowed configuration near the primary mirror and then deployed to a near-nominal position once in orbit; further fine alignment can then be performed.

The deployment mechanism consists of three concentric carbon-fibre barrels. The lower most and narrowest is mounted to a bulkhead which interfaces to the spacecraft. The middle and upper barrel are deployed post-launch simultaneously by lead screws spaced at 120° around the circumference, driven by a ring gear and redundant motor arrangement in the base.

Once is deployed some adjustment of the secondary mirror alignment will be required. A three-armed spider will hold the secondary mirror, with each arm having a linear actuator driving it in the z-direction (piston). When operated together, this provides piston of the secondary mirror, and when driven individually, it provides tip and tilt; it is therefore a 3-axis system.



Figure 1.5. Deployable telescopic barrel, stowed (left) and deployed (right)

1.2.5 FalconSAT-7

FalconSAT-7 (FS-7) is a 3U CubeSat with a deployable diffractive membrane solar telescope. The purpose of the mission is to demonstrate a deployable telescope with an aperture significantly larger than the spacecraft structure. The primary element deploys from one end of FS-7 and has a clear aperture of 20 cm, twice the cross-section of the host spacecraft structure. This novel payload is made possible by use of a thin (28 μm) membrane optic using diffractive principles to focus H-alpha light from the Sun onto an onboard camera. The diffractive optic is deployed using a set of spring-loaded pantographs that tension the membrane and hold it flat.

About the mission, the FS-7 program is targeting five key mission success criteria: first and foremost the goal of FS-7 is education; the second mission goal is to demonstrate the ability to deploy the

photon sieve, to collect and transmit to the ground at least one image of the Sun; to optimize and characterize image performance; to demonstrate flight heritage of the polyimide photon sieve material.

Regarding the FS-7 Mechanism, its deployment system is based on a two-stage, spring-loaded deployment mechanism. In the first stage of the deployment system, the assembly moves linearly out of the bus. The second stage is held in place using three sequencing bars. Once the bottom of the assembly clears the top of the bus, the sequencing bars all disengage and allow the second stage of the deployment. During the second stage, three precision pantograph arms extend to their final position, putting tension on the photon sieve and ensuring high-quality imagery. The deployment system was designed and built by MMA Design LLC in Boulder, Colo., in early 2012. During the summer of 2012, an engineering model was subjected to dozens of deployments in a 1-G environment. In August 2012, the engineering model was further tested on NASA's G-Force One aircraft. On this flight, the FS-7 team demonstrated six successful micro-gravity deployments, giving us confidence that the deployment system would work on orbit as well as in 1-G.

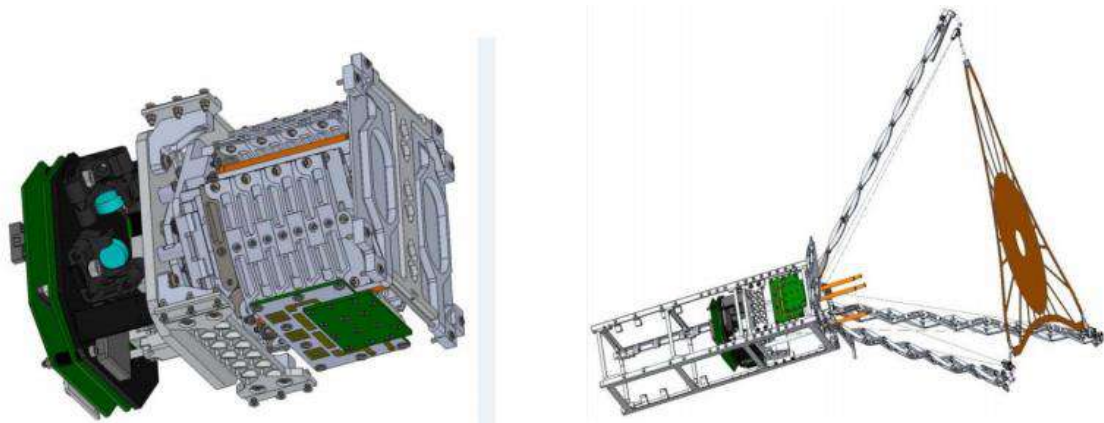


Figure 1.6. Stowed (1.5U) and deployed Peregrine payload schematics

While about the optical structure, the Peregrine membrane primary is an $f/2$, 20-cm diameter photon sieve. The Peregrine photon sieve consists of more than 2.5 billion cylindrical bumps etched from a 28-micron thick Kapton sheet. The bumps vary in diameter from 2 to 277 μm with the largest near the center. In the case of FS-7, the diffractive optic is a phase-type with a 50-percent fill factor, having a 38-percent theoretical focusing efficiency. The secondary optics of the photon sieve telescope collimate the focused beam from the photon sieve for transmission through a narrowband H-alpha filter and provide a magnification of around 6. Reflecting through three-fold mirrors, the beam illuminates a 10-bit monochrome CCD camera. The secondary elements are mounted to an electronically controlled translation stage with encoder providing 8 mm of travel in increments of 1.5 nm. This variable focusing capability allows for the ability to compensate for changes in focal length of the photon sieve caused by thermal or creep effects.

1.2.6 Collapsible Space Telescope for Nanosatellites

Between different types of deployable satellites, it could be found an important and innovative application in NASA Ames Research Center where, in August 2013, started a project about Collapsible Space Telescope for Nanosatellites. They started analysing all the optical and mechanical performances of a collapsible Cassegrain telescope in order to collect some information to design a compact spectropolarimeter. For that project they required a high-throughput entrance aperture. The main goal was to create a particular instrument for future trends about remote sensing application on Earth either other planets, but trying to get that kind of performances in a small packaging. In the past

versions their baseline aperture diameter for their instrument compatible with a small-sat architecture was 76.2 mm (3-inch), but with the introduction of this new collapsible telescope they had increased their collecting area by a factor of four, doubling the aperture diameter. They tried to focus all their attention following these requirements needed for science instrument application:

1. The telescope shall have at least a 91.6 cm² unobscured collecting area.
2. The telescope's effective f/# shall be faster or equal to f/6.
3. The distance between the back of primary mirror and system focus shall be at least 6 cm.
4. At best focus, the telescope system shall deliver spot size and shape repeatability of 10 % at center of the field of view (FOV) and 25 % at the edges of the FOV.
5. The telescope system shall deliver a bore-sight repeatability of at least 100 microradians.
6. The telescope system shall deliver a fully illuminated field over the 50um x 5 mm slit area.

In order to show their idea, it follows the concept description and how it works:

- *Concept description:* A strain Deployable Ritchey Chrétien Cassegrain 15.42 cm telescope occupies a volume of only 1Ux4U (U is a 10-cm cube) and then extends after orbit insertion to provide the target focal length, providing greater resolution capability than achievable with a fixed structure telescope to integrate in nanosatellites.
- *How it works:* Carbon composite coiling strain members provide the energy needed for deployment mechanism controls both the deployment rate and final tensioning force to rigidize the structure. Optical resolution is ultimately limited to aperture, thus limiting the optical utility of small spacecraft. By collapsing the telescope into the 1Ux4U space a scientifically useful aperture of 15-17cm is achievable.



Figure 1.7. Deployable Telescope Truss Concept

1.3 ACTIVE CONTROL SYSTEM FOR TELESCOPE MIRROR

One of the key challenges in the development of a deployable optical system, however, is making sure that it can meet its performance targets following its deployment. To accomplish that, an active control system must be implemented.

1.3.1 Active optics

Active optics (or active control) is a technology developed in the 1980s. It's used in the field of reflecting telescopes and its purpose is preventing deformation who could possibly characterize the telescope's mirrors. The deformations can be caused by external influences such as wind, temperature and mechanical stress

Historically, primary mirrors were quite thick and could reach a maximum diameter of 5-6 meters, in order to maintain the correct surface configuration. With the active control is now possible to

construct 8-10 meters diameter mirror and also to segment them. Lighter and thinner mirrors are now designed, but they can be maintained rigidly in the correct shape thanks to different elements, like actuators who apply variable forces to the body. Moreover, the mirror segmentation reduces the sagging due to weight typical of the monolith mirrors. The choice of the segment size is a result of a cost optimization. Smaller segment can be designed thinner and so, even lighter, but they will also require more complex software and more expensive hardware.

The actuators and their control, combined with an image quality detector to obtain the best image, is called active optics. That said, we can affirm that the telescope is actively still, in its optimal shape.

Telescopes based on active optics technology can correct errors resulting from optical manufacturing errors, deformations and misalignments due to gravitational or thermal reasons. A wavefront sensor monitors image quality, providing feedback in terms of aberration coefficients; the shape of the primary mirror and the position of the secondary mirror are adjusted accordingly. Misalignments and axial dispersion between the primary and secondary mirrors give rise to blurring and coma; coma refers to aberration inherent to certain optical designs or due to imperfection in the lens or other components that results in off-axis point sources such as stars appearing distorted, appearing to have a tail like a comet. These problems are commonly removed in active telescopes by the movements of the secondary mirror; while all other modes of aberration are compensated by the shape of the primary mirror. Different types of secondary mirror positioning devices have been proposed.

1.3.2 VST – Solutions for the positioning of the secondary mirror

The Very Large Telescope array (VLT) is the flagship structure for European terrestrial astronomy at the beginning of the third millennium. It is the most advanced optical instrument in the world, consisting of four unitary telescopes with main mirrors of 8.2 m in diameter and four mobile auxiliary telescopes of 1.8 m in diameter. The Very Large Survey Telescope (VST) is a 2.6 meters detection telescope installed on Cerro Paranal in the ESO VLT observatory, and it's operative since 2011.



Figure 1.8. ESO VLT observatory facility

VST covers a wide range of wavelengths from ultraviolet to near infrared optics (0.3 to 1.0 microns) and, while larger telescopes such as the VLT can only study a small part of the sky at any time, it is designed to photograph large areas quickly and deeply. With a total field view of $1^\circ \times 1^\circ$, twice as wide as the full Moon, the VST supports the VLT with wide-angle images by detecting and precharacterising the sources, which the VLT can then observe further.



Figure 1.9. VST at Cerro Paranal

The VST is composed of two mirrors, a primary mirror (M1) with a diameter of 2.61 and a smaller secondary mirror (M2) with a diameter of 93.8 cm. The telescope is also equipped with a single instrument on the dedicated focal plane: OmegaCAM. With its camera, the quality of its optics and the exceptional visibility conditions of the Paranal, the VST is making important discoveries in a variety of areas of astrophysics and cosmology. In planetary science, the investigation telescope aims to discover and study remote bodies of the Solar System as transneptunian objects, as well as to search for transits of extra-solar planets. The galactic plane has also been widely studied and is providing astronomers with crucial data to understand the structure and evolution of our galaxy. Further away, the VST is exploring nearby galaxies, extragalactic and intra-cluster planetary nebulae, and investigating weak objects and micro-lens events. In the field of cosmology, the VST is targeting mid-red shifting supernovae to help define the scale of cosmic distance and understand the expansion of the Universe. The VST is also looking for mid-red shift cosmic structures, active galactic nuclei and quasars to deepen our understanding of galaxy formation and the primordial history of the Universe. The VST project is a joint venture between the Southern European Observatory and the Capodimonte Astronomical Observatory (OAC), part of the Italian National Institute of Astrophysics (INAF). The Italian center designed the telescope while ESO is responsible for the civil engineering work on the site. The active control system is used in the case to control the positioning of the secondary mirror.

The de-focusing and coma compensation can be studied by applying the theory of two-mirror telescopes. The rotation of the secondary mirror, around the axes of the reference frame centred on its center of curvature, produces the coma but not the displacement of the image: this property makes it possible to correct the coma during exposures, provided the mirror positioner can form such movements. The de-focusing is corrected with a movement along the optical axis Z while the coma with movements along the optical axes X and Y combined with inclinations around the same axes; so periodic compensation during both defocus and coma exposures requires movements of the secondary mirror in five degrees of freedom. In these five degrees of freedom, without moving the image to the focal plane, you can use the image during exposures. A hexapod has been implemented in VST: it has six degrees of freedom, including redundant rotation around the optical axis. The hexapod motors adjust the length of its legs by guiding the mirror to the desired position and orientation.



Figure 1.10. Active optics control of the secondary mirror

In this case the problem is to determine the leg lengths of the hexapod to guide the secondary mirror into the desired position and orientation. In VST an iterative algorithmic approach is used. Each of the six legs of the hexapod is a linear actuator, controlled by the Hexapod Counter Unit (HCU) which manages the forward and reverse kinematics in real time and for each movement calculates the trajectories of the six actuators that start and stop simultaneously.

The trajectories are kept coherent, moving only the desired coordinates, avoiding cross talk about the degrees of freedom of the other mirror.

The control system has proven to meet the specific needs of active optics. The control system incorporates real-time kinematics mutations, using, in practice, no approximations in the translation from the joint space to the workspace and vice versa.

A Local Control Unit (LCU) sends position commands to the HCU and receives return telemetry.

The control diagram used by this secondary mirror is shown in the figure, which demonstrates how a dual loop is implemented; the PID logic of the controller is divided into 2 loops: the internal one performs the D function and the external one the function P and I.

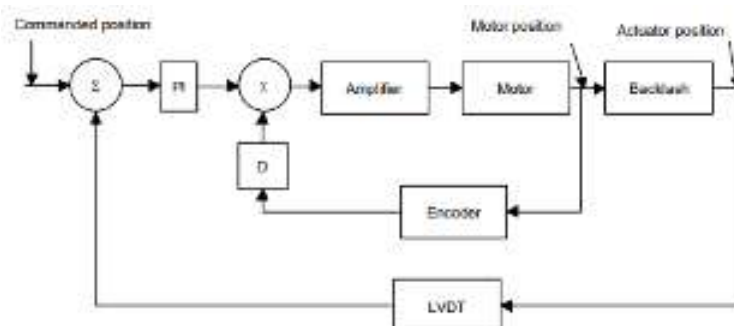


Figure 1.11. Dual loop control of single actuators

1.3.3 W.M. Keck Telescopes – Segmented primary mirror active control system

Situated at 4145 meters in the summit of Mauna Kea, Hawai'i, the William Myron Keck observatory makes an excellent astronomical observation site thanks to its nearness to the equator.



Figure 1.12. Keck Observatory atop Mauna Kea

There are two telescopes, Keck I and Keck II, which both measure 10 meters across. The mirrors of each one of them are made of 36 segments that work together similarly to a single mirror. The telescopes are housed in insulated domes of 700.000 cubic feet, where giant air conditioners run all day to keep the temperature at or below freezing; this helps reduce mirror and steel deformation.

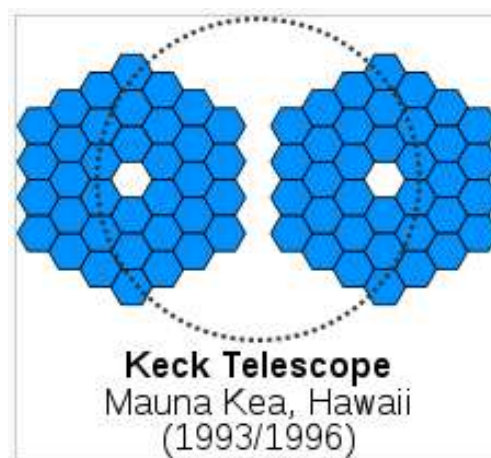


Figure 1.13. Representation of the Keck Telescopes mirrors, divided in 36 segments each

The entity was created in 1954 by William Myron Keck, who founded the Superior Oil Company. Nowadays, funding for the observatory comes from a philanthropic organization known as the W.M. Foundation. Its mandate includes funding science, and in 1985 it granted 70 million dollars to construct the first telescope, Keck I. During construction, 68 million dollars came through for the second telescope, Keck II. They began their science work in 1993 and 1996 respectively.

Keck's twin telescopes have made a host of discoveries since construction finished. In 1999 the observatory discovered how stars move in the Andromeda Galaxy, helping the astronomers understand just how immense the galaxy is. In 2015, Keck helped track down 'EGSY8p7', which was then called the most distant galaxy ever found (13.2 billion light-years from Earth).

The Keck Telescopes possess a 10 meters diameter primary mirror, segmented in thirty-six hexagonal mirrors and stabilized by an active control system. The components of the control system are position sensors, electronics, computer, actuators and software.

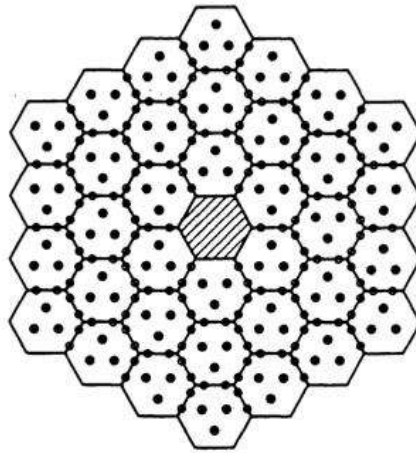


Figure 1.14. Locations of actuators and sensors from the concave side of the primary mirror

The control system function is to turn the segmented mirrors into a monolith by electronically gluing them together and controlling three of their six DoF, piston and x and y tilts. The function is accomplished by measuring the relative position of each mirror, calculating the errors and then repositioning the mirrors two times a second. In the figure above the positions of the 108 actuator (filled circles) and the 168 relative position sensors (open circles) are illustrated. Actuator motion commands are calculated every half second from the filtered position sensor data.

The Active Control System (ACS) receives three external inputs:

1. The Drives and Control System (DCS) controls the telescope slewing and tracking, by sending commands and deciding when to start and stop the ACS;
2. The Phasing Camera System (PCS) provides commands to piston and tilt the various segments. It also measures the optimal configuration of the whole mirror;
3. The star tracking camera system measures the overlapping of the 36 images from the segments.

ACS REQUIREMENTS:

- Control loop bandwidth > 0.1 Hz
- Control loop period < 500 ms
- Step response settling time: 10 seconds
- Maximum correction for primary mirror cell distortions during tracking: 27.5 nm/second

A simplified illustration of the position control system for one actuator and sensor is shown in the figure below. The reference unit, which is a function of the zenith angle z and the mirror temperature T , prescribes the desired sensor reading, derived from the calibration data. The difference between the position sensor reading and the reference input is passed through a lowpass filter and amplifier. The resultant signal causes the actuator to move and correct the sensor reading.

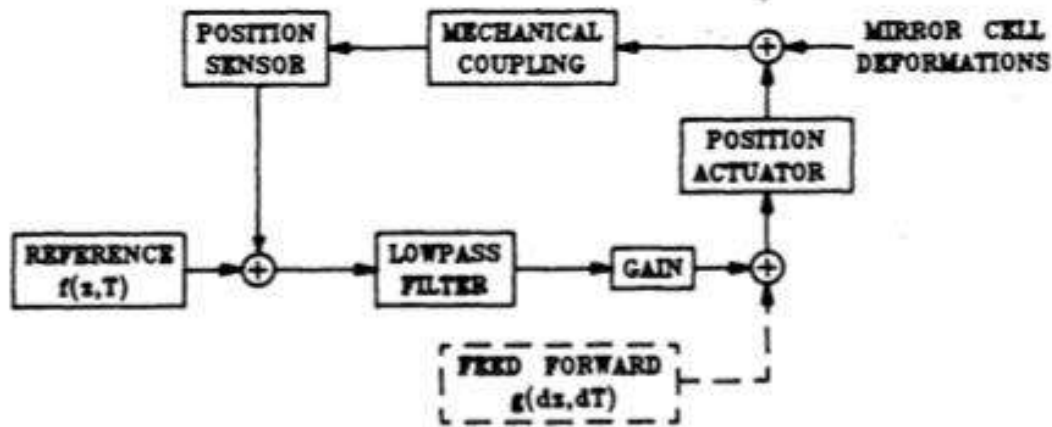


Figure 1.15. Simplified position control system

1.3.4 SOFIA Telescope – Extremely compact secondary mirror unit

SOFIA (Stratospheric Observatory For Infrared Astronomy) is a Boeing 747SP aircraft modified to carry a 2.5 meter reflecting telescope. Flying into the stratosphere at 15000 meters above sea level puts SOFIA above 99% of Earth's infrared blocking atmosphere, allowing astronomers to study the solar system and beyond in different ways. It is made possible thanks to a partnership between NASA and the German Aerospace Center (DLR).



Figure 1.16. SOFIA flying during the day

NASA awarded the contract for the development of the aircraft, operation of the observatory and management of the American part of the project to the Universities Space Research Association (USRA) in 1996. SOFIA's telescope first saw light on May 26, 2010, as the successor to the Kuiper Airborne Observatory. During 10-hour, overnight flights, it observes celestial magnetic fields, star-forming regions, comets, nebulae, and the galactic center. It is scheduled to operate until the early 2030s.

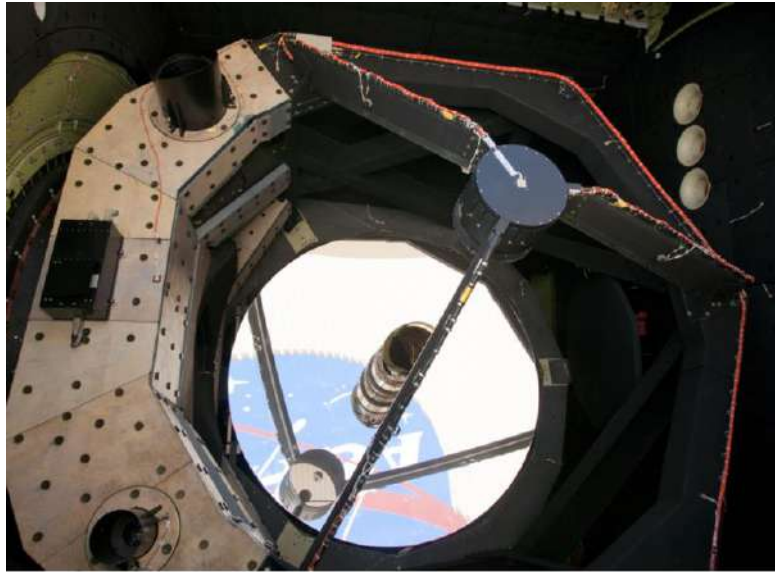


Figure 1.17. The NASA logo reflected in SOFIA's primary mirror

The astronomical community has fully embraced the platform's unique resume of skills. In the summer of 2015, it has been used to observe Pluto's atmosphere in two colours, allowing scientist to finally confirms that dwarf planet's fuzziness indicate haze.

Recently, SOFIA has embarked on two legacy programs. One aims to study groups of stars of different sizes to determine whether their bubbles and shockwaves make it easier or harder for other stars to form nearby. The other is targeting a large tract of the center of the Milky Way about the size of four full moons. Despite an abundance of star ingredients, something seems to be stopping stellar birth in this region, and researchers hope more detailed images will help them figure out what.

CSEM (Swiss Center for Electronics and Microtechnology, Switzerland) has been contracted for the development and manufacturing of the secondary mirror assembly (SMA). It includes all the actuations mechanisms, the sensors, the acquisition and drive electronics as well as the embedded software.

The secondary mirror has a diameter of 350 mm and is actively driven along five degrees of freedom in order to be able to center and focus the telescope light beam. The major goal is to provide the mechanism of the capability to 'chop' the secondary mirror, and therefore the field of view of the telescope. This chopping mode should be used in the infrared wavelengths, where the object luminosity is only marginally greater than the sky background, by observing the object and subtracting the background noise.

The Secondary Mirror Mechanism (SMM) is subdivided into two system named Tilt-Chopping Mechanism (TCM) and the Focus-Center Mechanism (FCM). The first is designated for the fast tip-tilt action and chopping actuation, while the latter is generally responsible for all the offsets of the mirror positions.

The FCM essentially consist in a Stewart platform hexapod-like mechanism. Such a platform is a parallel manipulator with six prismatic actuators involved. The mechanism is capable of orienting the mobile base along all six DoF, even though only five of them are actually used. The hexapod is divided in two actuators groups, one with three vertical actuating legs, and one with three horizontal actuating legs.

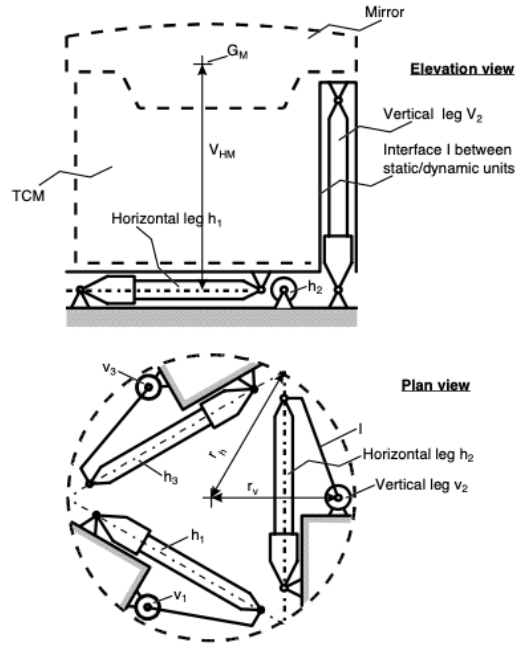


Figure 1.18. Principle of hexapod with free vertical and three horizontal legs

The TCM consist of:

1. A flexure pivot linked to the center of mass of the mirror holder, that constrains the x, y, z directions and the torsion R_z ;
2. Three actuator systems located in a 120 degrees symmetry, constituted of a lever actuated by a linear motor that transmits the motion to the mirror and to a reaction compensation ring. Each lever has then three pivots and the central one of this is mounted on the base. The other two transmit an uncoupled vertical motion respectively to the mirror and to a reaction compensating ring;
3. The reaction compensation ring, that move in the opposite angular direction of the mirror, in order to compensate the angular momentum.

The angular position of the mirror is detected by three non-contact displacement transducers located in correspondence with the mirror attachments, while the three axial forces are monitored by load cells. The three radial attachments will receive only axial forces with average value equal to zero, and so effectively controlling only the tip-tilt actions.

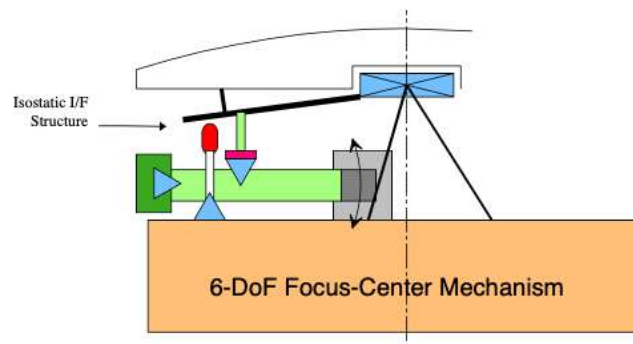


Figure 1.19. Schematic Diagram of the TCM

The chopping controller is constituted of three uncoupled SISO controllers, which operate in mirror coordinates. One controller is for the elevation axis, one for the cross elevation axis, and the last one is a sum-of-force controller which reduces any vertical force acting on the mirror suspension and guiding mechanism. All of them generate command voltages, which are then transformed to coordinates so as to determine the command voltages of the three real power amplifiers. These amplifiers apply in turn input voltages to the three actuators driving the TCM. The three forces transmitted through the mirror are measured using load cells. Three proximity sensors measure the position of the mirror, and each one of them corresponds to a single actuator displacement, no velocity sensor is used.

TCM REQUIREMENTS:

Settling time		≤ 5 ms for amplitude $\leq 0.06^\circ$ ≤ 7 ms for amplitude $\leq 0.24^\circ$ ≤ 10 ms for amplitude $\leq 0.6^\circ$
Square wave frequency	range	1 Hz to 20 Hz
	set increment	1 Hz
	phase adjustment	$\pm 180^\circ$ in 1° increments

CHOPPING MODES:

<u>Frequency (Hz)</u>	<u>Predominant mode shape</u>
29	Fundamental tilt mode
29	Fundamental tilt mode (orthogonal)
350	Mirror vertical displacement
390	Lever torsion about pivot
390	Lever torsion about pivot (orthogonal)
402	Mirror torsion

1.3.5 The Deployable Telescope PistonCam

The Deployable Telescope, being developed at Delft University of Technology, aims to provide the same resolutions as current state-of-the-art commercial Earth Observation (EO) systems at a drastically reduced cost. The telescope has been designed for the specifications based on the Worldview-3:

- Aperture Diameter: 1.5 m
- Focal Length: 11 m
- Orbital Altitude: 500 km
- Swath Width: 5 km
- Cross-Track Field of View: 0.6°

The design was optimized to provide a diffraction limited image quality across the entire field of view. During the optimization process, degradations due to deployment tolerances and stability tolerances were taken into account. A freeform tertiary mirror was used to improve the optical performance in the corner, while maintaining a compact design. The primary mirror of the telescope has been split up in four mirror segments, that can be folded alongside the instrument bus during launch. The aperture segments are tapered in the center. This creates additional area in the center of the aperture, leading to improved image contrast. Furthermore, the segments are now adjacent to one another in the

central ring of the pupil, which proves to be a valuable feature that can be used by the phasing systems.



Figure 1.20. Mechanical design of the deployable telescope

One of the key challenges that must be addressed to ensure that the deployable telescope can meet a diffraction limited performance is the alignment and co-phasing of the mirror segments. After deployment, the four segments need to be pointed in the same direction and brought into phase with an accuracy of just 10 nm. Meeting such tight requirements is impossible without a sophisticated active optics system: to speed up the phasing process, a novel phasing sensor concept was developed, called PistonCam.

Instead of looking at properties of measured point spread functions, the sharpness criterion of each sub-image is optimized. Each sub-image is influenced by only two segments at a time. As a result, the relative piston error between two segments can be found with comparative ease. To bring the segments into phase, several steps must be taken. The first step is the co-alignment of the segments, which ensures that all segments are pointed into the same direction. Phasing is only possible after co-alignment; the light coming from two adjacent segments must interfere in the image plane to notice changes in sharpness when the vertical location of a segment is varied. Once all segments have been co-aligned, two of the segments, at opposite sides of the telescope, are scanned through the adjustment range and the image sharpness is recorded throughout the process. After moving the mirror segments to their calculated position, the sharpness in each channel is measured and compared to the sharpness of the reference channel. If one or more of the channels shows a significantly lower sharpness value than the reference channel, the process is restarted. This typically occurs only when the relative piston error between any two adjacent segments exceeds 4 microns.

Throughout the phasing process, the telescope must track a fixed point on Earth. The sharpness criterion depends to a large extent on the scene. Normalization of the sharpness criterion using the data obtained with the reference channel does not fully remove this scene dependence. The scene that is used for the phasing process can, however, be switched at several instances; each co-alignment step can be performed with a different ground scene.

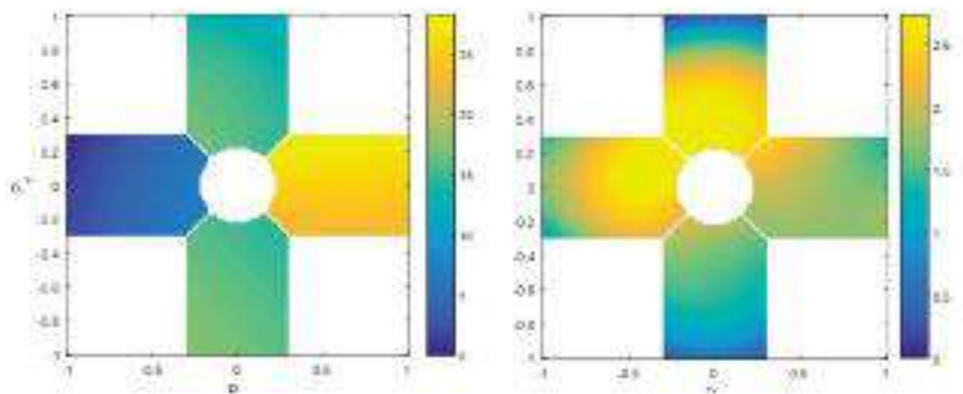


Figure 1.21. OPD in waves before and after the phasing procedure with PistonCam

Extensive Monte Carlo analyses were performed to check the robustness of the phasing algorithm and to see how many iterations are required. In more than 95% of these cases, the system could successfully bring the segments into phases. The cases which were unsuccessful suffered from intersegment piston errors larger than 5 microns. Running the systems again with an extended scan range, led to successful convergence for these cases as well.

It was found that generally between 400 and 700 mirror positions had to be evaluated before the segments were brought into phase, depending on the severity of the deployment errors. The total duration of the procedure depends on the actuation frequency that can be achieved, but conservative estimates suggest that an actuation frequency of 10 Hz should be achievable, leading to an average phasing duration between 40 and 70 seconds. At this point, the wavefront can be considered to be continuous, allowing the remaining aberrations to be corrected by a deformable mirror in the exit pupil plane.

1.3.6 ATLAST – Hybrid Sensor Active Control (HSAC) Architecture

This ATLAST (Advanced Technology Large-Aperture Space Telescope) reference design (Figure 1) is a 10 m-class observatory under assessment as a candidate for selection by the 2020 NRC Decadal Survey, which is a partnership between the National Academies and the astronomical community to identify key priorities in astronomy and astrophysics and develop a comprehensive strategy for agency investments in the upcoming decade.

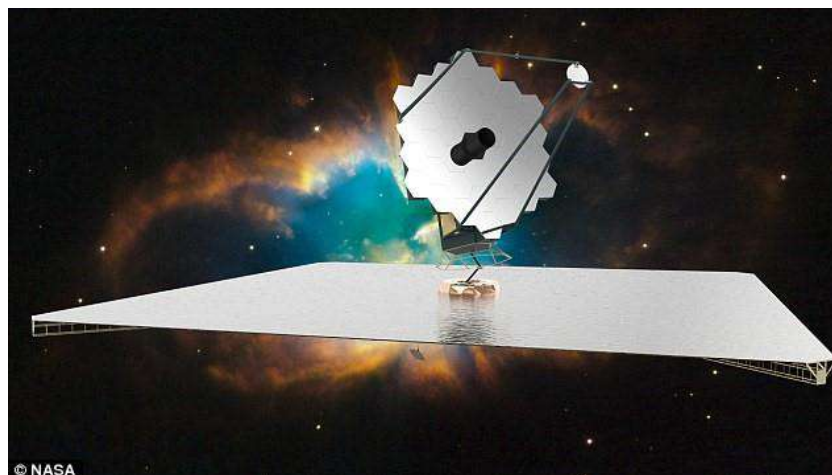


Figure 1.22. ATLAST reference design

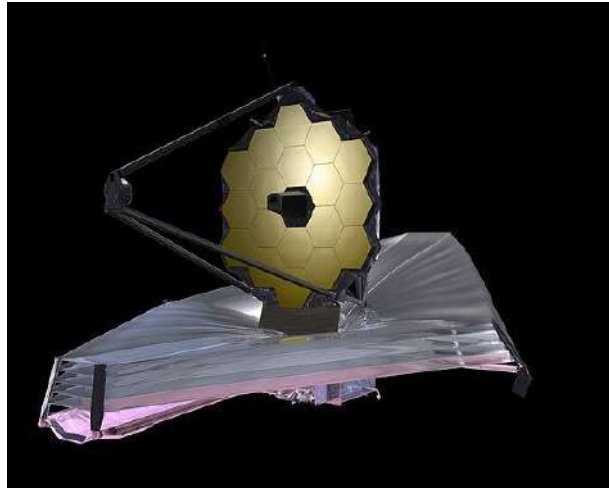


Figure 1.23. JWST design

With the completion of the final servicing mission to the Hubble Space Telescope (HST) and the development of the James Webb Space Telescope (Figure 2, to be launched in October 2021), the astronomical community has begun planning the next great observatory. An important mission study for the next generation UV-optical observatory was the Advanced Technology Large-Aperture Space Telescope (ATLAST). Like JWST, ATLAST would be launched into an orbit around the L2 Lagrange point that provides consistent full-sunlight, no temperature variations, and gives ST-2020 a constant field of regard (FOR) of about 3π steradians. A telescope in this orbit has roughly double the observational efficiency of a telescope in low Earth orbit (LEO), and is very stable thermally.

The ATLAST project includes a 9.2-meter diameter, 36-segment primary mirror. The design employs both a 2-mirror Cassegrain channel for ultraviolet science and planet-finding instruments and a 4-mirror WFOV channel. The design leverages highly from the JWST telescope architecture with modifications to address the size and UV-optical performance. The deployable configuration consists of two wings (each wing contains 7 segments), a secondary mirror support structure, as well as a central baffle for controlling stray light. The primary mirror segments are held on a large composite backplane.

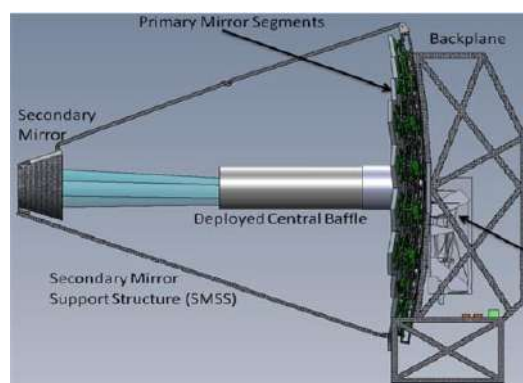


Figure 1.24. ATLAST configuration project

A new approach employed in the ATLAST architecture is the use of an active WFSC architecture that updates the primary mirror every 5 to 30 minutes and updates the secondary mirror once a day. The active architecture replaces the passive thermal architecture of JWST, which updates the primary mirror every 14 days and the secondary mirror approximately once a year.

For ST-2020 (a mission concept), dynamic disturbances (causing jitter) could be controlled with the use of technology developed for JWST (James Webb Space Telescope) for dynamic isolation, leaving the key challenge to be thermal stability. A solution to the temporal problem of thermal stability is more frequent updating of the segmented primary mirror alignment. This poses an additional challenge because slewing the observatory between fields of interest and a WFSC (Wavefront Sensing and Control) target star would be both inefficient and by itself cause further instability. To solve this problem, a novel Hybrid Sensor Active Control (HSAC) architecture has been proposed, which combines the Fine Guidance Sensor (FGS), used to guide the telescope, and the wavefront sensor (WFS) into a single instrument that uses guide stars in the observatory field-of-view (FOV) to perform frequent updates to the telescope alignment.

In order to reduce dynamic disturbances by nearly a factor of 4 times over JWST, ST-2020 employs a Disturbance Isolator Precision Pointing System (DIPPS), which actively isolates the telescope from the spacecraft. ST-2020 also employs an actuated Pointing Arm (PA), which allows the Optical Telescope Assembly (OTA) to be pointed at targets without changing the solid angle view of the sunshield with respect to the sun, a key source of instability on JWST.

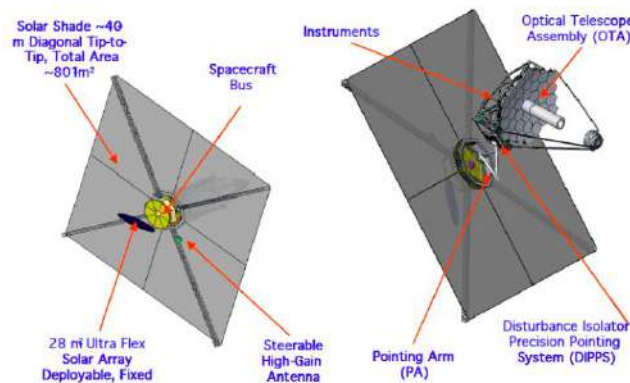


Figure 1. 25. ST-2020 spacecraft, OTA, and sunshield mission concept

Guidance and wavefront sensing will be performed by a set of four, nearly identical instruments known as Hybrid Instruments (HI). A schematic of one of the HIs is shown in Figure 8 and a mechanical model is shown in Figure 9. Light from the OTA is fed to the HI via a pick-off mirror. A bi-directional star selection mirror (SSM) is used to access a 4 x 4 arcmin FOV and steer an isolated bright star onto the Fine Guidance Sensor (FGS) and Wavefront Sensing (WFS) detectors. The beam of light is split three ways, with 20% of the light going to the FGS detector for guidance and 40% of the light going to each of two WFS detectors.

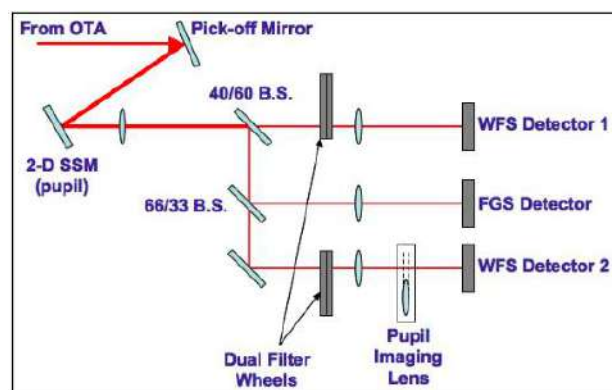


Figure 1. 26. Schematic of a HI

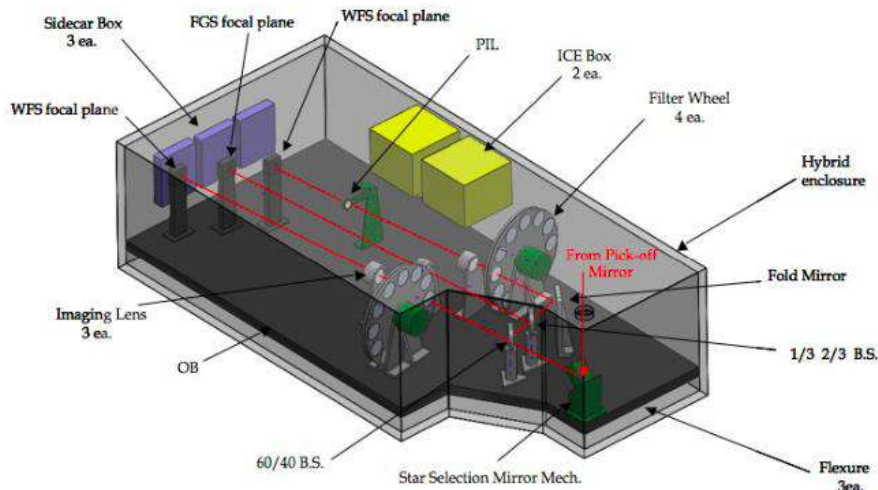


Figure 1. 27. Mechanical model of the HI

At any given time, two HIs are required for guiding ST-2020. In one HI, closed loop centroiding will be performed to generate commands for the attitude control system and SSM. The second HI, looking at a field point well separated from the first, will be used to sense observatory roll. Initially, a full 2k x 2k frame will be captured and read-out. The location of the guide star in the frame is determined and the SSM is actuated to move the star to the center of the frame. A second full frame is captured and again the location of the guide star is determined. An 8 x 8 pixels window about the star is read and centroiding at 5 Hz is performed at a noise-equivalent angle of 0.57 milli-arc-seconds. The FGS and attitude control will maintain pointing to within 1.3 milli-arc-seconds of image motion.

For ST-2020, there will be three operating modes of WFSC: primary mirror (PM) maintenance, secondary mirror (SM) maintenance, and one as a commissioning/diagnostic mode.

PM MAINTENANCE:

The PM maintenance mode is completely automated and will occur approximately every 5 – 30 minutes, depending on the radiometric brightness of the source. In this mode, only a single HI is used for WFSC. The filter wheels are actuated to place a narrow band ($\Delta\lambda/\lambda \approx 1\% - 5\%$) spectral filter in each beam path, as well as a weak lens. One path will use a positive weak lens while the other uses a negative weak lens to produce two out-of-focus images of the star. The images are sent to an on-board processor where the phase-retrieval algorithm processes the data and segment-motion commands are generated. To achieve the 5 – 30 minutes duty cycle, the phase-retrieval algorithm is implemented on dedicated, specialized parallel computing hardware.

SM MAINTENANCE:

The SM maintenance mode will also be automated but will require the use of 3 of the HIs and will be performed nominally about once a day. In this mode, each HI captures two out-of-focus images from three different points in the telescope's FOV. All six images are sent to the on-board computer where multi-field phase retrieval is performed. In this mode, SM-motion commands are generated in addition to segment-motion commands. This mode is required to remove possible wavefront ambiguities that may be introduced by SM motion. For example, a misplaced SM may introduce astigmatism, which is indistinguishable from astigmatism of the PM when sensed at a single field point. Sensing at multiple field points helps to remove this ambiguity.

COMMISSIONING / DIAGNOSTIC:

As the name implies, the commissioning/diagnostic mode will be used only after the initial deployment of the observatory, or as a diagnostic mode should the observatory become misaligned. The mode is entirely manual, allowing all images collected by the HIs to be sent to a ground-based control center for further analysis. There are several steps involved in commissioning, as shown in Figure 10. The initial 4-steps after deployment (SM Focus Sweep through Segment-Image Array) involve identifying the point-spread functions (PSF) of each segment on the WFS detectors. Global alignment uses phase retrieval on spots from each segment to characterize each segment individually, as well as provide information for better SM placement. Image stacking then moves each PSF to the center of the array by tilting each segment in the PM. Coarse phasing uses dispersed Hartmann sensor (DHS) elements in the filter wheels to sense and correct the piston errors between each segment to approximately 200 nm. The PM fine-phasing routine is similar to the PM maintenance routine and multi-field alignment is similar to the SM maintenance routine.

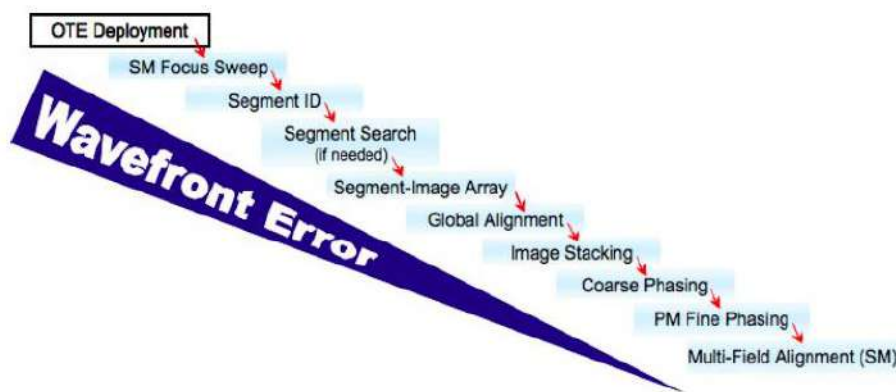


Figure 1.28. Flow chart of the commissioning process

Also, during commissioning, pupil images will be acquired using the PIL. The pupil images are used as Fourier constraints during the phase-retrieval processing of the data. Occasionally during the observatory's lifetime, updated pupil images will be required for diagnostic information or for updating the constraint used in the phase-retrieval processing.

1.3.7 Adaptive optics

Space based telescopes have always been a better choice when it comes to image resolution and quality, however they are considerably more expensive than their ground-based counterparts (at comparable specs) and the maintenance is still an obvious challenge.

The main issue with ground-based optics telescopes, is the distortion caused by the earth's atmosphere, whose chemical composition and physical properties can broadly vary both in space and time: this can result in blurry images, despite how powerful the telescope might be.

In fact, the optical turbulence of the atmosphere is intrinsically unavoidable. However, in the last decades, adaptive optics have been of great interest in the field of Astronomy. Its key concept is analysing the interfering noise with specialized equipment, which scans the atmosphere at a high frequency, then letting an algorithm elaborate the raw input data and using it to manipulate the so called "deformable" lenses that will take into account the analysed changes and correct their movement accordingly. These lenses' movement is finely and reactively adjusted by dedicated

actuators. The results are really accurate and rapidly improving as the technologies involved keep getting better, to the point where ground-based telescopes can use their optics' full potential and finally trade blows with their space-based counterparts.

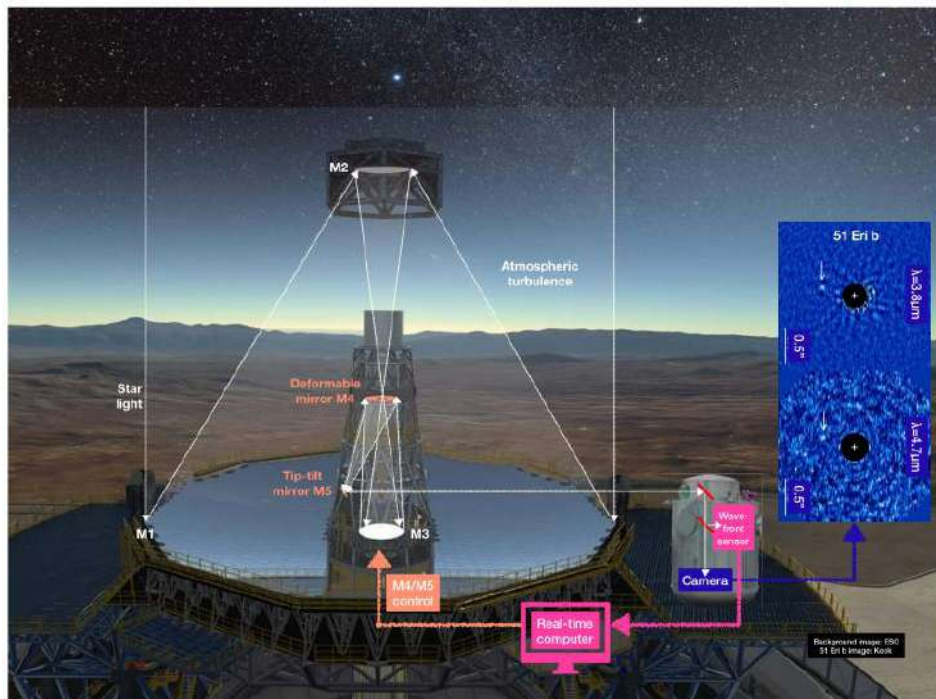


Figure 1.29. Single-conjugate AO scheme for the ELT METIS instrument

In the figure above, is shown how the atmospheric turbulence distorts the observed starlight collected by the ELT's segmented primary mirror M1. Reflected up to mirror M2 and down to mirror M3, passing through a hole in the DM M4, the last reflection on the tip-tilt mirror M5 forwards the light into the METIS cryostat on the right, outside of the telescope. The METIS internal WFS measures the atmospheric distortions. A real-time computer estimates the actual wavefront and delivers the corresponding information to properly shape the DM M4 as well as to properly steer the tip-tilt mirror M5 with the M4/M5 control unit. Eventually, an exoplanet like 51 Eri b can be detected at two different wavelengths (see blue-coloured inset on the right).

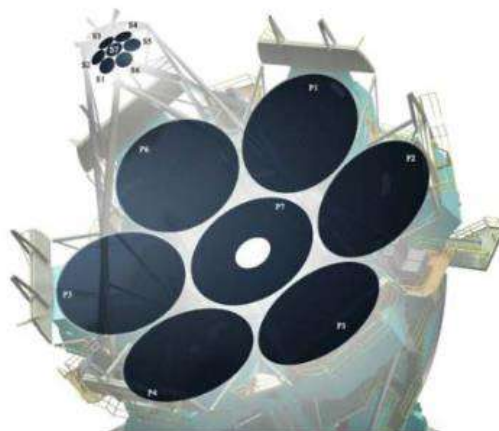


Figure 1.30. GMT optical layout of the seven segments, primary mirror (P1–P7) and secondary mirror (S1–S7)

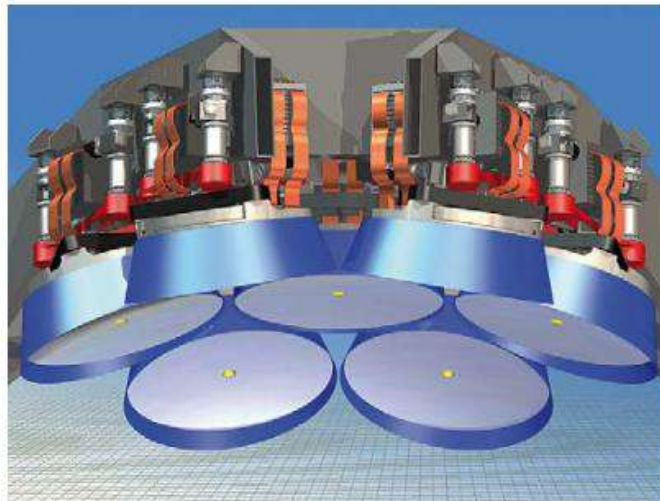


Figure 1. 31. Detailed rendering of the seven segments of the secondary mirror

Several concepts and designs are used to implement Adaptive Optics and are being currently utilized by many ground telescopes, including the most advanced ELTs (Extremely Large Telescopes), but the key principle remains the same.

As of right now, the usage of AO has been a game changer in astronomy. The common scenario for AO is in infrared astronomy, while usage in the visible spectrum is still currently a huge challenge, but further developments are expected in this field as well.

1.3.8 MMT – Unconventional secondary mirror used as corrector for the adaptive optics system

The MMT Observatory (MMTO) is an astronomical observatory on the site of the Fred Lawrence Whipple Observatory. The Whipple Observatory complex is located on Mount Hopkins, Arizona, USA (55 km south of Tucson) in the Santa Rita Mountains.



Figure 1. 32. View of the MMT Observatory

The MMTO is the home of the MMT (ex Multiple Mirror Telescope), which has a primary mirror of 6.5 m in diameter. The name is derived from the six smaller mirrors originally used before the single primary mirror was installed in 1998. The primary mirror has a special lightweight honeycomb design made by the University of Arizona's Steward Observatory Laboratory Mirror. The MMT is housed in a building which allows the walls and roof around the telescope to be completely tilted, allowing it to

cool down very quickly to improve observation. One of the reasons for its original multiple-mirror design was the difficulty of fusing large mirrors. A solution to this problem was found by Roger Angel of the Steward Observatory, University of Arizona, who projected mirrors with a honeycomb structure into a rotary oven. This made it possible to replace the six mirrors with one only 6.5 meters long. The original building and part of the structure could be reused. The new mirror was the first of its size to be cast and polished in the Steward Observatory Mirror Laboratory. The updated MMT, whose name is no longer an acronym, was rededicated on 20 May 2000.

At the end of 2002, a new deformable secondary mirror was added to the telescope. The MMT AO system contributed to the design of the Large Binocular Telescope, which achieved record-breaking Strehl ratios (a measure of the quality of optical image formation) with its AO system in 2010. From 2004 to 2010, approximately 8% of MMT observation time was made accessible to the entire astronomical community through the National Science Foundation's Telescope System Instrumentation Program (TSIP), administered by the National Optical Astronomy Observatory (NOAO).

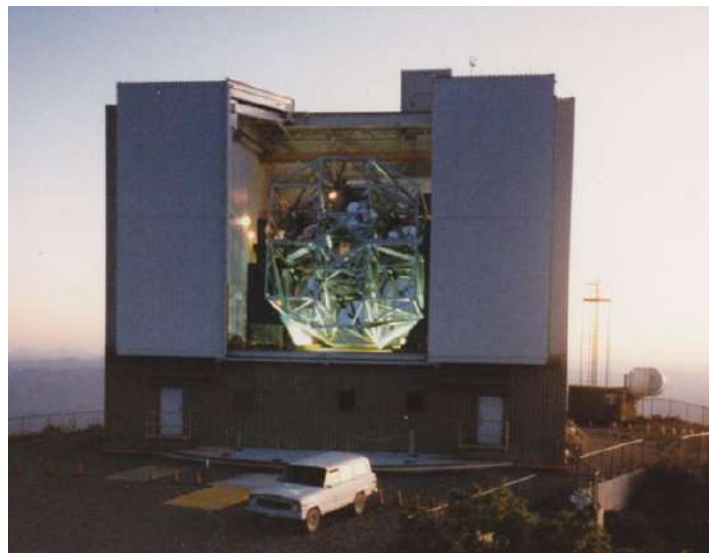


Figure 1.33. MMT and its six primary mirrors

We will now talk about the unconventional secondary mirror used as corrector for the adaptive optics system of the 6.5m conversion of MMT.

The design of the unit consists of a deformable convex mirror of 2mm thickness and 642mm convex diameter whose figure is controlled by 336 electromagnetic force acts, a shell used as position reference and an aluminium shell as support and cooling of the actuator. The actuator forces are controlled using feed-forward and decentralized closed loop compensation using the 40kHz feedback signals of the 336 capacitive position sensors.

In the current generation of adaptive optics systems (AO) wavefront correction is usually performed by a piezo-electric deformable mirror (DM); the DM is part of an AO module placed one of the focal points of the telescope. This, however, intrigues some limitations to the AO system itself.

The unconventional use of an adaptive secondary mirror (AS) as a wavefront corrector can solve the limitations. It uses electromagnetic actuators that have a large chord eliminating the need to have a separate tip-tilt corrector; it provides feedback on the current position of the mirror eliminating the need to monitor the active surface.

The six main components of the AS are:

- An intermediate flange bolted to the mobile hexapod M2-F/9 of the telescope which provides a mechanical interface to the unit;
- Three cooled boxes for electronics.
- An aluminium plate (*cold plate*) that provides support and cooling for the actuators.
- The 336 electromagnetic actuators;
- A ULE glass *sheet* with perforated holes, fixed to the cold sheet by means of a second hexapod and a central shaft.; this plate is used as a position reference for the thin deformable mirror.
- A deformable Zerodur convex aspherical shell; this shell has a central hole to which a central membrane is attached to provide a lateral constraint.

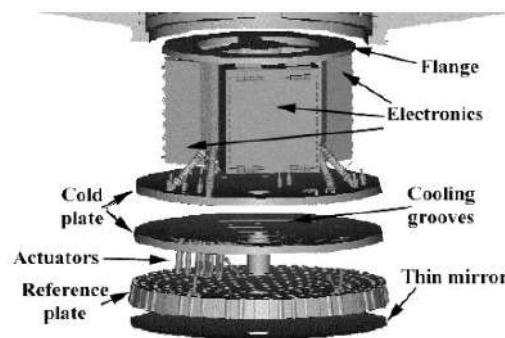


Figure 1.34. Exploded view of the MMT Adaptive Secondary unit

1.4 PASSIVE CONTROL SYSTEM – POINTING

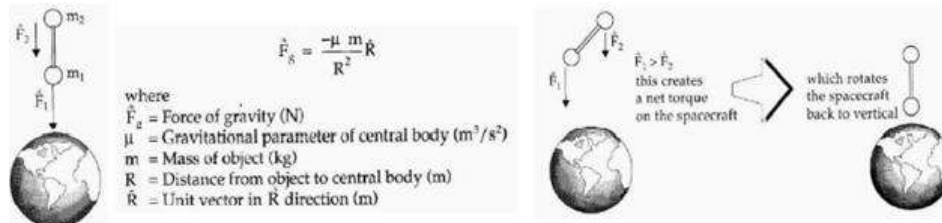
Passive pointing control systems are essential to moderate disturbance torques present in the space environment, to guarantee a continuous pointing at the required target. Disturbance torques are exploited, in positive, to emphasize a particular one of these to minimize others. Unlike active systems, passive control systems are advantageous for no power supply required, like electrical or propellant power, so operative life is not limited by this factor. Moreover, as there are no moving parts there is no mechanical usury of components. For these reasons passive systems are chosen for their reliability. On the other hand, accuracy and manoeuvre velocity are limited by conditions of space environments, variable during time and not directly controllable. Most used passive control systems use one or more of these three principal methodology: *Spin Stabilization*, *Magnetic Stabilization* e *Gravity Gradient*.

1.4.1 Gravity Gradient

Gravity gradient passive control systems take advantage of properties of elongated shape bodies which, in a gravitational field, will tend to stabilize with their minimum inertia axis in vertical orientation. As a consequence of this peculiarity, satellites that use this system of pointing are bounded by geometry and layout of masses.

To overcome this restriction, once the satellite has reached the desired orbit, two booms can be extracted at the end of one of the axes to create a minimum inertia axis, independent from the starting shape of the body. However, this system needs two motors for the extraction of booms, so the system becomes unavoidably more complex. With this method we cannot control the rotation of the satellite around the principal axis. To solve this problem, it can be used as an active component, like a momentum wheel orthogonally disposed to the principal axis, to contrast the undesired rotation. Moreover, oscillatory movements could arise around the axis which is perpendicular to the orbit plane

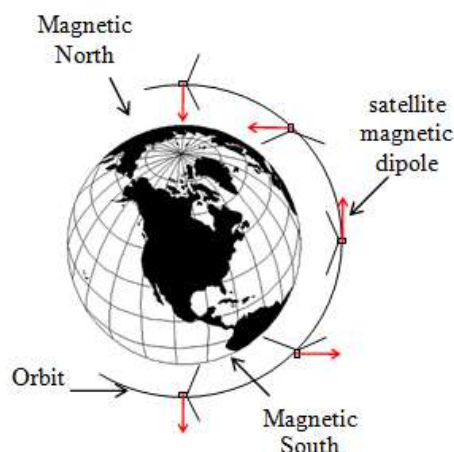
as a consequence of a perturbation. These are caused by the not perfect sphericity of Earth or by satellite's structures that are not perfectly stiff or have internal imbalance. These phenomena can be particularly disturbing if they will not be damped by particular devices as RW. This system is profitable if the pointing on the celestial body is in a nadir or zenith configuration. The precision of pointing is between +5 and -5 degrees. It depends strongly on external conditions in which the satellite operates, so also by the celestial body around which it orbits. Indeed, thanks to the external forces, this method can be used only in low orbits in which control couples exploited are however low.



Accuracy	$\pm 5^\circ$
Advantages	Simple, excellent for nadir pointing
Disadvantages	Geometry bound
	Efficient only in low orbits
	Low control couples
	Oscillatory mode caused by perturbations

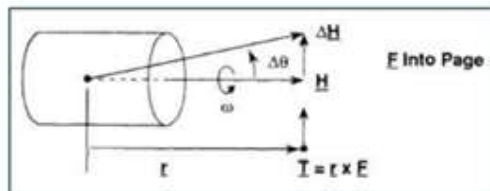
1.4.2 Magnetic Stabilization

This passive pointing method is implemented by inserting permanent magnets. The magnets align the satellite along the Earth's magnetic field lines. This method results strongly connect to the shape of the earth's magnetic field. Therefore, near the poles there is an overturning of the satellite pointing, whereas at the equator there is the stabilization of only one axis. In both cases, the control torques are minimal and feasible only in certain orbits. Moreover, it is necessary to provide for systems to dampen the oscillations deriving from the response to disturbances. This type of control is particularly useful in satellites in equatorial orbit, where the magnetic field is almost constant in intensity and direction.



Accuracy	$\pm 5^\circ$
Advantages	Semplicity
Disadvantages	Dependence on Earth's magnetic field
	Efficient only in low orbits
	Oscillatory way in response to disturbances

1.4.3 Spin Stabilization



$$\frac{dH}{dt} = T = rF \cong \frac{\Delta H}{\Delta t}$$

$$\Delta H = 2H \sin\left(\frac{\Delta\theta}{2}\right) \cong H\Delta\theta = I\omega\Delta\theta$$

$$\Delta\theta \cong \frac{rF\Delta t}{H} = \frac{rF\Delta t}{I\omega}$$

The higher the angular momentum value, the smaller the perturbation angle $\Delta\theta$ that a given disturbance torque will introduce.

Spin stabilization is a very simple passive system that exploits the property of gyroscopic stiffness, when the satellite is released into orbit the satellite has a rotation motion around its own axis, called spin motion, in order to stabilize it. When a torque is applied, with components in the two non-rotating axes, the body instead of assuming an angular acceleration begins to move with a constant angular velocity. At this point a nutation angle is formed and a dumper will remove this angle, in a lack of time of seconds or minutes, until the axis of rotation and the angular momentum coincide again. H is the angular momentum and is given by the product of the moment of inertia, with respect to the rotation axis, for the angular velocity. For H to be large it must have a large inertia or rotate very quickly, in fact bodies with a large H are usually disc shaped. The spinning motion is stable if the spacecraft rotates around the axis which has the highest moment of inertia value. The problem with this type of spacecraft is that its mass properties must be controlled to ensure the right direction and stability of motion; moreover, non-zero angular momentum requires more fuel to reorient the aircraft, reducing the effectiveness of this type of stabilization for objects that need to be oriented frequently. Although very effective and accurate in the case of telescopes, it is difficult to apply because of the rotation of the body itself, which causes intermittent pointing. To overcome this problem dual spin stabilization can be used, in which you have a high-speed rotor on a low speed rotating coaxial platform. The platform is used for the assembly of the instruments that must maintain the pointing. Pointing levels remain similar to the previous one. The advantage, however, is in the continuous pointing unlike the spin. On the other side, it will be necessary to check the reliability of the mechanical joint of the electrical connection system between rotor and platform.

Accuracy	$\pm 0.1^\circ - \pm 1^\circ$
Advantages	Good accuracy
Disadvantages	Constraints on the geometry of the masses of the system.
	Intermittent pointing possibility.
	Requires fuel for repointing manoeuvres.

1.5 PASSIVE DEPLOYMENT OF SPACE TELESCOPES

Passive deployment relates to the field of the mechanisms for deployment of space tethers from earth orbiting spacecraft or satellite carriers and in particular to the class of expendable tether mechanisms for passive on orbit deployment of end-masses tethered by a long space tether and that do not require to be retrieved. Deployment mechanisms remain in a stowed state during the launching process and deploy when the satellite reaches its target orbit. Research on these mechanisms has focused on minimizing satellite payload volumes. These structures can maximize the storage efficiency of a launch vehicle, enabling the launch of multiple satellites.

1.5.1 Spring Hinge

A high precise deployment mechanism for a deployable space telescope to facilitate satellite miniaturization is represented by a mechanism utilizing a spring hinge. This solution uses elastic force stored in spring to produce a relative motion of hinge, able to deploy structure in their required final position. This method can produce alignment errors because isn't controllable during deployment and allows only the designed structure's motion. To minimize this alignment errors a customized module or an assembly jig can be designed for a specific project. Usually to confirm the feasibility of the designed mechanism, three alignment errors that influence the optical performance of the structure-tilt, de-center, and de-space are theoretically analysed for various degrees of chosen model and final for the full model. Based on the theoretical analysis, a deployment mechanism of a simplified model is fabricated, and the alignments of the deployment mechanism are experimentally examined with a measurement platform consisting of some non-contact-based laser displacement sensors to measure error. In addition, the influence of gravity on the alignment error is analysed and compensated by investigating the tendency of the alignment error.

Deployment mechanisms generally work using the relative motions of the mechanism components, therefore there is joint clearance to allow the motion. An example of this solution is represented below. The spring hinge deployment mechanism consists of upper and lower panels on which the secondary and primary mirrors are installed, there are eight different customized linkages connecting the two panels, four spring hinges that drive the mechanism, and eight supporting modules preventing linkage twisting. Initially, in the stowed state configuration, the restoring force of the spring hinge was suppressed by the separating device. When the satellite reaches the wanted position, the separating device is released for deployment and then the linkages are deployed by the restoring force of the four spring hinges. When deployment is completed, the linkage is positioned upright at 90° as shown in figure.

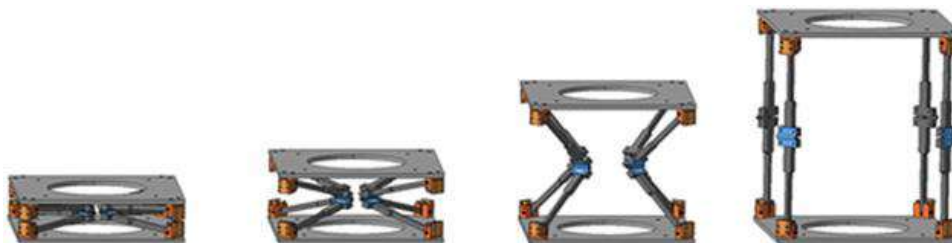


Figure 1.35. Deployment sequence concept

This method is particularly efficient and reliable, in fact it is composed of a very simple structure and the entire deployment phase doesn't need an active control on mechanical parts. There is also a reduction on the failure rate because the deployment of the structure is totally related to a structural property of the hinges.

1.5.2 Shape Memory Alloys

Shape Memory Alloys (SMAs) are interesting for space applications because they are able to recover large deformations by generating high specific force, with reasonable energy consumption and high precision movement, a useful property of the atomistic nature of the austenite-martensite crystalline phase transformation. Even if considerable advancements have been made in the evolution of reconfigurable, tuneable, and actuation material systems, the innovation of this technology is represented by our ability to model the thermo-mechanics of the SME (Shape Memory Effect).

SMAs components represent the primary alternative to pyrotechnic devices. Applications include the back-up boom release on-board the ISEE-B Spacecraft (1977), a solar array bearing pin off-load and unlatching mechanism for the Hubble Space Telescope solar panels (1990), and a solar panel release on-board the Clementine Spacecraft (1994). Just as these vehicles incorporated the SME, deployment and inservice challenges of gossamer spacecraft will further showcase the leverage SMA technology extends to structural reconfiguration and tuning.

The objectives of this research were to develop and refine an SMA modelling approach, the MTR subroutine, to predict the transient response of a spaceborne membrane optic SMA deployment actuator spine system as we can see in the figure below. The motivation for this figure acquisition concept can be found by studying thermonasty, or plant movement response to temperature. Tensioning a compliant concentrator with thermally activated spines falls under the same rubric as the blooming of a flower as we can see in the figure below.

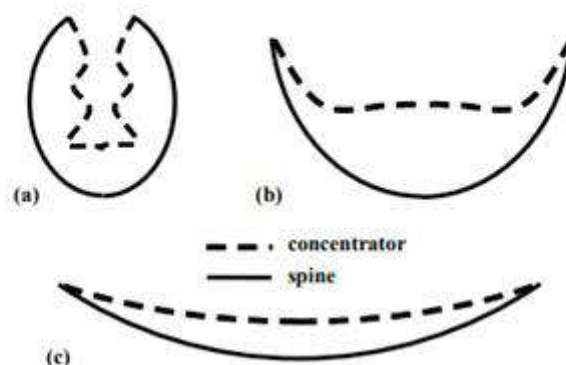


Figure 1.36. Deployment sequence concept

The deployment sequence shown is completely related with the SME of the Shape Memory Alloys and then the whole process is passive, without any active control or power consumption.

1.6 FUTURE TRENDS

Space agencies from all over the world are working on future space missions and some of them regard new space telescopes, in order to expand our knowledge of the whole universe. The main space telescopes that will be completed and launched in the coming years are presented in the following section.

1.6.1 James Webb Space Telescope

The James Webb Space Telescope, called also JWST, is a [space telescope](#) planned to [succeed](#) the [Hubble Space Telescope](#) as [NASA](#)'s flagship astrophysics mission. The JWST and its modern infrared cameras will provide improved infrared resolution and sensitivity over Hubble, and will enable a broad range of investigations across the fields of [astronomy](#) and [cosmology](#). One of the main tasks will be the observation and the study of some of the most distant events and objects in the [universe](#), such as the [formation of the first galaxies](#).

The JWST is being developed by a collaboration between [NASA](#), ESA and the [Canadian Space Agency](#) and is named for [James E. Webb](#), who was the [administrator of NASA](#) from 1961 to 1968 and played an integral role in the [Apollo program](#).

Its development began in 1996 for a launch that was initially planned for 2007 but the project has had numerous delays and cost overruns and underwent a major redesign in 2005. The JWST's construction was completed in late 2016, after which its extensive testing phase began and finally its launch date is programmed for the 31 of October 2021.

The James Webb Space Telescope will operate beyond the Earth's atmosphere at a distance of approximately 1,500,000 kilometres beyond Earth's orbit, near the Lagrange point, so it will be able to complete a large number of complicated analysis, also thanks to its infrared cameras. It will operate at a temperature of about 50 K ($-223.2\text{ }^{\circ}\text{C}$), necessary for infrared observations.

Telescope's main characteristic is represented by its incredible optical system, composed of a very large primary mirror that is a 6.5-meter-diameter gold-coated beryllium reflector with a collecting area of 25.4 m^2 . The entire mirror has a diameter too big to be contained in the payload capsule of an Ariane V rocket and then it is not a single mirror, but it is composed of 18 hexagonal segments which will unfold after the telescope is launched. Then the telescope is a deployable telescope and it will reach its fully open configuration only 6 months after the launch. The mirror has a system of active optic that uses 126 small motors to occasionally adjust the optics as there is a lack of environmental disturbances of a telescope in space.

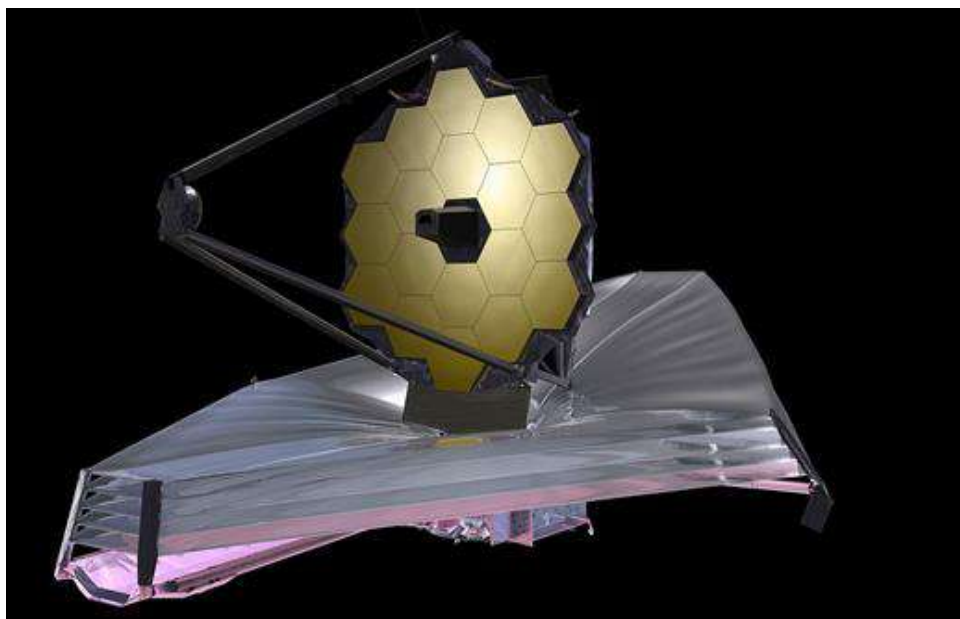


Figure 1. 37. James Webb Space Telescope in operative configuration

As shown in figure, the telescope has a five-layer sunshield, in order to maintain the optical module at the right operative temperature. Each thin layer is composed by Kapton E, a particular polyimide film, with membranes specially coated with aluminium on both sides and doped [silicon](#) on the Sun-facing side of the two hottest layers to reflect the Sun's heat back into space.

The [Integrated Science Instrument Module](#) (ISIM) could be considered the brain of the satellite, it provides electrical power, computing resources, cooling capability as well as structural stability to the Webb telescope and it is made with bonded graphite-epoxy composite. It is composed by four main scientific instruments:

1. NIRCam (Near InfraRed Camera)
2. NIRSpec (Near InfraRed Spectrograph)
3. MIRI (Mid-InfraRed Instrument)
4. FGS/NIRISS (Fine Guidance Sensor and Near Infrared Imager and Slitless Spectrograph)

Thanks to the new technologies and the huge work done on this new telescope, we will be able to discover and to understand unexplored areas of the universe.

1.6.2 LUVOIR

Large UV Optical Infrared Surveyor, simply called LUVOIR, is essentially a beefed-up version of the [Hubble Space Telescope](#). Like Hubble, this instrument would observe the universe in ultraviolet, infrared and visible wavelengths of light.

However, with a diameter of about 15 meters, LUVOIR's mirror would be more than six times wider than the one in Hubble. This means that LUVOIR would see the universe with six times the resolution of Hubble. And with 40 times the light-gathering power of the older telescope, LUVOIR would see fainter, smaller and more-distant objects.

NASA has come up with two different options for LUVOIR's design. The larger version, LUVOIR-A, would be built to launch on NASA's upcoming [Space Launch System](#) (SLS) mega rocket, at the moment it represents the maximum dimensions that could be fitted on a rocket.

There is a second prototype version, called LUVOIR-B that would have a mirror with a diameter of 26 feet (8 m) and could be used with the actual rockets, and the smaller size would entail a slightly lower resolution than for LUVOIR-A.

LUVOIR is designed to tackle a variety of astronomical research projects, like searching for habitable [exoplanets](#); studying the formation and evolution of stars and galaxies; mapping dark matter throughout the universe; and imaging objects in the solar system, like planets, comets and asteroids. Similar to the James Webb Space Telescope, LUVOIR will have a composite modular primary mirror that will be deployed in deep space during the mission.

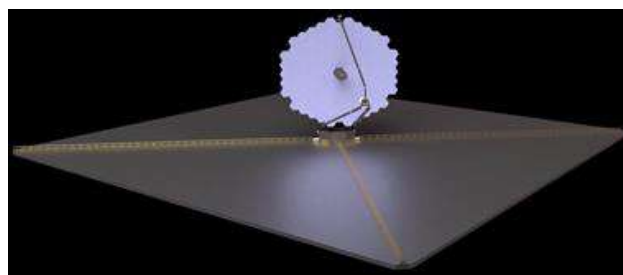


Figure 1.38. LUVOIR with his huge modular primary mirror

1.6.3 HabEx

The Habitable Exoplanet Observatory, shortly called HabEx is designed to observe potentially habitable exoplanets around [sun-like stars](#).

While looking for "biosignatures" like water and methane, which may indicate the presence of life on another planet, HabEx would also become the first telescope to directly image an Earth-like exoplanet. To be considered potentially "Earth-like," an exoplanet must be terrestrial, or rocky, and must orbit its parent star in the habitable zone, where the temperature is just right for liquid water to exist.

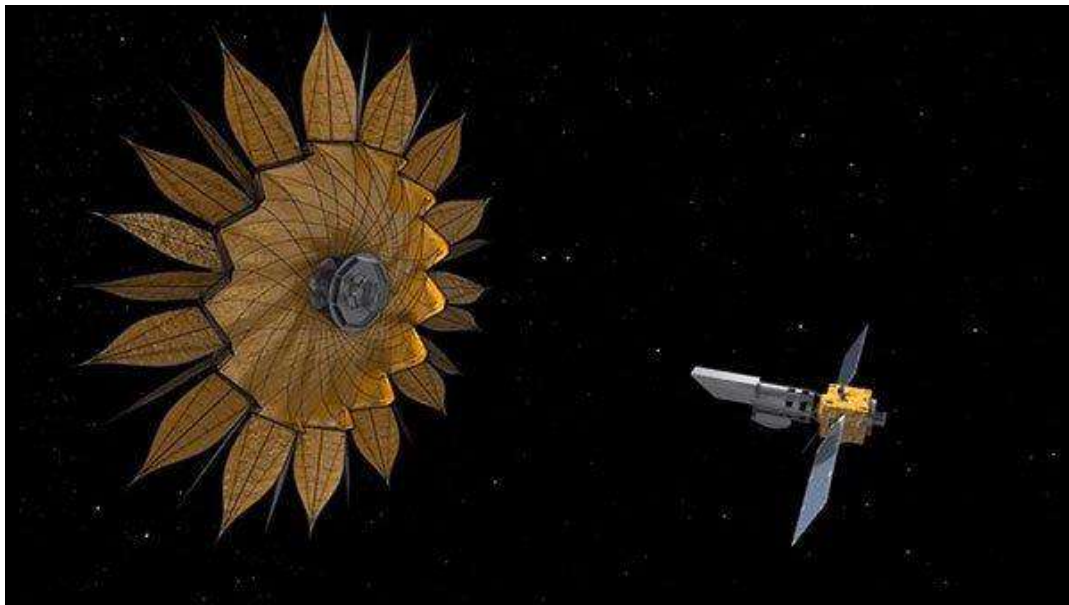


Figure 1. 39. HabEx will use a star shade to observe exoplanets around bright host stars.

The main innovation of HabEx from the past is represented by a large, sunflower-shaped "starshade", that will be deployed to block light from stars that have planets, allowing the telescope to study faint exoplanets in unprecedented detail. The HabEx telescope itself would have a diameter of 4 to 8 m, it isn't decided yet because NASA is still studying different design options with different sizes, but the starshade would be far larger, with a diameter of 72 m. HabEx will take advantage of the origami technique in order to stow and deploy the starshield.

In addition to collecting visible-light images, HabEx would also conduct ultraviolet and infrared observations of the cosmos, making this observatory useful for more than just exoplanet research. Using the same instruments that it would employ for studying exoplanets, HabEx could also observe and map stars and galaxies, study the expansion of the universe, and investigate [dark matter](#).

1.6.4 Lynx X-Ray Observatory

A potential successor to NASA's [Chandra X-Ray Observatory](#) is Lynx, a proposed space telescope that would uncover the "invisible" universe by detecting high-energy X-ray radiation that is not visible to the human eye. This means researchers could use the instrument to look for things like supernovas and black holes.

Lynx was designed to peer through space and time to look at the earliest black holes in the universe, allowing researchers to better understand how these objects form and grow. The telescope would also observe the formation and evolution of galaxies and galaxy clusters in order to understand more of our past.

It would also be able to investigate the birth and death of stars and capture high defined maps of exploding stars, like [Chandra did with its image of Tycho's supernova](#), but with 100 times the resolution of Chandra, Lynx would produce even more-spectacular images thanks to its improved primary mirror. While Chandra can study stars located up to about 1,300 light-years away, the instruments on Lynx would see more than 16,000 light-years away, or 12.5 times the distance.

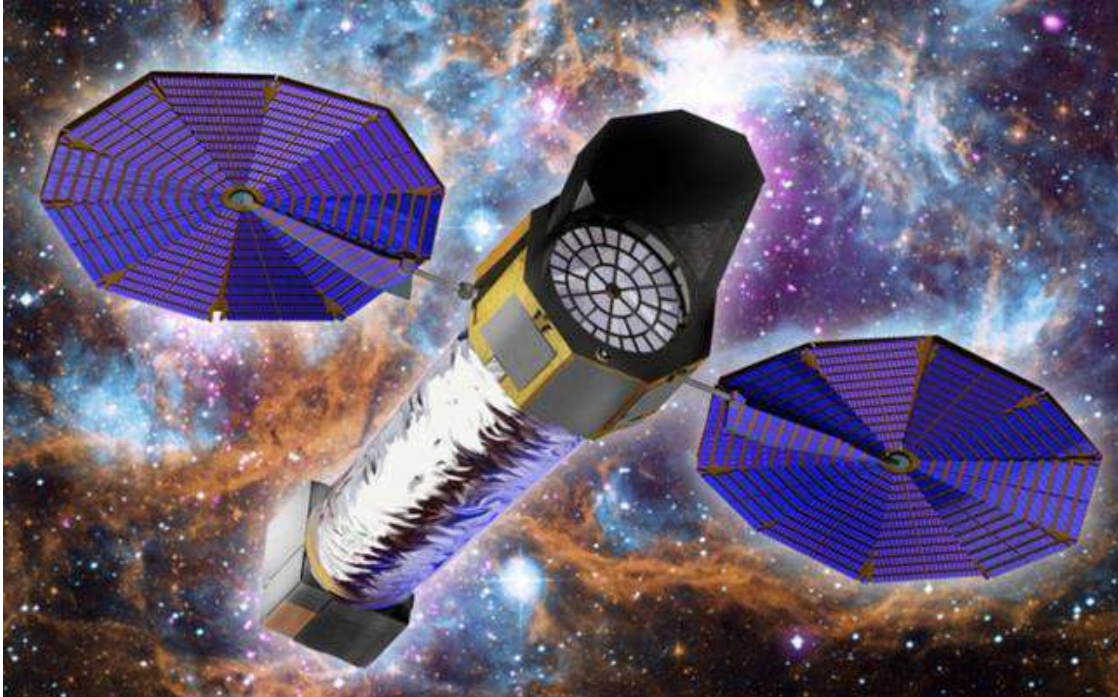


Figure 1. 40. Lynx concept art

With a primary mirror diameter of about 3 m, Lynx would be only slightly wider than Hubble. However, the opening of the tube-shaped telescope would be five times larger than Chandra's, which measures only 1.2 m in diameter.

1.6.5 Origins Space Telescope

Origins Space Telescope (OST) is a concept study for a far-Infrared Surveyor space telescope mission which seeks to answer the big mysteries of life in the universe, like how habitable planets are formed. The Origins Space Telescope would help scientists break down the steps in that process by tracking the ingredients for life from the earliest stages of star and planet formation.

This far-infrared surveyor mission would be able to peer through obscuring dust clouds to get a clear view of stars and exoplanets in star-forming regions. It could be considered a next-generation version of the [Herschel Space Observatory](#), a European mission that observed the universe in infrared for four years before shutting down in 2013.

The Origins Space Telescope would have a modular primary mirror with a diameter of about 15m and then it would be about the same size as LUVOIR and four times the size of Herschel. Like Herschel, this proposed telescope would require a special "cryocooler" system to keep its instruments from getting too hot. Working with a temperature of the optics of about 4 °K, the telescope would increase its sensitivity and the telescope could be up to 1,000 times more sensitive than any other infrared telescope launched to date.

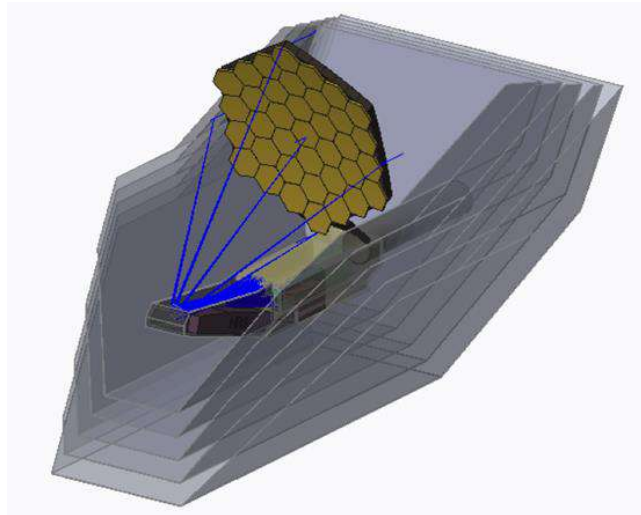


Figure 1.41. Origins Space Telescope concept art

1.6.6 WFIRST

Formerly the Wide Field Infrared Survey Telescope or WFIRST) but also known as Nancy Grace Roman Space Telescope (shortened as Roman or the Roman Space Telescope), is a NASA [infrared space telescope](#) currently under development. It will be launched in the middle of '20 and will substitute the Hubble Space Telescope. The science objectives of Roman aim to address cutting-edge questions in [cosmology](#) and [exoplanet](#) research, including:

- Answering main questions about [dark energy](#), complementary to the [European Space Agency \(ESA\) EUCLID](#) mission. Roman will use three independent techniques to probe dark energy: baryon acoustic oscillations, observations of distant [supernovae](#), weak gravitational lensing.
- Completing a census of [exoplanets](#) to help answer new questions about the potential for [life in the universe](#). The census would also include a sample of [free-floating planets](#) with masses likely down to the mass of Mars.
- Establishing a guest investigator mode, enabling survey investigations to answer diverse questions about our galaxy and the universe.
- Providing a [coronagraph](#) for exoplanet [direct imaging](#) that will provide the first direct images and spectra of planets around our nearest neighbours similar to our own giant planets.

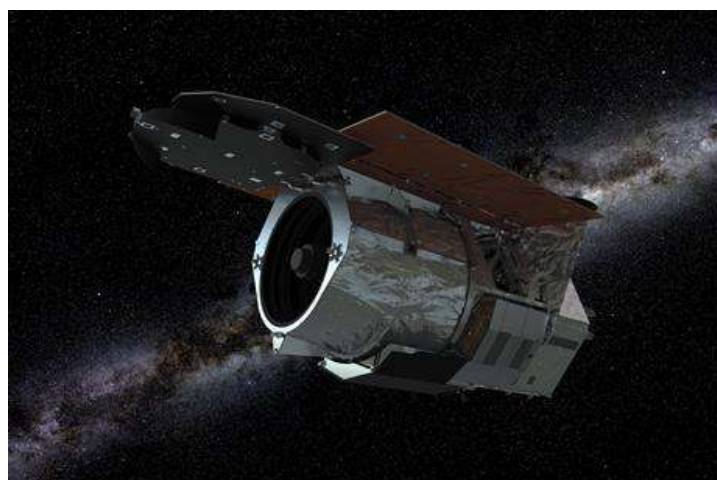


Figure 1.42. WFIRST concept art

Roman will have two particular instruments. The first is the Wide-Field Instrument (WFI) that is a 300.8-megapixel camera providing multi-band visible to near-infrared (0.48 to 2.0 micrometers) imaging using one wideband and six narrowband filters. It also carries both high-dispersion prism and low-dispersion prism assemblies for wide-field slitless spectroscopy. The second instrument is a high contrast coronagraph covering shorter wavelengths (0.5 to 0.8 micrometers) using dual deformable mirror starlight-suppression technology. It is intended to achieve a part-per-billion suppression of starlight to enable the detection and spectroscopy of planets as close as 0.15 arcseconds away from their host stars.

2 BOOMS: MATERIALS AND CONFIGURATIONS

2.1 WOVEN MATERIALS

The concept of deployable structures brings in the aerospace industry new types of materials, one that certainly needs to be discussed are **woven composites**.

Fiber composites have gained increasing popularity over metal alloys due to their high strength to weight ratio and the wider range of design possibilities offered by tailorable material properties. **Textile composites** provide better performance in intra- and inter-laminar strength and damage tolerance over composites made from unidirectional lamina.

The two basic constituents of textile composites are the *tows*, i.e. continuous strands of fibres, and the *matrix* which keeps them together. There are two sets of interlacing tows acting as reinforcement. The weave style controls the draping, surface smoothness and stability of the fabric.

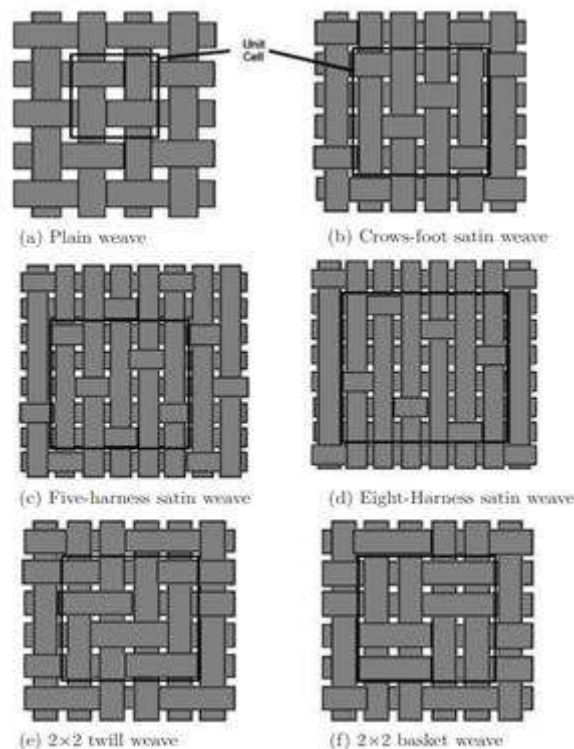


Figure 2.1. Examples of woven fabrics

In general, the lamina properties are estimated using the rule of mixtures while the laminate properties are calculated with the **classical lamination theory (CLT)**. However, in the case of textile composites it is difficult to use simple analytical models due to their complex architecture. *Soykasap* showed that such estimates can result in errors of up to 200% in the maximum bending strains or stresses, and up to 400% in the bending stiffness.

Researchers have used **finite element analysis** to understand the behaviour of textile composites by modelling a *representative unit cell (RUC)*. *Karkkainen* presented a direct micro-mechanical based

finite element model for the analysis of a single-ply plain weave laminate (Fig. 2.3.). This model considers bending effects which had been ignored in the conventional models by assuming a uniform stress state in the unit cell. The *RUC* is modelled as a linear-elastic thin Kirchhoff plate whose properties are defined by a homogenisation technique and periodic boundary conditions are assumed at the boundaries. The constitutive relationship for the homogenised plate is:

$$\begin{Bmatrix} N_x \\ N_y \\ N_{xy} \\ \hline M_x \\ M_y \\ M_{xy} \end{Bmatrix} = \begin{pmatrix} A_{11} & A_{12} & A_{16} & | & B_{11} & B_{12} & B_{16} \\ A_{21} & A_{22} & A_{26} & | & B_{21} & B_{22} & B_{26} \\ A_{61} & A_{62} & A_{66} & | & B_{61} & B_{62} & B_{66} \\ \hline B_{11} & B_{21} & B_{61} & | & D_{11} & D_{12} & D_{16} \\ B_{12} & B_{22} & B_{62} & | & D_{21} & D_{22} & D_{26} \\ B_{16} & B_{26} & B_{66} & | & D_{61} & D_{62} & D_{66} \end{pmatrix} \begin{Bmatrix} \varepsilon_x \\ \varepsilon_y \\ \gamma_{xy} \\ \hline \kappa_x \\ \kappa_y \\ \kappa_{xy} \end{Bmatrix}$$

Figure 2. 2. Constitutive Relationship for the homogenized plate

where N and M denote force- and moment-resultants, while ε and κ denote mid-plane strain and curvatures. γ denotes the engineering shear strain. The 6×6 constitutive matrix is denoted by ABD and its 3×3 submatrices are denoted by A , B and D .

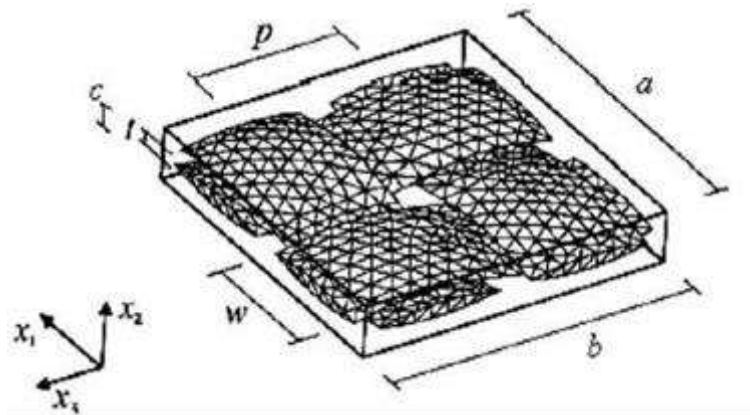


Figure 2. 3. Representative unit cells for plain-weave and tri-axial weave laminates

2.2 FABRICATION PROCEDURES

Before going deeper into the materials used for these deployable structures, let's focus on the fabrication procedure of a composite tube made of two-ply plain weave carbon fibre laminates.

2.2.1 Resin impregnation

At this stage is important to be careful when working with dry fabric as it can be easily distorted. The edges of the required piece of fabric were marked with masking tape and cut along the centre lines. Then the fabric was smoothly laid on top of a semi solid film of *913 resin*, provided on a release paper. The film was released from the paper by ironing over the surface. Next the resin impregnated fabric with release paper was cut to the exact dimensions and refrigerated. Once the resin had solidified the release paper could be smoothly peeled off.

2.2.2 Lay-up

A 38 mm diameter stainless steel tube was used as a male mould for constructing the composite tubes, while the surface was cleaned with acetone and dried before later operations. To facilitate the release of the cured tube from the mandrel a layer of *PTFE* based dry film mould release was sprayed onto the steel tube (*Sprayon MR311* was used). The resin impregnated fabric was laid on top of a *Wrightlon 5200B P-3* perforated release film and carefully rolled onto the sprayed male mandrel. Next, the lay-up was inserted into a 50 mm diameter 2:1 heat shrinkable tube and heated using a heat gun set to 300°C. Then the entire component was covered with another release film and then with a breather blanket to facilitate the flow of air around the tube and so to make it permeable to create uniform vacuum.

2.2.3 Vacuum Stage

Two different arrangements were tested for applying vacuum to the tubes. The first was to place the whole lay-up on top of a flat steel plate, to cover it with a *Wrightlon 7400* vacuum bag, and to seal it with tape sealant applied all around (Fig 2.4a). In the second method the vacuum bag was placed around the tube and sealed to itself with tape (Fig 2.4c). This allows the vacuum bag to fully cover the tube and hence avoids any distortions (Fig 2.4d).

Vacuum injection molding might be a better alternative because it shall minimize delamination at the edges, especially using two molds instead of only one, so that might improve the surface finish and repeatability.

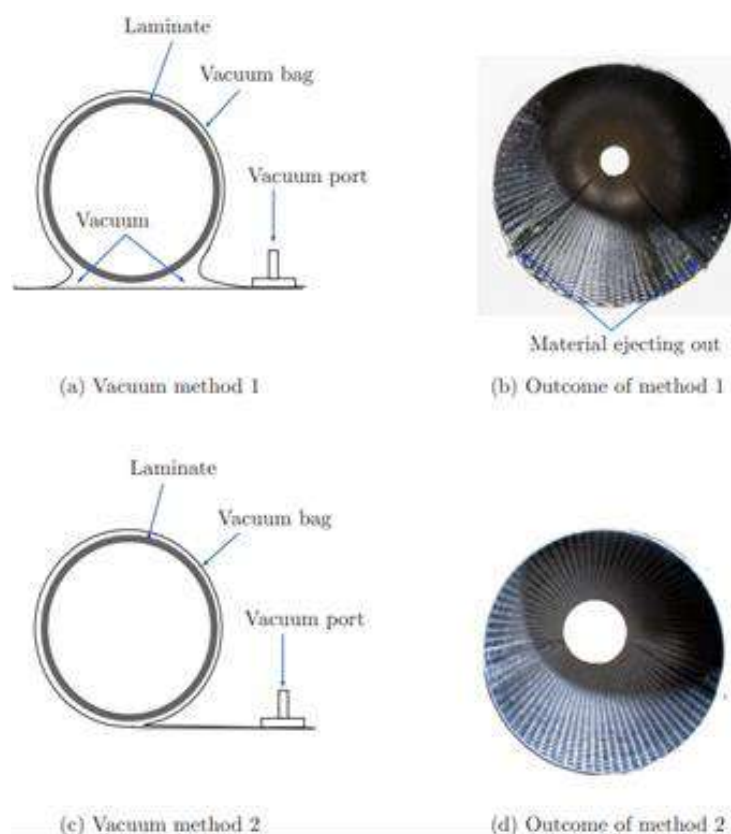


Figure 2. 4. Different vacuum bagging techniques

2.2.4 Autoclave curing

Two thermocouples were attached to the autoclave and a temperature-controlled curing process was carried out. The laminate curing process is determined by the curing cycle of the resin used. After

cooling down, all wrapping materials were removed, and the composite tube was pulled out from the mandrel by twisting the tube.

2.2.5 Hinge fabrication

Standard milling machines as slow rpm rotary tools tend to introduce micro cracks. Therefore, a *Dremel 400 XPR* high speed rotary tool set to $25'000\text{ rpm}$ was used for machining. Straight portions of the slot were cut with a diamond cutting tool and the curved parts were ground with silicon oxide or carbide grinders. After cutting the edges were carefully smoothed with sandpaper.

2.3 COMMON MATERIALS FOR DEPLOYABLE STRUCTURES

Inside a launch vehicle, the satellite or payload must be compact and light, but when it reaches space it should be as large as needed for the mission.

There are three main criteria leading the development of such structures: mass, stiffness and volume. When the tape is coiled the material stores *strain energy* thanks to bending and stretching. During the deployment phase, the tape spring goes back to a practically unstressed configuration, creating a *high stiffness tubular shape*, typically overlapping to increase the shear and torsional stiffness. The strain energy released by the tape spring drives the deployment.

A large number of developed and conceptual architectures are making such advancements with the implementation of flexible Fiber Reinforced Polymer (FRP) composite laminate components in place of traditional hinge and latching mechanisms. FRP composite laminates are the preferred material because of the increased strain capability, higher modulus, lower mass, and increased dimensional stability over conventionally used metallic materials for flexure applications.

The velocity of the deploying end of the boom accelerates and suddenly stops when the deployment is completed. This impact increases the risk for boom failure or even spacecraft damage. Once the tape spring is entirely deployed, a locking system maintains the deployed state and the whole structure becomes rigid. Materials commonly used for the shells are *Beryllium Copper*, *Stainless Steel*, *Glass Fiber Reinforced Plastics (GFRP)* and *Carbon Fiber Reinforced Plastics (CFRP)*.

2.3.1 Materials models and stability

As mentioned in previous chapter, to make the connection between stress and strain, the laminate theory uses the ABD matrix. More precisely, it relates the mid-surface strains and curvatures to the corresponding stress resultants in the shell.

$$\begin{bmatrix} N \\ M \end{bmatrix} = \begin{bmatrix} A & B \\ B & D \end{bmatrix} \begin{bmatrix} \epsilon^0 \\ \kappa \end{bmatrix}$$

To define the quantities that take place in the matrix, are used the laminate plate equations. These equations define the force and moment in plate.

$$N = \int_{-h/2}^{+h/2} \sigma \, dz$$

$$M = \int_{-h/2}^{+h/2} \sigma z \, dz$$

As for the stable configurations of the boom, the one that is more stable is when the boom is not coiled as there is no energy stored. When it is coiled, the local energy minimum contains non-zero energy. The transition between the less stable state to the more stable state is feasible by adding enough energy called activation energy. So, the stored strain energy will be released into kinetic energy during the snap-through, which is interesting to control.

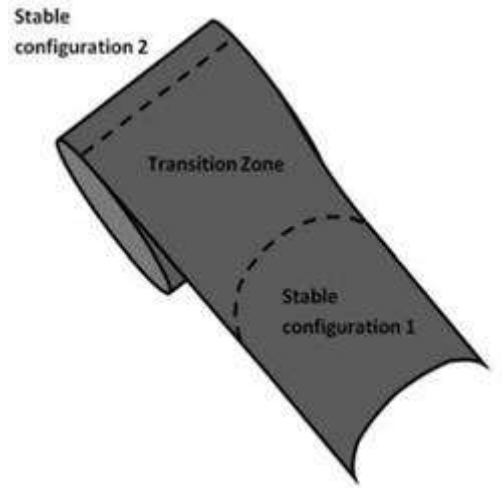


Figure 2. 5. Stable configurations of deployable booms

Bi-Stability in structural mechanics is a quite young field which consists in the capability of changing the Gaussian curvatures from one position to another. The tape springs are bi-stable when they have *two stable configurations* which correspond to two different local minima, from the stored strain point of view.

Some material characteristics examples:

Properties	Value	Chosen mechanical properties	Value
Nominal weight	48 g/m ²	Young modulus E_1	26.6 GPa
Warp / Weft	56 / 44 %	Young modulus E_2	2.97 GPa
Weave style	Plain	Shear modulus G_{12}	1.39 GPa
Finish	TF970	Major Poisson ratio	0.4
Thickness	0.04 mm	Minor Poisson ratio	0.045

Table 2.1: Glass Fiber Properties

Name of the solution	Mass proportion	Mechanical properties	Value
Dion 9102	100	Tensile strength	79 MPa
Accelerator 9802	2	Tensile modulus	3400 MPa
Accelerator 9826	0.5	Tensile elongation	4.5 %
Inhibitor 9853	0.2	Flexural strength	130 MPa
Norpol peroxide 11	2	Flexural modulus	3250 MPa
Gel time, in minutes	43	Heat distortion temperature	100 °C

Table 2.2. Vinylester Properties

The configurations used are usually [+45F/-45F/0/+45F/-45F] layup and [+45F/-45F/+45F/-45F/0/+45F/-45F/+45F/-45F] layup.

2.3.2 Flexible Composite Tape Springs (FCTS)

Air Force Research Laboratory Space Vehicles directorate has developed a general composite laminate construction that is dimensionally stable, exhibits high stiffness and can be subjected to high strains, which are essential characteristics for this class of tape spring applications.

The laminate is composed of two plies of *Hexcel Hexply IM7 12K/8552* unidirectional prepreg oriented at 0° (parallel to the long axis of the tape spring) sandwiched between two plies, oriented at 45° , of *JPS AstroQuartz II Style 525* plain weave impregnated with *Patz PMT-F7 resin*. While simple, these $[\pm 45^\circ/0^\circ/\pm 45^\circ]$ LAMINATES balance several conflicting requirements: they are high in axial stiffness, high in bending stiffness, have a very small coefficient of thermal expansion and are resistant to creep. Moreover, the unidirectional centre plies provide structural performance.

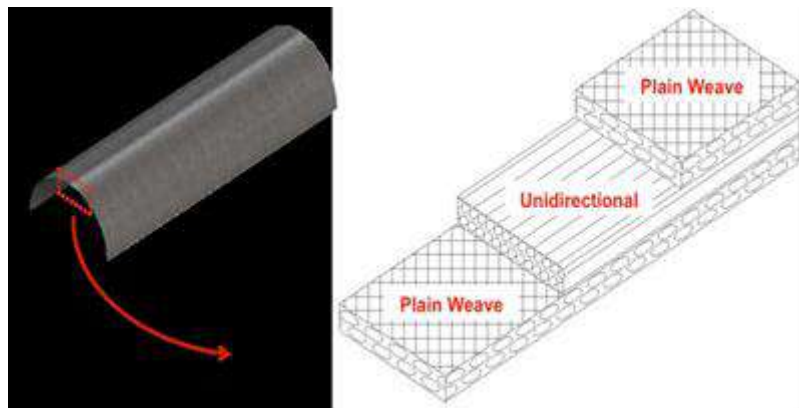


Figure 2. 6: Representation of FCTS

Property (units)	Astroquartz II/PMT-F7 Plain Weave
E_1 (GPa)	21.7
E_2 (GPa)	21.7
G_{12} (GPa)	3.4
ν_{12}	0.11
t (mm)	0.254

Table 2.3. Material properties

Property (units)	Astroquartz II/PMT-F7 Plain Weave	S2 Glass/PMT-F7 Unidirectional
E_1 (GPa)	21.7	56.9
E_2 (GPa)	21.7	16.8
G_{12} (GPa)	3.4	6.1
ν_{12}	0.11	0.27
t (mm)	0.1016	0.0762

Table 2.4. Laminate properties

2.3.3 Carbon Fiber Reinforced Polymer (CFRP)

Carbon Fiber Reinforced Polymer (CFRP) composite materials offer many advantages over conventional materials, such as lower mass, lower thermal expansion, anisotropic design flexibility and high stiffness/strength-to-weight ratio. This type of polymer can be categorized under two

families: *unidirectional laminates* (Fig. 2.7a), for low coefficient of thermal expansion (CTE), and *multidirectional laminates* (Fig. 2.7b), for plane structures.

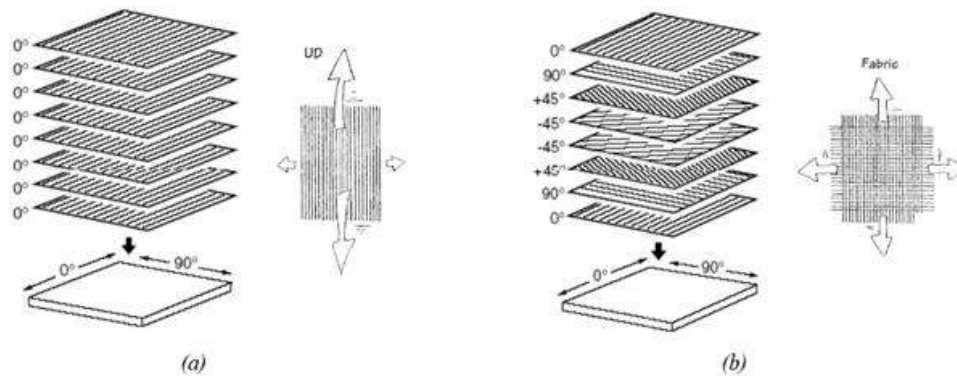


Figure 2. 7. CFRP laminates unidirectional (a) and multidirectional (b)

Both of configurations have very important properties:

- **Low mass**
With a density of around 1.6 g/cc , CFRP allows to have low inertial problems and low energy requirements.
- **Low Thermal Expansion**
CFRP materials exhibit low coefficient of thermal expansion (CTE), which implies less complexity in Mirrors and structure design, without the need for continuous corrections due to material expansion.
- **Anisotropic design flexibility**
With the unidirectional properties of CFRP, ideal properties can be applied in specific directions in-plane where it is needed, offering design flexibility and minimizing mass.
- **High Stiffness/Strength-to-Weight Ratio**
With High Stiffness/Strength-to-Weight Ratio, CFRP materials are ideal for stiff lightweight telescope structures, especially deployable systems that rely on the stiffness.

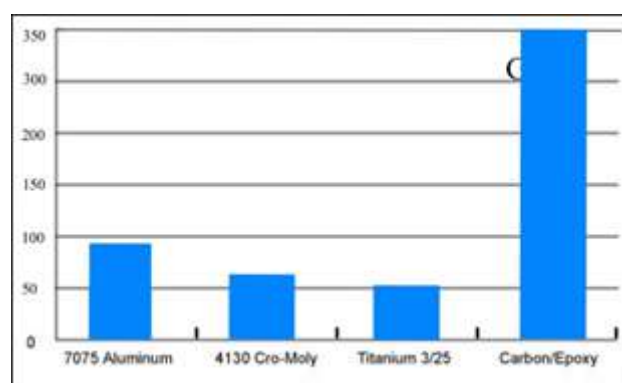


Figure 2.8. Stiffness to weight ratio

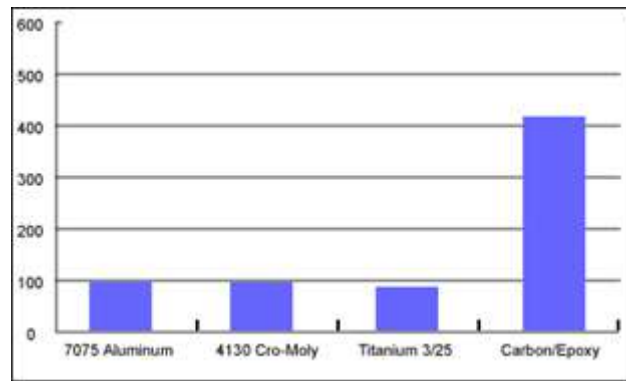


Figure 2.9. Strength to weight ratio

2.3.4 Large Strain Composite Materials

As already known, nowadays, the aerospace challenge in deployable structures is to balance material stiffness versus foldability (large strains) in the way to achieve both *compact packaging and efficient deployed performance* against mechanical, thermal and environmental loads with design flexibility. It is often desirable if the material modulus decreases during packaging because it reduces the need of energy required to be folded, because less strain energy is stored. Some companies developed carbon fiber based material systems with heat softenable matrix materials: when heated, the materials can be folded to very large strains repeatedly without degradation of their unfolded ability to resist buckling and general loads. The primary limitation of these materials is that they require *heating and cooling systems*: heating is needed in the packaging phase, while cooling is necessary to ensure that the structure remains cool enough to be rigid in orbit. Therefore, *thermal control system* is needed to guarantee the proper temperature condition, with the consequence of reducing mass efficiency.

For these reasons, a renewed interest in materials that do not require heating or cooling is now born: these are *passive large strain materials*.

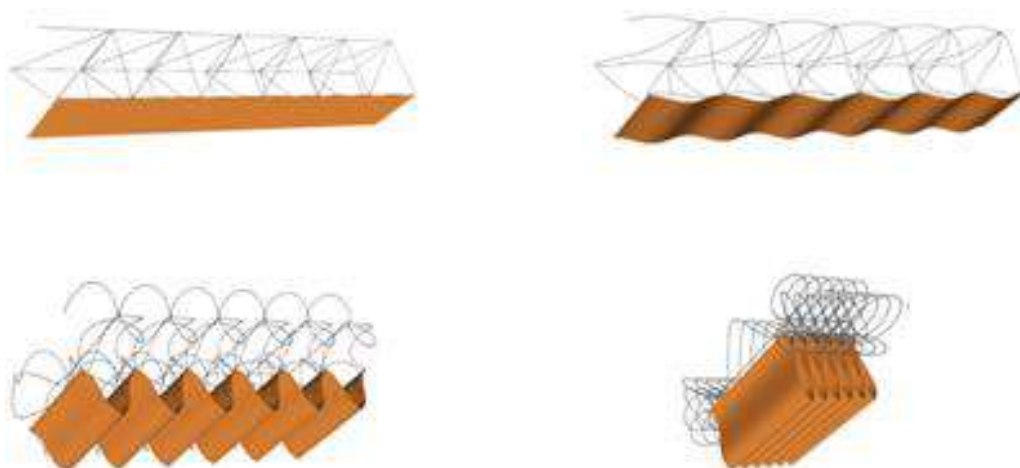


Figure 2. 10. Examples of Large Strain Materials

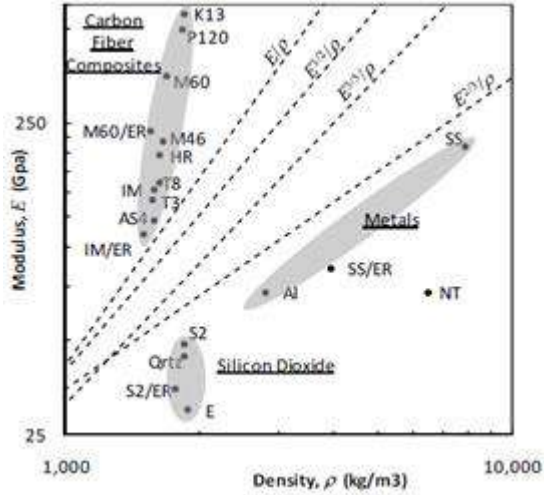


Figure 2.11. Modulus vs density plot

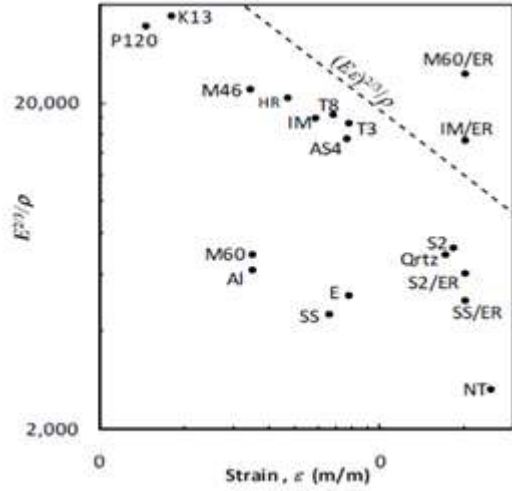


Figure 2.12. Strain vs density plot

Higher modulus, lower density, greater strain and greater strength are better for deployable structures but the selection of materials taking into account those several properties is challenging. In order to compare materials, analytically derived metrics were calculated: in the deployed configuration, material stiffness and density are important, while for deformed parts in packaging configuration, high strain is a “must”.

In the *Modulus-Density* plot (Fig. 2.11), materials towards the upper left corner of the plot (higher modulus and lower density) are better. In the specific *Density-Strain* plot (Fig. 2.12), materials towards the upper right (higher modulus and larger strain) are better.

As shown in the plots, *carbon fiber composites* are the best choice: neglecting them, considering metrics and application, S2 fiberglass or steel composites can be considered.

The best material depends on which metric is used to assess the materials:

- Metric for *beams application*: $\mu = E^{3/5}/\rho$, the beam is designed to meet a bending stiffness requirement and a bending strength requirement.
- Metric for *column application*: $\mu_c = E^{2/5}/\rho$, the column is designed to meet global and local longeron buckling requirements and must carry a compressive load over a length.
- Metric for *stiffness designed beam of arbitrary cross-section* in bending application: $\mu_b = E^{1/2}/\rho$, (suitable for the tape-spring booms also called STEM)
- Metrics for a *distributed strain continuous longeron* mast subject to a bending strength requirement and weight optimized (suitable for STEM): $\mu_b = (E\epsilon)^{3/5}/\rho$.

There are two different approaches to deployable: mechanical approach (sliding contact joints) or material deformation approach. The second one reduces freedom in material selection and deployed performance, but it is less expensive, because of the reduction of design costs associated to hundreds of hinges in a single structure (typical of mechanical approach). Moreover, in material deformation approach there are two possibilities of design:

- **Distributed strain (DS)**: the strains are distributed throughout the structure;

- **Concentrated strain (CS):** flexure hinges of high strain material in discrete locations and less flexible but more efficient materials elsewhere; it is a sort of mixture between mechanical and deployable approach.

The analysis of the plots has demonstrated the success of *carbon and S2 glass fiber composite materials* in relatively simple distributed strain deployable structures. Concentrated strain deployable structures that use less efficient flexible materials only in hinge locations have the potential for greater efficiency, design freedom and compact packaging. However, their implementation is limited by the low strain capacity of engineering materials.

2.3.5 Future Large Strain Materials

One approach to achieving passive (no heating required) large strain materials currently under investigation is to exploit microbuckling in composites with large strain elastomeric matrix materials. Under tensile or small compressive loads and before microbuckling, straight fibers will generate considerable axial stiffness and are suitable for deployed structures. With larger compressive loads, fibers will microbuckle but the elastomeric matrix will allow for *large effective compressive strains*. Moreover, microbuckling allows *large elastic strains* (approximately 4%) in this composite material while maintaining a *high compressive strength* (results are independent of fiber diameter). These new materials, once tested and proven, could be used for CS DESIGNS.

2.3.6 Deployment Repeatability Testing of Composite Tape Springs for Space Optics Applications

The use of carbon-fiber composites as tape-spring material increases the *stiffness-to-mass ratio*, *allows greater flexibility and customization* in their design, and takes advantage of *superior thermal expansion properties*. The combination of the tape-spring configuration and carbon-fiber-composite material is an optimal solution to the complex problem of selecting space hardware to maximize strength and stiffness while minimizing mass and packaging volume. Carbon-fiber-composite tape springs have the potential to be widely applicable in satellite systems and are particularly attractive to the space optics community driving for lighter, larger, and therefore more efficient and powerful telescopes.

Two types of tape springs are going to be investigated: one comprised *solely of carbon-fiber composite* and one that incorporates *thin metal strips at the hinges* to add robustness and increase the packaging ratio. In addition, two configurations are studied, as single springs in simple *2D bending tests* and in an inherently *stable 3D tripod configuration* simulating a *Ritchey-Chrétien-shaped Cassegrain telescope* (Fig. 2.13).

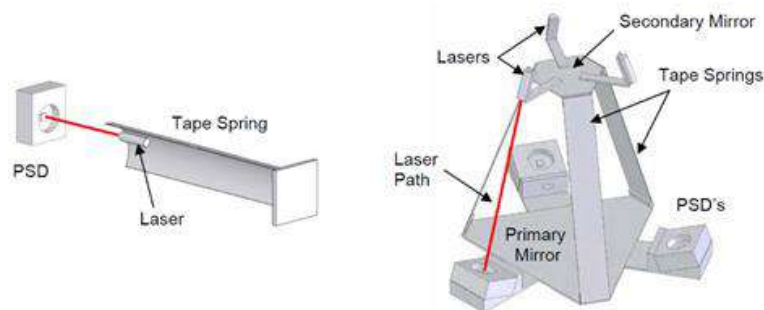


Figure 2. 13. Tape spring configurations: 2D (left) and 3D (right)

When the 2D tests of single tape-springs with one hinge in the middle of each spring are studied, it is demonstrated that the presence of the metal strip in the hinge of the tape springs leads to a significant improvement in repeatability. The metal strips achieve a three-fold increase in the precision of the

deployments, from 189 μm to 50 μm (the total displacement was calculated as the square root of the sum of the squares of the absolute averages). In almost all cases a similar steady-state value was observed with consistent recovery of stowage-induced strains. Even with slippage in the clamp (resulted from the bending of the spring), the data indicate that carbon-fiber-composite tape springs are capable of micron-level deployment precision after multiple stow-and-deploy cycles, returning to within 100 μm of their original positions on average. The data also indicate no significant performance degradation as a function of number of cycles.

The subsequent 3D tests show that the tape springs in the tripod configuration are more stable than they are individually. Unlike the 2D results, these data show little or no improvement in repeatability from using metal strips in the hinges. All data is well within the desired 100 μm bound for optical-level controllability. The average displacements in both hinges without and with metal strips were below 50 μm , a three-fold improvement from the single tape spring 2D configuration. The tape-spring tripod is more compliant in the lateral XY plane because the tape springs are much stiffer in axial compression than in lateral cantilever bending, and this increased compliance appears consistent with the wider range of final position.

2.3.7 EMC materials

Elastic Memory Composites materials are fiber reinforced composites with a polymer resin that exhibits shape memory properties. This thermoset shape memory resin that enables EMC materials to achieve much higher failure strains than traditional composites and much lower stored strain energy than traditional high-strain materials when exercised through a very specific thermo-mechanical load cycle. This unique behaviour is exploited in the development of EMC structural components that exhibit very high packaging-strain capability while guaranteeing predictable post-deployment structural performance. These high strains are induced by elevating the temperature of the EMC material above the glass transition temperature of the resin then applying mechanical load to the material. Due to the shape memory characteristics of the EMC resin, the high packaging strains can be “frozen” by cooling the EMC below T_g , while maintaining the mechanical load and the packaged shape indefinitely.

The main system-performance cost associated with the introduction of EMC materials is the requirement for a heater to cause and control deployment. Small EMC components will likely include simple embedded or bonded resistive-heating elements, whilst large EMC components will require more sophisticated distributed heating systems.

EMC materials and structural components exhibit significantly different responses to loading at temperatures above and below the T_g of the EMC resin. At temperatures below T_g , an EMC laminate exhibits linear stiffnesses that are predictable using classical laminate theory. When deployment is desired, the shape memory response can be activated by elevating the material above the T_g , so that the material exhibits dramatically reduced stiffnesses due to significant softening of the resin and lateral micro-buckling of the fibers and it will regain its original shape with little or no external force. Designing deployable structural components with EMC materials requires an ability to predict the stiffnesses of the material in both temperature regimes as well as the strains induced during packaging, deployment, and operation. A well-controlled heating rate is not as critical for the elastic memory approach as for one involving chemical cure, and therefore a simple solar concentrator may offer adequate energy for deployment. This packaging/deployment cycle is reversible.

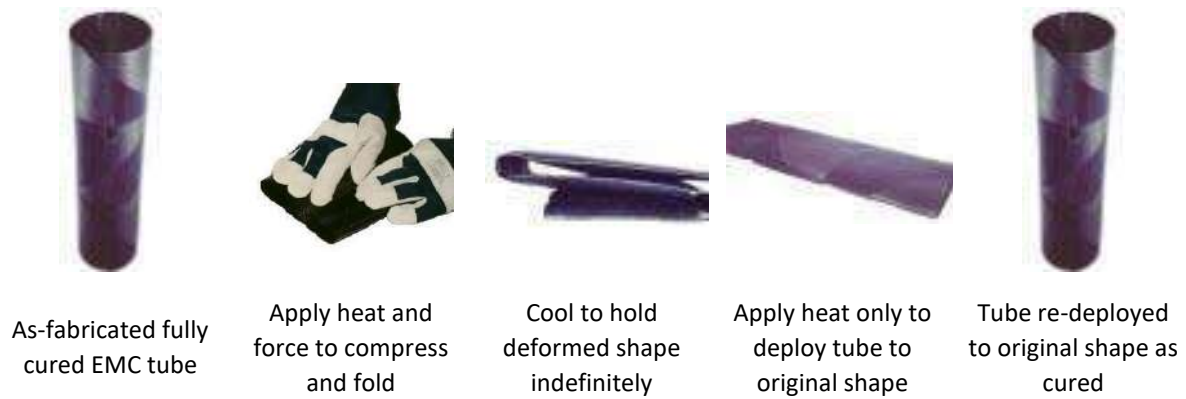


Figure 2. 14. Demonstration of Fully Cured Elastic Memory Composite Material

2.3.8 Micro-Structural model

In fiber reinforced composites the primary compressive deformation mode is associated with fiber micro-buckling. The stiff embedded fibers cannot experience the large compressive strains (necessary for a small R/t) without micro-buckling.

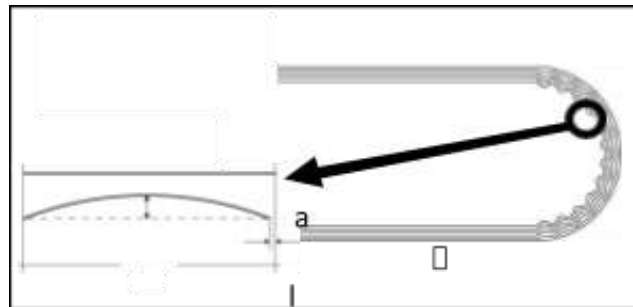


Figure 2.15. Micro-buckling

A heated EMC matrix has a very low shear modulus, and micro-buckling occurs at large bend ratios. The fiber micro-buckling enables a large effective fiber compressive strain because the length of a buckled fiber is significantly shorter than that of an unbuckled fiber. Effective strains well above traditional material strain failure limits are achievable through fiber micro-buckling. However, the geometry dictates that the amplitude of a micro-buckle needs to be large to accommodate a small change in the length, on the inner compressive side of the beam.

To demonstrate that EMC materials can be bent to relatively small R/t values without damage to the material, the fibers must be allowed to micro-buckle on the compressive side, to sufficient amplitudes to accommodate the length differences between the outer and inner sides of the beam. Furthermore, the most stable mode of fiber micro-buckling is out of plane (in the direction perpendicular to the rotation axis of the beam). Out of plane micro-buckling subjects the matrix to very localized strains, which often result in delamination. However, if the fibers can be forced to micro-buckle in plane, the localized strains in the matrix are lower, and higher fiber buckling amplitudes can be achieved, resulting in lower R/t values.

Fibers can be forced to micro-buckle in plane rather than out-of-plane by a combination of tailoring the EMC composite to provide a high strain capability matrix, and the development of special bending techniques and tools.

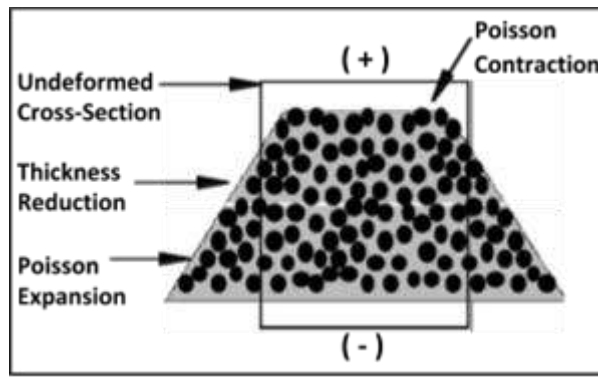


Figure 2.15. In-plane buckling of fibers

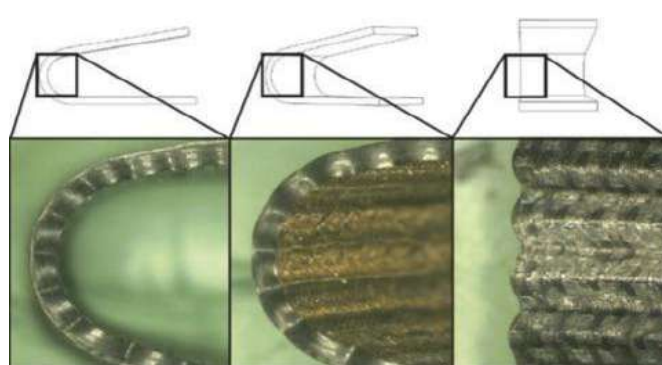


Figure 2.16. Poisson effect in a bent beam with a low modulus

The matrix must allow enough fiber mobility to avoid extreme local stresses/strains and permanent damage caused by local elastic buckling or thickness contraction (by Poisson effect, bending a beam with a low shear modulus matrix causes an **expansion** along its width, which reduces the required strain by lowering t , and allows bending to tighter radii)

2.4 BOOM CONFIGURATIONS

Booms are the main components to support the tip payloads in satellites. Traditionally, the design of a space deployable boom has involved complex assembly and control mechanism and high weight because the materials selected are mostly metals. There are a large number of classes of deployable booms, including telescoping booms like those used on the James Webb Space Telescope and cylindrical booms that can be inflated with internal pressure and become rigid once in space, for instance via a reaction with UV light. These booms are relatively complex, heavy, and often require motors for deployment, which increases the number of potential failure points in the spacecraft design and operations if these motors fail. One solution is strain-energy deployed booms such as thin-walled composite tubes with tape-spring hinges. In this case a dog-bone shape is cut out from the cylindrical tube to form tape spring hinges.

Inside a launch vehicle, the satellite or payload must be compact and light, but when it reaches space it should be as large as needed for the mission. So, during launch these booms are packaged thanks to a hold down and release mechanism added to the system, while elastic strain energy is introduced through bending and stretching. Once in orbit, the strain energy released by the tape spring drives the

deployment of the boom (either with or without a control system) that reach a practically unstressed configuration. Therefore, with fewer ancillary mechanisms required, the booms require less space than more complex solutions.

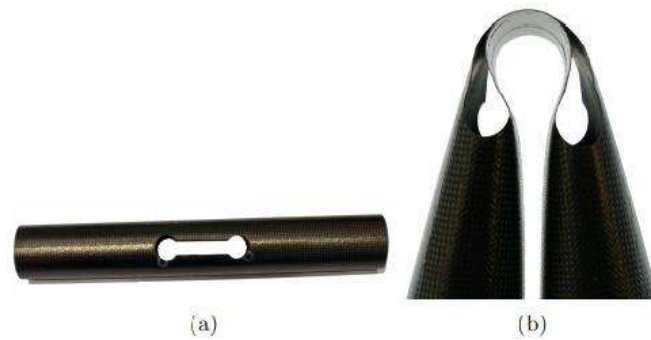


Figure 2. 17: Cylindrical carbon fiber boom incorporating dog-bone cut-out hinge in (a) deployed configuration and (b) folded configuration

Two main categories of deployable booms can be considered: **foldable booms** and **coilable booms**. In particular, coilable booms are distinct from other stored energy booms because they can be flattened and tightly wrapped around a centralized hub. This allows a very efficient packaging scheme ideal for space-based applications. In the sections below, the various type of coilable booms are going to be discussed.

2.4.1 Tape springs

Tape springs, also known as carpenter's tapes, are the simplest of the coilable booms and because of this are often used in cubesat missions. This kind of boom is a thin-walled, straight strip of material with curved cross-section. Metallic tape springs, typically made of beryllium-copper or high-strength steel, have been used for many years as components of deployable spacecraft structures, but the current trend towards simpler, cheaper and more reliable deployable structures is pushing towards tape springs made of carbon fibre reinforced plastic (CFRP), which will potentially offer tailorable performance, low mass, and low coefficient of thermal expansion, as well as integral construction with other parts of a deployable structure. However, Tape springs do have low torsional stiffness and, in most cases, do not have equal stiffness in both bending directions.

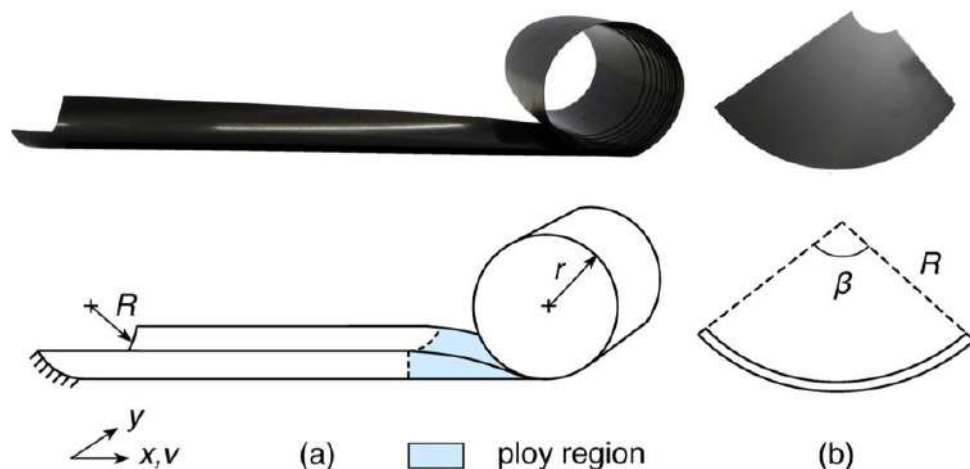


Figure 2. 17: Example of tape spring structure

As shown in Figure 2.17, the geometry of the coiled structure is characterized by the coiling radius r , while the extended configuration has transverse radius R , subtends an angle β , and has a total length L . The transition between coiled and uncoiled regions of the tape spring is called ploy region, which has no significant effect on the deployment behaviour.

2.4.1.1 Basic mechanics of tape springs

A tape spring is bent in the opposite sense when it is loaded by a uniform bending moment $M > 0$ that applies a tensile stress along the longitudinal edges and, conversely, it is bent in the equal sense when the uniform bending moment $M < 0$ applies a compressive edge stress (Fig. 2.18a). For sufficiently small amplitudes of the rotation θ the tape spring is uniformly curved and its moment-rotation response is linear. For larger end rotations the tape spring snaps, suddenly in the case $M > 0$ or through a gradual process involving bending and twisting in the case $M < 0$, and the moment greatly decreases. For further increases of θ , the bending moment remains constant while the arc-length of the elastic fold increases (Fig. 2.18b).

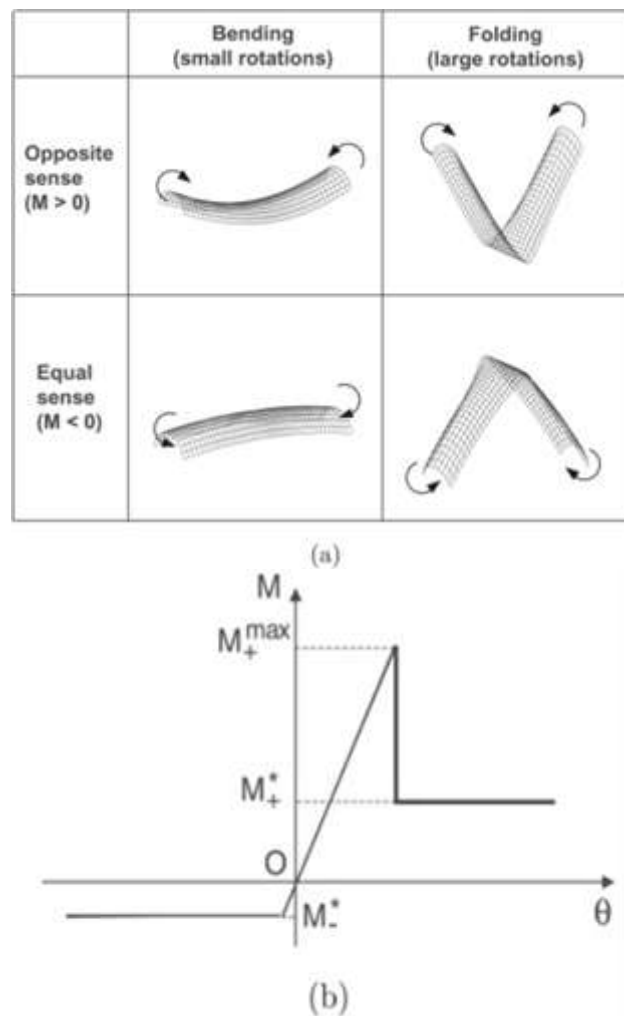


Figure 2.18. Two ways of folging a tape spring: equal sense bending and opposite sense bending

If, however, the tape spring is subject to equal-sense bending, it deforms by gradually twisting over two adjacent, but separate regions whose lengths grow until the two folds merge into a single, localised fold. Once this single fold has formed, further increasing the end rotation results (again) only in an increase of the arc-length of the fold region.

Important characteristics of this behaviour are that:

- the elastic folds have, for both senses of bending, a longitudinal radius of curvature that is independent of the end rotations imposed on the ends of the tape spring;
- the peak stresses and strains in a folded tape spring are largely independent of the end rotations imposed on the ends;
- upon unfolding, a tape spring snaps back into the straight configuration and this is most noticeable for a tape spring that has been bent in the opposite sense. This characteristic makes tape springs ideal as self-latching components in deployable structures.

2.4.1.2 Bi-stable tape-springs

Bi-stable tape-springs are carbon fiber reinforced plastic tape-springs stable in both the extended and rolled-up configurations. Lay-up angles are defined in order to adjust the longitudinal coefficient of thermal expansion very close to zero. Besides finite elements model developed with ABAQUS, models based on the calculation of the strain energy, on beams or shells theory were used to compare the radius of the final states with the experimental results. The main advantage is the safety provided by the stability of the coiled state, since the tape-spring cannot deploy by itself. On the other hand, an external energy income is required to jump over the energy peak between both equilibrium states. This energy can be classically brought by motors or by more innovative actuation based on shape memory polymer: the latter provides a low actuation force which limits the design of the tape-spring, especially the stiffness.

2.4.2 Storable Extendible Tubular Member (STEM) booms

A slight modification of a tape spring is the Storable Tubular Extendible Member (STEM), which was invented in Canada in the 1960s by Rimrott and commercialized a few years later by Astro Aerospace, an affiliate of Northrop Grumman. It has flown successfully on various spacecraft, such as the Voyager probe, the Hubble Space Telescope, and the Mars Pathfinder rover, and is still in production to this day.

Like a tape spring, it consists of a thin-walled cylindrical shell that can be elastically collapsed and reeled around a cylindrical hub or drum, but it can return to its circular shape on deployment via motor command. The main difference is that the arc-length of the cylindrical cross-section exceeds 2π radians. The increased arclength results in higher axial and bending stiffness but the torsional stiffness is still low due to the open cross section, which can be very different depending on the stiffness required. STEM booms are capable of pushing or pulling and are useful also as a deployment structure for other booms.

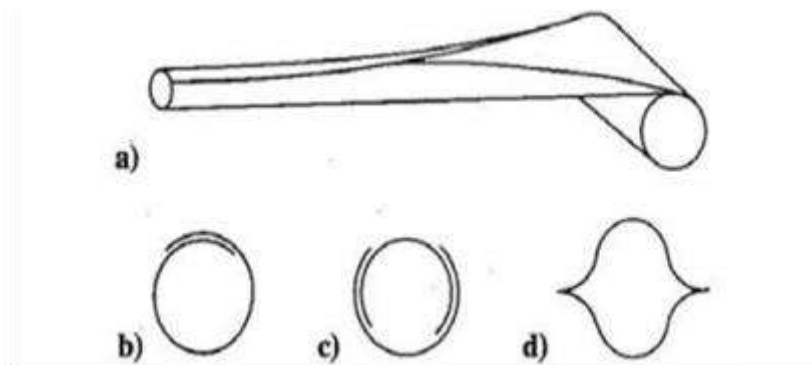


Figure 2.19. STEM types

When the tape is coiled, it can fit into a small place, where the material stores strain energy thanks to bending and stretching. During the deployment phase, instead, the tape spring goes back to a practically unstressed configuration and can extend many times its stowed length, creating a high stiffness tubular shape, typically overlapping to increase the shear and torsional stiffness. The strain energy released by the tape spring drives the deployment, but if this is not well controlled, STEM booms will deploy in a highly dynamic, uncontrolled and difficult to predict manner.

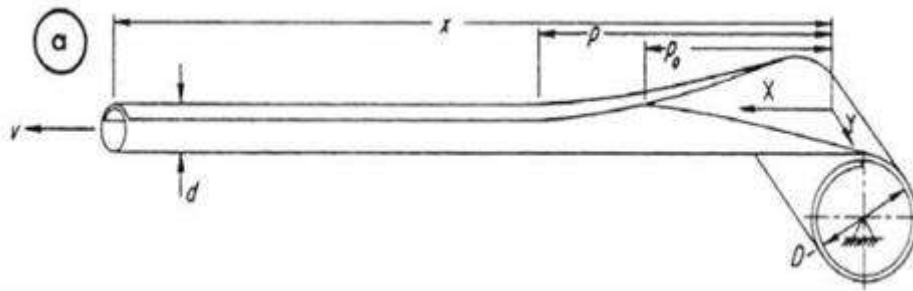


Figure 2.20. STEM deployment

The velocity of the deploying end of the boom accelerates and suddenly stops when the deployment is completed. This impact increases the risk for boom failure or even spacecraft damage. Once the tape spring is entirely deployed, a locking system maintains the deployed state and the whole structure becomes rigid.

2.4.3 Bi-STEM

The Bi-STEM booms are characterized by two STEM booms overlapped during deployment. This kind of boom has very high accurate positioning characteristics and it can also be used as a deployable and retractable telescoping mast. On the other hand, a telescoping mast can employ a Bi-STEM boom as an actuator and stabilizer, which alleviates the need for the deployed telescoping mast segments to overlap. Due to this feature and also because the segments can be fully overlapped when stowed, the mast enables an unusually lightweight and compact launch configuration. An interlocking STEM boom has a higher torsional stiffness than a standard bi-STEM boom.

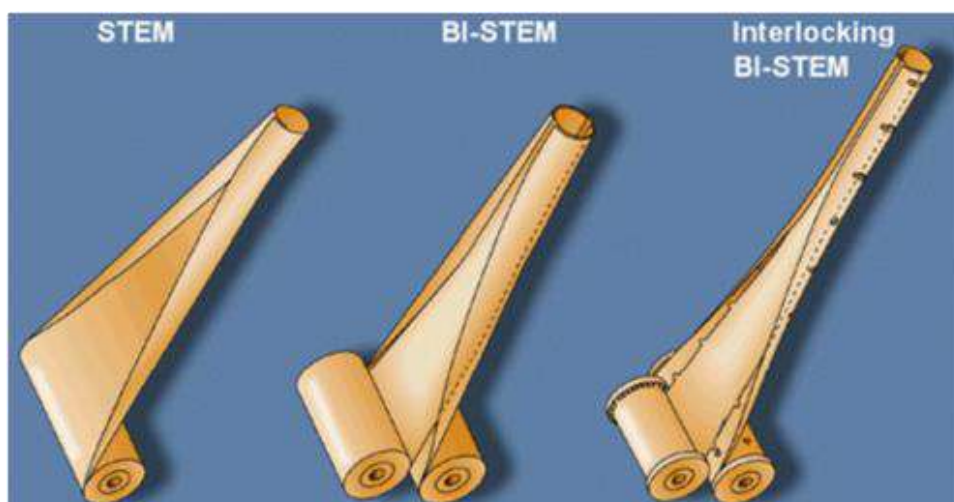


Figure 2.21. Variations of the STEM Boom Concept: Pure STEM, bi-STEM and Interlocking BI-STEM

2.4.4 Collapsible Tube Mast (CTM) Booms

The Collapsible Tube Mast is a boom originally made of a Beryllium Copper alloy that can be rolled and deployed from a canister. It consists of two thin flexible shells with a biconvex shape bonded at their edges by resistance seam welding. The boom can be flattened and coiled up on a cylindrical core that enables very long masts to be stowed within a very small volume because it is almost independent from the mast's length. They are made of thin metal sheet, aside from some early samples in the late 1970s fabricated in carbon fiber reinforced plastic (CFRP).

Given to its superior structural performance capabilities, the closed-section tubular masts with joined seams has been the rollable boom more investigated since its inception in the USA in the 1960s. It's formed by bonding two Ω shaped thin-shells together along the shared edge regions.

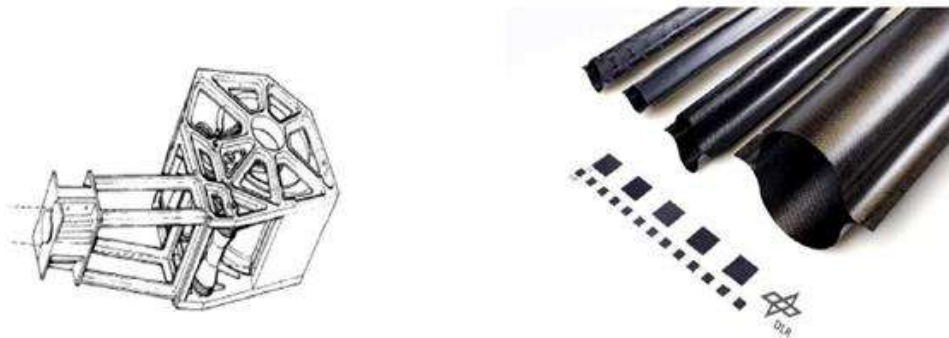


Figure 2.22. Deployment module of the CTM-Mast (left) and DLR composite booms of different scale (right)

The first notable use in space for CTM boom was to collect soil samples on Mars during the Viking 1 and 2 missions in 1976. From the late 1990's, DLR start using more modern composite materials and analysis software packages and begin to develop both scaled up and scaled down versions. A current NASA aim is to develop the smallest CTM composite booms of just 45 mm in flattened height, in order to comply the requirements of CubeSat-based deep space solar sail missions.

Boom designers face conflicting requirements of the stowed and deployed configurations. For the former, it is desired to have a laminate design with a high strain to yield/failure, for compact stowage and towards creep reduction, as well as a low axial Young's modulus, to have a less energetic system while stowed and during deployment. For the latter, it is desired to have a high axial modulus for boom stiffness (EI) maximization, as the failure modes are determined by stiffness rather than material strength, so normally it is realized by maximizing the percentage of fibres running parallel to the axis of the boom. From a cross-section geometry standpoint, in general, it is desired to have a small radius of curvature in the cross-section to maximize the moment of area but at the expense of an increase in flattening stresses and strains that tend to pose a limit on the design space.

2.4.5 Bi-CTM

The Bi-CTM design includes a secondary stable low-energy state aside from the rigid deployed state, so that the boom is not under high-spring stress when coiled up. This feature simplifies the stowage process as well as enabling a more controllable extension of the boom, reducing size, mass and complexity of the storage and deployment mechanism system.

Compared to the majority of deployable thin-shell booms, which have at best a semi-open cross section, this true closed-cross-section boom is stronger, while keeping the compact nature of rollable booms, and is able to overcome both bend and twist buckling related limitations.

These booms offer a great deal of boom customization in terms of stable coiled diameter and structural properties, since they use omega-shaped cross sections with optionally circular, parabolic or ellipsoidal segments, where each half of these thin-shell composite booms can use equal (symmetric boom) or different (asymmetric boom) cross section geometry and/or composite laminates.

Bi-CTM boom design optimization provides for maximized area moments of inertia and torsional constant, which related to the boom stiffness and the loading capacity, while remaining a bistable design.

2.4.6 Triangular Rollable and Collapsible (TRAC) Booms

A concept for coilable booms that is of particular interest is the Triangular Rollable And Collapsible (TRAC) boom by Murphey and Banik and developed by the Air Force Research Laboratory. The TRAC boom cross-section consists of two circular arcs (tape springs) attached along one edge, forming two curved flanges and a flat web (Fig 3.8). The main properties are:

- High ratio of deployable bending stiffness-to-packaged height
- More packaged height compared to other structures
- Low mass
- High packaging efficiency (interesting for CubeSat community)
- Ability of self-deploy
- Design parameter: r = curvature flange radius, t = thickness, θ = opening angle, w = bended section width

Booms of this type have been flown on three different solar sails demonstrations, NASA's NanoSail-D, the Planetary Society's LightSail-1 and LightSail-2. In all three cases, the booms were made from a metal alloy. Recent research has shown that metallic TRAC booms are sensitive to thermal gradients, causing large tip deflections when one flange is facing the sun in space, while the other flange remains in shadow. This has led to TRAC booms made of composite materials being studied extensively in recent years.

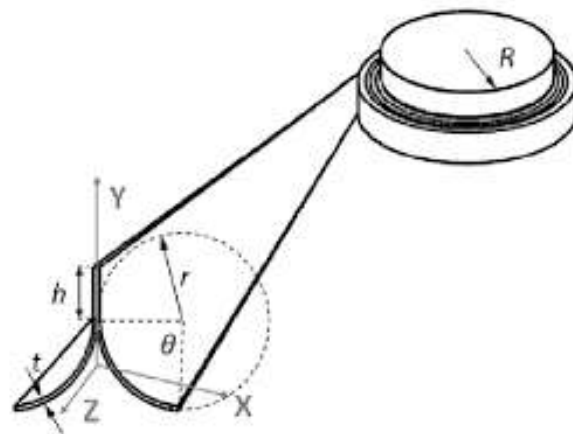


Figure 2.23. TRAC boom architecture and cross-section.

As for the manufacturing process, ultra-thin composite TRAC booms were manufactured from a 17 GSM uni-directional tape supplied by North Thin Ply Technology, a material that has a cured thickness of about $18 \mu\text{m}$ per ply. Alternatively, two aluminium components can be manufactured thanks to pressure by a v-shaped groove, generating a stronger between-flanges bond.

Regarding the coiling behaviour, considering the case $R \gg r$, a tension force is required to conform to the wrapped radius R and this could cause problems during deployment. The localized bend radius can be estimated from minimizing the bond region between two flanges, while the energy in transition can be neglected because it is almost identical for any bond radius, unless other localized bends are close enough to interact with one another. In the bond region, $\Delta\kappa_t = 0$, $\Delta\kappa_l = 1/R$, $s = w$ which is the height of the bond region itself. The initial equilibrium state has two localized hinges, but when a displacement is applied to one side, localized bends increased the radius. As the bend radii get bigger, the inner flange opens while the outer one goes on to conform to the localized bend radius.

2.4.6.1 Deployment behaviour

Due to its thin-walled open cross-section, the TRAC boom shows a complex, nonlinear behaviour both in the deployed configuration and during coiling. In fact, during flattening and coiling of these booms (but it has been observed also under pure bending) local buckling occurs, which can lead to material failure. [Banik and Murphey](#) showed that nonlinear finite element analysis can accurately predict the bending behaviour of booms that are relatively thick ($t \approx 1$ mm), while [Bessa and Pellegrino](#) studied numerically the behaviour of ultra-thin ($t < 100$ μ m) TRAC booms under pure bending and presented an optimization of the cross-section that reduces the effect of shape imperfections on the moment for which the boom collapses. Both studies considered rather short booms, with lengths of 0.6 m and 0.5 m, respectively.

The Finite Element Analysis used a mesh with four-node quadrilateral thin shell elements. Four different behaviour were considered: bending around Y, bending around X towards the opening, bending around X away from the opening and torsion. The torsional behaviour was obtained using only implicit non-linear static analysis, while the buckling and post-buckling behaviour was obtained under a bending moment. Four different cases were considered:

- Preliminary prediction of buckling moment by a linear perturbation (linear eigenvalue)
- Implicit static analysis to the predetermined buckling moment or until the analysis stopped due to buckling
- Linear perturbation buckling analysis
- Post-buckling analysis using a modified RIKS method, introducing an imperfection in the structures' geometry

	SIMULATION	EXPERIMENT
Bending stiffness in X [Nm^2]	8.91	8.15
Bending stiffness in Y [Nm^2]	7.94	3.32
Bending stiffness in Y [Nm^2] (post-buckling)	3.42	3.32
Torsional stiffness [Nm^2]	0.00071	0.0013
Bending Stiffness: $EI = (dM/d\theta) L$, Torsional Stiffness: $GJ = (dM/d\theta) L$		

Table 2.4. Comparison between simulation and empirical results of TRAC

A recent system-level study of deployable space solar power satellites envisages simply supported structural elements with the TRAC cross-section and up to 60 m long, requiring a relatively small bending stiffness of around 5 N m² (elements are mainly loaded in bending). This study also shows that the packaging efficiency of these satellites increases significantly by reducing the flange thickness of the elements with TRAC cross section.

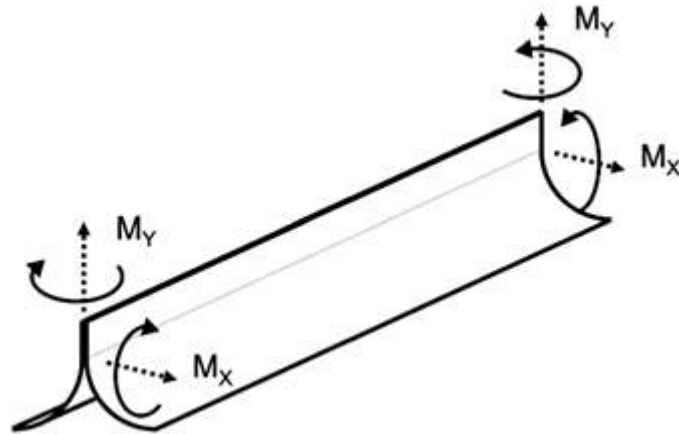


Figure 2.24. Sign convention for positive bending moments

2.4.7 Other Traditional Configurations

2.4.7.1 *Shearless*

NASA's Langley Research Center has developed the Shearless composite boom with a final cross-section shape that is lenticular and is flexible enough to allow elastic flattening and subsequent coiling around a cylindrical reel drum. The boom is fabricated from joining two independent tape-springs front-to-front with the use of a durable seamless polymer sleeve. This sleeve allows the two parts to slide past each other during the coiling/deployment process to minimize shear and its derived problems.

The torsional stiffness of the structure is also greatly increased and becomes two orders of magnitude larger than that of the individual tape-spring components alone. The innovation enables a lightweight structure that can be stowed on a reel without appreciable shear stresses developing in its constitutive composite parts. This allows for unprecedentedly small coiling diameters for the total thickness of the structure, which can enable highly compact designs such as those required in CubeSat/small satellite applications.



Figure 2.25. Shearless boom stowed, showing efficient coiling diameter (left) and deployed, with gravity off-loaded (right)

The shearless composite boom was presented to solve some of the challenges of joined-shell boom concepts. It shows great potential for finding a middle ground between the higher performance and

more scalable CTM boom, the larger moment of area per unit of stowed height TRAC boom, and the more accessible and lower cost production tape-springs or STEMs.

There's a bi-stable configuration also for this kind of booms: the cylindrical shell that coils in an equal-sense way around the hub needs have a bi-stable laminate, and the other mono-stable shell needs to be soft and compliant enough to allow the bistable shell to still develop a strain energy that dwells around the coiled configuration, preventing the snap-through or bifurcation to the extended state. The natural coiling diameter of these two-shell bi-stable booms can be tailored by the bending stiffness ratio of the inner and outer shells. The less stiff the mono-stable shell is, the closest the natural coiling diameter of the complete boom would be to that of the bi-stable shell coiled alone. In fact, less compliant and stiffer mono-stable shells can prevent the boom from attaining a stable, minimum energy secondary state in the coiled configuration.

2.4.7.2 Truss Booms

Truss booms provide higher strength and stiffness than tubular booms. However, these systems are much larger in size and typically require a canister for packaging that exceeds the volume allowed on small satellites. Nevertheless, these truss booms have proven to be highly reliable and efficient for many space missions over the past few decades.

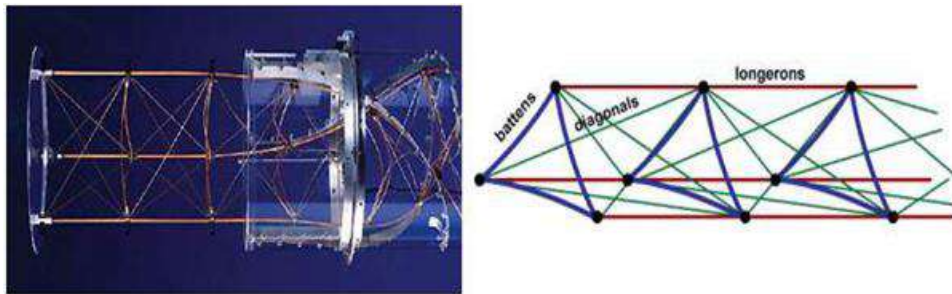


Figure 2.26. CoilABLE Truss booms



Figure 2.27. Deployment of ADAM-Mast

ATK's CoilABLE masts have continuous carbon fiber longerons that can be elastically coiled for stowage. For deployment there are two possibilities: a motorized stowage canister which enables deployment without tip rotation; or the stowed package is located at the tip and will rotate during deployment. The canister guarantees that at least one fully deployed mast segment is attached to the spacecraft so that force transmission is given even in the beginning of deployment phase (for the tip rotation deployment a canister is needed as well but with a shorter length). The stowage length for

the CoilABLE mast is between 0.5% and 2% of the deployed length but the canister size must be taken into account. The CoilABLE masts were developed for high ratios of bending stiffness to weight and are flight proven.

ATK's Folding Articulated Square Truss (FAST) and the Able Deployable Articulated Mast (ADAM) have much higher strength and stiffness than the CoilABLE boom. However, the higher stiffness results from larger diameters and stiffer components that result in a weight above 1 kg/m.

The ADAM-Mast is rectangular truss developed for applications where a high bending strength and a high geometrical accuracy is required. It is folded in a similar way to the CoilABLE-Boom but, instead of deformation of longerons, each joint is equipped with hinges to connect the longerons.

Eight FAST-Masts are currently used on the International Space Stations (ISS) to support the solar arrays. The masts are 32 m long, have a diameter of 1.09 m and are deployed out of 2.32 m long containers. Every second batten ring is flexible, so they are able to buckle and two cells of the boom can be collapsed. The deployment is driven by the strain energy of the buckled battens and is controlled by a retaining mechanism in the container.

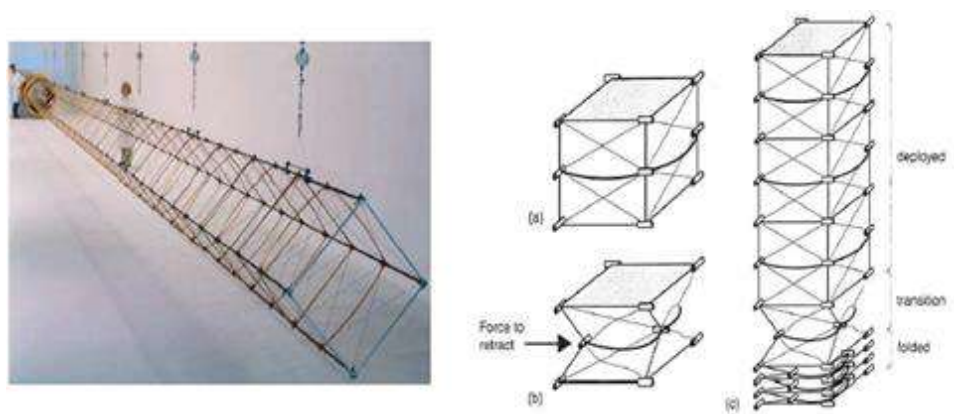


Figure 2.28. FAST-Mast deployment as used on ISS

2.4.7.3 Telescopic Mast

This boom locks in place after deployment and can be provided with a retractability feature. This mast can be sized for various stiffnesses and strengths, based on specific needs. The materials suitable for the mast include aluminum, steel, fiberglass, graphite/epoxy and carbon-carbon.



Figure 2.29. Telescopic mast

2.5 SHAPE MEMORY POLYMERS AND THEIR COMPOSITES

To overcome the problem of complex assembly and control mechanisms, as well as the high weight caused by the material used, the focus has shifted towards a new class of booms: SMPC booms. This new kind of booms are based on the use of EMC materials, which give them simple structure, but also the ability to package and deploy without mechanical devices. Therefore, the unique characteristics of SMPCs have aroused increasing interest for space-deployable structures

As discussed in the EMC paragraph in chapter 2, these materials offer, thanks to their thermoset shape memory resin, a high failure strains and low stored strain energy. In fact, when deployment is desired, the shape memory response can be activated by elevating the temperature of the EMC material above the glass transition temperature of the resin and then applying mechanical load. Due to the shape memory characteristics, the high packaging strains can be “frozen” by cooling the EMC below T_g , while maintaining the mechanical load and the packaged shape indefinitely. Therefore, this novel types of SMPC booms requires the ability to predict the stiffnesses of the material in both temperature regimes as well as the strains induced during packaging, deployment, and operation.

2.5.1 Foldable SMPC truss booms

When the work started in 2002 for the United States Air Force Academy (USAFA) FalconSat-3 mission, the Air Force Research Laboratory (APRL) selected two types of boom configurations: a two-longeron truss boom (Fig. 2.30b), which form a tubular-shaped boom after deployment, and a three-longeron truss boom (Fig. 2.30a), which can extend to a space triangle-shaped truss boom. Starsys Research Corporation (SRC) verified the feasibility of EMC truss booms and indicated that each model has some drawbacks during the deployment process. The moving of the two-longeron truss boom and the fastening of the three-longeron truss boom were potential failure factors to successful deployment: finally, the low-moving-part design and light weight of the two-longeron truss boom led to its selection as the preliminary baseline model. However, after a series of designs, fabrications, measurements and evaluations, the final truss boom included three semi-cylindrical EMC longerons packaged into a ‘z-shaped’.

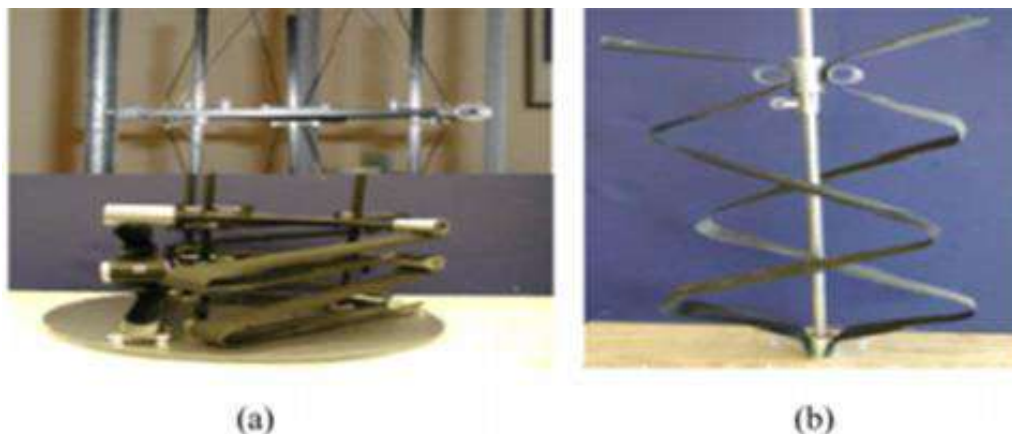


Figure 2. 30. Preliminary SMPC booms design of FalconSat-3 mission: three-longeron truss booms (a) and two-longeron truss booms (b)

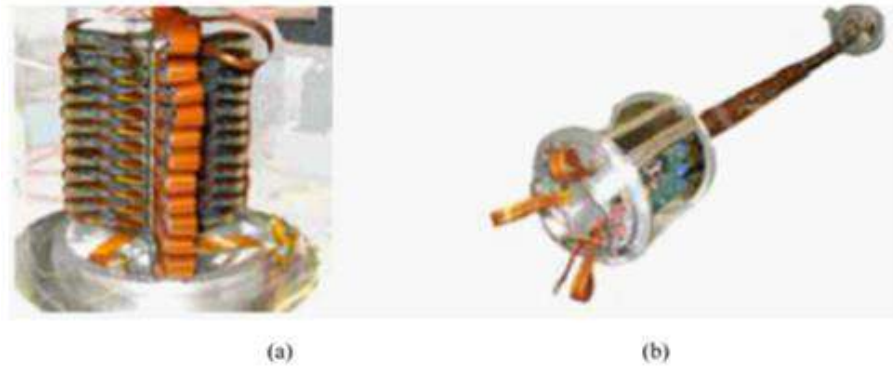


Figure 2.31. The proposed FalconSat-3 three-longeron EMC tubular boom: packaged configuration (a) and deployed configuration (b)

Recently, a novel type of SMPC truss boom has been developed: it consists of 18 pieces of laminate tapes with a semi-cylindrical section in the equal sense, one extendable central bracket, which includes six levels of short hollow rods that assume good contact area without friction (each rod contains three laminate tapes with an angle of 120° to each other) and a resistor heater stuck to the concave surface center-line along the axial direction of every laminate tape. The boom is packaged into a 'M-shape' before the experimental evaluation: when performing the experiment, the boom will gradually deploy with increasing time. The structure is very different from the CTD production, since the sections are in the equal sense, which is related to the deployable recovery force, and the structure is divided into some small parts.

2.5.2 Coilable SMPC truss booms

The coilable truss boom is another advanced deployable boom, which consists of simple structure and low weight, but also high strain energy and large relaxation phenomena in the stowed state. Nowadays, there are two types of configurations: three-longeron (Fig. 2.32a) and six-longeron (Fig. 2.32b) coilable truss booms. The longeron of the boom can be bent or stowed in a small volume and provide stiffness and strength after deployment, which can mainly consist of free deployment, cable deployment and tubular deployment. S2-glass/epoxy is the main material to fabricate the longeron due to the deformation and mechanical properties. However, the boom will store lots of strain energy and be a safety challenge when packaged by twisting the boom into a helical or snake shape for launch. To overcome the drawback, the fiber-reinforced composite concept was considered, resulting that the EMC longerons can significantly reduce the weight of the boom and the packaged strain energy while simultaneously maintaining the high strain capability.

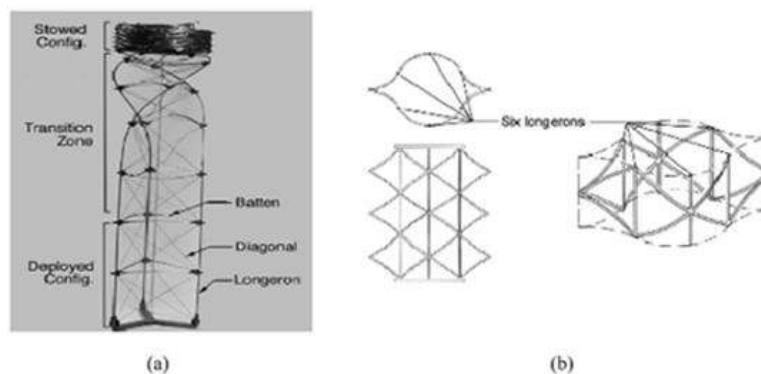


Figure 2.32. Coilable truss booms: three-longeron boom (a) and six-longeron boom (b)

2.5.3 SMPC STEM booms

The SMPC STEM boom is the other kind of advanced boom extant since the early stages of space exploration, characterized by lower weight, higher packaging strain and stronger specific modulus than the traditional STEM boom. Additionally, the SMPC STEM boom has a simpler structure and larger cross-section compared with conventional SMPC booms.

To date, an interesting prototype boom is a seven meter horizontal SMPC STEM, usually composed of some small member sections (lap joints are a key part to assemble these sections). Moreover, a deployment control system is imperative for deployment of the structure, which essentially consists of support structure, heater, driving motor and computer controller. Due to SMPC hinges/booms needing to undergo flexural deformation and have minimal strain, the bending mechanical properties and deployment dynamics must be considered in aerospace application.

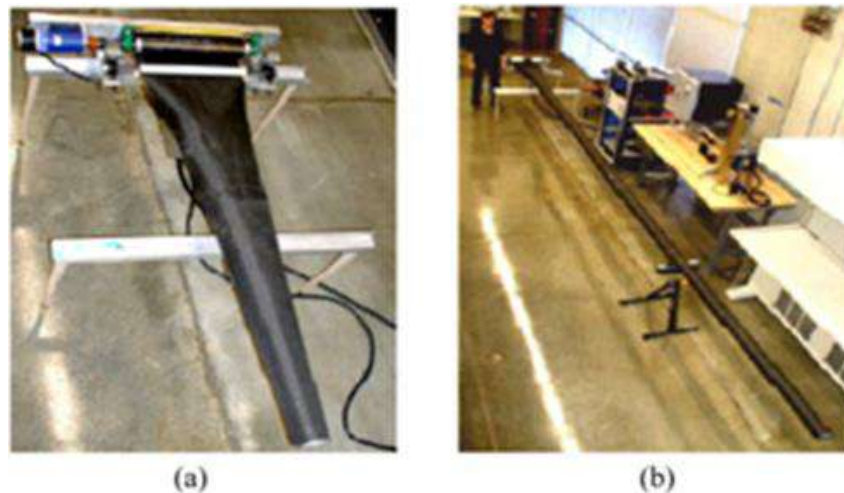


Figure 2.33. EMC STEM deployed system: partial deploying state (a) and deployed state (b)

As for the microbuckling response of reinforced EMC laminates, CTD Co. has reported that the fiber microbuckling can undergo higher flexural deformation without damage at high temperature and store the deformation state at low temperature and, when the EMC material is reheated, the microbuckling gradually disappears and the original shape is gradually recovered.

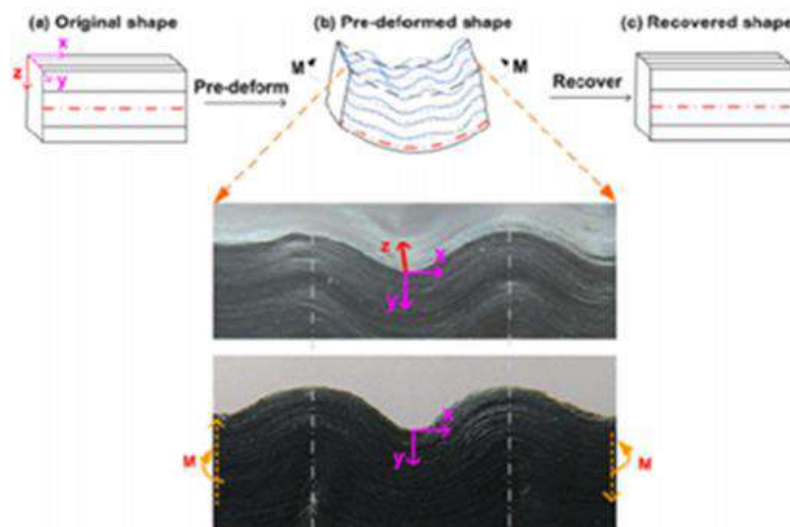


Figure 2.34. Microbuckling deformation of SMPC laminate

2.6 IN-DEPTH INFORMATION ABOUT OTHER MATERIALS, TAPE-SPRINGS AND CONFIGURATIONS

2.6.1 High-Strain Composite (HSC) Hinges

HSC materials are flexible, thin composite laminates, that undergo strain larger than metals and show very little plastic deformation, whose main function is serving as a deployment drive actuator. The hinges are made of two bonded opposite tape sections and cylindrical composite mandrels are used: the terminal fixed part is made of Invar, for reducing the thermal change, whilst the terminal free part is made of aluminum, to minimize the effects of gravity loadings at the tip. To enforce the bond line thickness, a composite based on glass fibers and Hysol EA 9394 epoxy is used.

The phenomenon of stress relaxation is present: in the folded position, the hinge is under stress and the fibers inside the matrix are in relaxation, so the creep recovery depends on materials' properties, applied strain and the duration of this one. Different tests have been conducted:

- **Environmental Tests:** a stabilized thermal chamber has been utilized to reduce temperature and humidity variation during the process. The first one was about 0.1 °C, while the second one was about $\pm 1\%$. The thermal expansion of the reference plate was less than 50 nm. Some sensors need to be mounted to cold plates, to achieve the temperature stability, so they were cooled using Polyscience FF-12930-38, a benchtop chiller.
- **Instrumentation Tests:** the sensors chosen were Keyence LK-H027 1D laser triangulation sensors, a temperature measurement system composed by two thermistors. Moreover, five thermocouples was required and two accelerometers monitor the vibration environment, even if no particular vibration levels were observed.

The composite materials expand and contract as a function of humidity absorbed by fibers and matrix: a change in humidity determines a change in the deployment precision. Using this Invar specimen, the changes in position, with that environmental conditions, were about 50 nm or less.

<i>Piston deployment precision</i>	<i>2.3 μm</i>
<i>Pitch and Roll precision</i>	<i>< 6.3 μm</i>
<i>Axial precision</i>	<i>0.24 μm</i>

Table 2.5. Properties of structure with multiple HSC Hinges

2.6.2 Viscoelastic Effects in Tape-Springs

As already said, one of the most challenging aspects of spacecraft applications consists of the packaging and orbit deployment. Focusing phenomenological features of linear viscoelastic behavior of polymers and choosing LDPE, whose glass transition is below 0°, viscoelastic effects can be studied in structures near room temperature.

Linear viscoelastic properties of LDPE were characterized through a series of tensile relaxation tests on rectangular test coupons that had been subjected to the same thermal cycle as during the manufacture of the tape-springs. In the following experiment specimens have defined dimensions, thickness, density and material constants C1 e C2.

A thermocouple was attached to the surface of a dummy LDPE specimen close to the test specimen to monitor the actual specimen temperature. As a test for stable temperature conditioning a temperature impulse was imposed and the subsequent temperature variation over time was

recorded. In this way it was found that the temperature readings from the two thermocouples became identical 30 minutes after the impulse: this indicates that thermal equilibrium can be established within such time frame, and this thermal conditioning time was allowed prior to each test. So, prior to testing, the position of the load frame was adjusted so that a zero axial preload was obtained.

The initial portion of the relaxation test data after loading with a finite strain rate deviates from that in the case of ideal instantaneous straining. For this reason, the data obtained during the first seconds after loading were discarded.

2.6.3 Material Model

The theory of linear viscoelasticity is well established to describe the time and temperature dependence of the mechanical properties of polymers. The relaxation modulus under isothermal conditions is written as:

$$E(t) = E_{\infty} + \sum_{i=1}^{\infty} E_i e^{-(t/\rho_i)}$$

where t is time, E_{∞} is the long term modulus, E_i are the Prony coefficients, and ρ_i are the relaxation times. The constitutive relation for uniaxial deformation is expressed in the form of the Boltzmann superposition integral in which σ is stress and ϵ is strain:

$$\sigma(t) = \int_0^t E(t - \tau) \frac{d\epsilon}{d\tau} d\tau$$

For thermorheologically simple materials, the effects of time and temperature on the material behavior can be treated in the same manner through the time-temperature superposition principle,

$$E(t, T) = E(t', T_0) \quad t' = \frac{t}{\alpha_T(T)}$$

t' is the reduced time, T_0 is the reference temperature, $\alpha_T(T)$ and is the temperature shift factor. The time-temperature superposition principle states that the modulus at temperature T and time t is the same as the modulus at a reference temperature T_0 at a reduced time t' . Thus, one can relate the viscoelastic behavior at one temperature to that at another temperature by a shift in the time scale. When temperature varies with time, the reduced time is obtained by integration.

One commonly used relation for the temperature shift factor of polymers is the empirical Williams-Landel-Ferry equation:

$$\log \log - \frac{C_1(T - T_0)}{C_2 + (T - T_0)}$$

in which C_1 and C_2 are material constants that depend on the particular polymer.

2.6.4 Quasi-static Folding and Stowage

Folding and stowage tests were performed by vertically compressing tape-spring. A fold-stow procedure was adopted in order to achieve fully controlled load profiles and boundary conditions. Moreover, to allow the ends of the tape-spring to rotate freely, the connection between the end of the tape-spring and the load frame was established through point contact with thin aluminium plates attached to the testing machine.

During the test, a vertically downward displacement of 80 mm was applied to the tape-spring for 5000s through two displacement rates, 1 mm/s and 5 mm/s. As a result, the load drops significantly after buckling and continues to decrease even though the displacement is still decreasing.

This is different from the post-buckling behavior of an Euler structure for two reasons: first, the moment at the fold of an elastic tape spring is constant due to propagation of a local instability; second, due to viscoelastic effects the moment actually decreases. Load relaxation takes place at the same time as buckle propagation and offsets the expected increase in force. The results also show that a higher load is required to fold the tape-spring at lower temperatures and faster rates. This behaviour is typical of viscoelastic materials whose modulus increases with decreasing temperature and increasing rate.

2.6.5 Dynamic Deployment

Deployment tests were conducted at room temperature on a tape-spring clamped on the bottom and positioned vertically, first folded to a certain angle and then held stowed for determined time span.

The deployment process can be divided into three stages with distinctive features: a dynamic response during the first seconds, which includes a low vibration magnitude about a finite displacement that decreases with time; then it involves quasi-static deployment with the lateral displacement; finally, a slow creep recovery of the fold cross section leads to a nearly zero lateral displacement over a long period.

2.6.6 Comparison of Results

Simulations of the quasi-static folding and stowage test were carried out with a model including thousands of quadrilateral shell elements while simulations of the dynamic deployment test were carried out with a similar mesh, but with less elements. The mesh density in the circumferential direction was twice the longitudinal direction for more precise computation of the localized fold. The behavior of thin shells is known to be sensitive to geometric imperfections from sources such as manufacturing and load misalignment, so in the analysed model the imperfections considered are the uneven distribution of thickness resulting from the fabrication process and the residual deformation resulting from previous loading cycles, due to residual viscoelastic deformation.

The results from the experiments and the simulations highlights that the simulations show good agreement with the observed response however it slightly overpredicts the peak load for different temperatures. The error between simulated and experimental results become insignificant after the the fold has completely formed at about 7s. As far as displacement of tape-spring during deployment regards, the general features of the response have been reproduced by the simulations, but there are also some noticeable discrepancies from expemiments. It is believed that the main reason for the discrepancies, expecially during quasi-static recovery phase, is thermal variations in the experimental environment

In conclusion, deployment begins with a weak dynamic response accompanied by a low magnitude vibration. This short dynamic phase is followed by a quasi-static deployment that returns the tape-spring to almost the straight deployed configuration and this full recovery requires a final slow creep recovery process. An interesting observation is that the fold location stays constant during deployment, opposed to the moving fold of elastic tape-springs. Moreover, locking of the fold is due to local stiffness reduction at the fold after relaxation and this particular deformed configuration becomes a stable equilibrium (for elastic tape-spring only neutral equilibrium is attained).

2.7 THE USE OF BERYLLIUM: JAMES WEBB SPACE TELESCOPE'S MIRRORS

The main requirement of the future space-based telescopes will be precision. For large primary mirrors, it is necessary a sort of on-orbit deployment or assembly: this is possible only with a high positional accuracy and many factors, responsible for errors, derive from environment loads; the level of positional accuracy depends on telescope, wavelength and mission.

Mirrors are a critical part of any space telescope because they must endure the extremely frigid temperatures in space, be highly reflective, lightweight and tough. Most mirrors are designed for visible light, however, there are mirrors that work at other wavelengths of electromagnetic radiation, such as X-ray, infrared, microwave, or even radio wavelengths.

Actually, the most used material is the beryllium: a metal, steel-gray in color, very strong for its weight and good at holding its shape across a range of temperatures. Beryllium is also a good conductor of electricity and heat but it doesn't exhibit magnetic properties. It also has one of the highest melting points of the light metals and it contracts and deforms less than glass and remains more uniform in the space's temperatures.

Beryllium has been used to develop parts for supersonic airplanes and also for the Space Shuttle. Moreover, it is used in gyroscopes, computer equipment, watch springs and instruments where light weight, rigidity and dimensional stability are needed. However, beryllium is highly toxic to plants, animals and humans.

For all these reasons, the optics of the Spitzer Space Telescope were entirely built of beryllium metal and it will be used as well on the James Webb Space Telescope's mirrors. In this last case, the particular type of beryllium used is called "O-30" and is a fine powder of high purity. The powder is then placed into a stainless steel canister and pressed into a flat shape. The steel canister is then removed and the resulting chunk of beryllium is cut in half to make two mirror blanks about 1.3 meters across. Each mirror blank will be used to make one mirror segment; the full Webb's mirror will be made from 18 hexagonal (six-sided) segments.

3 DEPLOYMENT TECHNOLOGIES

3.1 ONE SHOT DEVICES

The function of one-shot devices is to change the structural configuration of the spacecraft. In all cases, this function is performed by allowing and then forcing the relative motion of some of the parts of the mechanism that were previously constrained by a hold-down device. It's possible to distinguish two separate functions: the first one is to restrain one or more parts of the mechanism for all the time up to the moment when the system is finally 'fired', the second one is to enforce a predetermined movement of particular parts of the mechanism.

3.1.1 Pyrotechnics

In space engineering, space launchers and satellites use many devices for igniting, moving, separating or activating various elements. These devices are generally activated by electro-pyrotechnic components in which at least one phase involves the rapid decomposition of pyrotechnic substances at high pressure and temperature.

This sort of release devices are probably the most used release mechanisms in space missions. They are part of the family of one-shot release devices and include several types of devices such as explosive bolts (used also for the fairings) or pyrocutters.

Pyro devices are very energy efficient (> 10 KJ/g), operate very rapidly (< 10 ms) and they are very reliable. Space pyrotechnics always have two initiators for each charge and fully redundant firing circuits. The pressure generated by combustion is used to operate a simple mechanical device such as a pin-puller or pin-pusher, cable-cutter or valve-actuator.

They can only be operated once and are then discarded, and this feature seriously limits ground and pre-flight tests. Safety is a very important aspect and must be create protections systems to absorb electrostatic discharge.

There are many drawbacks to the pyrotechnic devices. The first problem is handling the pyrotechnic devices safely and avoid an uncontrolled trigger. In fact, launch vehicle manufacturers have high costs to guarantee the safety standards for storage, transportation and handling the devices during ground operation and storage.

The second problem, which is even more important, is the impact on the payload upon release. This kind of devices generate high shock levels which can seriously affect the payload and make the mission fail. But even if this does not happen, the pyrotechnic devices can generate a number of debris that can, in the worst case, jeopardize the mission. In fact, on some spacecrafts, like those with delicate optical elements, pyrotechnic release devices are forbidden.

These drawbacks become even more critical for small satellites, like CubeSats and nano-satellites, because of their size. For these reasons space launcher companies are looking for new low-shock release devices and for non-pyrotechnic release devices, like memory metal actuators or fusible wire-actuated separation nut design.

An example of the difference between Pyrotechnic and Non-Pyrotechnic release devices is shown in Fig. 3.1.

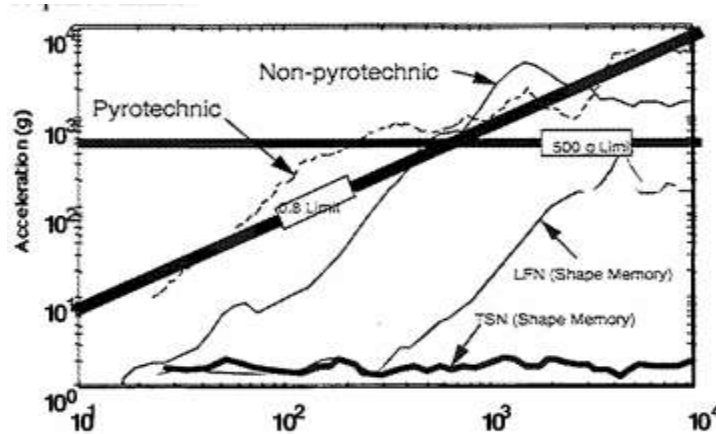


Figure 3.1. Acceleration of pyrotechnic devices

3.1.2 High energy paraffin actuators

With small dimensions and weight, the paraffin actuators are used in application where weight savings have a great importance.

Paraffin Actuators are characterized by high loading, average stroke and low speed. In this category of parameters, they can replace. Electrical motors when complexity and magnetic materials are not acceptable or pyrotechnic equipment when resettable aspects and no shock are required.

Paraffin Actuators use the volume expansion of paraffin that occurs during the solid to liquid phase change in order to produce mechanical work in the form of linear motion of a piston. This type of actuator is based on various technologies and four critical points were identified and solved: the heater and its electrical connections, the paraffin selection, the sealing system, and the retraction system.

A commercial paraffin was selected to answer to the requirements. The volume expansion of the selected paraffin was measured (see Fig. 3.2.). At atmospheric pressure, a high expansion of 25% was obtained. When the pressure increases, the expansion at melting decreased.

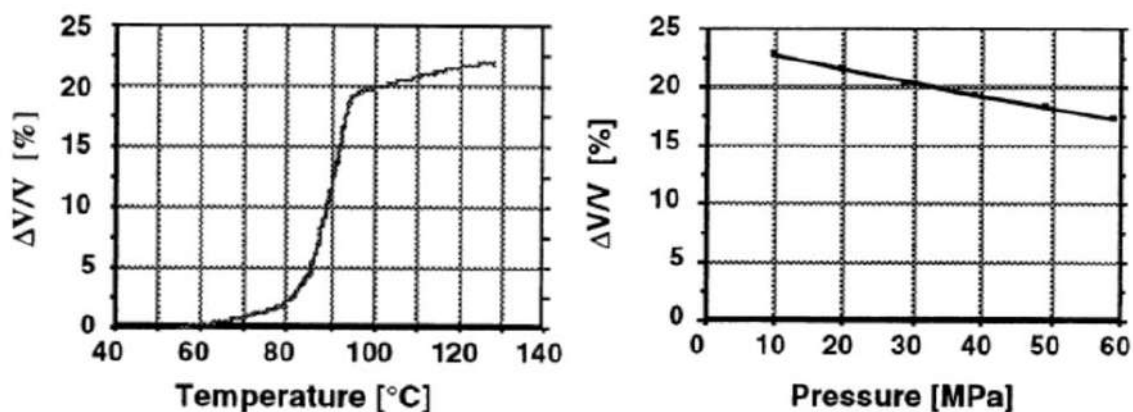


Figure 3.2. Variation of paraffin volume

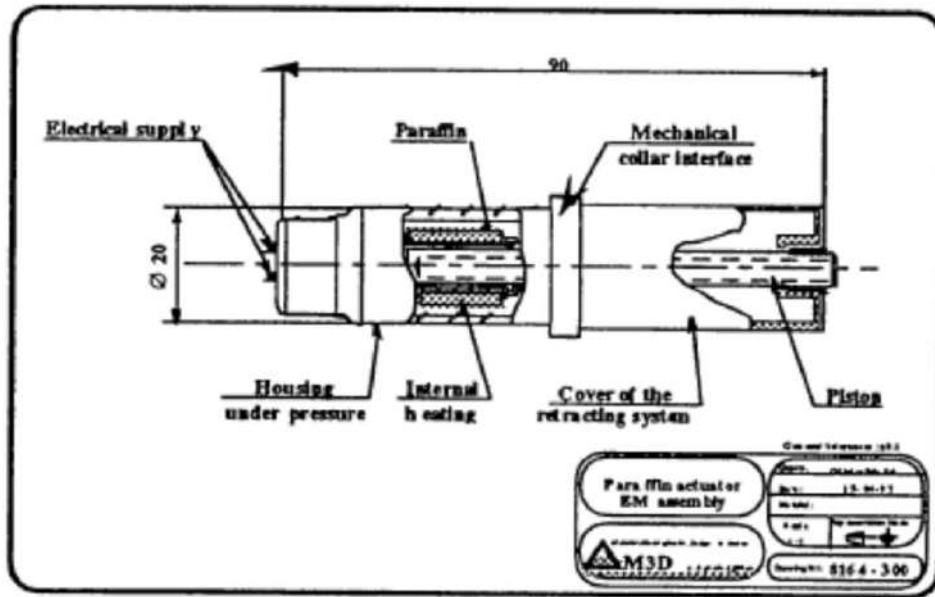


Figure 3.3. Example of a paraffin actuation device

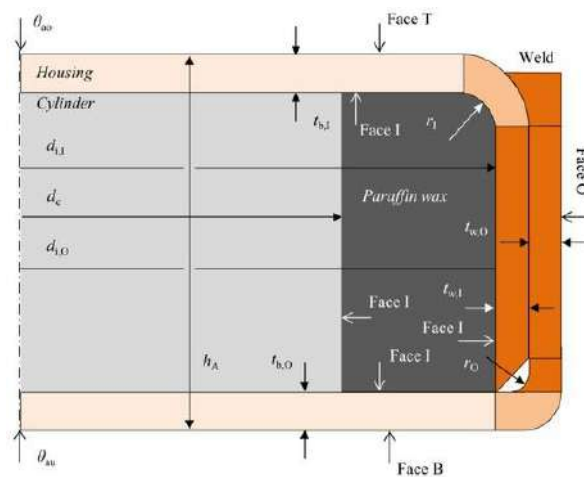


Figure 3.4. Example of a paraffin cylindre

A cylinder ($d_c = 20.0$ mm, 22.5 mm, 25.0 mm) is positioned in the middle of the actuator surrounded by paraffin wax. The actuator is surrounded by an aluminum ring which generates an even temperature distribution over the perimeter and contains three thermocouples to control the target temperature θ_t . There is a heater band around the aluminum ring. The paraffin wax volume can be adjusted by changing the cylinder diameter d_c in the actuator. The paraffin wax exerts pressure on the inner Face I due to its volume increase. The pressure leads to an axial force F_a . At face T and face B compression pads (steel grade 1.2379) are positioned to apply the counter pressure and connecting the actuator to the compression test machine (Zwick Roell 100). Thermocouples are placed in the center of the compression pads to record the temperatures θ_{ao} and θ_{au} . In the simulation, the vertical displacement of the compression pads is set to zero by a boundary condition. In the experiment, an optical system references the upper and lower compression pad to a constant distance. Between compression pad and test machine, two insulation plates are positioned to reduce the heat transfer into the test machine. The film coefficient for the outer surfaces is set between 0.01 – 0.03 W/(mm² K) in the simulation to model natural convection.

3.1.3 Burn wire mechanisms

This type of system has been used, due to their simplicity, to trigger release mechanism. The separate parts of the mechanism are pushed together, overcoming the force of some preloading device that tries to push them apart. These are then tied together by a wire. To release the mechanism, a current is passed through the wire, heating it up and ultimately melting it, thus allowing the parts of the mechanism to separate. This mechanism can be actuated in fractions of a second, depending on the characteristics of the wire.

For example, this mechanism is particularly useful for CubeSats application. In this case, the configuration is a Nichrome Burn Wire Release Mechanism. The nichrome burn wire release mechanism uses a nichrome burn wire which when activated heats up and cuts through a Vectran tie down cable allowing the deployable on the satellite to actuate. The release mechanism was designed from scratch with the goals to make it small, inexpensive, simple, reliable and easy to use by anyone.

The release mechanism, shown in Figure 3.6. utilizes a two-saddle design with compression springs to apply a spring stroke and force to the nichrome wire to thermally cut through the Vectran cable when heated. Through a test program and using a design of experiments (DOE) approach it was determined that the applied current to the nichrome wire and the diameter of the nichrome wire were the key parameters governing successful performance. To activate the Nichrome wire, a constant current of 1.60 ± 0.05 Amps is applied to ensure a successful and reliable cut. The tight tolerance constant current source is necessary in order to reliably: 1.) thermally cut the cable and 2.) prevent overheating failure of the nichrome wire to allow the mechanism to be reusable for many actuations without replacing the nichrome filament. The tight tolerances on the current prevent failure of the nichrome wire from overheating under too much current in a vacuum while also providing adequate thermal margin to cut through the Vectran tie down cable in air which requires more current than in vacuum.

The maximum length of the nichrome wire is limited by the free length which causes the wire to lose structural stability when heated. At a free length greater than 32 mm the apex of the nichrome wire when heated becomes very hot and the resulting loss in tensile strength can cause necking of the wire which would impact further use of the nichrome wire. The entire burn wire release mechanism once assembled has dimensions of approximately 32 mm.



Figure 3.5. Assembled burn wire release mechanism

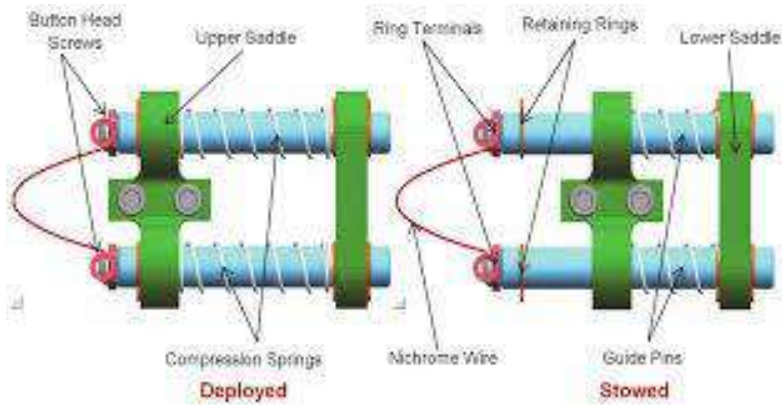


Figure 3.6. Nichrome burn wire release mechanisms in the deployed and stowed configurations

3.1.4 Marmon clamp

It is one of the most reliable and commonly used mechanism. The basic version of this device, whose cross-section is shown in the picture, is a circular belt with shoes having a V-shape horizontal groove, secured to its internal surface.

When the clamp-band-pyrocutter or similar device that cuts a retaining bolt. For flanges with a diameter larger than around 1m, the clamp-band is usually divided into two or more segments of equal length, joined by the pyrotechnically actuated bolts.

Once this bolt is cut, the belt opens (usually helped by torsional springs) pulling out the shoes, and thus freeing the flanges. At this moment, the satellite is pushed away by separation springs secured to the structure on the launch vehicle side.

These devices must be perfectly synchronized to make sure that the belt opens up at all the junctions simultaneously. The main advantage of this type of mechanism, apart from its simplicity, is that the two flanges are held together along the whole perimeter, thus transferring the loads in a relatively uniform manner, especially when the structure is cylindrical.

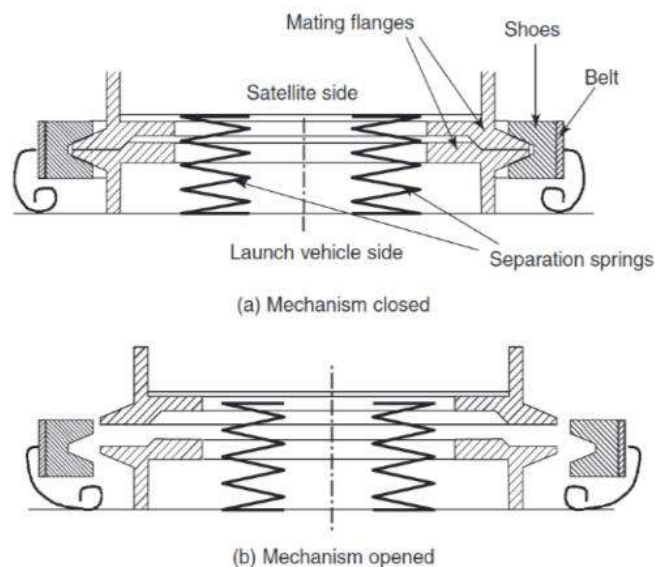


Figure 3.7. Example of a marmon clamp mechanism

3.1.5 CarboNIX

Thanks to its unique pusher arm, the CarboNIX system achieves very efficient separation with a near-zero tumble after separation, and with a nearly negligible shock over the satellite. In fact, CarboNIX achieves an average tip-off rate of less than 1 deg/s, with no axis that exceeds 2 deg/s, regardless of mass distribution across the satellite, instead of traditional separation systems (tip-off rate of about 20 deg/s) which need a satellite attitude control system to keep the orientation of the satellite. The absence of shock separation makes CarboNIX able to significantly reduce risks of damaging satellites' optical payloads and electronic components. CarboNIX is a very light separation system (3.07 kg in total and a flyaway mass of 330 g), with a 15" diameter and suitable for microsatellites weighing up to 200 kg.

3.1.6 Motorized Lightband (MLB)

It is made of aluminum alloys. MLB is characterized by 2 rings. The lower one has hinged leaves (mechanically-actuated), springs and a dual redundant release motor that separates the upper ring from the lower one. The upper one has a load-bearing hinge interface. This system is suitable for payloads from 50-1200 lb.

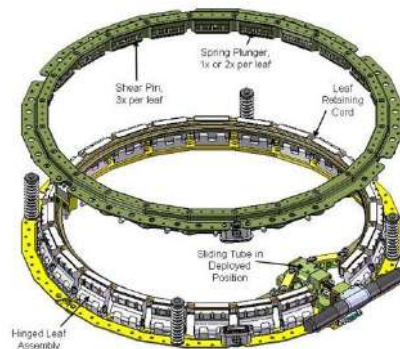


Figure 3.8. MLB elements view

3.1.7 Mechanical diode

Various kinds of mechanisms are used to deploy the appendages of a spacecraft. Usually, a simple method that allows to deploy these structures, is represented by a knuckle joint (formed by a simple hinge coupled with a torsional spring) placed at the root of the appendage. For instance, the knuckle joint is used to deploy antennas: The torsional spring forces the rotation of the hinge, driving the deployment of the antenna and keeping the structure fully deployed once the hinge reaches its end-stop.

Sometimes, it is necessary to use self-locking mechanisms in order to ensure that the structure remains deployed preventing the application of external loads from causing it to close. For this reason, a device called "mechanical diode" is used to guarantee that the deployment of the structure cannot be reversed. A mechanical diode will only allow rotation in one direction to be transmitted through the system. In general, it can be defined as a particularly robust kind of one-way clutch allowing rotation in one direction, while resisting it in the other, by keeping the stator locked to the transmission shaft in one direction and allowing rotation in the opposite direction. The mechanical diode works by using spring-loaded struts to engage in notch plate when turned in one direction and override the notches in the opposite direction. Each strut fits into a cavity in the hub and is pressed into engagement with an arch plate by a spring underneath. The springs fit into the center of the strut cavity, with the outer legs on the bottom, and the loop center of the spring sticking above the inner cavity. Each strut has tabs on one side, with different widths: the narrow tab goes towards the center

of the hub and the wider tab faces outward. The struts also have edges machined at an angle, to properly engage the hub and knock plate. The correct assembly of the struts and springs is crucial: it is important to pay close attention to their orientation.



Figure 3.9. hinge of whip antenna



Figure 3.10. Sonnax Mechanical Diode® CH-MD-1

3.1.8 Deployable lattice mast

Another class of deployment mechanism is the deployable lattice mast. The one shown in the figure is the AEC- ABLE CoilABLE mast, used successfully as an integral part of the Imager for Mars Pathfinder (IMP) system, which sustain a stereoscopic camera that provided panoramic views of Mars. The lattice structure consists of unidirectional fiberglass/epoxy rods, aluminium fittings, stainless steel diagonal cables and stainless-steel fasteners, with a mass of less than 0.7 kg. The structure extends and retracts by twisting about the longitudinal axis. The main part is the deployable 18.3 cm (7.2 in) diameter mast: when stowed, it acts like a compressed spring that, when released, pushes out toward the extended position. Once fully deployed, the structure becomes rigid and it's about 0.7 m long, while in the stowed position it retracts to 10% of its extended length, coiling up inside a 76 mm long canister. This type of structure has been used frequently in space: it has the advantage of exploiting stored strain energy, that allows the mast to self-deploy without motors that add weight, cost and complexity.

Alternatively, lattice masts can be in the form of articulated trusses composed by rigid segments hinged together.

New units design of this class of mechanisms will be scaled from existing designs, focusing on using proportionally stronger structural elements while minimizing the weight of all other components.

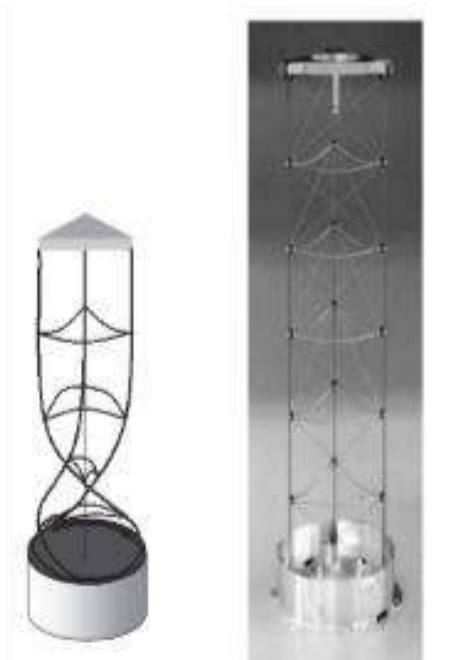


Figure 3.11. Deployable mast

3.1.9 Cold hibernated elastic memory (CHEM)

The material technology called “cold hibernated elastic memory” (CHEM) utilizes shape memory polymers in open cellular (foam) structures. The CHEM foams are self-deployable and use the foam’s elastic recovery plus their shape memory to erect structures. The key to this technology lies in the shape-memory-polymer-material systems. These materials behave very differently, depending upon whether they are above or below the glass transition temperature (T_g). Above T_g these materials are flexible and rubbery; below T_g they are glassy and rigid. Most important of all, structures formed initially below T_g “remember” their shapes and sizes through successive warm/cold cycles and, if unconstrained, return to their original shapes when warmed up. Thus, a structure can be formed below T_g , warmed above T_g to make it flexible, folded or rolled for stowage, cooled below T_g so it can be stored in the compressed state without external forces. Then, it can be transported to space, warmed above T_g to allow it to self-deploy back to its original shape and cooled below T_g to rigidize it for use.

CHEM advantages are: low mass and stowage volume; high reliability and low cost; self-deployment and simplicity; high dynamic damping and clean deployment and rigidization; none long-term stowage effects and ease of fabrication; impact and radiation resistant and thermal and electrical insulators.

The biggest disadvantage of CHEM structure is that heat energy is needed for deployment, but the solar heating deployment appears to be feasible.

Some possible space applications are: self-deployable wheels for mobility system, horn and radar antennas, sensor delivery system and precision soft lander.

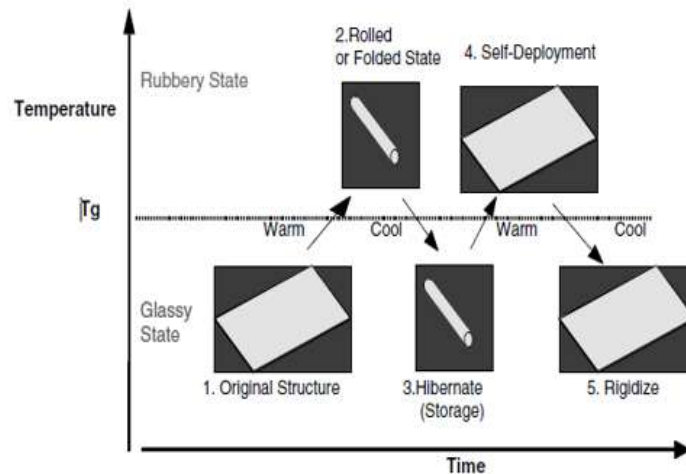


Figure 3.12. CHEM processing cycle

3.1.10 Air driven mechanisms

Inflatable technology has been used in aerospace applications over many decades, as the ability to deploy large structures with a low mass and very high packing efficiency has always been a requirement in the space industry.

For most space inflatable structural systems, the basic building-block structural elements are long, tubular beams and struts that are commonly known as inflatable booms. When being stowed for launch, these inflatable booms are generally flexible and can be rolled up or folded up to achieve high packing efficiency. After reaching the desired orbit, the stowed booms will be inflated and deployed by internal pressurization to attain their stiffness and design configurations. For a space mission that lasts only a few days, post-deployment rigidization may not be needed and maintaining a constant internal pressure will stabilize the booms. However, for missions of longer lives, it is necessary to rigidize the deployed booms by using one of many space rigidization methods for long-term space survivability. Therefore, depending on the need of space rigidization, a space inflatable boom can be classified as either a pressure-stabilized boom or an inflatable rigidizable boom.

The pressure-stabilized booms are typically constructed with thin polyamide films or urethane-coated fabrics. On the other hand, a space rigidizable boom can be further classified as resin-rigidizable or self-rigidizable. A resin-rigidizable boom typically consists of three elements, the bladder, fabric layer and outer enclosure. The bladder and the outer enclosure act as pressure barrier and restraining layer, respectively. The fabric layer (or layers) of a space inflatable/rigidizable boom is commonly made of woven graphite, Kevlar, or Nylon fabric and impregnated with a space-curable resin such as hydro-gel, thermal set, or UV-curable. The self-rigidizable booms, typified by those constructed with aluminum laminates do not rely on the use of space curable resins. An aluminum laminate boom will be subjected to two-step inflation pressurization in space. After being deployed by the first inflation, the boom will be subjected to a second inflation at higher internal pressure and its aluminum layer is stretched beyond the material yield point. After the inflation pressure is vented, space rigidization of the aluminum laminate boom is achieved by deriving stiffness from plastic deformation of the aluminum layer. Due to many inherent advantages, such as not requiring space power, low contamination and out gassing, and simpler construction, the aluminum laminate booms have received much attention for application to future space missions.

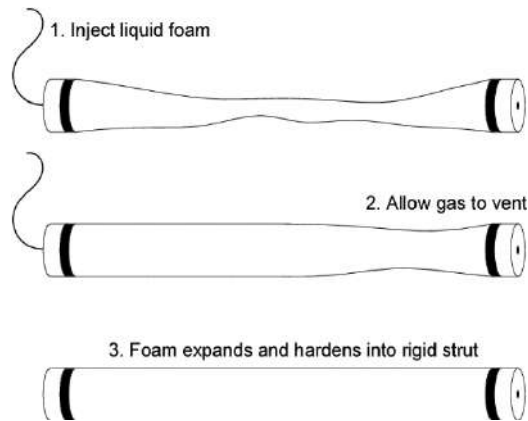


Figure 3.13. Inflatable booms expansion

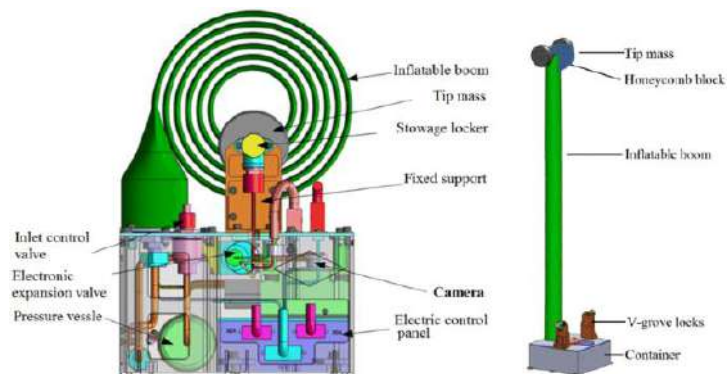


Figure 3.14. Inflatable booms deployment

3.1.11 Spring mechanisms

Spring loaded mechanisms are simple devices that transform stored elastic energy in mechanical work when released. Springs can be considered one shot devices in the absence of a repositioning mechanism, and the actuation of the spring needs a secondary system (pyrocutter, block pin, actuators, etc.) that allows the movement from the high energy compressed state to the expanded state. Springs are reliable and allow a diverse set of more complex mechanisms to work (I.e.: clamps) because they can store a high amount of energy in a small and lightweight configuration (usually made with metals or polymers).

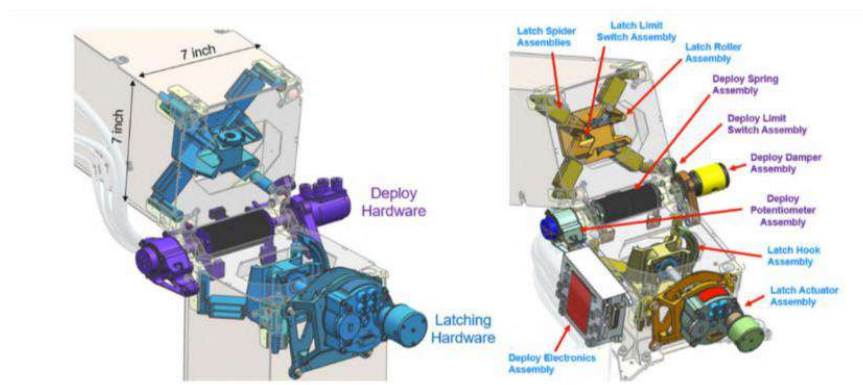


Figure 3.15. Appendix deployment using springs

The figure shows the NISAR hinge that deploys an appendix from the main body using a spring wrapped around a hinge. In this configuration. Each spring is supported by an aluminium outer mandrel which in turn rotates on glass-filled Teflon bushings on an inner mandrel, torsion spring solution is advantageous where limited volume and high output torque are needed for the deployment mechanism.

Nevertheless, metallic springs suffer from the space environment variation of temperature, and in some cases, they need to be lubricated to achieve maximum performances. The twisting needs to be studied in order to avoid deformations and torque variations.

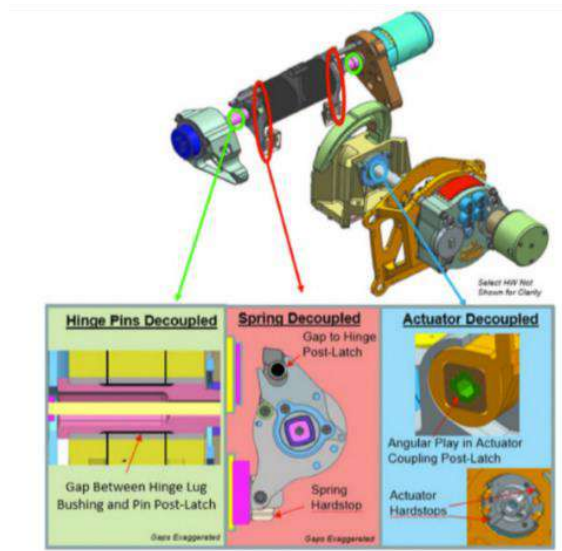


Figure 3.16. Spring deployment mechanism

3.1.12 Tensegrity structures

To improve packaging efficiency and avoid a high number of joints with complicated mechanisms, a concept of deployable tensegrity structure may be a good candidate. Tensegrity structures are internally prestressed, free-standing pin-jointed networks, in which the cables or tendons are tensioned against a system of bar or struts. The great advantage of this type of structure is that the compressive elements are disjointed. This provides the possibility to fold these members and hence the structure can be compactly stowed. Moreover, these structures have the ability to passively (i.e. without active control) achieve the required surface accuracy.

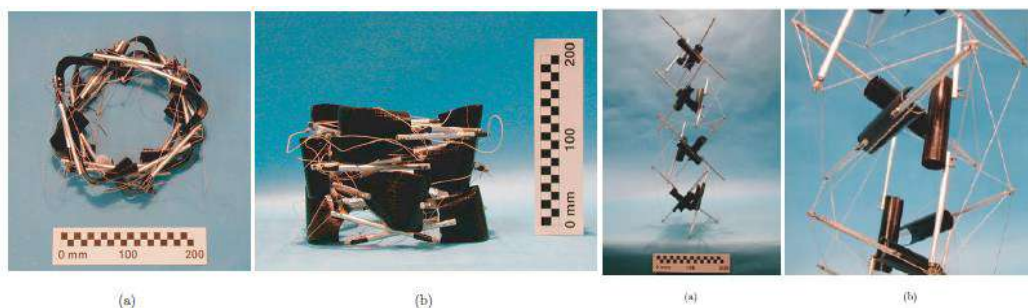


Figure 3.17. The stowed and deployed four-stage mast with bi-stable struts

An example of tensegrity structures for space field is a reflector antenna concept based on the tensegrity and tension truss concepts.

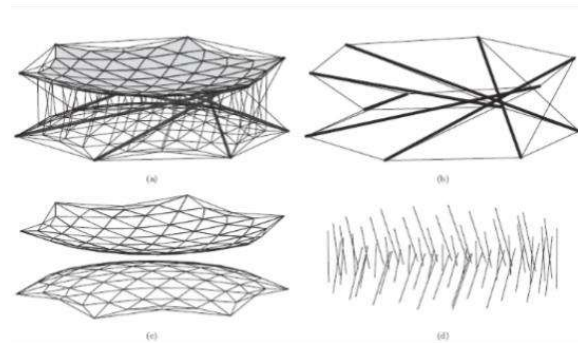


Figure 3.18. The new tensegrity reflector concept for small satellites

3.1.13 Tape springs

Interested readers are referred to Chapter 2, where tape springs and other deployable (mainly composite) booms have already been discussed.

3.1.14 A tape spring hexapod for deployable telescopes

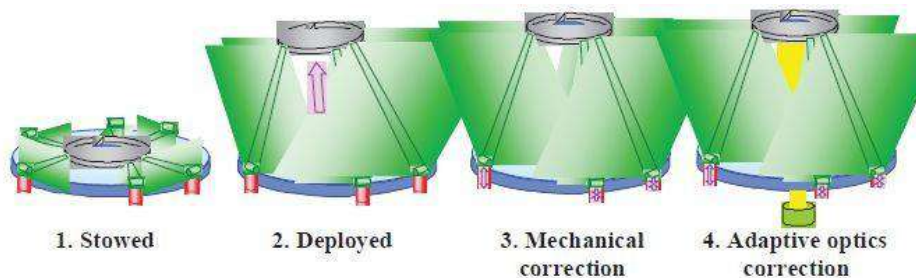


Figure 3.19. Mirror deployment

An hexapod based telescope concept whose legs are deployable has been investigated by L. Blanchard, G. Aridon, F. Falzon, D. Rémond, R. Dufour in order to stow the secondary mirror during launch and to self-deploy it in orbit thanks to a tape-spring based mechanism.

During the launch phase the tape springs which are coiled around a hub provide a strong volume reduction.

These mechanisms contain a rotating roll module in which the tape-spring is flattened and coiled. Because of this induced stress, the tape-spring naturally tends to go out to recover its natural curved section. Inside the roll module, a fixed axis is connected to the cradle by a ball bearing. The coiling device has an internal coil-spring which helps the self-deployment when the locking system is released. The coilers also have an internal deployment length tuning mechanism which was used to deploy the same length on each leg. The active wrist concept used for this hexapod implies the use of one spherical joint at the top and one universal joint at the base of each leg which means that 2 DOF are required at the base of the tape-spring and 3 DOF at its top. For the base junction, and hinge made of two ball bearings and the coiling of the tape-spring have been used as two degrees of freedom in rotation. The top junction is constituted with an axis mounted on two ball bearings, a thin metallic flexural blade and with the torsion of the tape-spring itself.



Figure 27: Deployment mechanism



Figure 28: Tape springs & hinges



Figure 29: Tape spring hinge



Figure 30: Uncoiling actuator

Figure 3.20. Tape spring actuator and constituents

4 CONTROL CONCEPTS

In order for satellites or space vehicles, like space telescopes, to accomplish their mission, their orientation and position in space often require extremely precise management: this is performed by onboard control systems.

A control system manages, commands, directs or regulates the behaviour of other systems using control loops.

In a space telescope the control concept includes: the attitude control to achieve the right position and orientation of the whole system and the pointing control of the internal mirrors to point the observatory maintaining the stability of the optical system with strict requirements.

Both systems depend on each other and rely on use of a set of sensors and control laws implemented with software and electronics.

The report deals with all the features mentioned.

4.1 ATTITUDE DETERMINATION AND CONTROL SYSTEM (ADCS)

Attitude and determination control system is one of the most important systems to consider for the design of a space mission. Different configurations of this system could be necessary depending on the mission and its purposes. Two key features must be considered when dealing with ADCS: attitude and attitude rate. These two quantities describe the rotational motion of the spacecraft around its center of mass and they are the subject of attitude analysis or spacecraft dynamics.

The attitude control is generally occasional, but it could be continuous when facing a problem like a failure.

4.1.1 Actuation systems

The task of the control function is to provide the force and torque commands which will be executed by the reaction control system of the spacecraft to correct deviations of the current state vector from the nominal one.

The motion control is always needed because we have at least to counteract natural and artificial disturbances, like atmospheric drag, gravitational force, solar pressure, magnetic dipole, or even to provide antenna pointing or payload pointing.

In our case of study, we must develop a space telescope, so our attitude control system must also achieve precise pointing requirements and maintain relative orientation, like in a satellite constellation.

Attitude control systems can be classified in two categories:

1. Passive systems.
2. Active systems.

4.1.2 Passive systems

Those systems use disturbance torques like aerodynamic drag, Earth's magnetic and gravitational field, solar pressure etc. as stabilization forces to control the spacecraft. So, this control methods use the interaction between the spacecraft and natural phenomena happening in the operational environment (that depends on the specific mission).

Examples of passive methods are:

- Spin stabilization: the gyroscopic stiffness of the spinning body is used to maintain its orientation in an inertial reference.
- Gravity – gradient stabilization: a spacecraft in a reasonably low orbit will tend to stabilize with its minimum – inertia axis along the local vertical orientation. So, the vehicle must have an axis such that the inertial moment around it is considerably lower than the inertial moment in the other directions.
- Magnetic methods: permanent magnets can be used to align one of the body axes with the lines of the Earth's magnetic field.

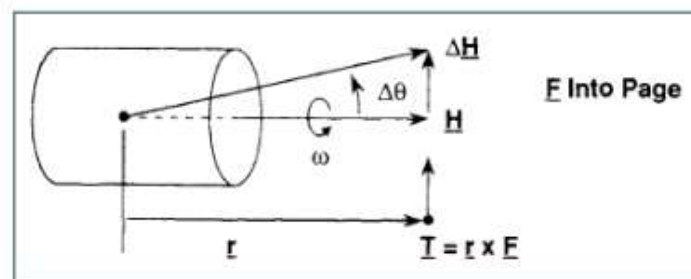


Figure 4.1. Spin Stabilization

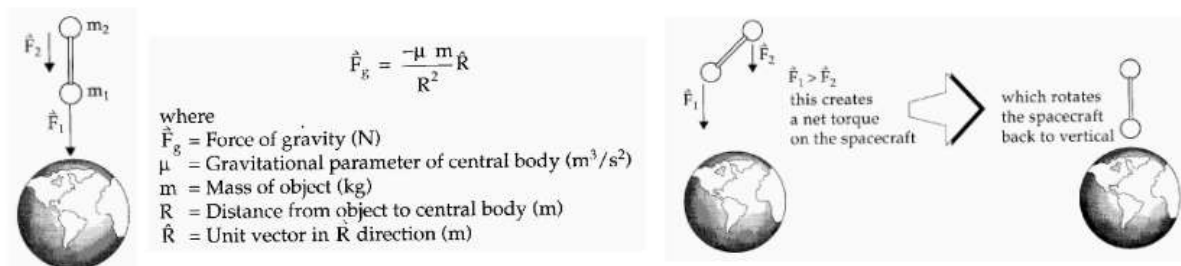


Figure 4.2. Gravity gradient stabilization

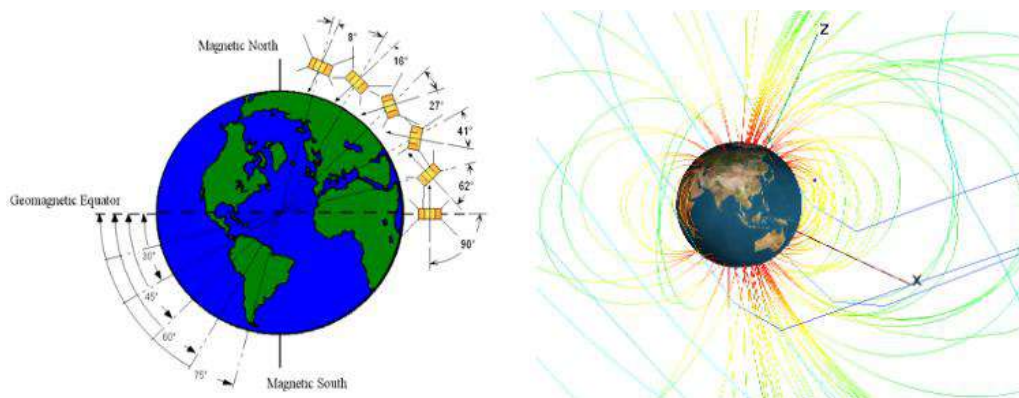


Figure 4.3. Magnetic stabilization

The major advantage of passive control systems is the ability to attain a very long spacecraft lifetime, not limited by onboard propellant. Main disadvantages are the low pointing accuracy that can be reached and the impossibility to react in response of external events. Moreover, natural phenomena can change over time.

4.1.3 Active systems

Those systems are based on the concept that attitude is measured and compared with a desired value. The error signal so developed is then used to determine a corrective action through control torque to generate a maneuver by means of the onboard actuators.

Clearly, active control systems give a very high pointing accuracy and a very high attitude rate can be obtained. Disadvantages are the cost, the technical complexity and the limited lifetime.

Example of active methods are:

- Reaction control jets;
- Reaction wheels;
- Momentum wheels;
- Control moment gyros;
- Magnetorquers.

4.1.3.1 Reaction wheels

The Wheel Drive Electronics controls a motor attached to the wheel. If the motor brakes the wheel, slowing it down, the satellite will start to rotate in the same direction as the wheel. If it accelerates the wheel, then the satellite rotates in the opposite direction.

Note that you cannot spin the wheels faster forever. At some point you must slow them down and start over. Of course, slowing the wheel will make the spacecraft turn in the opposite direction, and this is not desirable once the spacecraft is pointed in the right direction. That is where the reaction control system comes into action. Jets/thrusters or similar forces are used to push against the force/torques resulting from slowing down the wheels in a procedure called "angular momentum desaturation." The two opposing forces essentially cancel each other out so that the spacecraft maintain the intended orientation.

Reaction wheels do not provide any satellite gyroscopic stability, so they are usually smaller and spin slower than momentum wheels. There wheels are often implemented (one per axis). A fourth one is used in redundant configuration (tetrahedral configuration).

Advantages

No fuel consumption; power efficient

Disadvantages

Need of periodic momentum – dumping; micro vibrations can be generated

4.1.3.2 Control Moment Gyros

This actuation system consists of a spinning rotor and one or more motorized Cardan suspensions that tilt the rotor's angular momentum. As the rotor tilts, the changing angular momentum causes a gyroscopic torque that rotates the spacecraft.

So, CMG consist of a spinning wheel with an angular velocity of ω rotating around the x-axis. If the spinning wheel has a moment of inertial, then the angular momentum produced would be $h=I\omega$ along

the x axis. Rotating the spinning wheel about the gimbal axis z at a rate of δ , would then result in a large torque output about an axis normal to the angular momentum axis and the gimbal axis.

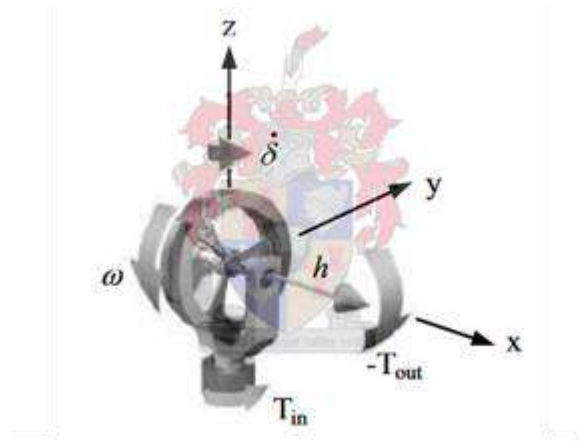


Figure 4.4. Control moment gyro

CMG provide high values of torque (over 100 Nm), high agility and authority control. Moreover, by turning the spin axis of the wheel a torque can be produced in the other directions. The spin axis control itself can be performed by changing the wheel speed as normal

4.1.3.3 Magnetorquers and rods

Magnetorquers and magnetic rods generate a dipole by sending electrical current through a coil. The interaction with a magnetic field generates a torque perpendicular to the plane on which the coils are placed. The torque magnitude depends on the number of coils (N), the area of the actuator (A) and the intensity of the current that flows in the magnetorquer (i). Magnetic rods are coils wound up around a ferromagnetic core that amplifies the dipole moment generated by the coils when current flows through them. However, the ferromagnetic core introduces a residual dipole moment that must be taken into account.



Figure 4.5. Magnetic rods (left) and square magnetorquer (right)

Advantages

no fuel expelled; generation of an external torque (can dump momentum).

Disadvantages

application restricted to LEO; small amount of torque provided; it cannot generate torque around the geomagnetic field vector typically needed at high inclination orbit.

4.1.3.4 Thrusters/jets

Thrusters can be used to control attitude but at the cost of consuming fuel. They are often used for momentum dumping, large (but not fine) maneuvers, detumbling of large spacecraft, station keeping in GEO and orbit maneuvers.

A three axes complete control requires at least 6 thrusters (12 or more for a more precise control) for attitude control and at least 8 (16 or more for high performance) for orbit control.

Different categories of thrusters/jets are commonly used:

- Cold gas: pressurized gas is accelerated through a nozzle;
- Solid: solid fuel is burnt and accelerated through a nozzle;
- Monopropellant (hydrazine): liquid fuel is decomposed into a gas through heat and a catalyst then is burnt and accelerated through a nozzle;
- Liquid bipropellant (LOx, Kerosene or H₂): a fuel and an oxidizer are mixed and produce a gas then accelerated;
- Electric (arcjet, resistojet, ion plasma, etc....);

Advantages

Fast; powerful

Disadvantages

Attitude/translation coupling; fuel consumption; complexity of the system; safety issues, reaction by-products may interfere with the field of view

4.2 SENSORS

Sensors are used for attitude and orbit determination and they can be:

- Star sensors;
- Sun sensors;
- IRES;
- Magnetometers;

4.2.1.1 Star sensors

This type of sensors can use scanning or tracking. The first one is used on spinning spacecrafts and it exploits the scanning of several stars in view around the spacecraft to determine the attitude. The second one instead is a camera that takes photos of the stars and compares the acquired photos with others in a catalogue and it allows attitude control system to exploit the offsets among the photos to determine the angular position and correct it.

The problems with the star trackers are:

- Blinding -> they cannot be used with Sun/Earth/Moon inside the field of view;
- "Holes" in the sky -> the sensor could track a portion of sky with no stars;
- Sensitivity:
 - Too sensitive -> too many stars are detected.
 - Too insensitive -> an insufficient number of stars is detected.

Advantages

High accuracy

Disadvantages

Expensiveness, high mass and volume

4.2.1.2 Sun sensors

Sun sensors could be:

- Analog -> this sensor is based on materials that allow current passage when they are illuminated by Sun radiation. Through electric output it is possible to evaluate the Sun incidence angle and so the attitude of spacecraft when Sun-vector is known.
- Digital -> this sensor is composed of an appropriately pierced box that lets the sun radiation passes to create the output signal that is interpreted as a binary signal.

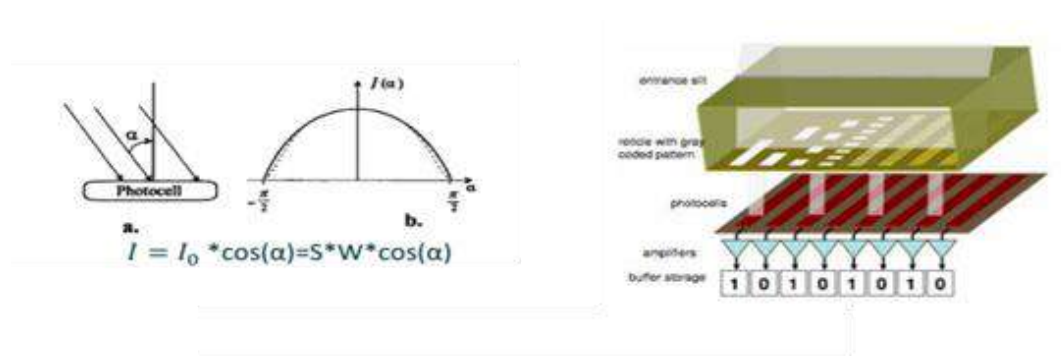


Figure 4.6. Analog (left) and digital (right) sun sensors

4.2.1.3 IRES

An Infra-Red Earth Sensor works by exploiting the gradient of temperature between deep space and the warm Earth. It is composed of several bolometer traces.

In nominal conditions IRES detects a constant temperature gradient of all bolometers and so the attitude of the spacecraft does not have to be corrected. However, when the attitude is not correct the bolometer tracers see some differences in the temperature gradients, and the attitude is corrected until the gradients are the same again.

The problems of this sensor are the external sources of heat (e.g. Sun or Moon) that can interfere with the measurement. In fact, when the Sun is inside the "field of view" of a bolometer, a peak of temperature occurs. This difference of temperature is detected from sensor and the ACS adjusts the attitude even when correct. To avoid this problem, the measurements of the bolometers known to be flashed by sunlight are ignored.

Advantages

High performance.

Disadvantages

Expensiveness, high complexity.

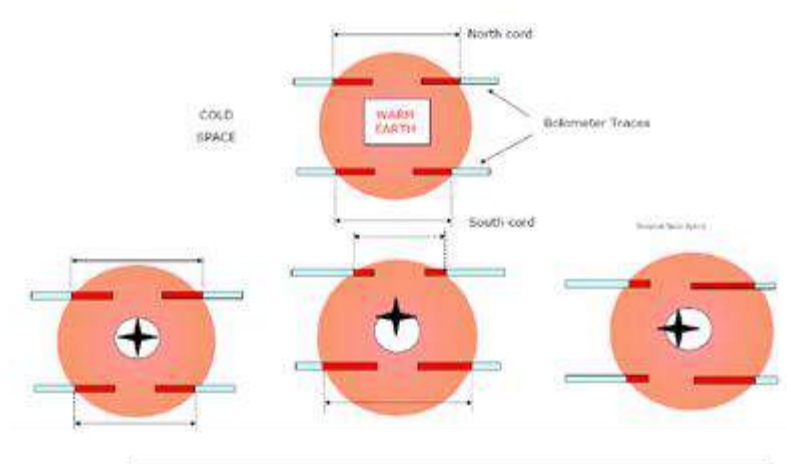


Figure 4.7. IRES in different “position”

4.2.1.4 Magnetometers

These sensors measure both the direction and the magnitude of Earth’s magnetic field. These measurements are compared to a set of data so that the ACS can adjust the angular position of the spacecraft.

Advantages

High simplicity.

Disadvantages

Noise, misalignment, non-linearity.

4.3 MIRROR CONTROL MECHANISMS

When it comes to high resolution imaging, a precise attitude determination and control system is required, but it does not solve the whole problem. Precise pointing does not imply precise imaging. Telescope’s mirrors and their frame are subjected to mechanical stresses, thermal expansion and contractions, manufacturing errors, misalignments, and many other different external disturbances. In general terms, a very important dimensional stability of the structural elements which connect optics together is required for high resolution observations.

Nowadays, two kinds of solutions are used to carry this out:

- To use specific materials with thermoelastic and hygroelastic sensitivity almost negligible and setting up integration procedures with fine metrological adjustments
- To exploit in-flight adjusting devices able to correct the optical defects during the mission

These latter solutions are active methods (also referred to as active optics) and we generally say they are “Mirror Control Mechanisms” (MCM). These mechanisms have the purpose of correcting, through proper actuators, the focusing and the tilts of the mirrors, in order to prevent aberrations which are generally detected by a wavefront sensor (which is a device that specifically measures aberrations in optical wavefronts) which, in turns, sends a signal to the actuators in a closed loop feedback.

While aberrations come from primary-mirror-related issues, blurring and coma (i.e. defects in the images for example due to lens imperfections) depends on a misalignment between primary and secondary mirrors. It is clear that while actuators on the back of the primary mirrors can be placed without too many constraints, when it comes to secondary mirrors, particular care must be taken: it is necessary to be sure that whatever mechanism it's used, its obstruction does not occult the field of view of the telescope, which implies that the whole frame containing actuators and their electronics, must remain in a cylinder of diameter equal to the secondary mirror's one and must be placed behind it.

4.3.1 Actuators requirements

When it comes to the selection of the actuators, different aspects must be considered. First, these devices must have a high output energy to mass ratio since the electric power available onboard is can be as limited as 100W in a small satellite and the mass must be as little as possible. On the other hand, the MCS must withstand launching vibrations, which can be challenging considering levels of vibrations as high as 20g rms. Moreover, the spacecraft is exposed to an extended range of temperatures that could vary between -150°C to 150°C or even wider for particular missions. This last condition could lead to difficulties in getting the heat off the actuators because of the vacuum of space, which also may cause outgassing near optics. Also, radiation resistance must be taken into account.

4.3.2 Mirror control with piezoelectric actuators

One of the most used type of actuators that satisfies all the requirements above is the piezoelectric actuator. A piezo actuator is based on the piezoelectric effect: applying a small voltage to a crystal such as quartz, the crystal shape is changed, either expanded or contracted, in a way that, once the tension is fixed, only depends on the initial shape of the crystal.

The actuators based on this effect offers high positioning resolution, high stiffness, low power consumption and direct drive linear motion. It is also possible to design piezo ceramics to operate in cryogenic temperatures as the James Webb Space Telescope requires.

One of the main drawbacks of piezoelectric actuators though is that their displacement is limited to some tens of μm . In order to overcome this limit, Cedrat Technologies, a high-tech French SME leader in this sector, has engineered the solution showed below.

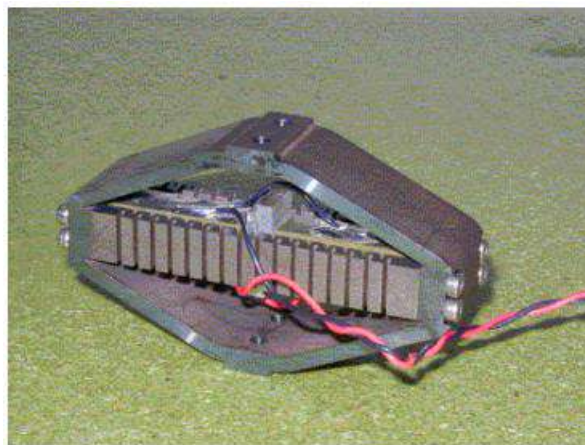


Figure 4.8. Cedrat piezoelectric actuators APA120ML-PP

It consists in a longitudinal piezoelectric actuator which integrates an elastic mechanical amplifier that consists of an elliptic shell. It is necessary to notate that the amplification must not be too high in

order to maintain a good stiffness. This solution is commonly referred to as “Amplified Piezo-electric Actuator” (APA).

This type of actuators has already been used in several important missions such as:

- Rosetta (ESA), for the Micro-Imaging Dust Analysis System (MIDAS) which uses 8 APA50Ss and 1 Parallel Prestress Actuator PPA10M
- Mars Science Laboratory (MSL, NASA), exploiting tens of APAs.
- AELUS (ESA), for the LIDAR Atmospheric LAsER Doppler Instrument (ALADIN)

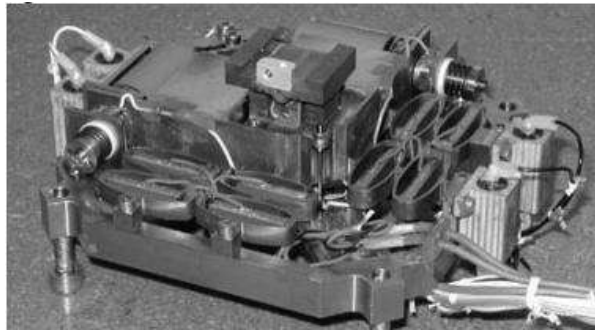


Figure 4.9. Rosetta MIDAS APAs

Let us now analyze a possible concept of application of these actuators to provide control to the secondary mirror assembly.

As mentioned before, secondary mirror control (SMC) must compensate for misalignments and focus related issues. Considering a classical Cassegrain telescope as depicted below, if the actuators could be placed at will, it would only be necessary to control 3 DOF to secure the precise high-resolution imaging (2 tilts and the refocusing). Instead, the SMC must be placed behind the mirror due to the constraints above. This implies that the kinematics of the SMC will be off set with respect to the optical center around which the corrections must be done. Hence the need of two additional DOF (X and Y translation in the picture below) to ensure any rejection is compensated.

The mirror positioning control is performed through a set of dedicated capacitive sensors.

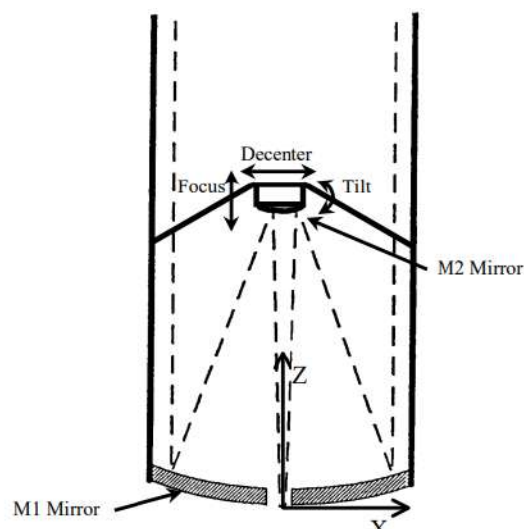


Figure 4.10. SMC concept

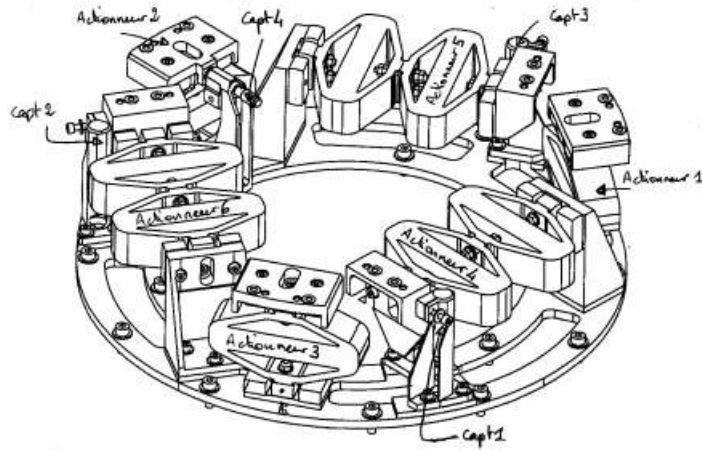


Figure 4.11. SMC APAs

A state-of-the-art concept for the SMC could implement the piezoelectric actuators as shown in the following picture:

This configuration uses 3 vertical APAs to correct focusing and provide tilting while three horizontals couples of APAs generate transverse movements. This displacement ensures the best stiffness around the z axis. Five capacitive sensors then provide information about the correct position of the mirror for the closed loop control. Such kind of solution can provide positioning accuracy in the order of some tenth of μm and some μrad .

$$e_1^2 = 1 + 2f_1'^2 (f_1' - t) / t f_1'^2$$

$$e_2^2 = \left(\frac{f' - f_1'}{f' + f_1'} \right)^2 + \frac{2f' f_1'^3}{t(f' + f_1')^3}$$

Figure 4.12. RTC focal lengths criterion

The classical Cassegrain configuration is nowadays evolved in the so-called Ritchey-Chretien telescopes (RTC). RTCs are a specialized variant of the Cassegrain in which both the primary and the secondary mirror are hyperbolic. This helps to reduce off-axis optical errors as ideally the mirrors' eccentricities combine to avoid no spherical aberrations or comas (even third order defects are avoided in this configuration). In RTCs the focal lengths of the mirrors must satisfy the following equations:

4.3.2.1 Deformable Mirror for Space Telescope

Another state-of-the-art solution is offered by the *Unimorph Deformable Mirror based on Piezoelectric Actuation*. *Deformable Mirror (DM)* technology is one of the most important devices of active optics, offering fundamental advantages in term of:

- Compensation of *wavefront distortions* induced by external disturbances
- Minimization of the ocular *aberration*

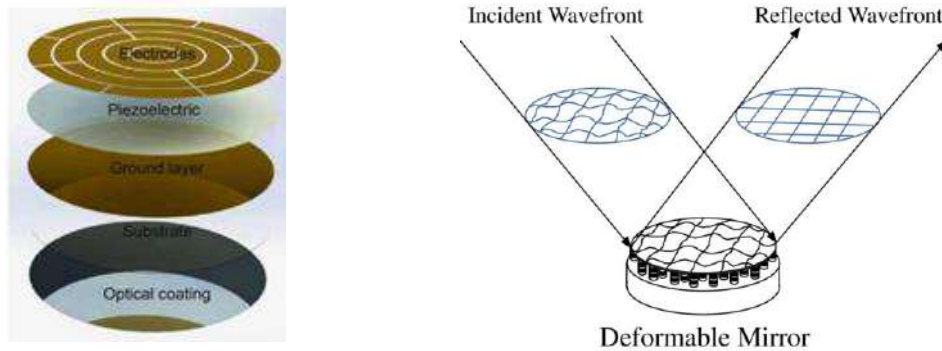


Figure 4.13. The image above on the left shows some examples of electrodes distribution of a 'deformable mirror' DM with piezoelectric actuation. On the right the effect of a deformable mirror DM in wavefront distortion.

The mirror concept is based on the *unimorph principle*. This kind of mirror is able to maintain its shape during change of ambient temperature and differently from more common actuation principle using out-of-plane forces, its shape is induced by in-plane-forces.

The unimorph principle has no need for a fixed based plate as support for electromechanical actuators thus resulting in lightweight and simple configuration of the mirror and actuator.

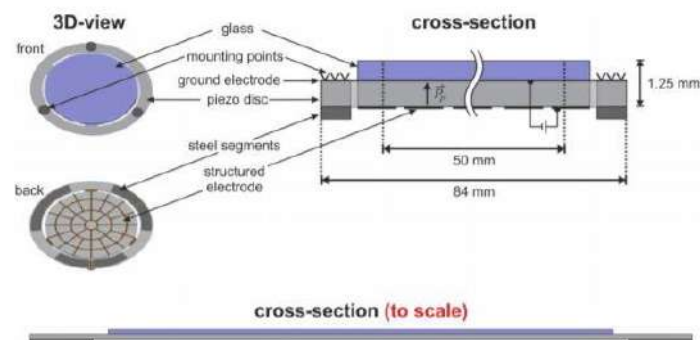


Figure 4.14. At the top the 3D-view of the mirror structure with a cross sectional view where PP represents the Piezoelectric Polarization. At the bottom there is a cross sectional view drawn true to scale.

Fig. 4.14 shows the classical structure of a deformable mirror with piezoelectric actuation. *Unimorph DM* is made of a piezoelectric disc with a polarization $P \rightarrow P$ perpendicular to the mirror surface. The disc is sandwiched between two metallic electrodes and a super polished and dielectrically coated glass is glued onto the side of the piezoelectric disc.

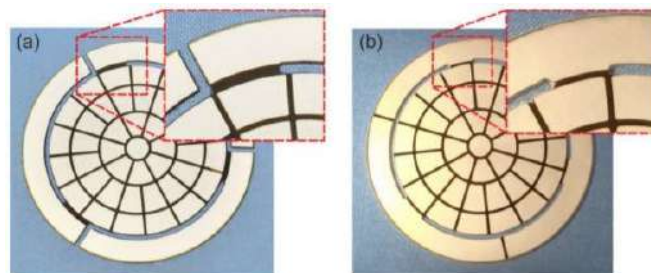


Figure 4.15. The image above shows two examples of piezo structure (piezoelectric disc) with the back-side electrode pattern.

When a voltage is applied to the electrodes the piezoelectric disc deforms parallel and perpendicular to the piezoelectric polarization $P \rightarrow P$ due to the *inverse piezoelectric effect*. Thanks to the perpendicular polarization (proportional to the piezoelectric coefficient of the material) is possible to induce stress between the disc and the glass thus resulting in a local deformation. It is self-evident that the *connecting joints* of the deformable structure have to be chosen carefully in order to adequate the dynamic response of the system to vibrations requirements, but at the same time allow the mirror to deform with manageable forces.

4.3.2.2 Piezoelectric Actuation issues and solutions

One of the main problems of piezoelectric actuator applied to deformable mirror is the requirement of surface *fidelity*. During the manufacturing process, the thin layer of glass is subjected to *coating induced stresses* which result in a spherical deformation that must be compensate with a *stress-compensation layer* applied to the opposite side of the glass substrate. It has been demonstrated that when the mirror is unpowered the deformation is greater than the powered configuration.

Another concern is connected to the *hysteretic behavior* of piezoelectric materials. The piezoelectric disc has a polarization non homogeneously oriented since it is a polycrystalline PZT (like PZT ceramics) material where each grain is divided into domain with different polarization. When poled, the grain directions are all oriented in the poling direction, however there are some domains which are oriented unfavorably that can be easily reoriented while other require strong fields to reorient, this lead to the *hysteretic behavior*. The domains in the area between two electrodes are not reoriented during actuation hence in these areas, the disc will deform differently from the areas under the effect of an electrode.

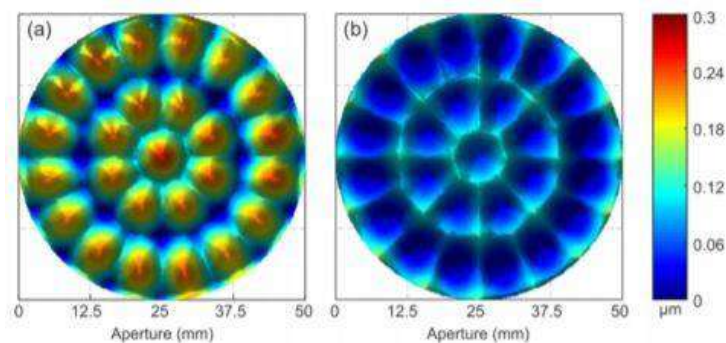


Figure 4.16. On the right side there is a numerical representation of different strains due to domain reorientation. On the left a numerical representation of negative strain around the electrodes.

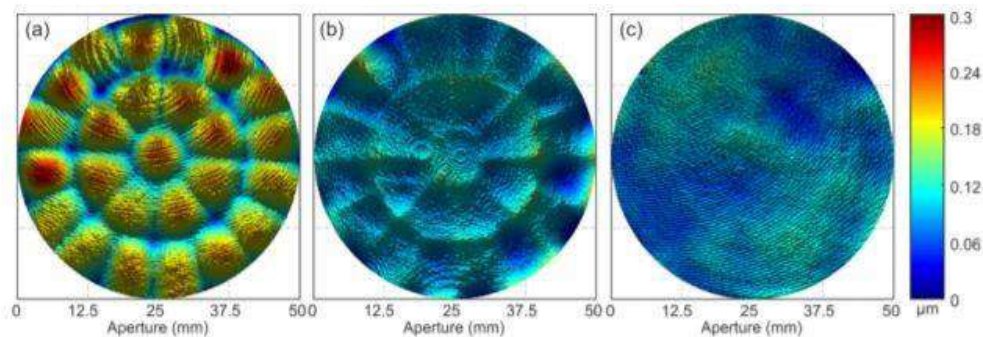


Figure 4.17. The figure above from left to right represents the residual surface deviation from best sphere of three deformable mirrors. In figure (c) the print-through is no longer visible thanks to pre-load.

As represented in Fig. 4.17, it is possible to induce a deformation with opposite sign if the piezoelectric material is subjected to a strong electric field in the pooling direction (is 'pre-poled') before the application of the mirror's glass. This result show that it's possible to steer in both directions (positive or negative) the *print-through* (inhomogeneous strains due to electrodes position). Hence it should be possible to find the right intensity of pre-poled electric field to realize a *DM* without print-through.

4.3.3 Primary mirror control with mechanical actuators

The most advanced mechanical actuators used on space telescopes for precise maneuvering of the primary mirrors have been designed by Ball Aerospace and are implemented on the James Webb. They are specifically designed to both meet precise mirrors control specifications and to provide wide coarse movements for the deployment of the telescope. To do this, two important features are implemented: fine stage flexure and coarse drive coupling.

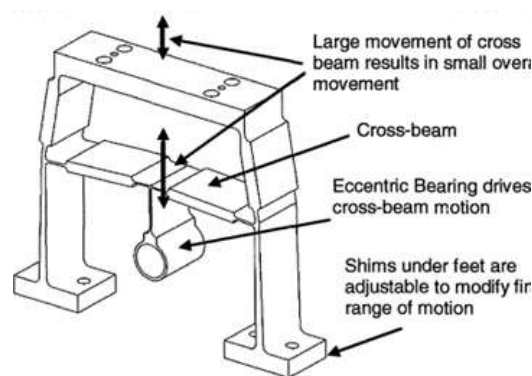


Figure 4.18. Fine stage flexure concept

The picture above shows the concept of the A-shaped fine stage flexure. It works the opposite way the piezoelectric actuator with amplification worked: this time, a wide and coarse movement provided by the electric stepper motor that drives the cross-beam motion through an eccentric bearing, must be transformed into a fine controlled movement through staging: the cross-beam motion deflects the side beams which, in turn, change the height of the flexure fine producing a sinusoidal displacement patterns as the cam shaft rotates.

The precision reached with this solution is in the nanometer-range.

The whole mechanism is able to operate at 30K to meet JW operational temperature requirements and all bearings and gears must be properly coated with a special lube.

The fine displacements provided by these actuators cannot match the wide movement necessary to deploy the telescope once it is released from the fairing after launch. An option would be to equip the mirror with different actuators for this maneuver but doing so additional mass would be installed onboard. As the least-weight principle is adopted in all space applications to reduce the costs, the more complex but convenient solution adopted on JWST is the coarse drive coupling.

The coupler consists of two rotating disks that can both engage or disengage depending on the drive direction. When the motor is in a direct drive, the two pins on the disks are engaged and a coarse motion is permitted exploiting the whole drive power of the motor. Instead, when the engine is driven in reverse the pins back away and the coarse motion is decoupled from the drive train, leaving only the fine stage engaged for precise positioning.

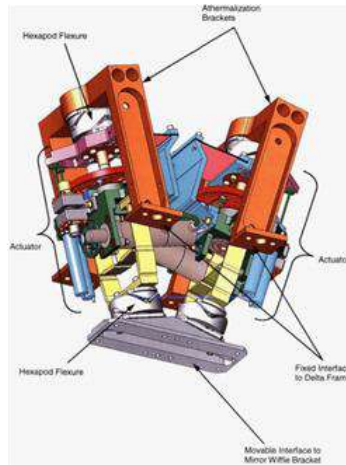


Figure 4.19. JWST's cryogenic actuators concept

4.3.4 Thermal refocusing

Space environment could be particularly challenging for spaceborne high resolution optical imagers due to vacuum and thermal excursions. In order to let optical and electrical systems work at their best, it is fundamental to maintain the temperature within a desired optimal range. So, thermal distortion is one of the operational issues. However, in orbit, thermally driven geometric variations are reduced by using nonexpanding Carbon Fiber Reinforced Polymer (CFRP) structures that also reduce weight and increase stiffness. Unfortunately, CFRP materials have an intrinsic dimensional instability, mainly as a result of moisture absorption during manufacturing processes. A humid environment and moisture absorption can elongate dimensions of the structure. When the aligned camera is sent into a vacuum environment, moisture leaves the body and the structure shrinks. Very precisely aligned camera optics can defocus after a few days in the vacuum. Moreover, stress relief, mechanical settling effects, internal micro cracking, and radiation damage can affect the camera structure and cause defocus in the camera. These structural variations perturb the alignment, and the optical system becomes remarkably defocused. A few microns displacement along the optical axis in the secondary mirror causes measurable decay in the image quality metrics. The camera design is much more sensitive to deformations along the optical axis rather than in other directions. So it is possible to tolerate non axial deformations caused by launch vibrations or recontaminations by means of allowable tolerances.

For these reasons, it is essential to use refocus mechanisms such as mechanical, piezoelectric or thermal actuators. Thermal refocusing methods are based on the controlled thermal expansion of mechanical components in designated directions. These methods do not require any joints and are therefore beneficial in avoiding mechanism-related problems in orbit. Moreover, because of staying away from piezo or electromechanical actuators, no aging effect is expected. For example, Pleiades and SEOSAT cameras are equipped with this kind of thermal refocusing mechanisms.

The secondary mirror (M2) is very sensitive, and small displacement of the mirror can provide good focus compensation, for this reason usually refocusing actuators are mounted on this mirror. This mirror is selectively heated to control its radius of curvature which changes the focal distance of the optical system. By changing the focal distance, it is possible to compensate defocus in the camera alignment. Zero-expansion glass ceramics, i.e., ZERODUR (made of silicon carbide SiC), are very common for spaceborne camera mirrors. Instead of using a zero-expansion mirror substrate for the secondary mirror, it is also possible to use materials with relatively high thermal conductivities and thermal expansion coefficients, like aluminum, to provide dynamic responses to thermal changes.

Nevertheless, the primary, folding, and tertiary mirrors are made of zero-expansion glass ceramic material to get rid of thermal.

The radiative and conductive heat transfers are designed to be always from the M2 mirror toward the camera structure and to outer space. M2 is always losing heat, therefore, to compensate for heat loss, a dedicated heater and temperature sensor unit is mounted on the rear side of the mirror (M2). When the mirror is warmed up in a controlled manner, the radius of curvature will change linearly by regarding the basic thermal expansion equation as follows:

$$R' = R(1 + \alpha\Delta T)$$

where R is the initial radius of curvature, R' is the expanded radius of curvature, α is the linear thermal expansion coefficient, and ΔT is the temperature change. The following figure shows how this method works.

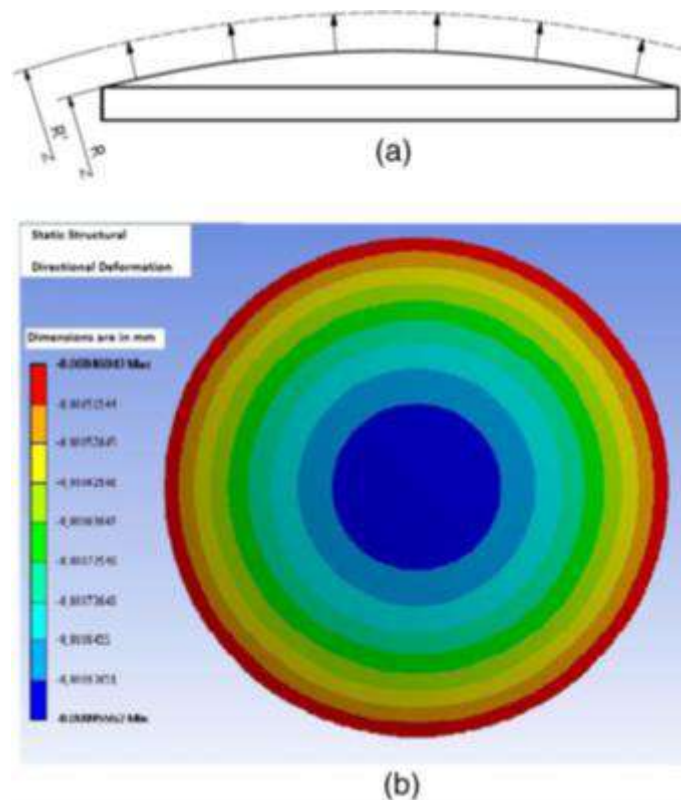


Figure 4.20. (a) Schematic description of the changing radius of curvature (b) FEM analysis shows expansion of the mirror for a 10°C temperature change

This control method is very robust and does not require any kind of extra equipment to refocus the camera. In addition, it does not require a launch lock device or any lubricants for operation. The mechanism also requires very low power for operation

4.4 CONTROLLERS

4.4.1 PID Controller

A proportional–integral–derivative controller (PID controller or three-term controller) is a control loop mechanism employing feedback that is widely used in industrial control systems and a

variety of other applications requiring continuously modulated control. A PID controller continuously calculates an *error value*: it automatically applies accurate and responsive correction to a control function.

The first applications in the aerospace industry was related to the control of electromechanical actuators in 1930's and nowadays 95% of aerospace industrial applications continue to be operated by PID controllers.

4.4.1.1 PID Theory

- P: is the proportional term that acts on the rise time but the steady state error is invariant
- I: is the integrative term that erases the steady state error but worsening the transitory response
- D: is the derivative term that decreases the overshoot improving the transitory term

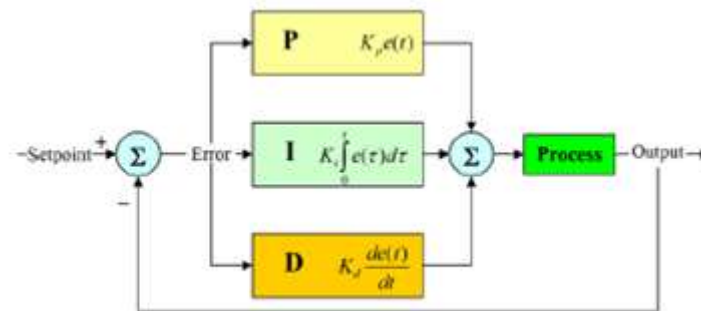
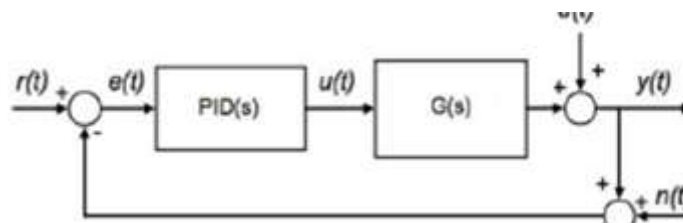


Figure 4.21. PID controller

Considering the following system



$$u(t) = K_P e(t) + K_I \int_0^t e(\tau) d\tau + K_D \dot{e}(t)$$

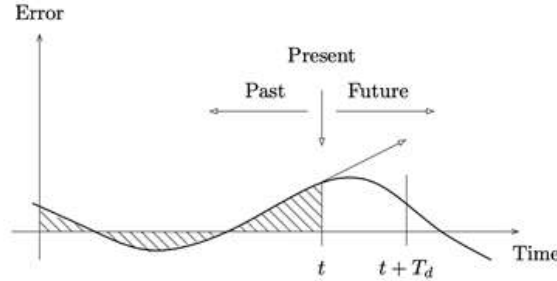
where

- K_P is the proportional gain (tuning parameter)
- K_I is the integral gain (tuning parameter)
- K_D is the derivative gain (tuning parameter)
- $e(t) = SP - PV(t)$ is the error (SP is the setpoint and $PV(t)$ is the process variable)
- t is the time or instantaneous time (the present)
- τ is the variable of integration (takes on values from time 0 to the present t)

Equivalently, the transfer function in the Laplace domain of the PID controller is

$$PID(s) = \frac{u(s)}{e(s)} = K_p + K_I \frac{1}{s} + K_D s$$

This type of controllers is based on past, present and prediction of future control error, as it is shown in the following figure.



The PID can be parameterized as

$$u(t) = K_p(e(t) + \frac{1}{T_I} \int_0^t e(\tau) d\tau + \frac{1}{T_D} \dot{e}(t))$$

where T_I is the integral time constant and T_D is the derivative time constant.

4.4.1.2 Proportional term

The proportional term produces an output value that is proportional to the current error value. The proportional response can be adjusted by multiplying the error by a constant K_p , called the proportional gain constant.

The proportional term is given by

$$P_{out} = K_p e(t)$$

A high proportional gain results in a large change in the output for a given change in the error. If the proportional gain is too high, the system can become unstable. In contrast, a small gain results in a small output response to a large input error, and a less responsive or less sensitive controller. If the proportional gain is too low, the control action may be too small when responding to system disturbances.

Because a non-zero error is required to drive it, a proportional controller generally operates with a steady-state error.

Steady-state error (SSE) is proportional to the process gain and inversely proportional to proportional gain. SSE may be mitigated by adding a compensating [bias term](#) to the setpoint AND output, or corrected dynamically by adding an integral term.

4.4.1.3 Integral term

The [integral](#) in a PID controller is the sum of the instantaneous error over time and gives the accumulated offset that should have been corrected previously. The accumulated error is then multiplied by the integral gain (K_i) and added to the controller output.

The integral term is given by:

$$I_{out} = K_I \int_0^t e(\tau) d\tau$$

The integral term accelerates the movement of the process towards setpoint and eliminates the residual steady-state error that occurs with a pure proportional controller. However, since the integral term responds to accumulated errors from the past, it can cause the present value to [overshoot](#) the setpoint value

4.4.1.4 Derivative term

The derivative term is given by:

$$D_{out} = K_D \frac{de(t)}{dt}$$

Derivative action predicts system behaviour and improves settling time and stability of the system.

An ideal derivative is not causal, so that implementation of PID controllers include an additional low-pass filtering for the derivative term to limit the high-frequency gain and noise.

4.4.1.5 Loop tuning

Tuning a control loop is the adjustment of its control parameters (proportional band/gain, integral gain/reset, derivative gain/rate) to the optimum values for the desired control response.

Stability (no unbounded oscillation) is a basic requirement.

PID tuning is a difficult problem because it must satisfy complex criteria within the limitation of PID control.

PID controllers often provide acceptable control using default tunings, but performance can generally be improved by careful tuning, and performance may be unacceptable with poor tuning. Usually, initial designs need to be adjusted repeatedly through computer simulations until the closed-loop system performs or compromises as desired.

4.4.1.6 Stability

The total loop transfer function is

$$H(s) = \frac{K(s)G(s)}{1 + K(s)G(s)}$$

where $K(s)$ is the PID transfer function and $G(s)$ is the plant transfer function.

The system is unstable when the closed loop transfer function diverges for some s and this happens when $K(s)G(s) = -1$.

Stability is guaranteed when $K(s)G(s) < 1$.

4.4.1.7 Tuning methods

There are several methods for tuning a PID loop. The most effective methods generally involve the development of some form of process model, then choosing P, I, and D based on the dynamic model parameters. Manual tuning methods can be relatively time-consuming, particularly for systems with long loop times.

4.4.1.8 Limitations

PID controllers, when used alone, can give poor performance when the PID loop gains must be reduced so that the control system does not overshoot, oscillate or [hunt](#) about the control setpoint value.

They also have difficulties in the presence of non-linearities, may trade-off regulation versus response time, do not react to changing process behavior (say, the process changes after it has warmed up), and have lag in responding to large disturbances.

There is also a problem with the derivative term because it amplifies higher frequency measurement or process noise that can cause large amounts of change in the output. It is often helpful to filter the measurements with a low-pass filter in order to remove higher-frequency noise components (but the amount of filtering is limited).

4.5 KALMAN FILTER

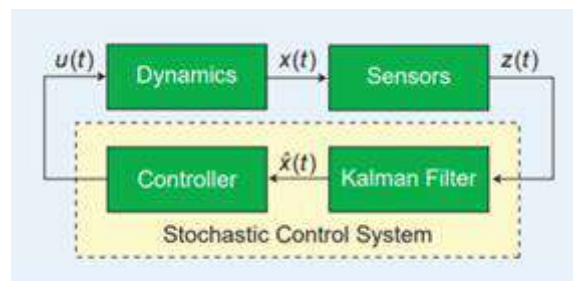


Figure 4.22. Kalman filter scheme

The Kalman filter, invented in 1960 by Rudolf E. Kalman, is a recursive algorithm that provides an optimal estimate of the state vector for a linear dynamic and stochastic system, starting from a series of measurements perturbed by noise and observed over time. It is also known as linear quadratic estimation (LQE). The KF has numerous applications as guidance, navigation, and control of vehicles and can be used in trajectory optimization. The first implementation of the Kalman filter was at NASA's Ames Research Center where it was applied to the non-linear problem of trajectory estimation for the Apollo program.

4.5.1.1 Kalman filter method

Kalman filter is used to estimate states based on linear dynamical system. The process model defines the evolution of the state from time k-1 to time k:

$$x_k = Fx_{k-1} + Bu_{k-1} + w_{k-1}$$

where:

- x_k is the state vector at the time k
- F is the translation matrix applied to the previous state vector x_{k-1}
- B is the control input matrix applied to the control vector u_{k-1}
- w_{k-1} is the process noise vector assumed to be zero-mean Gaussian with the covariance Q.
 $w_{k-1} \sim N(0, Q)$

This model is paired with the measurement model that defines relationship between the state and the current time step k :

$$z_k = Hx_k + v_k$$

where:

- z_k measurement vector
- H measurement matrix
- v_k is the measurement noise vector assumed to be zero-mean Gaussian with the covariance R . $v_k \sim N(0, R)$

The assumption upon which the Kalman filter is based, is that w_k and v_k are uncorrelated noises (covariance = 0), white and zero-mean Gaussian. The role of the Kalman filter is to provide estimate of the state vector x_k at time k , with given initial estimate x_0 , the measurement vector z_k and the information F , B , H , Q and R , these matrices are assumed invariant over time, as in most applications.

The covariance matrices are supposed to reflect the statistics of the noises, but the true statistics of the noises is not known or is not Gaussian in many practical applications. Therefore, the Q and R are usually used as tuning parameters to get specifically performance.

4.5.1.2 Algorithm

The Kalman filter algorithm consists of two stages: “prediction” and “update”, sometimes called “propagation” and “correction”

Prediction:

Predicted state estimate	$\hat{x}_k^- = F\hat{x}_{k-1}^+ + Bu_{k-1}$
Predicted error covariance	$P_k^- = FP_{k-1}^+F^T + Q$

Update:

Measurement residual	$\tilde{y}_k = z_k - H\hat{x}_k^-$
Kalman gain	$K_k = P_k^-H^T(R + HP_k^-H^T)^{-1}$
Updated state estimate	$\hat{x}_k^+ = \hat{x}_k^- + K_k\tilde{y}_k$
Updated error covariance	$P_k^+ = (I - K_kH)P_k^-$

- The hat operator “ $\hat{\cdot}$ ” means an estimate of a variable, $\hat{x}$ is an estimate of x .
- - and + means predicted (prior) and updated (posterior) estimates, respectively.

The predicted state estimate is advanced from the updated previous updated state estimate. P is the state error covariance. It encrypts the error covariance that the Kalman filter thinks the estimate error has. The covariance is defined also as:

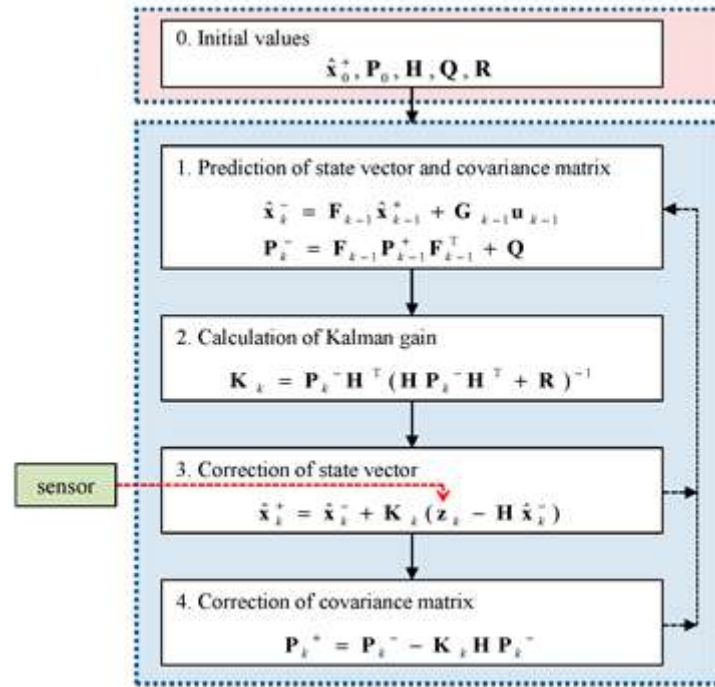
$$cov(x) = E[(x - \hat{x})(x - \hat{x})^T]$$

E is expected (mean) value of its argument. The error covariance becomes larger at the prediction stage, clearly, due the summation with Q , that means the filter is more uncertain of the state estimate after prediction step. In the update stage, the measurement residual is computed first, this is the

difference between the true measurement, z_k , and the estimated measurement, $H\hat{x}_k^-$. Then the residual is multiplied by the Kalman gain K_k , to provide the correction to the predicted estimate $\hat{x}_k^-$. After obtained the updated state estimate, the Kalman filter calculates the updated error covariance, P_k^+ .

As initial values, we need the initial guess of state estimate, $\hat{x}_0^+$, and the initial guess of the error covariance matrix, P_0^+ . Q and R , $\hat{x}_0^+$ and P_0^+ play an important role to obtain a desired performance.

Finally, Kalman filter can be obtained by implementing the prediction and update stages for each time step, $k=1, 2, 3, \dots$, after the initialization of estimate.



4.5.2 LQR Controller

The theory of optimal control is concerned with operating a dynamic system at minimum cost. The case where the system dynamics are described by a set of linear differential equations and the cost is described by a quadratic function is called the LQ problem.

One of the main results in the theory is that the solution is provided by the linear–quadratic regulator (LQR), a feedback controller.

The quadratic linear regulator (LQR), within the framework of excellent control, and more generally of automatic controls and time-invariant linear dynamical systems, is a dynamic compensator obtained by minimizing a cost index $J(x,u)$, function of state $x(t)$ and control $u(t)$.

We set a particular performance index (cost function) which governs the performance of the closed-loop system

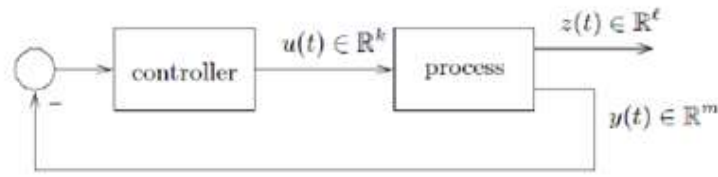
$$J_\infty = \int_0^\infty x^T(\tau) Q x(\tau) + u^T(\tau) R u(\tau) d\tau$$

The cost has two quadratic terms involving two weighting matrices

- $Q = Q^T \geq 0$ penalizes the states (stability)

- $R = R^T \geq 0$ penalizes the control input.

The matrix Q can be written as $Q = C_z^T C_z x = z^T z$, where $z = C_z x$ can be viewed as a controller output.



Find the control input that minimizes the following function:

$$J_\infty = \int_0^\infty \|z(\tau)\|^2 + \rho \|u(\tau)\|^2 d\tau \rightarrow J_\infty = \int_0^\infty x^T(\tau) Q x(\tau) + u^T(\tau) R u(\tau) d\tau$$

4.5.2.1 Theorem: existence solution

For each positive semidefined Q matrix and for any positive definite R matrix there is always a solution u_{ott} of the optimal LQR control issue that minimizes the cost index $J(x,u)$.

4.5.2.2 Theorem: existence of stabilizing solution

If the LTI invariant time linear system is stabilizable and detectable, then minimizing the cost index (making it limited) also stabilizes the system.

The control obtained $u_{ott}(t)$ is a linear function of the state and some matrices including $P(t)$ the solution of the Riccati differential equation (DRE) in the event that the control is finite-time, or P (constant) solution of the Riccati algebraic equation (ARE) if the control is infinite-time.

4.5.2.3 Finite-time

- Control $u_{ott}(t) = -K_{ott}(t)x(t)$
- State feedback controller: $K_{ott}(t) = R^{-1}B^T P(t)$
- DRE whose solution provides $P(t)$:

$$-\frac{dP(t)}{dt} = A^T P(t) + P(t)A + Q - P(t)BR^{-1}B^T P(t)$$

4.5.2.4 Infinite-time

- Control $u_{ott}(t) = -K_{ott}(t)x(t)$
- State feedback controller: $K_{ott}(t) = R^{-1}B^T P$
- ARE whose solution provides: $[0] = A^T P + PA + Q - PBR^{-1}B^T P$

Essentially checking a finite or infinite interval means only pointing to infinity ($tf \rightarrow \inf$) the upper end of the integral that defines $J(x,u)$. The effect of an infinite-time control is a stationary (time-independent) controller, that is, a constant and optimal matrix K_{ott} with respect to the index that you wanted to minimize.

4.5.2.5 Conclusion

The settings of a (regulating) controller governing either a machine or process (like a telescope) are found by using a mathematical algorithm that minimizes a cost function with weighting factors supplied by a human (engineer). The cost function is often defined as a sum of the deviations of key measurements, like position or attitude, from their desired values. The algorithm thus finds those controller settings that minimize undesired deviations. The magnitude of the control action itself may also be included in the cost function.

The engineer needs to specify the cost function parameters, and compare the results with the specified design goals. Often this means that controller construction will be an iterative process in which the engineer judges the "optimal" controllers produced through simulation and then adjusts the parameters to produce a controller more consistent with design goals.

The LQR algorithm is essentially an automated way of finding an appropriate state-feedback controller.

Advantages: provides the best possible performance with respect to some given measure of performance, robust with respect to small uncertainties

Drawbacks: it is difficult to synthesize a unique controller for the whole flight envelope.

4.5.3 Sliding Mode

Sliding mode control (SMC) is a well-established method for control of nonlinear systems. It's a variable structure control method with the interesting feature of the robustness versus imprecise knowledge of the plant to control. Consider a generic nonlinear dynamical system described by its state equation:

$$\dot{x}(t) = f(x, t) + B(x, t)u(t), \quad x \in R^n, B \in R^{n \times m}$$

So which is function of $x(t)$ and $u(t)$. Where:

- $x(t) \in R^n$ is the system state.
- $u(t) \in R^m$ is the system input (control input).

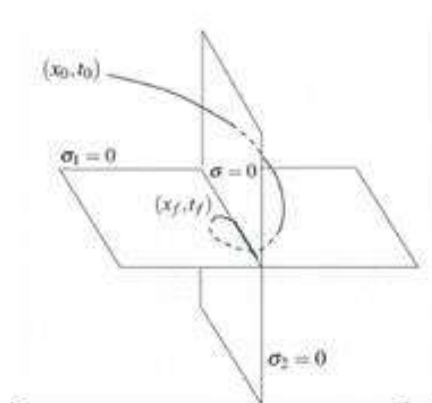
Consider a function of the system state:

$$\sigma(x) = [\sigma_1(x), \dots, \sigma_m(x)]^T$$

$\sigma(x(t)) \in R^m$ which is the sliding variable (switching function) that represents a kind of distance that the states X are away from a sliding surface.

The associated manifold (sliding manifold) that is verified when the state is on the sliding surface: $\sigma(x(t)) = 0$

When the state x is outside of the sliding surface: $\sigma(x) \neq 0$.



We need to fix the sliding manifold and the control law.

The sliding manifold is a subspace of the system state space having dimension $n - m$, which can be a single surface or be given by the intersection of several surfaces. The sliding surface is a subset of the

state space, on which the trajectory of the plant is desired to lie. The sliding mode is enforced when the state trajectory continuously crosses the sliding manifold, so it defines the desired system's dynamics and is designed so that the system in sliding mode evolves in the desired way (e.g. it results in being linearized and its state is asymptotically regulated to zero, or it satisfies some optimality requirement, etc.).

The control law has to be chosen in order to enforce a sliding mode in a finite time from any initial condition and must keep the trajectories sliding on the surface until we reach the goal. The feedback law is designed to bring the plant trajectory towards the sliding surface and, once there, to stay close to this surface. This is not a **continuous function** of time. Instead, it can switch from one continuous structure to another based on the current position in the state space.

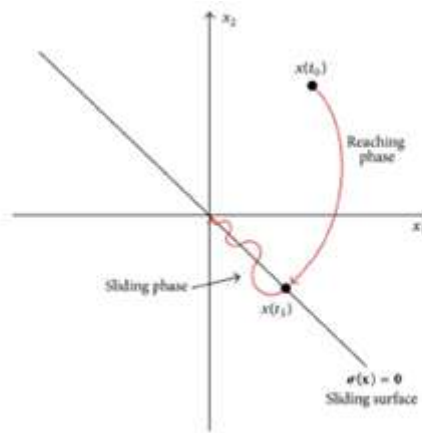


Figure 4.23. Sliding Mode control concept

The method is based on three theorems:

4.5.3.1 The theorem of the existence of the sliding mode

Considering a Lyapunov function candidate: $V(\sigma(x)) = \frac{1}{2} \sigma^T(x) \sigma(x) = \frac{1}{2} \|\sigma(x)\|_2^2$

Where:

- $\|\cdot\|$ is the Euclidean norm.
- $\|\sigma(x)\|_2$ is the distance away from the sliding manifold.

A sufficient condition for the sliding mode is:

$$\underbrace{\frac{\partial V}{\partial \sigma} \frac{d\sigma}{dt}}_{\frac{dV}{dt}} < 0 \quad (i.e., \frac{dV}{dt} < 0)$$

In a neighbourhood of the surface.

4.5.3.2 The theorem of the region of attraction

The subspace for which $\{x \in \mathbb{R}^n: \sigma(x) = 0\}$

Is given by $\{x \in \mathbb{R}^n: \sigma^T(x) \dot{\sigma}(x) < 0\}$

4.5.3.3 The theorem of the sliding motion

Considering a non-singular $\frac{\partial \sigma}{\partial x} B$. Along sliding mode trajectories $\sigma(x)$ is constant and so sliding mode trajectories are described by the differential equation: $\dot{\sigma}(x) = 0$

For the equivalent control $u(x)$. So u has to be:

$$\frac{\partial \sigma}{\partial x} \overbrace{(f(x, t) + B(x, t)u)}^{\dot{x}} = 0$$

$$u = - \left(\frac{\partial \sigma}{\partial x} B(x, t) \right)^{-1} \frac{\partial \sigma}{\partial x} f(x, t)$$

For example, we can choose u :

$$u(x, t) = \begin{cases} u^+(x), & \text{for } \sigma > 0 \\ u^-(x), & \text{for } \sigma < 0 \end{cases}$$

But this sign function has a discontinuity in 0. The repeated discontinuities in the control effort generate the so-called "chattering" that is a zigzag behaviour of the controller output which affects the state of the system. To overcome these problems, the hyperbolic tangent function is used instead of the sign function

4.6 APPLICATIONS OF TELESCOPES CONTROL SYSTEMS

4.6.1 The Spitzer Space Telescope's control system

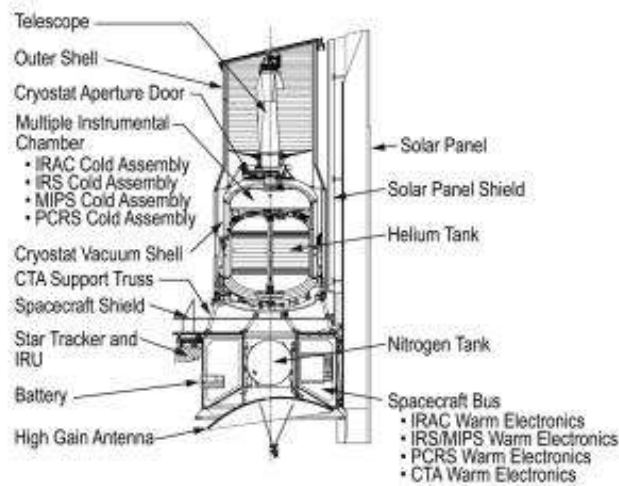


Figure 4.24. Spitzer space telescope

The Spitzer space telescope combines the sensitivity of a cryogenic telescope in space with the imaging capability of three infrared sensors: Infrared Array Camera IRAC, Infrared Spectrometer IRS and Multiband Imaging Photometer for Spitzer MIPS.

Due to the control and maintenance requirements of pointing, *beryllium* has been used to produce some of the components of the telescope such as the primary and secondary mirrors and the metering tower that connects them.

The reason is that this material minimizes the stresses generated from the process of cooling down to the temperature of 5 K, which needed to get clearer images from this kind of infrared camera.

The Secondary Mirror is the only one that can move due to a *cryogenic focus mechanism* which provides high-precision motion in the axial direction and so it is used in orbit to focus the telescope with the help of Pointing Calibration Reference Sensor PCRS and other elements such as:

- An appropriate design of the three cameras and the PCRS so that, if one of the cameras is focused, the others are too. This allows to get appreciable images from the space.
- Software programs that with the help of the Secondary Mirror allow the measurement of the direction and the depth of pointing. There are two types of software:
 - 1) *Simfit*, which relate the measured images to a catalog of predicted images for the focal position of all four of the IRAC modules;
 - 2) *Focus Diversity*, a method that uses the noise pixel statistic (developed by Wright) to determine the image quality, which can be related to the amount of time required for observation to a fixed sensitivity level.

4.6.2 Chandra's control system

The Chandra X-ray Observatory (CXO), previously known as the Advanced X-ray Astrophysics Facility (AXAF), is a Flagship-class space launched aboard the Space Shuttle Columbia STS-93 by NASA on July 23, 1999.

The system of sensors and control hardware that is used to point the observatory, maintain stability, and provide data for determining where the observatory had been pointing is called the *Pointing Control and Aspect Determination (PCAD) system*.

Chandra pointing requirements are not very stringent because Chandra detectors are essentially single-photon counters and therefore an accurate post-factor history of the spacecraft pointing direction is enough to reconstruct the X-ray image.

The main components of the PCAD system are:

- *Aspect camera assembly (ACA)*: 11.2 cm optical telescope, stray light shade, two CCD detectors (primary and redundant), and two sets of electronics;
- *Inertial reference units (IRU)*: two IRUs, each containing two 2-axis gyroscopes.
- *Fiducial light assemblies (FLA)*: LEDs mounted near each X-ray detector which are imaged in the ACA via the Fiducial Transfer System.
- *Fiducial transfer system (FTS)*: directs light from the fiducial lights to the ACA, via a retroreflector collimator (RRC) mounted at the X-ray telescope center, and a periscope.
- *Coarse sun sensor (CSS)*: provides all-sky coverage of the sun.
- *Fine sun sensor (FSS)*: has a 50 ° FOV and 0.02 ° accuracy.
- *Earth sensor assembly (ESA)*: conical scanning sensor, used during the orbital insertion phase of the mission.
- *Reaction wheel assembly (RWA)*: six momentum wheels which change spacecraft attitude.
- *Momentum unloading propulsion system (MUPS)*: liquid fuel thrusters which allow RWA momentum unloading.
- *Reaction control system (RCS)*: thrusters which change spacecraft attitude.

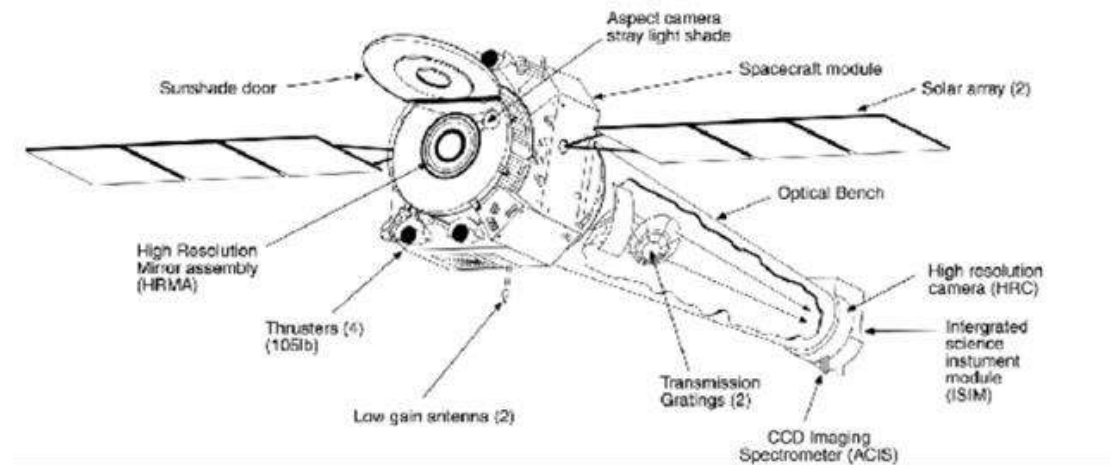


Figure 4.25. Chandra telescope assembly

The *aspect camera assembly (ACA)* includes a sunshade (~2.5 m long, ~55cm in diameter), a 11.2 cm, F/9 Ritchey-Chretien optical telescope, and CCD detector.

This assembly and its related components are mounted on the side of the X-ray telescope.

The camera's field of view is $1.4^\circ \times 1.4^\circ$ and the sunshade is designed to protect the instrument from the light from the Sun, Earth and Moon. Having either the Moon or the Earth in the field-of-view only saturates the detector output without incurring damage and therefore only limits the aspects camera's utility.

The ACA electronics track a small pixel region (either 4×4 , 6×6 , or 8×8 pixels) around the fiducial light and star images. There is a total of eight regions available for tracking. Typically, five guide stars and three fiducial lights are tracked, the average background is subtracted on-board, and image centroids are calculated by a weighted-mean algorithm. The image centroids and fluxes are used on-board by the PCAD and are also telemetered to the ground along with the raw pixel data.

Surrounding each of the focal-plane detectors there is a set of light emitting diodes, or "fiducial lights", which serve to register the detector focal plane laterally with respect to the ACA boresight.

Each fiducial light produces a collimated beam at 635 nm which is imaged onto the ACA via a collimating lens, corner-cube retroreflector and periscope.

Two inertial reference units (IRU) are in the front of the observatory on the side of the X-ray telescope, each IRU contains two gyroscopes, each of which measures an angular rate about two axes. This gives a total of eight channels; data from four of the eight channels can be read out at one time. The gyros are arranged within the IRUs, and the IRUs are oriented such that the eight axes are in different directions and no three axes lie in the same plane. The gyros output pulses represent incremental rotation angles.

Control of the spacecraft momentum is required both for maneuvers and to maintain stable attitude during observations. Momentum control is primarily accomplished using *six Teldix RDR-68 reaction wheel units* mounted in a pyramidal configuration. Momentum is shed from the reaction wheels by firing small thrusters in the MUPS and simultaneously spinning down the reaction wheels.

The aspect system serves two primary purposes: on-board spacecraft pointing control and aspect determination and post-facto ground aspect determination, used in X-ray image reconstruction and celestial location.

The PCAD system has nine operational modes (six normal and three safe modes) which use different combinations of sensor inputs and control mechanisms to control the spacecraft and ensure its safety. In the normal pointing mode, the PCAD system uses sensor data from the ACA and IRUs, and control torques from the RWAs, to keep the X-ray target well within $\sim 30^\circ$ of the desired location. This is done by smoothing (filtering) the data that have been taken during the preceding time intervals using aspect camera star centroids (typically five) and angular displacement data from two of the 2-axis gyroscopes.

On short time scales ($\sim$ seconds) the spacecraft motion solution is dominated by the gyroscope data, while on longer timescales it is the star centroids that dominate.

The important PCAD system performance parameters and a comparison to the original requirements are shown in the next Table. In each case the actual performance far exceeds the requirements.

TABLE I
Aspect system requirements and performance

Description	Requirement	Actual
Celestial location	1.0" (RMS radius)	0.6"
Image reconstruction	0.5" (RMS diameter)	0.3"
Absolute celestial pointing	30.0" (99.0%, radial)	3.0"
PCAD 10 s pointing stability	0.25" (RMS, 1-axis)	0.043" (pitch) 0.030" (yaw)

4.6.3 CHEOPS's control system

CHEOPS is the first mission dedicated to search for transit of exoplanets using ultrahigh precision photometry on bright stars already known to host planets.

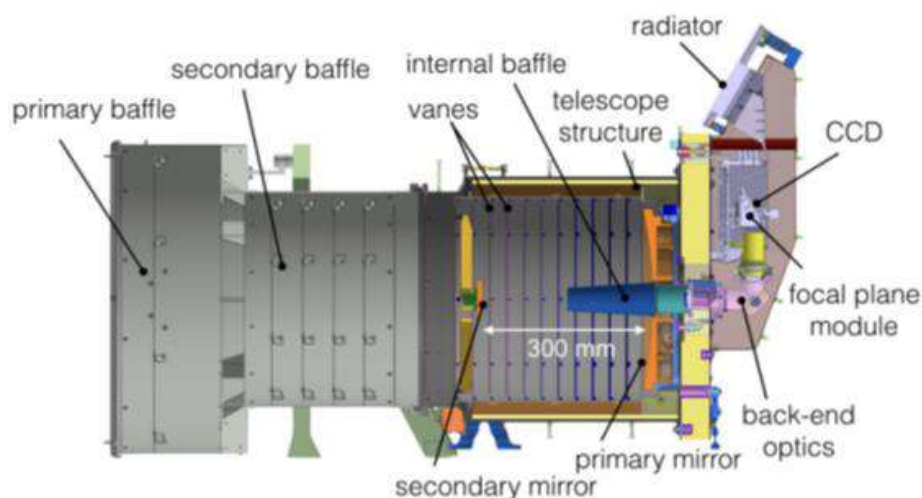


Figure 4.26. Cheops space telescope assembly

Required photometric precision and stability are achieved by using a single, frame-transfer, back-illuminated CCD detector at the focal plane assembly of 33.5 cm diameter, on-axis Ritchey-Chrétien

telescope. This telescope, along with all the detector and support electronics, instrument computer, and thermal regulation hardware, is called CIS. The CIS telescope tube assembly is passively cooled and thermally controlled with on-board heaters. CIS consists of 4 major hardware units: OTA (Optical Telescope Assembly - is the structure carrying the telescope, the Focal Plane assembly Module, FPM, and the radiators. Star trackers are mounted on the isostatic mounts of the OTA), BCA (Baffle and Cover Assembly), BEE (Back End Electronics Box) and SEM (Sensor Electronics Module).

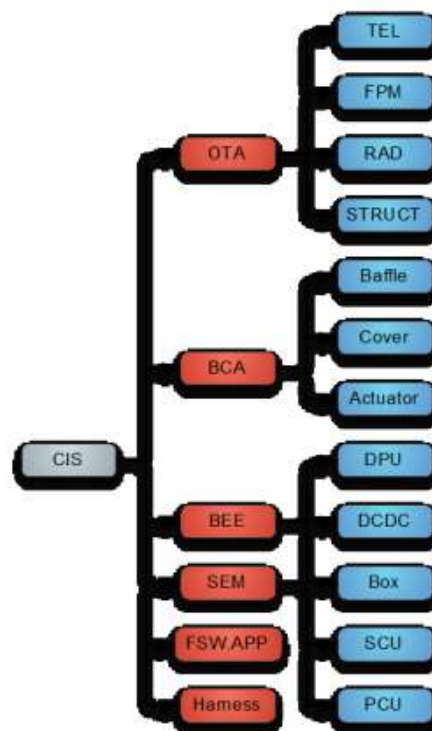
The BEE is the electronics box including the DPU and power converters with aluminium housing, it contains two redundant DCDC converters and two redundant instrument computers (DPU).

The BEE is installed inside the S/C structure. The sensor electronics module contains the digital electronics and interfaces to the DPU. The SEM is instead attached to the S/C panel.

The satellite design is based on the use of the AS-250 platform, an Airbus Defence & Space product line designed for small and medium-size missions operating in LEO.

The key avionic units and On-Board Control Software are inherited from the AS-250 product line. Part of the Attitude and Orbit Control System (AOCS) equipment also belongs to the AS-250 line, with the exception of the reaction wheels (provided by MSC1, Canada), the absence of GPS, and Coarse Sun Sensor, and the down-scaling of the magneto-torquers (Zarm, DE).

The two Hydra star trackers have been provided by Sodern (FR) and have been installed on the instrument optical telescope assembly to minimise misalignment to the instrument line of sight induced by thermo-elastic distortion.



The optical elements are mounted in a structure of carbon-fibre reinforced polymer, which is used to reduce the susceptibility of the instrument to thermal variations, which might be significant in near-Earth orbit.

The target operational temperature is -10°C for the telescope structure.

Straylight (primarily from the Earth) is potentially a major noise source and hence the telescope must be baffled; the instrument baffling takes account of the rotation of the spacecraft.

The temperature stability of the Focal Plane Assembly (FPA) must be maintained to a level of ~ 10 mK and the gain of the readout electronics must be stable at the 5-ppm level.

The FPA and the readout electronics require radiators to meet their operational temperatures and allow for a thermal control approach, which avoids use of energy intensive thermo electric or Sterling coolers.

The radiators will passively cool to temperatures below the target temperature and the FPA will be heated in a control loop to the specified value.

The sensor electronics module (SEM) supports the readout electronics and interfaces to a digital processing unit (DPU), which is part of the back-end electronics assembly (BEE).

The DPU controls all aspects of the instrument and allows image acquisition over a range of framing rates.

The DPU also analyse data in real-time autonomously and pass the centroid of the stellar images to the spacecraft AOCS thereby allowing more accurate control of the spacecraft pointing (requirement of $< 8''$ jitter).

This is needed to reduce the jitter of the system and thereby reduce noise arising from the error in the knowledge of the flatfield.

The BEE also contains a power unit (DCDC), which converts the input voltage from the spacecraft to the voltage supplies necessary for the instrument; the DC-DC converter has dedicated DC-DC converters for the DPU and the SEM.

In addition, switches are provided to control up to four external heaters used to stabilize the telescope. The S/C interface includes an EMC filter, solid state switches for ON/OFF control, and the status monitoring.

The CHEOPS spacecraft includes a compact, mono-propellant Propulsion Module (PM) from Arianespace Group (DE), inherited from the Myriade Evolution design and including a Hydrazine tank with a capacity of 30 litres from Rafael (IS), 4 small 1N thrusters, 1 pressure transducer and 2 pyro-valves (one of them for passivation at end of line).

The PM has been assembled and tested as a separate sub-system before integration on the satellite. Its mechanical interface, directly on the Satellite Interface Ring, has been designed to ensure maximum modes decoupling between the PM and the rest of the S/C.

The propellant tank has been sized for providing a total delta-V in excess of the total of 110 m/s required to perform:

- a) launcher dispersion correction manoeuvre;
- b) collision avoidance manoeuvres;
- c) final de-orbiting at the end of the operational phase.

4.6.4 Cryogenic Nano-Actuator for James Webb Space Telescope

An extremely precise positioning mechanism has been developed in the James Webb Space Telescope project for use in optical positioning of large mirrors: the *cryogenic nano-actuator*. The design

incorporates traditional mechanical components such as gears, bearings and flexures in a configuration covered by two patents. This linear actuator is capable of 10 nanometer position resolution over a range of 20 mm and can operate under *cryogenic conditions*.

The James Webb Space Telescope (JWST) is a large deployable spacecraft that will be launch on October 2021. A key component of JWST is the Optical Telescope Element (OTE), which consists of all the components along the optical path including the Primary and Secondary mirrors. The Primary mirror is about 6.5 meters in diameter and is made up of 18 segments, these mirrors are folded up during launch and deployed in space; once unfolded, the mirrors must be deployed away from the launch restraints and then adjusted very precisely.

Each primary mirror segment and the secondary mirror is supported and controlled by *six linear actuators* to obtain six degrees of positioning control. Each mirror can be positioned in tip, tilt, piston, horizontal & vertical decentering and clocking. Two actuators are assembled into a bipod assembly and the final hexapod configuration is made up of three bipods. These 18 mirrors are divided into 3 segments and a central set of actuators is used to adjust the radius of curvature of each segment according to the required specifications and the information of the feedback sensors.

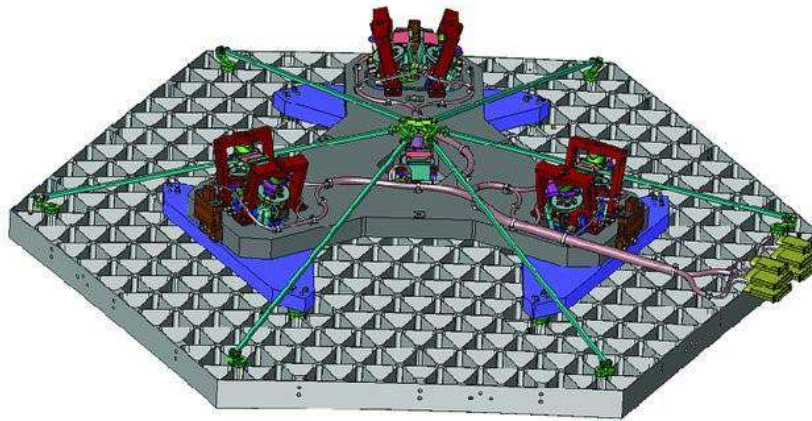


Figure 4.27. JSWT primary mirror actuation assembly

The positioning and focusing of the primary and secondary mirrors require a total of 144 actuators.

The Actuator for JWST has two top level requirements applied to the actuators supporting the 18 segments of the primary mirror as well as to the actuators that support the secondary mirror:

- Positioning of the mirror segments;
- Support the mirror segments during ground test and launch.

The first requirement of positioning was used to generate the following derived requirements:

1. Move the mirror from the stowed position to the nominal deployed position.
2. Move the mirror from the nominal deployed position with 6 degrees of motion.
3. Position each segment to nanometer resolution.
4. Support the segments in a hexapod configuration.
5. Operate at cryogenic & ambient conditions

6. Operate over the life of the mission.

The second requirement of support was used to generate the following derived requirements:

1. Support the mirror segments during launch.
2. Support the mirror segments during ground optical testing.
3. Support the mirror segments during ground transportation.
4. Hold the mirror segments in place with power off.
5. Support the segments in a hexapod configuration.
6. Operate at cryogenic & ambient conditions

A summary of the requirements derived are presented in the table that follows:

Property	Requirement	Capability	Compliance
Fine range	>7 micrometers	10 micrometers	yes
Fine step size	<10 nanometer	7.7 nanometer	yes
Fine repeatability	<3 nanometers	2 nanometers	yes
Coarse range	>20 mm	21 mm	yes
Coarse step size	<1.0 microns	0.058 microns	yes
Axial load	1890 N (425 lb)	2650 N (595 lb)	yes
Axial stiffness	24,500 N/mm (140,000 lb/in)	25,200 N/mm (144,000 lb/in)	yes
Axial holding	380 N (85 lb)	890 N (200 lb)	yes
Position feedback	20 microns	12 microns	yes
Cryogenic operation	30 K	20 K	yes
Nominal length	138.8 mm (5.5 in)	138.8 mm (5.5 in)	yes
Mass	700 gram (1.55 lb)	665 gram (1.47 lb)	yes

The success of this actuator is primarily due to two important inventions: the *fine stage flexure* and the *coarse drive coupling*, these two inventions enable the fine adjustment capability over the long range of motion required for the actuator.

NASA engineers use light waves to align the mirror segments to each other. Seven actuators attached to the back of each one of the mirror segments are used to adjust the positions and shapes of the mirror segments to achieve precise alignment. The actuators on each mirror segment are capable of extremely minute movements, which allow engineers to align the entire primary mirror by finely adjusting each mirror segment. They can move in steps that are a fraction of a wavelength of light (about 1/10,000th the diameter of a human hair). These actuators can also be used to precisely reshape each mirror segment to ensure they all match up once aligned and all at their operational temperature of about 40 K (or about -388 °F/-233 °C).

The primary mirror segment assemblies (PMSAs), mounted to the primary mirror backplane support structure, a graphite/epoxy composite truss structure, also have actuated radius of curvature (ROC) adjustment capability.

A wavefront sensing and control algorithm aligns the telescope mirrors after deployment and maintains alignment over the life of the telescope. For the eighteen primary mirror segments to act as a single mirror the reflected wavefront from each segment must be phased relative to the other seventeen segments. During this phasing process the radius of curvature of each segment is actuated

to match those of the other segments to within a budgeted amount. Though the design of the PMSA RoC actuation system has been optimized, the mirror curvature change is accompanied by a small figure deformation, known as RoC actuation residual. There is approximately 24 nm rms of figure change per millimeter of radius of curvature actuation. Since the actuation residual degrades the telescope overall performance, the amount of RoC actuation in-flight must be limited. Therefore, the mirror's radius of curvature is manufactured to a tight tolerance to minimize the need for RoC actuation.

4.6.5 ASTERIA attitude and pointing control systems

ASTERIA (Arcsecond Space Telescope Enabling Research in Astrophysics) is a 6U CubeSat deployed from the ISS on 20 November 2017. The principal goal of the mission was to image and perform photometry on bright, nearby stars and possibly detect transiting exoplanets orbiting these stars.

In order to successfully perform the technology demonstration, the payload must be pointed with a stability of 5 arcseconds RMS over 20-minute observations and a repeatability of 1 arcsecond RMS across multiple observations. To achieve these pointing requirements, *reaction wheels* were used to control attitude of the spacecraft bus while a *piezo stage* translates the focal plane array to control the pointing of the payload.

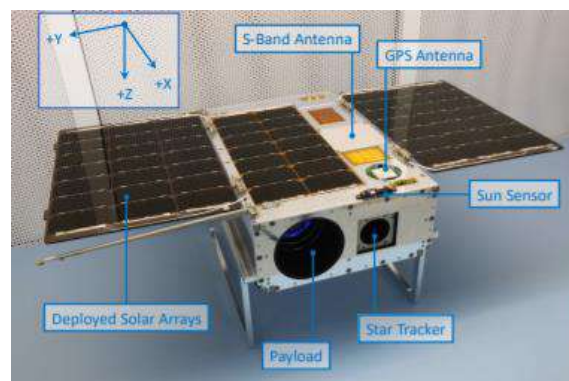


Figure 4.28. ASTERIA with deployed solar arrays

Figure 4.28 shows a picture of the final spacecraft with the solar arrays deployed. Other parts highlighted include the payload, star tracker, sun sensor, GPS antenna, and S-band antenna. The picture also shows the direction of the spacecraft-fixed x-, y-, and z-axes; the x-axis is aligned with the nominal payload boresight, the z-axis is antiparallel with the nominal solar panel normal vector and sun sensor boresight, and the y-axis completes a dextral triad.

4.6.5.1 Attitude Control Subsystem Hardware & Software

The attitude control subsystem (ACS) used on ASTERIA is the Blue Canyon Technologies (BCT) flexible Attitude Control Technology (XACT), shown in Figure 4.29. This is a fully integrated subsystem with both hardware and software necessary to control the spacecraft's attitude. The sensors include a star tracker, inertial measurement unit, sun sensor and magnetometer. The sun sensor, consisting of four diodes, is mounted on the z- face of the spacecraft and is aligned with the nominal solar array normal vector, this allows the XACT to point the solar arrays to the sun but does not provide full-sky coverage. The actuators include three reaction wheels for attitude control and three torque rods for momentum control.

A GPS unit was also added to the unit to assist with providing accurate time, position, and velocity information to the XACT and spacecraft. The software on the XACT includes algorithms for star tracking, attitude estimation, attitude control, momentum control, and orbit propagation.

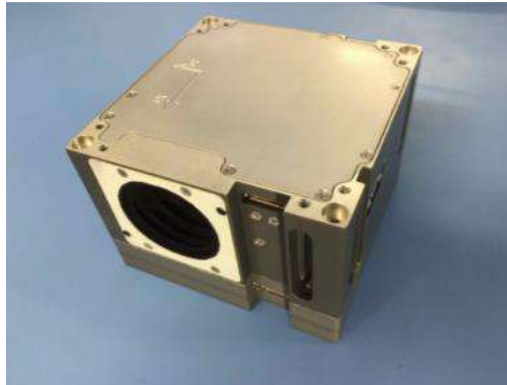


Figure 4.29. ASTERIA's Flight XACT unit

The primary commands used by ASTERIA to control the XACT include: setting the time, setting the position and velocity, pointing to the Sun, pointing to an inertial attitude, pointing to a specified target, setting the momentum bias and setting the attitude control gains.

4.6.5.2 Pointing Control Subsystem Hardware & Software

During science observations, while the ACS provides attitude control with reaction wheels, the pointing of the payload is further improved with the *pointing control subsystem (PCS)*. This consists of three main pieces of hardware: a lens assembly, a piezo stage, and an imager, which are all components of the payload.

The lens assembly was nominally designed to have a focal length of 85 millimetres and a f-number of 1.4. The two-axis piezoelectric nanopositioning stage is mounted behind the lens assembly and provides the ability to translate the imager in the plane orthogonal to the payload boresight, providing a tip/tilt correction to stabilize the image of the star field being observed. The electronics contains a digital-to-analog converter to command the stage, an analog-to-digital converter to read the position of the stage, two high-voltage amplifiers to drive each axis, a strain gauge feedback loop to remove the hysteresis and a notch filter to avoid exciting the resonant frequency of the moving mass of the stage. Mounted to this piezo stage is a Fairchild Imaging CIS2521F0111, this is a frontside-illuminated, monochrome CMOS image sensor with 2592 by 2192 pixels and a 6.5-micrometer pixel pitch.

With the lens, the imager has a field of view of 11.2 by 9.6 degrees and each pixel is 15.8 arcseconds on a side. A picture of the piezo stage is shown in Fig. 4.30.

The software that performs the fine pointing control is depicted in the block diagram shown in Figure 31. The software is made up of four algorithms: centroiding, target star centring, pointing control, and piezo & roll offload. These algorithms all run at 20 Hz.

The centroiding algorithm receives up to eight 64-by-64-pixel windows from the imager, calculates the centroid of the star in each of these windows, and reports the validity of these centroids.

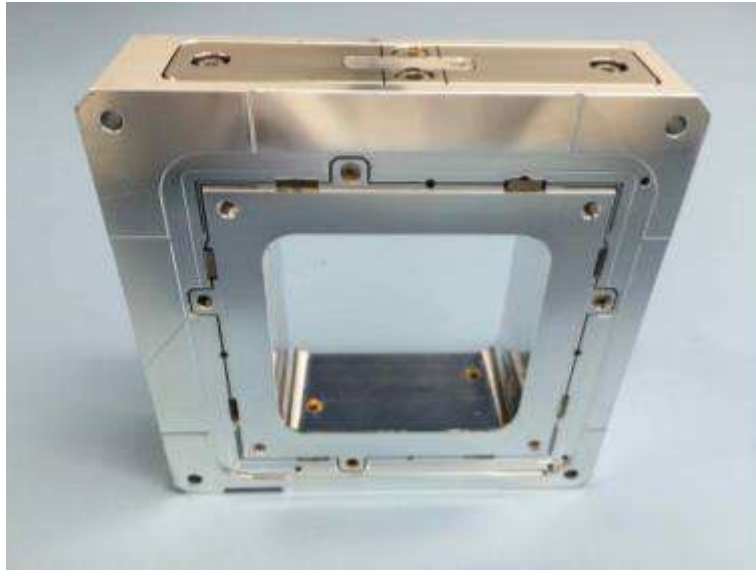


Figure 4.30. ASTERIA's piezo stage

They are then fed into the three other pointing control algorithms, the first of which is the target star centering algorithm. The purpose of this algorithm is to compute an offset that, when used by the pointing control algorithm, centers the target star. This target star offset, along with the measured centroids of all the guide stars are fed into the pointing control algorithm. This algorithm first computes the weighted average of the error of the guide stars; this combined guide star error is summed with the target star offset, forming the control error. This command is then sent to the piezo stage to stabilize the image.

Finally, the piezo stage and roll offload algorithm is used to command the ACS to roughly center the piezo stage, keeping it away from the saturation limits, as well as roughly zero the roll-about-boresight angle observed by the payload. This is done by constructing an attitude quaternion command based on a tip and tilt command offset and a roll command offset. The tip and tilt command offsets are computed by taking the error between the measured and centered piezo stage position, integrating it, and multiplying it by a gain.

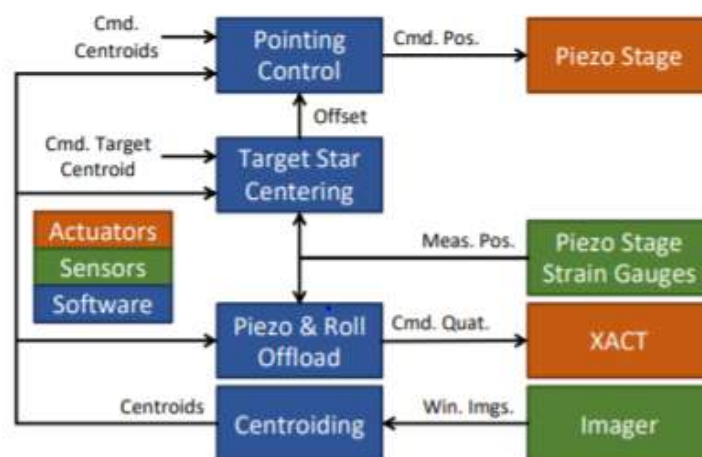


Figure 4.31. Block diagram of Pointing control software

The roll command is computed from the roll error, which can be computed from the relative motion of the guide stars spread across the imager. The gains used for computing the tip, tilt, and roll

commands determine how quickly the piezo stage and roll error are offloaded to the attitude controller. While the commanded quaternion is computed at 20 Hz, the commands to the XACT are down sampled to 5 Hz.

4.6.6 GAIA's control system

GAIA, for Global Astrometric Interferometer for Astrophysics. Gaia is the space-astrometry mission of the European Space Agency. The primary science goal of Gaia is to examine the kinematical, dynamical, and chemical structure and evolution of our Galaxy.

Gaia consists of two telescopes providing two observing directions with a fixed angle of 106.5° between them. The spacecraft rotates continuously around an axis perpendicular to the two telescopes' lines of sight. The main aim of Gaia is to measure the three-dimensional spatial and the three-dimensional velocity distribution of stars and to determine their astrophysical properties, such as surface gravity and effective temperature, to map and understand the formation, structure and evolution of Milky Way.

4.6.6.1 Mechanical Service Module

4.6.6.1.1 Attitude and orbit control

The extreme centroiding needs of the payload make stringent demands on satellite attitude control. This requires that rate errors and relative-pointing errors be kept at the milli-arcsecond per second and milli-arcsecond level, respectively. These requirements prohibit the use of moving parts, such as conventional reaction wheels.

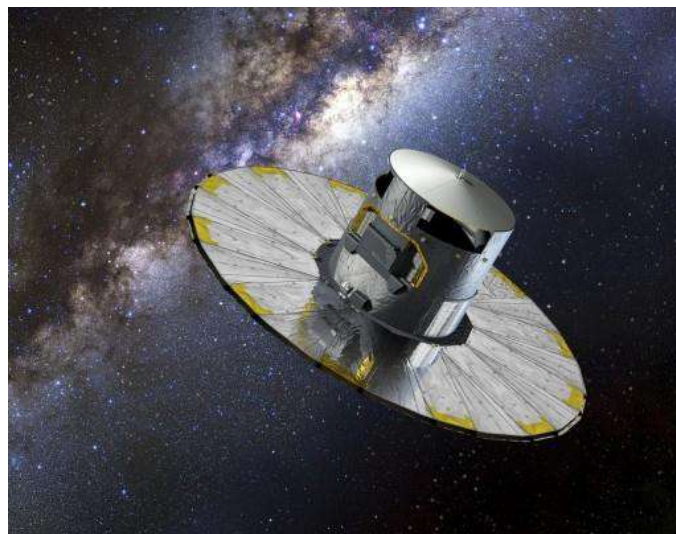


Figure 4.32. GAIA concept

The attitude- and orbit control subsystem (AOCS) is based on sensors and thrusters.

Sensors include:

- two autonomous star trackers;
- three fine Sun sensors;
- three fiber-optic gyroscopes (**without rotating wheels**);
- low-noise rate data that provided by the payload measurements of star transit speeds through the focal plane.

Gaia uses two categories of **actuators**:

Chemical-Propulsion Subsystem (CPS), two sets of eight bi-propellant (NTO oxidizer and MMH fuel) newton-level thrusters forming the for spacecraft manoeuvres and back-up modes, including periodic orbit maintenance. Chemical Propulsion System using a Bipropellant System: two Tanks are filled with a total of ~400kg of propellant and are mounted on the central tube via support struts.

Gaia's Chemical Propulsion System uses Monomethyl hydrazine as fuel and Nitrogen Tetroxide as Oxidizer. The tanks feed two banks of six 10-Newton (1.02kg) thrusters that are mounted in the +/-X axis directions. Four thrusters of each bank are oriented to the -X direction and two are oriented to the +X direction at an angle of 45 degrees to the Service Module structure. The 10-Newton thrusters consisting of a platinum alloy combustion chamber and nozzle that tolerates the operational temperature of 1,500°C. The thruster can be operated in a thrust range of 6 to 12.5 Newtons with a nominal thrust of 10-Newtons which generates a specific impulse of 291 seconds. It operates at an inlet pressure of 10 to 23 bar and a chamber pressure of 9 bar. The 10N thruster has sized for a burn life of 70 Hours with up to one million duty cycles.

Micro-Propulsion Subsystem (MPS), micro-newton-level thrusters forming the for fine attitude control required for nominal science operations. The MPS consists of the Micro Propulsion Feed Module, Micro Thruster Assemblies and Micro Propulsion Electronics. The thruster system uses high-pressure Nitrogen propellant to provide very small impulses with a thrust range of 1 Micronewtons to 500 Micronewton, each tank containing 28.5 Kilograms of N₂, stored at a pressure of 310 bar. The two N₂ tanks supply gas to the propulsion system via a High-pressure Transducer and a High-pressure Latch Valve that pass the gas to two Pressure Regulators that reduce the pressure.

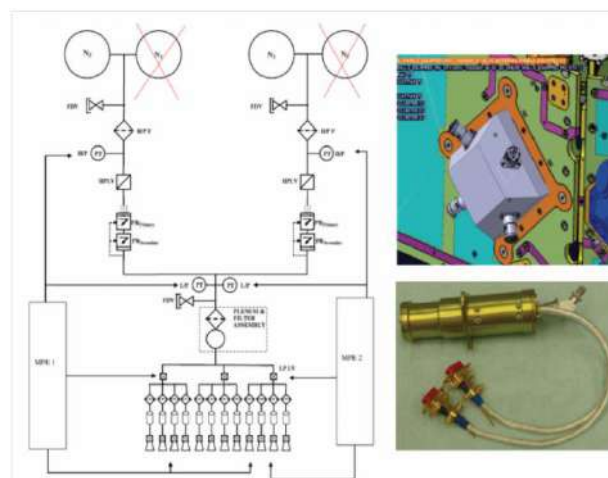


Figure 4.33. Micro-propulsion subsystem

Two Low Pressure Transducers then direct the gas to three Low Pressure Latching Valves that are installed in the Micro Thruster Assemblies – each feeding four thrusters.

The thrusters tolerate more than 500 million on/off cycles and have a lifetime of 20,000 hours at an operational temperature of -20 to +50°C.

4.6.6.2 Thermal Control

4.6.6.2.1 Deployable Sunshield Assembly

The DSA is 10 meters in diameter, it is folded up during launch and is deployed early in the flight. It protects the payload unit from direct sunlight that could compromise instrument accuracy.

The DSA is an umbrella-type structure that consists of Multilayer Insulation as the primary shield material and six rigid deployment booms as well as six secondary stiffeners.

These booms have a single articulation on the base of the Service Module for easy deployment in the radial direction by a spring system.

4.6.6.2.2 Thermal Tent

A large Thermal Tent Structure protects the Payload Module from thermal loads and stray light from the sun or other objects. The large cylindrical structure consists of Carbon Fiber-Reinforced Polymer (CFRP) sandwich panels. It is attached to the top floor of the Service Module.

The Tent is covered with Multi-Layer Insulation and facilitates a thermal radiator to keep the instruments at a stable temperature. It also protects the electronics from radiation and contamination.

Three openings in the tent serve as entrance apertures of the telescopes and the radiator.

4.6.6.3 Material

The use of silicon-carbide (SiC) ultra-stable material for mirrors and telescope structure provides low mass, isotropy, thermo-elastic stability and dimensional stability in a space environment.

This allows to meet the stability requirements for the basic angle between the two telescopes with a passive thermal control instead of an active one.



Figure 4.34. Deployable sunshield assembly

4.6.7 Hubble Space Telescope pointing control system

Hubble uses some very basic physics to turn itself around and look at different parts of the sky. Located on the telescope are six Gyroscopes (which, like a compass, always point in the same direction) and four free-spinning steering devices called reaction wheels.

Five types of sensors make up the *Pointing Control System*: the *Coarse Sun Sensors*, the *Magnetic Sensing System*, the gyroscopes, the *Fixed Head Star Trackers*, and the *Fine Guidance Sensors*.

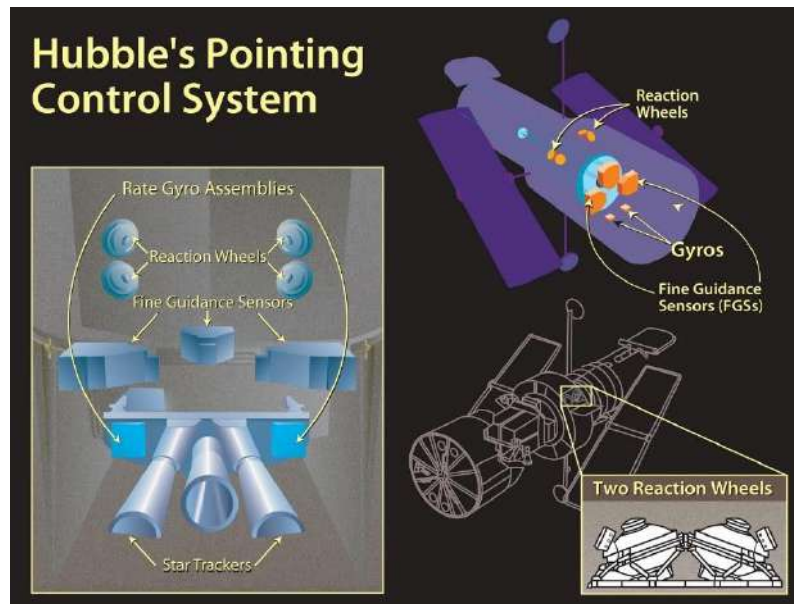


Figure 4.35. Elements of HST pointing control system

All the sensors work in tandem to adjust Hubble's attitude on command.

- The *Coarse Sun Sensors* determine the orientation of Hubble in relation to the Sun. This information is used to help point the telescope and to protect the observatory sensitive instruments and optics from damages. The sensors use silicon diode detectors to determine whether the Sun is present in their field of view and, if so, the angle of the Sun relative to the sensor. Because Hubble was designed to pick up very faint light, the telescope must be pointed at least 50 degrees away from the Sun line of sight.
- The *Magnetic Sensing System* measures the telescope orientation in relation to Earth magnetic field. The system consists of magnetometers and dedicated electronics, which, like a compass, assess the orientation of the telescope.
- HST* uses gyroscopes to detect and measure any rotations so it can stabilize itself in orbit and point accurately and steadily at astronomical targets. Hubble has some of the most accurate and stable gyroscopes ever built. The hair-thin wires that carry electricity in the gyroscopes can degrade over time, however, causing incorrect rate measurements, called biases, and eventual failure. Three gyroscopes are normally required for operations; observations are still possible with two or one, but the area of sky that can be viewed would be somewhat restricted, and observations that require very accurate pointing are more difficult. Since 2018, two gyroscopes are used at a time to extend their operational life.
- The *Fixed Head Star Trackers* determine Hubble's attitude by measuring the locations and brightness of stars in their field of view. This increases the accuracy of the observatory attitude to within 60 arcseconds, and it allows the computer to correct attitude errors following vehicle manoeuvres so that the *Fine Guidance System* can lock onto guide stars.
- The *Fine Guidance System* is Hubble's most accurate pointing sensor, and it is comprised of three Fine Guidance Sensors (FGSs).

The FGSs use starlight captured by the telescope mirrors to find and maintain a lock on guide stars to ensure that the spacecraft attitude does not change.

The level of stability and precision that the FGSs provide, gives Hubble the ability to remain pointed at a target with no more than 0.007 arcsecond of deviation over extended periods of time.

It can also be used as a scientific instrument to determine a star position.

Two actuator systems rotate Hubble: the *Reaction Wheel Assemblies* and the *Magnetic Torquers*. Neither of these systems uses propellants because the by-products of reactions could contaminate the telescope field of view.

- Hubble utilizes four reaction wheels, which are large, massive wheels that spin under the control of Hubble's computer.

If one of the reaction wheels turns clockwise, Hubble will turn counterclockwise.

The combined changes in the spin speeds of the wheels allow Hubble to manoeuvre to point at any location in the sky.

The four reaction wheels are each approximately two feet wide (smaller than a car tire), but in microgravity they can move the heavy Hubble because of their high spin rate and their mass. However, at its fastest, Hubble only rotates approximately 90 degrees in 15 minutes.

- Magnetic Torquers are eight-foot iron rods wrapped in wire that produce a magnetic field when commanded by Hubble's computer.

The magnetic field produced by the magnetic torquers pushes or pulls Hubble toward Earth magnetic field, rotating the telescope.

There are four magnetic torquers installed on Hubble, located at 90-degree intervals on the outside of the spacecraft.

The magnetic torquers are used to help reduce reaction wheel speeds, which can build up because of drag caused by Earth atmosphere.

4.6.8 HALCA Radio Telescope's control system

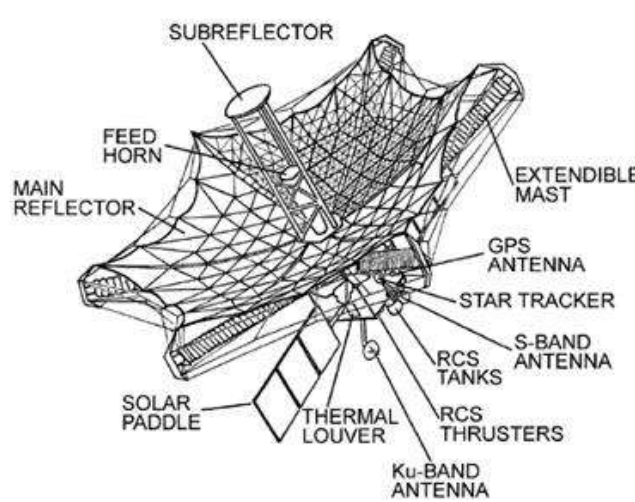


Figure 4.36. HALCA antenna structure

The scientific satellite HALCA (original project name MUSES-B) was developed for the Very Long Baseline Interferometry (VLBI) Space Observatory Programme (VSOP) in the Institute of Space and Astronautical Science (ISAS) and the National Astronomical Observatory (NAO), and launched on February 12, 1997. The satellite performs as a radio telescope, which is designed as a receiver-only system at three observation frequency bands: L-band (1.60–1.73 GHz), C-band (4.7–5.0 GHz) and Ka-band (22.0–22.3 GHz). The deployable antenna on HALCA consists of the main reflector, sub-reflector and feed horn as shown in figure. The antenna was folded to be installed in the rocket fairing and was deployed in orbit after the launch. The structure of the deployable antenna is based on the tensioned cable truss concept. The main reflector is composed of six extendible masts, cables and reflecting mesh.

Cables are tensioned by the masts to form 144 triangular trusses, which approximate a parabolic surface with smaller subdivisions. The antenna has a hexagonal aperture.

4.6.8.1 Large Deployable Antenna

The development of the large deployable antenna was the biggest engineering challenge of the MUSES-B (later HALCA) project. A light weight, high surface-accuracy antenna was developed based on the wire-tension-truss concept.

The main reflector is composed of wire networks, meshes and six extendible masts.

Extension of the six masts deploys the main reflector, and the same masts give tensions to the wire network and meshes such that the reflecting surface forms an accurate parabola.

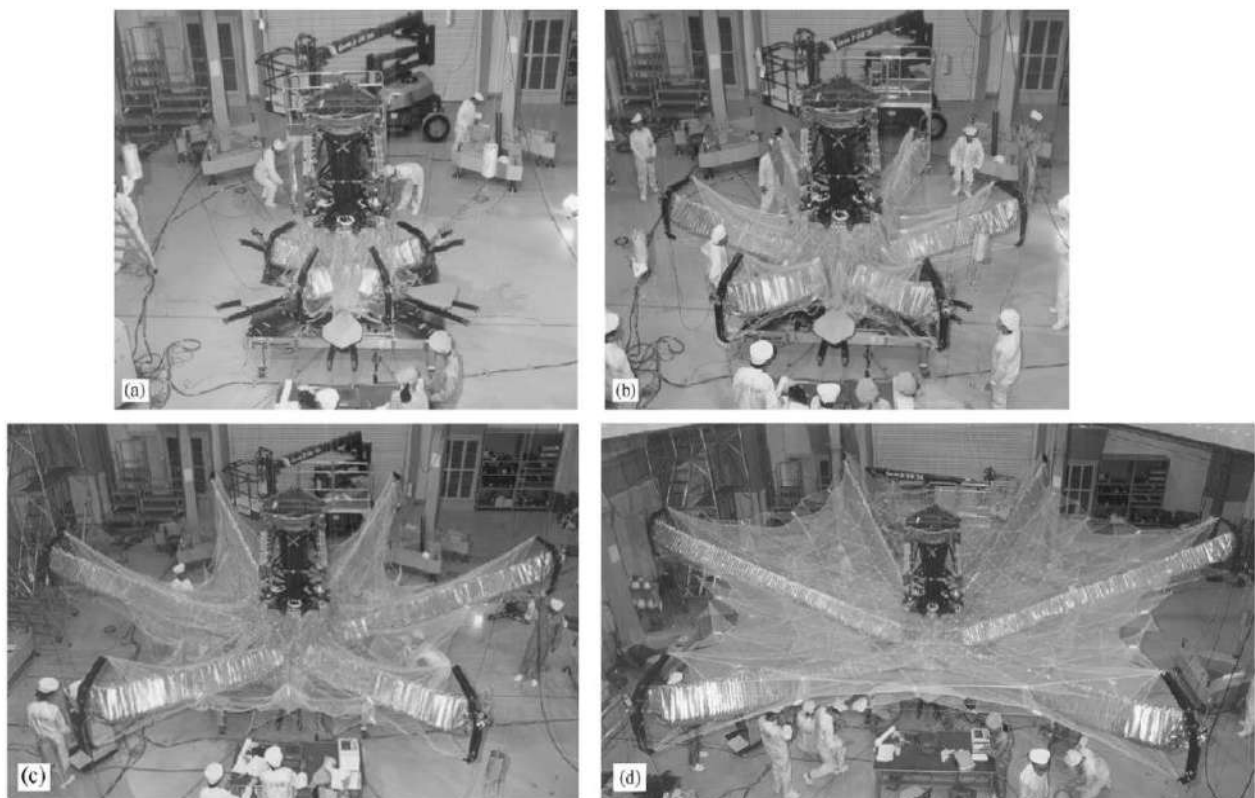


Figure 4.37. Deployment of hexapod antenna of HALCA

The wires are made of Kevlar core with Cornex envelope, and the meshes are made of gold-coated molybdenum wires.

The surface of the main reflector was adjusted on the ground, and the final surface accuracy attained was 0.6 mm rms. The sub-reflector of the antenna is supported by three columns; the columns were folded during launch and extended in orbit.

Deployment of the antenna was conducted over February 24–27, 1997. First the sub-reflector supporting columns were extended successfully. Then, the main reflector was deployed also successfully. On February 26, the deployment started. The most critical operation was the six masts simultaneous extension to their final position, lasting for about 11 min. The deployment was brought to completion in ~21h. The figure on the left shows the angle error of the attitude control of the spacecraft during the main reflector deployment in February 26.

The attitude of the spacecraft was kept stable under the large structural change of the whole spacecraft.

4.6.8.2 Attitude Control

The attitude control system is a zero-momentum, three-axis stabilized one, with four reaction wheels, an inertial reference unit (IRU), two star trackers (STT), magnetic torquers, and other attitude sensors. Pointing of the spacecraft and its large telescope is carried out by reaction wheels, applying torque to the spacecraft to rotate it in the desired direction: all four of them were used, allowing zero-momentum attitude control, until 1999 October, after which the satellite has been operated in a three-reaction-wheel (biased momentum) mode.

Manoeuvres are performed in two stages, with a first slew around the satellite-Sun axis followed by a slew about the solar-paddle axis.

The maximum slew rate is 2.25° per minute.

Mechanical gyroscopes are used to measure the angular motions while slewing.

Spacecraft attitude is detected by the STTs and the IRU, and drifts of the attitude and the IRU are estimated on-board by a Kalman filter: after every manoeuvre, they are used to check pointing and to calibrate the gyroscopes.

The angular momentum accumulated by the reaction wheels is dumped by magnetic torquers during perigee passes. The reaction wheels have a linear rotation range of ± 3500 rpm and 6 Nms angular momentum.

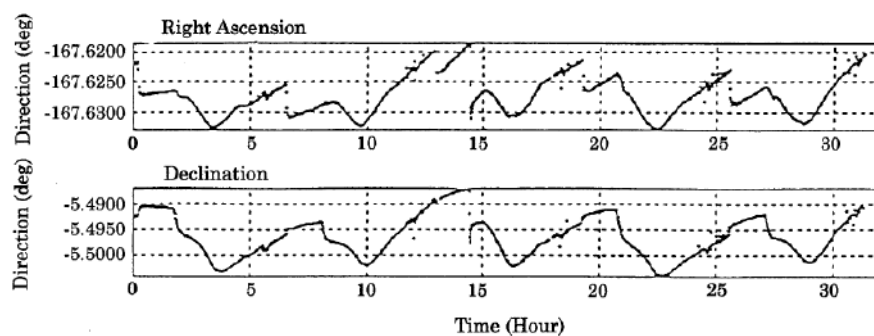


Figure 4.38. Pointing direction variation in a fully deployable test

The pointing accuracy aimed at the system design was 0.01° , the accuracy required in 22 GHz observations for the radio antenna: solar pressure and complex thermal effects are the main disturbances.

As described in the previous section, the *attitude control system* maintained the spacecraft stable during the large antenna deployment. After then, the performance of the *attitude control system* was examined for the fully deployed configuration of the spacecraft and was confirmed to be satisfactory.

The figure above shows an evaluation result: a variation of pointing direction (given in right ascension and declination) for a continuous 32 h when the z-axis of the spacecraft was directed towards a fixed direction. The variation was $\pm 0.00715^\circ$ in right ascension and $\pm 0.00855^\circ$ in declination.

The nearly 6 h period variation seen in figure is due to the STTs operation; the STTs are operated in some finite duration of the orbital period of 6.3 h and in the remaining time, the attitude is estimated by integrating the IRU data.

4.6.9 Keck space telescopes primary mirror

Keck is a ground-based observatory, it consists of two big and twin telescopes, which work with a very sensitive primary mirror system 10 meters wide.

It allows accurate results, but it needs continuous control to check it is working in the right way: it is composed of 36 smaller hexagonal shaped mirrors (called segments) that work as one.

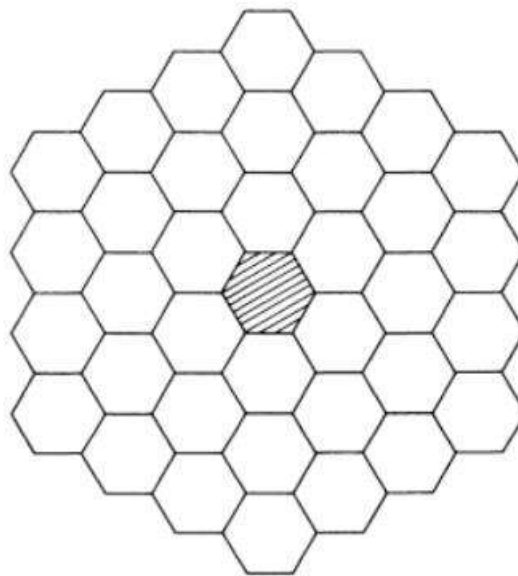


Figure 4.39. Hexagonal segmentation geometry of primary mirror

One of the problems that this kind of systems could present it is the misalignment of the segments (shown in figure) composing the primary mirror, due to gravity. So, it needs a computerized system of sensors and actuators which checks and realigns the position of every segment. The control system checks the inclination of every segment twice per second, and if it finds a wrong positioned segment, repositions it based on the adjacent ones. This is possible thanks to 108 actuators and 168 relative position sensors which form the control system.

The *Active Control System* (ACS) can be divided in three element: the *Drive and Control System* (DCS) that decides when to start and stop the activity of ACS, the *Phasing Camera System* (PCS) that controls

segments position, and the star tracking camera system that collects the images of every segment and measure the overlapping.

Kecks are Ritchey-Chrétien telescopes based on a complex system of lenses, so to prevent blurring of the image, adaptive optics is used: it concerns the use of a deformable mirror, able to change its own shape up to 2000 times per second to reduce atmospheric distortion.

5 REVIEW AND REQUIREMENT ANALYSIS OF SCIENCE AND EARTH OBSERVATION MISSIONS

5.1 INTRODUCTION: TYPICAL SCIENCE AND EARTH OBSERVATION MISSIONS' REQUIREMENTS

In this section, the focus will be on the project requirements. A requirement is a wide concept that could include any desired function, capability or quality of the system under development, so that it acquires a certain value and utility to a customer, organisation or other stakeholder. A set of requirements is used as an input into the design stages. The main reviewed and analysed requirements, in this report, concern the typical Science and Earth Observation missions, in terms of mechanical, thermal, control, optical corrections with respect to different existing missions.

A choice was made to analyse 5 LEO missions after a confrontation with the working team resolved in the decision to design the telescope for a mission operating in such an orbit.

5.2 HUBBLE SPACE TELESCOPE

5.2.1 General overview

The Hubble Space Telescope (HST) is a space telescope that was launched into low Earth orbit (LEO) in 1990 and remains in operation. Over 100,000 observations of more than 20,000 targets have been produced for retrieval. Above the distortion of the atmosphere, far above rain clouds and light pollution, Hubble has an unobstructed view of the universe. Scientists have used Hubble to observe the most distant stars and galaxies as well as the planets in our solar system.

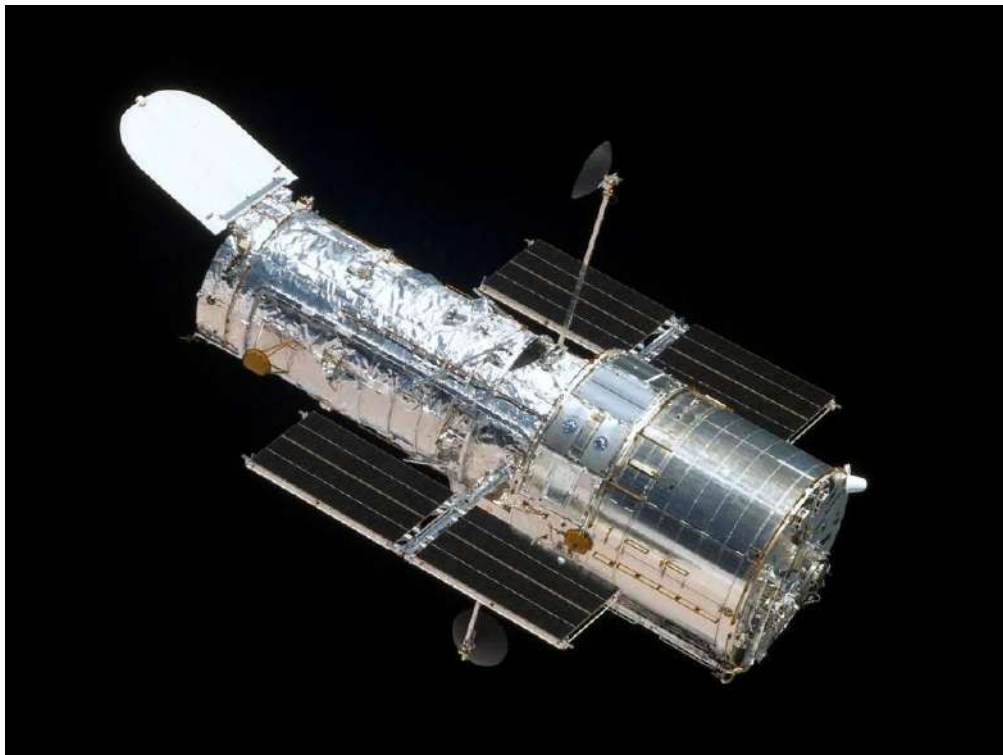


Figure 5.1. Hubble design

The Hubble Space Telescope (HST) has three interacting systems:

- The Support Systems Module (SSM), an outer structure that houses the other systems and provides services such as electrical power, data communications, and pointing control and manoeuvring
- The Optical Telescope Assembly (OTA), which collects and concentrates the incoming light in the focal plane for use by the science instruments
- Eight major science instruments, four housed in an aft section focal plane structure (FPS) and four placed along the circumference of the spacecraft. With the exception of the Fine Guidance Sensors (FGS), all are controlled by the Science Instrument Control and Data Handling (SI C&DH) unit.

Additional systems that also support HST operations include two Solar Arrays (SA).

5.2.2 Mission requirements

Considering the mission purpose, the orbit was chosen to be Low Earth Orbit, inclined of 28.5° from the equator. Hubble was launched using Space Shuttle Discovery, whose launch loads weren't included in this report due to the dismission of this launcher. Initially the mission duration was only 10 years but it was extended up to today, because of different service missions (latest: 2009).

The table below shows the main mission requirements.

Mission	
Orbit	LEO (593 km) inclined 28.5° from equator
Orbital Period	97 minutes
Period in Shadow	28-36 minutes
Orbital velocity	7.59 km/s
Mission Duration	>20 years
Launch mass	11110 kg
Launch Vehicle	Space Shuttle Discovery
Launch Site	Kennedy Space Center

Table 5.1. Hubble mission requirements

5.2.3 Mechanical requirements

The table below shows the main mechanical requirements, such as weight, length, shell, etc.

Mechanical	
Weight	11.110 kg tot (HST) 318 kg (FOC) 281 kg (WF/PC2) 374 kg (STIS) 390 kg (NICMOS) 229 kg (FGS) 290 kg (COSTAR)
Length	15.9 m
Light Shield and Forward shell	3.1 m
Diameter max (Equipment section and Aft Shroud)	4.2 m
Instruments dimensions	0.9 x 3 x 2.2 m (FOC) 1 x 1.3 x 0.5 m (WF/PC2) 0.9 x 0.9 x 2.2 m (STIS) 2.2 x 0.88 x 0.88 m (NICMOS) 0.5 x 1 x 1.6 m (FGS)

Table 5.2. Hubble mechanical requirements

Referring to 'Light Shield and Forward shell', the figure above shows in detail the overall HTS configuration.

The axial science instruments consisted of:

- Faint Object Camera (FOC)
- Faint Object Spectrograph (FOS)
- Goddard High Resolution Spectrograph (GHRS)
- Corrective Optics Space Telescope Axial Replacement (COSTAR).

In addition to the four axial instruments, four other instruments are mounted radially (perpendicular to the main optical axis). These radial science instruments are:

- Wide Field/Planetary Camera II (WF/PC II)
- Three Fine Guidance Sensors (FGS).

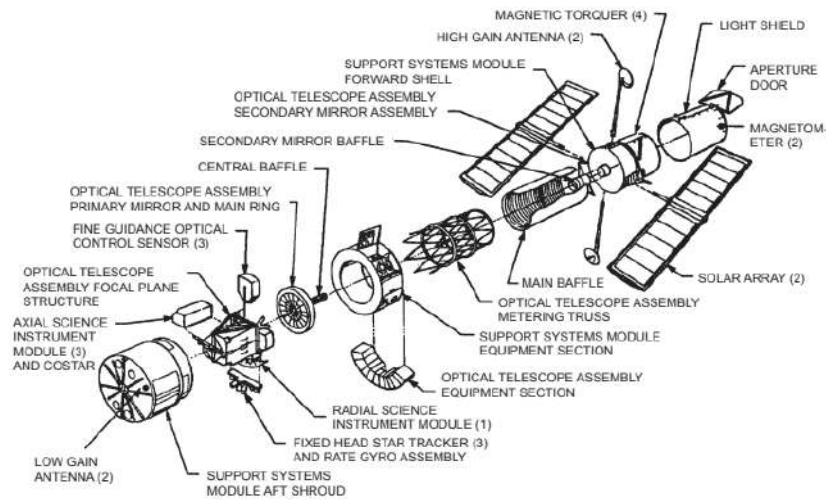


Figure 5.1. Hubble design breakdown

5.2.4 Thermal requirements

The table below shows the main thermal requirements.

Thermal	
MLI	Covers 80% of exterior
The insulation blankets	15 layers of aluminized Kapton and aluminized Teflon on the top
Tape on the aperture door surface facing the sun	Aluminum FOSR
Temperature sensors and thermistors	More than 200 temperature sensors and thermistors placed throughout the SSM, externally and internally, to monitor individual components and control heater operations.
Exterior surface temperature	from -150°F to +200°F (-101°C to +93°C)

Table 5.3. Hubble thermal requirements

It is important to point out that the mechanical and optical properties of the TC materials on HTS have degraded over the first 7 years in orbit. For SM2, there were large cracks in the metallized Teflon FEP and the emissivity of bonded metallized Teflon FEP radiator surfaces has increased over time. Close inspection of Teflon FEP revealed through-thickness cracks in areas with the highest solar exposure and stress concentration. The observation of HST MLI and ground testing of pristine samples indicate that thermal cycling with deep-layer damage from electron and proton radiation are necessary to cause the observed Teflon FEP embrittlement and the propagation of cracks along stress concentrations. Ground testing and analysis of retrieved MLI indicate that the damage increases with the combined total dose of electrons, protons, UV and x-rays along with thermal cycling.

5.2.5 Control requirements

A unique PCS (Pointing Control System) maintains Telescope pointing stability and aligns the spacecraft to point to and remain locked on any target. The PCS is designed for pointing to within 0.01 arcsec and holding the Telescope in that orientation with 0.007-arcsec stability for up to 24 hours while the Telescope continues to orbit the Earth at 17,000 mph. To point precisely, the PCS uses the gyroscopes, reaction wheels, magnetic torquers, star trackers, and FGSs.

The table below shows the main control requirements.

Control	
Pointing accuracy	0.007 arcsec for 24 hours
Angular resolution	0.1 arcsec at 6328 Å
Pointing control operation	gyroscope reaction wheels magnetic torquers star tracker FGCs
Fine Guidance Sensor (FGS)	Each FGS has a large (60 arcmin) FOV to search for and track stars
	Each FGS has 5.0 arcsec 2 FOV used by the detector prisms to pinpoint the star.
Reaction wheels	The momentum change caused by the reaction moves the spacecraft at a baseline rate of 0.22 degree per second or 90 degrees in 14 minutes

Table 5.4. Hubble attitude control requirements

5.2.6 Optical requirements

The OTA (Optical Telescope Assembly) consists of two mirrors, support trusses, and the focal plane structure. The light is collected by the concave 94.5-in. (2.4 m) primary mirror and converged toward the convex secondary mirror, which is only 12.2 in. (0.3 m) in diameter. The secondary mirror directs the still-converging light back toward the primary mirror and through a 24-in.hole in its centre into the Focal Plane Structure, where the science instruments are located.

The table below shows the main optical requirements.

Optical	
Focal length	57.6 m folded to 6.4 m
Primary mirror diameter	2.4 m
Secondary mirror diameter	0.3 m
Field of view	Faint Object Camera (FOC) 14.28 arcsec² Wide field/Planetary Camera II (WF/PC II) 4.7 arcmin² (WP), 0.3 arcmin² (PC) Space Telescope Imaging Spectrograph (STIS) CCD: 51 x 51 arcsec Near Infrared Camera and Multi-Object Spectrometer (NICMOS) 51.2 x 51.2 arcsec 19.2 x 19.2 arcsec 11.0 x 11.0 arcsec Fine Guidance Sensors (FGS) Access: 60 arcmin² Detect: 5 arcsec
Magnitude range (visual magnitude)	5 m to 29 m

Table 5.5. Hubble optical requirements

5.3 SENTINEL-3

5.3.1 Orbit Selection

The orbit selected for the Sentinel 3 is the result of a compromise between the requirements of all the missions that S-3 must complete, both topographic and optical. For the topography mission, the orbit has been optimized to facilitate the understanding and forecasting of the mesoscale variability of the oceans, and to make easier the observation of the surface wind and waves of the oceans, rivers, lakes ice sheets and sea-ice. All these requirements cannot be satisfied by the single altimeter of the S-3, and other altimeter systems “in constellation” must be considered. Due to the low inclination of the orbit (66°) in which the altimeter is foreseen for some time, the S-3 should maintain an ENVISAT type polar orbit: this way the mission is completely in line with GMES (Global Monitoring for Environment and Security, now re-branded into Copernicus) recommendations. The Ocean Colour mission requires a polar sun-synchronous orbit, with a local time of ascending node (LTAN) at the equator $\geq 10:00$ and a Solar Zenith Angle (SZA) $> 80^\circ$, preventing morning haze and cloud formation. It is fundamental to choose an orbit that minimize the negative impact of the sun-glnt, which can alter the ocean colour measurements. In order to allow an adequate carry out of the Sea Surface Temperature (SST) mission, it’s important to avoid diurnal stratification impacts, morning haze and afternoon clouds, and for these reasons is required a LTAN $< 11h$. After a trade-off analysis between all of these requirements, has been selected a sun-synchronous orbit, with an altitude of 814.5 km and a local equatorial crossing time of 10:00 a.m. The main parameters of the orbit are shown in the following table.

Mission Requirements	
Orbit	LEO at 814.5 km
Inclination	98.65° (Sun-Synchronous)
LTAN	10:00
Launch mass	1250 kg
Launch Vehicle	Vega / Rockot
Launch Site	Plesetsk Cosmodrome, Russia

Table 5.6. Sentinel-3 mission requirements

The configuration of S-3 presents two satellites (S-3A and S3-B), with an in-plane separation of 180°: the S3-B ground track is in the middle of the gap left by the S3-A, and this allows to maintain a balance between optical and topographic mission coverage, improving at the same time the payload coverage. The two satellites configuration can complete a full optical imaging of the oceans in just two days, even considering all the negative impacts of sun-glints, and deliver global land coverage in one day at the equator (even less at increasing latitudes).

5.3.2 Mechanical specifications: Sentinel-3 launch loads

The preliminary design of the main structure of Sentinel-3 was based on launch loads, as it generally happens with spacecrafts. These include high levels of vibrations associated with structures and noise field, accelerations, mechanical shocks due to pyrotechnic devices operations, a thermal environment which differs from both testing and space environment and pressure drop rate. The main peaks in the acoustic loads happens at lift-off, when the exhaust of the rocket motors fire at maximum thrust and the vibrations are reflected from the ground on the vehicle, and during transonic flight at Max Q because of shock waves and turbulences. The launcher user's manual provides the loads that the spacecraft is subjected to during all launch phases. In the picture below the acoustic environment of Vega from the user's manual is shown:

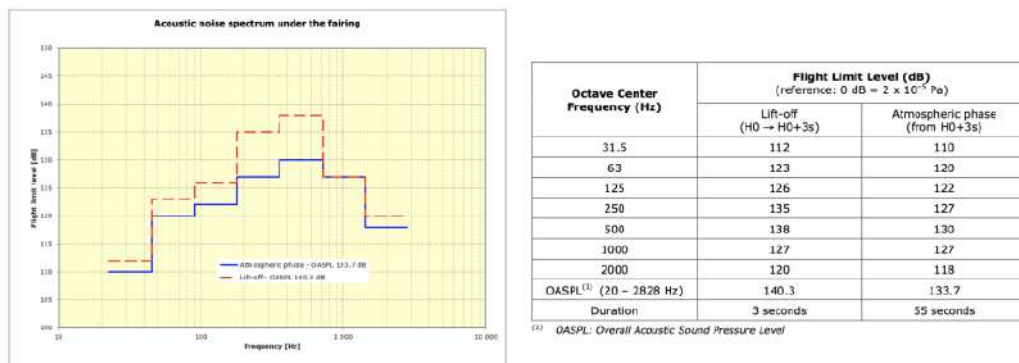


Figure 5.2. Vega acoustic environment

As structures vibrate randomly in response to acoustics, an accurate analysis and testing campaign has to be made to predict vibration environments for the spacecraft components and to verify the life span and the strength of sensitive structures. Generally, lightweight and wide structures are more subjected to acoustics than heavy and little ones. The picture below refers to the comparison of calculated random vibration levels for the Sentinel-3 solar array drive mechanism (SADM) with the levels required by the test specification.

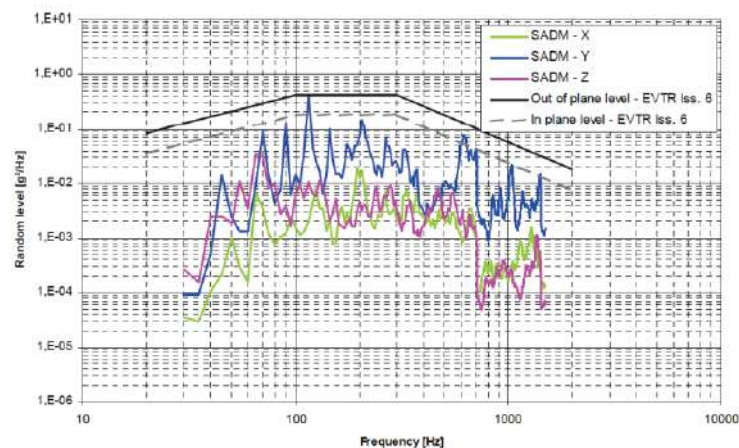


Figure 5.3. SADM random vibration test levels and requirements level

Vega was indeed one of the two launch vehicles that were proposed for Sentinel-3 mission: the other one is Rockot. The diagram below shows the envelope of quasi-static loads at qualification level for Sentinel-3 for these two launchers.

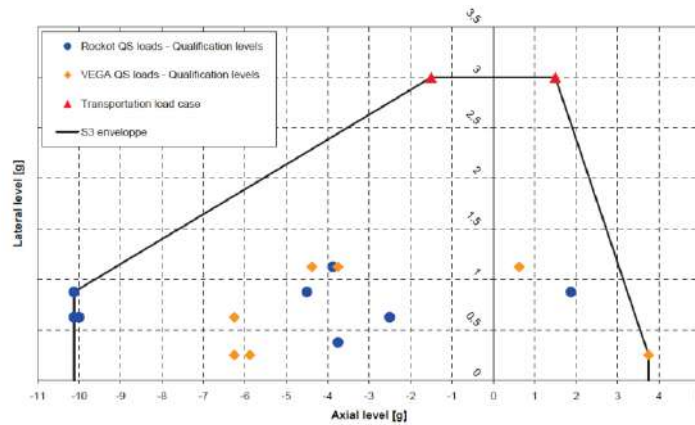


Figure 5.4. Launch and transportation loads for Sentinel-3 on Vega and Rockot launchers

It is also necessary to consider the eventual loads due to transportation, both in containers or airborne, which are also pointed out in the diagram. The qualification test loads were defined to verify the global stiffness, both axially and laterally, with subsystems and interfaces included. A choice was made to validate the structure with criteria of no yield or ruptures in the prototype at qualification loads.

A common test used to verify the strength of primary and secondary structures of spacecrafts is the sine vibration test. Generally, the real dynamic excitation profile is aperiodic, but these tests have been proved really helpful in the validation of structures during qualification and protoflight testing. The use of a swept-sine excitation with frequencies up to 100 Hz can lead to both under and over testing of the hardware tough. In fact, exciting only one component resonance at a time instead of multiple resonances as would be induced by the real excitation profile, results in an undertesting. On the contrary, a swept-sine test applies a larger number of stress cycles to the components, resulting in over testing. Also, the natural modes of the structures are generally different during test phase with respect to launch because of the coupling with the adapter and the rest of the structure. To make sure that the test does not exceed design capabilities, the sine input spectrum is notched considering the maximum axial and lateral quasi-static load levels specified in the launcher manuals. The picture below

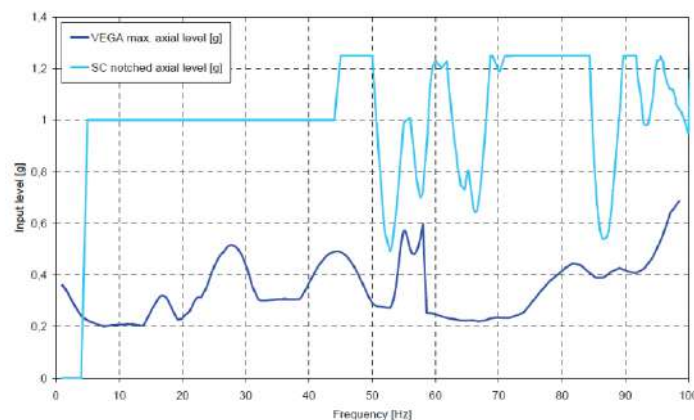


Figure 5.5. Sentinel-3 sine test predicted notching compared to the levels from the coupled load analysis

shows the comparison between the predicted axial load levels and the spacecraft/launch vehicle coupled load analysis results, denoting a sufficient margin between the two as it is desirable.

Another important constraint to deal with during launch is the thermal environment. During ascent, the spacecraft is subjected to an aerothermal flux which can be considerable because of the supersonic speeds reached. This requires designating the adequate point in time and space after lift-off where the fairing can be jettisoned. Jettisoning too low would increase too much the aerothermal flux on the payload and worsen the aerodynamics, while maintaining the fairing for too long would mean carrying an undesired mass, slowing the ascent.

As depicted in the graph below, in the case of Vega, the peak heat flux experienced by Sentinel-3 payload after the fairing is jettisoned is around 200 W/m². The graph on the right shows the typical heating profile inside and outside the fairing for a Scout rocket launch vehicle.

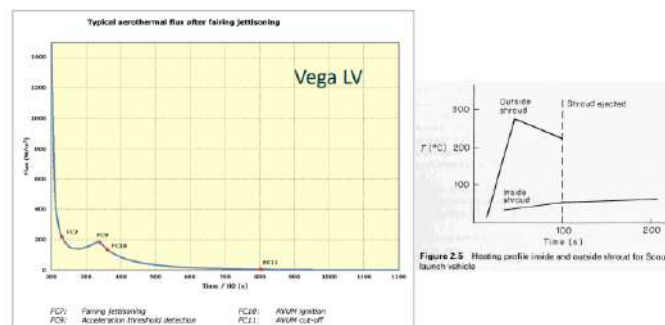


Figure 5.6. Typical aerothermal flux of Vega payload after

5.3.3 Mechanical specifications: S-3 stepper *SADM* 'Solar Array Drive Mechanism'

Solar Array Panels provide power to run many satellite subsystems such as sensors, active heating, telemetry and even spacecraft propulsion (Solar-Electric Propulsion). In order to meet all these power requirements, it's necessary to get the maximum sun light onto solar cells, hence a continuous correction of panel's orientation must be considered. The rotating mechanism performing this task is the *Solar Array Drive Mechanism SADM*. This mechanism acts as an *actuator* and as an *electrical power transfer element* between the Solar Arrays and the main body of the spacecraft.



Figure 5.7. KARMA-4 SADM with Slip Ring used on Sentinel-3.

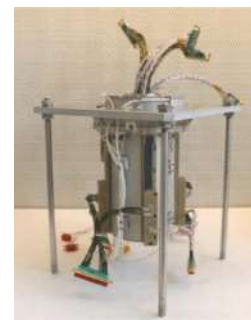


Figure 5.8. KARMA-4 SADM visualization of Driveline and Transfer Unit

Many *SADMs* are currently flying on satellites such as *Sentinel-1*, *Sentinel-3* and *ExoMars*. As an example, *Sentinel-3* is equipped with the 'Kongsberg Adaptive Rotational Mechanism Assembly Second Generation' *KARMA-4 SG* (Figure above).

KARMA-4 SG consists of two main subsystems:

- The *driveline* contains all torque generating components and a structural element to interface with the wall of the spacecraft. Thank to this item, it's possible to keep the solar array (cantilever structure) attached to the main body and to react forces resulting from in orbit manoeuvres and launch;
- The *rotatory solar power and signal transfer unit* consists of a twist capsule, for limited rotation, or a slip ring for continuous rotation. This unit uses stepper motor to rotate panels in the desired direction and transfers the power and signals from rotating frame (solar array) to stationary frame (spacecraft).

Mechanical Requirements		
Solar Array Drive Mechanism SADM	Mass	3.8 [kg] to 4.7 [kg]
	Dimensions	240 [mm] x Ø 150 [mm]
	Range Temperatures	−30 [°C] to + 85 [°C]
	Rotation Speed	2 [°/s] (resolution 0.0136 [°])
	Power Requirements	3 [W]
Solar Array Deployment Mechanism DM	Mass	1.5 [kg]
	Dimensions	122 [mm] x 110 [mm] x 70 [mm]
	Range Temperatures	−30 [°C] to + 50 [°C]
	Deployment Range	180 [°] (accuracy 0.006 [°])
	Torque	4 [Nm] to 10 [Nm]
	Axial Rotation Stiffness	$2.3 \cdot 10^4$ [Nm/rad]
Solar Array Hold-Down Mechanism HDM	Mass	0.15 [kg]
	Dimensions	63 [mm] x Ø 62 [mm]
	Range Temperatures	−70 [°C] to + 80 [°C]
	Pre-Actuation Load	≥ 400 [N]
	Activation Time	120 [s]
	Life	≥ 50 cycles

Table 5.7. Sentinel-3 mechanical requirements

5.3.4 Optical requirements

Sentinel-3 uses infrared, visible and microwave bands both for ocean and land measurements. For this reason, optical requirements strongly depend on the type of mission considered:

- Sea Surface Heights (SSH) mission objective is to perform a measurement of wave heights and wind speed using a microwave radiometer.
- Sea Surface Temperature (SST) mission rely on IR and MW bands to retrieve skin temperature changes to the order of 0.1-0.3° K.
- Ocean Colour (OC) acquires both physical (dynamics) and biological (colour) ocean data
- Land Colour (LC) mission objective is to monitor vegetation health, crop management, deforestation, rivers and lake levels, etc.
- Land Surface Temperature (LST) main objective is to monitor active fire and burned surface

Optical Requirements		
Sea Surface Heights (SSH)	Spectral bands	Ku Band
	Spatial resolution	100 km / 25 km
	Temperature accuracy	-
	Revisit Time	2-3 days
	Observation Time	-
	Coverage	3-10 days
Sea Surface Temperature (SST)	Spectral bands	-
	Spatial resolution	1 km
	Temperature accuracy	0.1-0.5 K
	Revisit Time	1-3 days
	Observation Time	Local time around 10:00
	Coverage	Global in 2-3 days at equator
Ocean Colour (LC)	Spectral bands	Min. 15 bands from 400-1050 nm.
	Spatial resolution	2-4 km (global); 0.2-0.5 km (coastal)
	Temperature accuracy	-
	Revisit Time	1 day (coastal); 2-3 days (global)
	Observation Time	Optimized to minimize sun glint and cloud cover
	Coverage	-
Land Color (LC)	Spectral bands	Minimum of 15 bands from 443-1085 nm
	Spatial resolution	0.25-0.5 km
	Temperature accuracy	-
	Revisit Time	1-3 days
	Observation Time	Local time around 10:00
	Coverage	Global in 2 days at equator
Land Surface Temperature (LST)	Spectral bands	3.5-12 μ m
	Spatial resolution	-
	Temperature accuracy	0-650 K
	Revisit Time	1 day
	Observation Time	-
	Coverage	-

Table 5.8. Sentinel-3 optical requirements

5.3.5 Control specifications and requirements

Sentinel-3 is 3-axis stabilized. It uses a new generation of avionics and a software designed after PROBA program heritage. The Canadian space agency provided the GNC algorithms.

Sentinel uses a multi-head star tracker (HYDRA), a coarse sun sensor, magnetometers and a GNSS receiver for attitude and orbit positioning determination. Thrusters are used only for OCS purposes, while 4 reaction wheels with magnetic off-loading provide attitude control.

AOCS Requirements		
Actuators	Reaction wheels	ACS
	Magnetorquers	Magnetic off-loading
	Thrusters	OCS
Pointing accuracy	Absolute pointing error	<0.1°
	Absolute measurement error	<0.015°
	Maximum control loop pointing error	5 arcsec
	Steady state angular stability accuracy	10-15 arcsec
Sensors	Multi head star tracker	HYDRA
	Coarse sun sensor	-
	Magnetometers	-
	GNSS receiver	Orbit determination
	Coverage	-

Table 5.9. Sentinel-3 attitude and orbit control requirements

5.3.6 Thermal Requirements

Sentinel-3 structure has been designed to guarantee the correct carrying out of the mission in every phase, even in case of unpredictable events, making sure that each element of the system is kept at the right temperature. To make this possible, all payload items and all onboard equipment have been decoupled, making them as independent as possible one from each other.

The S/C Thermal Control must ensure the proper temperature for each subsystem in every phase of the mission, provide the sensors and the heaters for the S/C temperature monitoring and control and minimize thermal gradient and mass and heater power demand.

To fulfil all these requirements, the thermal control system is based on a passive design, with the addition of an heaters-based active control. The main rejection of the heat is carried out by dedicated radiators, and layers of MLI blankets reduce the heat flow to and from the external environment. To reduce thermal gradients, internal units and panels are black painted, to maximize emissivity. In addition, the external layer is made of Black Kapton to provide electrical conductivity to avoid surface charging under UV light. In the following table the key aspects of the MLI lay-up are defined.

Type	Use	Lay-up			Temp. range [°C]	Thermo-optical properties		
						ϵ	α_{BOL}	α_{EOL}
EXT-1	Ext. Surfaces	LAY	QTY.	MATERIAL	LOCATION	+190/ -190	0,77	0,9
		1	1	1 MIL BLACK KAPTON x VDA INTERNALLY	EXTERNAL			
		2	6	DACRON SPACER Platest	LAY-UP			
		3	6	0.3 MIL VDA x KAPTON x VDA perf.	LAY-UP			
		4	6	POLYESTER SPACER PV-8g	LAY-UP			
		5	6	0.25 MIL VDA x MYLAR x VDA perf.	LAY-UP			
		6	1	POLYESTER SPACER PV-8g	LAY-UP			
		7	1	1 MIL VDA x MYLAR x VDA perf.	INTERNAL			
INT-1	Tank, RCS component, internal PIM	LAY	QTY.	MATERIAL	LOCATION	+150/ -190	0,05	0,15
		1	1	1 MIL VDA x MYLAR x VDA perf.	EXTERNAL			
		2	1	POLYESTER SPACER PV-8g	LAY-UP			
		3	8	0.25 MIL VDA x MYLAR x VDA perf.	LAY-UP			
		4	8	POLYESTER SPACER PV-8g	LAY-UP			
INT-2	PIM Struts	LAY	QTY.	MATERIAL	LOCATION	+150/ -190	0,77	0,9
		1	1	1 MIL BLACK KAPTON x VDA INTERNALLY	EXTERNAL			
		2	1	POLYESTER SPACER PV-8g	LAY-UP			
		3	1	1 MIL VDA x MYLAR x VDA perf.	LAY-UP			
		4	1	POLYESTER SPACER PV-8g	LAY-UP			
		5	8	0.25 MIL VDA x MYLAR x VDA perf.	LAY-UP			
		6	8	POLYESTER SPACER PV-8g	LAY-UP			
INT-3	RCS piping, propulsion items	LAY	QTY.	MATERIAL	LOCATION	+150/ -190	0,05	0,15
		1	1	1 MIL VDA x MYLAR x VDA	EXTERNAL			
		2	1	POLYESTER SPACER PV-8g	LAY-UP			
		3	2	0.25 MIL VDA x MYLAR x VDA perf.	LAY-UP			
		4	2	POLYESTER SPACER PV-8g	LAY-UP			
INT-4	Battery	LAY	QTY.	MATERIAL	LOCATION	+150/ -190	0,05	0,15
		1	1	1 MIL VDA x MYLAR x VDA perf.	EXTERNAL			
		2	1	POLYESTER SPACER PV-8g	LAY-UP			
		3	8	0.25 MIL VDA x MYLAR x VDA perf.	LAY-UP			
		4	8	POLYESTER SPACER PV-8g	LAY-UP			
HT-1	Thrusters	LAY	QTY.	MATERIAL	LOCATION	+290/ -190	0,77	0,9
		1	1	1 MIL BLACK KAPTON x VDA	EXTERNAL			
		2	19	VDA x 0.3 MIL KAPTON x VDA perf., emb.	LAY-UP			
		3	1	1 MIL KAPTON x VDA	INTERNAL			

Table 5.10. Sentinel-3 thermal requirements

Heaters are used to provide the temperature control of delicate components of the S/C (e.g. battery, propellant tank and external sensors), and to ensure correct temperature keeping. The design of the heater presents main + redundant heater lines (M+R HTRs). Each of these lines is controlled by a set

of three thermistors, which command heater circuits and monitor the on-board temperature. Thermistors used on Sentinel-3 are Betatherm G15K4D489.

5.4 ASTERIA CUBESAT

5.4.1 Overview

Overview of this mission is already considered in a previous paragraph of this dissertation, so the small table below is considered only a brief reminder of that.

Mission	
Mission	ASTERIA
Orbit	LEO
Orbit Altitude	404 km
Inclination	51.6°
Launch mass	12 kg
Launch Vehicle	Falcon 9
Deployment from ISS	NanoRacks CubeSat Deployer (NRCSD)

Table 5.11. ASTERIA mission requirements

5.4.2 Mechanical Requirements

One of the most notable features of this CubeSat is its design, developed to be able to be loaded in the deployment system of the International Space Station.

First, ASTERIA CubeSat was stowed in a special configuration and integrated for launch with a Falcon 9 rocket (for the launch requirements of this rocket, see another paragraph in this section). When Falcon payload is finally arrived at ISS, the crew of station “unboxed” partially the special stowage of ASTERIA and then, loaded it into the sophisticated trails system of CubeSat deployer, a special set-up that allow to insert in orbit small CubeSat directly from the station, orbiting at 400 km.

In this small section, requirements from this stage of the mission are analysed. They derive in a large amount from the Interface Definition Document (IDD) of the deployer but also from the policy and requirements documents for the safety of ISS and from the Mobile Servicing System (MSS) Interface Control Document. Last one refers about a robotic arm of Station, the JEM, that is used to shift the payload and the deployer in the final spot outside the Station, in order to reach the correct parameters for the mission, e.g. orbit altitude and inclination.

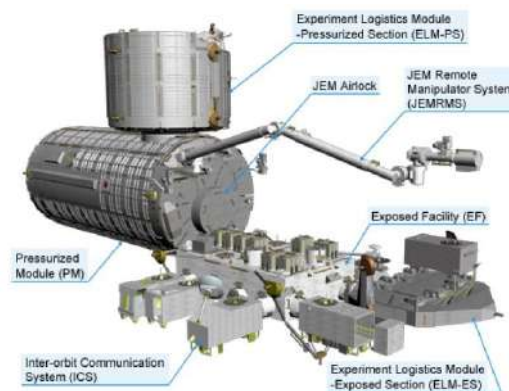


Figure 5.9. JEM system overview

NanoRacks CubeSat Deployer (NRCSD) consist of a railway like structure, where CubeSat are loaded thanks to an access panel, located on top of the apparatus. Then, a set-up of springs at aft of cubicle, push the CubeSat outside the path and so, outside the Station; deployment of CubeSat in a nominal orbit is so achieved.

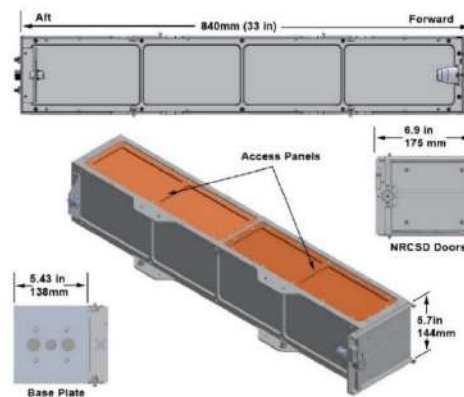


Figure 5.10. NanoRacks Cubesat Deployer (NRCSD) design features

Deployment from ISS	
CubeSat rail along the longitudinal axis	• Number of rails: four (4) • 1U rail length: 113.50mm • 2U rail length: 227.00mm • 3U rail length: 340.50mm • 4U rail length: 454.00mm • 5U rail length: 567.5mm • 6U rail length: 681 to 740.00mm
CubeSat mass shall be maximum	• 1U: 2.40 kg • 2U: 3.60 kg • 3U: 4.80 kg • 4U: 6.00 kg • 5U: 7.20 kg • 6U: 8.40 kg
CubeSat centre of mass (CoM) shall be located within the range aside, relative to the geometric centre of payload	• X-axis: (+/- 2cm) • Y-axis: (+/- 2cm) • Z-axis: • 1U: (+/- 2cm) • 2U (+/- 4cm) • 3U (+/- 6cm) • 4U (+/- 8cm) • 5U (+/- 10cm) • 6U (+/- 12cm)
CubeSat shall be capable of withstanding a deployment velocity in the range	From 0.5 to 2 m/s
CubeSat shall be capable of withstanding a tip-off rate of	5 deg/sec per axis
CubeSat critical structures shall withstand when exposed to accelerations aside	• Nx: +/- 7.0 g • Ny: +/- 4 g • Nz: +/- 4 g • Rx: +/- 13.5 rad/s²

	• Ry: +/- 13.5 rad/s² • Rz: +/- 13.5 rad/s²
CubeSat shall be capable of withstanding the random vibration environment for flight with appropriate safety margin for 60 secs in any axis	Maximum Expected Flight Level (MEFL) + 3dB grms (soft-stow): 5.76 g ² /Hz grms (hard-mount): 9.47 g ² /Hz
CubeSat shall be capable of withstanding the loads inside of the NRCSD when exposed to the acceleration environment during the Extra Vehicular Robotics operations	• On-orbit: 2.0 m/s² • Airlock carry out: 1.5 m/sec² • Emergency stop: 0.69 m/s²
The CubeSat shall be capable of withstanding a force of [] across all rail ends in the Z axis.	12000 N
The CubeSat shall be capable of withstanding the pressure extremes and depressurization / pressurization rate of the airlock as defined aside	• Airlock Pressure: 0 to 104.8 kPa • Airlock pressure depressurization/re-pressurization rate: 1.0 kPa/sec
The Maximum Effective Vent Ratio (MEVR) of the CubeSat structure and any enclosed containers internal to the CubeSat shall be	<5080 cm
CubeSat shall not contain any pyrotechnics devices	-
Satellites shall not utilize any non-metallic materials with specifications aside	• Total Mass Loss (TML) > 1.0% • Collected Volatile Condensable Material (CVCN) value > 0.1%

Table 5.12. ASTERIA mechanical requirements, deployment from ISS

5.4.3 Thermal Requirements

To achieve the payload temperature control objective, ASTERIA required two separate but non-independent thermal designs: the system-level thermal design and a design for focal plane temperature stability.

A significant driver of the thermal design was the inclusion of body-mounted solar panels, which impart a non-trivial thermal load on the spacecraft when illuminated. The combination of this and other internal dissipations, and environmental loads, led to a surface coating of 10-mil silver Teflon over nearly the entire exposed surface of the spacecraft. All the chassis walls were well connected thermally, so the entire spacecraft body served as a “radiator.” The components with high power dissipation (flight computer, payload electronics, radio) were purposefully well-coupled to the bottom plate. The telescope, the batteries and the interface board were thermally isolated from the bottom plate using titanium bipods for the telescope and low conductivity polymer spacers for the batteries and interface board. The batteries were isolated to avoid reaching the low-temperature limit during a worst-case deployment, and the Interface Board was isolated to reduce the amplitude of the temperature disturbance on the temperature-sensing circuitry.

ASTERIA implements a multi-layered approach to thermal control. First, the OTA (optical telescope assembly) is isolated as much as possible from the payload electronics and the spacecraft bus. The only connection (besides electrical wiring) between the telescope and any part of the flight system is through titanium bipods that affix the OTA to the bottom panel of the spacecraft. The estimated total conductance of the bipods is 0.015 W/K. With the OTA isolated from the rest of the spacecraft, additional layers of thermal control are implemented as shown Fig. 5.12.

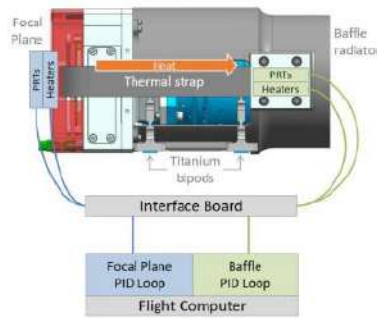


Figure 5.11. ASTERIA thermal control architecture

The imager, located inside the red enclosure, generates over 1 W of power while operating. This heat is moved from imager through a thermal strap (~ 1 W/K thermal conductance) to the baffle, which acts as a radiator to space. The OTA has two active thermal control loops, one at the imager and one at the baffle end of the thermal strap. Each control loop consists of several co-located platinum resistance thermometers (PRTs) and resistive heaters. A proportional-integral-derivative (PID) controller reads the PRTs and applies small amounts of heat to maintain the desired set points at the baffle and the focal plane. The control loop around the baffle acts as a “coarse” control, compensating for relatively large environmental disturbances and controlling to ± 0.5 K. The control loop around the imager acts as “fine” control, compensating for residual fluctuations and controlling to ± 0.01 K.

Figure 5.12 shows the effect of the optical telescope isolation and active thermal control over many orbits. Each sinusoidal variation corresponds to a single orbit and the spacecraft chassis varies over a range of $\pm 7^\circ\text{C}$. The Interface Board, which is mounted to the chassis, varies too ($\pm 2.5^\circ\text{C}$) albeit with a lower amplitude thanks to low thermal conductivity material at the mounting interface. The activation of thermal control is clearly visible in Figure 2, causing a step increase in focal plane temperature. Prior to active control, the focal plane, baffle, and piezo stage temperatures oscillate in phase with the chassis, although at a lower amplitude due to the low-conductivity titanium bipods between the OTA and the rest of the spacecraft. During this pre-control period the baffle temperature varies $\pm 1^\circ\text{C}$ and the piezo stage and focal plane both vary $\pm 0.75^\circ\text{C}$, both very close to thermal model results. Once the control system is activated, the mean focal plane temperature increases to the set point value. Because the imager and thermal control system dissipate heat and are well coupled to the other payload components, the mean temperature of the baffle, piezo stage, and to some extent the spacecraft chassis all experience a corresponding temperature increase. The baffle and piezo stage variation after thermal control is active decrease to $\pm 0.5^\circ\text{C}$.

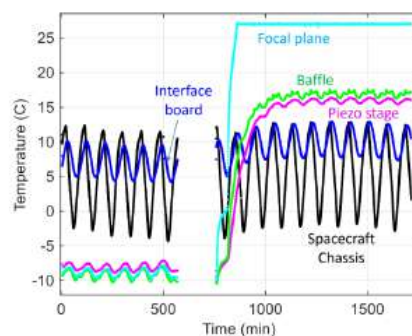


Figure 5.12. ASTERIA flight data showing temperatures of spacecraft and payload components both before and after thermal control is activated

Thermal control system	
Thermal stability	Temperature stability of ± 0.01 K at a single location on the focal plane over a 20 minute observation.
Thermal strap	A thermal strap shall connect the back of the imager with the baffle providing a thermal path for rejecting heat dissipated by imager.
Titanium bipods	The estimated total conductance is 0.015 W/K.
Control loops	The control loop around the baffle shall compensate for relatively large environmental disturbances and control to ± 0.5 K.
	The control loop shall compensate for residual fluctuations and control to ± 0.01 K.
Surface coating	A 10-mil silver Teflon surface coating shall cover nearly the entire exposed surface of the spacecraft
Chassis walls	Temperature shall vary over a range of $\pm 7^\circ\text{K}$
Interface Board	Temperature shall vary over a range of ± 2.5 K
Batteries and interface board	They shall be thermally isolated from the bottom plate using low conductivity polymers spacers
Flight computer, payload electronics, radio	They shall be coupled to the bottom plate
Maximum orbit-average power dissipation	24 W (corresponding to Safe Mode operation in full sunlight)
Deployer (phases)	Dragon pressurized cargo: 18.3°C to 29.4°C
	On-orbit, EVR Prior to Deployment: -10°C to 45°C

Table 5.13. ASTERIA Thermal requirements

5.4.4 Attitude Control Requirements

The objective of ASTERIA is to demonstrate capabilities that enable photometry in a nanosatellite platform: specifically, regarding attitude and pointing control, to demonstrate optical line-of-sight pointing stability of 5 arcseconds root mean square (RMS) over a 20-minute observation and pointing repeatability of 1 arcsecond RMS from one observation to the next. The extended mission objectives are, among others, to:

- demonstrate onboard orbit determination in Low Earth Orbit (LEO) using autonomous navigation (AutoNav) without the Global Positioning System (GPS), to achieve fully independent means of spacecraft orbit determination for Earth orbiters using only passive imaging;

- collect data on ASTERIA that will feed into future missions by characterizing the spacecraft pointing jitter as a function of target brightness, reaction wheel speed, controller gain, and number of guide stars.

The attitude determination and control subsystem (ADCS) used on ASTERIA is the Blue Canyon Technologies (BCT) fleXible Attitude Control Technology (XACT). This is a fully integrated subsystem with both hardware and software necessary to control the spacecraft's attitude: it features a star tracker, sun sensor, inertial measurement unit (IMU), magnetometer, reaction wheels, torque rods, and processor. The XACT's flight software is self-contained and capable of autonomously detumbling the spacecraft, searching for the sun, and maintaining a sun-pointed coarse attitude. A GPS unit was also added to assist with providing accurate time, position, and velocity information to the XACT and spacecraft.

Attitude control	
Optical line-of-sight pointing stability	5 arcseconds root mean square (RMS) over a 20-minute observation
Pointing repeatability	1 arcsecond RMS
3-axis attitude determination and control subsystem (ADCS)	Blue Canyon Technologies' (BCT) fleXible Attitude Control Technology (XACT), fully integrated subsystem with both hardware and software
ADCS lifetime	> 5 years
Sensors	• Star tracker (BCT's Standard NST) • MEMS inertial measurement unit (IMU) • Sun sensor • MEMS Magnetometer • GPS unit
Actuators	• 3 BCT's RWp015 reaction wheels for attitude control • 3 torque rods for momentum control
Flight software	Self-contained and capable of autonomously detumbling the spacecraft, searching for the sun, and maintaining a sun-pointed coarse attitude
Minimum coarse attitude control time during observation	20 minutes (more depending on observation duration)
Spacecraft pointing accuracy	± 0.003 deg (1-sigma) for 2 axes ± 0.007 deg (1-sigma) for 3rd axis
Momentum capacity	≥ 15 mNms
Slew rate	≥ 10 deg/sec

ADCS mass	≤ 1 kg
ADCS max dimensions	0.5 U (10 x 10 x 5 cm)
ADCS electronics input voltage	12 V
Reaction wheel (RW) spec torque	≥ 4 mNm
RW max momentum	≥ 15 mNms
RW diameter	≤ 4.3 cm
RW mass	≤ 0.130 kg
RW max power	≤ 1 W
RW nominal power	≤ 0.6 W
Sun sensor accuracy	< 2 deg
Star tracker (ST) accuracy	≤ 6 arcsec (bore-sight) ≤ 40 arcsec (roll axis)
ST minimum star magnitude	7.5
ST sun keep-out angle	± 45 deg
ST update rate	5 Hz
ST lost in space solution time	≤ 4 seconds
ST field of view	10 x 12 deg
ST sky coverage	$> 99\%$
ST mass	≤ 0.35 kg
ST max dimensions	10 x 6.73 x 5 cm
ST nominal power	≤ 0.75 W
ST peak power	≤ 1.0 W
ST operating voltage	5 ± 0.1 V

Table 5.14. ASTERIA attitude control requirements

5.4.5 Optical Requirements

The ASTERIA payload is a wide field-of-view optical telescope that contains specialized hardware to perform the pointing and thermal control functions. The optical telescope assembly (OTA) consists of a lens assembly, a two-axis piezoelectric nano positioning stage, an imager, a thermal strap, and a baffle. The lens assembly is a custom 5-element refractive design that is similar to a single lens reflex (SLR) camera lens but with additional features to withstand the environments of launch and low-Earth

orbit (LEO). The imager mounts to the moving portion of the piezo stage, and the stationary portion of the piezo stage mounts to the lens assembly.

The payload has two modes of operation:

1. Full frame: the payload acquires a single image containing all pixels in the array.
2. Windowed mode: mode used for observation that use up to 8 windows.

Optical Requirements	
Time of observation	in eclipse not during communications passes limited to a maximum of 30 minutes to avoid overflowing the image buffer in FSW memory
Position of target star	20 degrees away from the Moon to minimize stray light
Payload boresight	90 degrees away from spacecraft nadir to avoid stray light from the Earth limb
Time of exposure to the Sun of the solar arrays	up to a maximum of 43 consecutive minutes
Pointing Stability	5 arcsecond RMS over 20-minute observation to observation

Table 5.15. ASTERIA Optical requirements

5.5 ENVISAT

5.5.1 Introduction

Envisat (*Environmental Satellite*) was an Earth observation satellite developed by ESA. Its objectives were studying and monitoring the Earth environment to understand its *dynamics*. The purpose of the mission was to improve the measurements initiated by ERS^[1]-1 and ERS-2. Therefore, the main objectives of Envisat were:

- to enhance ocean and ice studies.
- to give an important contribution on atmospheric chemistry and biology.
- to allow the optimal monitoring and management of the resources on the Earth surface.
- to understand the solid Earth processes.

Other covered disciplines are meteorology and climatology.

In conclusion, Envisat was launched on 1st March 2002 aboard an Ariane V. Due to a sudden interruption of communications on 8th April 2012, ESA officially announced the end of the mission about one month later.

¹ European Remote Sensing Satellite

5.5.2 Mission requirements

The requirements for the mission are shown in the following table.

Mission requirements	
Orbit	<i>Sun synchronous polar</i>
Orbital period	<i>100.16 minutes (~1h 40')</i>
Semi-major axis	<i>7145 km</i>
Eccentricity	<i>0.00042</i>
Orbital inclination	<i>98.4°</i>
Orbital velocity	<i>7.47 km/s (average)</i>
Mission duration	<i>Planned: 5 years Final: 10 years, 1 month, 6 days</i>
Launch mass	<i>8211 kg</i>
Propellant mass	<i>319 kg</i>
Launch vehicle	<i>Ariane V</i>
Launch site	<i>Guiana Space Centre</i>

Table 5.16. Envisat mission requirements

As shown in the table, orbital parameters have an important role, in particular the orbital inclination and the semi-major axis. The satellite must cover the whole Earth surface in optimal visibility conditions, because of its purpose. Therefore, the two discussed parameters must guarantee a stable sun synchronous polar orbit. By adopting these parameters, the repeat cycle of the satellite is about 35 days within 501 orbits. This cycle implies that satellite points to the same Earth position only after 35 days.

5.5.3 Mechanical requirements

The design requirement from Ariane-5 is for a longitudinal quasi-static load of 4.55g and survival of 1g from 5 to 100Hz. Lateral accelerations are below 0.8g up to 100 Hz. In common with most large space structures, extensive notching was necessary to protect sensitive items, such as the solar array, the ASAR antenna, and several of the optical instruments. The vibration spectra, which were accepted by Ariane space on the basis of coupled dynamic analysis, were less than 0.3g in longitudinal and less than 0.35g in one lateral axis from 35 to 100 Hz. The other lateral axis (in the plane of the Earth-pointing face) was less than 0.6g in the same region.

Requirements	Value
<i>Static load</i>	<i>< 2.8g</i>
<i>Static load (frequency)</i>	<i>< 15 Hz</i>
<i>First lateral frequency</i>	<i>< 5.5 Hz</i>
<i>Longitudinal quasi – static load</i>	<i>< 4.55g</i>
<i>Longitudinal quasi – static load (frequency)</i>	<i>< 50Hz</i>
<i>Lateral accelerations</i>	<i>< 0.8g</i>
<i>Lateral accelerations (frequency)</i>	<i>< 100 Hz</i>
<i>Vibration spectra (longitudinal axis)</i>	<i>< 0.3g</i>
<i>Vibration spectra (lateral axis)</i>	<i>< 0.35g</i>
<i>Vibration spectra (other lateral axis)</i>	<i>< 0.6g</i>

Table 5.17. Envisat mechanical requirements

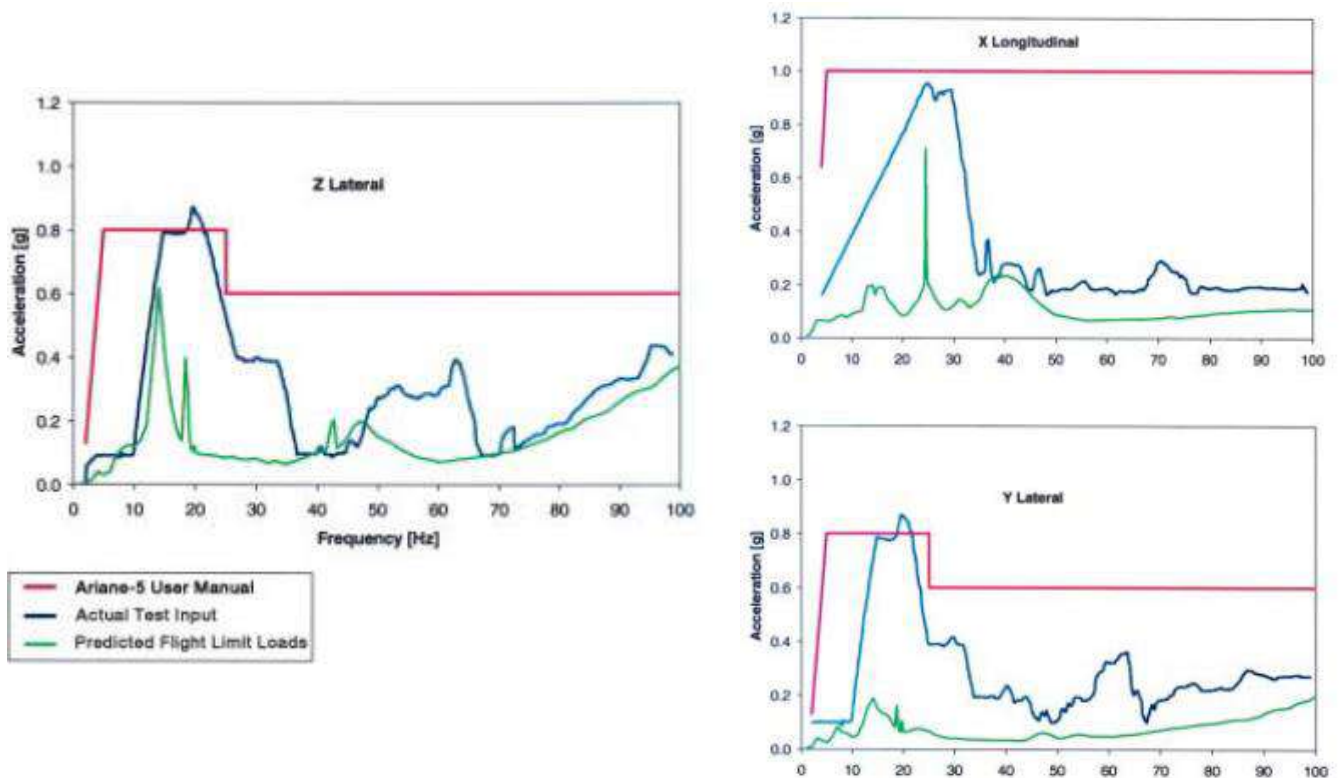


Figure 5.13. Spectra of vibration of Ariane 5, comparison with predicted

5.5.4 Thermal requirements

The spacecraft's thermal design is driven by the orbital conditions, and by the need to survive in the Sun-pointing safe mode. The thermal design of the box structure of the basic spacecraft is classically passive. Portions of the box's surface serve as radiators, whilst the rest is clad in multi-layer insulation. Externally mounted items, especially the instruments, are designed to have as little thermal interchange as possible with the box structure itself. Systems of heaters operated by thermostats protect equipment in safe mode. Specific thermostat-controlled heaters protect the hydrazine system and the batteries. Software controlled heaters are operating at other times when on-board computers are available to control them. The Instrument Control Unit of each instrument controls the thermal management of that instrument when it is on. A number of optical instruments incorporate heat pipes to transfer heat away from detectors to appropriately positioned radiator plates.

Instrument	Requirements
MIPAS	65-75 K
SCHIAMACY	Around 200K (VIS); around 135K (IR)

Table 5.18. Envisat thermal requirements

5.5.5 Optical requirements

Envisat carried on board many experiments, each one providing its own requirements to be met. As our focus centres on optical requirements the following paragraph reports the requirements developed for one of the main payload instruments involving a telescope: the GOMOS experiment. The goal of this experiment was to investigate atmospheric characteristics, such as ozone and temperature layering, plus some small-scale turbulence measurements. The goal was designed to be accomplished via a very sensible spectral analysis of the atmosphere, so a relatively large telescope (20 x 30 cm) and very low-noise detectors were needed.

In the following tables the requirements developed for the mission are reported.

Requirement description	Requirement	Design driver for:
Occulting stars characteristics	Visual magnitude range: Maximum -1.6 to minimum 2.4 to 4 for stars with 30,000K and 3,000K temperature respectively	High sensitivity and dynamic range requirements for the star tracker High sensitivity and dynamic range spectrometer detectors (especially in the UV) Large telescope and high transmission optics to collect sufficient signal from the faint stars.
Spectral range of the spectrometer:	250 nm to 675 nm for UV and VIS 756 nm to 773 nm and 926 nm to 952 nm	Wide spectral range, high transmission optics. Functional pupil separation between UVVIS and NIR. Very strict contamination control to avoid UV sensitivity degradation. Broadband sensitive and low noise detectors with high sensitivity in the UV. High NIR sensitivity.
Spectral sampling	0.3 nm in UVVIS 0.05 nm in NIR	Large sensors (ca. 2,500 used pixels on four sensors each 1,500 pixels wide)

		High dispersion, high efficiency gratings
Spectral resolution	1.2 nm in UVVIS 0.2 nm in NIR	High imaging quality optics Very high pointing system stability requirements
Spectral stability knowledge in dark limb	0.07 nm in UVVIS 0.016 nm in NIR	Star tracker and pointing system
Photometer spectral windows and sampling rate	470 nm to 520 nm and 650 nm to 700 nm 1 kHz sampling rate	Fast (1 kHz), high sensitivity detectors
Short term radiometric stability (over 150 seconds)	1%	Spatially uniform detectors and very high pointing stability
Linearity	1%	High detector and electronic chain linearity (very challenging for the extremes of the dynamic range)
Pointing stability	Better than 40 microradians peak to peak	High speed, high accuracy closed-loop pointing system
Number of occultations per orbit	45 on average	Requirement for the star pointing mechanism in terms of long term performance and reliability
Angular coverage	-10° to +90° with respect to the flight direction.	Total angular travel range for the mechanism

Table 5.19. Envisat optical requirements

These requirements are then developed in the requirements for the actual devices on board. In the following tables the requirements and the solution adopted for the telescope are reported, the front steering mechanism and the spectrometer segments.

TELESCOPE

Parameter	Requirement	Selected technology/design
Field of view	0.6°	Cassegrain design
Optical transmission	> 82% between 250 nm and 500 nm > 92% between 500 nm and 952 nm	Al coated UVVIS subpupil and AG coated NIR subpupil
Intermirror distance stability	better than 10 mm	coefficient of thermal expansion and coefficient of moisture expansion compensated CFRP structure
Imaging quality	UVVIS: 25 mm at 85% encircled energy NIR: 30 mm at 85% encircled energy	Cassegrain design with aspheric primary and secondary mirrors

Table 5.20. Envisat telescope requirements

STEERING MECHANISM

Parameter	Requirement	Selected technology/design
Angular range (optical)	100°	Two-stage (coarse + fine) design
Open loop pointing accuracy	better than +/-0.02° bias and +/-0.01 degrees dynamics	High resolution inductosyn angular sensors (1 LSB = 0.0009 degrees)
Torque resolution	5 micro nm	Voice-coil actuators together with high linearity electronics
Microvibration rejection	3 microradians residual above 10 Hz	Three-axis, frictionless mirror mounting
Number of angular travel cycles	> 1.5 million of 70° average	

Table 5.21. Envisat steering mechanism requirements

SPECTROMETER

Parameter	Requirement	Selected technology / design
Spectral range	250 nm to 950 nm	Thinned, backside illuminated, anti-reflection coated CCD
Quantum efficiency	20% at 250 nm to 350 nm, 60% at 500 nm to 675 nm , 20% at 950 nm	See above
Dark current at 20C	< 25 pA/cm ²	Inverted mode operation (MPP)
Linearity	better than 0.5%	Special design of the output stage
Geometrical design of the CCD	2 × 143 lines and 1353 columns with 20 × 27 micrometre pixels	

Table 5.22. Envisat spectrometer requirements

To complete the paragraph here it is the list of the other experiments whose requirements were not considered:

- **MERIS**, observation of many of Biophysical properties of earth surface.
- **MIPAS**, infrared spectrometer to measure temperature profiles.
- **RA-2**, monitoring of sea surface topography.
- **MWR**, a radiometer.
- **ASAR**, a radar.
- **SCHIAMACY**, measurements of trace gases via spectrometry.
- **AATSR**, measurement of sea upwelling.
- **DORIS**, orbit determination via Doppler effect.
- **LRR**, a passive laser reflector for satellite ranging.

5.5.6 Control requirements

The AOCS provides three-axis stabilization of the satellite body, which remains fixed in the local orbital reference frame. It ensures acceptable pointing and pointing-stability performance for the payload instruments and sensors. Attitude pointing: < 0.1° (3σ); attitude measurement: < 0.03° (3σ).

The AOCS actuators are:

- Sixteen thrusters, 8 allocated to each nominal and redundant actuator chains. They are used for initial attitude control, in- and out-of-plane manoeuvres, and safe mode. The backup thrusters are kept in cold redundancy.
- Five reaction wheels, each with a 40 Nm/s capacity (two wheels on Y- and Z-axis, one wheel on X-axis)

- Four magnetorquers, forming two pairs. They are actuated at the correct satellite orbital phase (depending on their alignment with the Earth's gravity field) to control the maximum spin rate of the reaction wheels on the relevant axis.
- Four tanks containing 300 kg of hydrazine for initial orbit acquisition and orbit-maintenance activities.
- A solar array drive mechanism (MEGS) to control the solar array's rotation (1 rotation per orbit). At a second level, the MEGS is also controlled to compensate for the solar array perturbations due to flexible modes, which are detected by the gyroscopes. Note: MEGS is of the SPOT platform family heritage.

Mode of Operation		Comments	Sensors and Actuators
Operational Mode	YSM or SYSM	Operational mode with fine satellite pointing	Reaction wheels with magnetic torquers YSM: 1 Digital Earth Sensor 1 Digital Sun sensor and 2 gyros SYSM: 2 SSTs and 2 gyros
	Geocentric pointing, correction for the local normal pointing, and yaw steering bias		
Orbit Control Mode	OCM	Out-of-plane orbit maintenance manoeuvres. $\Delta V > 0.05$ m/s along Y-axis after 90 deg rotation around Z -axis. Payload mission interrupted.	1 Digital Earth sensor 2 gyros 8 thrusters
	FCM or SFCM	In-plane orbit-maintenance manoeuvres. Mission not interrupted	8 thrusters FCM: 1 Digital Earth sensor 1 Digital Sun sensor and 2 gyros SFCM: 2 SSTs and 2 gyros
Satellite Safe Mode	SFM	+Z Sun pointing $0.4 \text{ deg/s} <$ Satellite rate $< 0.65 \text{ deg/s}$ TM/TC and AOCS control through T4S PROM software.	T4S PROM software-controlled 8 thrusters 2 SAS (+Z and -Z) 2 gyros
Attitude Re-acquisition (exit from safe mode)	RRM->CAM>FAM	Earth re-acquisition by Earth sensors Payload Module put into safe mode	8 thrusters 1 digital Earth sensor 1 digital Sun Sensor 2 gyros

Table 5.23. Envisat attitude control requirements

The prime satellite AOCS operational mode is stellar yaw-steering, deriving the satellite attitude from the SSTs (Stellar Star Trackers) and the satellite rates from the gyroscopes. Commands are sent to the SST sensors using a ground-based star catalogue tuned to the spectral characteristics of the SST optics. The time of detection of the commanded star, and its position within the SST field of view allow the onboard software to derive an attitude measurement for the satellite.

5.6 IXPE

The Imaging X-ray Polarimetry Explorer (IXPE) is a space-based observatory that will have the capability to measure the polarization of X-rays from astrophysical sources. IXPE will allow astronomers to explore, for the first time, the hidden details of some of the most extreme and exotic astronomical objects such as neutron stars as well as stellar and supermassive black holes. IXPE measurements will provide new dimensions for probing a wide range of cosmic X-ray sources, including active galactic nuclei (AGN) and micro quasars, pulsars and pulsar wind nebulae, magnetars, accreting X-ray binaries, supernova remnants, and the Galactic centre.

IXPE's payload is a set of three identical, imaging, X-ray polarimetry systems mounted on a common optical bench and co-aligned with the pointing axis of the spacecraft. Each system, operating independently, comprises a 4 m focal length Mirror Module Assembly (grazing incidence x-ray optics) that focuses X-rays onto a polarization-sensitive imaging detector. The focal length is achieved using a deployable boom. Each DU (Detector Unit) contains its own electronics, which communicate with the payload computer that in turn interfaces with the spacecraft. Each DU has a multi-function filter wheel assembly for in-flight calibration checks and source flux attenuation.

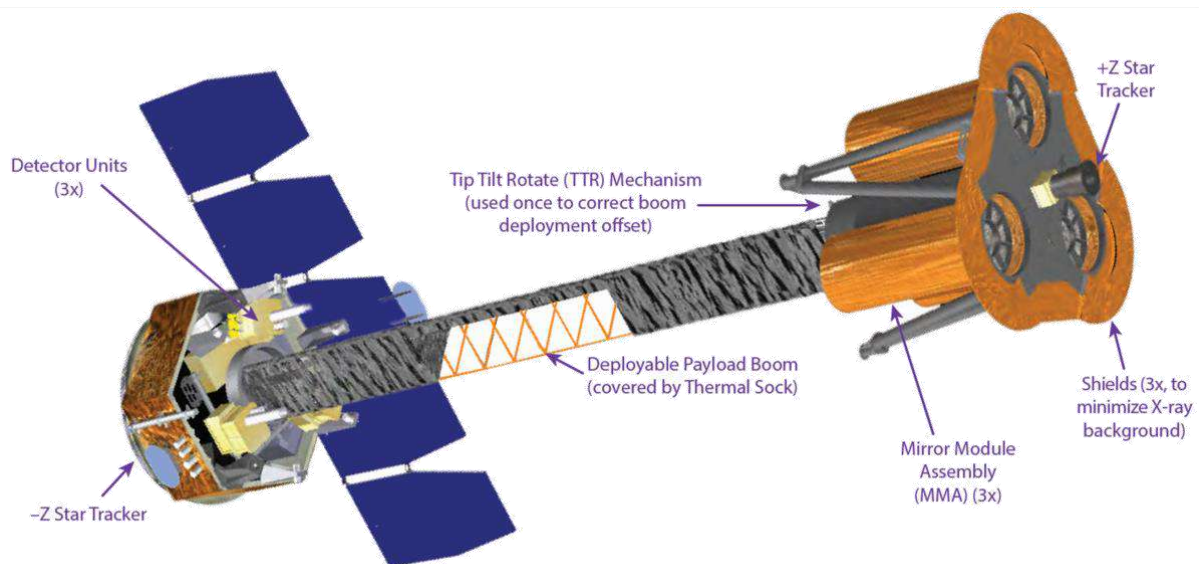


Figure 5.14. IXPE representation

This telescope will operate in LEO at an altitude of 600 km in a geocentric orbit with 0.2° of inclination. Typical requirements, in terms of mechanical, thermal, control and optical corrections, for this kind of science observation missions are analysed as follows.

Mission	
Mission	Imaging X-ray Polarimetry Explorer
Orbit	LEO
Orbit Altitude	600 km
Inclination	0.2°
Launch mass	4800 kg
Launch Vehicle	Falcon 9

Table 5.24. IXPE mission requirements

5.6.1 Control requirements

The Attitude Determination and Control System (ADCS) provides a 3-axis stabilized platform controlled by reaction wheels and torque rods. The primary attitude sensor is a pair of star tracker optical heads augmented by a magnetometer and sun sensors. They are mounted directly to and aligned with the deployed payload to minimize alignment errors between the spacecraft and payload. GPS is used for timing as well as spacecraft orbital ephemeris. A metrology system, with a camera is mounted on the underside of the MMSS (Metallic Mirror Module Support Structure) which images a metrology target (diode string) on the spacecraft top deck; it is used to monitor motions between the two ends of the Observatory during science observations. A tip/tilt/rotate mechanism allows on-orbit adjustability between the deployed X-ray optics and the spacecraft top deck-mounted DUs (Detector Units), providing system tolerance to variations in deployed geometry.

If on deployment, the X-ray image is not within the required position range of the detector centre point, the X-ray image can be re-aligned by using the TTR mechanism, while observing a bright source. Note that all three MMAs are moved in unison. This is possible because the forward star tracker is mounted with the optics, so that this adjustment effectively re-aligns the pointing axis with the new payload axis. Co-alignment of the individual MMAs with respect to each other and the star tracker, is performed during payload integration and test.

The Observatory is designed to support IXPE measurement requirements. Key design drivers include pointing stability in the presence of various disturbances, particularly gravity gradient, and minimization of SAA passes which makes the zero-degree inclination orbit the best available choice.

Control	
Stabilization method	3-axis stabilized controlled by reaction wheels and torque rods
Attitude control	40 arcsec (3σ) – x- and y-axis – Point State
LOS co-alignment accuracy x-axis	19.8 arcsec (3σ) [capability 9.5 arcsec (3σ)]
LOS co-alignment accuracy y-axis	26.7 arcsec (3σ) [capability 12.8 arcsec (3σ)]
Pointing method	Gravity Gradient
LOS pointing accuracy	53.1 arcsec (3σ) [capability 25.2 arcsec (3σ)]

Table 5.25. IXPE attitude control requirements

5.6.2 Optic requirements

Optimized for polarimetry at 2–8 keV, IXPE will measure the (linear) polarization of various types of sources as indicated above. The x-ray optics comprise 3 mirror module assemblies (MMAs, 3), with optical axes co-aligned to the Spacecraft's +Z star tracker. These are mounted on the mirror-module support structure (MMSS), which is skirted by an x-ray shield to preclude direct illumination by celestial sources of the focal planes of the collimated detector units (DUs). The x-ray instrument comprises 3 polarization-sensitive detector units (DUs), each aligned to its respective MMA node, plus a detectors service unit (DSU) that provides control and data handling for the DUs. Each DU is mounted through an interface plate to the top side of the Spacecraft top deck, while the DSU is mounted to the bottom side of the top deck. A deployable boom serves as the telescope system's optical bench, enveloped by a thermal sock to minimize thermally induced distortions. Connecting the boom to the MMSS centre tube is a Tip–Tilt–Rotate (TTR) mechanism, which enables collective alignment of the DUs to the respective MMA nodes during the commissioning phase.

MMA specifications

Each of the 3 IXPE MMAs (plus 1 spare MMA) utilize 24 electroformed nickel–cobalt shells, replicated from precision machined and polished metal mandrels. The principal components of the MMA are the front spider, central support tube, mirror shells (inner to outer), rear spider, outer housing, and thermal shields. The front spider is the primary structural element of the design, through which the MMA will mount to the MMSS during Payload integration at Ball. The rear spider has 18 spokes, each with a metal comb. After all 24 shells have been glued to the front spider, the rear spider is attached to central support tube and its metal combs aligned such that each mirror shell floats within the corresponding slot between the tines of the rear-spider combs. Unlike the front-spider combs, which hold the mirror shells, the rear-spider combs merely limit excursions of the shells under launch loads to preclude shell-to-shell collisions.

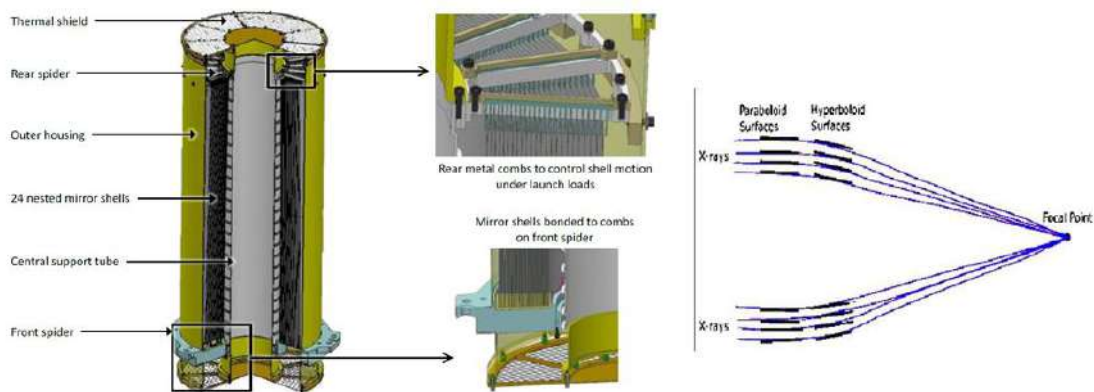


Figure 5.15. MMAs configuration (left) and mirrors operation (right)

Figure 5.16 summarizes the configuration and performance specifications of the IXPE mirror module assemblies. Each of the nested nickel–cobalt shells include the primary and secondary grazing-incidence mirror surfaces to achieve a 4001-mm focal length. As the x-ray optical properties of nickel are nearly ideal for IXPE’s 2–8-keV band, there is no need for a separate optical coating on the mirrors.

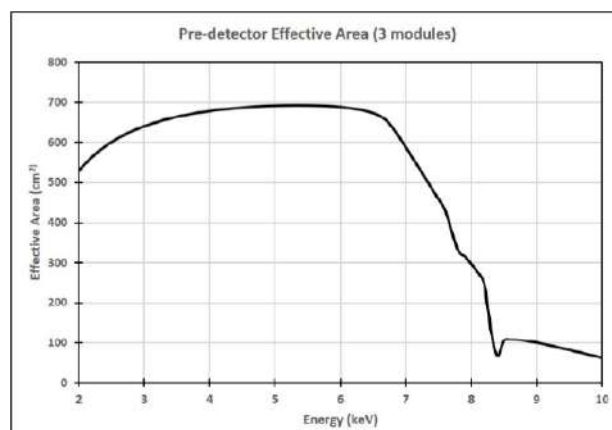


Figure 5.16. Configuration and performance specification of the IXPE Mirror Module Assemblies. Left panel tabulates various design and performance parameters. Right panel displays the expected effective area of 3 MMA optics, including attenuation by the thermal shields but not the efficiency of the detectors.

The principal optical-performance requirements on the MMAs are the effective area (AMMA) and the half-power diameter (HPDMMA). Current best estimates of AMMA and of HPDMMA support the level-1 requirements on polarization sensitivity and on imaging resolution, respectively. Calibration of IXPE’s optics is done in ground facilities before launch. Using its 100-m x-ray test facility, MSFC will calibrate each MMA using GSE detectors and sources at several x-ray energies, measuring MMA effective area and point spread function on axis and at several off-axis angles, ghost (non-imaged) x rays, and focal length. While calibration of a Telescope (MMA+DU) can be synthesized from separate MMA and DU calibrations, at least one MMA and DU will be calibrated together to validate the synthesized response. This validation will measure the Telescope point spread function and effective area for a subset of energies and off-axis angles used for the analogous MMA measurements with GSE detectors.

DU specifications

The dominant interaction process for a few keV photons is the photoelectric effect, which is in principle a perfect analyser for X-ray polarization measurements. In fact, the direction of emission of the photoelectron for a K-shell absorption and 100% linearly polarized incident radiation, is 100% modulated around the polarization direction with a $\cos^2$ function of the azimuth angle. Therefore, this process is well suited for the energy window below 10 keV, where the source fluxes are relatively high, and the grazing-incidence optics provide reasonable efficiency. This effect has been exploited with the *Gas Pixel Detectors* (GPDs).

The Gas Pixel Detector (GPD) proposes a solution based on the usage of an *Application-Specific Integrated Circuit* (ASIC) as a finely pixelized collecting anode of a proportional gas detector, and a *Gas Electron Multiplier* (GEM) as amplification stage. This concept is shown on the left side of Figure 4: a photon is absorbed in a gas gap emitting a photoelectron (and, possibly, an Auger electron), the tracks ionize the gas and an electric field allows electron/ion pairs to drift respectively to the GEM and top plane. The differential voltage on the GEM induces electron multiplication and the pixelized anode allows a detailed imaging of the track. The ASIC and the GEM pixel pitch can be small enough to permit a good sampling of the short photoelectron tracks. In the current generation both the readout and amplification stage exploit a hexagonal pixel pattern with only 50_μm pitch. On the right side of Figure 5.17 some performance specification is shown.

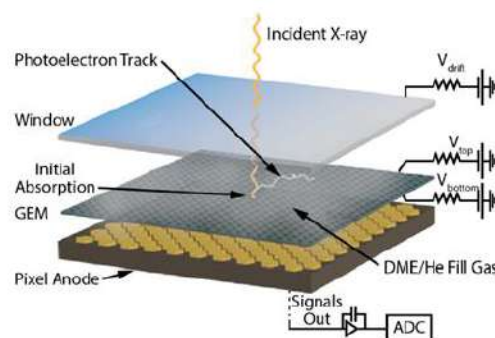


Figure 5.17. GPDs configuration (left) and performance specifications (right)

Optical	
Number of mirror modules	3
Mirror shells per module	24
Focal length	4.00 m
Total shell length	600 mm
Shell diameter	Inner: 162 mm Outer: 272 mm
Shell Thickness	Inner: 178 μm Outer: 254 C

Shell material	Nickel cobalt alloy
X-ray optical coating	None (bare NI-Co)
Effective area per module	183 cm ² (2.3 keV) > x > 210 cm ² (3.6 keV)
Angular resolution	≤ 25 arcsec HPD
Detector limited FOV	12.9 arcmin
Mass (3 assemblies)	95 kg*w/contingency
Modulation factor	20% (70%) at 2 (8) keV
Residual modulation	< 0.5 %
Spatial resolution	90 μm at 5.9 keV
Energy resolution	< 20% at 5.9 keV
Timing resolution	≈ 10 μs

Table 5.26. IXPE optical requirements

5.6.3 Thermal requirements

- **Mirror temperature variation rate:** the mirror temperature variation rate in whatever operating phase must be lower than 10 °C per hour, so as not compromise mirror life.
- **Detector and mirror Absolute temperatures:** the detector and mirror absolute temperatures must be between 5°C and 25 °C.
- **Maximum thermal gradient:** the maximum thermal gradient on the mirror must be 1 °C both in the longitudinal direction and in the azimuth direction.
- **Maximum launch temperature:** the structure of the IXPE shall withstand a maximum temperature of 94°C for 250 seconds.

Thermal	
Mirror temperature variation rate	< 10 °C/h
Detector absolute temperature	5 °C
Mirror absolute temperature	25 °C
Maximum thermal gradient	1 °C (longitudinal and azimuth direction)
Maximum launch temperature	94 °C for 250 seconds

Table 5.27. IXPE thermal requirements

5.6.4 Mechanical requirements

- **Natural frequencies of vibration:** the fundamental bending mode of the IXPE shall be greater than 10 Hz, and fundamental axial mode greater than 25 Hz using Falcon 9. This places a lower limit on the manifestation of the proper modes of vibration of the engine.
- **Quasi-static loads:** the structure must withstand a maximum load factor of 6 g along the axial direction and 2 g along the lateral direction.
- **Payload mass:** payload mass shall be smaller than 4800 kg (worst case scenario).
- **Encumbrance:** the dimension of the IXPE folded up shall be smaller than 5,3 m along lateral direction and 6,6 m along z axis.
- **Maximum Allowable Acoustic Sound Pressure Level (OASPL):** the Overall Sound Pressure Level (OASPL) is the total energy contained in the sound spectrum and shall be greater than 139.6 dB.

Mechanical	
Natural frequency	Lateral: 10 Hz Longitudinal: 25 Hz
Quasi-static Load	Lateral: 2 g Longitudinal: 6 g
Encumbrance	5.3 x 5.3 y 6.6 z
OASPL	139.6 dB
Traceability	Falcon 9 Launch Vehicle Payload User's Guide

Table 5.28. IXPE mechanical requirements

6 CONCEPTUAL DESIGN: COLLAPSIBLE BOOMS

6.1 BOOM REQUIREMENTS

In the preliminary phase, two categories of requirements have been identified: geometric and material requirements. Both are enlisted in the following sections.

6.1.1 Geometric requirements

- Deployed boom length ≥ 3860 mm
- Arc angle $\geq 65^\circ$
- Arc radius < 20 mm
- Flat tab l $< 0.5 \cdot$ Arc radius
- Thickness < 0.2 mm
- Stowed volume $< 1.1 \cdot 10^5 \text{ mm}^3$
- Total number of wraps of the coil ≤ 13
- External coiled radius ≤ 53 mm
- Internal coiled radius ≤ 50 mm

The first requirement was calculated through the measurements of the distance between two mirrors and the difference between their diameters, fixed by the statement of work written by ESA. Second and third are fixed by statistical analysis of data found in the literature and choosing a point ($\kappa = \frac{1}{\text{Arc radius}} = 0.05 \text{ mm}^{-1}$, Momentum = 250 Nmm) that guarantees a torque below the maximum one according to the Moment-Curvature chart.

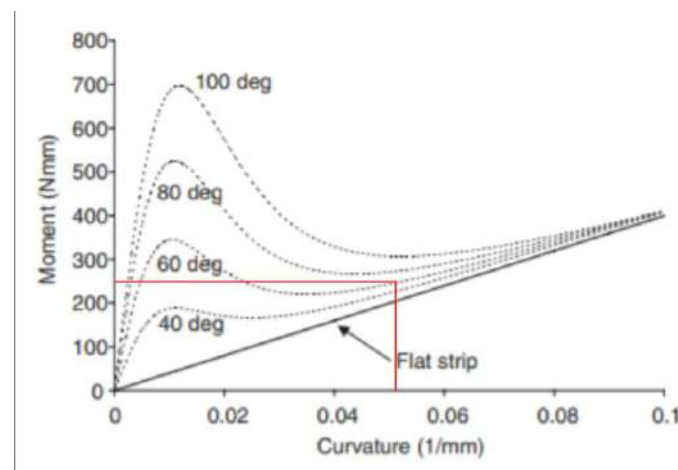


Figure 6.1. Momentum-Curvature chart

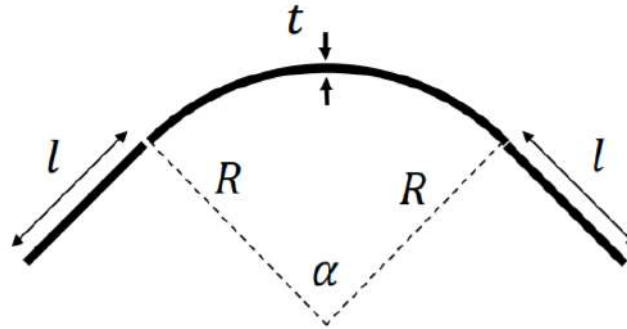


Figure 6.2. Cross-section of the tape-spring.

Likewise, the fifth requirement was selected by statistical analysis of similar structures. Then, from sixth to ninth requirement are calculated with the following procedure, taken from “Advanced Deployable Shell-Based Composite Booms for Small Satellite Structural Applications Including Solar Sails” by J. M. Fernandez.

```
t_sh = 0.2; %[mm] Thickness of the thin shell wall
l=3870; % length [mm]
mu = t_sh/100*25; %packaging efficiency
eps = 0.2/100; % elongation
n=1;% number of layers
t_t = n * t_sh * (1+mu); % total thickness of all the co-reeled booms
r_i = 0.5*t_sh/eps; %inner radius[mm]
w = sqrt((r_i/t_t)^2 + 1/(pi*t_t)) - r_i/t_t; %number of wraps
r_f = r_i + w*t_t; %outer radius [mm]
```

Figure 6.3. Matlab script for geometrical requirements

6.1.2 Material requirements

- Low susceptibility to creep/relaxation along the fibre direction
- Low transverse and shear stiffness
- Stowage creep effect should produce $\leq 30\%$ boom cross-section flattening
- Operational life ≥ 5 years
- Elongation $< 2\%$
- Coefficient of thermal expansion in the boom axial direction $\leq -0.5 \cdot 10^{-6} \frac{m}{m^{\circ}C}$
- Young modulus: $135 \text{ GPa} \leq E \leq 160 \text{ GPa}$ (single layer)
- Density: $1500 \frac{kg}{m^3} \leq \rho \leq 1700 \frac{kg}{m^3}$
- Mass of each boom $\leq 0.08 \text{ kg}$

In the case of material requirements, from the first to the sixth were selected considering the characteristics of existing telescopes. Then, from the seventh to the ninth were chosen considering the typical characteristics of space application's materials. Finally, the mass of the boom is computed by means of the following script:

```
Arc_radius=20; %[mm]
alpha=65; %[deg]
Arc=Arc_radius*(alpha*pi/180); %[mm]
ltot=Arc/2+Arc/2+Arc; %Total Width
Rex=53; % [mm] Outer Radius
Vol=pi*(Rex^2)*ltot % External Volume
R_int=50; %[mm] Inner Radius
Vol_int=pi*(R_int^2)*ltot; %Internal Volume
Vol_tapespring=(Vol-Vol_int)*10^-9 %[m^3] Coil Volume
rho=1600; % [kg/m^3] Material Density
m=Vol_tapespring*rho %[kg] Boom's Mass
```

Figure 6.4. Matlab script for material requirements.

6.1.3 Boom material selection

MATERIAL	PLOT NAME	% Resi na	Rho [Kg/m 3]	Cured Thickness [mm]	Tensile Strength [MPa]	Tensile Modulus [GPa]	Compression Strength [MPa]	Compression Modulus [GPa]
T3T095-12"-F155	T3-F155	38,4	1609,00	0,094	1552	126,2	1158	109,6
T3T095-12"-F263	T3-F263	37	1590,00	0,1	1247	134,8	1420	
M78.1/39%/U D120/CHS	M78-CHS	39	1530,00	0,13	2400	128		
8552-IM7-35%-134	8552-IM7	35	1570,00	0,13	2723	164	1689	109,6
M20/34%/134 /IM7	M20-IM7-12K	34	1570,00	0,13	2790	175		
M56/35%/UD 134/AS7-12K	M56-AS7	35	1530,00	0,134	2330	141	1308	129
T2C145-12"-F155	T2-F155	39	1590,00	0,14	1834	119	1331	117
T2T145-12"-F655	T2-F655	33,37	1545,00	0,14	1517	142,7	1671	
M48/35%/150/IM7	M48-IM7	35	1570,00	0,147	2895	167	1693	153
T3T145-12"-F263	T3-F263	35,4	1598,00	0,15	1362	132,3	1333	109,6

M35-4/38%/UD150/CHS/460	M35-4-CHS	38	1570,00	0,15	2000	142	1500	
M21/34%/UD194/AS7-12K	M21-AS7	34	1580,00	0,15	2350	148	1560	123
T7G145-12"-F584	T7-F584	35,39	1603,00	0,15	2337	156	1585	134
T9A145-12"-F584	T9-F584	35,39	1603,00	0,15	2799	156	1654	155
M18/1/42%/G947	M18-G947	38	1530,00	0,16	2400	130	373	121
M10R/38%/UD150/CHS	M10R-CHS	38	1570,00	0,16	2600	140	1200	
M91/34%/UD194/ AS7-12K	M91-AS7	34	1580,00	0,18	2580	142	1350	123
M21/34%/UD194/IMA-12K	M21-IMA-12K	34	1580,00	0,184	2860	160	1790	148
M21/34%/UD194/ IM7-12K	M21-IM7	34	1580,00	0,184	2860	160	1790	148
M91/34%/UD194/IM7-12K	M91-IM7	34	1570,00	0,184	2980	165	1860	150
M91/34%/UD194/IM10-12K	M91-IM10	34	1580,00	0,184	3520	176	1880	156
T2C190-12"-F155	T2-F155	38,4	1609,00	0,19	1518	124,8	1172	109,6
8552-AS4-35%-134	8552-AS4	34	1580,00	0,19	2200	141	1500	124
M47/38%/UD150/T700	M47-T700	38	1540,00	0,16	2750	140	1450	

Thanks to an Ashby map, it was possible to compare these materials in order to find the best one for our purpose. The metrics used to assess the materials for this map are the same described in the task 1.2 (page.53, Fig 2.11).

Materials from Hexel were chosen because it is a company specialized in space grade materials. It is important to take into account the fact that the material can't outgas (it can damage the mirror's performance), and the thermal expansion coefficient needs to be as low as possible (to avoid image).

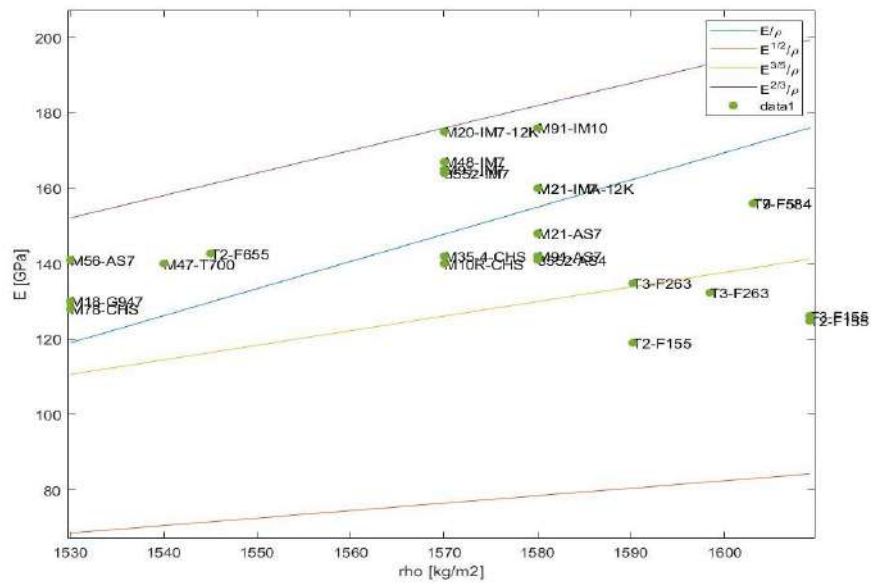


Figure 6.18. Young modulus vs density map

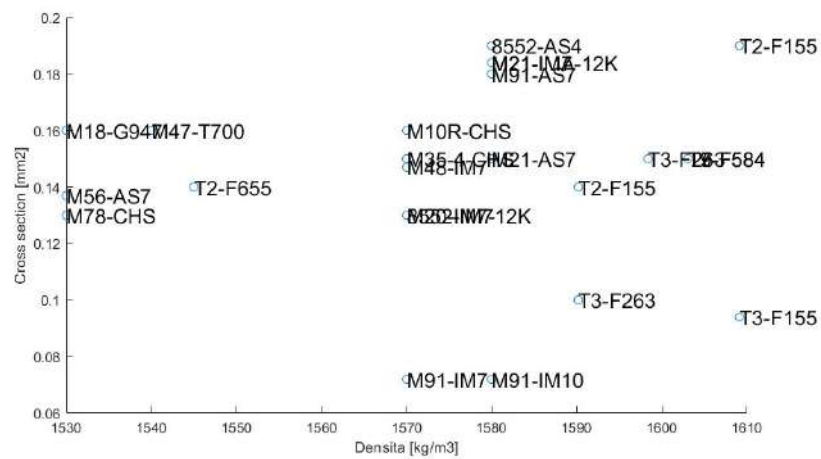


Figure 6.19. Cross section vs density map

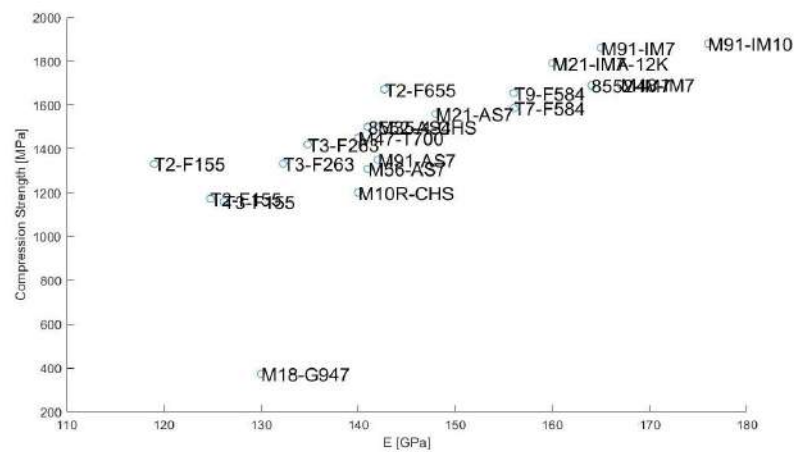


Figure 6.20. Compression strength vs Young modulus

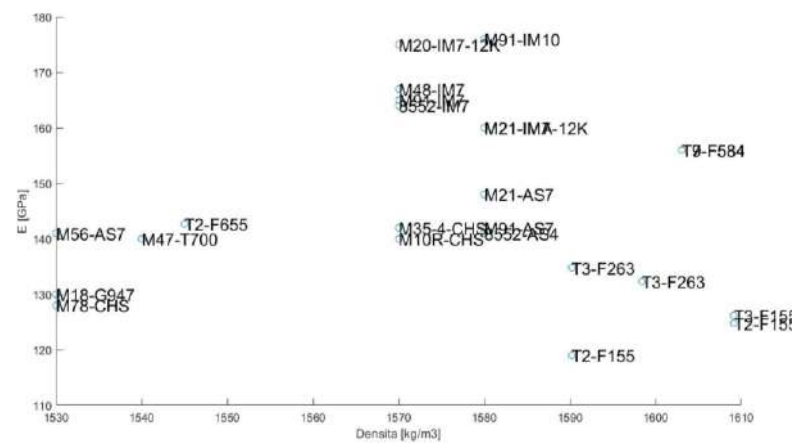


Figure 6.21. Young modulus vs density

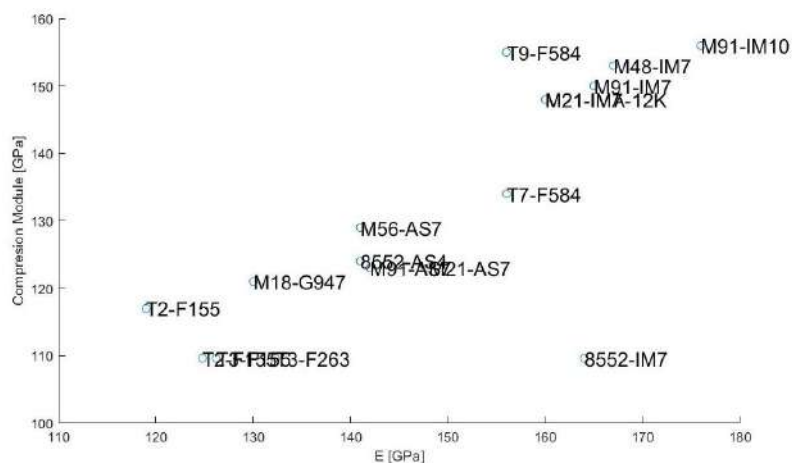


Figure 6.22. Compression modulus vs Young modulus

MATERIAL	%Resin	Density [kg/m³]	Cured Thickness [mm]	Tensile Strength [MPa]	Tensile Modulus [GPa]	Compression Strength [MPa]	Compression Modulus [GPa]
M56/35%/UD134/AS7-12K	35	1530,00	0,137	2330	141	1308	129
T2C145-12"-F155	39	1590,00	0,14	1834	119	1331	117
M20/34%/134/IM7	34	1570,00	0,13	2790	175	~	~

From these graphs it was possible to pinpoint three materials to be further analyzed with the finite element method. These have the best performance in terms of Tensile strength, tensile modulus and compression modulus. This prepregs have less density than the other ones (in the table above) and also a minimum cured thickness. A range of resin from 30 % to 40 % allows these materials to have excellent bending capability, important for the stowed configuration. A density range around 1500

Kg/m³ allows these materials to be very light, this is an important factor that will decide the material to be chosen in this first conceptual design.

By pondering pros and cons of these materials, the **Hexcel M56/35%/UD134/AS7-12K** was chosen, thanks to the low density and cross-sectional area while maintaining an optimum Young's modulus (neither too high nor too low), compression module and compression strength (further analyzed using the FEM results).

This prepreg consists of:

- **M56:** high-performance epoxy matrix;
- **UD134:** unidirectional carbon fiber tape, where 134 stands for the gsm (grams per square meter) of the carbon fiber tape.

The M56/35%/UD134/AS7-12K is made from resin (M56), reinforcing fibers (AS7-12K) and UD carbon tape (UD 134) that is a unidirectional carbon fiber. This product formats are designed to optimize fiber volume.

M56 is a high-performance epoxy matrix developed for out-of-autoclave curing of composite aircraft structures, with a matrix thinner than equivalent laminates cured under vacuum pressure only. This product is suitable for ATL (automated tape laying) and AFP (advanced fiber placement) processing.

6.1.3.1 Resin matrix properties

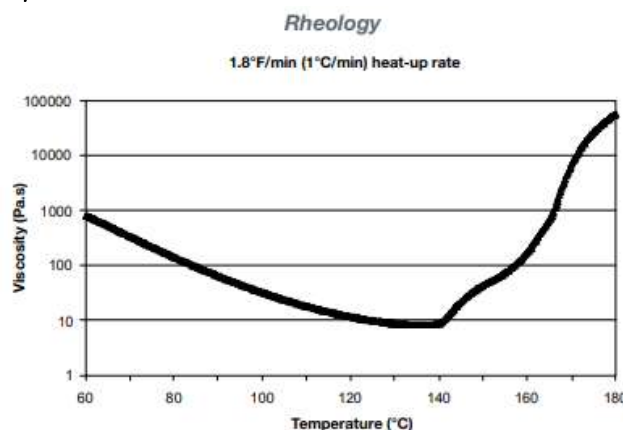


Figure 6.23. Viscosity - Temperature

6.1.3.2 Cure cycle viscosity profiles

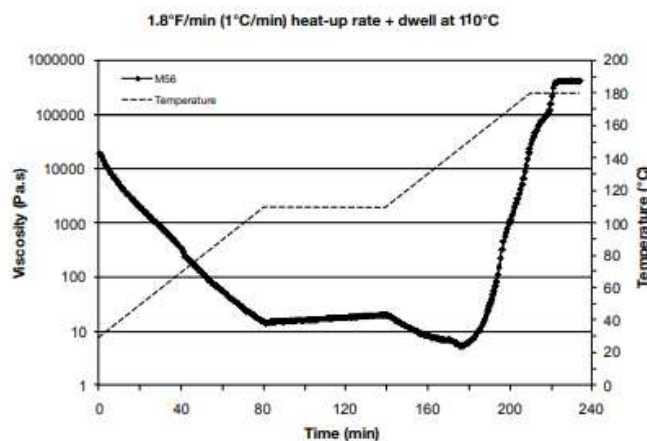


Figure 6.24. Viscosity - Time

6.1.4 Prepreg lay-up

To achieve the best monolithic laminate quality, the first ply should be vacuum debulked compacted to the mold cure tool at room temperature. The frequency of compaction depends on part size and complexity, but a good rule of thumb is to vacuum compact for 5 to 10 minutes every subsequent 3 plies to ensure removal of air trapped during the lay-up process.

During vacuum compaction of lay-up and for final laminate during cure, it uses a rigid edge breathing support (such as cork, high tack-high temperature sealant tape, cured silicone rubber or metal) to protect the plies at the edge of the part from pinching off and preventing air evacuation. The edge breathing support should be no more than 1/8 inch higher than the edge of the laminate panel.

6.1.4.1 Prepreg curing conditions

1. Apply full vacuum (-28 inches Hg, -0.95bar)
2. Ramp at 2 - 5°F/min (1 - 3°C/min) to 230± 10°F (110 ± 5°C)
3. Hold at 230± 10°F (110 ± 5°C) for 60 ± 5min
4. Reduce vacuum to -15 inches Hg (-0.5bar)
5. Ramp at 2 - 5°F/min (1 - 3°C/min) to 355 ± 10°F (180 ± 5°C)
6. Hold at 355 ± 10°F (180 ± 5°C) for 120 + 30, - 0 min
7. Cool at ≤ 5°F/min under vacuum
8. Release vacuum when laminate temperature is below 140°F

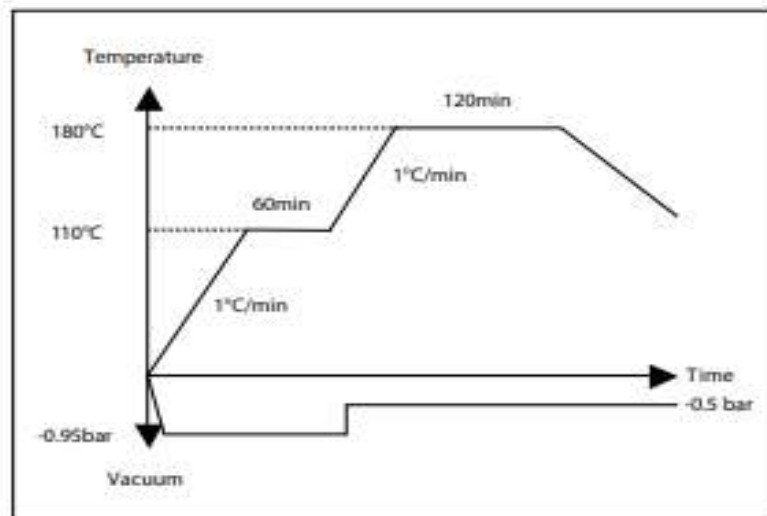


Figure 6.25. Temperature - Time

For maximum storage life, HexPly® M56 prepreps should be stored continuously at -18°C (0°F) for 18 months. After removal from cold storage, prepreg should be allowed to reach room temperature before opening the polythene bag to prevent condensation (a full roll in its packaging can take up to 48 hours).

The system of cure will be slow, at room temperature, so as to prevent reactions that reduce unbastability, drapability and out time, but fast enough to high temperatures to ensure reasonably short cure times.

The tack life, which is the time, at room temperature, during which prepreg retains enough tack for easy component lay-up, and it's about 30 days at 23°C (73°F).

After impregnating the fiber with the resin, the prepreg is stored at a low temperature. The material is heated to room temperature to carry out the lay-up.

About fiber cost, the greater the number of filaments per tow, the lower the cost. For the resin case, the lower the temperature performance, the lower the cost, and finally for the prepreg cost, the larger the product, the lower the cost.

This prepreg has been tested for aerospace application that has assessed the feasibility of using this material in advanced fields.

In order to get the best performance out of the prepreg, a **lamination +45/-45/+45** was chosen, therefore **the final thickness of the boom material will be around 0.41 mm**. This lamination configuration is commonly used in boom structures since it gives the best performance for axial and transverse loads while avoiding the micro buckling in the rolled-up configuration. This selection is preliminary since a more thorough analysis is needed. Once the effective structural and deployment loads are known, it will be possible to define the material and the lamination that fulfill the requirements.

6.2 FINITE ELEMENT ANALYSIS OF A TAPE SPRING STRUCTURE

The aim of this section is to evaluate the stiffness of the tape spring structure made of three different composite materials, such as **M56/35%/UD134/AS7-12K** (M56), **T2C145-12"-F155** (F155) and **M20/34%/134/IM7** (M20).

The selection of the geometry of the booms has been made by the criteria exposed beforehand, and are reported below. The thickness is 0.402 mm and the length is 170 mm.

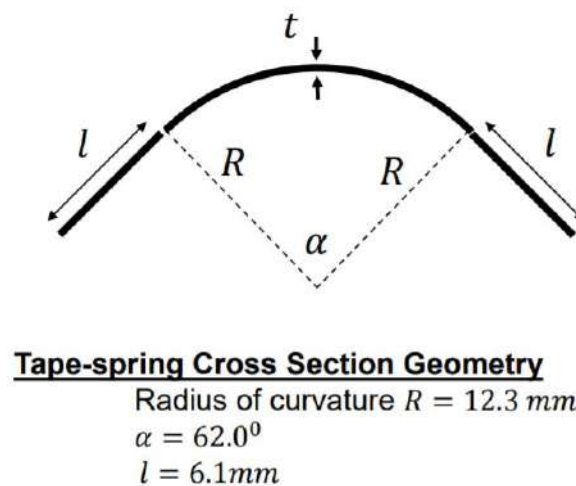


Figure 6.26. Tape spring section geometry

6.2.1 Finite element mesh

The geometry has been created in *Patran-Nastran* and discretized by using shell elements. The long sides are characterized by 30 elements, while on the short ones there are 16 elements, of which 8 on the arcs of circle and 4 on each straight segment.

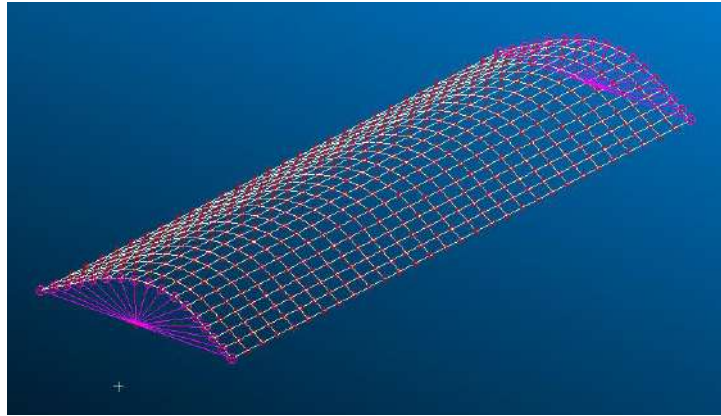


Figure 6.27. Mesh of the tape spring

The analysis has been performed to simulate the behavior of a deployed boom charged by a torque that makes the boom fold. In order to apply the loads and the constraints, two master nodes have been created and located in the geometric center of the tip cross sections, while the nodes on the arc are the “slave nodes”. A carriage rail has been applied in the master loaded node and a hinge has been located on the other one.

A 1 Nmm torque was applied in order to evaluate the boom's stiffness. This force is not the one that the booms will endure during the mission, but it is a simple approach that allows to perform an analysis without the knowledge of the real loads. This unit force also allows to perform a linear approach, further simplifying the analysis.

By measuring the rotation θ of the loaded master node, it is possible to calculate the stiffness of the tape spring, where:

$$D = \frac{M}{\alpha}$$

$$\alpha = 2 \cdot \theta$$

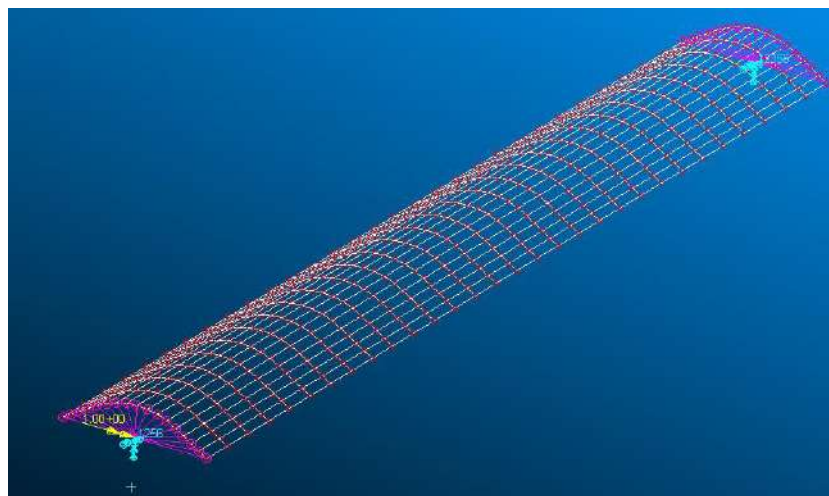


Figure 6.28. Applied torque and constraints

6.2.2 Numerical results

The results of the analysis are reported below:

MATERIALS	M56		M20		F155	
LOADS	M+	M-	M+	M-	M+	M-
ROTATION [rad]	$3,27 \times 10^{-4}$	$-3,27 \times 10^{-4}$	$2,6 \times 10^{-4}$	$-2,6 \times 10^{-4}$	$2,64 \times 10^{-4}$	$-2,64 \times 10^{-4}$
STIFFNESS [Nmm]	1529,05	1529,05	1923,08	1923,08	1893,94	1893,94

From these results it is possible to see that the Prepreg M56 has the biggest rotation with respect to the other materials, with a maximum of $3,27 \times 10^{-4}$ rad, while the others are in the range of $2,6 \times 10^{-4}$ rad. However, M56 has a lower stiffness, this is an advantage since the material needs to be bended before being deployed, therefore reducing the chances of suffering from micro buckling. The Cross-section of the M56 is the biggest with 0,134 mm, but a considerably lower density makes this material better suited for a space structure.

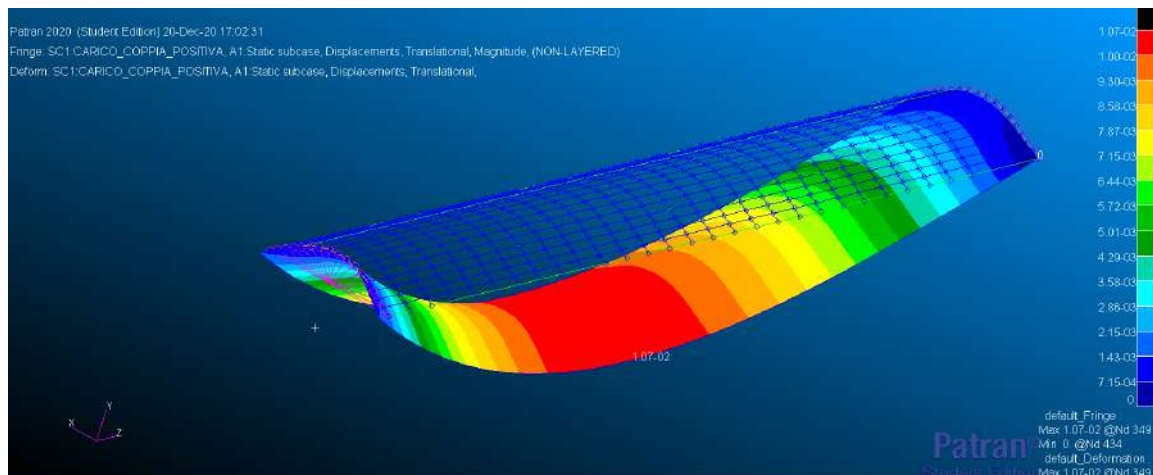


Figure 6.29. Translational displacements-M56 positive torque

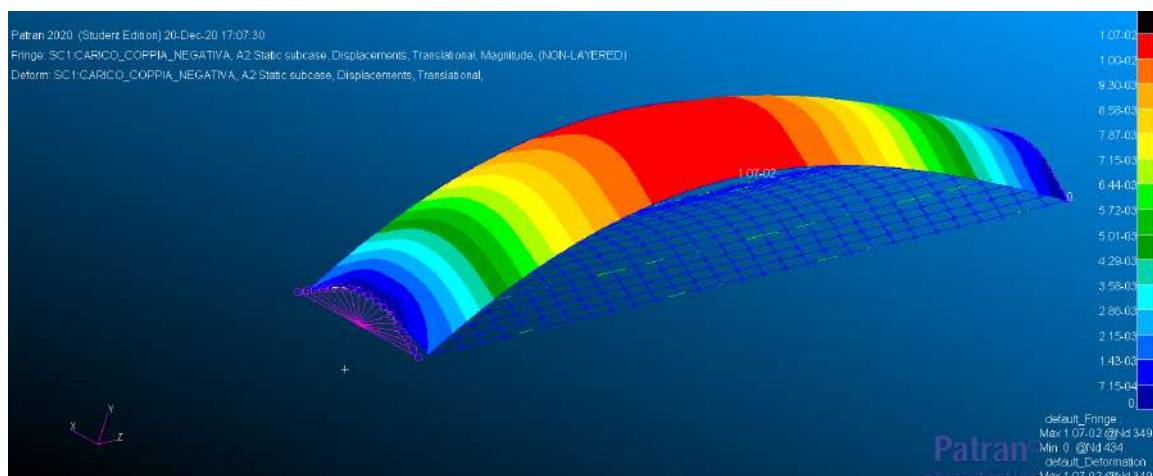


Figure 6.30. Translational displacements-M56 negative torque

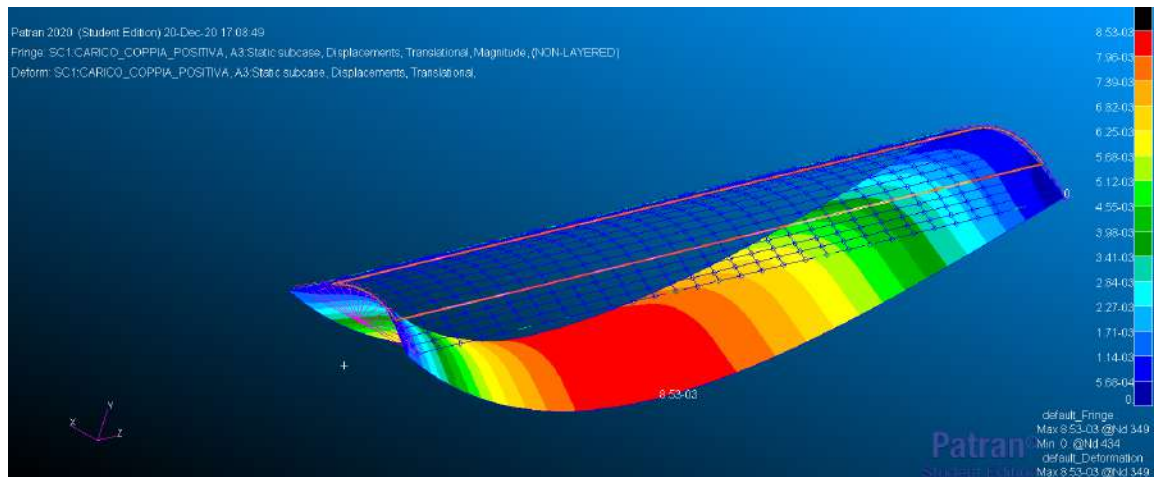


Figure 6.31. Translational displacements-M20 positive torque

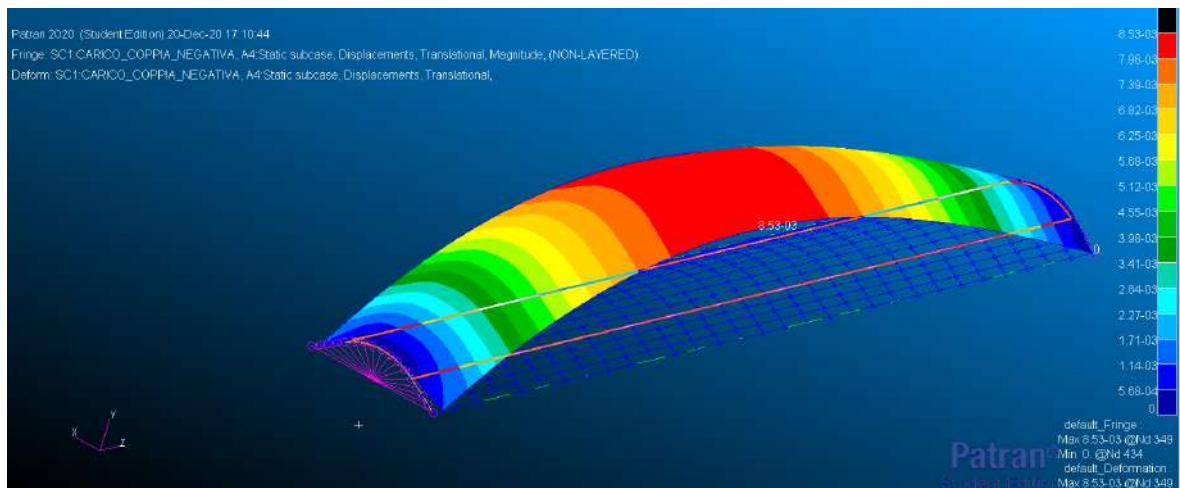


Figure 6.32. Translational displacements-M20 negative torque

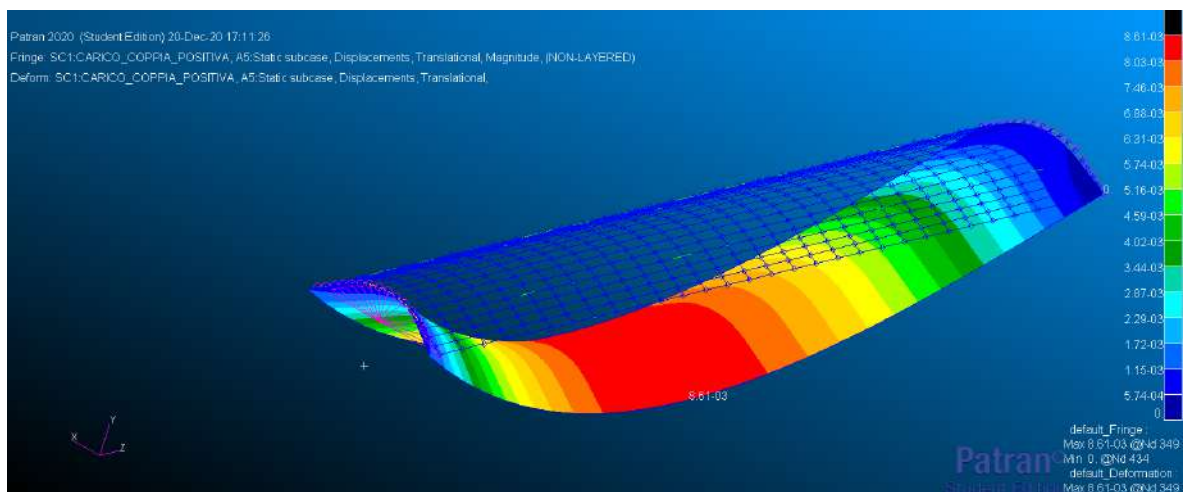


Figure 6.33. Translational displacements-F155 positive torque

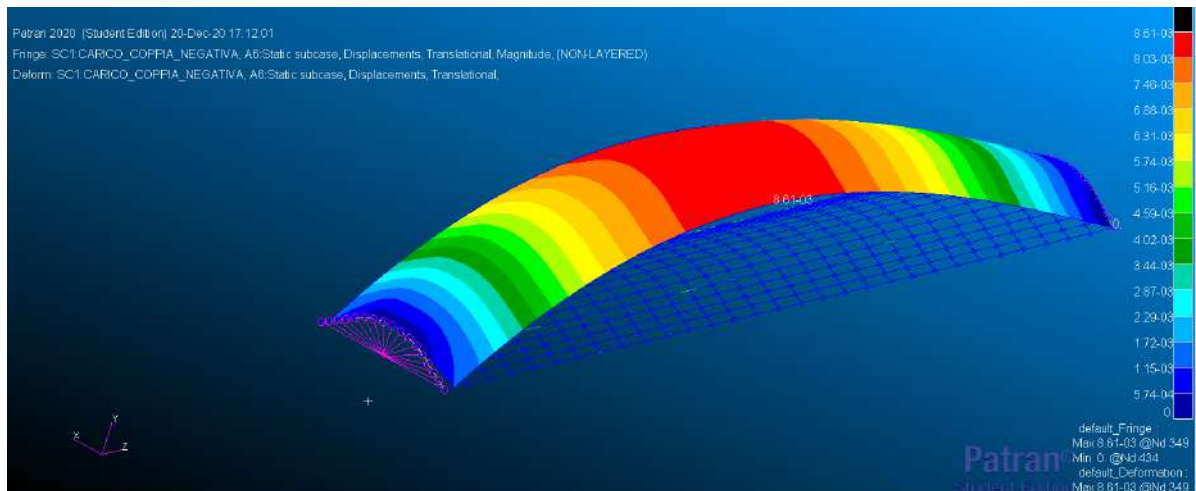


Figure 6.34. Translational displacements-F155 negative torque

The results from this linearized approach are very similar since the three materials chosen are carbon composites Prepregs with similar properties. From the figures is possible to compare the **displacements**, where the M56 has a bigger displacement with a maximum of $1,07 \times 10^{-2}$ at the center point of the inner curve. The displacement for the M20 and F155 are 25% lower than the M56 under the same load. Then, when the tape springs are under a 1 Nmm bending torque (linear) the positive and negative moments give the same displacements.

Once the effective forces and torques are known, a new study is needed in order to confirm that the material chosen for the boom deployment has the best performances in a more realistic environment.

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7 CONCEPTUAL DESIGN: KINEMATICS OF DEPLOYMENT

7.1 INTRODUCTION TO HEXAPODS

The hexapod is a common structure in robotics, and it is getting more and more interest by the space community thanks to its capability in optical observations or communications payload. The main advantage is the **possibility of being deployed in space**, giving the chance to build up **larger structures** that still fit in the rocket fairing. However, the structure must guarantee a severe *mirror stability* in the mission life and should offer correction capabilities to overcome disturbances.

7.2 DESIGN AND KINEMATICS OF THE 6-PUS PLATFORM

The case in point is an hexapod uses *six rolled tape-springs as legs*. The classic *Stewart* platform uses the variable length of leg actuators to control the upper platform. However, the design proposed by *Merlet and Gosselein* is a 6-PUS platform manipulator, also called “*active wrist*”. Its main advantage is that the **active prismatic joints are fixed in the lower platform** and move vertically to adjust the position of the base of the fixed-length legs.

The **PUS** refers to the joints that are used: **P** stands for *Prismatic*, **U** for *Universal* and **S** for *Spherical*. **P** is underlined because it is the active joint used at the base of the legs, to provide the 6-Degree of Freedom (DOF) design.

This design allows to leave the tape-spring fully deployed while moving the base of the legs, therefore avoiding the complex corrections of the deployed booms, reducing inertia of the actuators, while still guaranteeing the corrections capability of the upper mirror with respect to the main one. The tape-spring is folded during the launch and the first part of the mission. In particular, the tape-spring is flattened and coiled around a rotating drum. Once in the desired orbit, the locking system is released, and the tape-spring booms are deployed. This happens thanks to the **potential energy stored** in the tape-spring because of the stress imposed in the coiling phase. Moreover, the *coiling device* can be fitted with an *internal coil-spring*, which helps the deployment when the locking system is released. Indeed, load relaxation (the dissipation over time of energy stored during folding) may cause low internal forces in the tape spring, therefore a problematic dynamic of the deployment.

The **deployment phase** is critical and complex, and the final configuration depends on the deployment deviations and short-term or long-term instabilities. These deviations can be corrected by the prismatic actuators, so that the telescope can operate in the desired configuration.

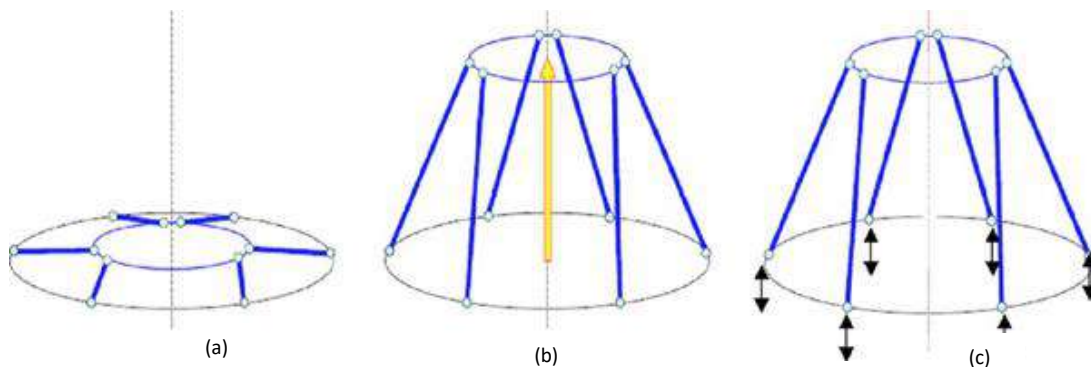


Figure 7.1. Stowed Configuration (a), Deployment (b) and Actuators (c)

7.2.1 Inverse pose kinematics

The control of the platform is governed by *inverse pose kinematics*, given the cartesian pose X (position and orientation) of the moving platform with respect to the base, find the required lengths $L = \{L_1 L_2 L_3 L_4 L_5 L_6\}'$. Referring to the figure below, the pose is described by $X = \{x y z \alpha \beta \gamma\}'$ (position of $\{P\}$ relative to $\{B\}$ and Euler angles).

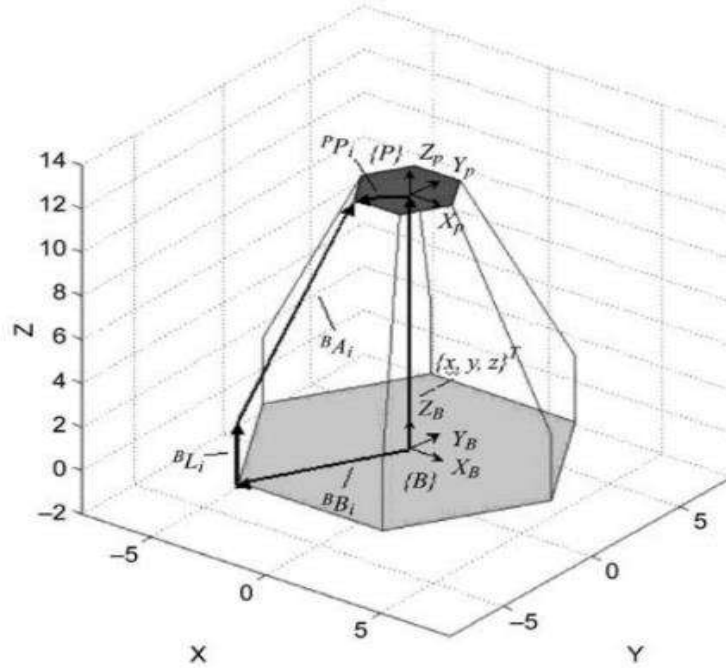


Figure 7.2. Pose kinematics

For each leg “ i ” we can write the following equation:

$$\begin{Bmatrix} B_{ix} \\ B_{iy} \\ 0 \end{Bmatrix} + \begin{Bmatrix} 0 \\ 0 \\ L_i \end{Bmatrix} + \begin{Bmatrix} A_{ix} \\ A_{iy} \\ A_{iz} \end{Bmatrix} = \begin{Bmatrix} x \\ y \\ z \end{Bmatrix} + \begin{bmatrix} r_{11} & r_{12} & r_{13} \\ r_{21} & r_{22} & r_{23} \\ r_{31} & r_{32} & r_{33} \end{bmatrix} \begin{Bmatrix} P_{ix} \\ P_{iy} \\ 0 \end{Bmatrix} \quad i = 1, 2, \dots, 6 \quad (1)$$

where:

$$\begin{bmatrix} r_{11} & r_{12} & r_{13} \\ r_{21} & r_{22} & r_{23} \\ r_{31} & r_{32} & r_{33} \end{bmatrix} = \begin{bmatrix} c\alpha c\beta & -s\alpha c\gamma + c\alpha s\beta s\gamma & s\alpha c\gamma + c\alpha s\beta c\gamma \\ s\alpha c\beta & c\alpha c\gamma + s\alpha s\beta s\gamma & -c\alpha c\gamma + s\alpha s\beta c\gamma \\ -s\beta & c\beta s\gamma & c\beta c\gamma \end{bmatrix} \quad (2)$$

Row 3 of Equation (1) contains two unknowns, resulting in two solutions for A_{iz} and the corresponding solutions for L_i . With six legs, this results in **64 inverse kinematic solutions**. Choosing only positive A_{iz} solutions ensure that the platform configuration lies above the actuators' extensions. Only one of the 64 solutions is specified for manipulator inverse pose kinematics.

7.2.2 Forward pose kinematics

The problem in *forward kinematics* becomes, given the active leg lengths, find the platform Cartesian pose. Equation (1) is rewritten and expanded:

$$\begin{Bmatrix} A_{ix} \\ A_{iy} \\ A_{iz} \end{Bmatrix} = \begin{Bmatrix} x + r_{11}P_{ix} + r_{12}P_{iy} - B_{ix} \\ y + r_{21}P_{ix} + r_{22}P_{iy} - B_{iy} \\ z + r_{31}P_{ix} + r_{32}P_{iy} - L_i \end{Bmatrix} \quad i = 1, 2, \dots, 6 \quad (3)$$

Writing the length of the passive leg in *Euclidean Norm*, combined with *Equation (3)* gives a function of the six Cartesian variables. Taking the partial derivatives results in the 6×6 *Newton-Raphson Jacobian matrix*, function of the unknown Cartesian pose $X = \{x \ y \ z \ \alpha \ \theta \ \gamma\}'$.

$$\{X_{k+1}\} = \{X_k\} - [J_{NR}(X_k)]^{-1}\{f_i(X_k)\} \quad (4)$$

An initial guess is made, and *Equation (4)* is used to update the solution.

7.3 STUDY OF THE DEPLOYMENT

The platform mobility and correction capabilities are made possible by implementing a *universal joint at the base of the leg*, and a *spherical joint between the tape-spring and the upper base* (which houses the secondary mirror). The upper junction acts as a spherical joint consisting of a **revolute joint** (first DOF), a **flexible metal blade** (second DOF) and exploiting the low torsional rigidity of the tape-spring leg. The base connections, instead, behave as universal joints by using the possibility of the tape-spring to coil around the hub, and a revolute joint that connects the coiling mechanism with the lower platform.

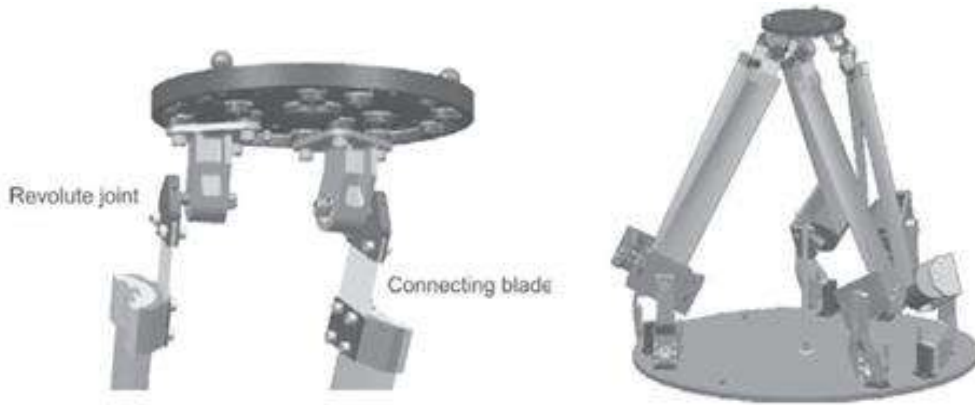


Figure 7.3. Representation of the joints

The deployment of twelve different deployments were successfully experimented by the photogrammetry technique. This method is based on several images taken during the deployment.

The *translational deployment deviations* were lower than 1 mm and the *rotational deployment deviations* were lower than 2 m-rad . These deviations can be subsequently corrected thanks to the designed *corrections capabilities* of $[2 \text{ mm}, 2 \text{ mm}, 2 \text{ mm}, 2^\circ, 2^\circ, 3^\circ]$. These results guarantee for the deployment repeatability of the hexapod and so a deployable hexapod with tape-spring legs can be developed.

Moreover, the results and the frequencies of a modal analysis depend on the analysed model, but the mode shapes are reported here as a reference to the possible critical modes that can evolve in the proposed hexapod.

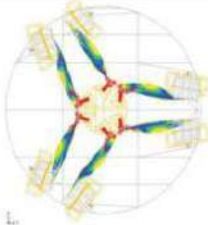
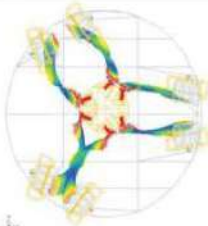
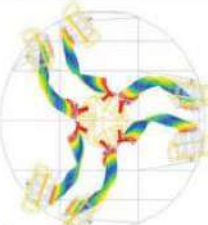
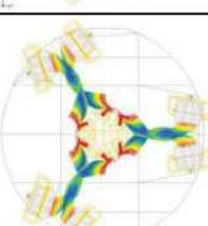
Mode shapes			Platform motion	Predicted frequencies
Mode 1		Bending of blades	Mainly Tilt	19 Hz
Mode 2		Torsion of blades	None	26.6 Hz
Mode 3		Torsion of legs (tape-springs + blades)	None	77.3 Hz
Mode 4		Bending of legs	Tilt	97.7 Hz
Mode 5		Bending of legs	Mainly Twist	141.9 Hz
Mode 6		Bending of legs	Pumping	149.5 Hz

Figure 7.4. Mode Shapes

Two of the major limitations of this space deployable structures are the **deployment accuracy** and the **dynamic stability**. Consequently, deployment processes must be investigated in order to optimize design and performances. The field of robotics points out *two dynamic models* to establish robot control diagrams:

- **an inverse dynamic model IDM** that links the external forces to the Cartesian and angular acceleration of the upper platform
- **a forward dynamic model FDM** which supplies the generalized coordinates of the platform versus the external forces.

The main difficulty lies in the analytical formulation of the FDM, which is not easy to solve due to the closed-loop structure and the interdependence of the parameters.

7.4 HEXAPOD DYNAMIC MODEL

Three main approaches are commonly used for solving the dynamics of parallel structures:

- 1) The **Newton–Euler approach** considers each part of the robot as a subsystem, thus the equation of motion for each body of the system must be established in relative coordinates. The resulting dynamic equations are a set of forward and backward recursive equations. A forward recursion propagates link velocities and accelerations, then a backward recursion propagates the forces and torques along the manipulator chain. This method is based on the principle of the manipulator being a serial chain; when a force is applied to one link, it may also produce motion in connected links. Due to this effect, there may be considerable duplication of calculation, which can be avoided if expressed in a recursive form. This reduction in computational load greatly reduces calculation time, allowing the forward and inverse dynamics calculations to be performed in real-time; therefore, it can enable real-time torque control methods of robot manipulators.
- 2) The **principle of virtual work** permits knowing the main model information without a high computational cost. It can be implemented in the control system due to its quickness.
- 3) The **Lagrangian equations** provide explicit state equations, highly nonlinear, for robot dynamics, calculating the kinetic and the potential energies of the system. Due to the imposed constraints of a closed loop kinematic chain, it requires long calculations of the Lagrangian partial derivatives using independent generalized coordinates.

Other approaches have been proposed, but many dynamic models require huge processing capacities and simplification for implementation in a real-time control system, hence many works have focused on the reduction of the computational time.

Different methods can be applied depending on the requirements of the application and/or whether the dynamics must be calculated for simulation or control purposes.

In our project we focused on describing the **mechanical behaviour** of self-deployment hexapod, based on the *release of stored strain energies*.

7.4.1 RESTORING FORCE MODEL

One of the most common methods used in literature to describe tape springs' behaviour is the *restoring force model* named **the generalized Dahl's model**. In physics, the restoring force is a **force which acts to bring a body to its equilibrium position**. It is a function only of position of the mass, and it is always directed back toward the equilibrium position of the system.

Linear actuators contain a rotating roll module with a spiral groove in order to guide the tape-spring. The coiling of the strip induces a flattening of the natural curved section. Because of the stress generated by the elastic deformation, the tape-spring tends naturally to go out to recover its curved section. Inside the roll module, the hub has an additional internal coil-spring which helps the self-deployment when the locking device is released (Fig. 7.5).

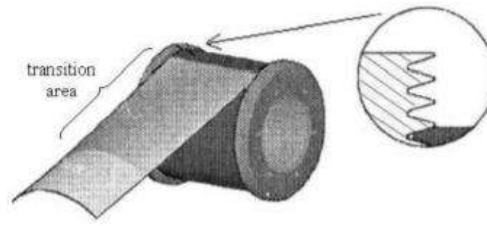


Figure 7.5. Tape spring coiling device

The actuator is characterized by two main phenomena: an **energy supplied by the deformation of the tape-spring** (this concerns the transition zone between the flattened section and the naturally curved section of the tape-spring) and an **energy dissipation because of radial friction** of the tape-spring at the entry of the groove.

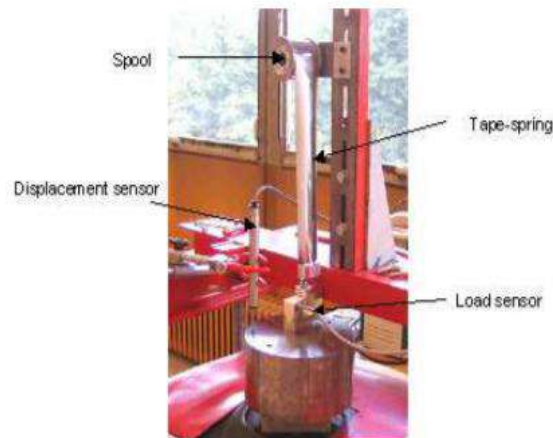


Figure 7.6. Experimental set up

The experimental set-up (Fig. 7.6) used for characterization presents the hub of the spool fixed on a frame allowing the change of the unrolled tape-spring length and the spool free to rotate. The unrolled end of the tape-spring is connected to a *load sensor and sine excited in displacement* thanks to an *electro-dynamic shaker*. The forced displacement is measured by using a *laser sensor* with a one-micrometre accuracy. Force deflection loops are recorded for several forcing frequencies and unrolled strip lengths. They exhibit a **nonlinear hysteresis behaviour** (Fig. 7.7).

It can be observed that the path is *anti-clockwise direction*: the self-unrolling-load provided by the tape-spring is lower than the necessary load to roll it on the hub. Both measured loads are positive because the strip is preloaded. Moreover, it appears that the *unrolled and rolled forces remain quite constant with elongation*.

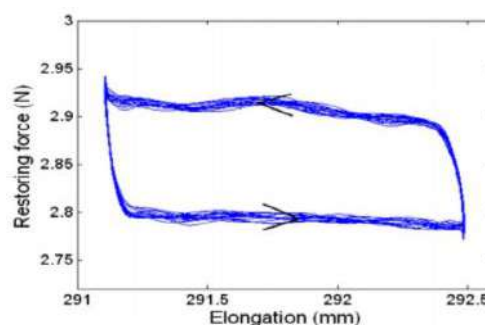


Figure 7.7. Example of force-elongation loop of the actuator at 1Hz

The model taken into account to describe the restoring force is a more general one developed by *Al Majid* based upon the *Dahl's model*.

The following first order differential equation is considered:

$$\dot{R} = \beta \dot{u} |h - R \operatorname{sgn}(\dot{u})|^\mu$$

where R is the restoring force, $\dot{u}$ the deflection velocity of the tape-spring length, β a parameter that adjusts the shape of the friction slope function, μ a constant acting on the orientation loop.

The hysteretic behaviour is captured by defining the function h :

$$h = \frac{(h_u - h_l) + (h_u + h_l) \operatorname{sgn}(\dot{u})}{2}$$

where μ is one, β is determined by comparing theoretical and experimental dissipated energy loop areas and h_u and h_l represent the *upper and lower boundaries lines* of the loop whose equations are identified thanks to the least squares method:

$$h_l = a_l u + b_l$$

$$h_u = a_u u + b_u$$

Constants a_l, b_l, a_u, b_u are the *slopes and the initial forces* of uncoiling and coiling, respectively.

The hub inertia is considered in *Al Majid's* model thanks to the asymptotic curves, which depend on the forcing frequency. Based on experimental data, Fig. 7.8 shows the evolution of the **straight-line equation parameters versus cycling velocity**.

Anyway, subtracting hub inertia from the measured forces, to distinguish the dry friction of the tape-spring in the groove from viscous friction provided by the internal roller bearing supporting the hub axis, it was proved that *frequency dependence was negligible* and thus the spring actuator is equivalent to a *stick slip system*.

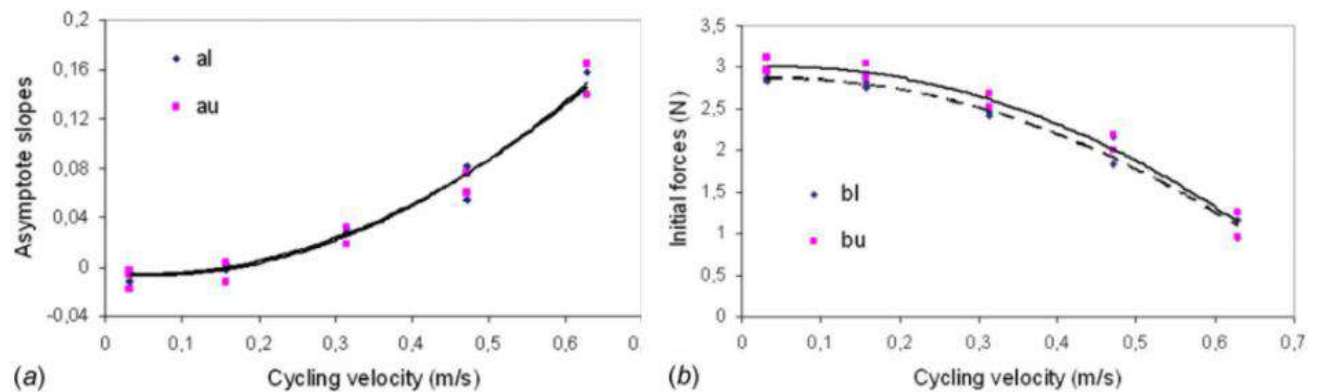


Figure 7.8. Evolution of the coefficients of the straight asymptotic lines: (a) Asymptote slope (b) initial force

7.4.2 Geometry

The model was created using *ADAMS software* and following the reference geometry in the *Statement Of Work* of the ESA document “Deployable Telescope using Tape Spring Booms”: it is made of **two mirrors, one upper and one lower**. The lower mirror has a diameter of *3700 millimeters*, while the upper mirror has a diameter of *400 millimeters*. In the first phase, **spherical joints** were inserted between the *levers* and the mirror and the deployment was simulated through **translational joints**, to which a law of motion was applied. Although the model worked, it could be improved by inserting **forces** that simulated the movement of the upper mirror as tape springs would.

The arrangement of the levers on the relative mirrors was considered according to the scheme shown in Fig. 7.9, estimating an angle of 90° between two consecutive levers in the absence of other data.

To simplify the model, spherical joints (Fig. 7.10a) have been replaced with fixed joints (Fig. 7.10b), so that the two-bodies *model* of action-reaction *force* can be used between the corresponding levers of the two mirrors. The direction of application of this force always lies on the line of conjunction between the levers, in this way degrees of freedom are not lost while introducing joints.

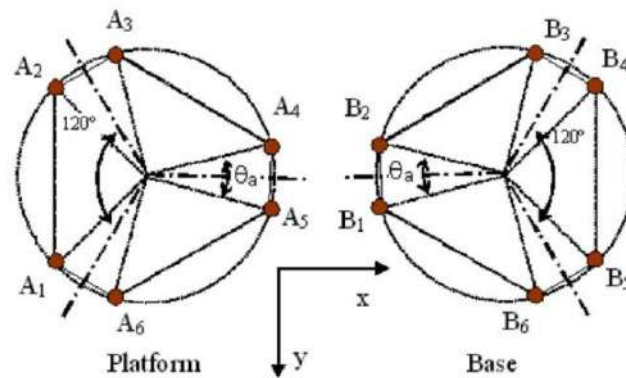


Figure 7.9. Levers location on platform and base

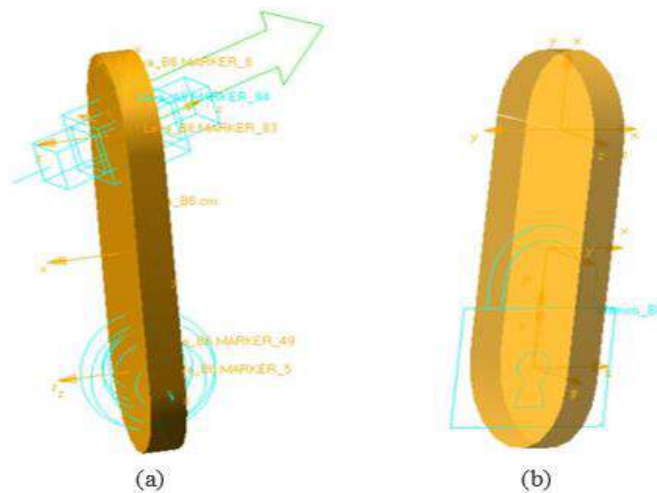


Figure 7.10. Representation of spherical joints (a) and fixed joints (b)

7.4.3 Simulation

While running the simulation, a problem arose due to the fact that, despite the symmetrical position of the levers, the forces on the horizontal plane did not completely balance, generating a torque that turned the upper mirror upside down. Since this inconvenience was at a greater distance than the maximum elongation of the tape springs, it would not have been an impediment. Despite this, it was decided to refine the model itself by introducing compensating **torque and force** proportional to the position and velocity (linear and angular) of the *center of mass of the upper mirror/levers aggregate* A. Once the absence of oscillations on the horizontal plane and the correct displacement of the upper mirror (only long z translation) were verified, a **stop system** was introduced through a mechanism that can be reconducted to a spring-damper system.

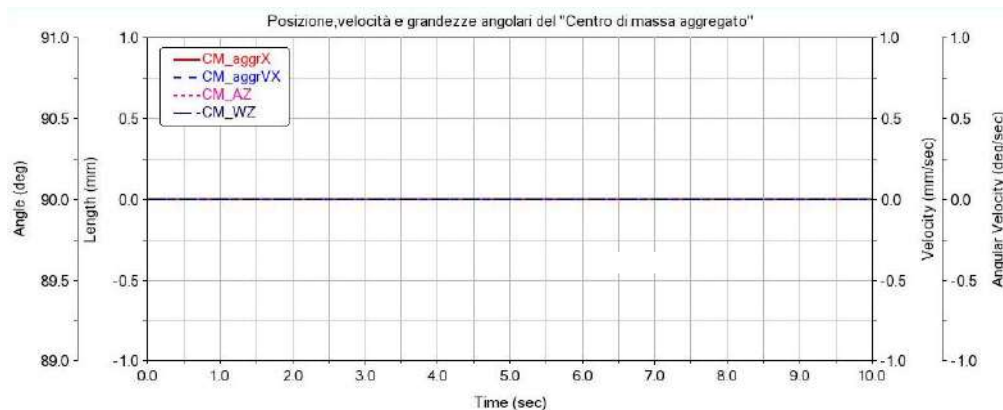


Figure 7.11. Trend of the main parameters

The operation principle of this mechanism is based on the activation of the **recall forces** of the upper mirror and the **damping** of its motion only once the desired deployment has been reached (in this case *3500 mm*). Activation is through a **distance sensor** between a pair of *A-B levers*. The result is a total stop of the upper mirror with a minimum oscillation. However, the inserted values are preliminary and should be changed in the later phases of the project.

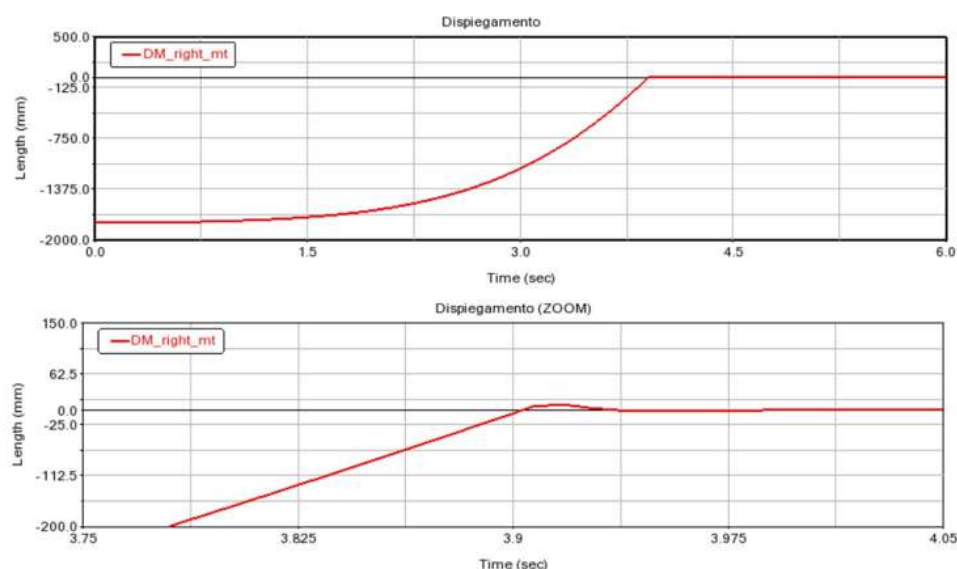


Figure 7.12. Elongation during the deployment

$$m_{zy} = D(k_l + \nu k_t)$$

$$m_{yz} = D(k_t + \nu k_l)$$

- $k_l = 1/r$, where r is the coil radius
- $k_t = 1/R$, where R is the tape spring curvature radius
- $D = Et^3/12(1 - \nu^2)$, where E is the young modulus, t is the tape spring thickness and ν is the Poisson modulus

It is proven that **to minimize the tape spring potential energy** it is possible to assume $r = R$, thus:

$$m_{yz} = m_{zy} = \frac{D}{R}(1 + \nu)$$

The total potential energy of the system can be calculated as follows:

$$\begin{aligned} P &= - \iint_A \mu dA = - \iint_A \frac{1}{2} m_{yz} \epsilon_{yz} + \frac{1}{2} m_{zy} \epsilon_{zy} dA = \\ &= - \iint_A \frac{D}{R} (1 + \nu) \frac{1}{R} dA = - \frac{D}{R^2} (1 + \nu) R \alpha (l - R \theta) \end{aligned}$$

Where:

- α is the angle under the tape spring curvature
- θ is the rotation angle of the coil
- l is the length of the uncoiled tape spring

To determine the **total kinetic energy of the system**, other two assumptions were taken:

- 1) during the deployment, *the secondary mirror goes vertically up* (i.e., velocity components lying on the mirror's plane were considered negligible)
- 2) since the *tape springs* mass and mean velocity are the lowest in the system, its *kinetic energy is considered negligible*

With these assumptions, the kinetic energy can be calculated as follows:

$$\mathcal{K} = \mathcal{K}_{mirror} + \mathcal{K}_{spool}$$

Where:

$$\mathcal{K}_{mirror} = \frac{1}{2} \frac{M_m v_m^2}{n}$$

- M_m is the mirror mass
- v_m is the mirror velocity
- n is the number of tape springs in the system

$$\mathcal{K}_{spool} = \frac{1}{2} I_s \dot{\theta}^2$$

- $I_s = 1/2 \rho_s \pi R^4$ is the spool moment of inertia
- $\dot{\theta}$ is the spool angular velocity

Considering the following scheme (Fig. 7.15), it is possible to reduce the problem to the study of a **1 degree of freedom system**, resulting in a differential equation in θ only:

- $v_m^2 = v_s^2 + v_r^2$
- $v_r^2 = \dot{\phi}(R\theta + C)$
- $\phi = a \cos\left(\frac{C}{R\theta + C}\right)$

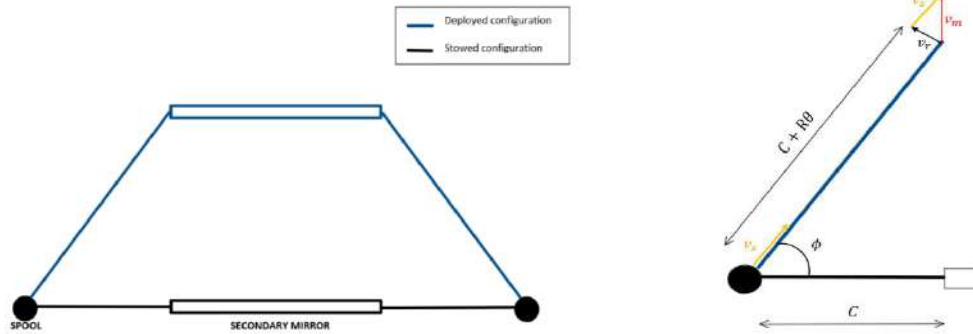


Figure 7.15. System representation e main

Substituting these expressions in the first equation we have that:

$$v_m^2 = \dot{\theta}^2 R^2 \left(1 + \frac{C^2}{(R\theta + C)^2 - C^2} \right)$$

As a result, a non-linear, second order differential equation, with non-constant coefficient is obtained:

$$-\left\{ M_m R^2 \left[\frac{C^2 (R\theta + C) R}{[(R\theta + C)^2 - C^2]} \right] \right\} \dot{\theta}^2 + \left\{ M_m R^2 \left[1 + \frac{C^2}{(R\theta + C)^2 - C^2} \right] + \frac{1}{2} \rho_s \pi R^4 \right\} \ddot{\theta} + D(1 + \nu) = 0$$

The previous equation was integrated on *Matlab*, using a *stiff-ode integrator (ode15s)*. To start the tape spring deployment, an initial arbitrary small angle of θ_0 was given (in this case $\theta_0 = 0.01$). The initial velocity $\dot{\theta}_0$ was set to 0. The following figures will show **trends of angle θ , angular velocity $\dot{\theta}$, mirror height y , linear velocity $\dot{y}$ and the force F applied to the mirror**.

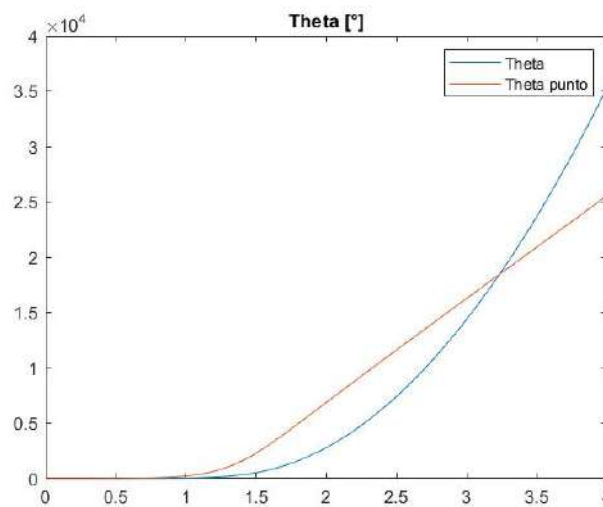


Figure 7.15. Angular position and velocity of the coil during the deployment

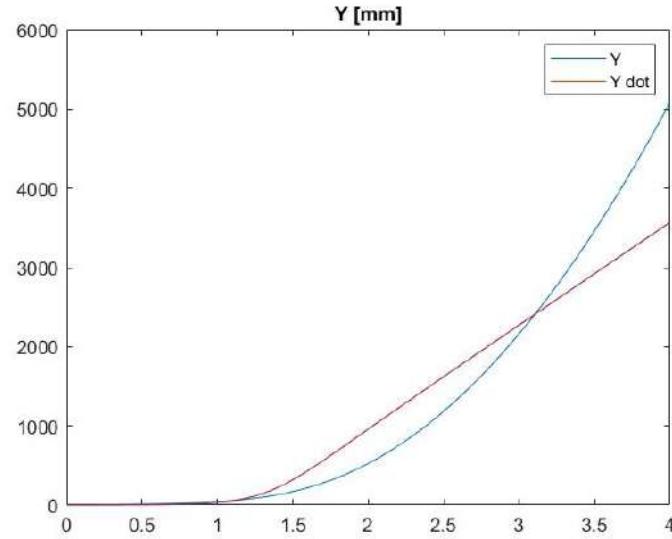


Figure 7.16. Linear position and velocity of the mirror during the deployment

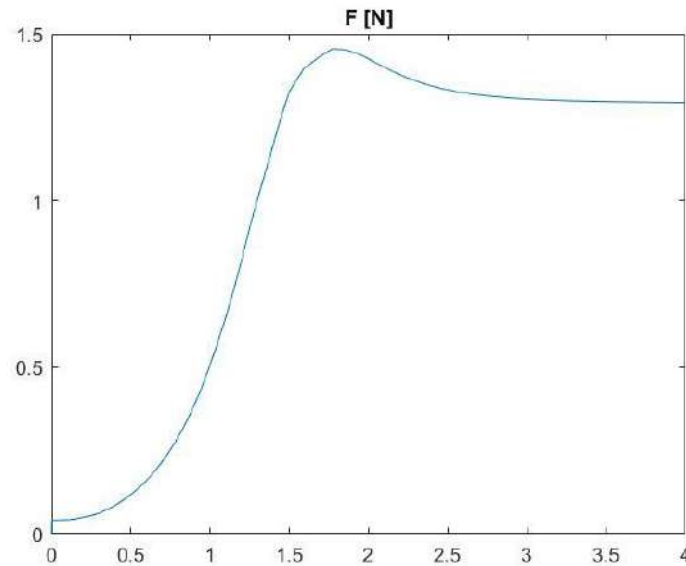


Figure 7.17. Total force applied to the mirror during the deployment

7.4.4.2 Solution of the controlled system dynamics

From the previous graphs it is possible to notice that after a transient, the restoring force becomes almost constant and consequently the angle θ described by every tape-spring rises in a parabolic way. **End stop springs** are introduced in order to limit this indefinitely motion and obtain a controlled deployment. *Linear springs* are chosen for ease of use and they have been attached to the upper joints linked to the mirror. The initial spring displacement matches the main platform level, so that the overall system turns out to be compact in the fairing during the launch. These assumptions led to a reformulation of the *Lagrange differential equation*. In particular, **the elastic energy is taken into account**, in addition to the kinetic energy of the mirror and the spool and the potential energy of the (coiled) tape-springs:

$$P_{spring} = -\frac{1}{2}Ky^2$$

where $y^2 = (R\theta + C)^2 - C^2$.

The spring potential energy is differentiated as follows, and added to the overall system dynamics equation:

$$\frac{\partial P}{\partial \theta} = -KR(R\theta + C)$$

The trends of *angle θ* , *angular velocity $\dot{\theta}$* , *mirror height y* and *linear velocity $\dot{y}$* show a **harmonic behaviour**. For instance, the mirror height oscillates between the initial position (the main platform level) and a value for which the elastic force equals the tape-spring restoring force. Therefore, a **damper is required**, and the model has to be modified to provide an end stop spring-damper component.

The *damping force* is calculated as follows and added to the Lagrange equation as an external force:

$$F_A = cv_y^2 = c\dot{\theta}^2 R^2 \left(1 + \frac{C^2}{(R\theta + C)^2 - C^2} \right)$$

The final differential equation of the controlled system is the following:

$$\begin{aligned} & - \left\{ M_m R^2 \left[\frac{C^2 (R\theta + C) R}{[(R\theta + C)^2 - C^2]} \right] \right\} \dot{\theta}^2 + \left\{ M_m R^2 \left[1 + \frac{C^2}{(R\theta + C)^2 - C^2} \right] + \frac{1}{2} \rho_s \pi R^4 \right\} \ddot{\theta} + \\ & + D(1 + \nu)\alpha - RK(R\theta + C) = c(R\dot{\theta})^2 \left[1 + \frac{C^2}{(R\theta + C)^2 - C^2} \right] \end{aligned}$$

The equation was integrated with the same method and initial condition described in the previous paragraph. The trends of angle θ , angular velocity $\dot{\theta}$, mirror height y , linear velocity $\dot{y}$ and the force F applied to the mirror are the following.

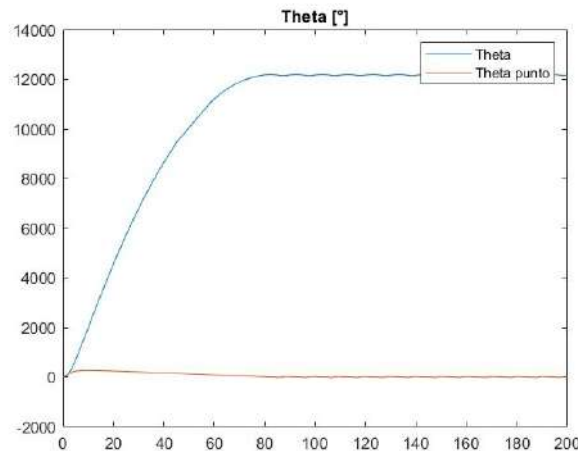


Figure 7.18. Angular position and velocity of the coil during the controlled deployment

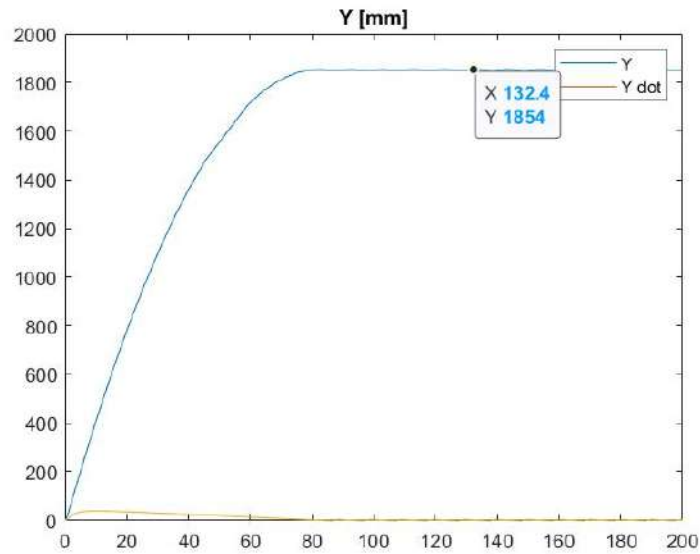


Figure 7.19. Linear position and velocity of the mirror during the controlled deployment

With the use of the dampers, it is possible to easily deploy the mirror at the desired height. After 60 seconds the maximum deployment height is reached with a significantly low speed. Even though the system tends to oscillate, the amplitude tends to decrease around the desired asymptotic value.

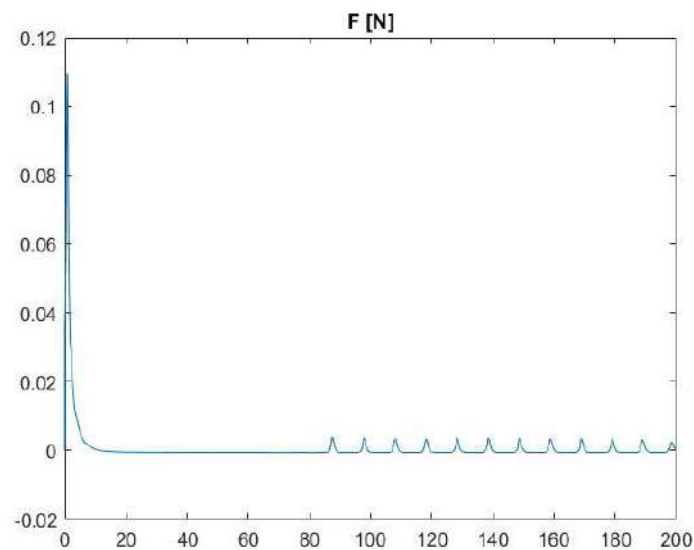


Figure 7.20. Total force acting on the mirror during the controlled deployment

In the graph concerning the force we can see a very high initial peak necessary to push the system to the desired height. This peak is followed by small oscillations also due to the presence of the damper. However, these residual forces are of the order of 0.001 N, hence completely neglectable. A further increase in the damping value may further decrease these residual forces to 0 N. The overall dynamic of the deployment system seems to have a good behaviour and the mirror does not go under excessive forces or velocities.

8 CONCEPTUAL DESIGN: POSITION AND POINTING

In this section it will be explained every choice it has been made up for position and pointing conceptual design.

8.1 ATTITUDE AND ORBIT DETERMINATION

First of all, it's necessary to select devices to determine attitude and orbit positioning. These devices will be different according to tasks and requirements.

8.1.1 Two-axis determination

To obtain the requested attitude determination accuracy we decided to use, for pitch and roll determination, a horizon sensor. The chosen sensor is MiDES, made by Servo corporation of America, and specifically designed for LEO satellites.

The MiDES (Mini Dual Earth Sensor) is specifically designed to provide horizon position in both Pitch and Roll for Earth orbiting unmanned LEO satellites. The MiDES derives its information using two pairs of 16 element pyroelectric detectors positioned 90 degrees apart, combining to provide 32 pixels that view the horizon. Each 16 elements array is spatially separated into two 8 element staggered columns shown below to provide for sun/moon rejection and for radiance compensation. Covering a FOV of 11 degrees, the pixels detect the temperature difference between the reference chopper, Earth and Space through measurement of the CO₂ band thermal gradient. An algorithm calculates the position of the horizon based on the voltages obtained from the pixels that subtend the horizon. Earth and Space looking pixels are used to reduce radiance errors.

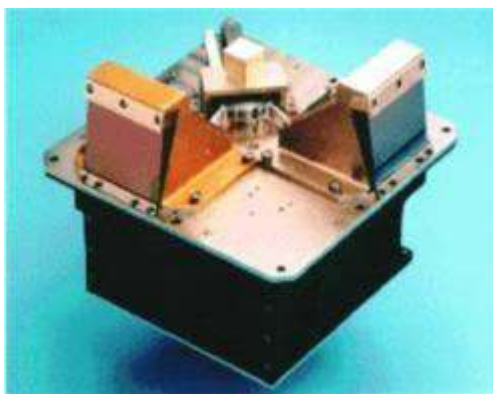


Figure 8.35. Horizon sensor MiDES, Servo corporation of America.

Model	MiDES
FOV [°]	11x5
Accuracy [°]	0.04
Optical Pass Band [μm]	14.6-15.8
Dimensions [mm]	133,35x133,35x128,78
Mass [kg]	1.5
Operating Temperature [°C]	-30 /+60
Power Supply [VDC]	28(+7 -6)
Max power consumption [W]	0.800
Flight Heritage	•Astro Spas - Astrium (DASA) •UOSAT 12 - SSTL •QuikToms - Orbital

Using four sensors, one on each vertex of a rectangular configuration offer redundancy and the capability of seeing the horizon.

8.1.2 Third axis determination

For Earth observation missions, both Sun sensors and star sensors are typically used, but given the accuracy requirements to which the mission is subjected, the choice fell only on star sensors, capable of providing the best possible accuracy.

Among the star sensors available on the market, one of Leonardo's star trackers was selected, the AA-STR model.

These star trackers are characterized by a very precise three-axis attitude determination, with a low mass and very low power consumption for the class of instrument to which they belong.

The AA-STR is a new generation product with a high level of technological maturity (TRL-9) with already numerous applications in the scientific field (Bepi Colombo - ESA, Astro-G - JAXA), Earth Observation (PRISMA) and other commercial programs.

Particularly suitable for GEO and LEO missions having a high level of radiation resistance and is guaranteed for an operational life of 18 years in GEO.



Detector	HAS APS
FOV [°]	20x20
Number of tracked stars	Up to 15
Tracking rate	Up to 2°/sec
Point accuracy (Pitch, Yaw [arcsec]	8,25
Point accuracy (Roll) [arcsec]	11,1
Update rate [Hz]	10/8/5/4
Dimensions [mm]	164x156x348
Mass [kg]	2,66
Operating Temperature [°C]	-30 / +60
Power Supply [V]	20-52
Max power consumption [W]	5.6-12.6
Lifetime GEO [years]	18

Figure 8.2. Star tracker AA-STR, Leonardo.

It has high performances that allow it to be a reliable instrument to be installed on space telescopes with extremely precise aiming and in agile or rotating satellites.

The operating temperature range is estimated from -30 ° C to 60 ° C and has a storage temperature between -35 ° C and 65 ° C.

Six-star trackers will be installed in a two-plate configuration. Two multiple heads star sensors will be installed in order to observe the operability requirements in all mission phases.

The manufacturing company declares its willingness to produce custom trackers for spacecraft with highly demanding mission requirements.

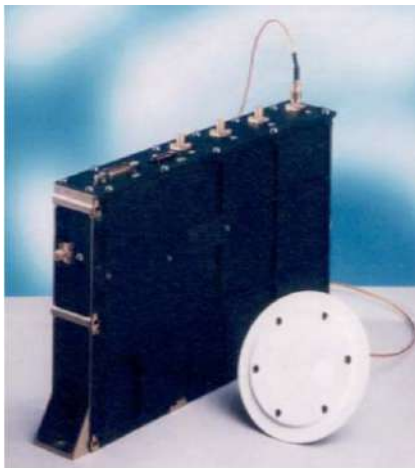
8.1.3 Orbit determination

Through the various options, it has been decided to rely on NAVSTAR GPS constellation.

8.1.3.1 GPS receiver

GPS receivers now provide an established system for tracking spacecraft in Low Earth Orbit (LEO). So far, the distinction between navigation and scientific applications is almost unequivocally reflected in the choice of single or dual frequency receiver technology. The large number of GPS in-orbit receivers is designed to provide navigation and timing information with an accuracy that is well compatible with the GPS Standard Positioning Service (SPS) and thus they use the single frequency technology. Scientific applications, on the other hand, require the availability of dual frequency measurements.

As we do not need extreme precision, since the attitude determination is done with horizon and star sensor, we chose to use the single frequency technology for our configuration. In particular, the selected receiver is the TOPSTAR 3000. Between its various features we remark the highly accurate and reliable navigation, allowing to cope with poor visibility conditions and spacecraft manoeuvres, the very high sensitivity and the modular design able to accommodate a variety of interface and mission requirements in terms of number of antennas, number of processing channels, class of receiver clock, data interfaces, power interfaces. It also is fully space qualified.



Model (Manufacture)	TOPSTAR 3000 (Alcatel)
Channel	12-16 C/A
Weight	1.5 kg
Power	1.5 W
Dimension (LxWxH)	276x42x170 mm
Radiation Tolerance	>30 Krad
Operating temperature	from -70°C to 70°C
Non-operating temperature	from -80°C to 80°C
Velocity accuracy	1 cm/s
Acquisition/tracking	50 bps
Position accuracy	<10 m

Figure 8.3. GPS receiver TOPSTAR3000 by Alcatel.

The receiver consists of a *signal-processing* and a *localisation module*:

- The *signal-processing* consists of a set of channels that perform GPS signal acquisition and tracking and produce measurements and 50 bps demodulated GPS message data. In order to cope with the wide signal dynamic range of space missions, TOPSTAR 3000 includes several acquisition algorithms.
- The *localisation module* is dedicated to position, velocity and time computation, and signal-processing management.

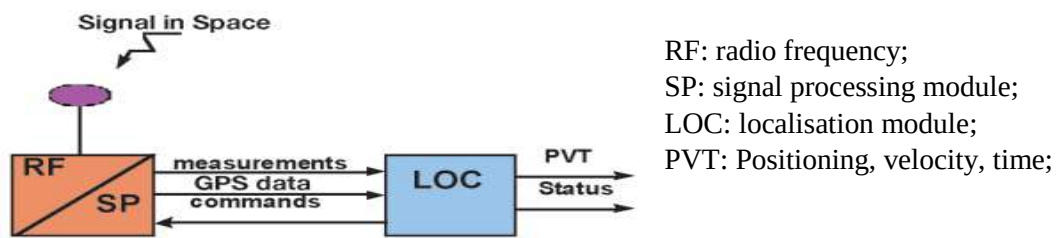


Figure 8.4. Functional architecture of TOPSTAR3000.

The CNES-developed orbital navigator implemented in TOPSTAR 3000 is called DIOGENE (Détermination immédiate d'orbite par GPS et navigateur embarqué). Based on a Kalman filter, it propagates the state vector with an accurate force model and updates it with all available GPS measurements.

8.1.3.2 GPS receivers' location

To achieve the determination of the telescope position along the orbit, it will be necessary that the receivers receive the signal from at least 4 transmitting satellites.

The NAVSTAR constellation is formed by a set of 24 satellites orbiting at an altitude of about 20200 km, and they are responsible for emitting the signals the spacecraft will have to receive. Their constellation is made up of 6 almost circular orbits, with 4 satellite orbiting along each of them, separated by 90° (in terms of *true anomaly*) from each other. Thanks to this configuration it is achieved that every receiver has in sight at least 4 of these satellites, or what is the same, every receiver has available their signals, at any times.

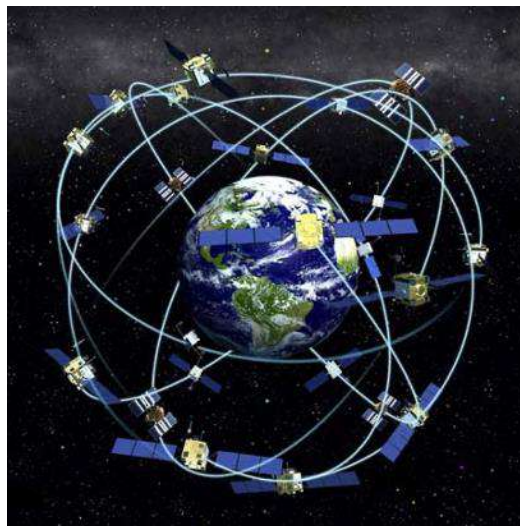


Figure 8.5. NAVSTAR constellation.

Once the position of the transmitting satellites is known we can proceed to make the following reasoning. The spacecraft will have the function of capturing images of the Earth, so it is obvious that its mirrors will point towards the Earth's surface. On the other hand, we know that the orbit in which the telescope will be located is a LEO orbit, in a lower altitude than GPS satellites. As we also know that it will need to receive the signals from the GPS satellites, coming from "above" our body, it is logical that we place the GPS antennas on the opposite face where the mirrors are placed. In this way a cleaner and clearer reception of the GPS signal will be achieved.

That said, 2 GPS receivers will be incorporate to correct data reception and thus be able to ensure orbit control in an adequate way. They will be positioned 180° apart in the face of the satellite facing the GPS/GNSS constellation, ensuring a simple and safe configuration.

The use of more antennas would be needed in applications such as attitude determination, where greater precision is required.

8.2 ATTITUDE AND ORBIT CONTROL

Considering the purpose of the mission, the Earth's observation, we need an autonomous attitude determination, that can provide a position knowledge better than 0.1deg to support the pointing requirements. Three axis control method provide the precision we want to reach (± 0.01 deg to ± 0.1 deg) , with different possible solutions, spacing form thrusters, momentum and reaction wheels and CMG. The method can be developed in two forms: momentum bias and zero momentum. The first usually uses just one wheel mounted along the pitch axis, normal to the plane orbit, giving the spacecraft gyroscopic stiffness. The latter, which we are choosing, the selected actuators work in order to delete the disturbances affecting the spacecraft. Zero momentum control has been using reaction wheels o CMG, with the support of the thrusters just to desaturate them when needed.

8.2.1 First proposal: thruster-based configuration

With new technologies, it is now possible to use thrusters also to balance the external torques with a great accuracy, thanks micro-propulsion.

For this configuration we thought about using an ADCS only driven by thrusters, which are not affected by the same concerns as momentum storage device (limited maximum momentum storable).

They produce torques and forces that:

- control attitude;
- adjust orbit;
- control nutation;
- control the spin rate;
- maneuver spacecraft over large angles.

On the other hand, the spacecraft life is limited by the propellant quantity that aliments the thrusters; even though for this matter, micro-propulsion system helps us, needing very few grams of propellant to conduct the maneuvers. We also need to get the thrusters in the best position to avoid their plumes to contaminate the surfaces, such as solar array and telescope's mirrors.

8.2.2 Micro-propulsion and FEEP

We decided to use an innovative technology developed in recent years, the Field Emission Electric Propulsion (FEEP). The FEEP thrusters can usually provide thrust in the range of $199\mu N - mN$. Unlike most propulsion systems, FEEP doesn't require external propellant tanks, tubing, valves, etc. Propellant feeding is made by capillarity, filling the assembly on ground, with the entire quantity of propellant needed for the mission. With this discussed configuration it's already possible, for small

and medium spacecraft, to guarantee a 5 years life mission. So, if necessary, we could think about a feeding system with external tanks to increase considerably the life duration.

Thrust is produced by exhausting a beam, mainly composed of singly ionized cesium atoms, produced by field emission. The actuator is mainly composed by three elements: the emitter, the accelerator and the neutralizer. The accelerator is placed in front of the emitter. When thrust is required, a strong electrical field is generated applying a high voltage difference between the accelerator and the emitter. When a certain value of the electrical field is reached the ionization process begins. The neutralizer is the external source of electrons that provide negative charges to maintain global electrical neutrality of the assembly.

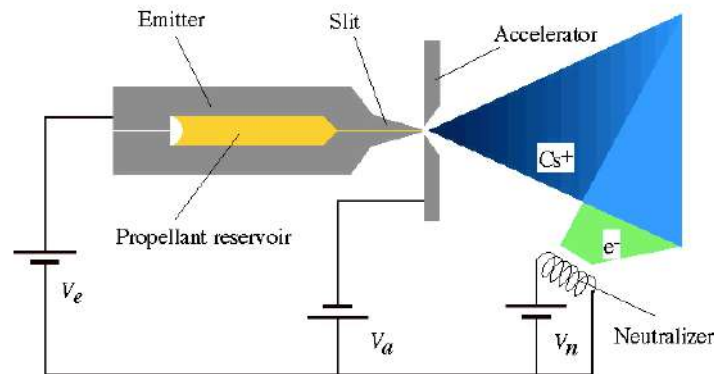


Figure 8.6. FEEP architecture.

Unlike most ion engines, FEEP doesn't require propellant vaporization to obtain ionization: ions are directly extracted from the liquid phase. A large number of liquid metal or alloys can be accelerated by the thruster and the specific impulse is in the 2000 to 8000 s range.

8.2.3 Thruster selection

The *Field Emission Electric Propulsion* (FEEP) thrusters can usually provide thrust in the range of μN -mN. Starting from the requirements it's possible to select from the thruster's catalogue a specific electric model: "**RIT μX** ". It is the smallest Radiofrequency Ion Thruster by "*Ariane Group*", able to provide levels of thrust from $50 \mu\text{N}$ to $500 \mu\text{N}$, a perfect range of values that guarantee a high level of orbit and attitude control, with minimum errors. Mini-ion engines are considered a suitable solution to overcome difficulties and limitations of low thrust propulsion application. A growing number of missions combines the demand for high specific impulse, long lifetime, high total impulse and highest thrust resolution, accuracy and controllability.

An electric propulsion system based on RIT μX answers to mission requirements thanks to special characteristics demonstrated during extensive tests, such as the high performance at low complexity, highest specific impulse that offers substantial mass saving, a narrow beam divergence, the robust design concept with a large domain of operational stability, the large throttle range and adaptability to available electric power and its very low noise and long thruster lifetime. Also, it has an high growth potential with increasing electric power.



Figure 8.7. RIT model by ArianeGroup.

TRHUST & POWER	
Nominal Thrust	50-500 μN
Nominal Power	<50 W
FUNCTIONAL PERFORMANCE	
Extended/ On Request	10-100 μN 300-3000 μN
Isp	300-3000 s
Max demonstrated	>3500 s
Divergence angle	<17°
LIFETIME	
Total Impulse	from 10 kNs to 200 kNs
Max operational cycles	>10000
Total lifetime	>20000 h
TECHNOLOGY	
Acceleration	Electrostatic
Ionisation	RF-principle
Grindsystem	2 grids
Propellant	Xenon
DESIGN	
Mass	440 g
Diameter	78 mm
Height	76 mm
ENVIRONMENT	
Operating temperature	from -40°C to 160°C
Non-operating temperature	from -60°C to 160°C

8.2.4 Thruster's location

The chosen configuration consists of 16 thrusters arranged in pairs at each vertex of the spacecraft main body. A neutralizer is placed on every edge, to be in total 4 neutralizer, each one maintaining the global neutrality of four thruster. This arrangement was chosen to provide full thrusting authority along all axes with high degree of redundancy.

For every rotation, two thrusters are needed to generate torque along the axis, and they become four in order to control torque in both verse (positive and negative). So, 4 thrusters are positioned to control roll and 4 to control yaw. Of these 4 actuators, 2 need to be placed in opposite faces with respect to the 2 others, and in the same face, the 2 actuators have to be positioned diagonally in opposite edges. In the figures below the red triangles represent the thruster that can produce a positive torque along the specific axis, and in green the ones that can produce a negative torque, referring us to the cartesian triad showed net to the spacecraft modelled as a cube.

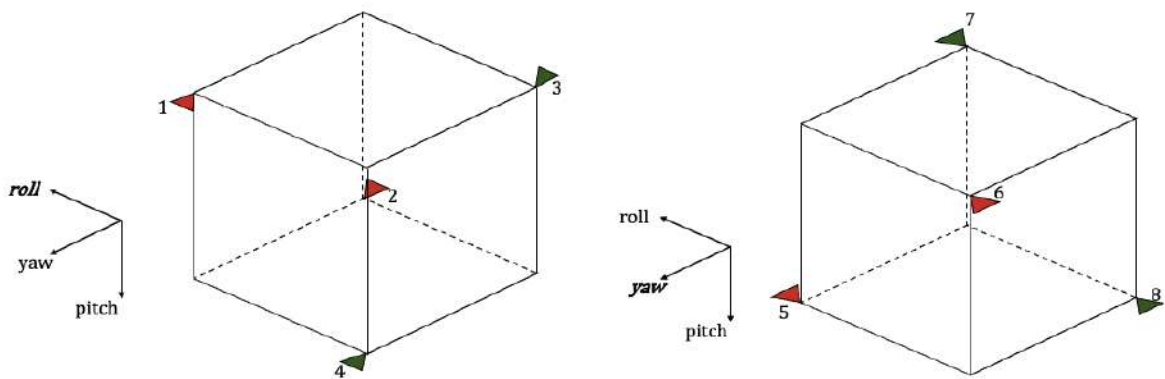


Figure 8.7. Roll and yaw balancing thrusters.

The same 8 thrusters will be used to control pitch by selecting accurately the combination with whom coupling them.

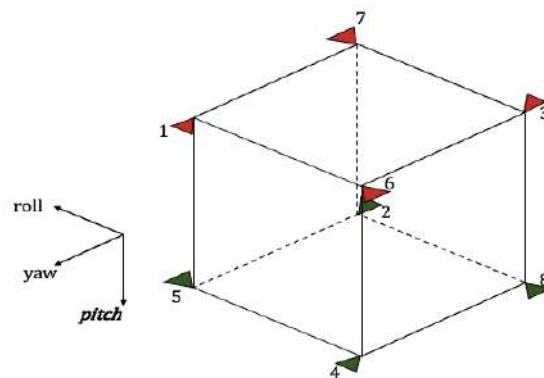


Figure 8.8. Pitch balancing thrusters.

Thus, with a degree of redundancy for roll and yaw, the thrusters are now 16. For the pitch axis the redundancy is even greater. In the figure below the black triangle represent the primary actuators while the blue ones represent the redundancies.

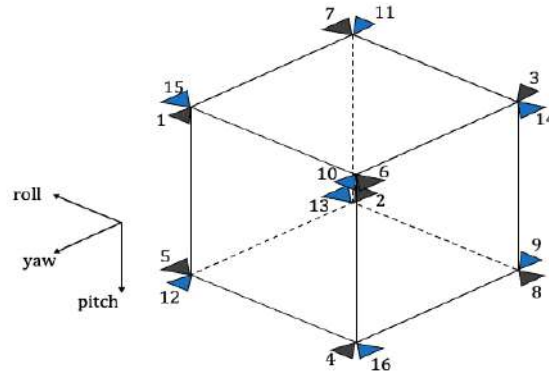


Figure 8.9. Thrusters assembly.

Thruster	ROLL		PITCH		YAW	
	+	-	+	-	+	-
Primary	[1,2]	[3,4]	[1,3] [6,7]	[2,4] [5,8]	[5,6]	[7,8]
Redundancy	[9,10]	[11,12]	[9,12] [13,16]	[10,11] [14,15]	[13,14]	[15,16]

8.2.5 Disturbances balancing

A simple model has been created to calculate a manoeuvre to correct the telescope's attitude due to different phenomena. Here the assumption made:

$$I_{sp} = 1600 \text{ s} ; \Delta t = 0.5 \text{ s} ; l = 3 \text{ m} ; M = 150 \mu\text{Nm}$$

Where l is the cube's side that round the body off. Furthermore, the momentum M has been taken from the sum of different kind disturbances related to LEO orbit:

- Gravity gradient;
- Solar radiation pressure;
- Magnetic dipole;
- Atmospheric drag.

Since the body is assimilated to cubic geometry, and the thruster are on the edge of this cube, the force could be calculated very simply:

$$\text{Yaw/Roll rotation: } M = F \cdot l \Rightarrow F = \frac{M}{l} = 50 \mu\text{N}$$

$$\text{Pitch rotation: } M = 2F \cdot l \Rightarrow F = \frac{M}{2l} = 25 \mu\text{N}$$

Where F is the force of each thruster: to underline that for pitching moment there are two couple of thrusters thus the force needed is half of the yaw or roll one.

With the specific impulse, the propellant mass to burn is given: $m_p = \frac{I_{tot}}{I_{sp} g}$

$$\text{Yaw/Roll} \rightarrow m_p = \frac{2 F \Delta t}{I_{sp} g} = 3.19 \cdot 10^{-9} \text{ g} ;$$

$$\text{Pitch} \rightarrow m_p = \frac{4 F \Delta t}{I_{sp} g} = 3.19 \cdot 10^{-9} \text{ g}$$

To improve an *orbit control*, the thrusters could be turned on the same face of the body and in the same direction: in this way it can be made a ΔV . Looking carefully, a translation around pitching axis can't be proved by the thrusters. There could be a manoeuvre that can achieve it: rotating around the yaw or roll axis with an angle of 90° and then actuate two devices in the demanded direction.

8.2.6 Second proposal: CMGs & thruster configuration

For low earth imaging spacecraft which demand high agility and spot-to-spot imaging, a high torque output is essential along with sufficient angular momentum. In order to meet these dual requirements, it is proposed to replace conventional reaction wheels with single gimbal control moment gyros (CMGs). CMGs are capable of generating more than 50 times the torque output of a typical reaction wheel. It also imparts adequate angular momentum to the spacecraft platform.

To guarantee the attitude control requirement our proposal consist of a configuration of 4 control moment gyros (CMGs) disposed as a pyramid cluster. In this way, the system has a redundancy in order to operate even if case of a failure.

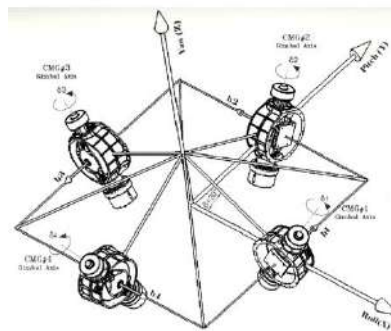


Figure 8.10. Schematic of 4-CMG pyramid cluster with all gimbal angles zero

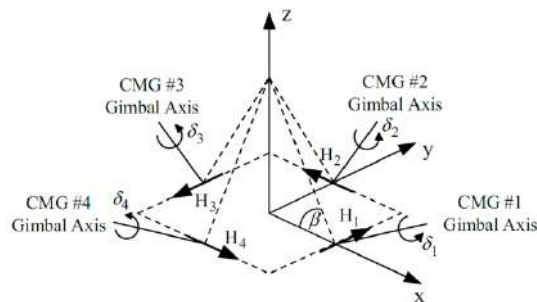


Figure 8.11. 4-CMG pyramid cluster with actual CMG

8.2.7 CMGs selection

Referring to the table of the existing control moment gyro (Appendix A), since masses and dimensions are not yet known, the choice fell on the device that provides the greatest range of angular momentum h : the M50 CMG model, manufactured by the Honeywell company.

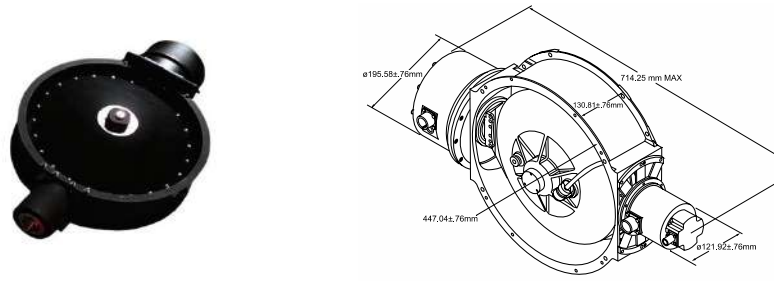


Figure 8.12. M50 CMG model.

Momentum h [Nms]	25 to 75
Torque [Nm]	75
Mass [g]	28000

After chosen this actuator we will consider the medium value of the momentum provided by the device: $h = 50$ Nms. This value will be useful in order to estimate the optimal slant angle β which satisfies the mission requirement.

8.2.8 Determination of the slant angle

The CMGs resultant angular momentum H could be decomposed along the three axes:

$$\mathbf{H} = \mathbf{H}_x + \mathbf{H}_y + \mathbf{H}_z$$

For our pyramidal cluster configuration:

$$\begin{aligned} [H_z]_{max} &= 4h \sin\beta \\ [H_x]_{max} &= 2h(1 + \cos\beta) \\ [H_y]_{max} &= [H_x]_{max} \end{aligned}$$

From which we are able to obtain the relation between the individual angular momentum h and the slant angle β :

$$\frac{[H_z]_{max}}{[H_x]_{max}} = \frac{4h \sin\beta}{2h(1 + \cos\beta)} = 2 \tan \frac{\beta}{2}$$

Applying these relations to multiple values of β , the following angular momentum values were obtained.

Pyramid Cluster			
β [deg]	H_x [Nms]	H_y [Nms]	H_z [Nms]
20	193,97	193,97	68,40
25	190,63	190,63	84,52
30	186,60	186,60	100,00
35	181,92	181,92	114,72

40	176,60	176,60	128,56
45	170,71	170,71	141,42
50	164,28	164,28	153,21
55	157,36	157,36	163,83
60	150,00	150,00	173,21
65	142,26	142,26	181,26

Having the angular momentum requirement, starting from the table we could be able to select the optimal slant angle.

Since we have not that requirement, a possible value of β could be 40° . This choice guarantees a high value of H_x and H_y , and a medium-high value of H_z . In fact, referring to our Earth observing mission, there is no need to have high momentum of H_z because this component is necessary just for attitude correction. Otherwise, H_x and H_y compete also for the pointing requirement.

8.2.9 Desaturation of moment gyros devices

Control moment gyros devices (CMGs) are used to momentum management of satellites. However, as they cannot spin at high rates due the limits fixed by structure. When CMGs are at their maximum total angular momentum, they need to be desaturated. This requires the use of reaction control system thrusters to maintain the desired satellite orientation, while the 4 CMGs return to their neutral position. This process, called desaturation or momentum dumping, can be done by thrusters or magnetic torquers; we decide to use thrusters because of their supplying of large and instantaneous torques at any point in the orbit.

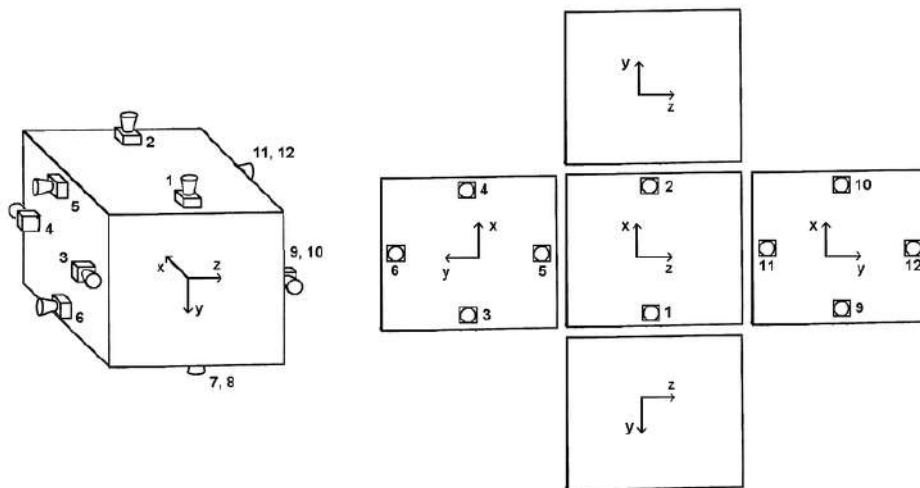


Figure 8.13. Thrusters' location and orientation for the assumed satellite

We assumed a configuration consisting of 12 thrusters: 4 acting in the direction of the x axis, other 4 in the direction of the y axis and 4 in the direction of the z axis. On each side of the satellite there will be two thrusters. The presence of 4 thrusters, oriented in each direction of the main axes, guarantees translational and rotational motion of the spacecraft on all, even in case of failure.

8.2.10 Sizing thrusters to desaturate CMGs

Before reaching saturation of the control moment gyros, it is necessary to dump the angular momentum excess. This can be done simply firing the thrusters in the opposite direction. The external torque produced by the thrusters must be equal to the maximum total angular momentum of CMGs.

$$2FLt = H_{max}$$

$$H_{max} = \sqrt{H_{x_max}^2 + H_{y_max}^2 + H_{z_max}^2} = 280,9 \text{ Nms}$$

We assume *thruster's moment arm* is $L = 1.5$ [m]. For each momentum-dump pulse we consider a *burning time* $t = 1.2$ [sec] obtained from literature.

$$F = \frac{H_{max}}{2Lt} = 78,03 \text{ N}$$

We need to choose a thruster that own this level of thrust. Referring to the table of existing thrusters, the choice falls on MONARC 90LT monopropellant thruster using hydrazine and produced by MOOG.



Figure 8.14. Monarc 90LT thruster

8.2.11 Desaturate manvre mass propellant estimation

The total impulse for desaturate 4 control moment gyros, using 2 thrusters, is

$$I_{tot} = 2Ft = 187,26 \text{ Ns}$$

Specific impulse for a Monarc 90LT thruster is $I_{sp} = 232,1$ s. The total mass of hydrazine required to desaturate the CMGs is

$$M_p = \frac{I_{tot}}{I_{sp}g} = 82,24 \text{ g}$$

8.2.12 Station keeping mass propellant estimation

Satellite are subject to various non-Keplerian forces and disturbance torques that affect their ability to maintain station in right position. However, for proper operation, the position and attitude of satellite must be maintained within tight specified ranges. Thus, to counteract the disturbance force and torques to meet these operational requirements the propulsion system is designed to have enough thrust and cyclical application.

Orbital station keeping maneuver is made by firing 2 thrusters in a single direction, taking longer than the desaturation maneuver. For this reason, we assume $t = 3 \text{ s}$ for on-time for each large maneuver pulse and we consider the steady state thrust $F = 90 \text{ N}$.

In this case, the total impulse is

$$I_{tot} = 2Ft = 540 \text{ Ns}$$

The total mass of propellant required for a single pulse is

$$M_p = \frac{I_{tot}}{I_{sp}g} = 237,16 \text{ g}$$

8.3 AOCS SYSTEM CHOICE

As a result of a meticulous trade-off between these two configurations, it appears clear to further the second one as the best option for the imposed requirements.

In fact, although the lower pointing accuracy, this configuration guarantees a longer lifetime which is essential to this kind of missions.

Right below it is shown a representative model of the chosen configuration created upon CATIA software.

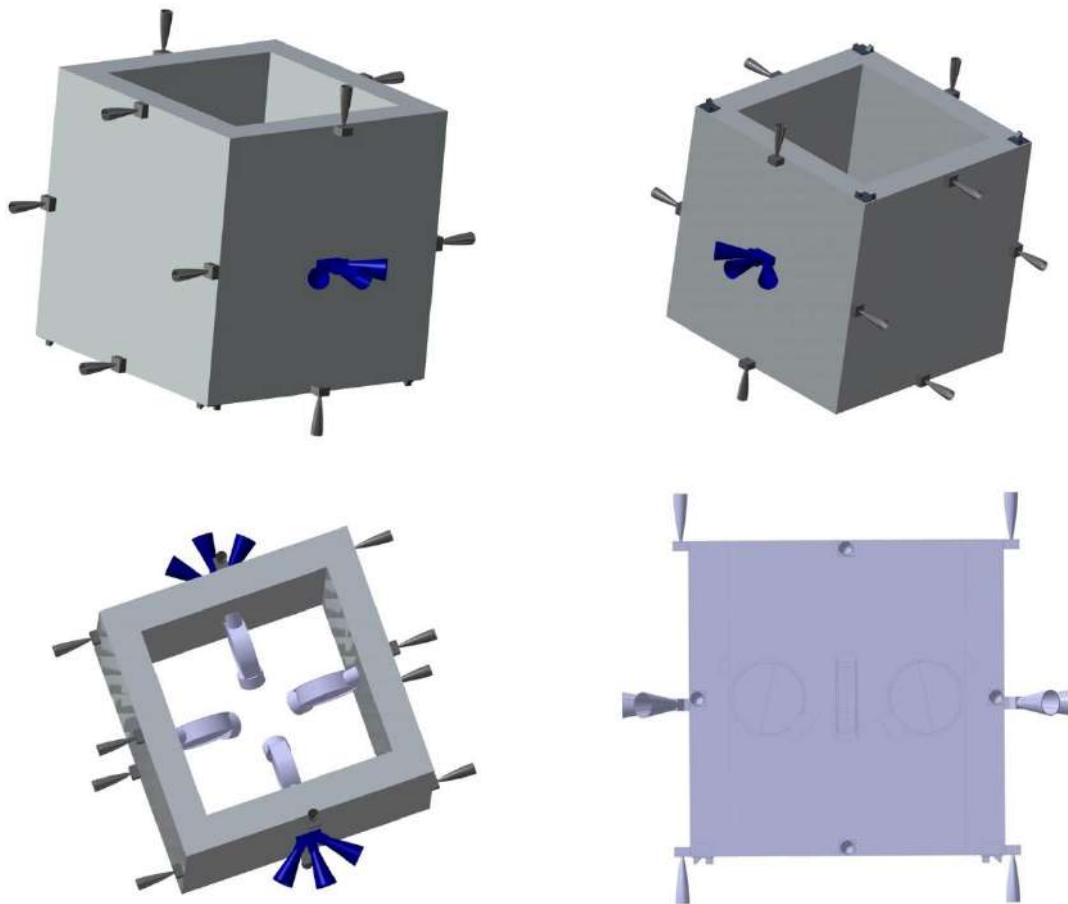


Figure 8.15. AOCS configuration concept.

8.4 CONTROL ALGORITHMS

In this section are analyzed possible algorithms for attitude and orbit control, focusing on the systems chosen before. In particular, thrusters' actuation is considered for orbit control, while a CMGs one is selected for attitude. These algorithms have been implemented in Simulink with the aim of guaranteeing the correct work of the ACDS.

8.4.1 Orbital control

A closed loop logic is chosen to provide the best orbital control. The feedback signal in input is the main characteristic of this method.

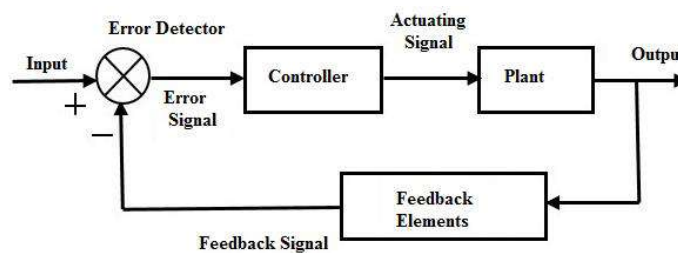


Figure 8.16. Closed loop logic.

This loop allows control of satellite desired position and velocity, in order to have a better point of view for onboard telescope. The difference between the desired vector and the state vector (output) generates an error. When the error passes through the controller, it's corrected by a signal sent to the actuators system. This correction drives the actuators to change satellite position or velocity.

The loop in the figure below describes how these maneuvers can be done. The starting point is y_r , the reference coordinates of the S/C that can be written either in physical trajectory coordinates or in orbital elements. This reference enters in the *NMPC law block*, a predictive algorithm of the position of the S/C, that supplies the thrust needed to reach the predicted point. This input enters in the *spacecraft dynamics block* where, by adding the *external disturbances*, the S/C actual state vector can be calculated. After adding the *measurement errors*, the vector is returned in feedback to the *NMPC law block*.

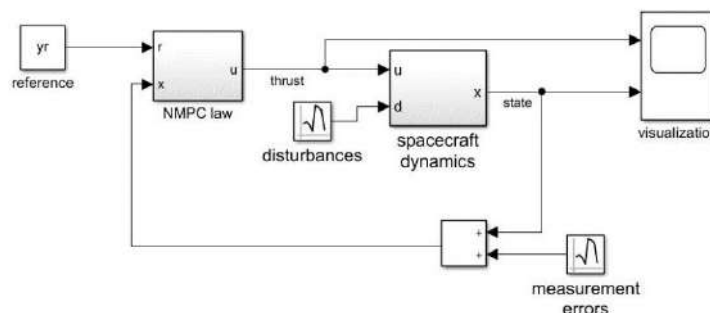


Figure 8.17. Maneuver algorithm.

Non-linear Model Predictive Control (NMPC) is the application of MPC algorithms to non-linear problems and it is finalized to orbital maneuvering/control. MPC, which stands for Classic Model

Predictive Control, uses an explicit dynamic plant model to predict future reactions' effect of the manipulated variables on the output and the control signal obtained by minimizing a pre-defined cost function. MPC operation can be schematized in the following figure.

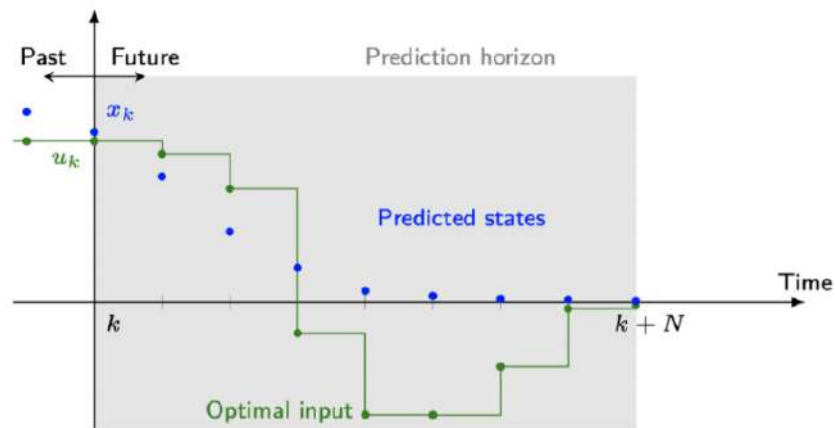


Figure 8.18. NMPC operation.

Model Predictive Control works in discrete time thus the problem is described by a discrete-time state space representation. At each time t , the optimal control sequence is found over a future horizon of N steps. Then, the problem is solved with the found solution. Afterwards, the first optimal move is applied, and the rest of the sequence is replied by a new optimization obtained by measurement at time $t+1$.

NMPC is not a classical approach for aerospace applications, but it gives some advantages:

- General closed-loop approach
- Systematic treatment of constraints (static or time varying)
- Efficient management of the trade-off performance/command activity
- Optimal trajectories (over a finite time horizon)
- Combined design of orbit/trajectory and control law for guidance and control
- Relatively simple to tune.

To improve law efficiency for proper orbital maneuver, the controller output can be defined in orbital parameters. In this way it is not required a reference trajectory, but it is found implicitly by NMPC, that now results easy to tune correctly. Using orbital parameters requires long time to obtain a high precision maneuver. To overcome this problem, it is possible to use either the NMPC algorithm in an early phase or a Gauss or Lagrange planetary equation to increase the precision.

8.4.2 Attitude control

Attitude control is implemented with a three axes control law, where controller is driven by feedback. Then, a filter transforms the torque signal into the real momentum applied by actuators. Other blocks are necessary to transform applied momentum in position. To gain a better precision in attitude control, CMGs replace classic actuation. This configuration is the best choice combined with a PD controller, since CMGs apply continuously variable momentum to the system, but they are necessary to consider the saturation limit of this method. For this reason, to control the actuators signal is used a low pass filter followed by a saturation one, as in following image. Thrusters are used as a

desaturation method controlled by an PWWF (*Pulse-Width Pulse-Frequency*) modulator to provide a quasi-linear steady state response. At least three CMGs are necessary to guarantee an efficient control.

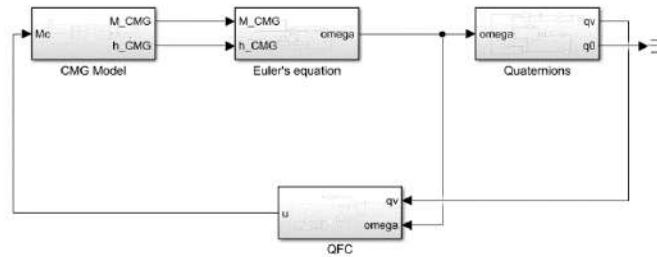


Figure 8.19. Attitude control law.

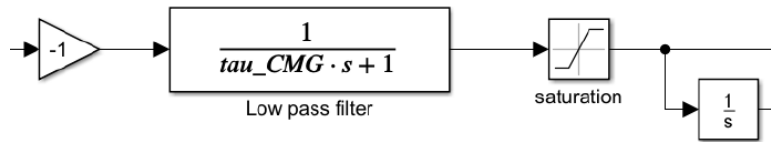


Figure 8.20. Filter stage.

As control law it is chosen Quaternion Feedback Control (*QFC*) to drive the satellite with a torque signal in the correct attitude. QFC presents relative simplicity in tuning and capacity to avoid other methods singularity error. Without making mistakes, it is possible to assume that the inertial reference frame coincides with the fixed desired attitude. Target attitude is defined as $(\pm 1, 0, 0, 0)^T$. While considering the proportional and the derivative part of the control, the command torque equation is defined as:

$$M_B = -\text{sign}(q_{\{0,des\}})K_p q_v - K_d \omega_b$$

The chosen controller gains are defined as:

$$K_p = [\alpha I + \beta \mathbf{1}]^{-1}$$

$$K_d = \text{diag}(d1 \ d2 \ d3)$$

Where K_p and K_d can be interpreted respectively as artificial stiffness and damping. The gain factors value influences the maneuver efficiency and stability as described in the following table, so it is important a correct tuning valued with the complete model of telescope.

	<i>Rise time</i>	<i>Overshoot</i>	<i>Settling time</i>	<i>Steady-state error</i>	<i>Stability</i>
<i>Increasing</i>	Decrease	Increase	Small increase	Decrease	Degrade
<i>Increasing</i>	Small decrease	Increase	Decrease	Minor change	Improve

8.4.3 External disturbances

The External Disturbance Torque due to orbit and to environment conditions will be considered for the calculation of the state vector of the system $\mathbf{x}$, for the considered orbit:

- **Solar radiation pressure:** $M_s = 10^{-5} \text{ Nm}$ $F_s = 10^{-5} \text{ N}$
- **Magnetic torque:** $M_{mg} = 10^{-6} \text{ Nm}$ $F_{mg} = \text{Negligible}$
- **Gravity gradient:** $M_g = 10^{-4} \text{ Nm}$ $F_g = \text{Negligible}$
- **Atmospheric Drag:** $M_a = 10^{-7} \text{ Nm}$ $F_a = 10^{-5} \text{ N}$

These are the typical torque and forces orders of magnitude. Torque was considered in attitude control, while forces in orbit control. The telescope operates in LEO (*Low Earth Orbit*), as usual for space telescopes: the atmospheric drag represents the main part of total disturbance force. Following this view, every kind of error added to the control loop is simulated as random signals with zero mean and standard deviation given as follows:

- *other perturbations/disturbances:* $\text{std}(\mathbf{d}(t)) = 0.01 \text{ kg} \cdot \text{km/s}^2$
- *measurement errors* $\mathbf{w} = (\mathbf{w}_r, \mathbf{w}_v, \mathbf{w}_m)$:
 - *position:* $\text{std}(\mathbf{w}_r(t)) = 10^{-3} \text{ km}$
 - *velocity:* $\text{std}(\mathbf{w}_v(t)) = 10^{-4} \text{ km/s}$
 - *mass:* $\text{std}(\mathbf{w}_m(t)) = 10^{-3} \text{ kg}$.

A complete treatment of this part is available in appendix, with a digression on possible laws to use and various correction manoeuvre types.

8.5 PRIMARY AND SECONDARY MIRROR

In order to achieve the imposed requirements, it has been chosen Mars observation as telescope's mission profile for the purpose of getting continuously more information about it. This conceptual design is based on various characteristics, starting from the choose of primary mirror structure to the number and type of different actuators.

First of all, it has been meticulously studied the arrangement for the primary mirror design; in fact, after an accurate analysis it has been selected a deformable primary mirror in a segmented structure. Advantages and disadvantages are multiple. For example, on a hand there is the possibility of having a stroke as large as needed with a reduced number of actuators (approximately 10^2), but in the other hand this technology is very expensive.

After several reviews, the primary mirror conceptual design is composed of twelve hexagonal-shaped mirrors arranged like the image below. It has, as sub-mirrors, regular one-meter-side hexagons, because of optimization of the space occupying a circular 4-meter diameter encumbrance in its deployed configuration.

Every sub-mirror has also 3 piezoelectric actuators each, model DTT35XS-SI-2, that allow to obtain a maximum out-of-plane rotation of 2.8 mrad around x-axis and y-axis, with the aim of modifying the relative position between different parts composing the primary mirror.

It is possible to notice that this kind of configuration leaves an empty space just at the center of the primary mirror that, with easy geometrical computations, appears to be extended 0.65 m^2 . This hole would define the space in which all deployment kinematics would be "stored" when they are extended, leaving at the center a window for the secondary mirror functions.

After this discussion about deployment and encumbrances, it has been selected two different types of technologies to apply to the secondary mirror.

First, it has 3 piezoelectric actuators, model TT60SM, that provide max z-translation of 58 μm and RX-Angular displacement of 11.70 mrad. This actuator specific choice is attributable at the different goals of the secondary mirror compared to the primary, in fact while the last is approximately blocked, unless any attitude maneuvers, the first one has the objective of analyze different targets in its FOV, so it is requested a more accurate precision and larger out-of-plane rotations to better follow and point the specific subject.

This secondary mirror mechanism structure has also implemented a thermal refocusing technology that, as seen in previous parts of this document, it could be a real improvement for image correction and capture.

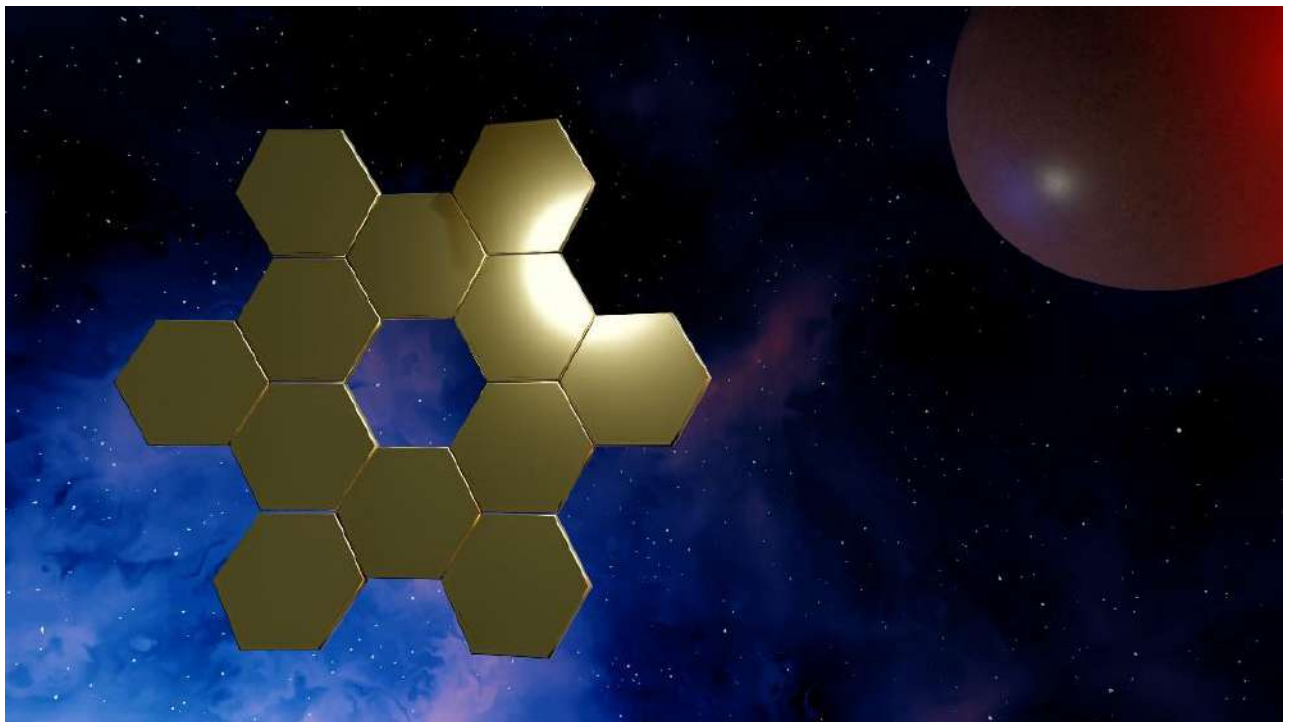


Figure 8.21. Primary Mirror Configuration deployed.

At [this link](#), it is possible to watch an animation about the Primary Mirror Configuration deployed.

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Appendix A: Disturbances in space applications

Considering the structure on which we are developing this work, we should be aware that it will be following a certain orbit during its life in space. The orbit should theoretically have a specific shape, however, there are elements in space itself, such as planets, stars or other bodies that make the trajectory of our structure change due to the attraction of these bodies or stars. This is what is known in orbital mechanics as a disturbance.

As an example of a disturbance, we could focus on the elliptical orbit followed by planets, such as the Earth's one around the Sun. The orbit in theory should be elliptical, however, as the planet moves along the orbit, this ellipse is deformed by disturbances.

We can distinguish the following classification considering the evolution of these as time progresses:

- Secular disturbances: These are disturbances whose growth is monotonous over time.
- Long-term periodic disturbances: These disturbances are periodic, as their name indicates, and the variation of these occurs with a long period, greater than the orbital period of the body that is disturbed.
- Short-term periodic disturbances: Unlike the previous ones, their variation over time occurs with a shorter period, generally on the order of the orbital period of the body that is disturbed.

It is important to say that although periodic disturbances can be quite interesting, generally the most important effects that can occur in space systems such as the one in this work are due to secular disturbances.

Finally, the following image shows the evolution over time of the disturbances described above:

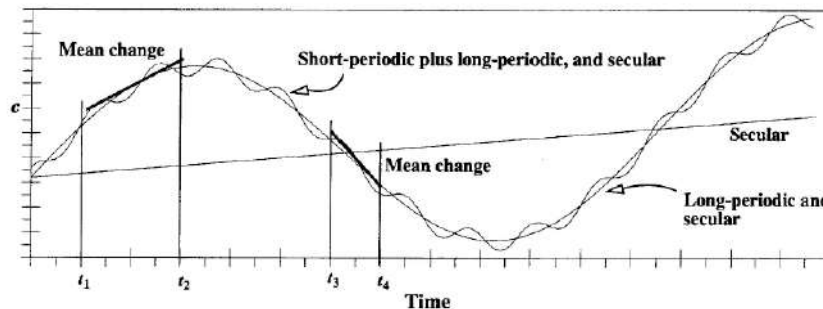


Figure A.1. Development over time of the different types of disturbances.

In addition to this type of classification, there is another one, perhaps more related to structure:

- External disturbances are those that occur around the vehicle, are dependent on the type of orbit developed and can cause elastic deformations in the structure or vibrations.
- Internal disturbances are those caused by some type of actuator belonging to the satellite systems.

Having said all this, we will go on to describe in a brief and clear way the main disturbances to which our structure may be subjected.

A.1 Disturbance due to flattening of the Earth

This type of disturbance is due to the shape of the Earth. While the gravitational field of a sphere is a uniform field, the gravitational field of a geoid is a totally asymmetric field. This makes it necessary to take into account the effects produced by the Earth in low orbits and try to obtain which orbital parameters are susceptible to being altered.

Through different calculations, it can be demonstrated that the components of this disturbance are inversely proportional to the radius of the orbit raised to 4, making the effect more “disturbing” in bodies located in low orbits.

It is necessary to indicate that the influence of the terrestrial flattening is shown on the right ascension of the ascending node, the argument of the perigee and the mean anomaly. Due to the elements’ changing, there are 2 disturbing effects:

- The regression of the nodes, produced by the variation of the right ascension of the ascending node, causes the plane of the orbit to change continuously.

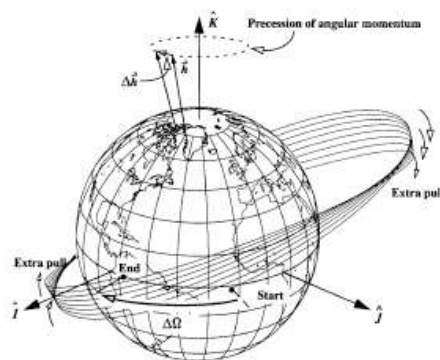


Figure A.2. Effect of node regression in the plane of the orbit.

- The variation of the perigee argument causes its advance to modify the geographical location of the line of apses, and therefore the points where the orbit remains longer (apogee) and less time (perigee).

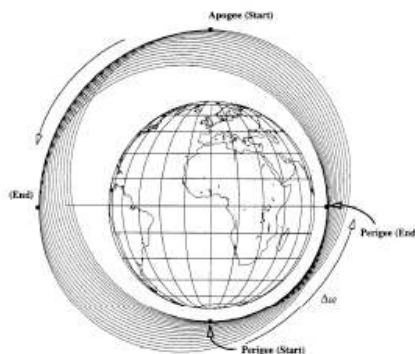


Figure A.3. Effect of variation of the perigee argument.

A.2 Aerodynamic disturbances

The atmospheric influence gives rise to an aerodynamic force in LEO orbits, below 1000 km altitude. This type of force can be considered as a disturbance, since depending on the external shape of the satellite, it will show some influence.

To take into account this type of disturbance, that is, the aerodynamic torques that can affect our structure, it would be convenient to consider these factors:

- Environment: we must take into account the atmospheric model when aerodynamic calculations are carried out during orbit
- Orientation of the surface of our structure: it is important that the external elements of the spacecraft are oriented as precisely as possible, in order to ensure that the smallest possible aerodynamic torques are generated
- Position of the center of pressure: depending on the orientation of our body within the orbit, the position of the center of pressures can vary. It is of vital importance to know this position when estimating the value of the aerodynamic torques that are produced in our structure
- Aerodynamic force: The flight configurations, the orientation of the vehicle in its orbit and the atmospheric environment are the most important factors that determine the net aerodynamic force that is exerted on the center of pressures.

A.3 Atmospheric resistance

Focusing on LEO orbits, one of the most influential disturbances that can appear is aerodynamic drag. Its effect is also important in orbits that present a high eccentricity with a low perigee.

The presence of this resistance causes a loss of energy in our structure, which should translate into a loss of altitude and consequently in a progressive degradation of the orbit, which can lead, if this disturbance is not corrected, to a process called reentry, that can lead to the destruction of our spacecraft.

For non-circular orbits, the initial effect that the disturbance would produce is to circularize the orbit. Once the orbit is circular, the disturbance would decrease the radius of the orbit in a slower way, producing a spiral fall until our structure re-enters the Earth.

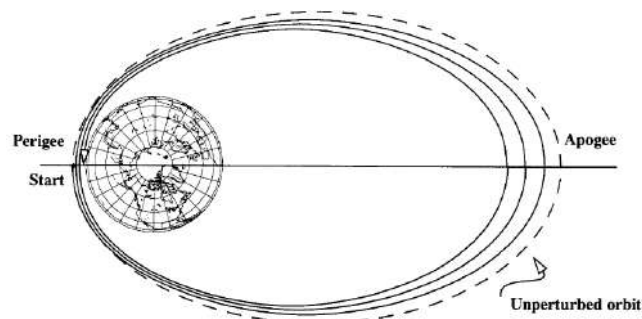


Figure A.4. Effect of declining apogee radius.

A.4 Solar radiation pressure

Due to the incidence of sunlight (photons) on the surface of a body, a mechanical effect such as the pressure of solar radiation is produced. This type of disturbance generates a force on bodies and / or structures that varies with the inverse of the square of the distance from the Sun. The pressure exerted by the solar radiation is independent of the altitude of the orbit above the Earth. It is a constant-type force, although it can be cyclical in Earth-oriented vehicles. Its effect is related to the geometry of the vehicle and the surface affected by the radiation. It is important to take into account the fact that when an eclipse occurs, that force disappears.

The influence of this disturbance is reflected in the variation of the right ascension of the ascending node and the argument of the perigee.

A.5 Magnetic field

Since our structure will generate a residual current, it can be said that magnetic-type torque will appear as a consequence of the interaction between the Earth's magnetic field and the residual currents generated on board. Magnetic type sources can be the telescope's magnetic moments, eddy currents or simply hysteresis. The last two magnetic components are produced by the rotational movement of our structure. In particular, hysteresis is produced by the magnetization of the permeable materials on board the satellite.

A.6 Control equipment

In order to keep the orbit under control, knowing and adjusting its parameters, different technologies has been implemented. We'll first take a look to the systems that allow has to continuously obtain information about our orbit, and then we'll analyses the on-board actuators systems that correct the previously listed perturbations.

A.7 GPS

The Global Positioning System (GPS) is a space-based radionavigation system, owned by the U.S. Government and operated by the United States Air Force (USAF). It can pinpoint a three-dimensional position to meter-level accuracy and time to the 10-nanosecond level, worldwide and 24/7. GPS is comprised of three different segments: the space segment, control segment and user segment.

Today the GPS constellation (the space segment) consists of over 30 operational satellites, each equipped with redundant atomic clocks and tracked by a ground control network (the control segment). Each satellite transmits its position and time at regular intervals and those signals are intercepted by GPS receivers (the user segment). The receiver is able to determine its position by calculating how long it took for the signals to reach it.

This increases spacecraft autonomy, enabling new methods of spaceflight operations and reducing the burden on NASA's tracking stations. Worldwide government and commercial spacecraft launch projections over the next two decades show that approximately 60% of future missions will operate in low-Earth orbit (LEO), and 95% of missions will operate at or below geosynchronous orbit (GEO). Many of these space users could meet their real-time navigation needs through GPS.

A.8 LIDAR

A LIDAR (Light Detection and Ranging) is an active remote sensing device that is typically used in space applications to obtain the range to one or more points on a target spacecraft. As the name suggests, LIDAR sensors use light (typically a laser) to illuminate the target and measure the time it takes for the emitted signal to return to the sensor. Because the light must travel from the source, to the target object, and back to the detector, the range to the observed point may be ideally computed as,

$$r = \frac{ct}{2}$$

Where ' r ' is the range from the sensor to a point on the target object, ' c ' is the speed of light and ' t ' is the laser time-of-flight (ToF).

There are various practical concerns that complicate this equation in practice.

Most LiDAR systems use four main components:

- Laser
- Scanners and optics
- Photodetector and receiver electronics
- Navigation and positioning systems

LIDAR sensors may be divided into a few different categories based on how they illuminate and sense the 3D scene in front of them. Broadly speaking, we propose that most current LIDAR systems can be categorized into three major groups:

1. *Scanning*: systems that illuminate the scene through sweeping of a “narrow” laser beam and sense the return with only a single detector
2. *Detector Arrays*: systems that illuminate the entire scene at once and sense the return with an array of detectors
3. *Spatial Light Modulators*: systems that illuminate portions of the scene in a pattern and then sense the return with a single detector (along with compressed sensing algorithms)

For the first two groups (Scanning and Detector Array), the system measures the laser ToF through one of a few common mechanisms. The first method is about sending out discrete laser pulses/flashes and then wait the return pulse at the detector. The second method modulates a signal onto the laser and then track the phase shift in the returning signal to measure laser ToF. The third method consist into encoding a pseudo-random number (PRN) sequence onto the laser and then perform an autocorrelation with the sensed return to determine laser ToF.

ToF Measure Method	Scanning	Detector Array	Spatial Light Modulator
Pulse/Flash	X	X	
Continuous Wave (CW)	X	X	
Pseudo-Random Number (PRN)	X	X	
Compressed Sensing (CS)			X

A.9 Optical System

These are cameras that work not only in the visible, but also in the near infrared, so starting from the images they obtain, an estimate of the navigation parameters is made using computer vision techniques (i.e. image manipulation algorithms), from which it is possible to define characteristics within the image that allow to extract information about the various objects in the image itself. Some examples of algorithms that can be used in the optical cameras are:

- The estimation of the centroid (the one that gives a lower accuracy) that provides to see inside the image what is the contour of a certain figure.
- The feature detection that allows to define on the spacecraft some characteristic points called key points and, starting from the position that they assume, you can apply triangulation algorithms to reconstruct the object.

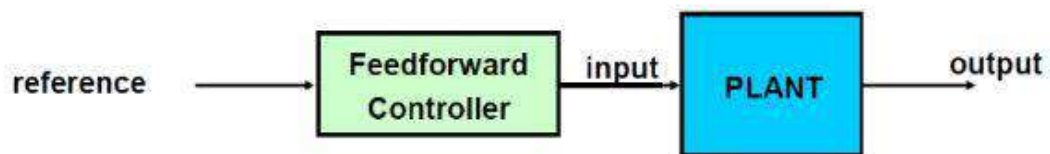
The marker detection that is used in the last stages of the approach, when you are very close to the target on which markers are placed (which can be bright points rather than objects with particular shapes or colors) that are taken by the camera: when you know the position of the markers on the target and what their size is, you make a comparison between the size you see in the image and the size you know. In this way you can define a much more accurate distance between the two objects.

Appendix B: Control algorithms

In this chapter we are going to explain some control algorithms typically used in space engineering. In particular we will focus on two different types of control: orbit control and attitude control. These controls will act on the three axes of the satellite changing its position and orientation in space. The logic behind them is the one of open and closed loop.

- **Open Loop**

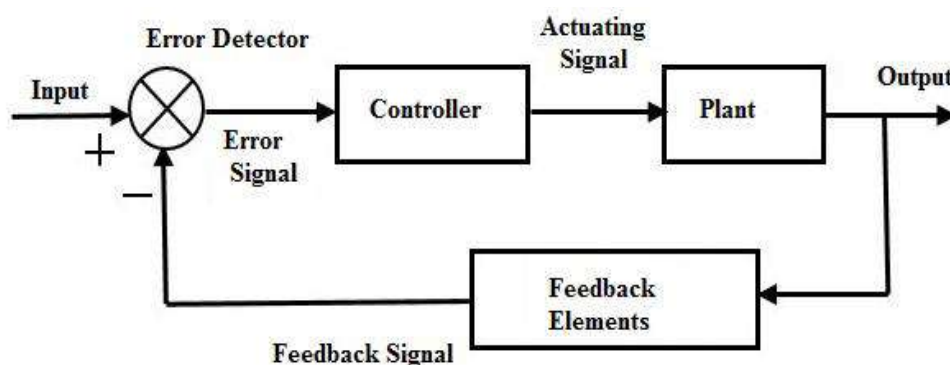
Open loop algorithms are the representation of a mathematical model in which the output has no influence on the control action of the input signal. The response of the system is, therefore, the combined response of the plant and the controller, excluding the effect of a feedback system, that we can find in closed loop algorithms.



This logic has a great limitation: if the model is perfect then we'll have no problems, but if not (and this is what always happens in real life systems), mistakes can't be avoided, since the loop has no way to correct them.

An example of an open loop system is the sprinkler.

- **Closed Loop**



The closed loop logic is similar to the open loop one, but here we add a feedback signal to the input. This lets us control the desired positions and velocities of our satellite, in order to have a better point of view for our telescope onboard. By the difference between the desired vector and the state vector (output) measured by the sensors, we generate an error that will pass

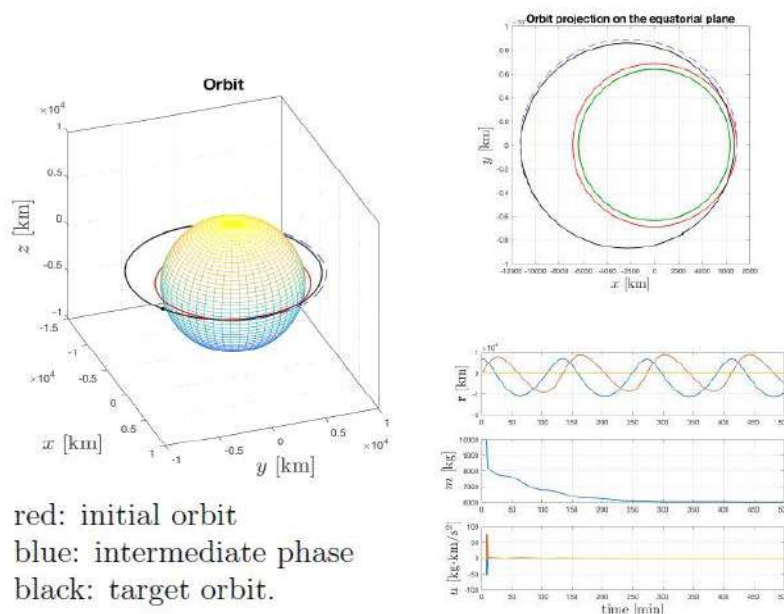
through the controller. At the end, the controller will correct the error by sending a signal to the actuators system that will change the position or the velocity of the satellite. For our aims we will certainly use this kind of logic that corrects the errors generated by the action of disturbances on our telescope in order to make it more precise.

Dynamical systems can be various, but in this case we focus on linearized, second order systems, described by a PD controller, that stands for proportional-derivative, meaning that the feedback algebraically summed to the input consists of a proportional term and a derivative term. The former acts on the rise time, but leaves the steady state error invariant, the latter decreases the overshoot improving the transitory term.

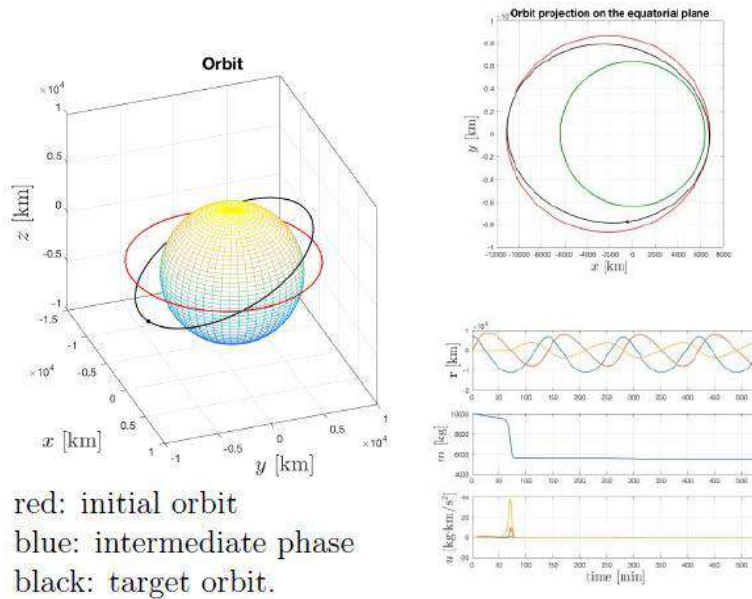
B.1 Orbit Control

Orbit control algorithms have six main purposes:

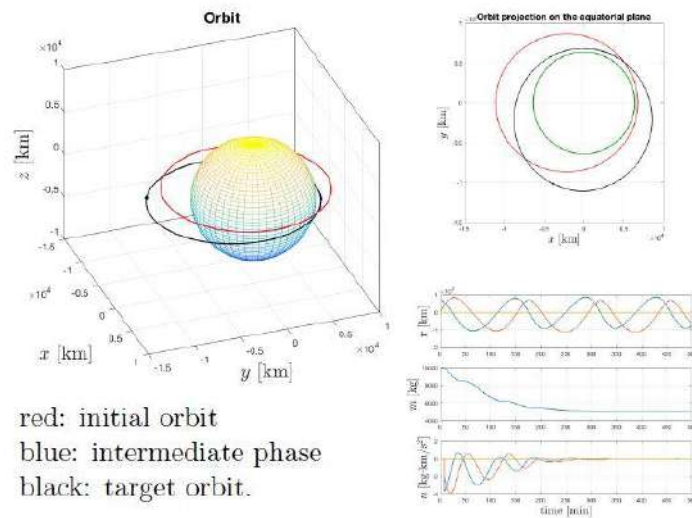
- **Shape change:** the shape of the orbit is changed, maintaining the orbital plane. Typically, it is done with the Hohmann transfer, a maneuver consisting of two phases, the first to go from a starting orbit to a transfer orbit and the second to go from the latter to the target orbit.



- **Inclination change:** the velocity of the satellite is changed in order to change the orbital plane. This is going to need large amounts of propellant, since the typical velocity of a satellite is very large.



- **Orbital plane rotation at constant inclination:** rotating the orbital plane around, while keeping the inclination fixed (change the right ascension of the ascending node Ω).
- **Period change:** the orbital period depends on the altitude and shape of the orbit. Maneuvers to change the shape and altitude of the orbit can be used to change the period.

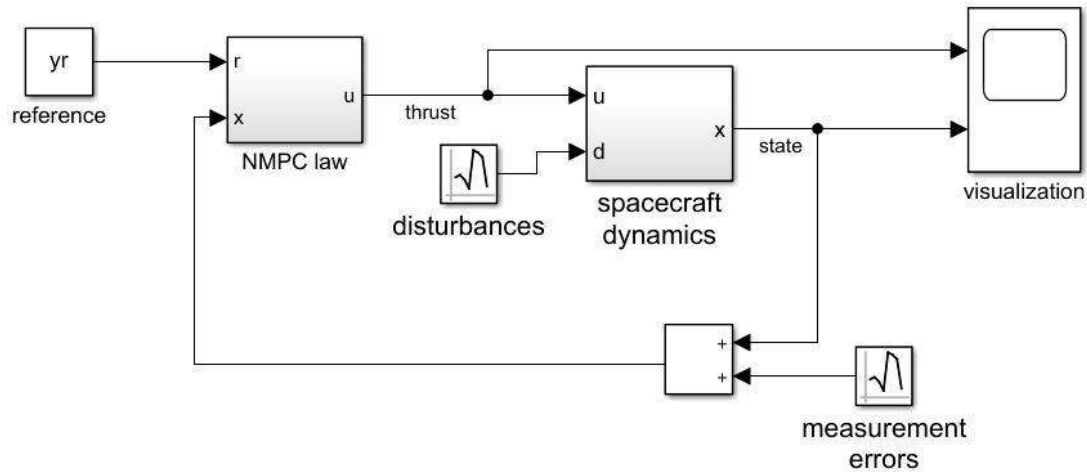


- **Relative location changes in the same orbit:** changing the speed does not work, since that changes the orbit. The satellite must be placed temporarily into a higher or lower orbit to change its period, and then moved back to the original orbit.
- **De-orbiting:** Thrusters are used to accelerate out of orbit and back toward the Earth.

The loop in the figure below describes how this manoeuvres can be done:

the starting point is y_r , the reference coordinates of the S/C that can be written either in physical trajectory coordinates or in orbital elements. This reference enters in the *NMPC law block*, a predictive

algorithm of the position of the S/C, that gives us the thrust needed to reach the predicted point. This input enters in the *spacecraft dynamics block*, in which, adding the *external disturbances*, we can calculate the actual state vector of the S/C. This vector is returned in feedback to the NMPC law block, adding the *measurement errors*.



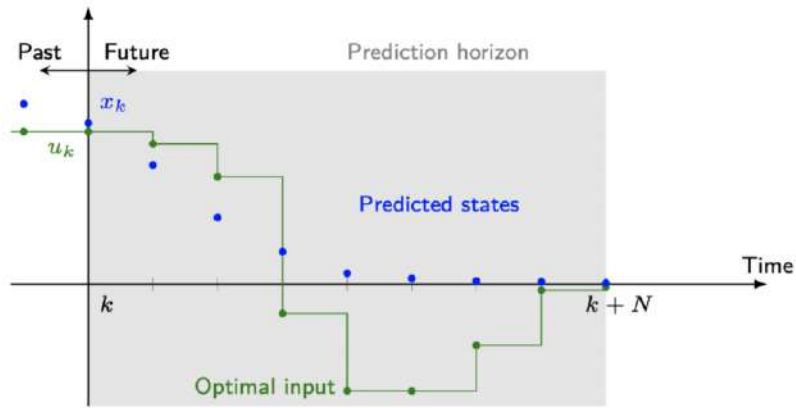
B.1.1 Measurement Errors

Every kind of error added to the control loop is simulated as random signals with zero mean and standard deviation given as follows:

- other perturbations/disturbances: $std(d(t)) = 0.01 \text{ kg} \cdot \text{km/s}^2$
- measurement errors $w = (w_r, w_v, w_m)$:
 - position: $std(w_r(t)) = 10^{-3} \text{ km}$
 - velocity: $std(w_v(t)) = 10^{-4} \text{ km/s}$
 - mass: $std(w_m(t)) = 10^{-3} \text{ kg}$.

B.1.2 NMPC Law

Non-linear Model Predictive Control (NMPC) finalized to orbital maneuvering/control is the application of MPC algorithms to non-linear problems. MPC, which stands for Classic Model Predictive Control, uses an explicit dynamic plant model to predict the effect of future reactions of the manipulated variables on the output and the control signal obtained by minimizing a pre-defined cost function. MPC operation can be schematized in the following figure:



Model Predictive Control works in discrete time that evolves in discrete increments k so that the problem is described by a discrete-time state space representation. At each time t , the optimal control sequence is found over a future horizon of N steps then the problem is solved with the founded solution. Next, the first optimal move is applied and the rest of the sequence is replied by a new optimization obtained by measurement at time $t+1$ and so on.

NMPC is not a classical approach for aerospace applications, but presents some advantages wrt classical approaches:

- General closed-loop approach
- Systematic treatment of constraints (static or time varying)
- Efficient management of the trade-off performance/command activity
- Optimal trajectories (over a finite time horizon)
- Combined design of orbit/trajectory and control law for guidance and control
- Relatively simple to tune.

To improve the efficiency of the law for proper orbital maneuver, the output of the controller can be defined in orbital parameters. In this way it is not required a reference trajectory, but it is found implicitly by NMPC, that now results simply to tune correctly. The disadvantage of using orbital parameters is that a long time is required to obtain a high precision maneuver. To overcome this problem, it is possible to use the NMPC algorithm in an early phase to rapidly get the desired orbit or use a Gauss or Lagrange planetary equation to increase the precision.

B.1.3 External Disturbances

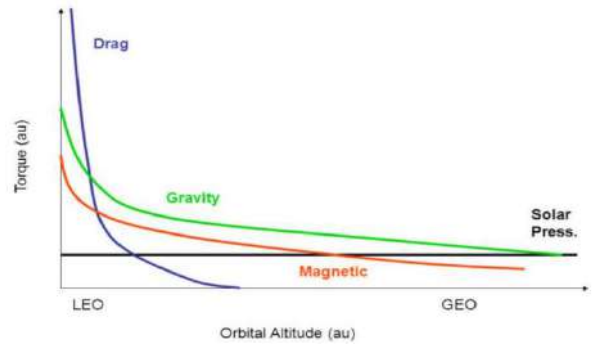
The block represented below is the Random Number generator, that generates normally distributed random numbers in order to approximate the External Disturbance Torque due to orbit and environment conditions. The variable $\mathbf{d}$ represents the sum of all contributions of the external disturbances and in the Simulink block a maximum is fixed. The disturbances will enter in a matlab function and they will be considered for the calculation of the state vector of the system $\mathbf{x}$.



The external disturbance torque is given by four main contributes:

- **Solar radiation pressure:** the photons from the sun product an effective pressure on spacecraft surfaces that generates a momentum;
- **Magnetic torque:** generated by the effect of the residual dipole and the interaction with Earth's magnetic fields;
- **Gravity gradient:** due to Earth's gravitational field, which is not uniform;
- **Atmospheric Drag:** represented by the aerodynamic drag due to the presence of neutral particles in the upper levels of the atmosphere, it is proportional to the spacecraft wetted area.

The telescope will operate in LEO (*Low Earth Orbit*), as usual for space telescopes: the atmospheric drag represents the main part of total disturbance torque, as we can see in the graph below.



B.1.4 Spacecraft Dynamics

The model describes the main aspects of a spacecraft's behavior in all the phases. In case of launch, it refers to a single stage vehicle. The discontinuous decrease of mass for multi-stage vehicles can be included. According to classical orbital control, the approach is based on the Δv , that is the change of speed required to reach a desired orbit. Satellites are typically controlled via ground operations with, at first, the ideal Δv calculation and then the Δv actuation by means of thrusters.

The Dynamic model is represented below:

$$\begin{aligned}\dot{\mathbf{r}} &= \mathbf{v} \\ \dot{\mathbf{v}} &= -\mu \frac{\mathbf{r}}{r^3} + \frac{1}{m} (\mathbf{F}_d + \mathbf{d} + \mathbf{u}) \\ \dot{m} = f(x) &= \begin{cases} 0, & \mathbf{u} = \mathbf{0} \\ -u/v_e, & \mathbf{u} \neq \mathbf{0} \end{cases} \\ F_d &= -\frac{1}{2} \rho C_D S v \mathbf{v}, \rho = \rho_0 \exp\left(-\frac{r-r_0}{H}\right) \\ \mathbf{u} &= \mathbf{0} \quad \text{if} \quad m \leq m_b\end{aligned}$$

As input variables we have:

- $(\mathbf{r}, \mathbf{v})$: S/C position and velocity,
- $\mathbf{u}$: rocket thrust,
- $\mathbf{F}_d$: atmospheric drag,
- ρ : atmosphere density,
- $\mathbf{d}$: other disturbances,
- m : S/C mass (body + propellant).

The parameters are:

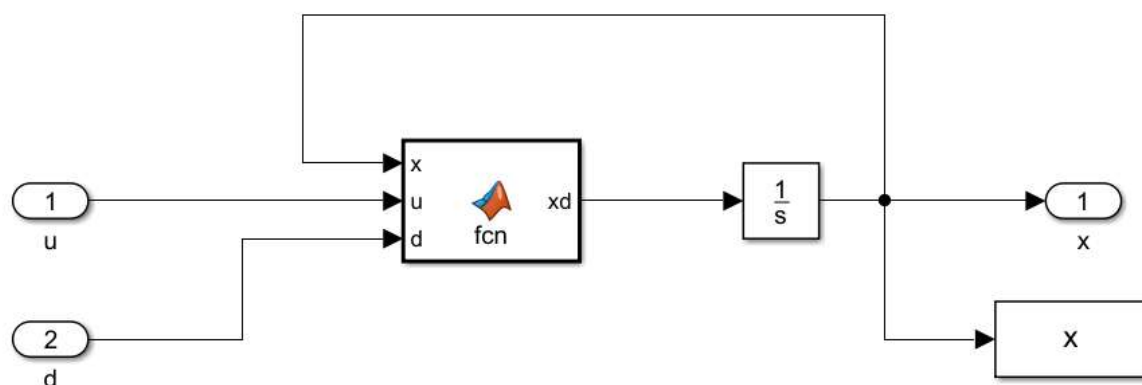
- μ : gravitational parameter,
- C_D : drag coefficient,
- H : scale coefficient,
- S : S/C projected area,
- ρ_0 and r_0 : reference density and height,
- v_e : effective exhaust velocity,
- m_b : body mass.

Furthermore $\mathbf{r}$, $\mathbf{v}$ and m are grouped in the state vector $\mathbf{x}$; $\mathbf{u}$ is the command input vector; $\mathbf{F}_d$ e $\mathbf{d}$ are the disturbances. All vectors are expressed in an Earth-centered inertial frame.

The output of the analysis control can be:

- A physical trajectory (tracking/set point maneuvers) or
- Constants of motion/orbital elements (proper orbital maneuvers).

The subsystem is represented in the simulink model below:



As we can see in the figure the inputs are the command input vector, the disturbances and the state

vector given to feedback after the integration in every iteration, and the output is the state vector itself. The *dynamics block* contains the function that implements the equations above.

B.1.5 Orbital Parameters

At the end of the Simulink model, we find the function *Orbital element calculation*, that lets us calculate some orbital parameters which we are probably interested in, for example the semimajor axis of the elliptical orbit (a), the eccentricity vector of the orbit ($\mathbf{e}$) and the inclination of the orbit plane (i). Below we can see some formulas used to reach this target:

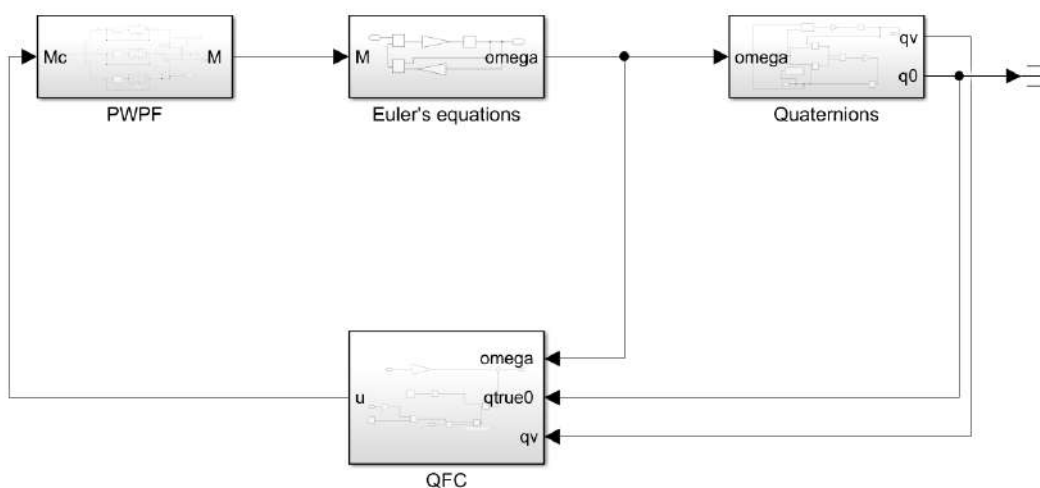
$$\mathbf{y} = (a, \mathbf{e}, ci) \in \mathbb{R}^5$$

$$\mathbf{h} = \mathbf{r} \times \mathbf{v}, \quad \mathbf{e} = \frac{1}{\mu} \mathbf{v} \times \mathbf{h} - \frac{\mathbf{r}}{r},$$

$$a = h^2 / (\mu(1 - e^2)), \quad ci = \cos i = \hat{\mathbf{K}} \cdot \mathbf{h} / h$$

where $\mathbf{y}$ is the vector containing the calculated orbital parameters. At the end we will compare this vector with a reference vector containing the reference orbital parameters.

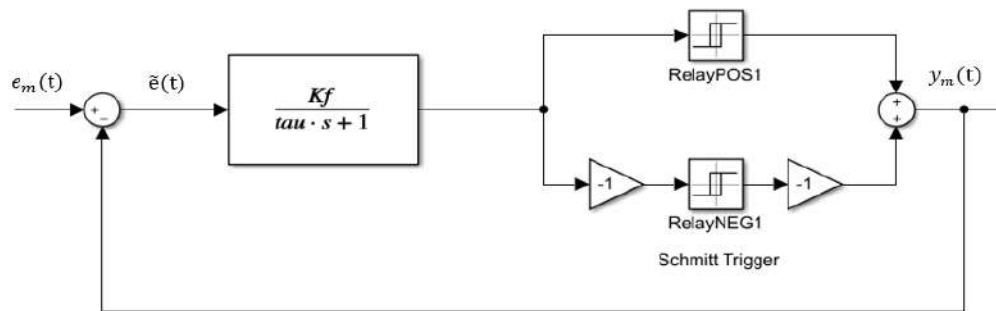
B.2 Attitude Control



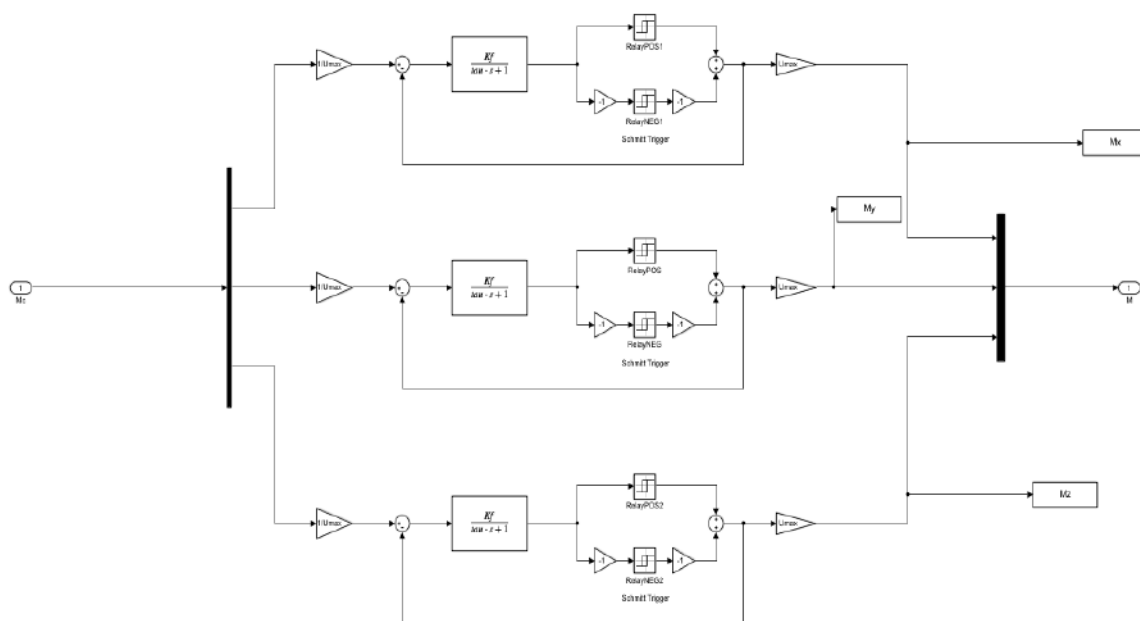
Attitude control is implemented with a control law for three axes. At the beginning it can be considered a closed loop with a *PWPF* modulator to obtain the torque command in considered axes to drive the satellite towards the prescribed direction.

B.2.1 PWPF Modulator

A Pulse–Width/Pulse–Frequency modulation is implemented with a *PWPF* modulator, which is composed of a *Schmitt Trigger* which consists of a double relay with hysteresis separated by a dead band. In addition, to provide a quasi-linear steady state response a modulator low pass filter is added represented by a linear first order system. A representation of the inner loop is in the following figure.



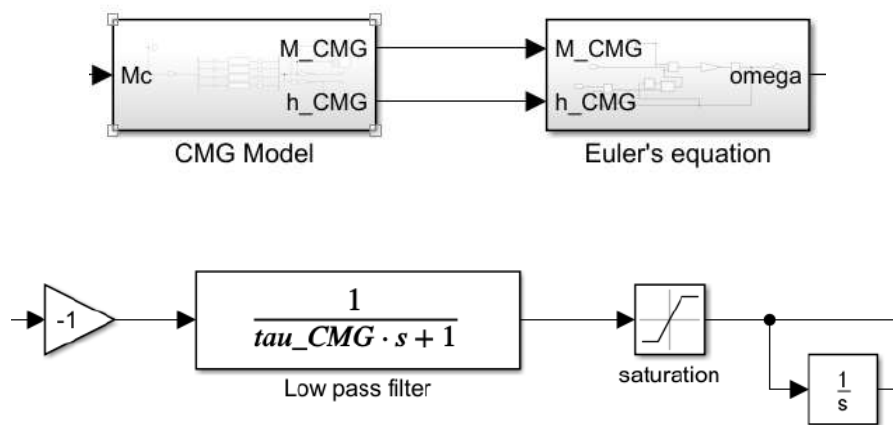
The signal is driven by the difference between the error e_m and the output y in accordance with the consideration that torque command is proportional to the error defined as the difference between actual and desired position. Instead, the output remains zero until the exit signal from the filter remains inside the activation threshold, named deadband, specified in the relay block. When the signal gets at the limit of this band, thrusters are activated by the trigger and deactivated when it is brought back in the limits.



For a three-axes control is implemented a *PWPF* modulator for every axis to obtain a command torque from the initial torque defined by control law.

B.2.2 PWPF modulator with CMG

In order to gain a better precision in attitude control, CMG can be replaced by a classic thruster actuation. This configuration is the best choice combined with a PD controller, since CMGs actuators apply continuously variable moments to the system, but they are necessary to consider the saturation limitation of this method. In the considered model four reaction wheels in a pyramidal configuration are necessary to guarantee an efficient control. In the following images it is possible to see how the modulator changes considering the new model.

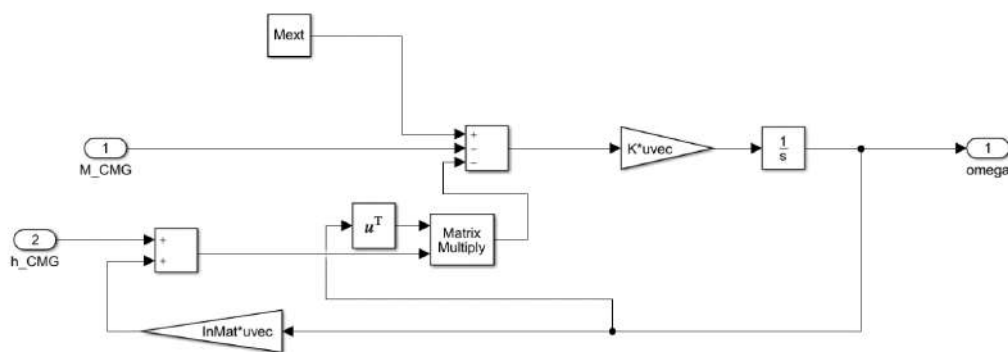


In this case the low pass filter is the same and the saturation block performs like the Schmitt trigger to maintain the control signal in the required band. Unlike the classic actuators, CMGs model give as output control momentum and torque, required by this type of actuators, and Euler's equation is modified as presented after.

B.2.3 Equation of Motion of a Rigid Body

In this section is implemented the Euler's rotation equation starting from the momentum $\mathbf{M}_B$, in which are considered all the moments created by actuators as thrusters, reaction wheels and environment, and the inertia's matrix:

$$\dot{\omega}_B = I^{-1}(\mathbf{M}_B - \omega_B \times (I\omega_B + \mathbf{h}_{RW}))$$

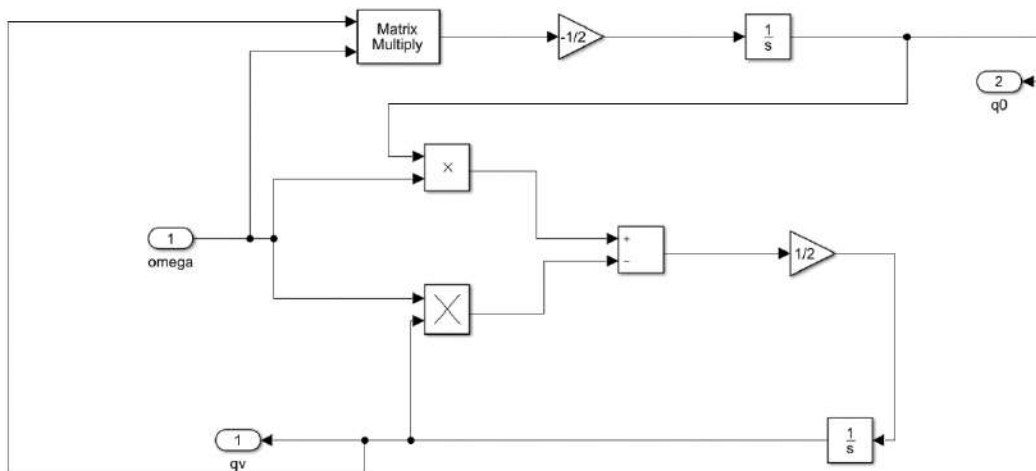


B.2.4 From Omega to Quaternions Block

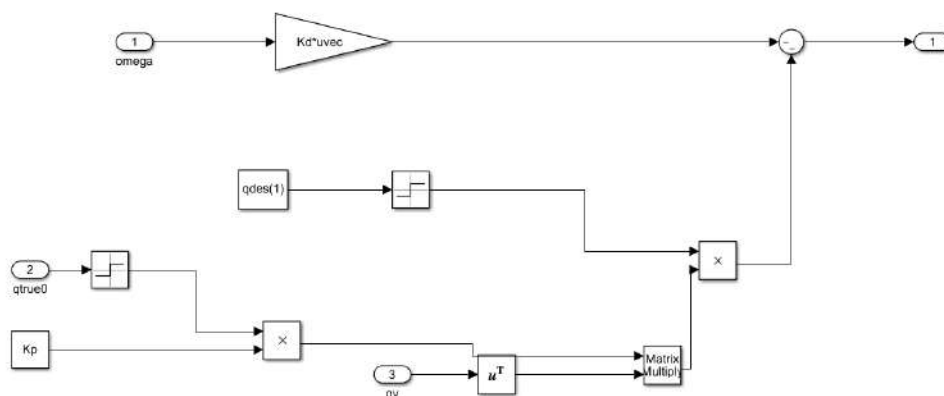
This block in the Simulink model is needed to convert the angular velocities, defined in a body reference, into four Euler's parameters (quaternions), that are the scalar parameter q_0 and the three components of $\mathbf{q}_v$. This representation has several advantages over Euler angles, in particular for the absence of singularities and because it's less computationally expensive.

The formulas used in the calculation are:

- $\dot{q}_0 = -\frac{1}{2} \cdot \boldsymbol{\omega}_B \cdot \mathbf{q} + k \cdot (1 - q_0^2 - \mathbf{q}^T \mathbf{q}) \cdot q_0$
- $\dot{\mathbf{q}} = \frac{1}{2} \cdot (q_0 \boldsymbol{\omega}_B - \boldsymbol{\omega}_B \times \mathbf{q})$



B.2.5 Quaternion Feedback Control



In this block is implemented a three – axis closed loop system with Quaternion Feedback Control (QFC). It is possible to assume, without making mistakes, the inertial reference frame coincides with the desired attitude when the desired attitude is fixed. The inputs are omega and quaternions, the

Euler's parameters. In case of small angles or a target attitude of $(\pm 1, 0, 0, 0)^T$, the equation of the command torque, considering the proportional and the derivative part of the control, is defined as:

$$\mathbf{M}_B = -\text{sign}(q_{0,d}) \cdot \mathbf{K}_p \cdot \mathbf{q}_v - \mathbf{K}_d \mathbf{w}_b$$

Gains are defined as:

$$\mathbf{K}_p = k \text{sign}(q_{true0}) \mathbf{1} \quad \text{and} \quad \mathbf{K}_d = \text{diag}(d_1, d_2, d_3)$$

Where $\mathbf{1}$ is the 3x3 identity matrix, k and d_i are positive scalar constants, $\text{sign}(q_{true0})$ is the sign function.

The effects of gain factors' variation $\mathbf{K}_p$ and $\mathbf{K}_d$ are described in the following table.

	<i>Rise time</i>	<i>Overshoot</i>	<i>Settling time</i>	<i>Steady-state error</i>	<i>Stability</i>
<i>Increasing</i>	Decrease	Increase	Small increase	Decrease	Degrade
<i>Increasing</i>	Small decrease	Increase	Decrease	Minor change	Improve

Appendix C: Devices market research

Model	Sensor Type	By	Measurement range[nT]	Lifetime [years]	Operating band[μm]	Earth acquisition mode[km]	Operational capability [km altitude]	FOV[deg]	Accuracy	Update rate[Hz]	Mass[kg]	Dimensions[mm]	Operating temperature[°C]	Supply Voltage[V]	Max Power consumption[W]
IRES-M2	IRES	Leonardo		>15	14-16,25	15,300-53000	up to 140000	1ra 5,5°2,5 a 23°14	<0,05deg random error	10	<2,5	169,9*163,8*156	-30 a +60	24 - 50	<4
ESA	IRES	NFC									1,55	121*157*123		31,0-50,0 Vdc	4
Digital Earth sensor	IRES	Sitael			7,5-14,5			37°44	<1deg	8	0,4	126*72*52	-25+70	15-50	<2
MIDES-G	IRES	Servo		15	14,6-19,8			33	0,025deg		1,5	133,33*133,35*128,78	-40+60	28Vdc	<4
LESA-T	IRES	Servo		7	14-16	910-2000	910-2000	48 16 arrays	+0,2deg		2,4		-20+55	22-40Vdc	
ESFS	IRES	Meisei						33°4arrays	1deg		0,25	40*60*55	-25+60	5	<1
MSS	Magnetometer	Meisei	+64000						1deg		<0,22	100*50*40	-30+60	10+34	<1
SX-MAG-04	Magnetometer	Sputnix	+100000						100nT		0,67	70*70*20	-80+60	6+36	0,8
MSS-0	Magnetometer	OCE	+60000	>5					0,05nTps		0,7	78*88*30	-30+60	12	<140mW
MAC_3	Magnetometer	SpaceQuest							0,75mTps		0,1	35,1*32,3*32,6	-55+85	15+34	
DMW	Magnetometer	Antrix Corporation Limited	+60000						+30nT				-25+65	+14+16	1,5
RMS-100	Magnetometer	PHI	+11.000.000						23nT	147			-40+85		
NVRM-Bn 25 or 85	Magnetometer	NewSpace Systems	+60000						1deg	<18	<0,085	99*63*17	-25+70	5dc	<0,75

Reaction and Momentum Wheels

Model	By	Lifetime [years]	Speed Range [rpm]	Speed control accuracy [rpm]	Max wheel torque [mNm]	Max wheel momentum [Nm.s]	Dimensions [mm]	Mass [g]	Operating Temperature [°C]	Supply Voltage [V]	Max Power Consumption [W]
Flywheel	Chang Guang Satellite	5	4000	±1 (speed >1000rpm)	≥50	2	160x160x120	3000±500	-10 to +45	28	≤70
NRWA - T005	NewSpace Systems		-8000 to +8000	<1	10	0.45	75x75x73	<1200	-30 to +60	28	
NRWA - T005			-5000 to +8000	<1	20	0.45	102x102x105	1550	-30 to +60	28	
NRWA - T10			-5000 to +1000	0.6	210	10.6	235 diameter x 73	<1000	-20 to 60	28	
RW - 40NMS	Spacetaq	7	-4000 to +4000	0.12	190	18	215x215x73	<1000	-20 to +60	22 to 34	110
RW210	Hypoxon Technologies		10000	±0.5	49.1	±0.0015	25x25x15	21	-20 to +60	3.25 to 3.5	6.8
RW400			6000	±1	44	±0.015	50x50x27	135	-40 to +60	4.5 to 5.25	15
RWA-405		5	5000	±2	20	0.3	105x106.5x109.3	<1700	-20 to +40	15 to 25	
SXC-4PK-402	Sputnix LLC		±6000	±10	1.213.24.64	25224.8	87x97x68	530	-30 to +85	5 to 14	<8
VIRW-1	VECTRONIC Aerospace GmbH	>5	±5000	0.25	±25	1	115x115x77	1800	-20 to +70		<25
VIRW-02		>5	±5000	0.1	±20	0.2	76x76x48	1000	-20 to +70		<25

Magnetorquers

Model	By	Lifetime [years]	Magnetic moment [Am^2]	Residual moment [Am^2]	Resistance [Ω]	Linearity [%]	Dimensions [mm]	Mass [g]	Operating Temperature [°C]	Supply Voltage [V]	Power Consumption [W]
Magnetic Torquer	Chang Guang Satellite	15	15	≤0.1	24		280x56x53.5	670±30	-10 to +45	12	≤3
INTQ	ISIS		0.2				85x60 1x17	196	-40 to +70	5	<1.2
NCTR-30002	NewSpace Systems		0.2	±0.005		±4.5	70x10 diameter	<30	-20 to +60	5	<0.2
NCTR-30012			1.19	±0.005		±1.5	94x15x13	<50	-20 to +60	5	<0.8
MQ20			±10	≤0.1			31x36.56x44	450±50	-25 to +60	10	≤0.4
MQ200	OCE Technology	>1	≥200	≤1			840x100x61	4100±100	-25 to +60	40	≥4
TQ-15	Sachin Interspace		15		250		221 (length)	400	-35 to +72	28	
TQ-40			42		250		331 (length)	825	-35 to +75	28	
MT-1	Sputnix LLC		1				140x36x20	200	-30 to +60		0.4
MT-5			5				220x30x35	350	✓30 to +60		0.6
MT-10			10				220x36x35	550	✓30 to +60		0.7
MT-15			15				400x36x35	630	✓30 to +60		0.8
MT-20			20				470x36x35	800	✓30 to +60		1
MT-25			25				470x36x35	1000	✓30 to +60		1.2
MT-30			30				440x36x40	1200	✓30 to +60		1.8
Magnetic Torque Rods	STRAS SPACE		5-15				250x60x30	350 ± 600	-30 to +100	5/14/28	0.75 - 3.5
VMT-35	VECTRONIC Aerospace GmbH		25/35		100/200	1	354x25.5x50	600	-20 to +65	16/28	

Control Moment Gyroscopes

Model	By	Lifetime [years]	Momentum [Nm.s]	Torque [Nm]	Pointing performance [mrad]	Global Maneuver	Dimensions [mm]	Mass [g]	Operating Temperature [°C]	Power bus [V]	Power maneuver [W]
CMG 6-48	Airbus	8	4	6	<10	> 5 000 000	270x225x313	15000		22-37	64
CMG 15-48S		10	15	45	<10	> 2 400 000	270 diameter x330	19400	-20 to +55	22-37	25
CMG 40-40S		10	30 to 40	60	<10	> 7 300 000	370x465x509	38000	-30 to +35	22-37	115
CMG75-18S		10	70 to 75	75	<2	> 2 800 000	550x500x579	49000	-20 to +40		160
CMG-8	Blue Canyon Technologies	>10	8	8		> 2 000 000				22-36	30
CMG-12		>10	12	12		> 2 000 000	540x43x310	15000		22-36	35
M30 CMG	Boreswell		25 to 75	75				28000			
CMGM-6-4-1A	Mama Aerospace	5	5	28			301.4x184.6x181.6	8000±500	0 to +45		≤35

Star Tracker/Scanner

Model	By	Picture	TRACKER SCANNER	Lifetime LEO (years)	Lifetime GEO (years)	Star sensitivity [m _v]	Point accuracy (Pitch, Yaw)	Point accuracy (Roll)	Update rate [Hz]	Maximum slew rate (tilt) [°/s]	Maximum slew rate (roll) [°/s]	Dimensions [mm]	Mass [g]	Operating Temperature [°C]	Supply Voltage [V]	Max power consumption [mW]
ST200	Hyperion Technologies		TRACKER	5			30 arcsec	200 arcsec	5	>0,3	>0,8	29x29x30,1	42	-20 / +40	5	600
ST400	Berlin Space Technologies		TRACKER				10 arcsec	120 arcsec	5	>0,5	>1	53,8x53,8x90,5	280	-20 / +40	3,6-34	700
Cube Star	CubeSpace		TRACKER			<0,3	<0,0215 deg	<0,061 deg	1	0,3		50x35x55	55	-10 / +60	3.03	142
Star Mapper	NewSpace Systems		SCANNER			-8	16 arcsec		>1			135x135x280	<800	-20 / +55	28	<2000
STAR-T3	Space Inventor ApS		TRACKER	5			<2 arcsec	<12 arcsec	5	>0,3	>0,3	60x60x100	350		5-12	1000
AURIGA-SA	Sodem S.A.		TRACKER	7			2 arcsec	11 arcsec				120x90x22	770	-30 / +60	5	
Auriga-CP			TRACKER	7			2 arcsec	11 arcsec	5			55x66x94	210	-30 / +60	5	1000
HYDRA			TRACKER	10	18				30							
HYDRA-CP			TRACKER		18				16			146,5x263	1400	-30 / +60		<1000
HYDRA-M			TRACKER	10					18					-30 / +50		<1600
SED28	CubeSat Pointing		TRACKER	7	18		4 arcsec	20 arcsec	10			100x170x290	3470	-30 / +60		
KH Leuven			TRACKER				2 arcsec	10 arcsec	10			50x55x95			5	<1000
KH Leuven Mini Star Tracker			TRACKER				20 arcsec	120 arcsec	2			20x20x10			3,3	<1000
NanoSat Sensor (NST-1)	Chung Gwang Satellite		SCANNER	5			7 arcsec	70 arcsec	10	4	4	55x50x132	0.27			1200
Mark-e Star Sensor	Antron Corporation Limited		SCANNER	15	15		<0,2 deg		8				1000	-10 / +50		
VST-66M	Vectronic Aerospace GmbH		TRACKER				5 arcsec	30 arcsec	5	<3	<3	60x30x138	470	-20 / +65	9-40	3000
VST-41M			TRACKER						4			80x100x180	900	-20 / +65	9-40	2500
The Standard NST	Blue Canyon Technologies Inc.		TRACKER	>5			6 arcsec	40 arcsec				100x55x50	350			<1500
The Extended NST			TRACKER	>5			6 arcsec	40 arcsec				250x100x100	900			<1500
ST-16RT2	Sinclair Interplanetary		TRACKER	13	0		5 arcsec		2	<2	<3	62x56x38	158	-40 / +50	9-34	500
Laser Downlink plus Star Tracker			TRACKER	13	9		5 arcsec	55 arcsec				78x68x98	335	-20 / +40	28	1000
ASTRO 10	Jena-Optronik GmbH		SCANNER	12						1	1			-40 / +50	28	
ASTRO 15			SCANNER	>15	25									-30 / +55		
ASTRO APS			SCANNER	>18	25					0,3	0,3	154x154x237	2000	-30 / +60	28	
A-STR	Leonardo Finmeccanica S.p.A.		SCANNER	18			8,25 arcsec	11,1 arcsec	10	2	2	105x175x290,5	3550	-30 / +60	20-50	8900-13500
AA-STR			SCANNER	18			8,25 arcsec	11,1 arcsec	10	2	2	104x150x346	2660	-30 / +60	20-52	5600-12000
SPACESTAR			TRACKER	18			7,7 arcsec	10,6 arcsec	4-10			164x164x284	1660	-30 / +60	5	<1500
HE-GAS	Terma		TRACKER				<1 arcsec	<5 arcsec	4	0,5	0,5	120x120x33	1000	-40 / +70	22-34	1500-5500
MAI-S8 Space Sextant	Adcole Space		TRACKER				5,7 arcsec	27 arcsec	4	>2	>2	55x65x70	135	-40 / +80	5	2000
NTS-1	O.O.E. Technology		TRACKER	>5		5,8	<3	<25	10	>3	>3	60x50x85	150	-30 / +60	5	1000
NTS-2			TRACKER	>5		5,8	<3	<25	10	>2,5	>2,5	50x50x32	130	-30 / +60	5	1000
PTS-1			TRACKER	>5		5,8	<5	<40	5	>1,5	>1,5	32x32x45	80	-30 / +60	5	500
PST8 pico star tracker			TRACKER	0-5		5,8	<5 arcsec	<70 arcsec	10	>1,5	>1,5	33x33x9,4	85	-40 / +40	5	500

Sun Sensor

Model	Model	Model	Model	Model	Model	Model	Model	Model	Model	Model	Model	Supply Voltage [V]	Max power consumption [mW]
CubeSense	CubeSpace		DIGITAL	>10		180	<0,2	2		80	-20 / +70	3	150
SS200	Hyperion Technologies		ANALOG			110	<1	100	24,66x15x3,5	3	-45 / +85	5	2.5-40
Fine Sun Sensor	Space Inventor		ANALOG	5		120	1		42x16x6,2	8	-40 / +70	3,3	90
NCSS-SA05	NewSpace Systems		ANALOG			114	<0,5	10	33x11x6	5	-25 / +70	5	10
NFSS-411			DIGITAL			140	<0,1	5	34x32x20	35	-25 / +70	5	37,5
ISS-A5 analogue	Solar MEMS Technologies		ANALOG	>10		10	<10		80x27	100	-40 / +85	5	
ISS-A15 analogue			ANALOG	>10		30	<10		80x27	100	-40 / +85	5	
ISS-A25 analogue			ANALOG	>10		50	<10		80x27	100	-40 / +85	5	
ISS-A60 analogue			ANALOG	>10		120	<10		80x27	100	-40 / +85	5	
ISS-D5 digital			DIGITAL	>10		10	<0,1		80x27	100	-40 / +85	5	
ISS-D15 digital			DIGITAL	>10		30	<0,2		80x27	100	-40 / +85	5	
ISS-D25 digital			DIGITAL	>10		50	<0,3		80x27	100	-40 / +85	5	
ISS-D60 digital			DIGITAL	>10		120	<0,4		80x27	100	-40 / +85	5	
MASS-5			ANALOG	>10		10	<10		80x27	100	-40 / +85	5	
MASS-15			ANALOG	>10		30	<10		80x27	100	-40 / +85	5	
MASS-25			ANALOG	>10		50	<10		80x27	100	-40 / +85	5	
MASS-60			ANALOG	>10		120	<10		80x27	100	-40 / +85	5	
ISS-T5			DIGITAL	>10		10	<10		80x27	100	-40 / +85	5	
ISS-T15			DIGITAL	>10		30	<10		80x27	100	-40 / +85	5	
ISS-T25			DIGITAL	>10		50	<10		80x27	100	-40 / +85	5	
ISS-T60			DIGITAL	>10		120	<10		80x27	100	-40 / +85	5	
SSOC-A60			ANALOG			120	<0,3		40x30x12	25	-40 / +85	5	36
nanoSSOC-A60			ANALOG			120	<0,5		27,4x14x5,9	4	-30 / +85	5	
SSOC-D60			ANALOG			120	<0,3		70x30x12	35	-45 / +85	5	
nanoSSOC-D60			ANALOG			120	<0,5		43x14x5,9	6,5	-30 / +85	5	
Coarse Sun Sensor (CSS)	Bradford Space		ANALOG			180	>1,5		110x110x30	215	-80 / +120	5	
Fine Sun Sensor (FSS)			ANALOG			128	0,3		108x108x52,5	375	-50 / +85	5	0,25
Mini Fine Sun Sensor (Mini-FSS)			ANALOG			128	1,5		50x46x17	50	-50 / +80		
4 Pi Sun Sensor	Antrix Corporation		ANALOG						55x40x30	50	-20 / +60		
SS Series Sun Sensor	O.C.E. Technology		ANALOG	>5		60	1		60x60x26	77	-80 / +80		-
Digital Sun Sensor	Chang Guang Satellite		DIGITAL	5		120	<0,5	10	48x36x21	40	-30 / +60	5	300
S3 (Smart Sun Sensor)	Leonardo Finmeccanica		ANALOG		>15	128	0,02		112x12x43	330	-25 / +60		700
SXC-SD-01	Sputnix		DIGITAL			120	<0,5	10	28x23x11	10	-40 / +85	4-15	
Coarse Sun Sensor	Adcole Space		ANALOG			360	1		89x89x43	130			
Miniature Spinning Sun Sensor			ANALOG			175	0,25		57x51,5x51	250			500
Fine Sun Sensor			ANALOG			100	0,01						
Fine Spinning Sun Sensor			ANALOG			128	0,1		33x25x66	109			400
Digital Sun Sensor			DIGITAL			64	0,1		96x94x53	300			
SmallSat Sun Sensor			ANALOG						50,8x19,05x2,03	5,5		5	5

