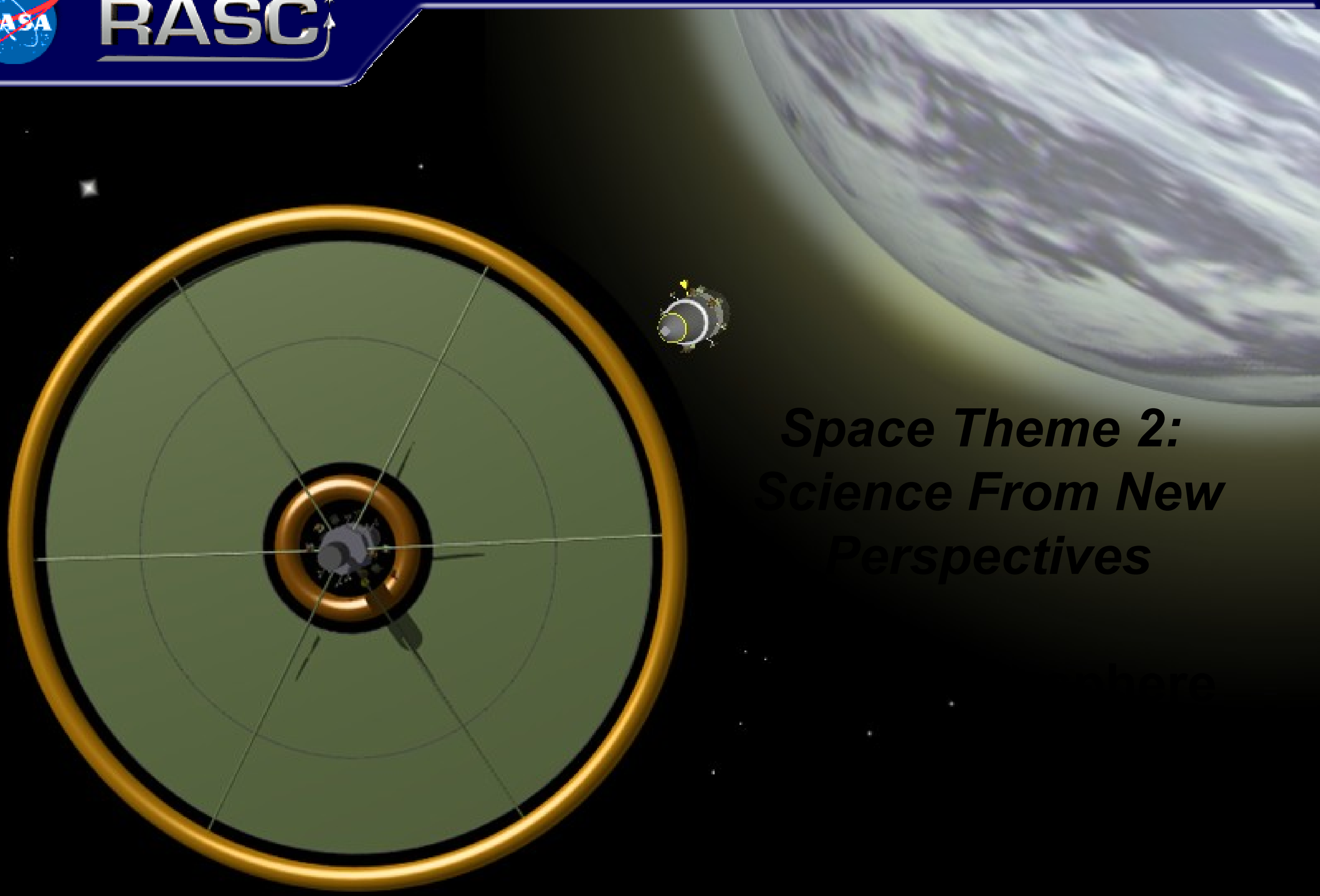


Space Theme 2: Science From New Perspectives

Space Theme 2: Science From New Perspectives

Earth Atmosphere Observatory at L2:

Introduction and Background

Space Theme 2: Science From New Perspectives

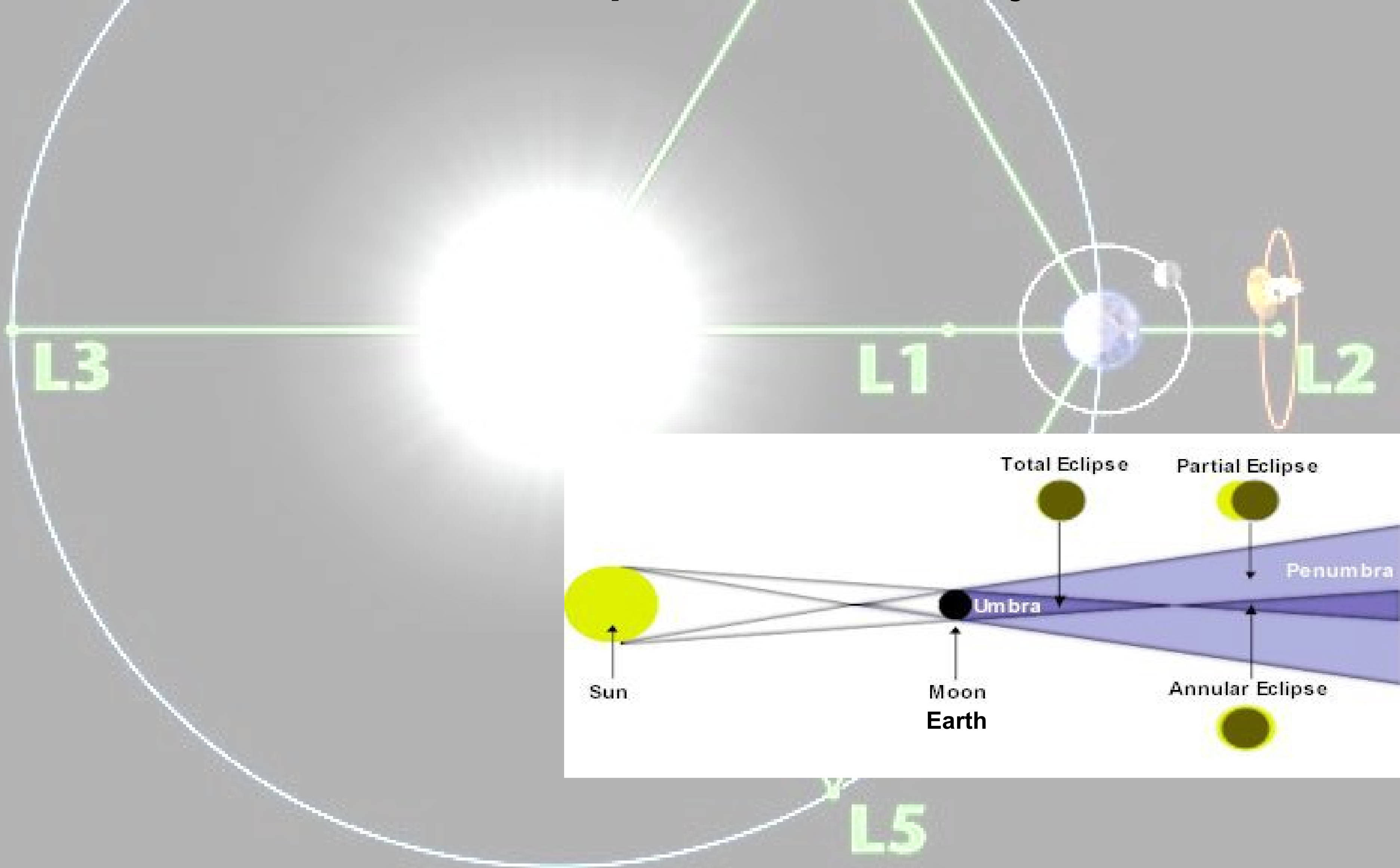
Earth Atmosphere Observatory at L2

- **Virtual Structure Gossamer Space Telescope**
  - **Edward Mettler - JPL**

- **Atmospheric Remote-Sensing Observatory at the L2 Lagrange point**
  - **Joseph Zawodny - LaRC**

- **Study Approach:**
  - **The "Virtual Structure Gossamer Space Telescope" selection serves as the focal point of the Earth Observation Telescope at L2 study**
  - **The "Atmospheric Remote-Sensing Observatory at the L2 Lagrange Point" selection provides the science requirement drivers**

Earth Atmosphere Observatory at L2



Study Objectives

- Develop a concept for an Earth observing capability at L2
- Define the science to be conducted and the associated instrumentation
- Develop conceptual designs for the instrument, telescope, and supporting spacecraft
- Define an end-to-end mission architecture
- Assess the technical challenges and identify enabling technologies

Groundrules and Assumptions

- **Timeframe is 2025 – 2030**
- **A crewed expedition to Mars has already succeeded and commercial development of low Earth orbit (LEO) is beyond its infancy**
- **International Space Station has been enhanced to provide a LEO satellite assembly and maintenance capability**
- **A crew tended Earth-Moon L1 gateway station has been established which also provides a satellite assembly and maintenance capability**
- **Infrastructure is in place to support routine operations between Earth and the L1 gateway. This infrastructure will be expanded during this timeframe to support deployment and servicing of observatories at Earth-Sun L2.**
- **Application of nuclear systems is no longer feared and the utilization of materials from the moon and Mars in support of permanent human presence beyond LEO has begun.**
- **A high bandwidth communication infrastructure and a Solar System "GPS" are being planned for support of deep space missions (including missions to Earth-Sun L2) Develop a concept for an Earth observing capability at L2**

Accomplishments

- Finalized the Observatory Science and Engineering System Requirements
- Developed the overall architecture of the Earth observing Telescope and its operation, and method of delivery from Earth to Deployment at L2
- Developed a unique Optical System Design to meet the specific Science requirements using a 25m aperture Primary Mirror combined with a secondary telescope at the Primary focus 125 m away in formation flying precision alignment
- Completed assessments of GNC, Power, Thermal Control, Communication, Propulsion, Structure and CD&H Subsystems, and Navigation to L2
- Developed the Initialization and Calibration strategy and scenario for the Observatory Operation
- Addressed the engineering infrastructure issues related to delivering the facility to L2 site, and its maintenance.

Study Team

- JPL
  - Edward Mettler – Principal Eng. Invest.
  - Ram Manvi – Technical Integration
  - Ahmet Acikmese – Controls/Dynamics
  - William Breckenridge – System Reqs
  - Serge Dubovitsky – Optical Metrology
  - Steve Macenka – Optical Design
  - Eldred Tubbs – Form. Flying/Metrology
- GSFC
  - Steve Cooley/Dave Folta/Greg Marr – Orbital Mechanics
  - Jay Herman – Science
  - Jesse Leitner – Distributed Spacecraft
  - Quang Nguyen/Ken McCaughey – C&DH
  - David Steinfeld – Thermal
- LaRC
  - Joe Zawodny – Science/Instrumentation
  - Jeff Antol – Study Integration
  - John Flick – Technology Assessments
  - Shawn Krizan – Configuration/CAD
  - Rosalind Echols – Mission Infrastructure
  - Jeff Murch – Graphics/Animations
  - Carlos Roithmayer/Linda Kay-Bunnell/ Renji Kumar/Haijun Shen – Orbital Mechanics
  - Fred Stillwagen/Jennifer Parker – Comm
  - Chris Strickland – Structures
- GRC
  - Stan Borowski – GRC Lead
  - Bob Cataldo – Power
  - Melissa McGuire/Thomas Packard – Launch Vehicle, Transfer Vehicle
  - Tim Sarver-Verhey – Propulsion

EARTH OBSERVATORY at L2

The mission has two major phases:

1. **Delivery and Deployment of Payload at L2:**

   Baseline is direct impulsive Earth escape, then low thrust to L2, and final impulsive zero velocity deployment using hybrid Chemical / NEP or SEP Carrier Vehicle

2. **Earth Atmospheric Science at L2:**

   Baseline is Two spacecraft Observatory Formation of a 25-m Primary Membrane Mirror and a Secondary Science Telescope in a precision orbit that provides continuous alignment of Earth - Sun occultation

In-Space Infrastructure Potential Options

Phase 1 Optional Elements

- Launch & Transfer to L2 point
- On-orbit Assembly vs. Self-Deployment
- Astronaut EVA vs. Robotic Operations
- Servicing & Maintenance
- Enabling Technologies

Potential Deployment Scenario

- **Launch/Deployment/Servicing:**
  - Launch and transfer to Earth-Moon L1
  - On-orbit deployment/assembly and checkout at E-M L1 using an OASIS-based assembly/servicing capability
  - Low-thrust transfer of deployed spacecraft to Earth-Sun L2
  - Potential for a refueling/maintenance mission at 5 years by autonomous servicing vehicle
  - Potential contingency repair mission to deal with unexpected situations
- **Infrastructure Technology Requirements:**
  - Assembly/servicing capability: increases reliability of mirror deployment
  - Reusable transfer vehicles
  - Autonomous servicing vehicles (primarily for refueling)
  - On-orbit assembly tools (robotic and EVA)

NASA Earth Science Objectives

- Monitor changes in the Forcing and Response of the Earth's Atmosphere
- Understand the mechanisms of change and quantify the attribution of change be it of chemical or dynamical origin
- Improve the short and long term predictive capability of weather and climate models through the use of near real time measurements and an improved understanding of the dynamical, chemical, radiative Feedbacks and Responses to the observed change in Climate Forcing

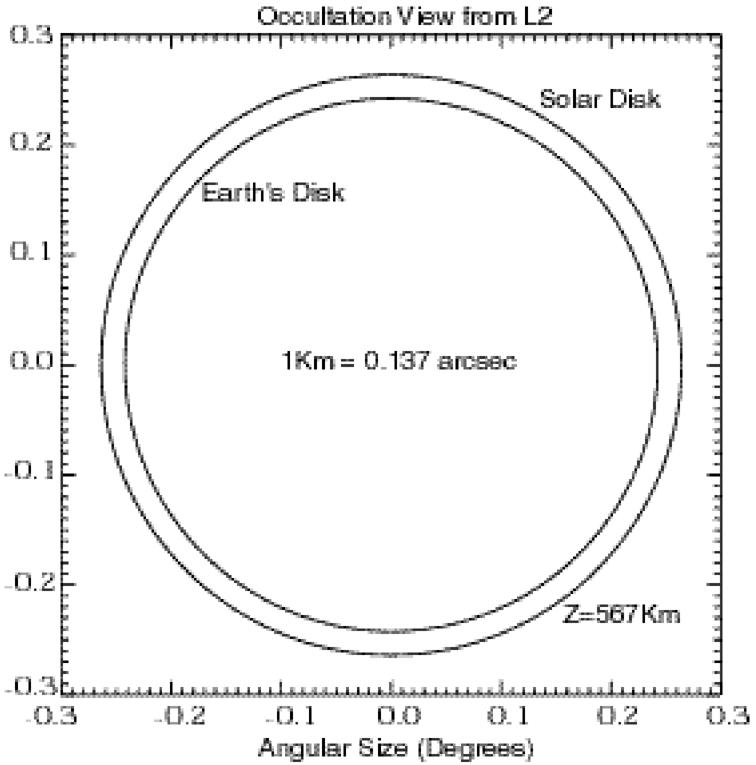
Mission Science Strategy

- **Solar Occultation Best Suited for Long-Term Climate Change Studies**
  - **L2 is the optimal place to deploy solar occultation instruments.**

- **Can obtain high vertical and spatial resolution maps of many species twice per day for use in near-real time predictive assimilation models. (forecast models)**

- **Similar capability would require multiple spacecraft in low Earth orbit.**

- **Observation Strategy:**
  - **Scan around the annular ring of the Earth's atmosphere at least 360 times per day for ~1° "longitudinal" sampling**
  - **Sample each rotation at least 360 times to provide ~1° "latitudinal" sampling**



From L2 the Sun is slightly larger than the Earth and places the annular ring of the Earth's atmosphere into permanent occultation. Spatial sampling is limited only by the speed of the instruments and the down-link bandwidth.

Improvements over Current Practice

- Spatial resolution can approach 0.1 degrees (10x improvement over Aura) through a combination of increased instrument sampling and algorithmic techniques (tomography or data assimilation)
- Trend-Quality observations of the dynamical response of the middle atmosphere (10-70km) to climate change.
- Similar capability would require multiple spacecraft in low Earth orbit.
- Near Real-Time production of final products for time-critical consumption (forecast models)

Science Observation Strategy/Orbit

- Remain close (within 200km) to the Earth-Sun axis for 24/7 100% duty cycle.
- Scan around the annular ring of the Earth's atmosphere at least 360 times per day for ~1° "longitudinal" sampling
- Sample each rotation at least 360 times to provide ~1° "latitudinal" sampling
- Refraction will limit the lowest altitude to ~8km
- Co-align all instruments and synchronize operation to provide sampling of the same air mass over all wavelengths (0.3 to ~10 microns)

Earth Atmosphere Observatory at L2

System Architecture, Optics, Design Requirements, L2 Deployment, and GNC

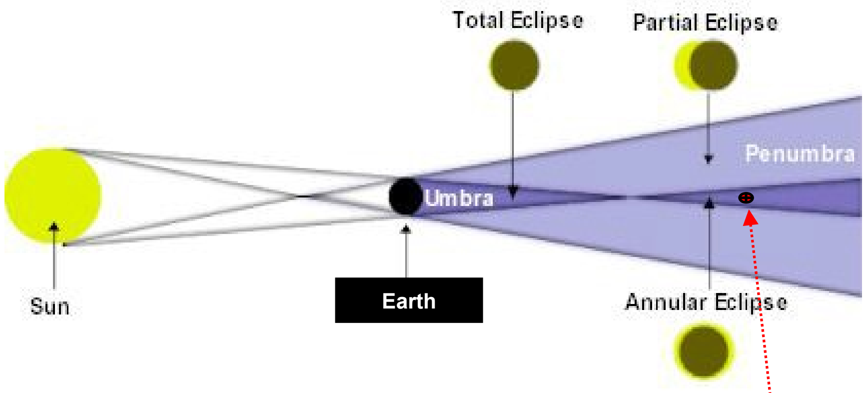
Edward Mettler

JPL

Principal Engineering Investigator

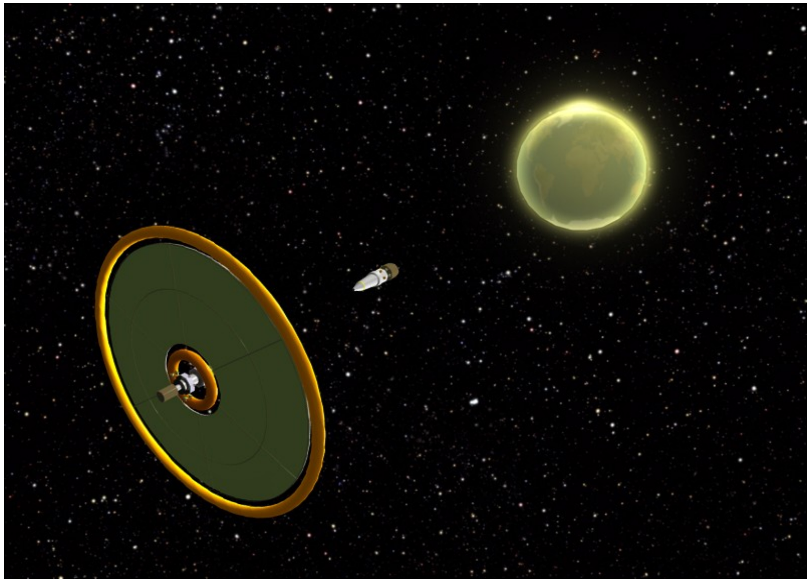
Approaching Occultation Alignment

Observatory Architecture



Credit: http://sci.esa.int/content/doc/1f/13855_.htm

Earth Observatory

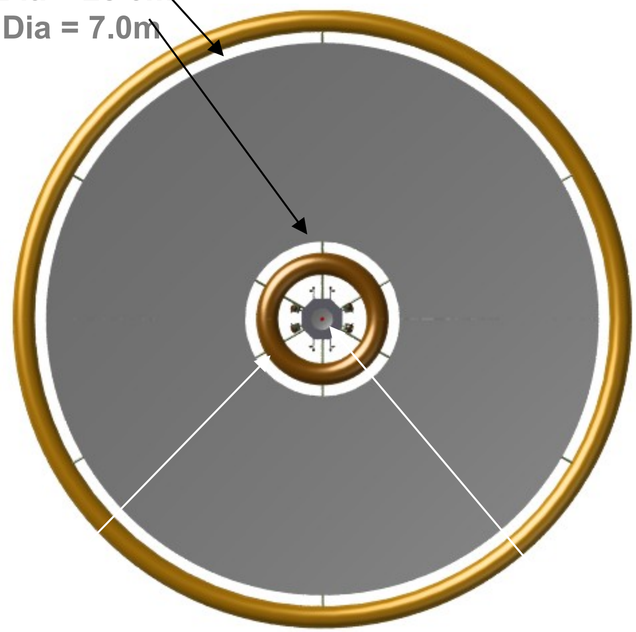


L2 Earth Observatory Two Spacecraft Formation

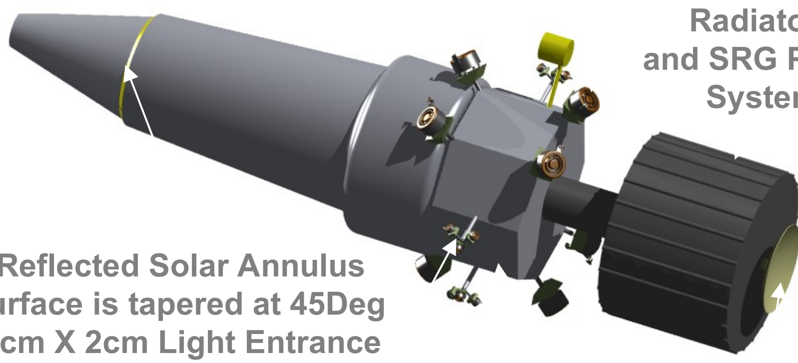
- Science S/C: This spacecraft contains the science telescope directed at the mirror in the Aperture S/C. It has sensors for orbit following, as well as metrology for formation flying and Primary Aperture Figure (spherical shape) determination.
- Primary Aperture S/C: This spacecraft contains a 25 m diameter membrane mirror inside an inflatable torus deployed from a bus. It has sensors for formation flying and orbit following.
- Each spacecraft is baselined at 1200 kg wet mass. Orbit analyses and simulation results are scalable to larger or smaller spacecraft mass.

Observatory Spacecraft Concepts

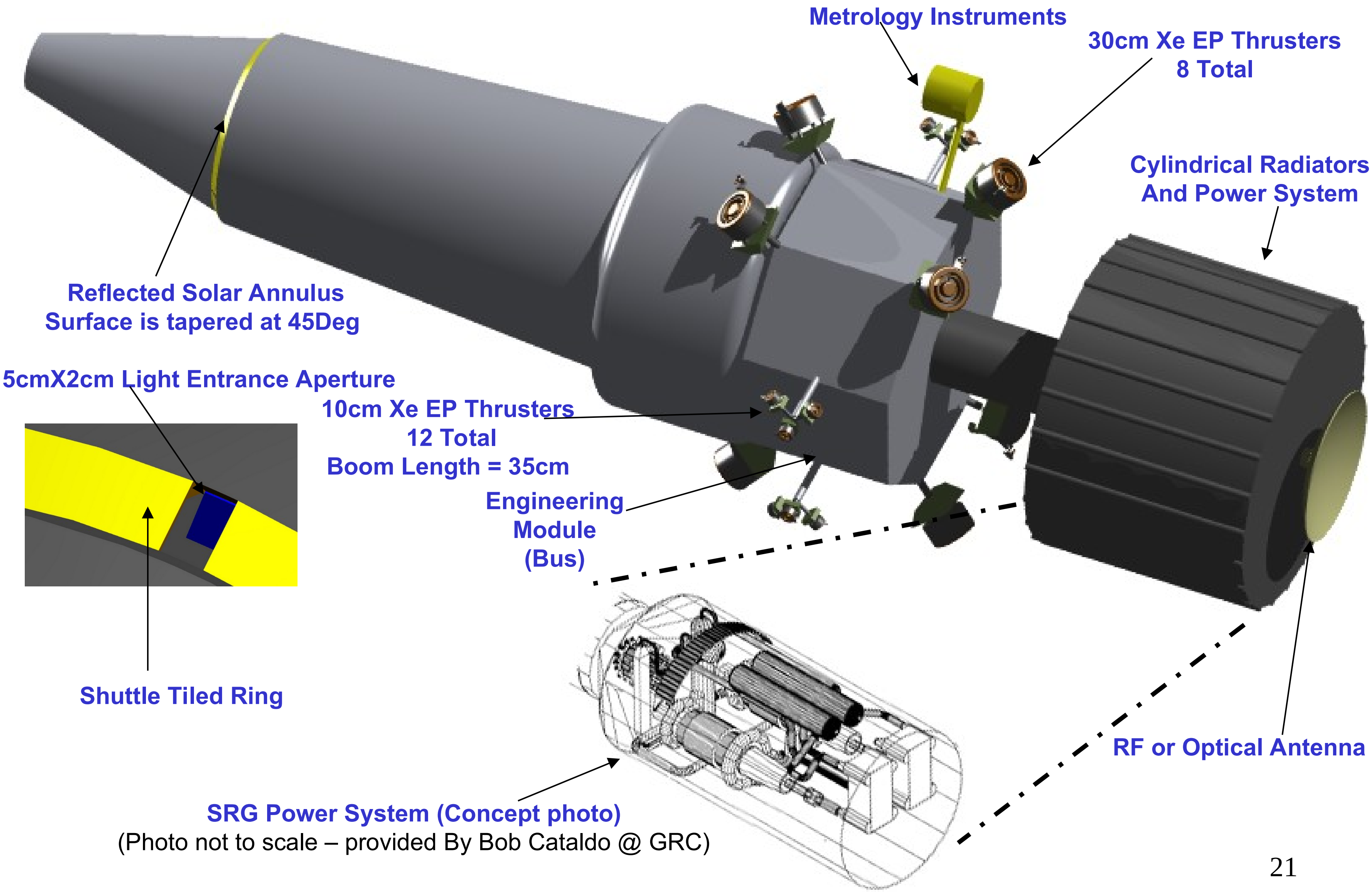
L2 Observatory Aperture Spacecraft



L2 Science Telescope Spacecraft

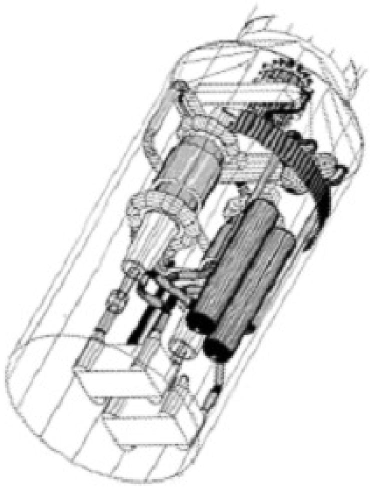


L2 Science Telescope

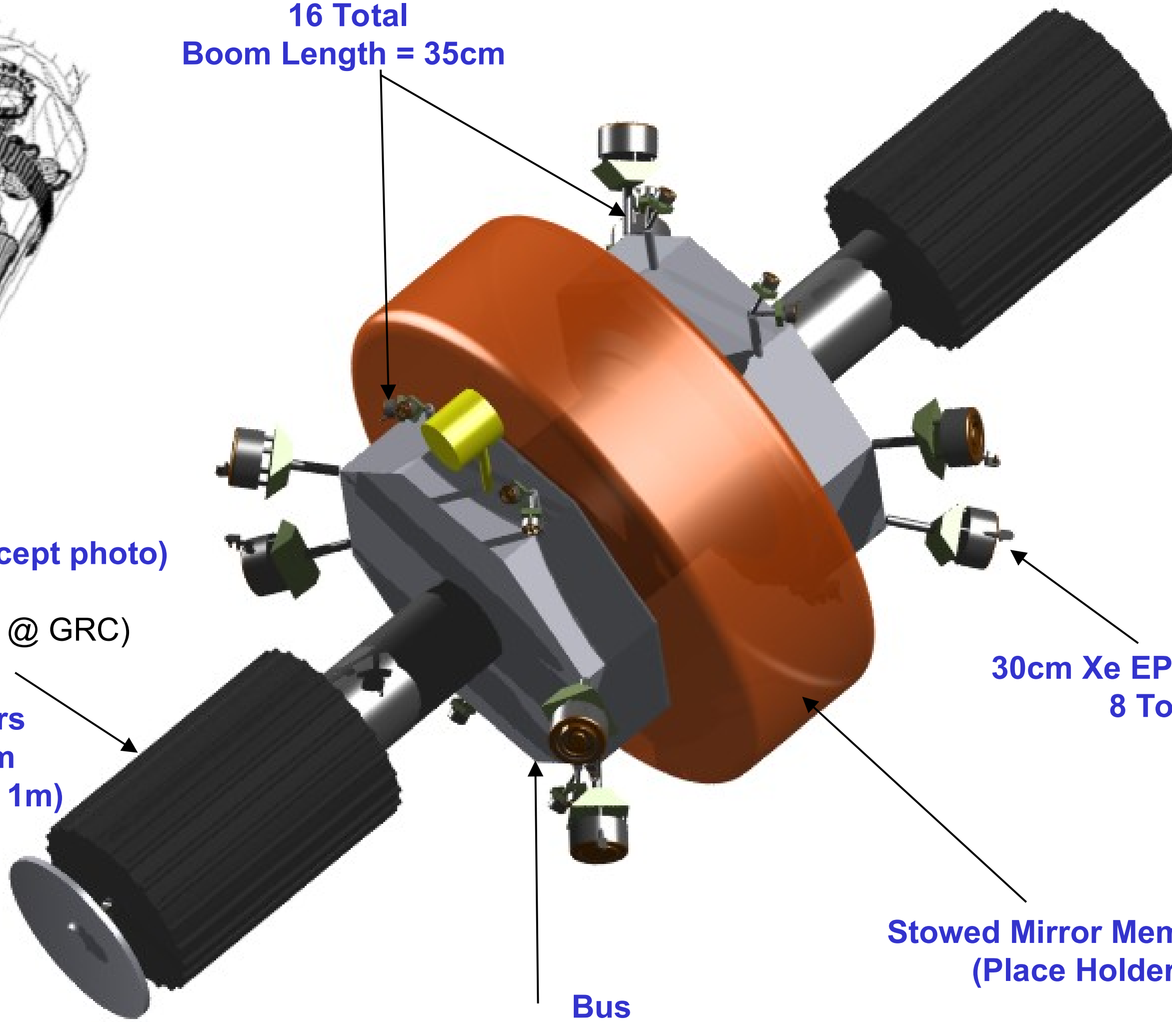


(Photo not to scale – provided By Bob Cataldo @ GRC)

L2 Observatory Aperture S/C
(Pre-deployment)

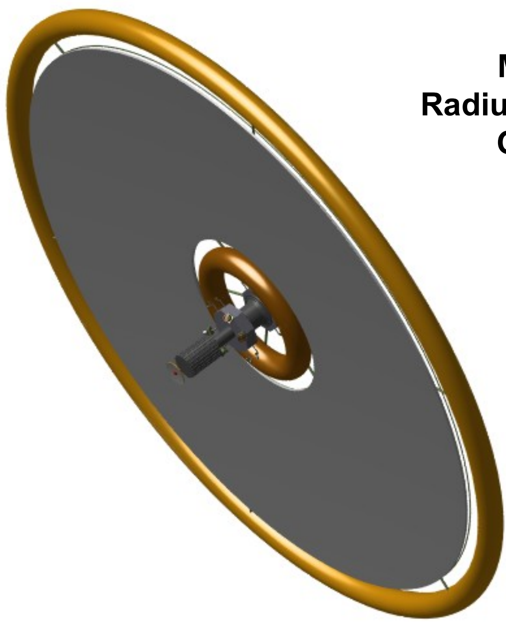


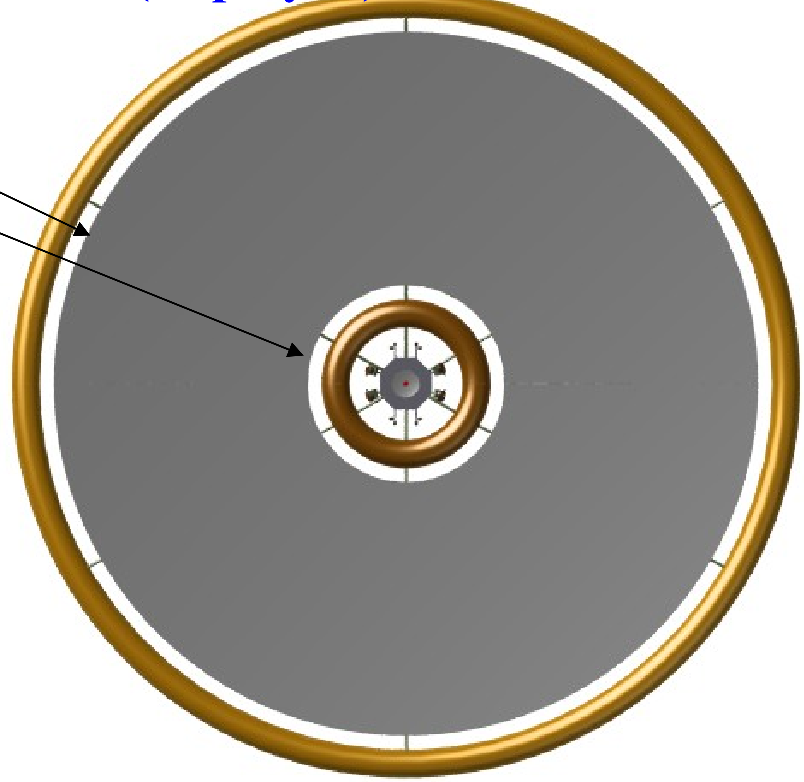
SRG - Power System (Concept photo)
not to scale
provided By Bob Cataldo @ GRC)

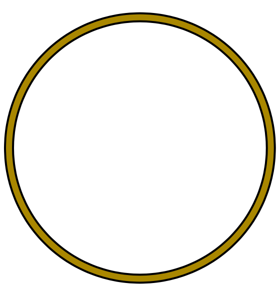


L2 Observatory Aperture S/C

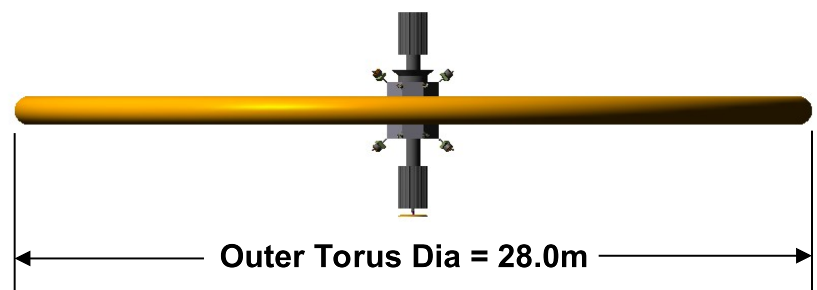
(Deployed)





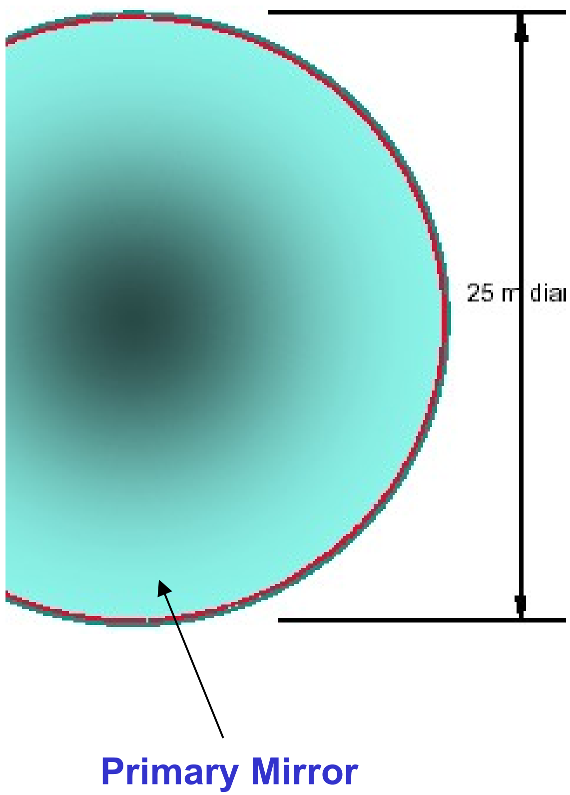


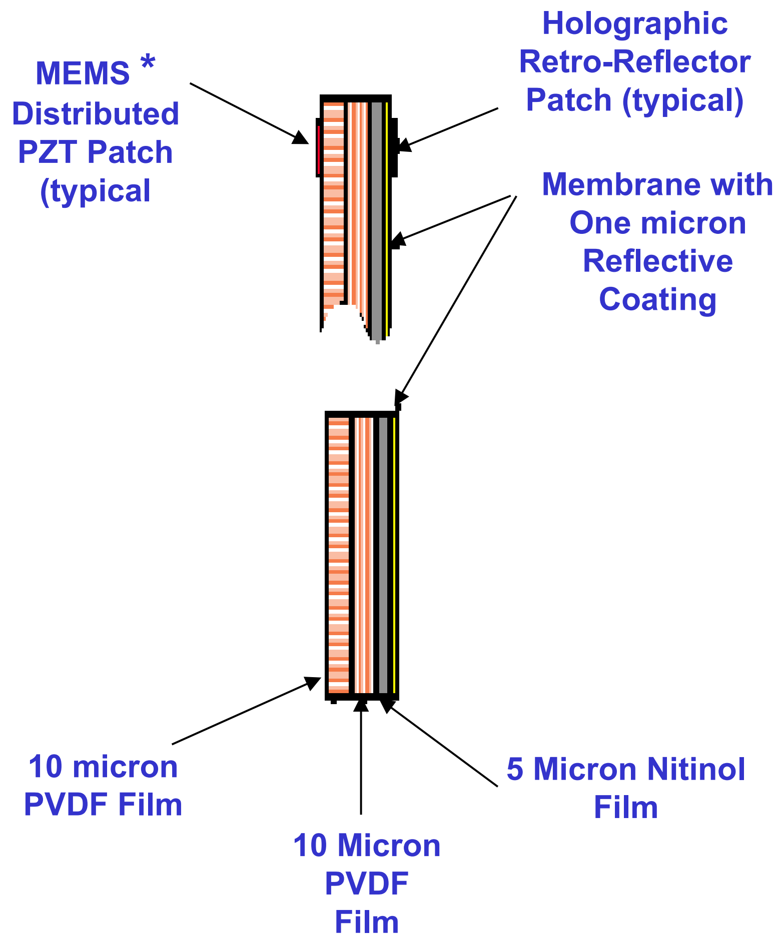
Cross Section of Torus – Dia = 1.0m



Membrane Primary Mirror

Active Figure Control Concept for Membrane Primary Mirror





* E-H Yang and S-S Lih, JPL , 2003 IEEE, 0-7803-7651-X/03

Observatory Telescope Optics

Design Drivers:

- Primary Aperture is 25 m diameter to satisfy science 1 km resolution at Earth over broadband spectrum, i.e., Diffraction limit of 67 micro-radian at 10.5 microns. Theoretical size is 19 m with added margin for membrane boundary conditions.
- Earth-Sun are extended objects viewed from L2 and require a **Spherical Aperture** system or ***Schmidt Telescope*** concept to handle wide angle and high resolution.
- Separated S/C optics are required by the desired Primary f/10 focal ratio (to minimize aberrations) with 250 m focal length and 500 m center of curvature (c-c).
- The ***Schmidt*** spherical aberration Corrector Mirror, normally located at the (c-c) in a monolithic system, must be re-imaged to locate inside the secondary (Science) S/C Telescope. This is called a **"Reduced Schmidt"** design and adds complexity.

Observatory Telescope Optics

Design Drivers (con't):

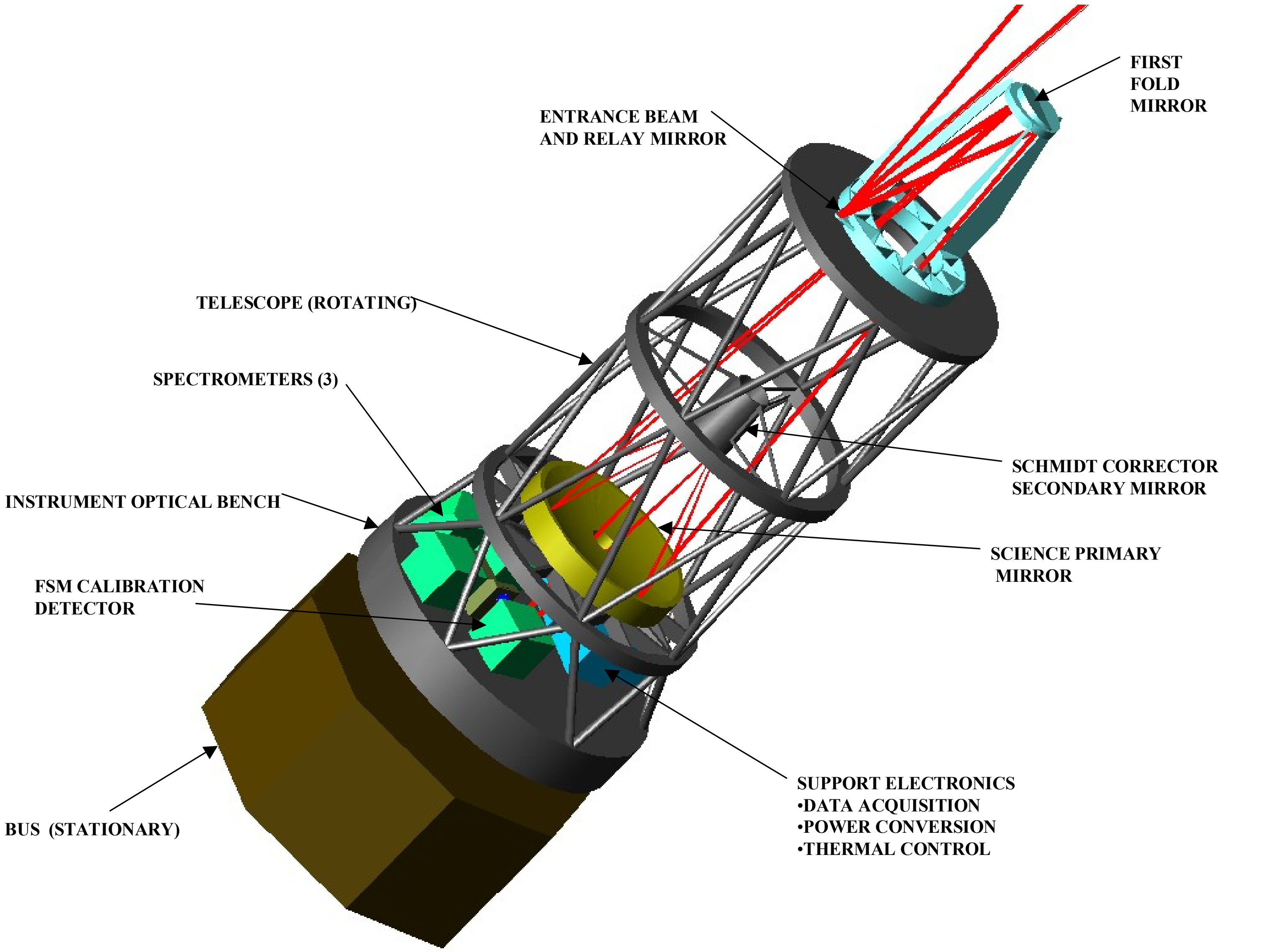
- **However, the desired f/10 optics creates an excessively large (2.5 m) Solar Light Annulus image at the Primary focal plane.**
- **Constrained to a practical f/5 Annulus size of 1.25 m in order to keep the Science Telescope optics and S/C size/mass within realistic limits. The f/5 mass is < 1/10 th the f/10 Telescope.**
- **Faster f/5 Primary is more prone to spherical aberration and less depth of field tolerance.**
- **Smaller size relay and corrector elements are more difficult to design (Lagrange invariant) for required performance.**
- **Ratio of 25 m to ~ 1 m pupil magnification increases distortion.**
- **Greater number of optical elements required reduces broadband energy throughput of optical train.**
- **Required Corrector Aspheric is more complex to design/fabricate.**
- **FSM (fast steering mirror) required for beam stabilization.**
- **Science Instruments mass, volume, interfacing optics space constraints.**

Observatory Telescope Optics

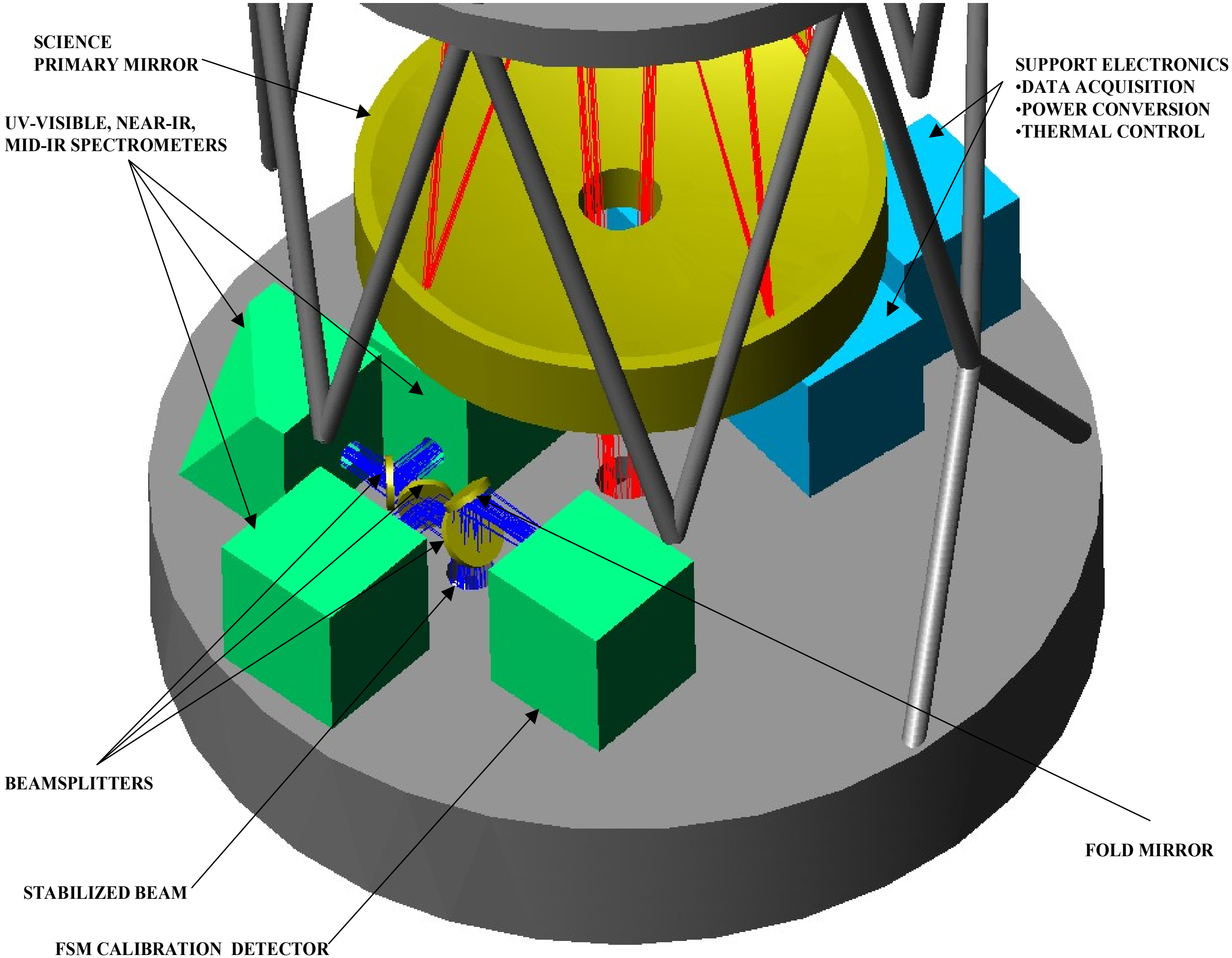
Implementation Trades:

- **All f/10 optical system was designed with excellent performance, however it was found to be extremely large and massive, i.e., not feasible for the L2 Observatory concept which must fly a powered orbit near L2.**
- **A much smaller and lighter Reduced Schmidt design was developed to work with an f/5 Primary Aperture. This allowed the Science Telescope to have an f/5 Front End with an f/10 Back End which recovers most of the original performance, albeit with a more complex optical train, transfer optics, and aberration corrector.**
- **The following diagrams depict the true to scale unique optical design of the receiving Secondary Telescope in the Science Spacecraft.**

Science Telescope Optics Design

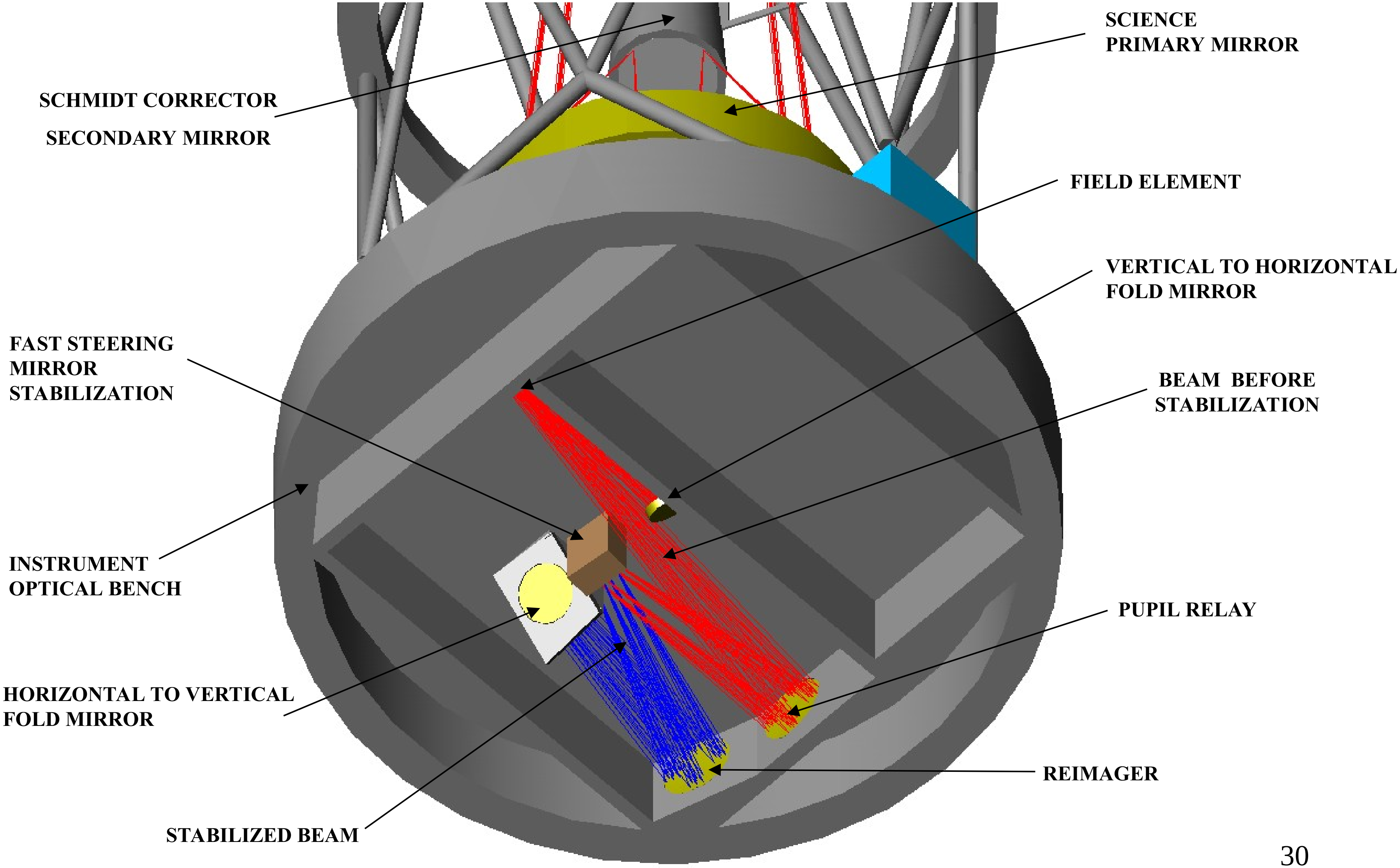


Science Telescope Optics Detail



SCHMIDT CORRECTOR
SECONDARY MIRROR

Science Telescope Optics Detail



Earth Atmosphere Observatory at L2
System Requirements

This section describes the overall Observatory quantitative system engineering requirements for accomplishment of the scientific mission at L2.

Earth Atmosphere Observatory at L2 System Requirements

Science Drivers of Observatory Design

- Earth Atmospheric Observation from the Earth-Sun L2 Point
- Scan the Earth Atmosphere with 100km surface (lat, long ≈ 0.9°) resolution, once per day
- Sample the atmosphere with 1km vertical resolution
- Deviate from the Earth-Sun line by ≤ 200km to maintain the Sun annulus around Earth
- Large Aperture Telescope for a resolution of < 0.67μr for observations at wavelengths 0.28μm to 10.5μm
- Angular jitter of ≤ 0.13μr(0.2km alt.), peak-peak, for frequencies >100Hz
- Know/control jitter to ≤ 0.2km for lower frequencies

Earth Atmosphere Observatory at L2
System Requirements

Instrument Sampling of Atmosphere

Sampling Resolution and Timing

Latitude: 400 samples per limb-scan
100km or 0.9° latitude resolution
0.54 sec per sample

Longitude: 400 limb-scans per day
100km or 0.9° longitude resolution
216 sec per limb-scan
(0.0291 r/s about Earth Line)

Earth Atmosphere Observatory at L2
System Requirements

Instrument Sampling of Atmosphere

Field of Regard

Altitude: 0 to 100km above Earth Limb

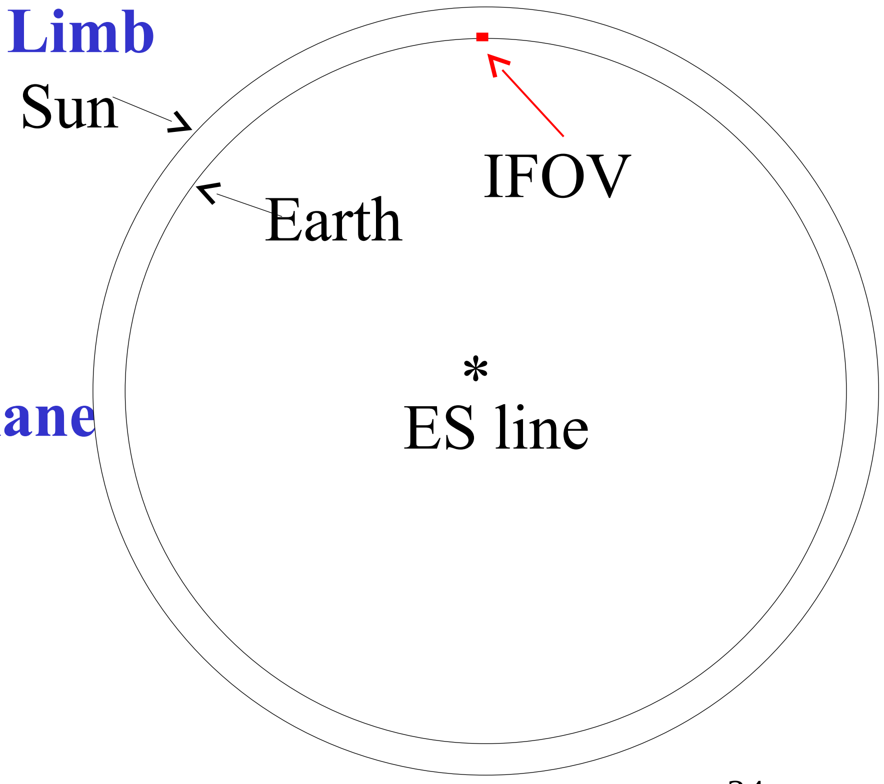
Instrument Field of View

(aligned with limb normal):

1km x 100km at Earth

0.66 x 66.2μr at L2

0.08 x 8.3mm at Focal Plane



Earth Atmosphere Observatory at L2 System Requirements

Observation Geometry

Ranges

L2 to Earth	1.51e6 km
L2 to Sun	151.1e6 km

Angular radius

Earth	4.2233 mrad
Sun	4.6192 mrad
Difference	0.3954 mrad

Mean Angular rate, Earth-Sun line

mrad/day	17.202
mrad/scan	0.043

Earth Atmosphere Observatory at L2
System Requirements

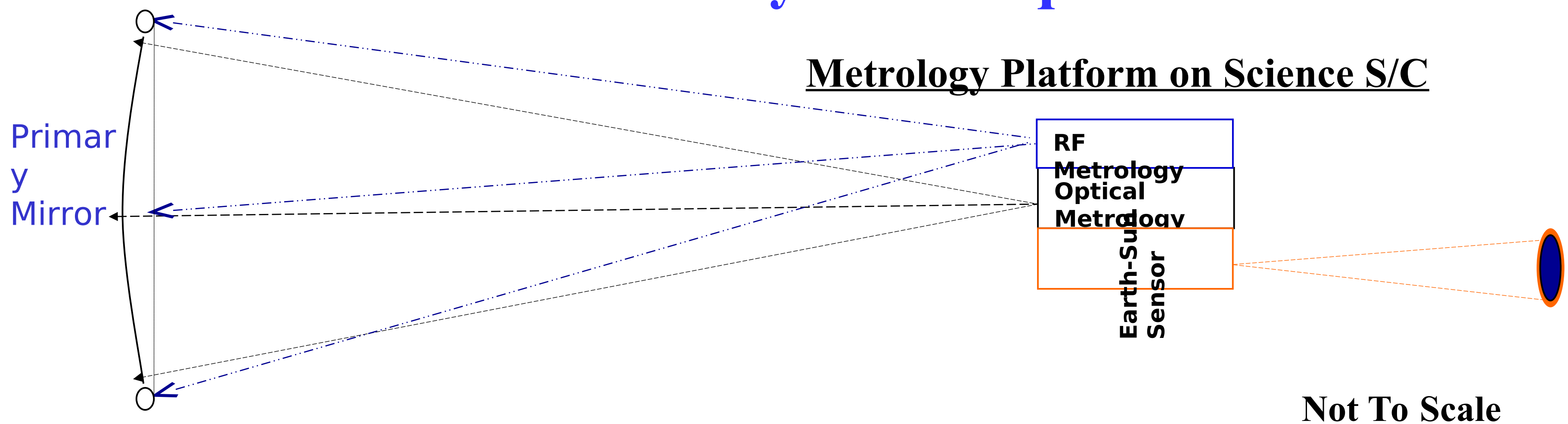
Primary Aperture Imaging Geometry

f #	5
Primary Mirror diameter	25m
Focal length	125m
Mirror radius of curvature	250m

Images in Primary Image Plane:

Earth, diameter	1.056m
Sun,diameter	1.155m
Difference	0.099m

Earth Atmosphere Observatory at L2
System Requirements



RF Metrology - **GPS-like range and phase measurement between transmitter and receiver, triangulated to get relative position and attitude of the Aperture S/C for acquisition and coarse formation control.**

Measurement (1σ): Range - 1cm Δphase - 10µm

Optical Metrology - **Laser range & bearing to retro-reflectors on the mirror to get precision relative location and attitude and mirror shape for fine formation control and Earth image location prediction.**

Measurement (1σ): Range - 1µm Bearing - 10µrad

Earth-Sun Sensor - **Image Earth & Sun to find points on the limbs and determine relative Earth direction, position offset from the Earth-Sun line and coarse Earth range.**

Measurement (1σ): Limb point - 20µrad

Earth Atmosphere Observatory at L2 System Requirements

Metrology Performance Estimates, 1-sigma

- **Image Location Error**

  -Based on estimate using one minimum data set (time point)
  -No predicted reduction from larger data set or filtering/smoothing*
  -Mirror shape sensing not included

RF Metrology	5.8	mm	Along LOS
	72.2	mm	Normal to LOS, per axis, range only
	12.5	mm	Normal to LOS, per axis, range&Δphase

*reduced to 25% with filtering and accel. noise of 0.1μg, 1000s correlation time

Optical Metrology	0.5	mm	Along LOS
	12.5	mm	Normal to LOS, per axis, range only
	0.7	mm	Normal to LOS, per axis, range&bearing

*reduced to 40% with filtering and accel. noise of 0.1μg, 1000s correlation time

- **Earth Relative Navigation**

  -Based on estimate using one minimum data set (time point)
  -No predicted reduction from larger data set or filtering/smoothing*

Earth-Sun Sensor	3552	km	Earth Distance (20μr, angular diameter)
	21	km	Earth or Sun Centroid accuracy, per axis (≈14μr)
	30	km	Offset from E-S Line, per axis (difference of E & S centroids)

*Readily reduced by a factor of 10 with 100 measurements.filtering

Earth Atmosphere Observatory at L2
System Requirements

Navigation Requirements and Error Allocations (3σ)

- **EARTH RELATIVE POSITION**

±200km		Lateral Position from the Sun-Earth Line	
	100	Sun-Earth Line Offset Knowledge, 3-sigma	
		63	Centroid of Earth, unfiltered
		63	Centroid of Sun, unfiltered
	100	Formation Position Control	
±5000km		Allowable Earth range variation	

- **EARTH RELATIVE POINTING**

±436μrad		Communication using HGA
	42	Earth direction, E-S Sensor centroid, 3σ unfiltered
	±175	Pointing Control
	80	HGA misalignment wrt E-S Sensor, 3σ .

Earth Atmosphere Observatory at L2
System Requirements

Formation Requirements and Error Allocations (3s)

SCIENCE S/C - EARTH IMAGE   RELATIVE POSITION
±2.0 cm Image Position Error ⊥ LOS,
(to fit within a 5 cm telescope entrance aperture)

Allocation, cm

0. 21	**Mirror knowledge, Optical Metrology 3ọ filtered**
0. 53	**Earth direction, E-S Sensor centroid 3ọ filtered**
0. 03	**5m aperture offset X S/C pointing control error**
±1.00	**Relative Position control error**

Earth Atmosphere Observatory at L2
System Requirements

Line-of-Sight Jitter Management

- **Frequencies > 100Hz :**
  Managed by mechanical design so that there is no significant jitter in this range.

- **Frequencies < 100Hz :**
  Jitter is removed using a Limb Detector and Fast Steering Mirror in the science telescope to keep the limb image stable in the science instrument FOV.

Earth Atmosphere Observatory at L2 System Requirements

Fast Steering Mirror / Limb Detector

- FSM Field of Regard shall be large enough to capture the limb image anywhere in the entrance aperture.
- Limb Detector (and limb position estimator) shall have sufficient accuracy to meet the science pointing requirement and track low frequency jitter to the required accuracy.
- The FSM controller shall have the bandwidth needed to track image motion up to 100 Hz at the required jitter accuracy.

Earth Atmosphere Observatory at L2 System Requirements

Options / Trades

- Spinning aperture for limb sampling rather than free flying relay mirror. Excessive fuel required, impractical.
- Selection of Earth-Sun sensor for Earth relative navigation rather than extensive, nearly continuous, DSN support. Still need DSN Nav of Carrier L2 deployment and periodic updates.
- Fine pointing control to target image with Fast Steering Mirror and Limb Detector in science telescope helps to ease requirements on Formation Control.

Earth Atmosphere Observatory at L2

This Section Describes the Observatory Initialization Sequence following Separation of Both S/C from LV/Carrier Transporter in L2 orbit space within specified position and velocity state accuracies for acquisition of the Earth-Sun Line by Onboard Sensors

Observatory Initialization Sequence following Separation

Delivery at L2
<u>By One Carrier</u>

- Delivery of both spacecraft on a single carrier allows a controlled separation and continuous knowledge and control of the relative position of the two spacecraft. This avoids inadvertent collision risk. The carrier must move away a safe distance and then return home.
- The delivery point should be the Sun-EM_barycenter L2 which is the mean position of the Sun-Earth L2 point. The Sun-Earth line sweeps past this point twice a month providing frequent rendezvous opportunities. The force free orbit at the delivery point is not stationary, because of the Earth-Moon motion varies the gravitational field, but can vary ±100 km in the tangential direction (don't care about the radial motion and the normal motion is smaller).
- Delivery accuracy, in the normal direction, should be <7000 km to guarantee at least 50% of the Earth Limb overlaps the Sun for operation of the Earth-Sun sensor (<600km for 100%). Beyond 13000 km there is <u>no</u> occultation and we would have to search for the Earth. In the tangential direction an extra 5000 km can be tolerated because the monthly motion of the Sun-Earth line sweeps out a larger space, you just have to wait for the Earth to come by.

Observatory Initialization Sequence following Separation

Delivery at L2 [2]
By Two Carriers (Not Recommended for Baseline)

- Delivery of the two observatory spacecraft by separate carriers does not allow autonomous initialization of relative position knowledge. A less accurate initialization would use data from ground tracking (or perhaps carrier to carrier relative navigation).
- The two delivery points should be on a common Sunline with the Science spacecraft on the Sun side of the Aperture spacecraft. The separation must be large enough to avoid collision and have relative direction error small enough to allow acquisition by the RF metrology.
- Delivery accuracy in absolute coordinates is the same as for one-carrier delivery, but the considerations in the previous paragraph drive the requirements on the knowledge of relative position, which in turn constrains the absolute position knowledge accuracy.

Observatory Initialization Sequence following Separation

Deployment from Carrier(s)

Separate Spacecraft from Carrier

- Prior to separation from the Carrier each spacecraft should be turned on, checked out and be ready for autonomous operation. Position and attitude estimators are initialized and operating. The Carrier is responsible for controlling separation direction. Separation paths of the two spacecraft shall not intersect, to avoid possible collision prior to full stabilization of each S/C.
- After separation each spacecraft will turn to a specified attitude (e.g. SunPoint), and stop after moving a specified distance, and deploy stowed appendages, not necessarily in that order. Inter-spacecraft communication (ISC) and RF metrology will be established as soon as possible.
  The specified stopping points should leave the spacecraft in position for RF acquisition.

Observatory Initialization Sequence following Separation

Deployment from Carrier(s) [2]

Control Spacecraft Relative Positions

- After separation, relative position is propagated open-loop, i.e., dead-reckoned by IMU inertial Nav, and ISC (inter-spacecraft communications) is needed if each spacecraft is to know what the other is doing. The separation distance of the stopping points should be consistent with the expected accuracy if a collision is to be avoided.
- Acquiring RF metrology is necessary to improve the relative position knowledge accuracy and further insure no collision due to growth of dead-reckoning error. The RF accuracy is limited until full deployment of the primary mirror but should be adequate for collision avoidance.
- Once relative position measurements are available the desired relative position of the spacecraft can be refined, needing less margin for collision avoidance.
- At this point we can designate one spacecraft as the leader and the other as the follower. Consider that the follower controls relative position and the leader controls absolute position (or position relative to the Sun-Earth line target position). These responsibilities can be shared but it is easier to talk about if separated.
- We still have not improved the absolute position knowledge. At this point all knowledge of absolute position comes from ground tracking and orbit determination. A spacecraft Sun Sensor with stellar attitude determination can provide some plane-of-sky position knowledge but it would not be very accurate (2.6E6km per degree of error).

Observatory Initialization Sequence following Separation

Deployment from Carrier(s) [3]

Primary Mirror Deployment and Figure Determination

- There is no reason, at this point in the deployment scenario, to deploy the mirror. Determining mirror figure should be deferred until on the Sun-Earth line.

- If the mirror is deployed then the collision "keep out zone" of the Aperture spacecraft includes not only the structure but also the area around the focused sunlight, 125m away.

Observatory Initialization Sequence following Separation

Sun-Earth Acquisition

Begin Sun-Earth Line Navigation

- If the delivery accuracy requirement is met the Earth-Sun sensor, pointed toward the Sun, should see the Earth occulting the Sun image. The degree of occultation depends on the delivery accuracy and the "time of month".
- Ephemeris knowledge gives the inertial Sun-EM_barycenter vector, the offset of the Earth from the EM_barycenter and the corresponding velocities.
- The Earth-Sun sensor data is used to estimate the center of the Earth-shadow and Sun limb images. There is a reduction in accuracy if there is only a partial limb.

  *The Sun direction is mapped to inertial coordinates using attitude information and can be used to estimate the absolute delivery accuracy to the Sun-EM_barycenter line (plane of sky) but that error need not be corrected unless it is so large that it adversely affects the occultation. Over time, we can also estimate the velocity.

  *The Sun-Earth offset is estimated and tells the offset from the moving target Sun-Earth line, and the rate of change. This provides the relative position and velocity information required to plan the rendezvous with the moving target point. Over the monthly cycle, the offset varies ±5000km and the degree of occultation varies, with the most accurate measurements occurring when there is a complete Earth shadow.

  *The Earth diameter can be estimated and thus the range, but a lot of filtering is required to get an estimate accurate enough to use for rendezvous in the radial direction.

Observatory Initialization Sequence following Separation

Sun-Earth Acquisition [2]

<u>Rendezvous with Sun-Earth Target Point</u>

- In a uniformly rotating Sun-EM_barycenter coordinate system, the ephemeris data provides the information required to compute the position and velocity of the Sun-Earth line target point vs time. The Sun-Earth offset estimate from the Earth-Sun sensor the is used to estimate the spacecraft position and velocity in that coordinate system. To have the spacecraft rendezvous with the target point we must plan a propulsive maneuver to move from the spacecraft position/velocity to a future target point position/velocity. At this point this spacecraft would then follow the target point acceleration profile (plane of sky).
- Degrees of freedom are the start and stop times of the maneuver. A constraint is the available acceleration. Optimization criteria may be minimum fuel use, earliest completion time, etc. The solution of this problem is the subject of future technology development.

Observatory Initialization Sequence following Separation

Absolute/Relative Position Control

<u>Follow Sun-Earth Target</u>

- **At this point this spacecraft will follow the target point acceleration profile using *feed-forward* acceleration commands, with correction (*feed-back*) accelerations derived from the Earth-Sun sensor estimation of the tracking error. The spacecraft will then follow the Sun-Earth line within the required accuracy.**
- **The relative positions of the two spacecraft will have been held at a safe distance to avoid collision and area of focused sunlight.**

Observatory Initialization Sequence following Separation

Absolute/Relative Position Control [2]

Deploy Primary Mirror

- If the Primary Mirror is not yet deployed we must do so now before proceeding further in this sequence of events. Once the ground verifies deployment we can proceed.
- The monitoring of the deployment for ground verification would be by some imaging sensor, not yet defined.

Manage Mirror Figure

- With a stable environment established, the Science spacecraft is moved to the predicted center of curvature of the Primary Mirror, 250m away. This is still well away from the focused sunlight.
- The Center-of-Curvature sensor will determine the quality of the spherical shape of the mirror
- The Optical Metrology sensor will survey the surface fiducials of the mirror to determine its shape, or deviation from the desired 250m radius sphere. This is the opportunity to cross-calibrate these two sensors.
- Mirror shape control is initiated to bring the mirror to the desired shape. This is continued until the shape is within the required tolerance.

Observatory Initialization Sequence following Separation

Absolute/Relative Position Control [3]

Establish/Maintain Science Observatory Configuration

-

- With the mirror shape known, the position of the Sun-Earth image from the Primary mirror is determined in Aperture Spacecraft coordinates using the spacecraft attitude and inertial direction to the Earth.
- At this point in the timeline the Science Spacecraft telescope must be rotating. This was not required earlier.
- The Science Spacecraft is now moved so that the center of rotation of the entrance aperture is at the optical model predicted center of the Sun-Earth image. Relative spacecraft position is measured using RF and Optical Metrology for closed-loop control of relative position.
  The primary mirror shape is continually monitored and readjusted as required.
- The telescope entrance aperture should be scanning the Earth limb.
  The Limb Detector in the telescope will determine the true limb position and control the Fast Steering mirror to place the limb in the Science Sensor FOV. This same correction can be applied as a vernier control on the Science Spacecraft position relative to the image, calibrating and correcting for errors in the image location prediction and relative position control.
- We are now ready to start taking science data.

Observatory Guidance, Navigation, & Control

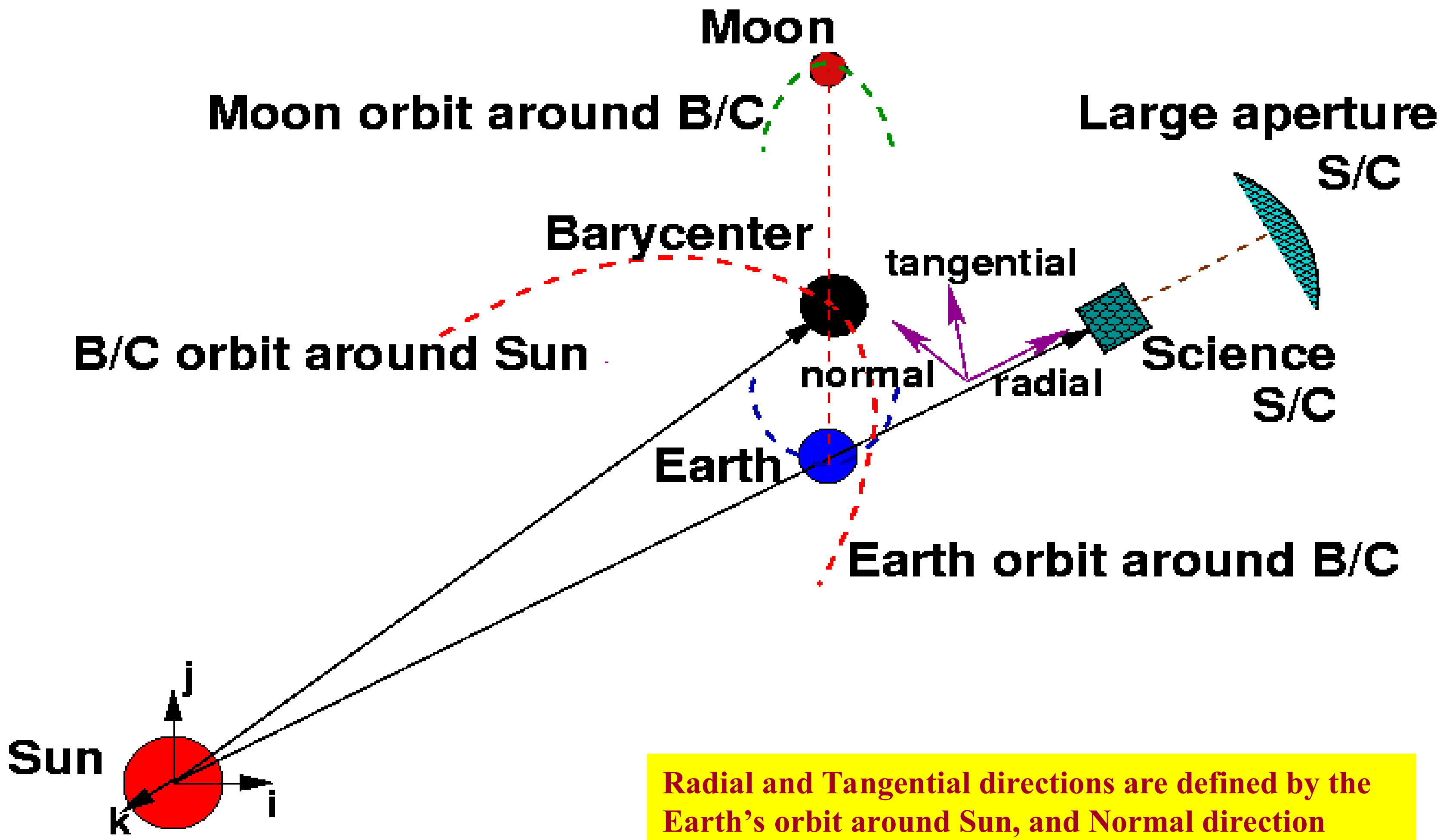
Formation Orbit and Stationkeeping Control Analysis, Design, and GNC Hardware Mass and Power

Basic Orbit Requirements for Science Phase

1. Stay on Sun-Earth line at a mean distance from Earth of 1.51 million km's, with approximately +/- 5000 km monthly oscillations

2. Stay within 200 km of Sun-Earth line

Orbital Geometry



Radial and Tangential directions are defined by the Earth's orbit around Sun, and Normal direction completes the coordinate system

NOT TO SCALE

ORBITAL DEFINITIONS

- **L2 Point:** The L2 point of Sun and Earth-Moon Barycenter
- **Barycenter:** Mass center of Earth and Moon
- **Sun-Earth Line:** The line connecting Sun and Earth
- **Projection of a point, P, on Sun-Earth Line:** The closest point on Sun-Earth line to P

Orbital Analysis

Orbital Analysis for Fuel/ΔV Requirements in Science Phase of the mission

We consider three orbits for the Leader S/C (Science S/C), where the leader follows a point P described as:

Orbit-1: P is on Sun-Earth line at a fixed distance of 1.51 million km from Earth

Orbit-2: P is at the projection of L2 on Sun-Earth line

Orbit-3: P is at L2 point (this orbit is not on Sun-Earth line)

Properties of the 3 Orbits

Orbit 1: **This is a feasible orbit, satisfying main mission requirements, with the greatest fuel use.**

Orbit2: **This is a feasible science orbit, with an optimal fuel use for the formation staying on Sun-Earth line. This is the current Baseline Orbit.**

Orbit 3: **This orbit has minimum fuel requirements because it is approximately the equilibrium point . But it does not satisfy the mission requirements.**

*** All Orbital Studies Assumed Advanced Xe Ion Drive Thrusters Based on Project Prometheus Data**

Summary of Results for the Orbits

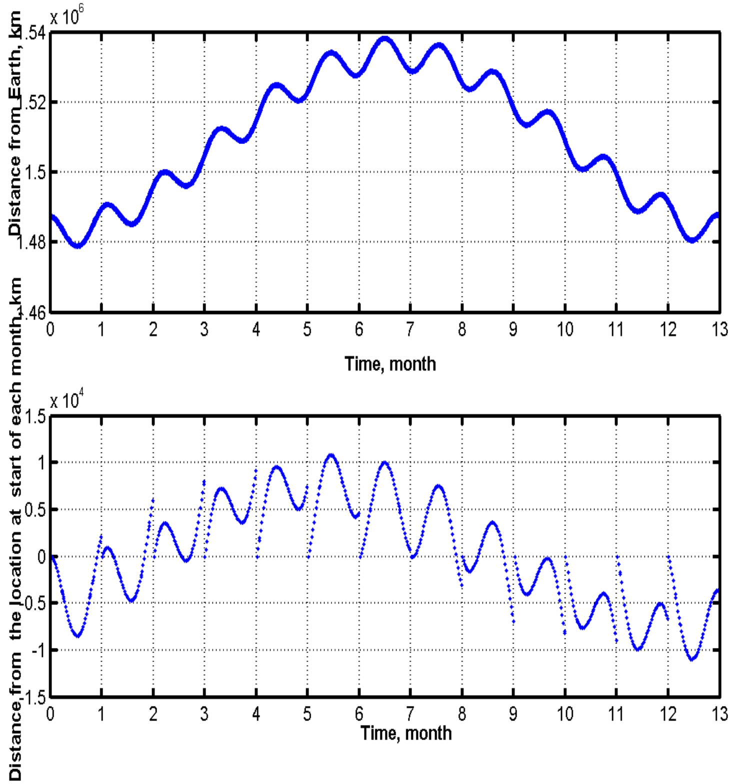
Parameter	Trajectory 1	Trajectory 2*	Trajectory 3
ΔV-radial (m/s/day)	1.939	0.234	0.025
ΔV-tangential (m/s/day)	1.809	1.561	0.015
ΔV-normal (m/s/day)	0.162	0.162	0.002
ΔV-sum (m/s/day)	3.909	1.957	0.042
Max. radial force for 1000 kg S/C (mN)	41.7	5.3	0.77
Max. tangential force … … (mN)	38.7	31.8	0.36
Max. normal force … … (mN)	3.3	3.3	0.06
•Fuel for 10 yrs (kg), 6000 s Isp, 1000kg S/C	242	121	2.6

(Used as Relative Measure Not Final Value)

* *Baseline Orbit*

For the simulations above, we only considered orbital accelerations due to gravity forces and IDEAL Thrusters

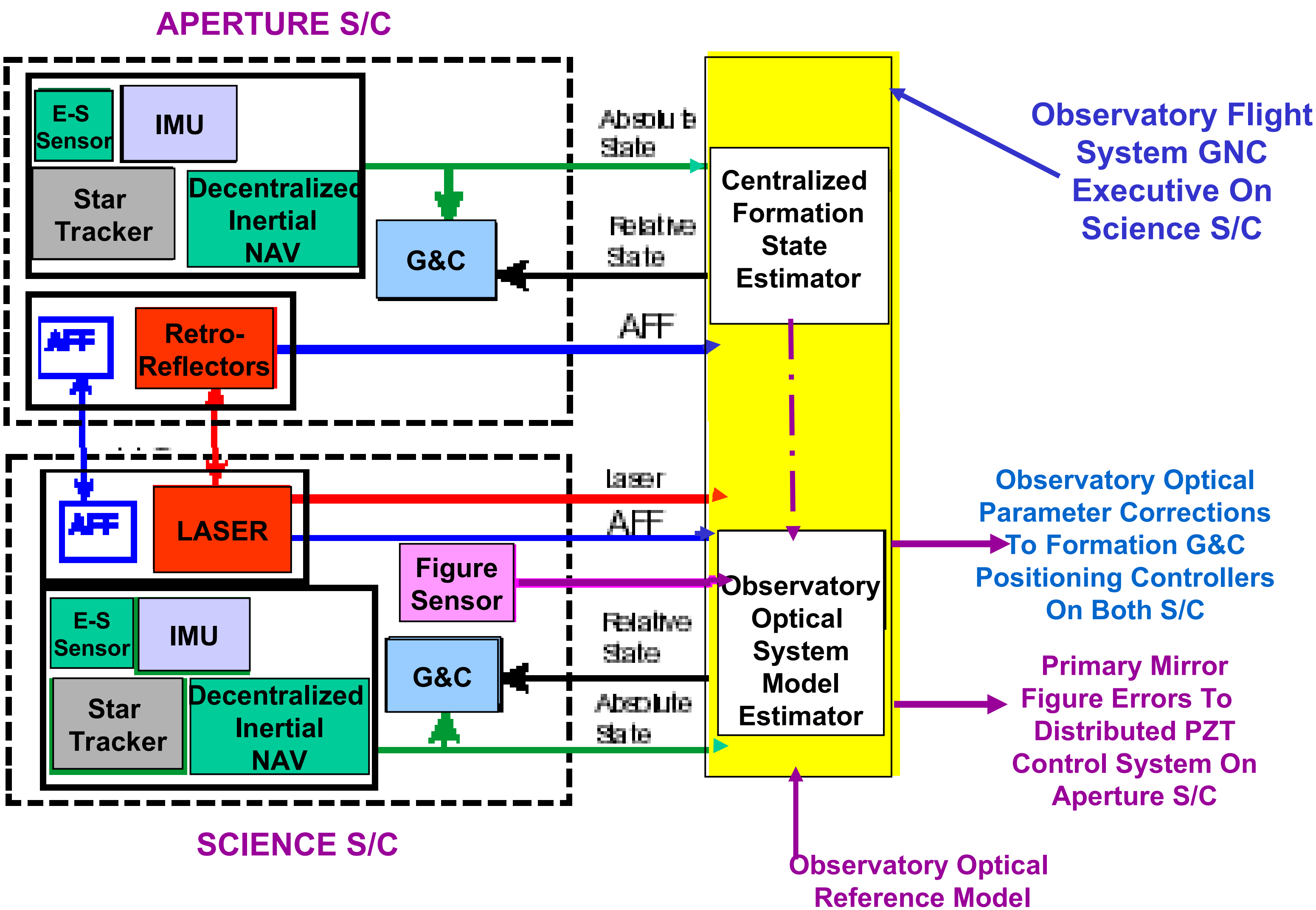
Motion of Leader (Science S/C) on Sun-Earth Line following Orbit-2



Observations

1. Leader (Science S/C) moves in and out nearly **29,700 km *annually***, about a mean distance of **1.5075 million km's** from Earth on Sun-Earth line
2. Leader S/C moves in and out nearly **4550 km's *monthly***, about a monthly mean location on Sun-Earth line

Formation GNC Concept



NASA RASC

Aperture S/C GNC Mass, Power, Equipment List

RASC - L2 Earth Observatory - Aperture Spacecraft														
Current Best Estimate							Future Best Estimate							
System	Source	Quantity	Total Mass CBE (kg)	Average Power CBE (W)	Peak Power CBE (W)	Comments	Source	Quan-tity	Total Mass FBE (kg)	Average Power FBE (W)	Peak Power FBE (W)	Technology SOA Date for FBE	Comments	Revisions
GNC	Subtotal		49	66	156		Subtotal		21	21	39			
Formation Ka-Band Metrology	Ka-Band Transceiver & 4-Patch Antennae; one cm range, one arcminute bearing relative accuracy	2	4	2	2	Ongoing funded development for TPF. Relative Range & Bearing bt S/C	Advanced Ka-Band Transceiver & 4-Patch Antennae; 0.5-cm range, 0.5-arcminute bearing relative accuracy	2	1	1	1	2015	Reduced mass & power from further miniaturization/ integration of electronics chips and packaging technology	
Sun-Earth Limb Sensor	Similarity Ref is Questar-7 Maksutov Cassegrain Catadioptric: New design is 0.4-m Folded, 2.5-m Focal length, 10-cm Dia, 1.0 Deg FOV Telescope w/4096 Square CCD	1	4	7	7	New Development needed for NAV function: Lightweight long focal length Telescope, Large format CCD for ~4.2 microradian resolutiopn, w/ internal detector and electronics Redundancy	Similarity Ref is Questar-7 Maksutov Cassegrain Catadioptric: Advance lightweight design is 0.4-m Folded, 2.5-m Focal length, 10-cm Dia, 1.0 Deg FOV Telescope w/4096 Square CMOS APS	1	2	2	3	2015	Athermalized Light-weight Structure & Optical Train of Titanium or Silicon Carbide or Beryllium barrel and Lens Cells, BK-7, Zerodour, Pyrex, coated lens and mirrors, Heat rejection filters.	
Star Cameras & electronics	(4) Star Camera Heads plus (2) Electronics, CCD Detectors	1 Set	3	4	4	DTU Adv Stellar compass; Orsted, SAC-C, ADEOS 11, CHAMP	(4) Star Camera Heads plus (2) Electronics, Adv. CMOS APS Detectors w/on-chip processing	1 Set	2	2	3	2015	Reduced mass & power from further miniaturization/integration of electronics & packaging and APS tech.	
IMUs	Litton SIRU HRG, Internal Redundant Gyros/Accels, Electr.	1	5	27	27	Cassini; EOS AURA, CHEMISTRY: Bias Stability 0.003 deg/hr, R-W 0.0001 deg/rt-hr	Advanced MEMS version Litton SIRU, Internal Redundant Gyros/Accels, Electr.	1	1	4	4	2015	Reduced mass & power from further miniaturization/ integration of electronics chips and packaging technology	
Coarse Sun Sensors	EDO-Barnes 0.7 Deg Accuracy	4	1	1	1	FOV: 128 X 128 Deg Acquisition Sensor	Advanced Tech Version: APS detector tech, 0.2 Deg Accuracy	4	1	1	1	2015	FOV: 180 X 180 Deg, Digital CMOS 1000 X 1000 format APS w/on-chip processing	
Reaction wheels & drive electronics	SMEX (Clagget) 4 Nms; 0.14 Nm	4	16	8	60	TRACE, WIRE, SWAS	Adv lightweight low power version SMEX (Clagget) 4 Nms; 0.14 Nm	4	5	3	15	2015	High speed low mass composite rotors on magnetic bearings & miniaturized lower power electronics	
Interface Electronics for GNC Sensors & Actuators, and RIU to Main Data Bus	I/O cards for Data I/F w C&DH Main Bus via RIU; Analog & digital I/O; A/D, D/A conversions; house keeping sensors; I/F w/ EP Thrusters Control and Metrology	4	2	2	2	All data channels are redundant. Mil-Std 1553B assumed for main data handling bus architecture, and distributed subsystem microprocessors I/F via RIU's (remote interface units); GNC LAN	I/O 3-D modules for Data I/F w C&DH Main Bus via RIU; Analog & digital I/O; A/D, D/A conversions; house keeping sensors; I/F w/ EP Thrusters Control and Metrology	1	1	1	1	2015 - 2020	Advanced 3-D stacked lowpower microchip technology or SUGAR CUBE architecture	
GNC Computers	Adv BAE SFC	2	1	5	26	Next Gen RAD 750 Microprocessor	3-D Ultra-Thin Chip Scale Integration SFC-Internal multi-Redundant w/ autonomous reconfiguration mngt.	1	1	2	5	2015 - 2020	Nanoscale Flight Computer w/ IC feature size <10 nm & ~ 9 Giga Gates/chip; 3-D Sugar Cube packaging architecture	
Non Volatile Memory	RAM NVM	4	4	2	6	12 GBIT Data per Card	Next Generation NVM RAM: FRAM, MRAM, CRAM Options	1	1	1	2	2015	Feature size 22 nm, 64GBit Data per Module, Non-volatile and Dynamic Random-Access	
DTCI I/F	Data, TLM, CMD, EP, Star Cameras Interfaces	2	1	3	7		3-D Ultra-Thin Chip Scale Integration- Internal Redundant w/ autonomous reconfiguration mngt.	1	1	1	1	2015	Advanced 3-D stacked lowpower microchip technology or SUGAR CUBE architecture	
CMIC I/F	Heartbeat, Sys Reset Tr, A/B Side Selects	2	1	1	2		3-D Ultra-Thin Chip Scale Integration- Internal Redundant w/ autonomous reconfiguration mngt.	1	1	1	1	2015	Advanced 3-D stacked lowpower microchip technology or SUGAR CUBE architecture	
ULDL I/F	RS-422 Data I/F to like units	2	1	2	8		3-D Ultra-Thin Chip Scale Integration- Internal Redundant w/ autonomous reconfiguration mngt.	1	1	1	1	2015	Advanced 3-D stacked lowpower microchip technology or SUGAR CUBE architecture	
GIF I/F	Discrete I/O, Mil-Std 1553B I/F	2	1	3	5		Ultra-high speed Duplex-Redundant Data bus	1	1	1	1	2015	Advanced 3-D stacked lowpower microchip technology or SUGAR CUBE architecture	
Backplane	6U cPCI	2	3				New Generation 3-D redundant architecture	1	1			2015	Advanced 3-D stacked lowpower microchip technology or SUGAR CUBE architecture	
Chassis	Aluminum Enclosure	1	4				Compact 3-D Al enclosure	1	1			2015		
Subsystem Shielding	GNC: TBD for Solar Energetic Particles over 10 years		TBD			Need Environmental-Equipment Analysis	GNC: TBD for low susceptability		TBD				Robust parts & packaging technologies	

Science S/C GNC Mass, Power, Equipment List

RASC - L2 Earth Observatory - Science Spacecraft

	Current Best Estimate						Future Best Estimate							
System	Source	Quantity	Total Mass CBE (kg)	Average Power CBE (W)	Peak Power CBE (W)	Comments	Source	Quantity	Total Mass FBE (kg)	Average Power FBE (W)	Peak Power FBE (W)	Technology SOA Date for FBE	Comments	Revisions
GNC	Subtotal		75	92	206		Subtotal		34	36	62			
Coarse Sun Sensors	EDO-Barnes 0.7 Deg Accuracy	4	1	1	1	FOV: 128 X 128 Deg Acquisition Sensor	Advanced Tech Version: APS detector tech, 0.2 Deg Accuracy	4	1	1	1	2015	FOV: 180 X 180 Deg, Digital CMOS 1000 X 1000 format APS w/on-chip processing	
Formation Ka-Band Metrology	Ka-Band Transceiver & 4-Patch Antennae; one cm range, one arcminute bearing relative accuracy	2	4	2	2	Ongoing funded development for TPF. Relative Range & Bearing bt S/C	Advanced Ka-Band Transceiver & 4-Patch Antennae; 0.5-cm range, 0.5-arcminute bearing relative accuracy	2	1	1	1	2015	Reduced mass & power from further miniaturization/ integration of electronics chips and packaging technology	
Formation Optical Metrology	Laser Transceiver & Platform	2	10	10	20	Ongoing funded development for TPF. Relative Range & Bearing bt S/C	Internal Redundant Laser Transceiver	1	6	7	7	2015	Reduced mass & power from miniaturization/ integration of electronics packaging & Internal Redundancy	
Sun- Earth Limb Sensors	Similarity Ref is Questar-7 Maksutov Cassegrain Catadioptric: New design is 0.4-m Folded, 2.5-m Focal length, 10-cm Dia, 1.0 Deg FOV Telescope w/4096 Square CCD Detector & Digital Electr.	1	4	7	7	New Development needed for NAV function: Lightweight long focal length Telescope, Large format CCD for ~4.2 microradian resolutiopn, w/ internal detector and electronics Redundancy	Similarity Ref is Questar-7 Maksutov Cassegrain Catadioptric: Advance lightweight design is 0.4-m Folded, 2.5-m Focal length, 10-cm Dia, 1.0 Deg FOV Telescope w/4096 Square CMOS APS Detector, on-chip processing, Redundant internal detectors & electronics	1	2	2	3	2015	Athermalized Lightweight Structure & Optical Train of Titanium or Silicon Carbide or Beryllium barrel and Lens Cells, BK-7, Zerodour, Pyrex, coated lens and mirrors, Heat rejection filters.	
Star Cameras & electronics	(4) Star Camera Heads plus (2) Electronics, CCD Detectors	1 set	3	4	4	DTU Adv Stellar compass; Orsted, SAC-C, ADEOS 11, CHAMP	(4) Star Camera Heads plus (2) Electronics, Adv. CMOS APS Detectors w/on-chip processing	1 Set	2	2	3	2015	Reduced mass & power from further miniaturization/integration of electronics & packaging and APS tech.	
IMU	Litton SIRU HRG, Internal Redundant Gyros/Accels, Electr.	1	5	27	27	Cassini; EOS AURA, CHEMISTRY: Bias Stability 0.003 deg/hr, R-W 0.0001 deg/rt-hr	Advanced MEMS version Litton SIRU, Internal Redundant Gyros/Accels, Electr.	1	1	4	4	2015	Reduced mass & power from further miniaturization/ integration of electronics chips and packaging technology	
Reaction wheels & drive electronics	SMEX (Clagget) 4 Nms; 0.14 Nm	4	16	8	60	TRACE, WIRE, SWAS	Adv lightweight version SMEX (Clagget) 4 Nms; 0.14 Nm	4	5	3	15	2015	High speed low mass composite rotors on magnetic bearings & miniaturized lower power electronics	
Telescope Rotation Drive Motor & Electronics	2-Phase DC Torque Motor w/ 12-Bit Absolute Optical Encoder	1	3	4	8	Redundant Motor Windings, Encoders, Commutation & Drive Electronics	2-Phase DC Torque Motor w/ 12-Bit Absolute Optical Encoder	1	2	3	6	2015	Reduced mass/power from Higher efficiency magnetic materials & electronics miniaturization	
Momentum Wheel/Drive Electr.	For Telescope Momentum Cancellation	1	10	10	20	Integral Electronics; Aligned to Telescope Spin Bearing Axis; Higher speed counter-rotating rotor	For Telescope Momentum Cancellation	1	5	5	10	2015	High speed low mass composite rotors on magnetic bearings & miniaturized lower power electronics	
Interface Electronics for GNC Sensors & Actuators, and RIU to Main Data Bus	I/O cards for Data I/F w C&DH Main Bus via RIU; Analog & digital I/O; A/D, D/A conversions; house keeping sensors; I/F w/ EP Thrusters Control and Metrology	8	3	3	3	All data channels are redundant. Mil-Std 1553B assumed for main data handling bus architecture, and distributed subsystem microprocessors I/F via RIU's (remote interface units); GNC LAN architecture assumed.	I/O 3-D modules for Data I/F w C&DH Main Bus via RIU; Analog & digital I/O; A/D, D/A conversions; house keeping sensors; I/F w/ EP Thrusters Control and Metrology	1	1	1	1	2015 - 2020	Advanced 3-D stacked lowpower microchip technology or SUGAR CUBE architecture	
GNC Computers	Adv BAE SFC	2	1	5	26	Next Gen RAD 750 Microprocessor	3-D Ultra-Thin Chip Scale Integration SFC- Internal multi-Redundant w/ autonomous reconfiguration mngt.	1	1	2	5	2015 - 2020	Nanoscale Flight Computer w/ IC feature size <10 nm & ~ 9 Giga Gates/chip; 3-D Sugar Cube packaging architecture	
Non Volatile Memory	RAM NVM	4	4	2	6	12 GBIT Data per Card	Next Generation NVM RAM: FRAM, MRAM, CRAM Options	1	1	1	2	2015	Feature size 22 nm, 64GBit Data per Module, Non-volatile and Dynamic Random-Access	
DTCI I/F	All Digital Data, TLM, CMD, EP, Star Cameras Interfaces	2	1	3	7		3-D Ultra-Thin Chip Scale Integration- Internal Redundant w/ autonomous reconfiguration mngt.	1	1	1	1	2015	Advanced 3-D stacked lowpower microchip technology or SUGAR CUBE architecture	
CMIC I/F	Heartbeat, Sys Reset Tr, A/B Side Selects	2	1	1	2		3-D Ultra-Thin Chip Scale Integration- Internal Redundant w/ autonomous reconfiguration mngt.	1	1	1	1	2015	Advanced 3-D stacked lowpower microchip technology or SUGAR CUBE architecture	
ULDL I/F	RS-422 Data I/F to like units	2	1	2	8		3-D Ultra-Thin Chip Scale Integration- Internal Redundant w/ autonomous reconfiguration mngt.	1	1	1	1	2015	Advanced 3-D stacked lowpower microchip technology or SUGAR CUBE architecture	
GIF I/F	Discrete I/O, Mil-Std 1553B I/F	2	1	3	5		Ultra-high speed Duplex-Redundant Data bus	1	1	1	1	2015	Advanced 3-D stacked lowpower microchip technology or SUGAR CUBE architecture	
Backplane	6U cPCI	2	3				New Generation 3-D redundant architecture	1	1			2015	Advanced 3-D stacked lowpower microchip technology or SUGAR CUBE architecture	
Chassis	Aluminum Enclosure	1	4				Compact 3-D Al enclosure	1	1			2015		
Subsystem Shielding	GNC: TBD for Solar Energetic Particles over 10 years		TBD			Need Environmental-Equipment Analysis	GNC: TBD for low susceptability							

Formation Translational Control Strategy

- The translational control design is reduced to a MIMO* tracking problem. Science and Aperture S/C follow the desired orbit while maintaining a separation distance of 125 m on Sun-Earth line, with an accuracy of +/- 1cm.
- Xe IonThrusters are used with continuous actuation.

*MIMO: Multi-Input/Multi-Output

- Control input for each S/C has two components:

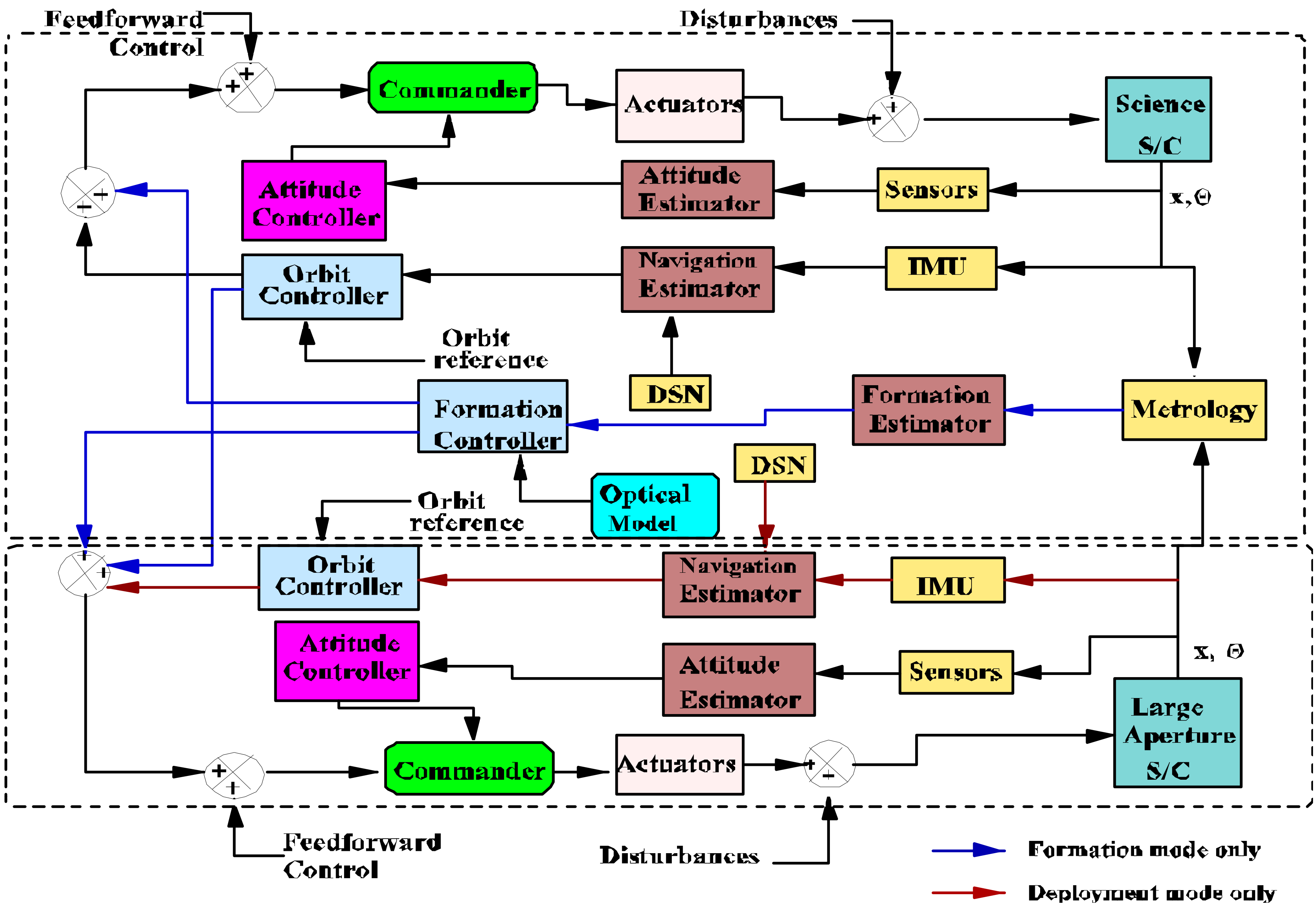
1. Feedforward control input for orbit following
2. Feedback control for formation stationkeeping and correcting errors in orbit following by rejecting disturbances: solar pressure, internal disturbances caused by non-ideal thrusters, and model uncertainties. A robust MIMO PID controller is designed – see control block diagram.

Formation Rotational Control Strategy

- The thruster placement/geometry minimizes force-torque coupling, the rotational control can be treated as a SISO* tracking problem about each S/C axis.
- Controllers align each axis with corresponding radial, tangential, or normal axis, determined by Sun-Earth line.
- A PID controller is designed for each axis, and Reaction Wheels provide necessary control torques.
- Since orbital and disturbance torques are very small, the control strategy involves a simple feedback torque determined by the PID controller for each axis.
- See Spacecraft Control Block Diagram

*SISO: Single Input-Single Output

Spacecraft Control Diagram



Spacecraft Thrusters Configuration, Placement and Sizing

Thruster Configuration

- **Based upon our orbit analysis under gravitational forces and solar pressure (which contributes disturbance force less than 1 mN), we require two sets of thrusters:**

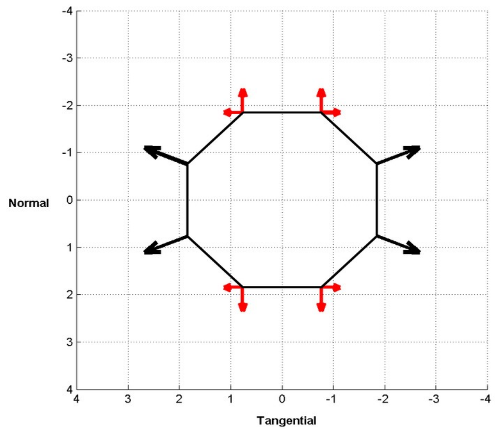
1. **LARGE THRUSTERS (Coarse control): 8 large 30 cm diameter Xenon EP thrusters (4 in alternative configuration), and 4 are redundant. These thrusters provide mean orbit following.**

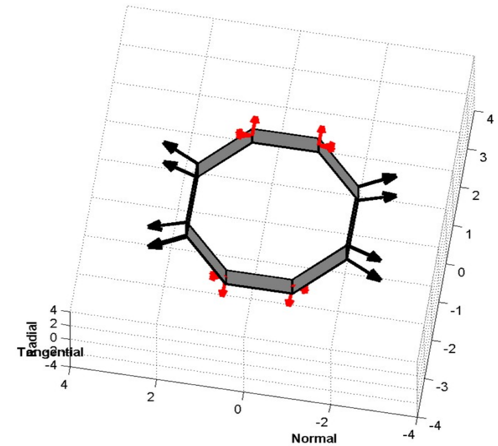
2. **SMALL THRUSTERS (Precision control): 12 small 10 cm diameter Xenon EP thrusters, which provide fine control forces for station keeping and orbit control.**

Thruster Configuration has Additional Properties

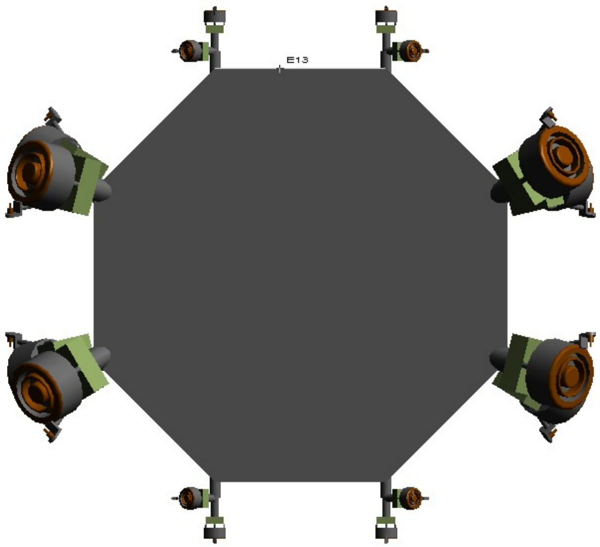
1. **There is significant redundancy in number of thrusters; at most 2 large thrusters fire at a given time. Small thrusters also have redundancy**
2. **The canted large thrusters have line of action passing through the center of mass, preventing unwanted torques, and plume impingement**
3. **The small thrusters in normal direction provide feed-forward control forces, that are much smaller than orbital forces in the other directions**

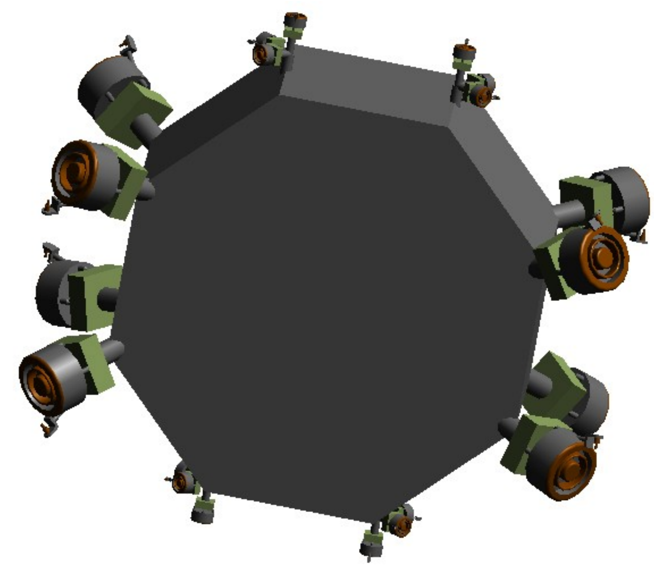
Thruster Configuration





<u>NOTE:</u> ***The radial, normal, and tangential directions are described with respect to the S/C (which are ideally aligned with orbital radial , normal, and tangential)***





Simulations with Non-Linear Canted Thrusters

These simulations evaluated fuel requirements, required peak thrust levels, and control accuracies with large and small canted thrusters in balanced CG operation. Includes all disturbances forces, thruster non-linearities, navigation sensor and metrology errors and noise

Simulation Conditions

- Simulations were conducted to assess the effects of realistic non-linear EP thrusters and system sensor/metrology errors on performance of formation control and orbit following.

- Conditions for both Large and Small Xe EP Thrusters:
  - MIMO PID controller with prescribed orbit acceleration Feed-forward function is designed.
  - Limited number of throttle levels (from 20 to 40 available thrust quantization steps).
  - Throttle levels act like disturbance errors since they are inexact responses to required control forces.
  - Thrust dead-band from 20% to 5% (zero response until DB is exceeded by required thrust)

Simulation Results are shown in the following slides.

Simulation Results of Controlled Formation Following Orbit-2 for a Lunar Cycle

Orbit 2: **6000 s Isp, 1000 kg S/C**

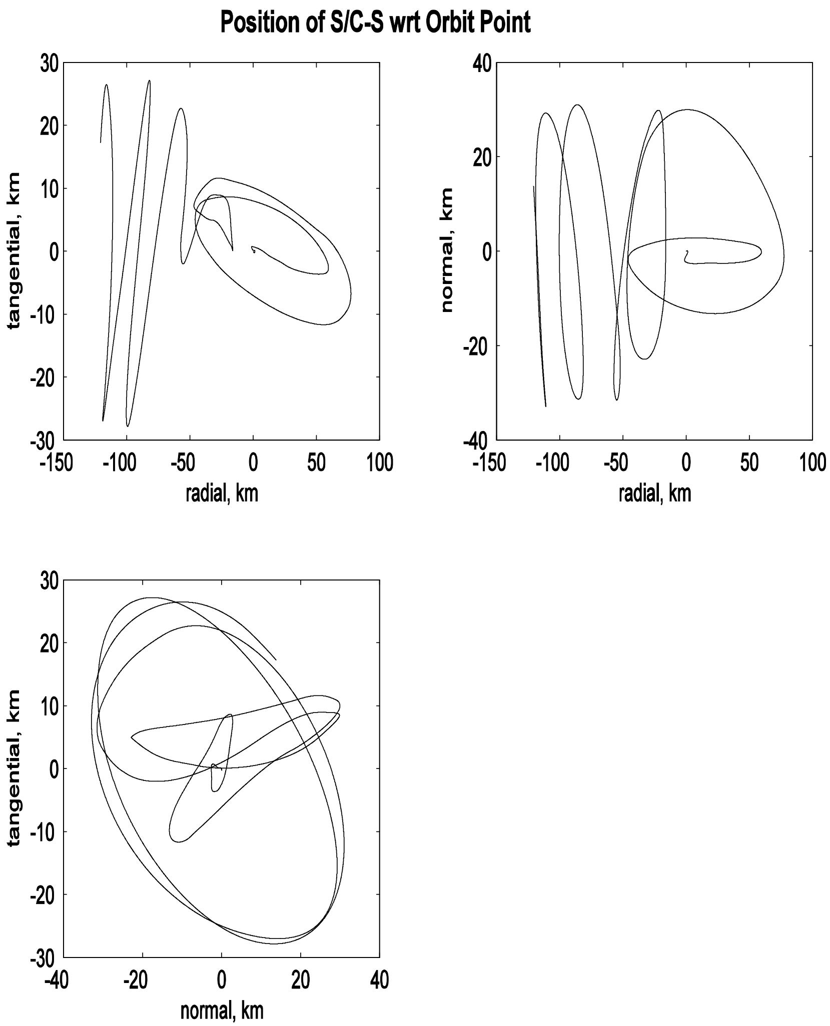
	S/C-A	S/C-S
Fuel for Orbit (kg/month)	0.98	0.98
Fuel for Formation (kg/month)	0.27	0.26
Total Fuel (kg/month)	1.25	1.24
Ratio to Ideal Orthogonal Thrusting Fuel	1.33	1.32
Peak total thrust (mN)	57.6	57.6

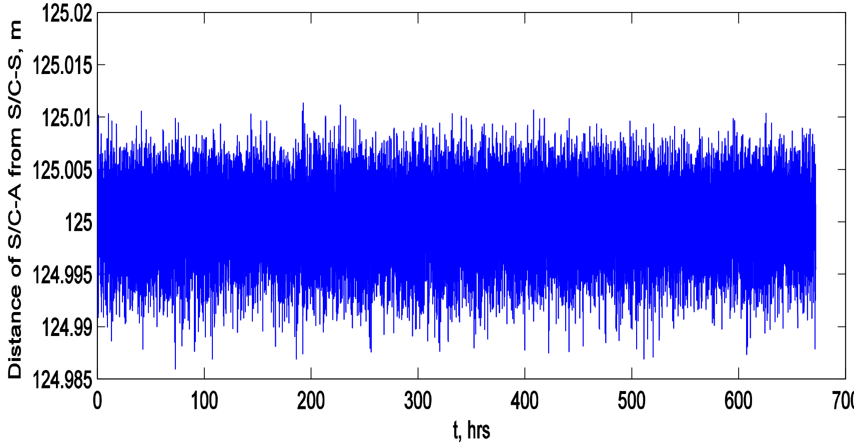
20% Thrust Deadbands and ~ 20 Thruster Throttle Levels for both sizes

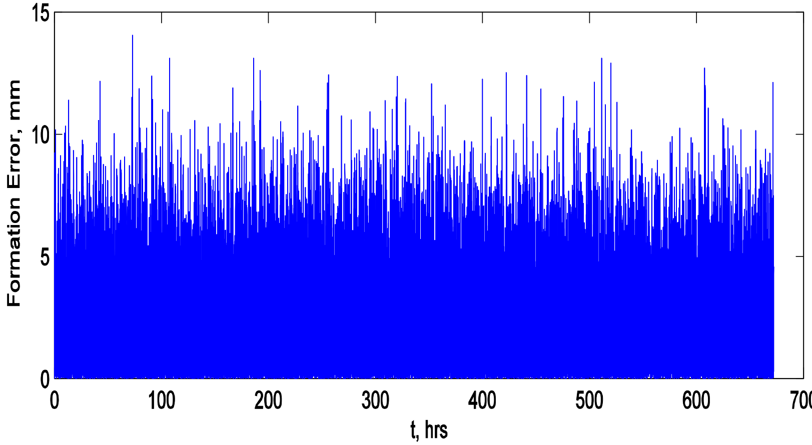
Max. Formation Error (mm)	14
Max. Tangential Orbit Error (km)	30
Max. Normal Orbit Error (km)	35

	Throttle Levels (mN)
LARGE Thrusters	0, 8, 9.6, …, 38.4, 40
SMALL Thrusters	0, 1, 1.2, 1.4, …, 5

Simulation Time Histories







Simulation Results of Controlled Formation Following Orbit-2 for a Lunar cycle

Orbit 2: 6000 s Isp, 1000 kg S/C

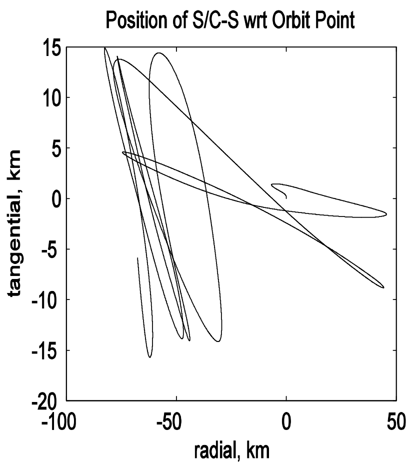
	S/C-A	S/C-S
Fuel for Orbit (kg/month)	0.98	0.98
Fuel for Formation (kg/month)	0.24	0.23
Total Fuel (kg/month)	1.22	1.21
Ratio to Ideal Orthogonal Thrusting Fuel	1.30	1.29
Peak total thrust (mN)	50.3	49.9

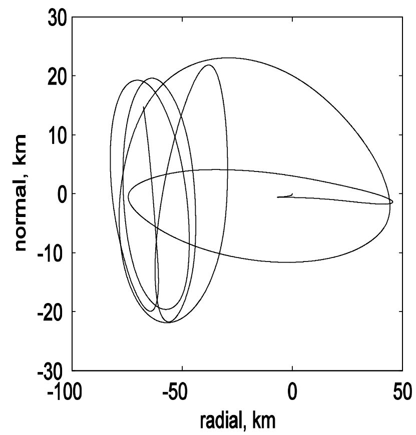
5% Thrust Deadbands and ~ 40 Thruster Throttle Levels for both sizes

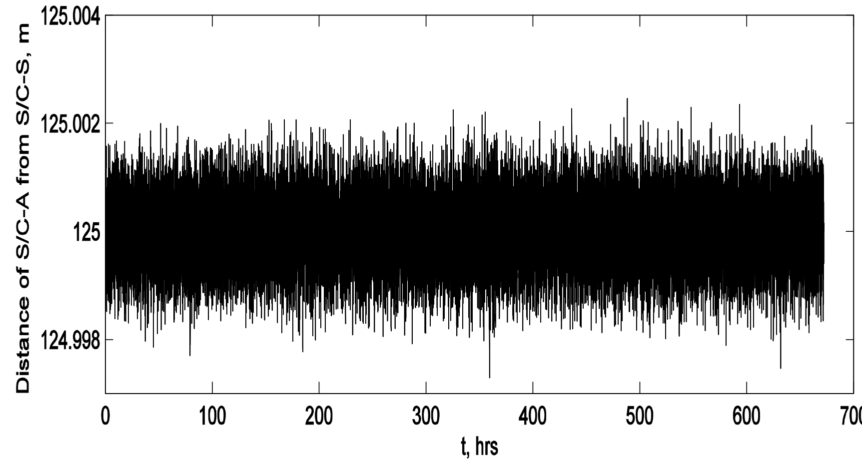
Max. Formation Error (mm)	2.8
Max. Tangential Orbit Error (km)	15
Max. Normal Orbit Error (km)	25

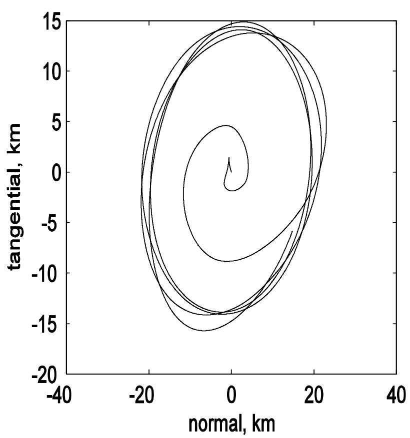
	Throttle Levels (mN)
Large Thrusters	0, 2, 2.8, …, 39.2, 40
Small Thrusters	0, 0.25, 0.35, …, 5

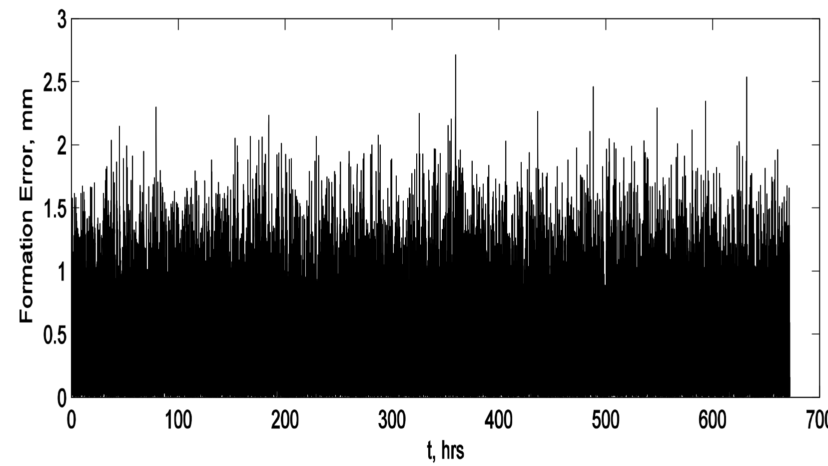
Simulation Time Histories











Effects of Thruster Design Limitations on Performance

1. Simulated performance of orbit following and formation station-keeping are within requirements when thrust quantization steps and deadbands are small enough.
2. However, large time dependent position errors are generated by current technology Xe EP throttle level step sizes and thruster deadband non-linearities.
3. Thruster deadband for current DS1 30 cm diameter EP is ~ 20 % of full thrust or ~ 18 mN. It has about 20 throttle points over ~ 90 mN full thrust with an Xe Isp ~ 3000 s.
4. The L2 Observatory control design requires about 40 throttle points and deadbands </= 5% of full thrust.
5. Isp of EP thrusters needs to be higher. Target should be 6000 s for the fuel mass required of the L2 Observatory.

Technology Development Needed for Xe EP Thrusters

1. **Robust MIMO PID controllers with acceleration feed-forward can not provide required formation performance with current non-linear Xe EP thrusters having coarse throttle levels and large thrust deadbands.**
2. **Also need linear small Xe EP thruster responsive to required variable thrust orbit and formation tracking environment.**
3. **Two combined solutions to this problem are:**

   a) Development of more linear thrusters in large/small sizes

   b) Development of more sophisticated, higher order controllers that may mitigate thruster limitations.
4. **Both paths of development are needed.**

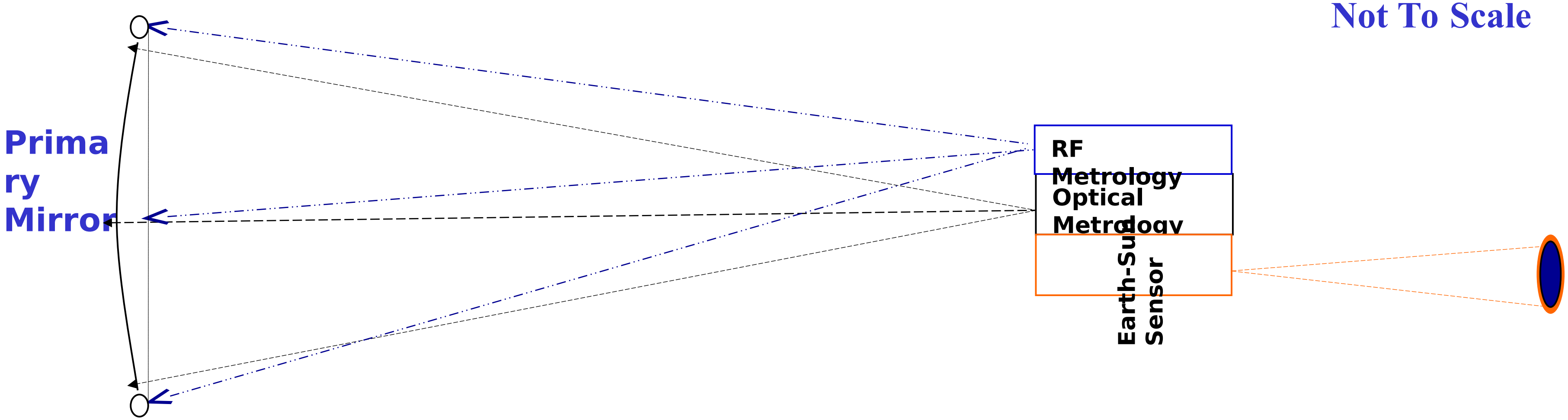
Mission-Specific Metrology & Sensors

The following specialized new sensor types are specific to this mission:

- Formation RF Metrology
- Earth-Sun Sensor
- Primary Mirror Surface Figure Metrology
- Primary Mirror Center-of-Curvature Sensor
- Only a Formation RF Metrology Sensor and an Earth-Sun Sensor are located on the Aperture Spacecraft.
- All four sensor types are located on the Science Spacecraft Metrology Platform.
- This discussion assumes that each spacecraft is also equipped with a suite of typical ACS sensors including Sun Sensors, IMU, and Star Cameras/trackers.

Metrology Platform on Science S/C

Not To Scale



RF Metrology - GPS-like range and phase measurement between transmitter and receiver, triangulated to get relative position and attitude of the Aperture S/C for acquisition and coarse formation control.

Measurement (1σ): Range - 1cm Δphase - 10µm

Optical Metrology - Laser range & bearing to retro-reflectors on the mirror to get precision relative location and attitude and mirror shape for fine formation control and Earth image location prediction.

Measurement (1σ): Range - 1µm Bearing - 10µrad

Earth-Sun Sensor - Image Earth & Sun to find points on the limbs and determine relative Earth direction, position offset from the Earth-Sun line and coarse Earth range.

Measurement (1σ): Limb point - 20µrad

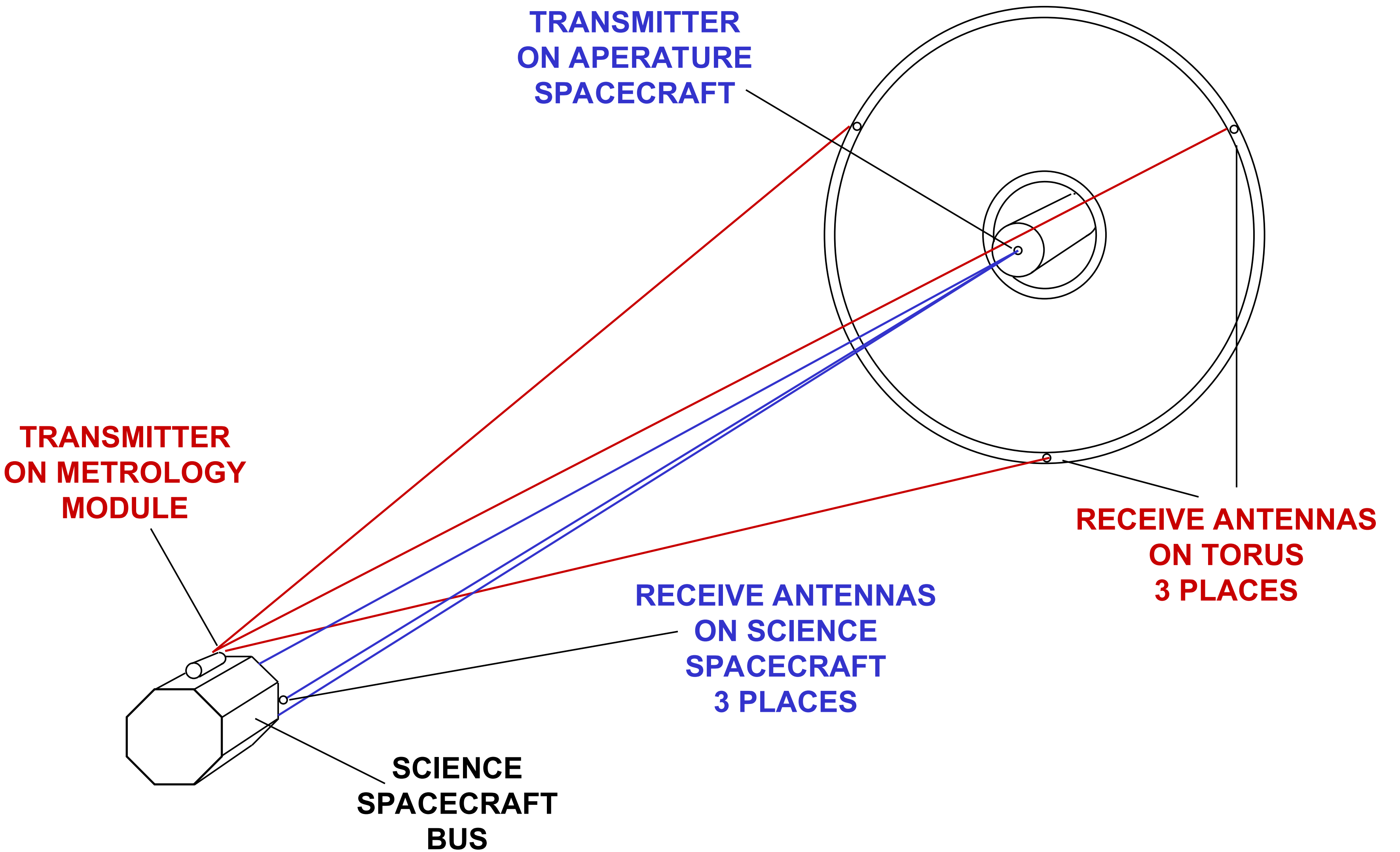
Center-of-Curvature Sensor - Provides initial global test of surface figure. Not separately shown in above diagram.

Autonomous Formation Flying RF System (AFF)

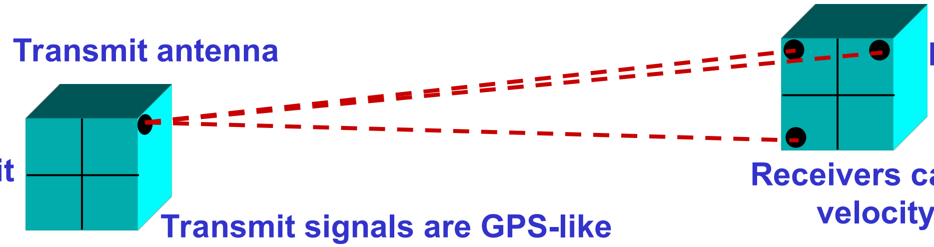
<u>Currently Funded JPL Development</u>

- Ka-band Duplex Links
- 3 Transceivers with Patch Antennas on each S/C
- Relative Range, Range Rate, Bearing, and Bearing Rate

Observatory Formation RF Metrology



Observatory Formation RF Metrology

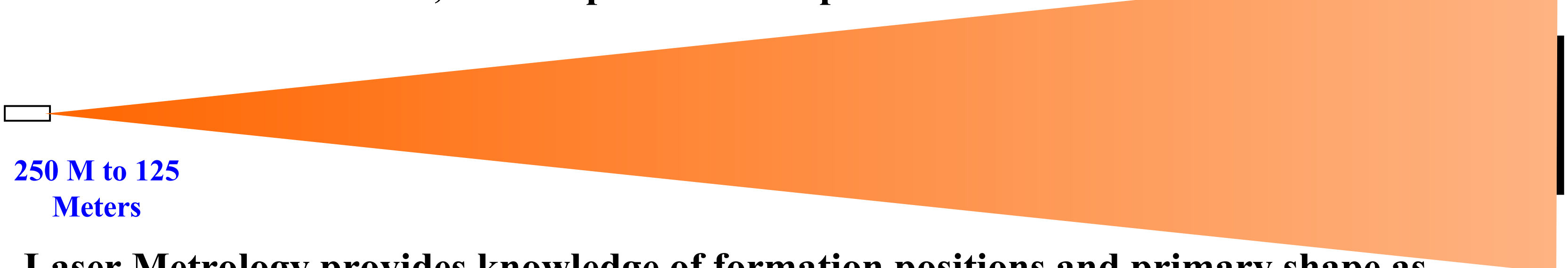


- Metrology is based on Transceiver- Antenna units placed on both S/C (Duplex Links)
- GPS-like transceivers determine relative ranges and attitudes
- No NAVSTAR GPS signals are required for relative measurements
- Will be based on JPL's existing "Turbo Rogue" system
  - Requires minimum apriori knowledge
  - Readily adaptable to variable baselines with 100 m to 1 km separation
- Will determine the coarse 3D formation geometry
- Characteristics per element:
  - Transceivers use Ka-band signals
  - Two small transmit and 6 small receive antennas
  - Current mass 2 kg, power<1 W. Much less in future with "System-on-a-chip"
  - Has up to 1,300 km range
- Current capability precision:
  - ±1 cm relative ranging
  - ± 1 arc minute relative orientation
  - ± 0.1 mm/sec relative velocity

Initialization Using Optical Metrology

At 250 Meters

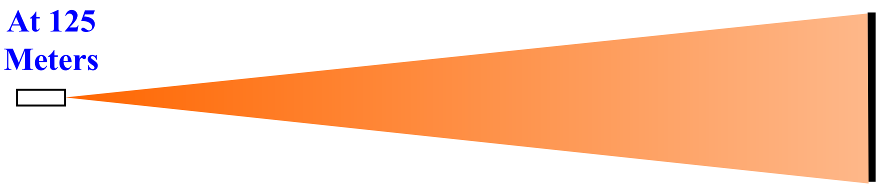
Science S/C moves to nominal center of curvature of primary under control of Formation Metrology. Center-of-Curvature Sensor and Shape Metrology calibrate the primary shape, true center of curvature, and shape control responses.



250 M to 125 Meters

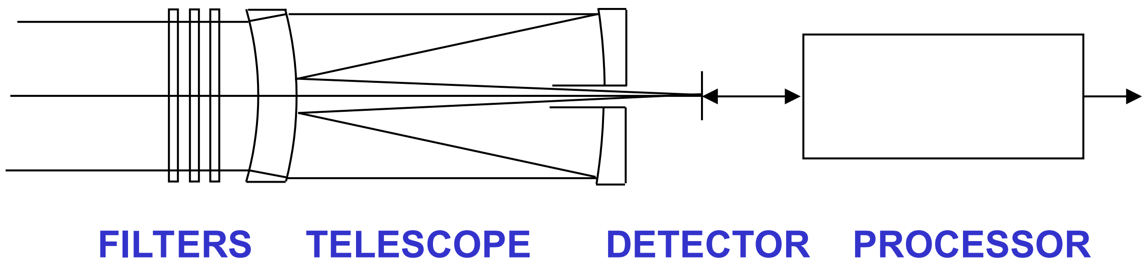
Laser Metrology provides knowledge of formation positions and primary shape as Science S/C moves from center of curvature to focal point.

At 125 Meters



Laser Metrology provides formation and shape knowledge during observation phase.

Earth-Sun Sensor Enables Accurate Orbit Nav



Design Concept

- A Maksutov telescope of 10cm aperture images the Earth and Sun on an array detector. The field of view is 1°x1°, and the detector is 4096x4096 pixels. The telescope is preceded by heat-rejection and narrow-band filters. The detector output is processed to yield the centroids of both the Earth and Sun images in all conditions of alignment.

- All the surfaces of the Maksutov design have a common center. This gives a simple system with a wide field of view and a short over-all length.

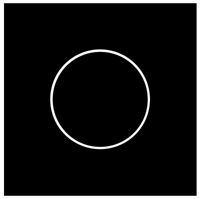


Image on Detector
in Aligned Condition

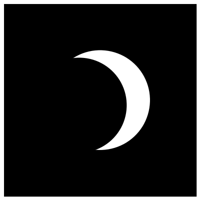


Image on Detector
in Unaligned Condition

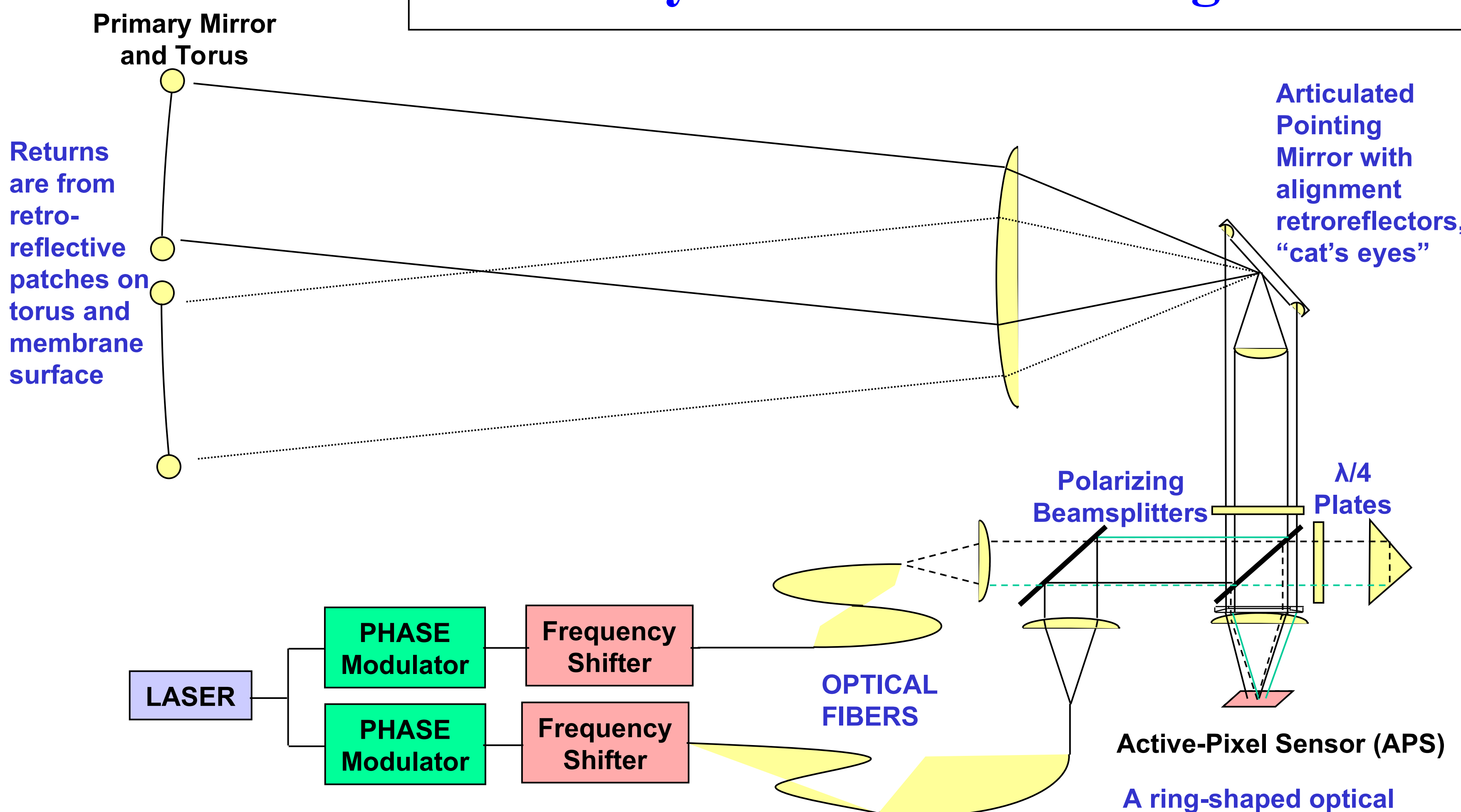
Primary Mirror Surface Figure Sensor

• The Mirror Surface Figure Sensor uses heterodyne modulation-sideband technology to provide unambiguous determination of range to retroreflective targets on the surface of the reflector and the torus. A single, frequency-stable laser provides the signal. The two isolated wavelengths necessary for heterodyne operation are generated by high-speed phase modulators followed by frequency shifters. The signal is detected by an active-pixel sensor that enables the returns from multiple targets to be processed simultaneously. The output of each pixel of the detector is at the heterodyne frequency. The phase of that output is the measure of target range. Phase modulation enables long range unambiguous distance measurement. Any remaining ambiguity is removed by comparison with the RF Metrology measurements.

• The angular position of the targets is determined from the position of the the returns on the active-pixel sensor and the angular position of the steering mirror. The latter is determined by the same interferometer that measures the range to the points on the primary mirror. The steering mirror is fitted with retroreflectors set into the mirror surface. The preferred form for these is the so called “cat’s eye” because it can be made to operate at large angles of incidence. The range to the cat’s eyes measured with interferometric precision determines the angle of the steering mirror.

• The optics require some special features. The cat’s eye retroreflectors are miniature Maksutov telescopes with the focus on the second surface of the corrector plate. This surface is made partially reflecting so that the rays retrace their paths with high precision without lateral displacement. This allows the use of a special optical element between the interferometer and active-pixel sensor. This is ring-shaped prism that deflects the rays from the cat’s eyes so that they fall on the detector in regions away from those occupied by the returns from the targets on the reflector and torus.

Primary Mirror Surface Figure Sensor



A ring-shaped optical element separates the returns from the pointing mirror from those from the primary mirror and torus.

Two-Frequency, Absolute-Distance Interferometer

Ranges to targets on the membrane combined with their angular positions determine the surface figure of the membrane. The angular position is calculated from the position of the image on the APS and the angular position of the mirror. The latter is calculated from interferometric range measurements to targets on the mirror.

Center-of-Curvature Sensor

• The Center-of-Curvature Sensor is used in the initialization of the Primary Mirror and for subsequent checks of the figure of the reflector. It operates only when the Science Spacecraft is at the center of curvature of the mirror. Light from an optical fiber floods the surface of the primary mirror from a point adjacent to the center of curvature. A perfect mirror will return the light to a focus symmetrically located with respect to the center of curvature and will produce two uniformly illuminated out-of-focus spots on the two detectors. Deviations form the correct location or the ideal surface figure are determined by photometric analysis of the out-of-focus images.

• The initialization sequence is the following: The Science spacecraft moves to the center of curvature of the primary mirror, a point 250 m from the mirror. The Formation RF Metrology Sensor is used for the initial positioning. The Surface-Figure-Metrology Sensor is then activated. It uses the return from the retroreflective targets in the inner and outer torus to refine the position of the Science Spacecraft.

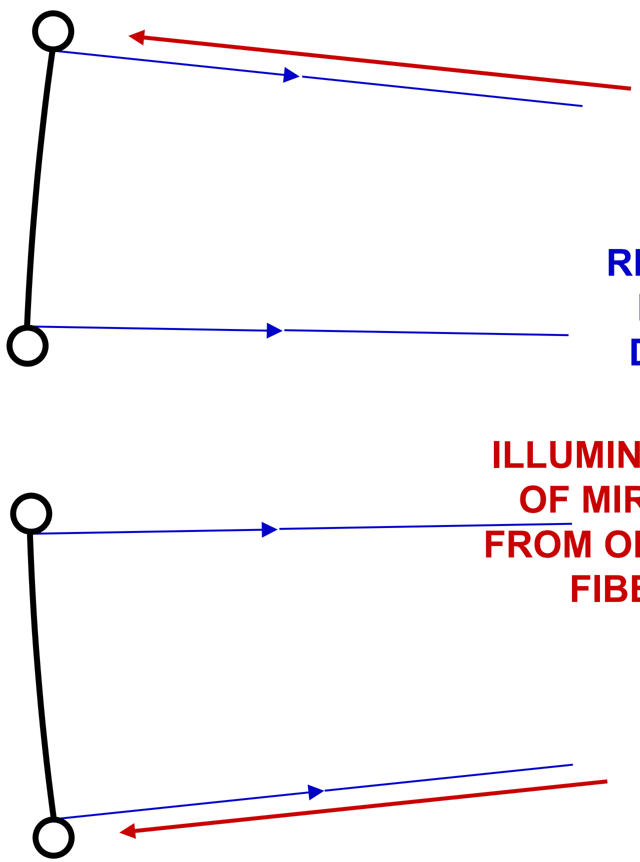
• The Center-of-Curvature Sensor is activated. Its output is first used to further refine the position of the Science Spacecraft with respect to the Aperture Spacecraft. When the two spacecraft have been stabilized at the correct relative positions, The output of the sensor is used to evaluate the surface figure of the mirror.

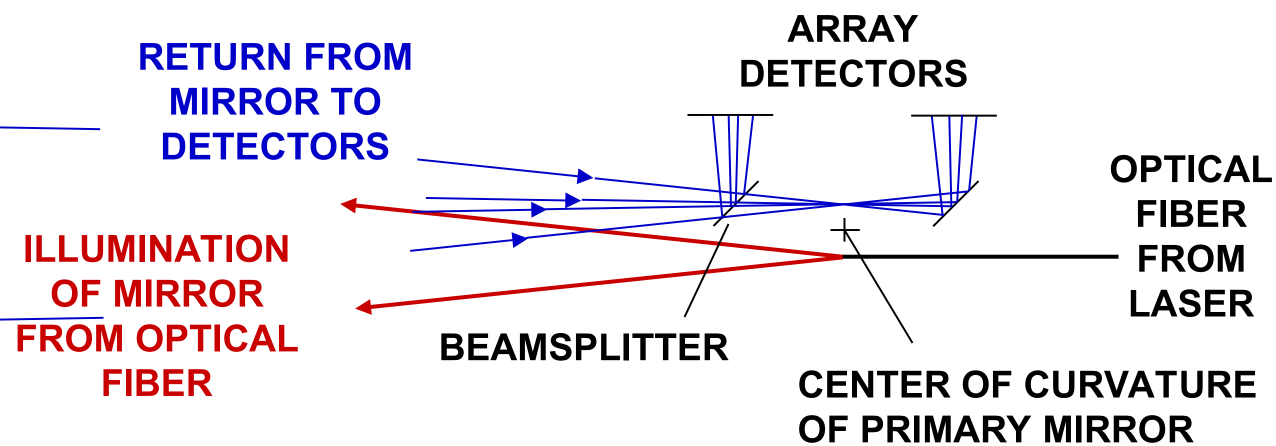
• Surface-Figure-Metrology Sensor is then used to measure the surface figure of the mirror. Since this sensor is a scanning system, some time is required for the measurement. During this time the Center-of-Curvature Sensor monitors the surface figure for change. A cross-calibration of the two sensors is then used to refine the algorithms that produce the output to the control system from the Surface-Figure-Metrology Sensor. That sensor now takes over the control of the the mirror surface, and the Science Spacecraft then moves to the focus of the primary mirror.

Center-of-Curvature Sensor

• Provides an initial global determination of the Primary Mirror surface figure

• Makes possible an independent check of the Primary Mirror figure metrology





Operating Principle

• Light from an optical fiber floods the surface of the primary mirror from a point adjacent to the center of curvature. A perfect mirror will return the light to a focus symmetrically located with respect to the center of curvature, and will produce two uniformly illuminated out-of-focus spots on the two detectors.

• Deviations form the correct location or the ideal surface figure are determined by a photometric analysis of the out-of-focus images.

Shape Estimation and Control of Primary Membrane Mirror

Membrane Mirror Control Strategy

- **Design a shape estimation and control technique for the large mirror in order to keep its *spherical* shape**

- **Ideally, light reflected from the surface of an a spherical mirror has only spherical wavefront aberrations, which will be corrected by the *Schmidt Corrector mirror* in the Science spacecraft Telescope**

- **An extension to current Schmidt Corrector mirror is to consider *adaptive optics*, that uses the Primary mirror Shape Metrology and Center of Curvature Sensor data to also estimate any residual wavefront aberrations in the incoming light from the Primary. Active figure control of a small adaptive corrector mirror would further reduce these residual aberrations**

Primary Mirror Shape Control Options

- ***Passive nominal global shape*** is obtained by using a shape memory alloy (SMA) membrane, such as a Nitinol film, with optical quality reflective surface layer deposition.
- Some options for ***active fine shape control***

  1. Bimorph piezoelectric (PVDF or ceramic film) actuation with electrode patches bonded to the back surface of the SMA.

  2. MEMS inchworm actuator patches bonded to the back surface of the SMA (with or without piezo Bimorph film).
- Laser metrology is used to ***measure the mirror surface*** for shape control .
- PVDF connector strips with MEMS inchworm actuators at the back surface, and large stroke piezo actuators at the torus connection, to ***establish a soft connection*** between the torus and the primary mirror, in order to dynamically isolate the mirror, and attenuate transmission of any undesirable disturbance forces from toroids and spacecraft to the ***"floating"*** mirror.

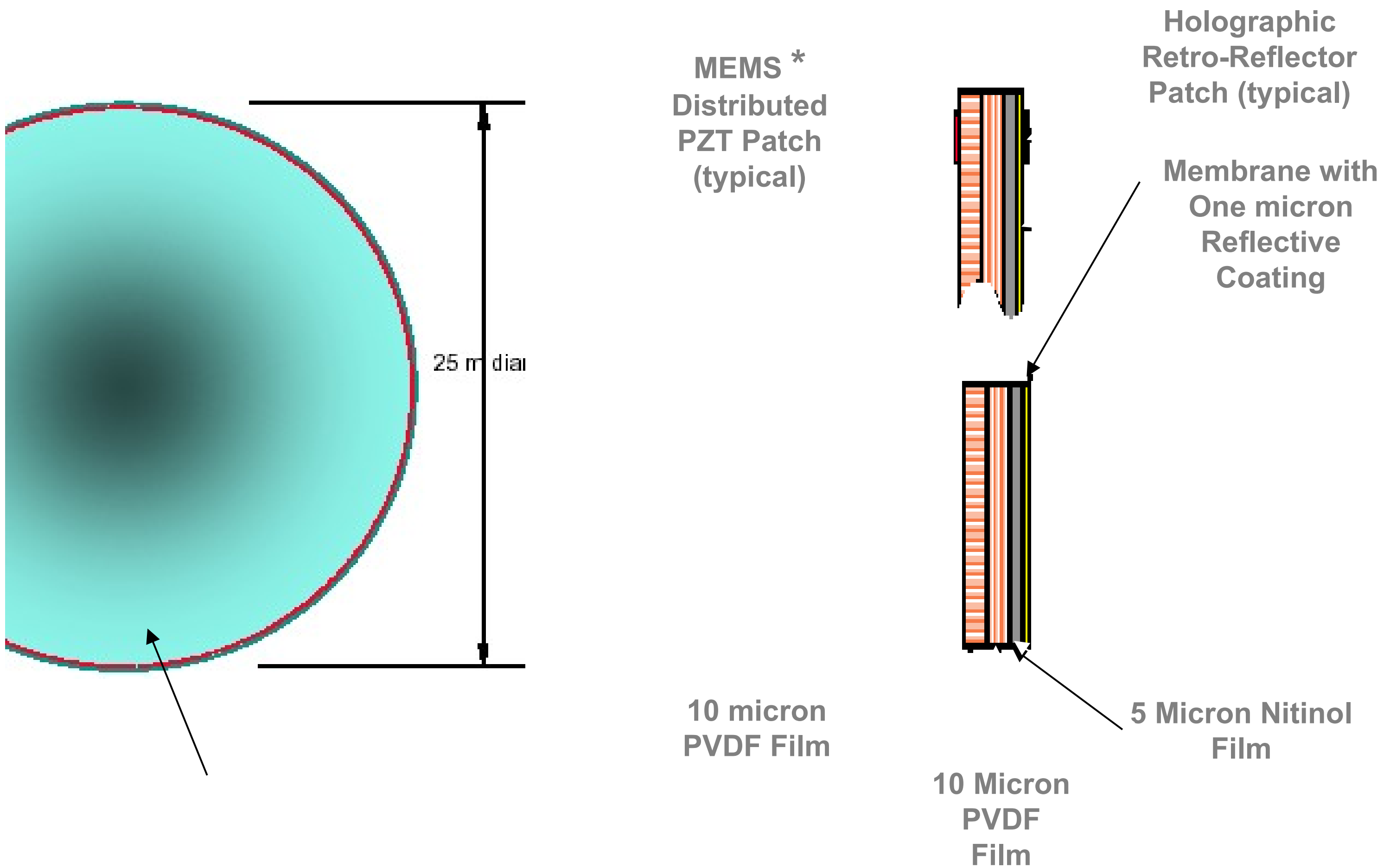
Key Material Properties

Material	PZT-5A	PVDF	Nitinol
Type	Piezo-ceramic: Lead Zirconate Titanate	Piezo-film: Polyvinylidene Fluoride	Shape Memory Alloy: NiTi
Poisson's ratio	0.31	0.35	0.33
Young's modulus (GPa)	62	1.2	35 (martensitic) 83 (austenitic)
CTE (ppm/deg C)	3.5	42	7 (martensitic) 11 (austenitic)
Density (g/cm3)	7.8	1.77	6.5

- PZT and Nitinol have comparable mechanical and thermal properties
- Manufacturing very thin films of PZT ceramic is in development
- PVDF has significantly different properties with respect to Nitinol, requiring thicker cross-section for equivalent stiffness
- PVDF manufacturing is current technology.

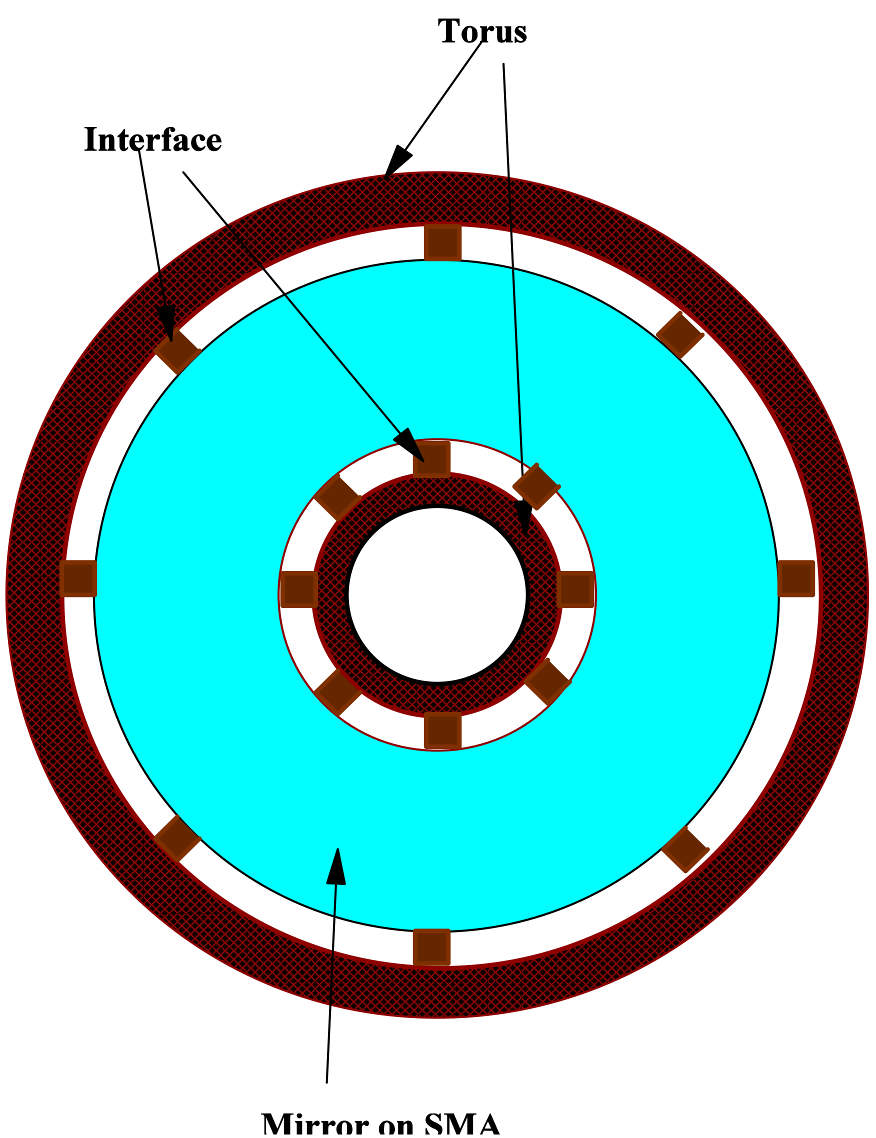
Large Membrane Mirror

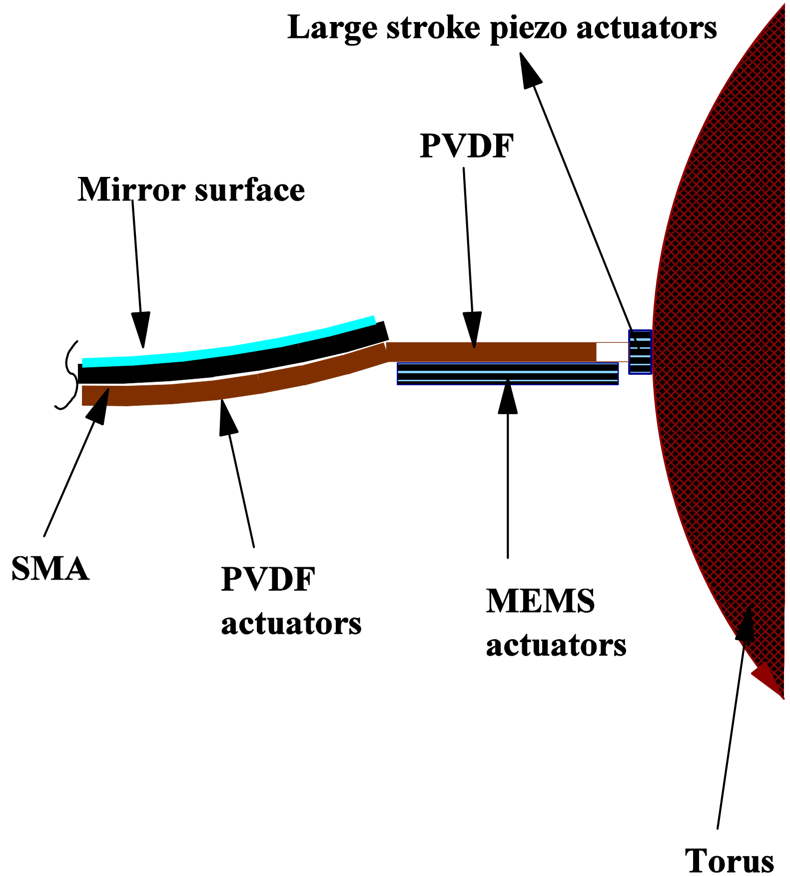
Active Figure Control Concept for Membrane Primary Mirror



* E-H Yang and S-S Lih, JPL , 2003 IEEE, 0-7803-7651-X/03

PRIMARY MIRROR CONTROL ELEMENTS



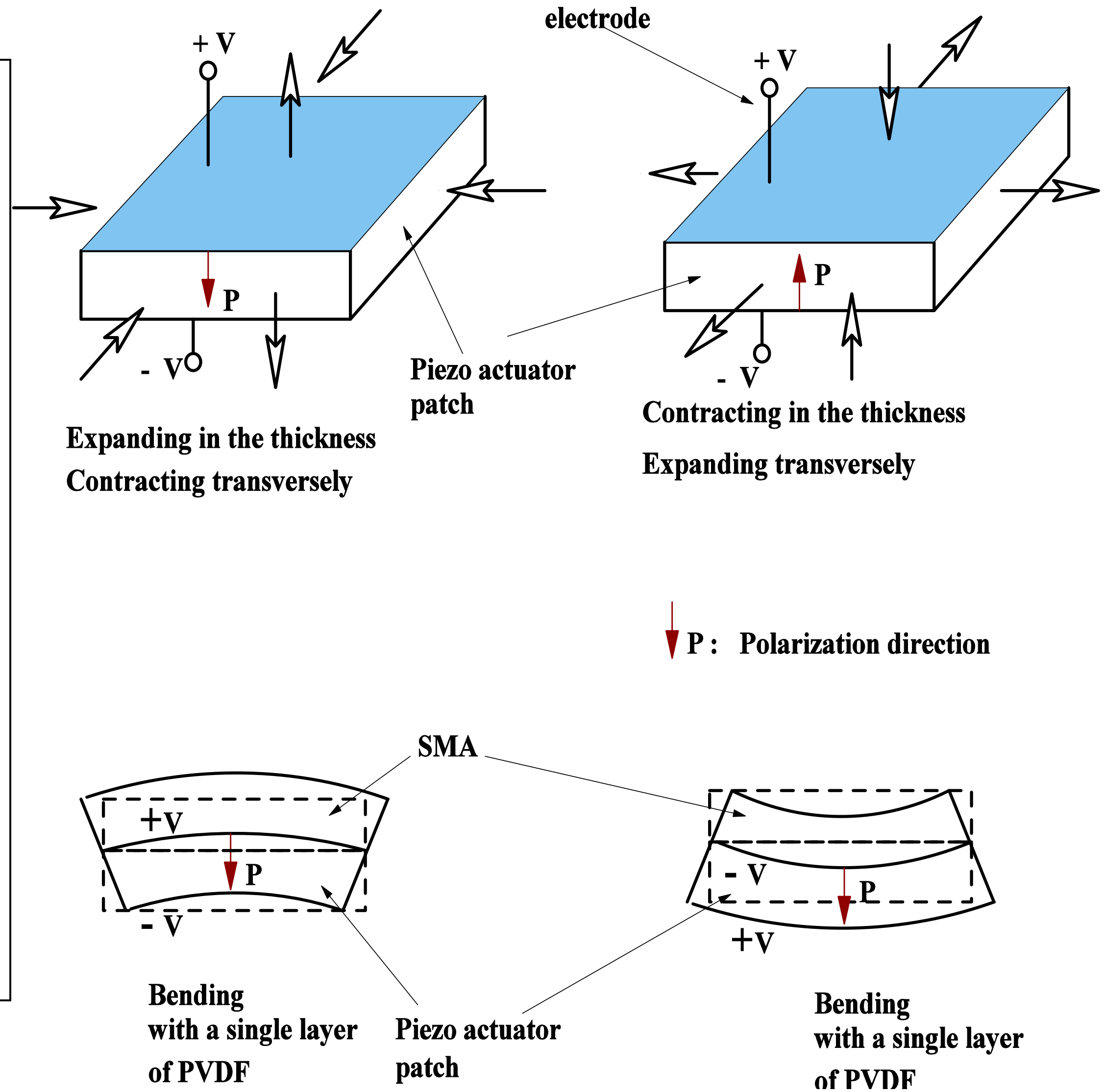


NOT to SCALE

Single Layer Piezo-Actuation Principle

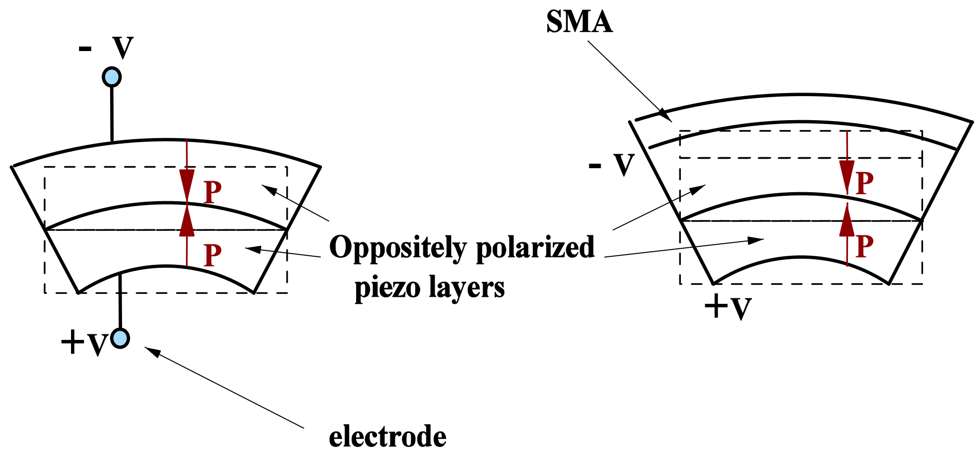
- Electrical field having the same polarity and orientation as the original polarization field is placed across the thickness of a single sheet of piezo layer, the piece expands in the thickness (along the axis of polarization), and contracts in the transverse direction

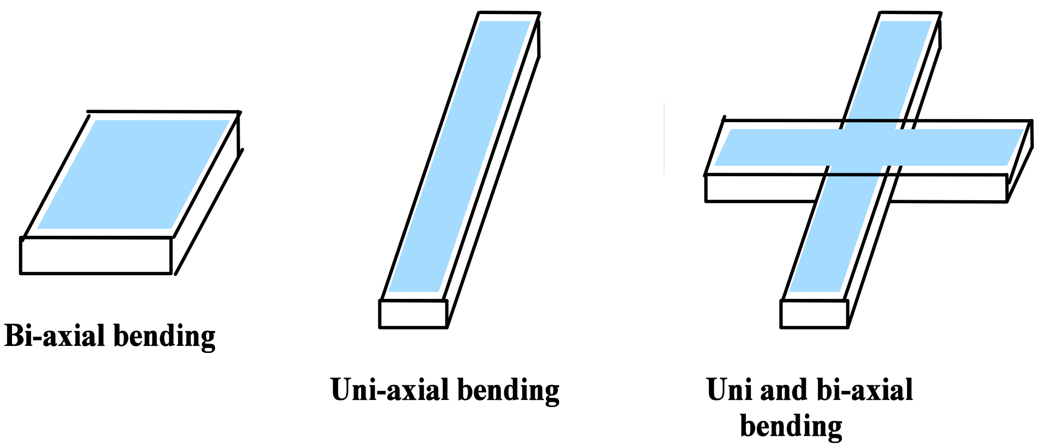
- By bonding the piezo-actuator patch onto SMA, and applying different voltages, we can obtain bending



Bimorph Piezo-Actuation Principle

- Bimorph elements can be made to elongate, bend, or twist depending on the polarization and wiring configuration of the layers. They produce curvature when one layer expands while the other layer contracts.
- Different patch geometries can be used to obtain uni-axial, and bi-axial bending.





Membrane Mirror Modal Estimation and Control

- **The Zernike polynomials can fully describe the mirror shape**
- **Obtain measurements at finite number of points on the mirror surface by using the laser metrology, with retro-reflectors on mirror surface**
- **Estimate the Zernike components (the polynomials contributing to the existing shape), and their amplitudes (Zernike coefficients) from measured data (analogous to classical "*Sinusoid estimation*")**
- **Model membrane and actuators in order to obtain influence functions, which establish a relation between the applied actuation voltages and Zernike coefficients describing the mirror shape**
- **Solve the inverse problem to compute the required voltages in order to obtain Zernike coefficients for the desired shape**
- **Apply voltages to the electrodes to excite the piezo material in order to apply desired deformations**

Zernike Polynomials

- Orthonormal polynomials on unit circle
- Can be used in order to describe the shape of the mirror

$$z(\rho,\theta) = \sum_{k=1}^{m} c_k Z_k(\rho,\theta)$$

where

Z_k is kth Zernike Polynomial

c_k is kth Zernike Coefficient

Membrane Modal Shape Estimation

- **Measurements give information at finite number of points**

$$z_j = \hat{z}_j + n_j, \quad j = 1, \quad \ldots, \quad M$$

where z is the measured variable,

$\hat{z}$ is the actual value, and n is measurement noise

- **We have to determine the statistical properties of the noise in measurements. Then, develop an estimation technique to estimate the Zernike coefficients of interest from data with the those properties, i.e.**

$$c_k = f_k(z_1, \ldots, z_M), \quad k = 1, \ldots, m, \qquad \text{where}$$

f_k is the function representing the estimation process

Membrane Modal Control

- **The control voltages are determined based on the difference between the desired Zernike coefficients, and estimated ones**

$$c_{k,c} = c_{k,d} - c_k, \quad k = 1, \ldots, m \qquad \text{where}$$

$c_{k,d}$ **is the Zernike coefficients for the desired shape**

c_k **is the Zernike coefficents for the estimated shape**

$c_{k,c}$ **is the correction that will be applied by control**

- **Influence functions are analytically and/or numerically determined, which correlates applied voltages to shape correction as**

$$c_{k,c} = g_k(V_1, \ldots, V_n), \quad k = 1, \ldots, m, \qquad \text{where}$$

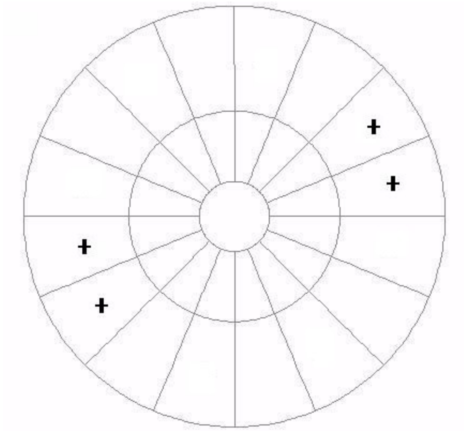
g_k **is kth influence function**

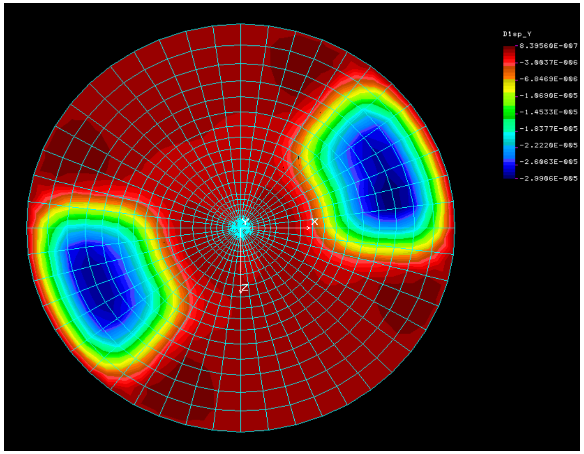
$V_j, \ j = 1, \ldots, n$ **are applied voltages on actuator patches**

- **Then, we can find necessary voltages which will give the optimal shape correction:**

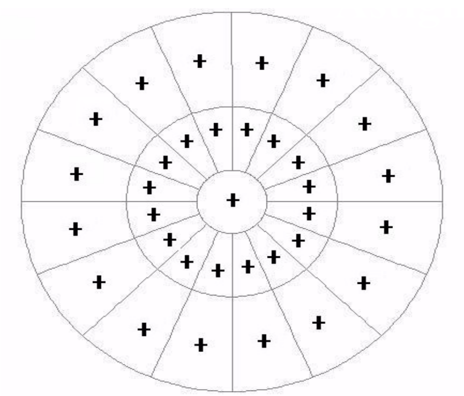
  For example, when influence functions are linear, the voltages can easily be computed by solving a least squares optimization problem.

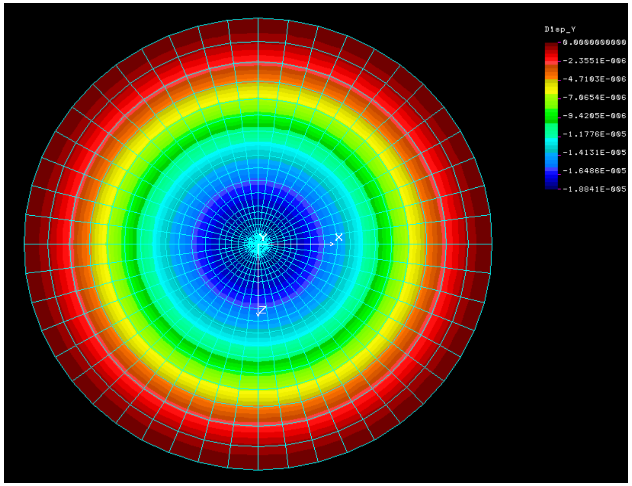
An Example of Aberration Numerical Influence Function Determination





Astigmatism





Focus

By M. S. Lih, JPL

Future GNC Research

1. Model and analyze advanced non-linear thruster performances, and develop higher order robust controller
2. Modeling, analysis, and simulation of metrology and optical sensor errors and dynamics
3. Optical analysis and control for Deployment Phase of the mission, including formation initialization and calibration
4. Detailed modeling of membrane mirror for Shape control
5. Determination of Piezo electrode distribution for sufficient control authority vs number of excitation nodes
6. Placement of retro-reflectors on mirror surface for laser metrology, and shape sensing, with minimal optical quality reduction

Enabling Technologies

- **Subsystems and science instruments emphasizing: miniaturization, 3-D electronics, integrated structures/cabling, MEMS sensors and actuators**
- **Structures**
  - Lightweight composite structural materials for Telescope and Spacecraft
- **Power**
  - Advanced Radioactive Isotope Power Sources with high specific power efficiency
- **GNC**
  - Combined High Accuracy Earth-Sun Sensor to maintain Orbit Tracking
  - RF and Laser-Optical Metrology Systems for Formation 3-D Knowledge
  - Precision Formation Flying and Orbit Navigation Methodologies
  - Sophisticated, higher-order station-keeping controllers that mitigate thruster limitations
  - Active Figure Control Concept for Membrane Primary Mirror
- **Propulsion and Control**
  - Advanced high specific-impulse linear Xe EP thrusters in large/small sizes

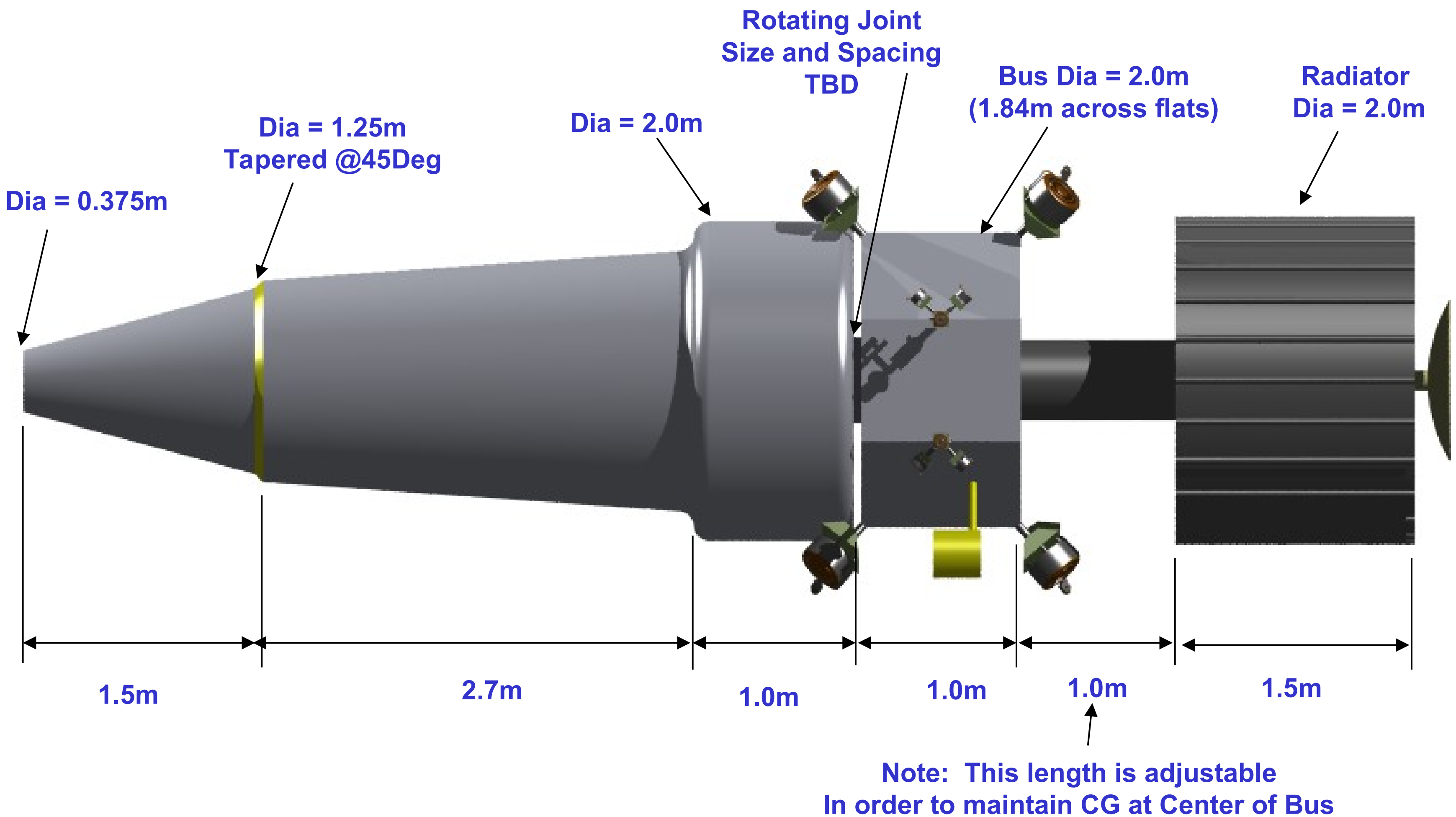
Looking Ahead:

The Focus for Continued System Development

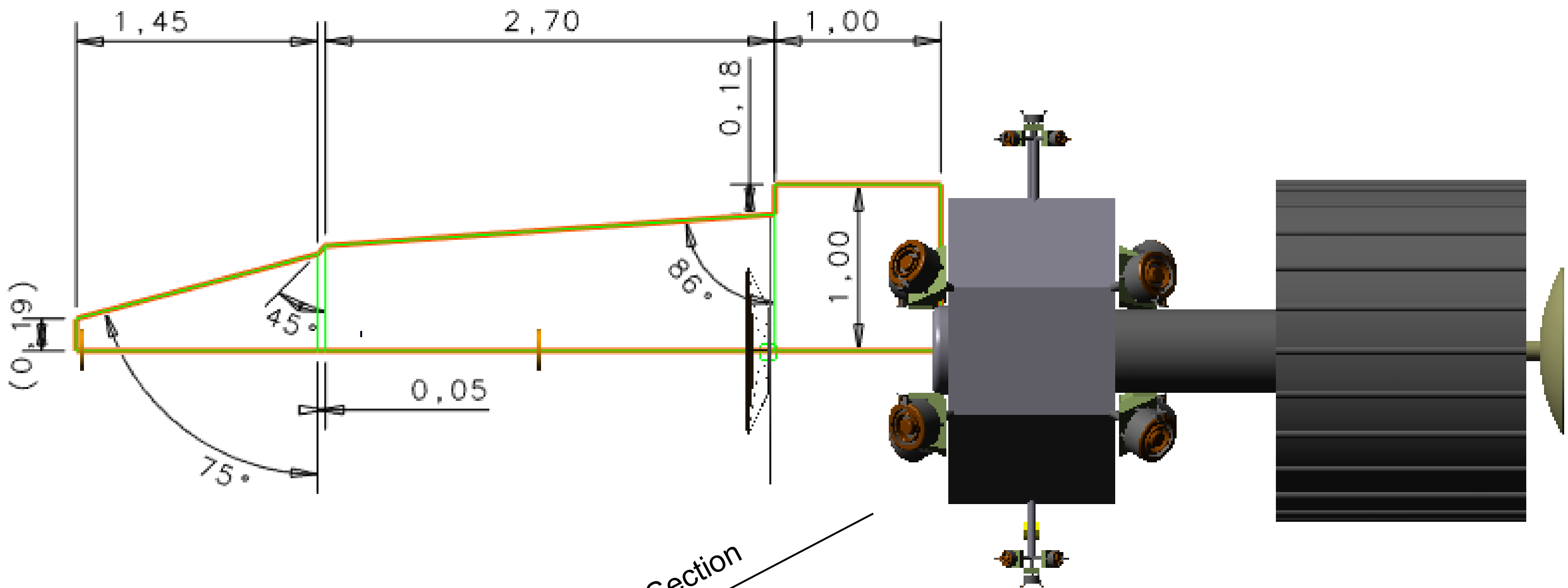
- Refinement of all subsystem concepts including science instruments emphasizing: miniaturization, 3-D electronics, integrated structures/cabling, MEMS sensors and actuators, high specific power and propulsion efficiencies, and lightweight composite structural materials consistent with 2025-2030 SOA projections.
- System Mass estimates based on above with the target for each S/C not to exceed 1200 kg wet for mission operations without Xe re-fueling of 10 years (goal) and 5 years minimum.
- Definition/refinement of Earth to L2 Deployment Phase: Options/trades for LV/Carrier, Direct vs E-M L1Gateway, L2 Insertion using Chemical, Low-Thrust NEP and SEP Hybrid propulsion.

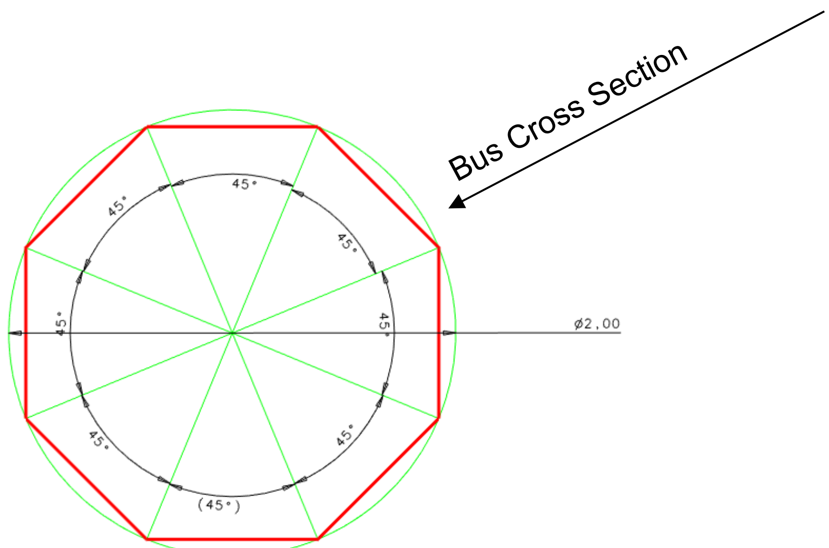
Appendix

L2 Science Telescope Dimensions



L2 Observatory Science S/C Telescope Dimensions





Wire frame showing detail Dimension of the science section of the spacecraft. All dimensions are in meters.

L2 Observatory Science S/C

Configuration Options/Trades

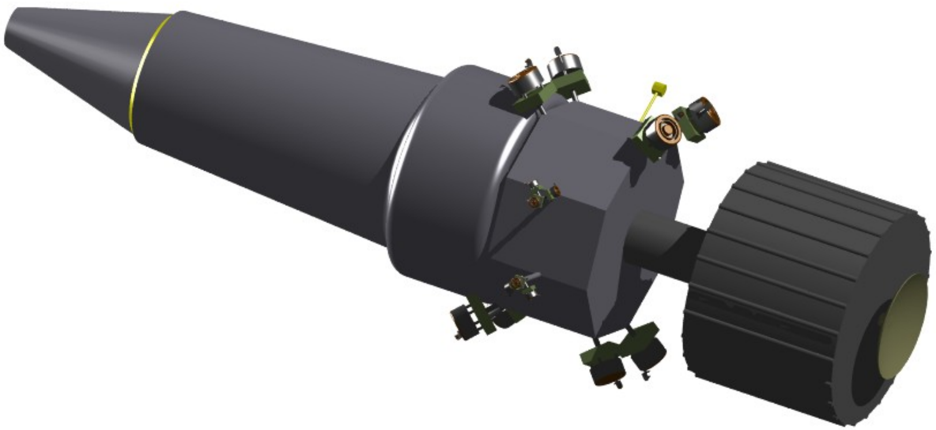


Figure 1 – Baseline Concept with SRG Integrated Power Module Placed AFT to balance the Spacecraft CG location

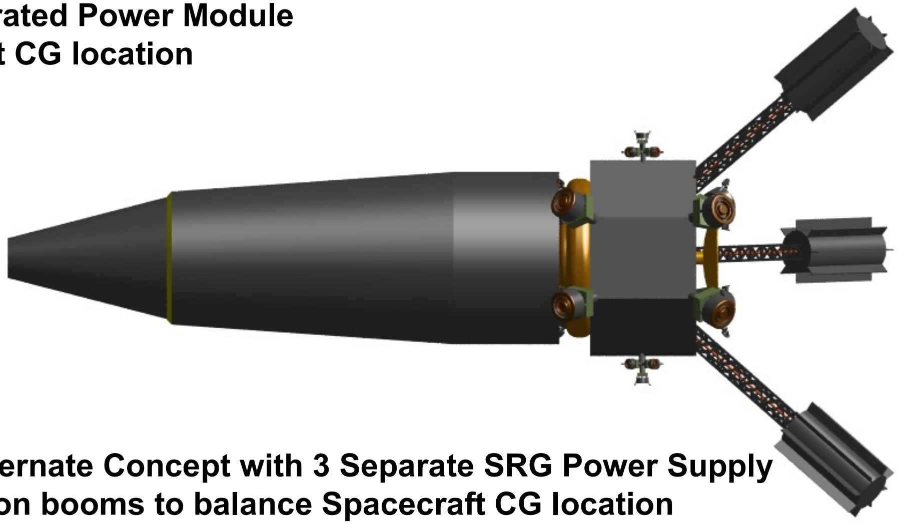


Figure 2 – Alternate Concept with 3 Separate SRG Power Supply Units on booms to balance Spacecraft CG location

Alternative Thruster Configuration