ME 261: Engineering Design
Fall Semester, 2009

Potential Projects for the International X-Ray Observatory
(Design Teams: 1. Please prioritize the four projects described below,
and 2. The referenced articles will be available for your perusal)
(Completion Date: 12/18/09)

Background

The International X-ray Observatory (IXO) is a new collaboration between NASA, ESA, and JAXA which is under study for launch in 2020. IXO will be a large 6600 kilogram Great Observatory-class mission which will build upon the legacies of the Chandra and XMM-Newton X-ray observatories. It combines elements from NASA's Constellation-X program and ESA's XEUS program. For more information reference:

D. Robinson and R. McClelland, "Mechanical overview of the International X-Ray Observatory" 2009 IEEE Aerospace Proceeding, March 6-13, 2009

The IXO Flight Mirror Assembly (FMA) IXO is currently being developed at NASA Goddard Space Flight Center (GSFC). The design addresses some unique engineering challenges presented by the unprecedented combination of high angular resolution and large effective area required to achieve the desired scientific objectives. To meet these requirements, the Soft X-Ray Telescope (SXT) optical design consists of about 14,000 0.4 mm thick glass mirror segments densely packed into a 3.4 m diameter FMA and supported with micron level accuracy and stability. The thin mirror segments are mounted into 60 intermediate wedge shaped structures called modules. Modules are kinematically mounted to the FMA primary structure which is optimized for minimum mass and obscuration of the clear aperture. Key engineering challenges addressed include ensuring positive stress margins for the glass segments with a high Factor of Safety, keeping the structure light enough to launch, providing a large effective area, preventing unacceptable thermal distortion, and assembling the many mirror segments into the FMA with the project schedule. For more information reference:

R. McClelland et al, "Preliminary Design of the International X-Ray Observatory Flight Mirror Assembly" Proceedings of SPIE Vol. 7437, 2009

Problem 1. Design of Kinematic Mount Mechanism between FMA Modules and Primary Structure

The 60 mirror modules will be attached to the FMA primary structure via three flexures near the mid-plane of the module, two on the sides and one in the front. In order to maximize the effective area of the FMA, the gap between the structure and modules must be ~2 mm. This does not leave room to bolt the module flexures to the primary structure during integration. The modules must be able to be installed in any order and removed and replaced at any time with minimal space to operate the mechanism connecting the two structures. The connection between the modules and primary structure must carry shear loads of ~1300N during launch in the plane of the module panels. The current concept consists of a deployable pin which can be extended from the flexure into a mating hole in the primary structure. The pin can be deployed and retracted by inserting a tool into the small gap between the module and primary structure. The design of this connection needs to be taken from the requirements to a solid feasible concept that is capable of being locked in place for launch.

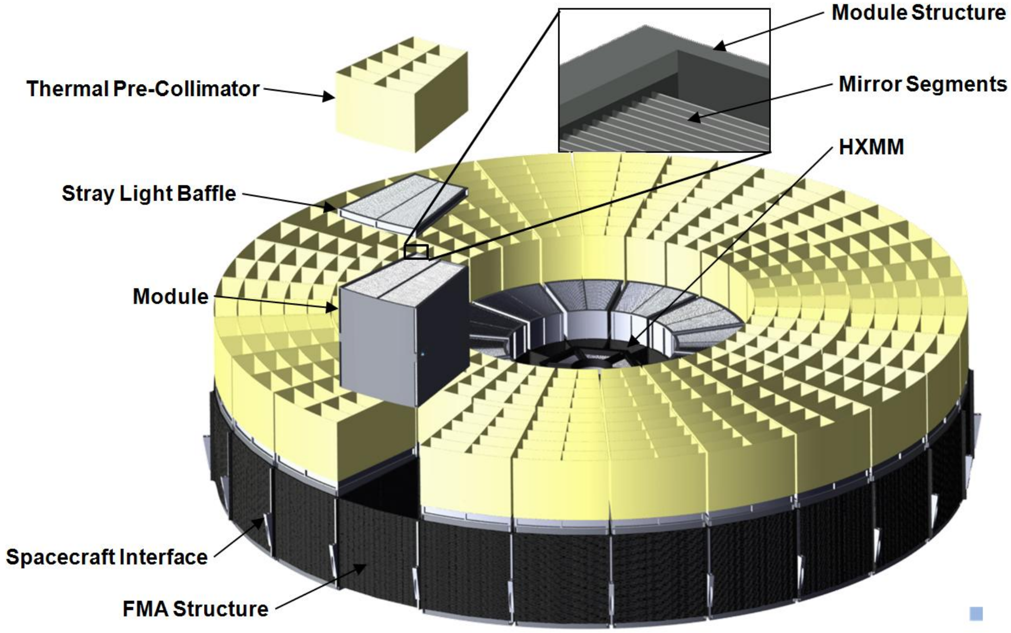

Figure 1. Exploded view of FMA and module illustrating the basic parts of the assembly. Each of the 60 SXT modules contains approximately 200-300 bonded-in mirror segments.

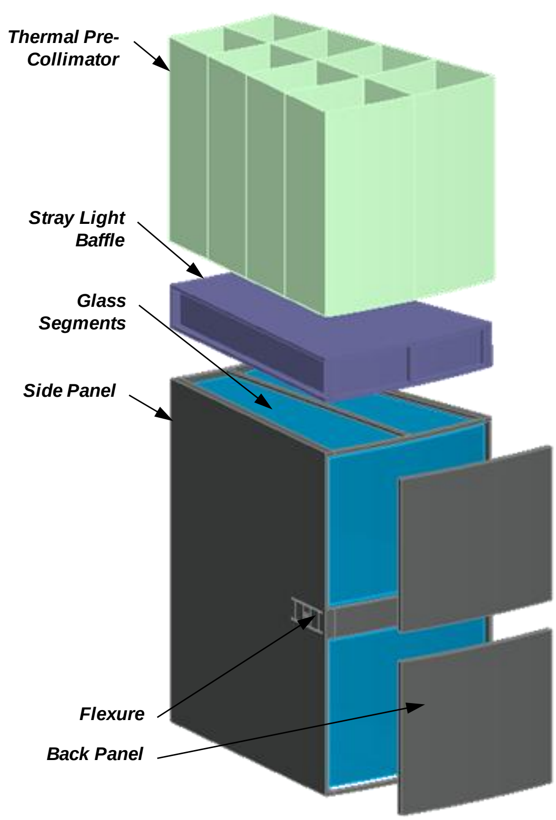

Figure 2. Exploded view of a middle ring module. The module structure consists of panels on four sides and radial rails to which the segments mount.

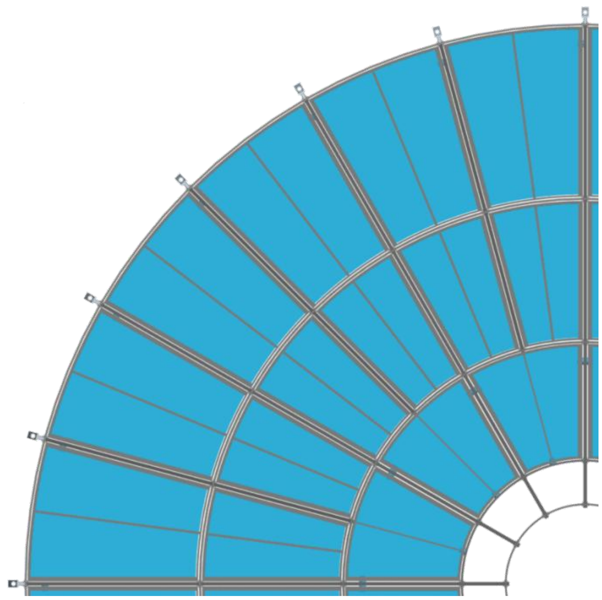

Figure 3. Quadrant of the FMA viewed for the perspective of incoming X-rays. The mirrors (blue areas) must be obscured as little as possible.

Problem 2. Design of Proof Testing Machine for Mirror Segments

Due to the unpredictable nature of glass failure, the 14,000 mirror segments must each be proof tested before installation into the module in order to guarantee they can be launched successfully. A proof test fixture must apply given stresses at 8 points on each segment. In order to perform this crucial task quickly, an automated proof test fixture that performs the test on 720 different mirror shapes is needed. The fixture must be able to adjust to the specified segment sizes and apply displacements near the mounting points in order to exercise each mirror at the design strength. The proof test must be performed in a dry air or nitrogen environment to prevent crack growth during testing. A small chamber which can be quickly filled with gas should surround the fixture and mirror. Each segment is 200mm long and 0.4 mm thick with width ranging from 150 mm to 400 mm and radius of curvature ranging from 370 mm to 1600 mm.

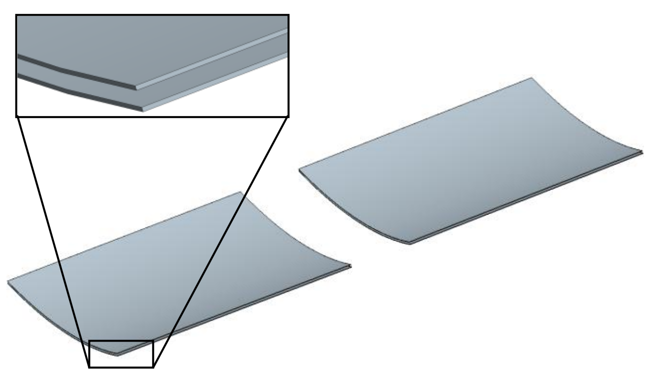

Figure 4. Two pairs of primary and secondary mirror segments spaced 1.5 mm apart. The 0.4 mm thick segments must be closely packed to achieve the required effective area.

Problem 3. Design of Retractable FMA Rear Contamination Cover

During launch, the FMA must be covered on both axial ends in order to prevent contamination (mostly particles loosened due to launch vibrations). The cover on the front end will be jettisoned into space after launch. The internal cover presents more of a challenge because it must be removed inside of the fixed metering structure without touching the FMA or getting in the way of the X-Ray beam. The retractable cover must be highly reliable and may need redundant systems due to its critical role.

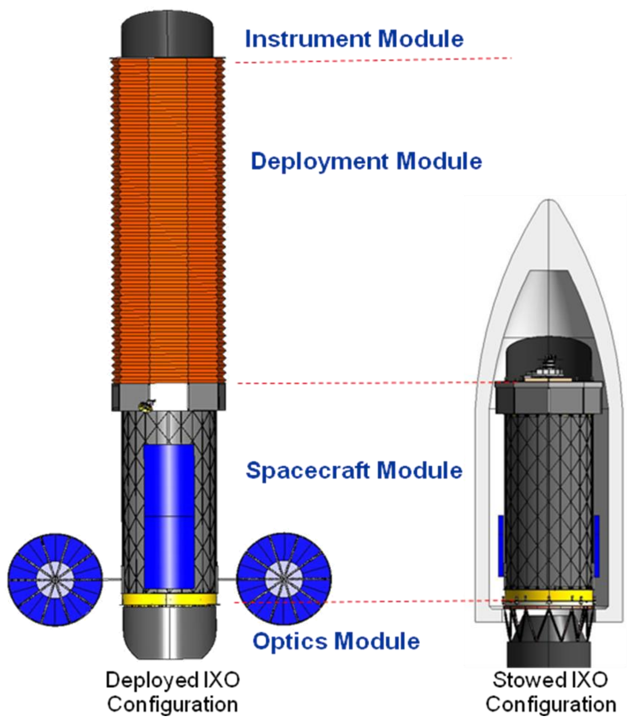

Figure 5. IXO Spacecraft shown in deployed and stowed configurations. The observatory is divided into four modules to simplify integration and testing. The FMA is contained within the Optics Module. IXO will have a mass of around 6600 kg, a length of approximately 23 m when deployed, and a 4 m diameter.

Problem 4. Design of Automated Mirror Segment Bonding Assembly

Each of the 60 mirror modules contains 200-300 mirror segments that must be glued into the module at 8 points. In order to minimize distortion and maximize assembly speed, a fixture is needed that mounts and applies injection pressure to 8 syringes simultaneously. UV light guides must then shine on the bond between the mirrors and structure in order to quickly cure the UV sensitive adhesives used. The fixture must automatically adjust to each of the 720 different mirror shapes in order to use a common design for the whole FMA. Each segment is 200mm long and 0.4 mm thick with width ranging from 150 mm to 400 mm and radius of curvature ranging from 370 mm to 1600 mm.

Deliverables

- Rationale for concept chosen
- Supporting engineering calculations and analysis
- CAD models of design
- Rationale for material and component selection
- Physical model of design (SLA, metal, wood, etc)

Mechanical Overview of the International X-Ray Observatory

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Abstract—The International X-ray Observatory (IXO) is a new collaboration between NASA, ESA, and JAXA which is under study for launch in 2020. IXO will be a large 6600 kilogram Great Observatory-class mission which will build upon the legacies of the Chandra and XMM-Newton X-ray observatories. It combines elements from NASA's Constellation-X program and ESA's XEUS program. The observatory will have a 20-25 meter focal length, which necessitates the use of a deployable instrument module. Currently the project is actively trading configurations and layouts of the various instruments and spacecraft components. This paper will provide a snapshot of the latest observatory configuration under consideration and summarize the observatory from the mechanical engineering perspective.

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1. INTRODUCTION

The International X-ray Observatory (IXO) is a new collaboration between NASA, ESA, and JAXA which is planned to launch in 2021. It combines elements from NASA's prior Constellation-X program[1] and ESA's XEUS program. IXO will be a large Great-Observatory class mission which will build upon the legacies of the Chandra and XMM-Newton X-ray observatories. Two major improvements in IXO over Chandra and XMM Newton are the high effective area for X-ray photon collecting in the 1 keV to 6 keV range, and a high spectral resolving power achieved with a micro-calorimeter X-ray detector cooled to 50 milliKelvin. The exciting science enabled by IXO includes exploration of black holes, growth and evolution of the largest structures in the universe and cosmic feedback. Details can be found in many other papers and will not be discussed here.[1]

IXO will have a mass in the neighborhood of 6600 kg and will be approximately 10 meters long and 4 meters in diameter in its launch configuration. It will fly on an Atlas 5 or an Ariane V rocket into an L2 halo orbit. The L2 Lagrange point orbit is approximately 1,500,000 km from Earth on the side opposite the Sun where the gravity of the Earth and Sun are balanced. The main advantages to this orbit are its very stable thermal environment, and the minimal shadowing from the Earth or Moon.

To collect as many photons as possible, IXO will have a 3.2m diameter Flight Mirror Assembly (FMA). Instead of the single large normal incidence primary mirror one finds in optical telescopes, the FMA will consist of nested concentric rings of mirrors which are nearly edge-on to the incoming X-ray photons. X-ray photons glance off the mirror surface at a very shallow angle and are focused 20 meters downstream at the instrument module end of the spacecraft. In the current concept, the FMA has approximately 360 concentric shells of glass, and 14,000 individual pieces of curved glass segments each 0.40 mm thick. Another X-ray focusing concept under development by ESA consists of "micropore" optics. Both concepts present major technological challenges to the IXO mechanical team, particularly in maintaining alignment and ensuring survivability during launch.

A major science driver for the observatory is a long focal length since that enables more photon collecting capability at the higher energy ranges. A focal length of 20 meters was selected for IXO as a reasonable balance between science needs and engineering constraints. No rocket fairing is large enough to fit a 20 meter long observatory, so IXO will have a deployable metering structure between the spacecraft bus and the instrument module.

1———————————

[2] IEEEAC paper #1580, Version 1, Updated 2009:01:07.

2. IXO CONFIGURATION

When deployed during the cruise to the L2 orbit, IXO will be 23m tall. It is divided into five major assemblies or modules: the Instrument Module (IM), the Deployable Metering Structure, the Spacecraft Bus, the Fixed Metering Structure, and the Optics Module. Each module may be integrated and tested separately at different sites nd then brought together for final integration.

Assembly. Other mechanisms include the rotational stages for the instruments, launch locks, and a mission unique launch vehicle payload adapter and separation system.

Another challenge in the design is to accommodate the Flight Mirror Assembly which focuses X-rays. The Spectroscopy X-ray Telescope (SXT) Flight Mirror Assembly (FMA) presents unique design challenges due to its 3.2m diameter and support of thousands of 0.4mm thick glass segments arranged in tightly packed concentric rings.

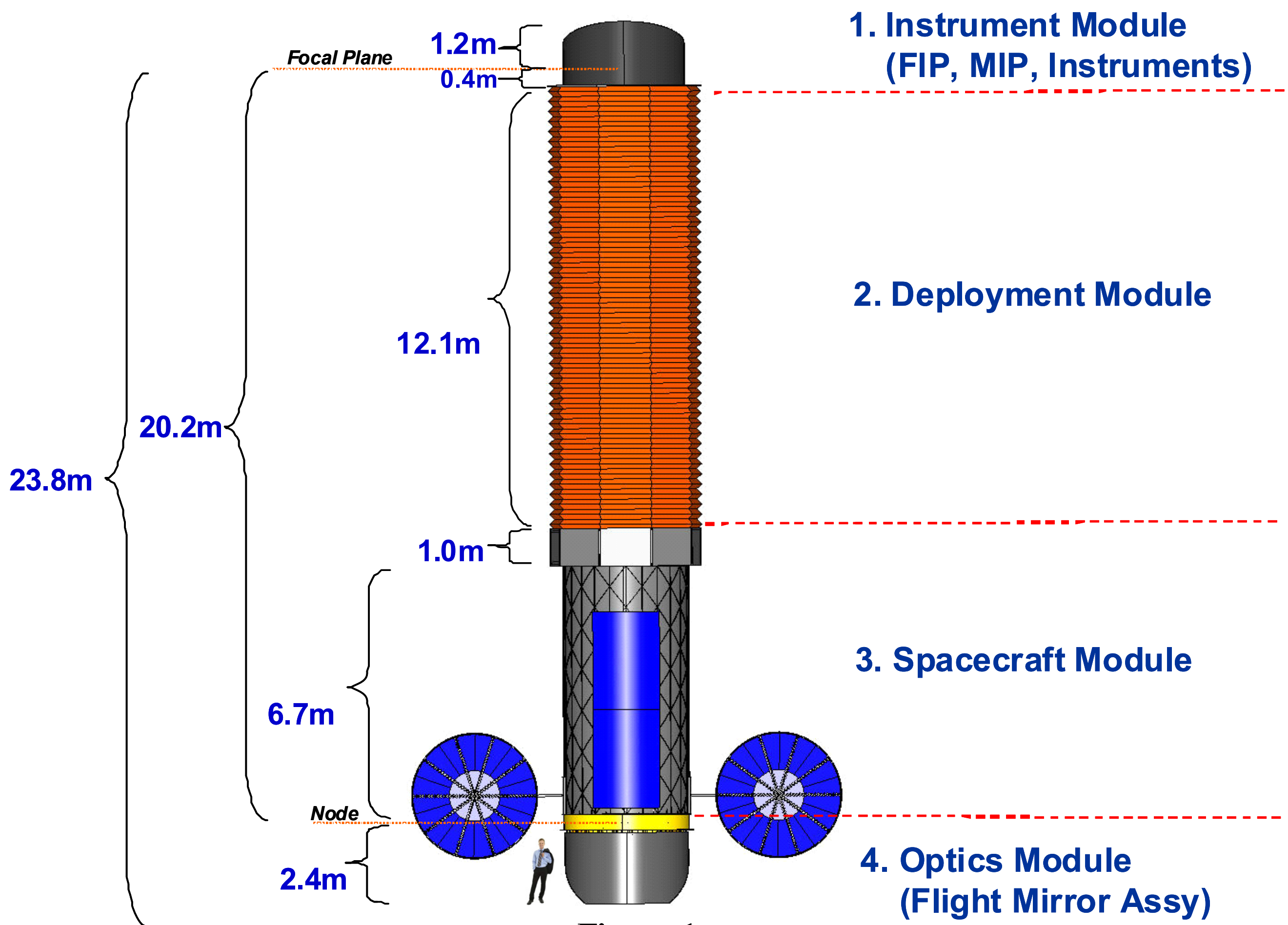

Figure 1

The primary structure of the observatory will be made from carbon fiber reinforced plastic (CFRP) composite materials. The 6.7m long metering structure is planned to be a one-piece composite cylindrical tube with isogrid stiffener ribs. It will be fabricated using advanced fiber placement technology recently developed for wide body airliners and other large aerospace structures.

One of the most interesting mechanical systems of IXO is the triad of deployable 12.1 meter long masts which will extend the instrument module to the focal point. The masts are similar to those used on the international space station solar arrays, and the upcoming Nustar X-ray mission. One challenge for the masts will be pulling up a 4 meter diameter light-tight shroud made of multi-layer insulation between the instrument module and the Flight Mirror

The glass segments are assembled into wedge shaped modules for integration and testing before the modules are kinematically mounted into a large stiff truss structure to form the FMA.

When stowed for launch, IXO will be about 10m long and 4 meters in diameter and fit into a medium Atlas 5 or Ariane V fairing.

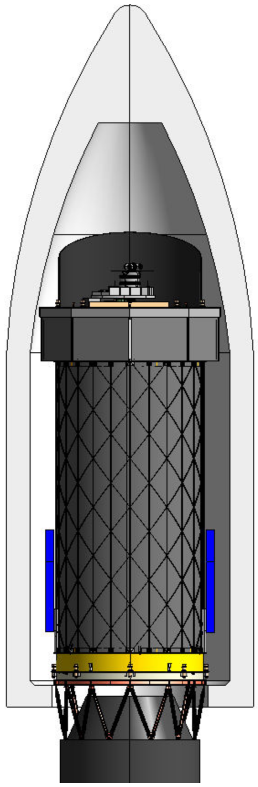

Figure 2 – IXO Stowed Configuration in Fairing

The following sections detail the observatory starting with the instruments and working down to the launch vehicle.

3. Instrument Module (IM)

The IM contains the instrument suite consisting of the X-ray Magnetic Spectrometer (XMS), Wide Field Imager (WFI), High Energy X-ray Imager (HXI), X-ray Gratings Spectrometer (XGS) CCD Camera, High Time Resolution Specrometer (HTRS), and the X-ray Polarimeter (XPOL). The fully integrated instrument module is estimated to weigh 900 kg. Because the X-ray beam cannot be folded as in typical visible light telescopes, five of the instruments must be rotated into the focus in turn. The XGS views x-rays which have been diffracted from the main beam by gratings to a CCD Camera which is off to the side of the main focal plane, and thus it operates continuously. The HXT instrument sits piggyback on top of the WFI and views high energy x-rays which have passed through the WFI's focal plane.

The instruments which must be at the focal plane of the main beam are placed on a Moveable Instrument Platform (MIP) which consists of a honeycomb panel with a redundant motorized rotation stage. The MIP rotates through about 140 degrees and has four stops.

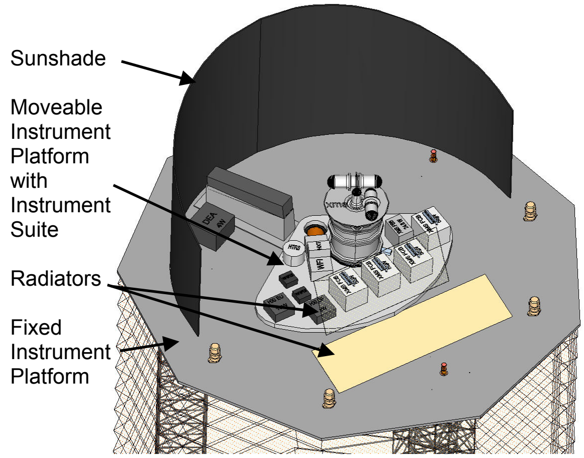

Figure 3 – Instrument Module Overview

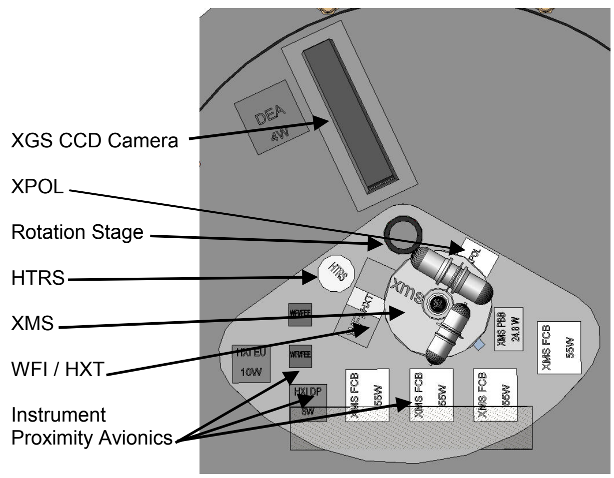

Figure 4 – Top View of the Instrument Suite

The instruments on the MIP will have their own fine focusing stage. This will allow them to compensate for any changes in the focus between ground-based testing and the space environment due to moisture desorption from the composites and thermal expansion. These effects may not be fully predictable prior to launch.

The XGS CCD Camera is planned to have a lateral stage which can provide fine positioning capability so that it can be centered on the 780mm long X-ray diffraction beam coming from the gratings. The gratings may be located on the aft end of the FMA or in the spacecraft bus where they project into the X-ray beam.

On the MIP, the four instruments each have their own electronics boxes which need to be in close proximity. These boxes process the analog signals from the instruments. Once the analog signals have been converted to digital, the data is sent to additional boxes on the bottom side of the Fixed Instrument Platform (FIP), and from there down to data recorders in the spacecraft bus.

One challenge in the layout of the IM is accommodating the thermal hardware. A 1.4m^2 louvered radiator will be installed onto the MIP to reject the heat of the local boxes via constant conductance heat pipes. The instruments may have their own radiators looking out to deep space. The remaining instrument electronics boxes have been located on the underside of the FIP to keep them within the large shroud around the observatory metering structure. The environment inside the shroud is more quiescent. The FIP is a honeycomb panel with embedded heat pipes to transfer heat from the boxes to a radiator which has a view to deep space.

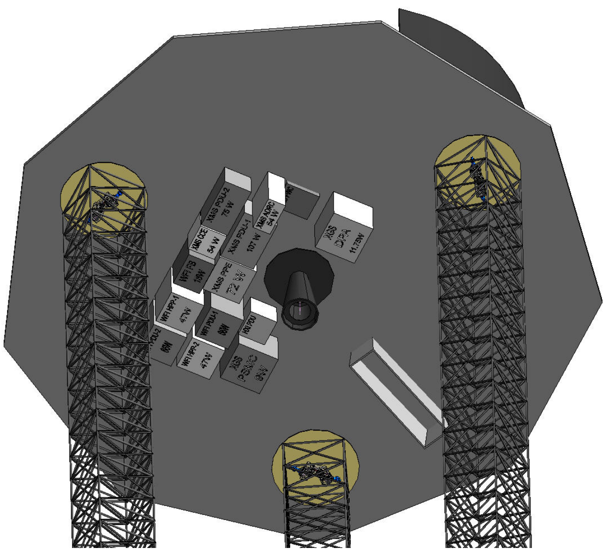

Figure 5 – View of avionics underneath the FIP

A fixed sunshield mounted to the FIP will keep the instruments in shadow at all times to maintain a very stable thermal environment. Further stabilizing the thermal environment is the requirement that the observatory will stay within +/- 10 degrees roll and +/- 20 degrees pitch of the incoming sunline.

4. Deployable Metering Structure

The job of the deployable metering structure is to push the stowed instrument module 12.1 meters away from the spacecraft bus to the X-ray focal point. The deployable metering structure consists of the deployment system, a light-tight shroud, baffles, and wire harness between the instruments and the bus. The goal for the deployment system is to place and maintain the instrument module in the focus within +/- 1 mm.

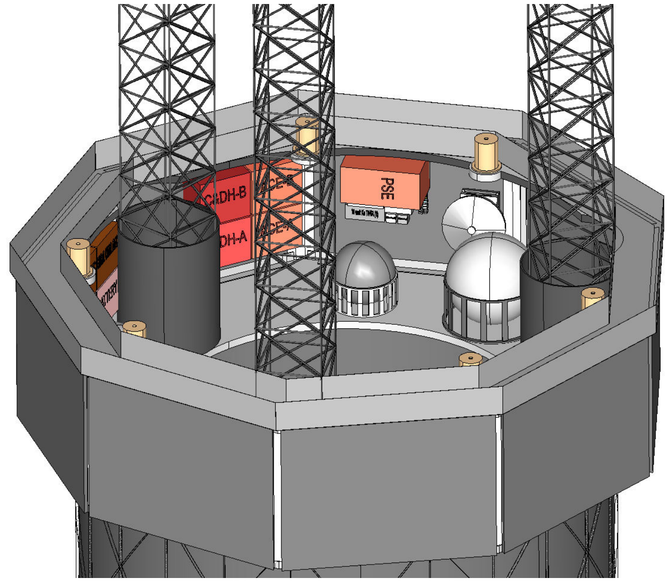

Figure 6 – 3 masts emerging from their stowage canisters

The deployment system is envisioned to be three masts which deploy from canisters located in the spacecraft bus. Three masts are required to maintain the stability of the IM and keep the fundamental resonant frequencies of the deployed spacecraft above 1 Hz. Preliminary analysis showed that the 1st bending mode is above 1.4 Hz and the 1st torsion mode is above 4 Hz. Another analysis showed that jitter from the reaction wheels will not excite these modes and that the displacement of the focal point will be quite small and well within requirements.

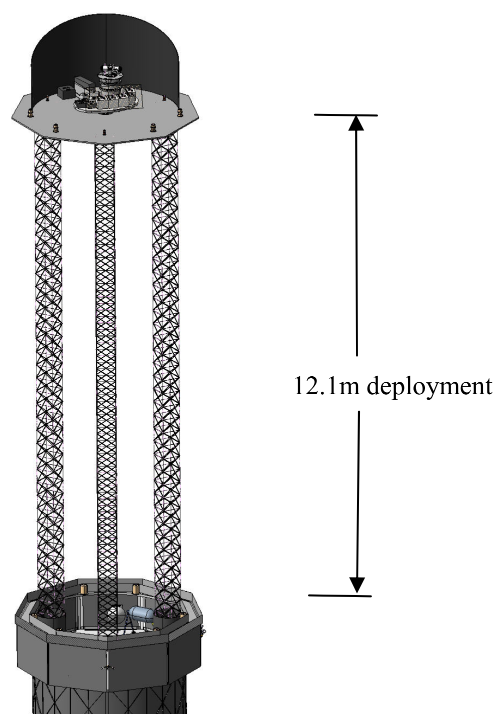

Figure 7 – The 3-mast system deploys 12.1m

Masts of this type are used to deploy the International Space Station solar array wings a distance of 35m. Another example flew on STS-99 in 2000 and deployed a radar topography instrument 60 meters from the shuttle while maintaining extremely tight stability.

The masts fold up inside the canisters for storage and deploy from the canister in a very precise and repeatable manner. As the mast deploys from the canister it forms a repeating series of cubic bays framed by vertical members (longerons), horizontal members (battens), and diagonal crossbraces. The longerons and battens are made of graphite rods and the crossbraces are stainless steel cables. The mass of each mast and canister assembly is estimated at 60 kg. Encoded motors in the canisters ensure that the three masts extend synchronously to deploy the IM.

The wire harness between the IM and the bus is expected to be at least several dozen twisted shielded pairs of wires. The wires will be braided through the battens and longerons in the canister when stowed and will deploy along with the masts. This technique has been successfully employed on several deployable masts in orbit.

Shroud

A shroud is required between the bottom of the IM and the top of the spacecraft bus and around the three masts. This shroud must block stray light from entering the instruments and flooding their detectors. The shroud will consist of multi-layer insulation blankets which are pleated like an accordion or camera bellows. The pleats allow the shroud to be folded up into a channel located on top of the spacecraft bus for stowage prior to deployment.

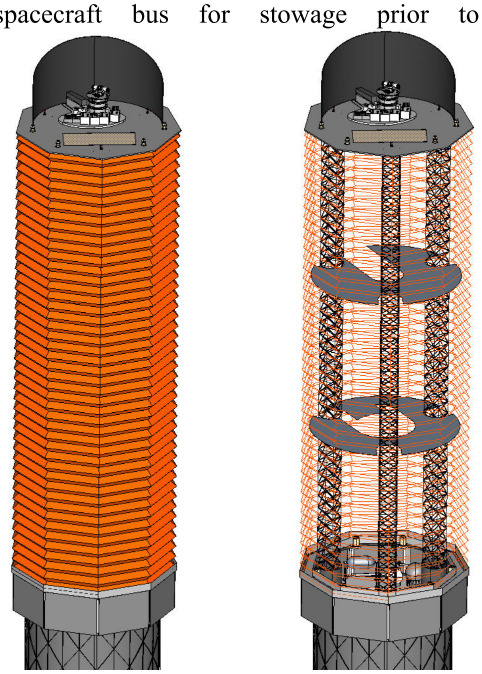

Figure 8 – IXO's shroud and baffles

To reduce the number of micrometeorite penetrations causing light leaks, the shroud consists of two concentric blankets separated by 10 cm to form a "Whipple shield." Each blanket will be constructed of 5-10 layers of ¼-mil aluminized Mylar and Dacron scrim cloth with thicker inner and outer layers. The mass of the shroud depends on the number and thickness of layers required but is expected to the about 150 kg. The blankets will maintain a steady-state thermal environment inside the enclosed volume which will minimize thermal expansion or contraction from the masts to help meet the observatory's stability requirements.

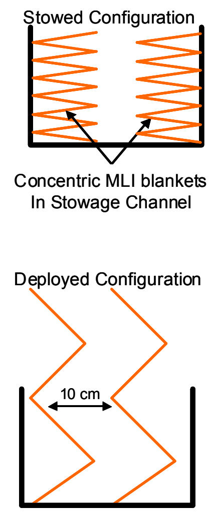

Figure 9 – Concentric accordian shrouds in stowage channel

A small-scale prototype of this accordion-shaped blanket was fabricated at NASA Goddard which demonstrated that IXO's 12.1m shroud can be folded and stowed into a stack only about 20 cm tall.

Figure 10 – Scale model of accordion-style shroud

Baffles

As seen in Figure 8, several baffles are also required inside the shroud to further block stray X-ray photons. These baffles are thin sheets of Tantalum foil on a film backer. They match the diameter of the shroud (~3.9m diameter) with cutouts that closely match the size of the X-ray beams passing through. The baffles will be attached to the interior of the shroud and deploy along with the shroud.

5. SPACECRAFT BUS

IXO's spacecraft bus contains the satellite systems that provide services to the instruments such as power distribution, attitude control, propulsion, and telemetry. The bus is a nine-sided box with a bottom deck. A riveted aluminum frame forms the basic shape and also serves as the spacecraft electrical ground. The nine equipment panels are graphite composite sandwich honeycomb panels about 1m tall, 1.3m wide, and 2 cm thick. Panels bolt to the aluminum frame. During Integration and Test (I&T), they will fold down or hinge open for easy access to the avionics inside. The bottom deck is also a composite honeycomb panel and will have a 2.4m diameter cutout in the center for the X-ray beams to pass through.

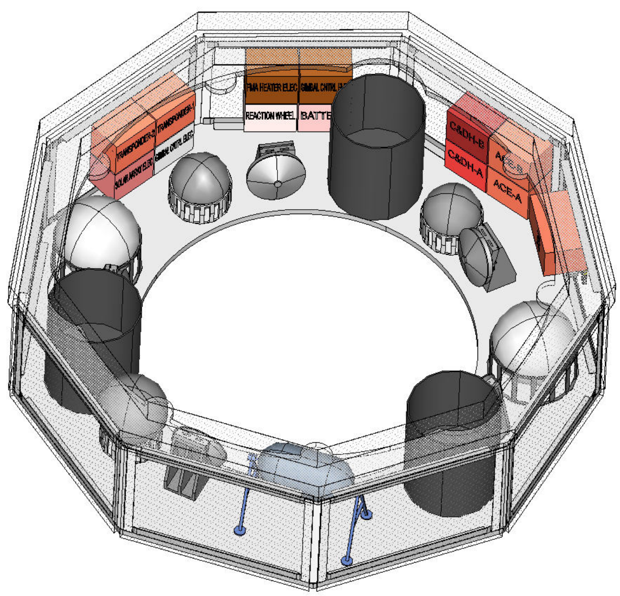

Figure 11 – Spacecraft Bus with avionics

The avionics will be located on the anti-sun side of the spacecraft so that their heat can be radiated away through the panels. Patches of white thermal paint will be applied on the exterior of the panels at the boxes footprints to facilitate the radiation heat transfer. The remainder of the exterior of the bus will be blanketed. The mass of the fully integrated bus is estimated at 900 kg with an additional 300 kg of propellant.

The hydrazine and oxidizer bi-propulsion tanks and a helium pressurant tank will be symmetrically located inside the bus. In this orientation, the center of gravity will not shift as the propellant is expended. The tanks will hold approximately 300 kg of fuel to maintain attitude control and L2 orbit position for at least 10 years.

Also inside the bus are five reaction wheels for fine attitude control, inertial reference units, the electrical power distribution system, command and data handling avionics boxes, a small battery, data recorders, and the Ka-transmitter electronics. Over 150 kg of wiring will connect them together.

There are several systems on the exterior of the bus. Several triads of 22 N attitude control thrusters will be installed around the perimeter of the observatory. The high-gain Ka-band antenna will be stowed against the side. Launch locks will secure the IM to the top deck during launch.

6. FIXED METERING STRUCTURE

The metering structure is essentially a 6.7m long by 3.4m diameter spacer between the spacecraft bus and the FMA mirror. The term "fixed" refers to the fact that it is a static structure that does not deploy. Its function is to space the bus far enough away from the FMA so that the conically-shaped X-ray beam can pass through the bus. Also, the solar arrays and the star trackers may be attached to its sides.

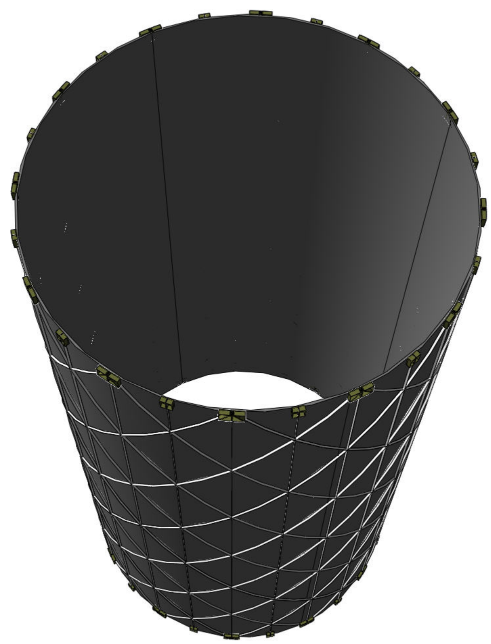

Figure 12 - Fixed Metering Structure

The metering structure will be a Composite Fiber Reinforced Plastic (CFRP) isogrid cylinder. It will consist of a 1-1.5mm thick facesheet of CFRP with 2-4cm tall ribs for stiffening. Recent analyses assume a composite layup of

M55J fiber with 954-3 cyanate-ester resin in a quasi-isotropic layup providing a thermal expansion coefficient (CTE) close to zero. This layup has a Young's modulus (E) of about 15 MSI, with greater strength and lower density than aluminum. The near-zero CTE helps to maintain the focal length of the observatory in various thermal environments during ground testing and on orbit. The high Young's Modulus helps the observatory meet the 8 Hz 1st bending mode requirement imposed by the launch vehicle.

The isogrid concept, also known as an Advanced Grid Stiffened structure (AGS), was chosen since it provides a higher stiffness to weight ratio to other types of cylindrical design approaches such as skin-stringer, stiffened skin, and semi-monocoque. Recent efforts, such as an Air Force program for the Minotaur fairing, have been successful in making AGS structures from CFRP that are lighter weight and have lower parts count than previous designs[2]. The mass of IXO's metering structure is estimated to be 290 kg, not including metallic fittings or inserts.

Manufacturing of the metering structure is anticipated to be via a large automatic fiber placement machine. The structure is made by laying CFRP prepreg tape or cloth onto a mandrel. After all the prepreg is layed down on the mandrel, the whole assembly is moved into an autoclave for curing. After curing the piece is removed from the mandrel and is ready for further processing such as installing inserts or bonding fittings.

Just a few years ago, structures this size were considered too large for automated machinery, but recent advances have now made such structures routine. Large airplane fuselages are now being made by automatic fiber placement machines in factories in the U.S. and Europe. The cost of mandrel fabrication has also been greatly reduced as well.

There is much work to be done to optimize the structure in terms of the thickness of the facesheet, and geometry of the ribs. Another task will be to find the best CFRP layup. There may be other fiber-resin systems available that can provide higher stiffness to weight ratios and meet all the other requirements.

Choosing the length of the metering structure is a balance between several factors. One factor is the desire to keep the center of gravity of the spacecraft low to stay within the CG height limitation of the payload adapter and to minimize launch loads. On one hand, the deployable system is lighter per meter of length than the fixed metering structure so there is a desire to maximize the mast length and minimize the fixed metering structure length. On the other hand, the deployable mast system is less stiff than the fixed metering structure and must have a high enough bending and torsion resonant mode to satisfy the attitude control system and the instrument pointing requirements. This argues for a short deployed length. Lastly, the spacecraft bus must be large enough allow the X-ray beam to pass through it. The X-ray beam exits the FMA with a diameter of 3.2m and tapers to ~10cm diameter at the focal plane 20m away. The spacecraft bus requires about 80cm of "wall space" all around the beam to accommodate the propulsion tanks and avionics. A maximum bus diameter of 4m was selected because this fits into the static envelope of the Atlas 5 fairing (4.57m diameter) with a little margin. Considering the 80cm distance needed from each panel, the size of the X-ray hole can be 2.4m diameter. At this size, the length of the fixed metering structure works out to 6.7m.

Titanium fittings will be bonded to the top of the metering structure to interface to the spacecraft bus bottom deck. Likewise, fittings bonded to the bottom of the metering structure will interface to the spacecraft adapter ring. Titanium (usually Ti-6Al-4V) is often used for fittings on composite structures because of its strength and relatively low CTE which is compatible with the near-zero CTE of the CFRP. Because of the low bearing strength and thin-walls of most CFRP structures, it is not advisable to thread or bolt through them directly in a heavily loaded joint.

7. FLIGHT MIRROR ASSEMBLY (FMA)

The FMA is the huge x-ray collecting and focusing device for the X-ray telescope. It weighs about 1700 kg and is arguably the largest design driver of the observatory. Another paper in this conference discusses the FMA design in detail[3]. The FMA's Wolter-I x-ray telescope optical design consists about 14,000 0.4mm thick glass mirror segments densely packed into a 3.2m diameter and supported with micron level accuracy and stability.

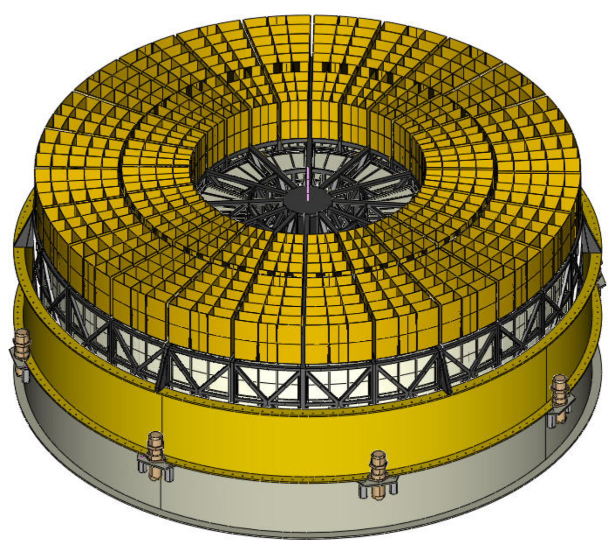

Figure 13 – Flight Module Assembly and spacecraft adapter.

A research and development program at NASA Goddard Space Flight Center has been in place for several years to determine the best way to form the glass mirrors and align them into a flight-like structure. The current concept involves placing flat glass sheets over a highly polished and figured mandrel and heating them to ~600C to the point that the glass slumps onto the mandrel and takes its shape. After cooling, the glass is removed, held in some fashion to reduce the effects of gravity, and bonded at the edges to a mirror module structure.

The primary structure of the FMA looks like a wagon wheel. It is currently envisioned to have 24 spokes which intersect at a central hub. Around the spokes are three rings. Modules will be kinematically mounted to the wagon wheel structure.

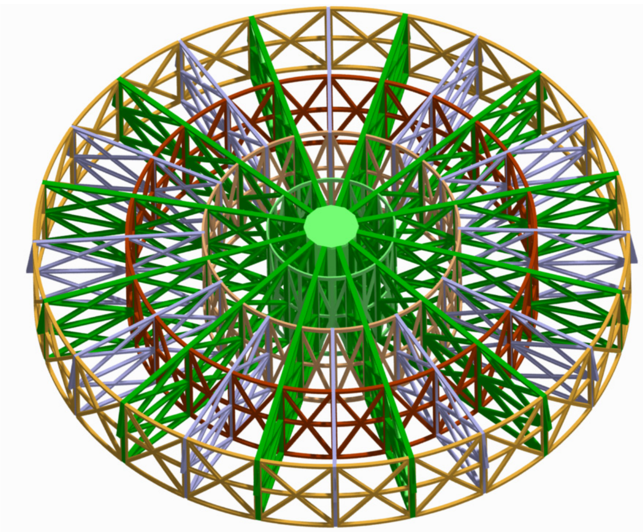

Figure 14 – One FMA primary Structure concept

The mirror module is the basic building block of the FMA. A module is a wedge shaped housing which contains about 240 glass segments and weighs about 20 kg. The current FMA design contains 24 modules around the FMA perimeter to form the outer ring. There are another 24 modules in the middle ring and 12 in the inner ring.

The FMA will be fastened to the spacecraft adapter ring in at least 6 points. The spacecraft adapter ring is essentially an aluminum cylinder with flanges on top and bottom. Shims and/or flexures will be used between the FMA and the adapter mounting flange to ensure that the FMA does not warp and come out of alignment when the bolts are torqued down, as well as relieve misalignments in the FMA caused by thermal gradients on the spacecraft adapter.

8. LAUNCH VEHICLE INTERFACES

The FMA and metering structure will be attached to the top flange of a 43 cm tall, 3.4m diameter aluminum cylinder called the spacecraft adapter. The bottom flange of the adapter is the interface to a separation ring which contains a mission-unique separation system. The separation ring is another 43 cm by 3.4m diameter aluminum cylinder with flanges on the top surface where the separation system secures the spacecraft adapter for launch. The separation ring is bolted to a 3302 Truss Adapter which is fastened to the top of the Centaur upper stage. The truss adapter is provided by Atlas.

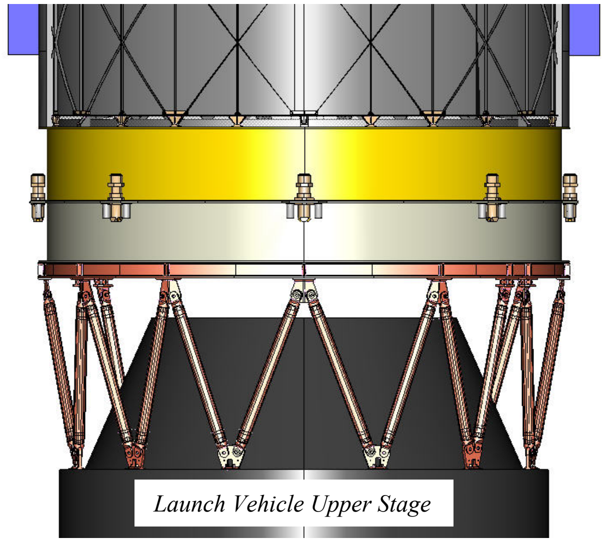

Figure 15: IXO on the 3302 Truss Adapter

The diameter of the spacecraft adapter and separation ring is driven by the FMA diameter of 3.3m. This is too large to allow IXO to use a standard payload adapter and separation system provided by Atlas. It could be possible to make an inverted frustum (truncated cone) adapter from the spacecraft to the largest standard size payload adapter (1.6 m diameter) but its smaller size obscures the aperture of the FMA. Therefore, the frustum adapter would have to be jettisoned in orbit which is an added complication and weight penalty. Another problem which rules out the 1.6m adapter is that IXO exceeds its center of gravity height limitation.

IXO’s mission-unique separation system could be either a Marmon clamp-band or a set of separation nuts and springs. Since no clamp-band of that size is known to be qualified for flight, currently IXO is planning to use a set of 8 pyrotechnic-actuated separation bolts with push-off springs to achieve the separation.

Truss Adapter

Large diameter truss adapters are offered by the Atlas Mission Planner’s Guide as an option for heavy spacecraft or those with a very high center of gravity. The 3302 truss adapter is nearly the perfect size for the current IXO configuration. To the best of our knowledge, the 3302 truss adapter has not yet been qualified for flight.

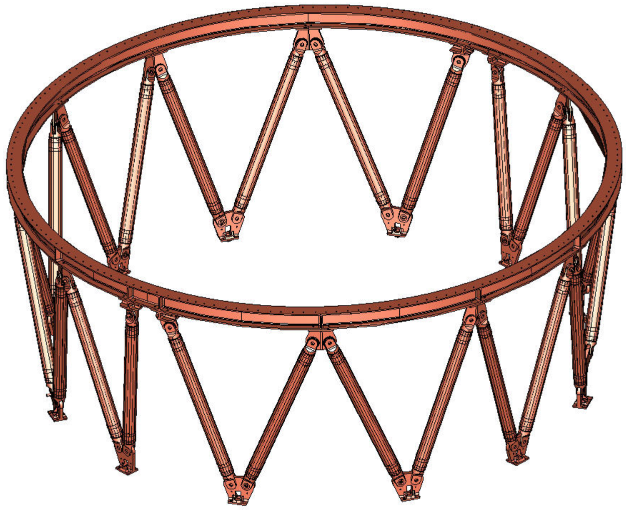

Figure 16 – 3302 Truss Adapter concept model

The center of gravity height limitation on the 3302 truss adapter is at 5.7m above the standard interface plane. IXO currently has a stowed center of gravity of 4.6m which is within the capability.

For the Ariane V launch vehicle option, a different payload adapter will be required. The current plan is to use a derivative of ESA's Separation and Distancing Module (SDM) which was used on the Jules Verne Automated Transfer vehicle. The SDM is essentially a large cylindrical ring of diameter 3.94m in diameter. It interfaces to the top of the Ariane V's upper stage. Regardless of the launch vehicle that is ultimately chosen, it appears that a mission unique payload adapter will be required for IXO.

9. MECHANISMS

There are many mechanisms required for IXO to achieve mission success. Beyond the mechanisms internal to the instruments, there are the following mechanisms starting at the instrument end of IXO:

- Launch locks for the Fixed Instrument Platform to the Bus. Expected to be pyrotechnic-actuated separation nuts.
- Rotation stage for the Moveable Instrument Platform to position the XMS and WFI in the focus. The motor is expected to have primary and backup stepper motors with a gearbox and engagement mechanisms.

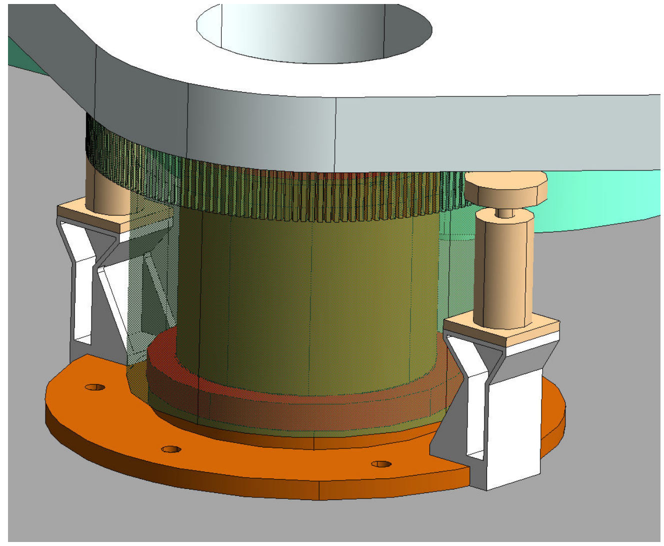

Figure 17 – Rotation Stage concept

- Focus mechanisms for XMS and WFI with a range of motion of +/- 1.5cm.
- Lateral positioning mechanisms for the XGS camera box with a range of motion of ~ +/- 1cm.
- Deployable Masts (discussed earlier).
- High Gain Ka-Band Antenna with azimuth and elevation gimbals to maintain pointing at the Deep Space Network ground antennas.
- Two solar array wings. Standard commercial arrays and pyrotechnic separation systems are planned.
- Launch vehicle separation system (discussed earlier)
- FMA internal cover.
- FMA external cover, to be jettisoned.
- FMA deployable sunshield

Mechanisms will be single failure tolerant if possible.

FMA Covers and Sunshield

The FMA will have covers on the fore and aft ends during Integration and Test (I&T) to reduce the contamination by particulates and condensation of offgassed substances, particularly silicones, onto the mirrors. The FMA will be purged by ultra-pure nitrogen during I&T and both covers must be able to withstand about 0.1 psi of differential pressure caused by the purge and launch depressurization. After launch and once the outgassing and moisture desorption of IXO has diminished sufficiently, the FMA covers will be removed. The exterior cover will be jettisoned by springs. The actuation device, or trigger, has not been selected at this point, but could be paraffin wax or solenoid. There is a desire to not use pyrotechnic devices so close to the glass mirrors on the FMA due to the shock loads caused by these types of devices.

The interior cover on the FMA cannot be jettisoned and several options have been discussed. One option is to put a hinge on the cover and rotate it out of the FMA field of view. Because the 3.4m diameter cover will interfere with the metering structure, it must fold up as it hinges. Another option is to construct the cover from a thin membrane such as Kapton and to gently pull it into the center of the FMA or off to the side with a motorized winch system. Another option is to avoid having an interior cover altogether and maintain strict contamination control around the FMA when it is installed into the observatory during I&T.

The FMA's sunshield is planned to be a simple pop-up frame. The frame will be made from a flexible material such as fiberglass or CFRP. An MLI blanket will be attached to the frame. When stowed, the frame will be bent to fit around the FMA, the blanket will be folded appropriately, and the frame latched. Deployment will occur with the unlatching of the frame via an actuator device which has not yet been selected. Once unlatched, the stored strain energy of the frame will cause it to pop up into position.

10. CONCLUSIONS

Most of the mechanical implementation on IXO will be based on technology with extensive flight heritage, although in some cases on a larger scale than ever before. The fixed metering structure may become the largest single composite piece to fly in space. IXO's synchronized triad of masts have not flown in space, although individual masts certainly have. Pull-up shrouds have flown in space but not nearly of the size and shape as planned for IXO. In addition to these, dozens of individual mechanisms must work reliably on IXO. The IXO mechanical engineering team has many challenges ahead in the years to come. There is much to do.

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BIOGRAPHY

David Robinson *is the Lead Mechanical Systems engineer of the IXO Project at the NASA Goddard Space Flight Center. He has worked on several scientific satellites including the Solar Dynamic Observatory and Swift. He started his career with NASA Glenn Research Center in 1990 working on the International Space Station and microgravity space experiments for shuttle and the Russian Mir space station. He received a B.S in Aerospace Engineering from the University of Virginia, a M.S. in Mechanical engineering at Cleveland State University, and a M.S. in Space Studies at the International Space University in Strasbourg, France.*

Ryan McClelland *is a Senior Mechanical Systems Engineer at SGT Inc. currently leading the design of the IXO Flight Mirror Assembly. His previous technology development experience includes work on aluminum foam core optical systems and non-linear effects of clearances in kinematic mechanisms. Ryan has also worked on flight missions with designs currently on orbit aboard the Hubble Space Telescope and Space Technology 5 spacecraft. He received a B.S in Mechanical Engineering, summa cum laude, from the University of Maryland.*

Preliminary Design of the International X-Ray Observatory Flight Mirror Assembly

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ABSTRACT

The Flight Mirror Assembly (FMA) preliminary mechanical design for NASA's next major X-ray telescope mission, the International X-Ray Observatory (IXO), has been developed at NASA Goddard Space Flight Center (GSFC). The design addresses some unique engineering challenges presented by the unprecedented combination of high angular resolution and large effective area required to achieve the desired scientific objectives. To meet these requirements, the Wolter-I Soft X-Ray Telescope (SXT) optical design consists of about 14,000 0.4 mm thick glass mirror segments densely packed into a 3.4 m diameter FMA and supported with micron level accuracy and stability. Key engineering challenges addressed include ensuring positive stress margins for the glass segments with a high Factor of Safety, keeping the structure light enough to launch, providing a large effective area, and preventing unacceptable thermal distortion. Standard mechanical design techniques such as FEM modeling and optimization, integrated optomechanical analysis, and development testing were applied to this unique problem. The thin mirror segments are mounted into 60 intermediate wedge shaped structures called modules. Modules are kinematically mounted to the FMA primary structure which is optimized for minimum mass and obscuration of the clear aperture. The preliminary design demonstrates the feasibility of building and launching a large space-based SXT using slumped glass mirrors which meets the IXO effective area, mass, structural, and thermal requirements.

Keywords: International X-Ray Observatory, IXO, Module, Flight Mirror Assembly, FMA

1. INTRODUCTION

The FMA provides IXO with the unprecedented combination of soft X-ray collecting area and angular resolution required by the science objectives for the mission [1]. The FMA design concept detailed in this paper illustrates how the slumped glass mirror technology being developed at NASA and the Smithsonian Astrophysical Observatory (SAO) can be used to build and launch a large space-based SXT meeting the requirement shown in Table 1. The preliminary FMA design leverages existing aerospace technology and requires no technology development beyond what is currently being pursued, namely the fabrication, alignment, and mounting of slumped glass X-ray optics [2].

Table 1. FMA preliminary driving requirements. The requirements necessitate a light stiff structure which obscures a minimum about of the clear aperture.

Requirement	Value
Effective Area	3.0 m^2 @ 1.25 keV 0.6 m^2 @ 6.0 keV
Angular Resolution	5 arc-sec
Focal Length	20 m
Mass	1750 kg
First Axial Mode	35 Hz
First Torsional Mode	15 Hz
Quasi-static design loads	7.5 g lateral 10.5 g axial
Operating temperature	20°C ±1

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The primary increase in performance, relative to past missions, required of the IXO FMA is the effective area for X-ray photon collection in the 1 keV to 6 keV range. Where the Chandra X-Ray Observatory had 4 primary/secondary mirror pairs, IXO must have ~360. Correspondingly, the mirrors must be much thinner in order to accommodate the mass and volume constraints of existing launch vehicles. Supporting this large number of very thin mirrors is the central challenge of the FMA design.

The baseline FMA design was pursed to a level of detail commensurate with the pre-Phase A IXO mission study including design trade studies, CAD modeling, Finite Element Analysis (FEA), preliminary material selection, thermal analysis, and X-ray performance sensitivity analysis. In some cases the design was developed past what one would expect at this early stage, particularly with respect to optomechanical and structural analysis of the glass mirror segments mounted in the SXT modules in order to mitigate perceived mission risks.

This paper encompasses a brief overview of NASA's concept for the IXO spacecraft, which was developed in tandem with the FMA, followed by detailed mechanical description of the baseline FMA design including the mirror segments, SXT mirror modules, and FMA primary structure. An alternate design based on the same mission requirements is being developed at the European Space Agency (ESA) using silicon micro-pore mirror technology [3] and is outside the scope of this paper. The FMA also includes a Hard X-Ray Telescope (HXT) based on existing technology [4] that is not described in detail in this paper.

1.1 IXO mission overview

The International X-ray Observatory (IXO) is a collaboration between NASA, ESA, and JAXA which is planned to launch in 2021 [5]. It combines elements from NASA's prior Constellation X program and ESA's XEUS program. IXO will be a Great Observatory-class mission which builds upon the legacies of the Chandra and XMM-Newton X-ray observatories. IXO will have a mass of around 6600 kg and will be approximately 23 meters long when deployed and 4 meters in diameter. It will fly on an Atlas 5 or an Ariane V rocket into an L2 halo orbit. On orbit roll and pitch on the spacecraft are limited so that the sun always shines on one side to ensure a stable thermal environment. The observatory is divided into four modules to simplify integration and testing of the observatory as shown in Figure 1.

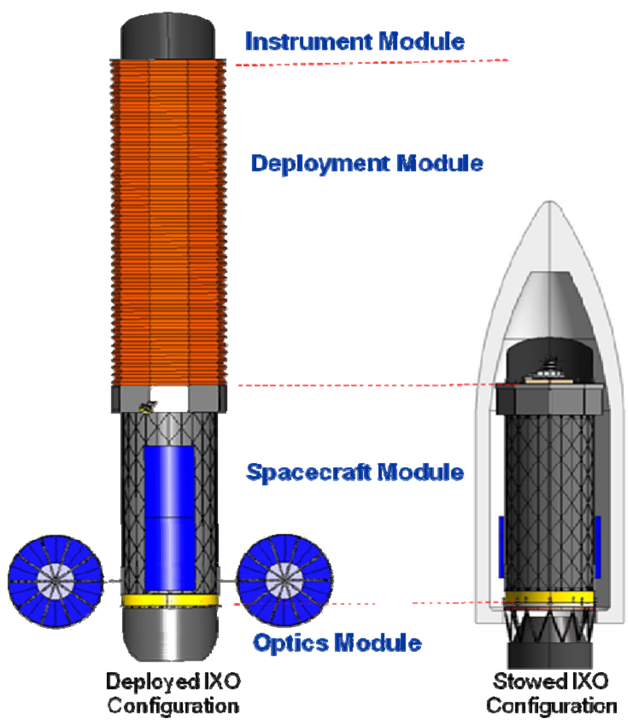

Figure 1. IXO Spacecraft shown in deployed and stowed configurations. The observatory is divided into four modules to simplify integration and testing. The FMA is contained within the Optics Module. IXO will have a mass of around 6600 kg, a length of approximately 23 m when deployed, and a 4 m diameter.

1.2 FMA preliminary design overview

The FMA consists of 60 SXT modules, each containing approximately 200-300 mirror segments, mounted into the FMA primary structure as shown in Figure 2. The inner ring has 12 modules and the middle and outer ring have 24 modules each. Table 2 gives a breakdown of the size, mass, and effective area of the three types of modules. Each module has additional thermal and optical elements mounted to it, including a thermal pre-collimator and a Stray Light Baffle (SLB) or thermal shield. The Hard X-ray Mirror Module (HXMM) is mounted into a central hole in the primary structure. Note that the term 'module' in this paper generally refers to the SXT modules rather than the HXMM.

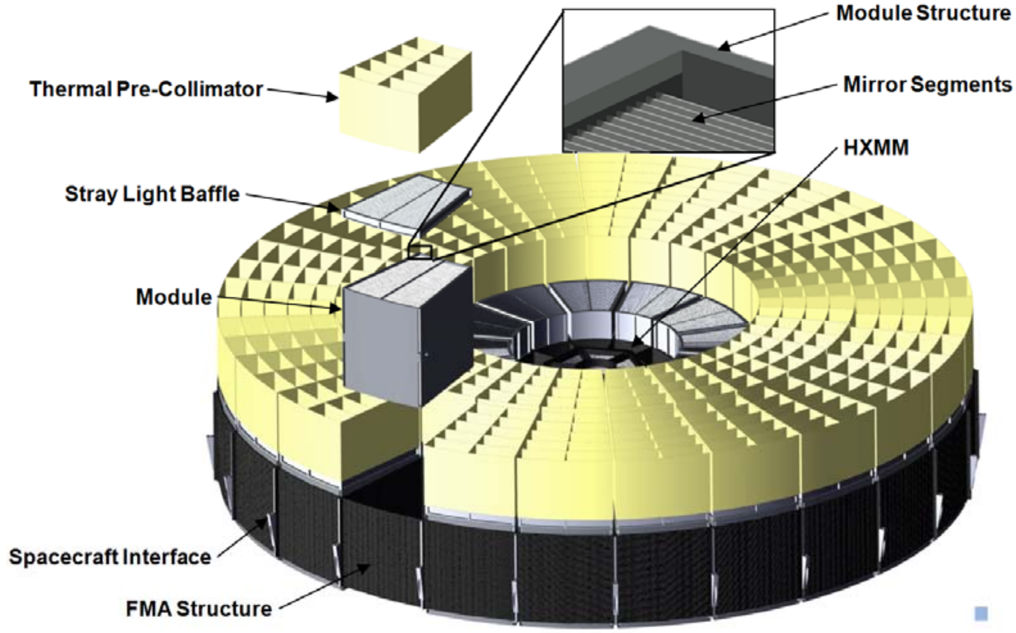

Figure 2. Exploded view of FMA and module illustrating the basic parts of the assembly. Each of the 60 SXT modules contains approximately 200-300 bonded-in mirror segments.

The FMA mounts to a spacecraft adapter ring via 24 bolted interfaces located around the perimeter of the primary structure. The spacecraft adapter ring in turn mounts to the launch vehicle payload adapter fitting with several pyrotechnic devices, forming the separation plane for the observatory.

Table 2. Parameters of the inner, middle and outer modules. The outer modules have greater effective area at lower X-ray energies and the inner modules have greater effective area at higher energies.

Ring	Modules per Ring	Azimuthal Span (deg)	Number of Mirror Segments	Azimuthal Span of Largest Mirror (mm)	Est. Module Mass (kg)	Effective Area at 1.25 keV (m2)	Effective Area at 6.0 keV (m2)
Inner	12	30	286	335	22.6	0.038	0.031
Middle	24	15	230	263	15.7	0.040	0.018
Outer	24	15	206	392	21.9	0.074	0.001

2. MIRROR SEGMENTS

2.1 Segment geometry

The fundamental elements of the FMA are the slumped glass mirror segments. In order to maximize effective area, the segments must be packed together as densely as possible without one primary segment shadowing the next as shown in Figure 3. The thinner the mirror, the more densely the shells can be packed. To meet the effective area and mass requirements the FMA uses 0.4mm thick segments arranged into 361 concentric rings of primary and secondary mirrors. The spacing between mirrors ranges from 1.5 mm to 4.5 mm.

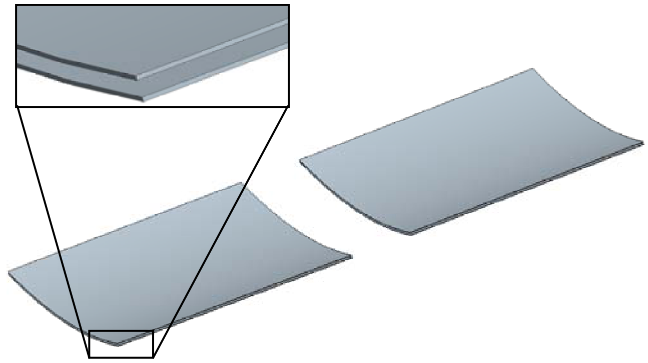

Figure 3. Two pairs of primary and secondary mirror segments spaced 1.5 mm apart. The 0.4 mm thick segments must be closely packed to achieve the required effective area.

Segments are slumped from commercially available Schott D263 glass onto polished mandrels to facilitate large scale production [2]. Limitations in the azimuthal size of segments that can be slumped, along with structural considerations discussed in Section 5, led to the 12/24/24 module layout previously described. Each segment is 200 mm in axial length and 167-392 mm in azimuthal span, roughly the size of an A4 sheet of paper. The total glass mass of all 13,896 segments is 730 kg.

2.2 Segment strength

In order to calculate stress margins and demonstrate the mirror segments can be launched with the required 3.0 Factor of Safety the strength of the glass segments must be well understood. Determining the strength of glass is more complex than for an analogous metal optic due to the nature of brittle glass failure which is dependent on the size and distribution of surface flaws. The statistical strength of a population of glass segments is effectively expressed by the two parameter Weibull distribution which describes the Probability of Failure (POF) as a function of the characteristic strength (σ_0) and the Weibull modulus (m) [6].

$$POF = 1 - \exp\left[-\left(\frac{\sigma}{\sigma_0} \right)^m \right] \tag{1}$$

The strength of the test specimens can be related to the strength of the glass segments as supported in the FMA, which have different stressed areas, by the following equation [7]:

$$\frac{\sigma_1}{\sigma_2} = \left(\frac{A_2}{A_1} \right)^{\frac{1}{m}} \tag{2}$$

Extensive materials testing has been performed on slumped glass segments in order to determine the Weibull parameters, including both folding tests and tests simulating the segment-to-module bonding geometry as shown in Figure 4. Results from the tests using a simulated bond joint, scaled by the number of bond areas, are used for margin calculations due to their superior representation of the actual stress state. The Figure 4 graph shows the design strength for slumped D263 glass segments bonded at eight locations to the module as a function of the POF based on the aforementioned test results. Detailed stress analysis of the segment and Margin of Safety determination are presented in Section 4.

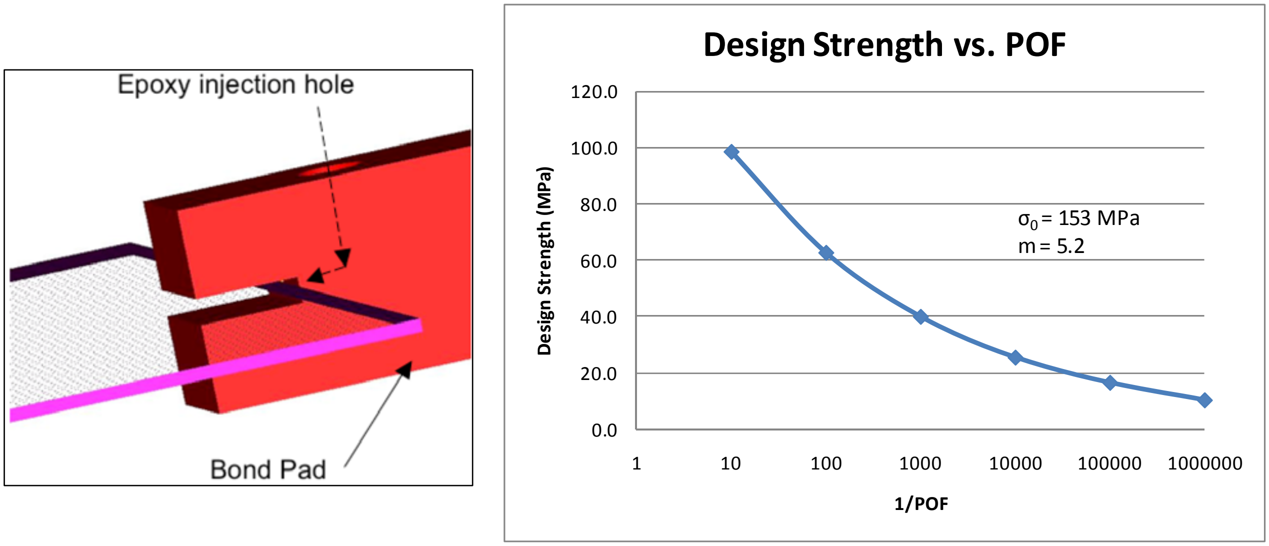

Figure 4. Glass strength test sample bonded to structure (left) and resulting statistical strength data as a function of POF using 30 test samples (right). Stress margins are very sensitive to the desired POF. Strength has been scaled by the increased area of eight bond points.

3. MIRROR MODULES

Many pairs of segments are bonded into a truncated wedge shaped structure to form the SXT modules shown in Figure 5. Modules are the optical building blocks of the FMA and the creation of modules meeting all requirements is the focus of the technology development efforts taking place at NASA and SAO. In parallel with the technology development efforts to fabricate, align, and mount segments into a structure, a baseline module mechanical design has been developed that demonstrates the feasibility of building the module structure, integrating the modules into the FMA, launching the populated modules, and sufficiently controlling the on-orbit thermal distortion. In addition to FEA, environmental tests have been performed that demonstrate the ability to model the dynamic response of the segments in simulated launch environments including stress prediction.

3.1 Benefits of the modular design

The slumped segments lend themselves well to the modular approach which has several important advantages versus a monolithic design:

- Reduces risk. If one segment or set of segments is damaged before launch, the module can be replaced with a spare.
- Allows for easier handling. Modules are designed to be a manageable size for assembly, transportation, and test.
- Reduces FMA fabrication time. Since integrating large numbers of segments will be time consuming, the modular approach allows for parallel assembly lines.
- Reduces load in mirror segments. Kinematically mounted modules take segments out of primary load path.
- Reduces thermal distortion of mirror segments. Kinematically mounted modules decouple the deformation of the primary structure for the deformation of the segments.
- Approach is applicable to X-ray mirrors of arbitrary size.

From a structural engineering standpoint, the kinematically mounted modules become payloads supported by the primary structure. This design decouples the module and primary structure stiffness's and greatly simplifies the structural analysis.

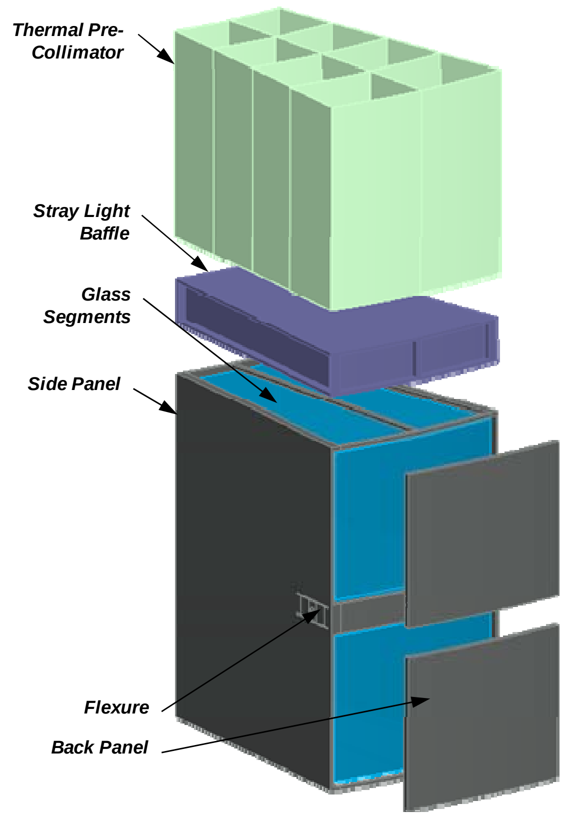

Figure 5. Exploded view of a middle ring module. The module structure consists of panels on four sides and radial rails to which the segments mount. The SLB and Thermal Pre-Collimator are also mounted onto the module.

3.2 Module design overview

The fully assembled module includes the mirror segments and the supporting structure as well as additional thermal and optical elements. The structure consists primarily of load bearing panels which close out the module on all but the axial ends, where X-rays must pass through. There are several advantages to closing out the module with panels:

- Panels provide lightweight structural stiffness needed to keep the segments aligned during integration, testing, and launch.
- Panels protect the mirror segments from Foreign Object Damage (FOD).
- Panels protect the mirror surfaces from direct impingement of acoustic energy, reducing launch stresses.

Rails rigidly fastened to the interior of the modules are used to mount the tabs to which the mirrors are bonded as shown in Figure 6. Each segment is bonded at three locations along each axial edge and one along the top and bottom edges for a total of eight bonds per segment. The side and front module panels have integrated flexures in a kinematic arrangement allowing for the decoupling of module and primary structure deformations.

A custom low Coefficient of Thermal Expansion (CTE) Titanium/Molybdenum alloy was selected for the module structure to closely match the 6.3 ppm/C CTE of the D263 glass segments. Other materials being considered include Carbon Fiber Reinforced Plastic (CFRP), Nickel/Iron alloys such as Kovar and Alloy 42, and various metal matrix composites.

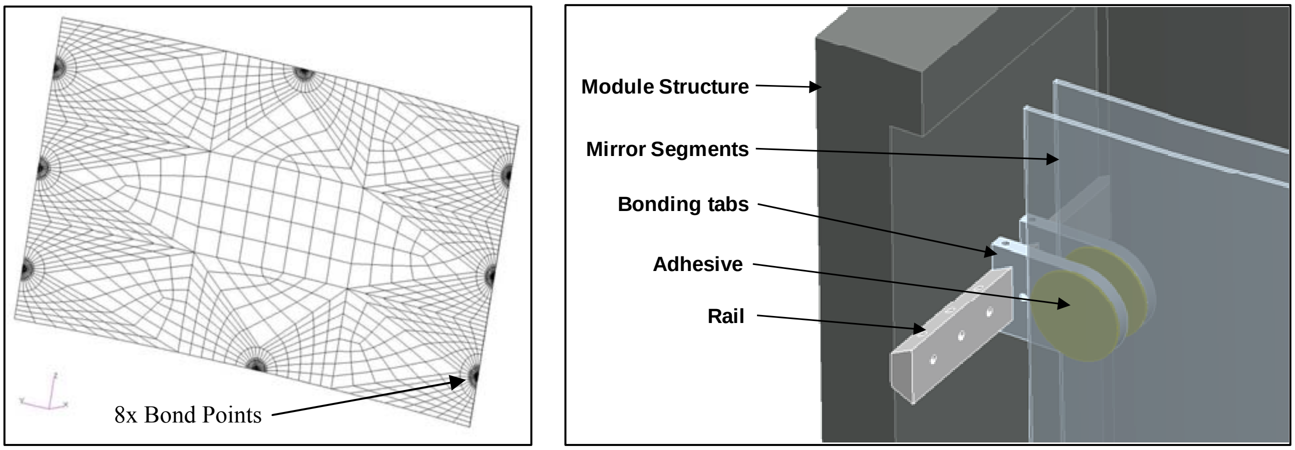

Figure 6. FEM of a segment showing the eight bonding areas (left) and close-up of two segments bonded into a development module (right). The segments are bonded to tabs which are attached to rails fastened to the module side panels.

3.3 Optomechanical sensitivity analysis

A detailed Finite Element Model (FEM) of an inner module was generated in order to determine the performance of the design under various thermal and structural loads (Figure 7). Generating the hundreds of unique mirror segment FEMs within the module with sufficient accuracy and element density to allow for ray traced X-ray performance prediction was particularly challenging. Custom software was written to allow the segment FEMs to be automatically generated with the desired mesh density based on an optical prescription file. Additional custom software was written to extract the FEA output and ray-trace the results to generate performance predictions for both individual segments and entire modules. Performance predictions are based on the low order surface deformations and equations published in reference [8].

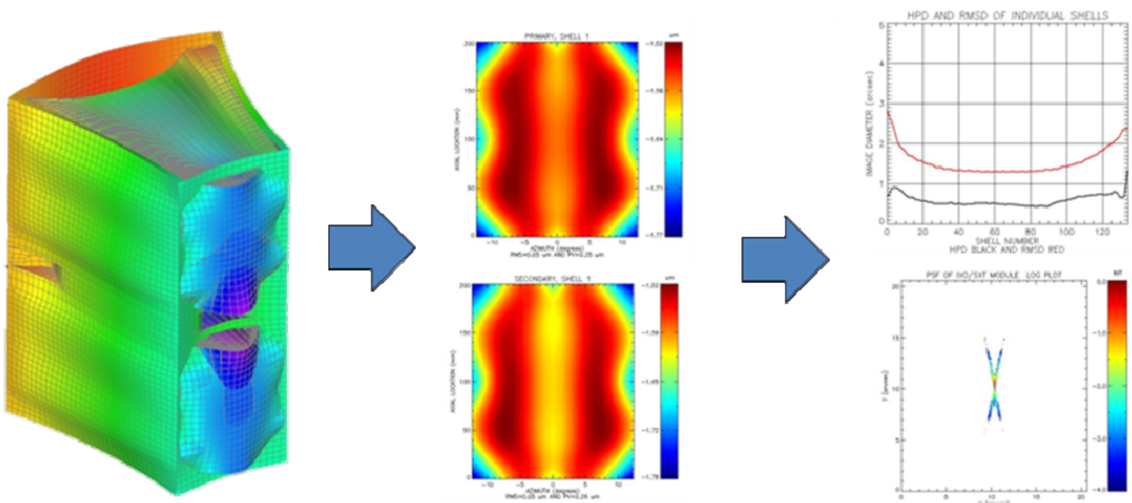

Figure 7. Analysis flow from FEA results using a detailed FEM (left) to segment surface distortion determination (center) and X-ray performance prediction for each segment (upper right) and the entire module (lower right).

In order to aid in the module structural and thermal designs, the sensitivity of X-ray performance was predicted with respect to various design parameters and thermal loads as shown in Table 3. These results indicate that performance is very sensitive to radial thermal gradients, gradients between the module structure and segments, and CTE mismatch between the D263 glass and module structure. These valuable insights were taken into account in material selection and thermal design. Full mapping of the predicted module temperatures to the FEM and subsequent optomechanical analysis will take place in a later project phase.

Table 3. Predicted sensitivity of module X-ray Half Power Diameter (HPD) performance to various thermal cases. These results were considered in structural and thermal design.

Case	HPD (arc-sec)
1°C bulk temperature change applied with 1.0 ppm/C CTE structure mismatch	3.0
1°C thermal gradient in radial direction with CTE matched structure	1.6
1°C thermal gradient in azimuthal direction with CTE matched structure	0.3
1°C thermal gradient in axial direction with CTE matched structure	0.1
1°C thermal gradient between module structure and segments with CTE matched structure	19.5

3.4 Module thermal design and analysis

In order to keep the thermal distortion of the segments within acceptable limits (~1 arc-sec HPD) during integration and on-orbit operation, the temperature of the modules will be tightly controlled near room temperature. This goal is aided by the relatively quiescent L2 thermal environment and limitations on the roll and yaw of the observatory (±20° yaw, ±10° roll). A sunshade prevents direct illumination of the FMA. The challenge then becomes replacing the heat lost to space by the segments while maintaining minimal thermal gradients over the modules. The baseline thermal design includes active heater control on the forward section of the metering structure, active heater control on the SLBs, and thermal pre-collimators that limit the view of the segments to space as shown in Figure 8. Using this strategy, approximately 1500 W of heater power is required to keep the modules at room temperature in the cold case [9].

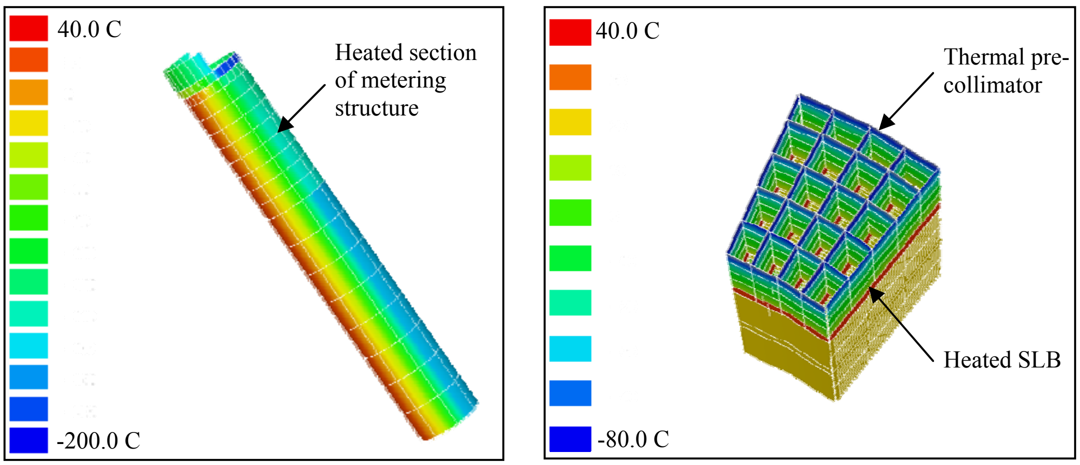

Figure 8. Thermal models of the observatory (left) and module (right). To replace the heat lost to space while minimizing the thermal gradient in the module, the forward section of the metering structure is heated, the SLBs are heated, and thermal pre-collimators are located between the modules and X-ray aperture.

4. MIRROR STRUCTURAL ANALYSIS AND TESTING

Of particular concern in the structural analysis of the FMA and modules is the stress experienced by the glass segments. In order to develop a Margin of Safety for the segments, the Ultimate Tensile Strength, appropriate Factor of Safety, quasi-static design loads, and maximum glass stress must be determined.

4.1 Ultimate Tensile Strength and Factor of Safety

Options for determining the Ultimate Tensile Strength for the design of glass parts are detailed in NASA-STD-5001. Due to the large number of segments, directly using the Wiebull parameters and a POF based on every segment having a 99% chance of survival leads to a very low strength (~10 MPa). In order to use a greater design strength, each segment must be subjected to a simple proof test which applies a stress equal to 1.2 times the Ultimate Tensile Strength before being assembled into a module. Using this method one can select a design strength that gives sufficient margins of safety while resulting in an acceptable number of segments rejected during proof testing. For instance, using a strength of 40 MPa one would expect to fail about 1 in 1000 segments during proof testing per the Figure 5 graph. This rejection rate and corresponding 40 MPa design strength are used in the margin calculations below. A 3.0 Factor of Safety is required for glass per NASA-STD-5001.

4.2 Quasi-static design loads

Due to the critical nature of this analysis more accurate quasi-static design loads were desired than are provided by a generic Mass Acceleration Curve (MAC) typically used at this project phase. In order to take into account the effect of the structural response of the FMA primary structure and IXO observatory on the module loads, a sine response analysis was performed on an integrated FEM of the observatory. A sine sweep in each axis was input at the base of the stowed observatory model and net center of gravity (CG) accelerations were recovered at the spacecraft, FMA, and module levels. The net CG accelerations of the spacecraft were scaled to the maximum Atlas 551 launch vehicle payload accelerations. The net CG accelerations of the FMA and modules were then scaled by the same factor to determine their respective maximum accelerations in each axis. The resulting quasi-static load environment for the inner module (worst case) was determined to be 8.5g lateral and 18g axial. Additional loads refinement will occur in later project phases when a true Coupled Loads Analysis is performed by the launch vehicle provider.

4.3 Glass stress and resulting Margin of Safety

Detailed solid element FEMs of the worst case segment were used to predict the maximum stress in the glass. The outermost segment of the inner module was chosen due to its large azimuthal span and relatively high curvature. Several bond geometries were investigated including the baseline semi-circular bond with a 3 mm radius shown in Figure 9.

Appling these quasi-static design loads to the worst case segment FEM resulted in maximum stress of 3.1 MPa in the worst case orientation. Using a design strength of 40.0 MPa and Factor of Safety of 3.0 as described above yields a Margin of Safety of 3.3. The maximum principal stress failure criterion was used. This result is a strong indication the modules can be successfully launched.

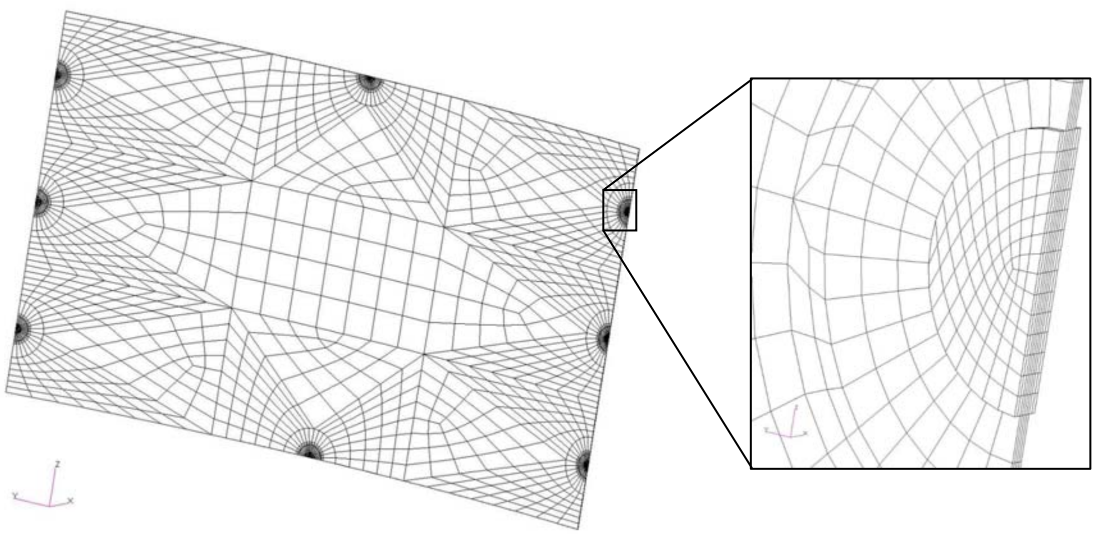

Figure 9. Detailed solid element FEM of mirror segment including a detail of bond area and epoxy modeling. This model of the baseline design shows positive margins with a 3.0 Factor of Safety.

4.4 Environmental testing

Development environmental testing and corresponding structural analysis were performed to ensure that the behavior and strength of the glass segments in the flight environments is well understood. The response to loading environments was investigated via static load testing, modal tap testing, random vibration testing, and acoustic testing including a successful acoustic test of three closely spaced segments at Atlas 551 qualification levels (Figure 10). Mirror segment response including modes and stresses correlated well with analysis predictions. Pre- and post-test mirror figure measurements show the mirror figure did not change as a result of environmental tests. A shock test simulating actuation of the pyrotechnic spacecraft separation devices is currently being developed.

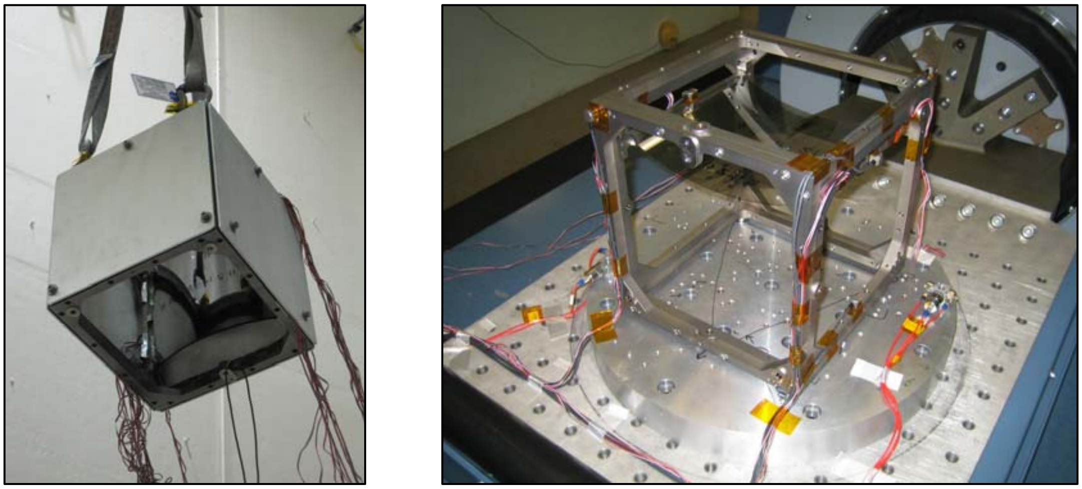

Figure 10. Acoustic test of three closely spaced segments in a simulated module with close-out panels (left). Instrumentation included 40 strain gauges on the mirror segments. Vibration test of a single mirror mounted in an open simulated module with 6 accelerometers and 9 strain gauges (right). Results correlated well with FEA predictions.

5. FMA PRIMARY STRUCTURE

The 3.4 m diameter primary structure supports all the modules during integration, launch, and on-orbit operation. It is constrained to the spacecraft at 24 locations around the perimeter. An azimuthally thin but axially thick structure is desired to minimize the projected area of the structure in the focal plane in order to maximize the effective area of the FMA while supplying a high bending stiffness. The Figure 11 graph demonstrates the effect of structure thickness on effective area.

The structure consists of primary radial beams which attach to the central cylinder and spacecraft interface, secondary radial beams, and five concentric cylinders as seen in Figure 11. The HXT is mounted within the innermost cylinder. The 12/24/24 module layout allows for each module to be attached to a primary radial beam, providing a good load path to the spacecraft.

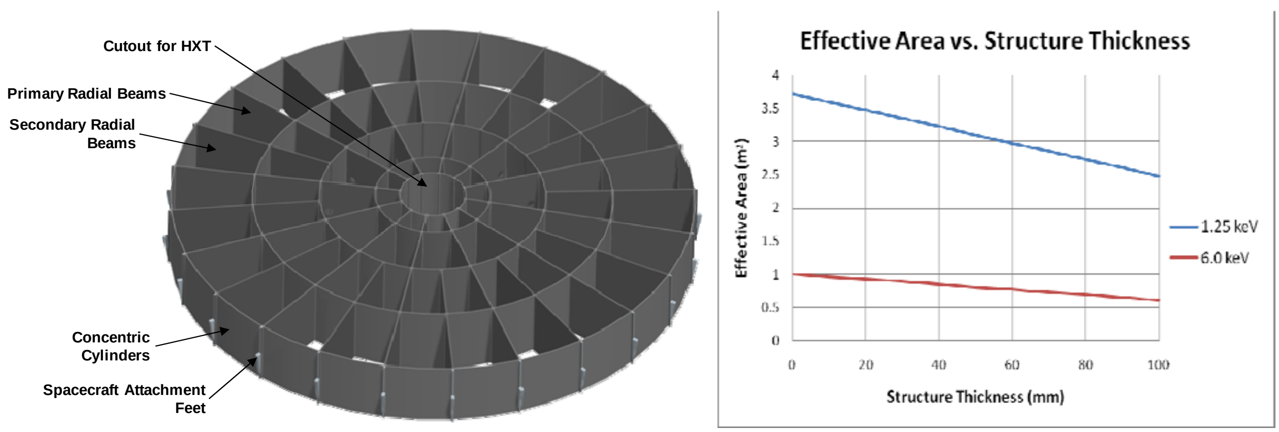

Figure 11. FMA primary structure CAD model illustrating major components (left). Graph demonstrating effect of structural member thickness on effective area at the requirement energies (right). Structure was optimized to maximize effective area.

To maximize stiffness while minimizing structural member thickness M55J/954-3 CFRP was selected as the material for the primary structure. The composite members are assembled with a wine-box style construction and bonded together with doublers to form the monolithic CFRP primary structure.

Extensive FEA was performed on the structure including software optimization to minimize member thickness while meeting the frequency requirements shown in Table 1. The resulting structure weighs 28% of the payload weight (ie modules), has a first torsional mode of 16 Hz and a first axial mode of 60 Hz. All member thicknesses are 10 mm or less, which allow for effective areas of 3.2 m^2 and 0.8 m^2 at 1.25 keV and 6.0 keV respectively, exceeding the requirements of 3.0 m^2 and 0.6 m^2. Since the design is stiffness driven, all stresses and interface forces are relatively low. The limiting stress will likely occur at the bonded interfaces which will be analyzed in detail during a later project phase. The CFRP layup chosen has a near zero CTE resulting in only 2 µm of displacement over the 3.4 m FMA diameter for a 1ºC bulk temperature change.

6. CONCLUSION

A baseline IXO FMA design meeting the sub-system requirements to a level of detail commensurate with this pre-Phase A mission study has been created and extensively reviewed. The FMA embodies new approaches to X-ray mirror design required by the unprecedented angular resolution and effective area requirements. Some of the key challenges addressed by the design study include demonstrating the ability to launch the thin glass segments, keeping the structure light enough to launch, and providing a large effective area. Extensive analysis and testing was performed to demonstrate positive stress margins for the glass segments with a 3.0 Factor of Safety. Standard mechanical design techniques such as FEM modeling and optimization, integrated optomechanical analysis, and development testing were applied to this unique problem.

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