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Constellation-X

Reference Mission Description Document

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SIGNATURE PAGE

The Constellation-X Reference Mission Description Document

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Constellation-X Reference Mission Description Document

1.0 Introduction

The Constellation-X is a follow-on project for the Chandra X-ray Telescope and this document describes its Reference Configuration. The document is divided into five sections. The first section introduces the Constellation-X. The second section gives an overall view of the Reference Configuration of the Constellation-X. The third section describes the scientific payloads and the optical benches. The fourth section describes the spacecraft bus. The final section analyzes the mission operations for the project.

The introductory section is further subdivided into purpose, scope and background subsections.

1.1 Purpose

The purpose of this document is to provide a top-level comprehensive description of systems and subsystems implemented for the Constellation-X in what is called the Reference Configuration. The configuration is used to facilitate discussion on costing, trades and enhancements to achieve the Constellation Mission Requirements. The document also provides the big picture for the interested personnel who are contributing a specific part of the whole mission. As new organizations and individuals join the Constellation-X team, the document will provide a convenient means of familiarization. It will also facilitate communication across the program.

1.2 Scope

This Reference Configuration was developed by NASA and SAO beginning in late 1998 through 1999. This will be revised in the future to reflect enhancements and new scientific requirements. This document does not reflect the equally viable configurations developed by TRW and Ball Aerospace teams under the NASA Constellation-X Mission Architecture Study Cooperative Agreement Notice (CAN-55546-232). The final selection of the configuration for the Constellation-X will incorporate further mission studies and technology developments, and the Reference Configuration may not be the final selected baseline configuration.

1.3 Background

The Constellation-X is modeled after the Keck Optical Observatory on Mauna Kea in Hawaii. Both observatories have superior collecting areas for analyzing components of the gathered light. Keck and the Constellation-X are complements to the great high angular resolution space telescopes: the Hubble Space Telescope and the Chandra X-ray Observatory, respectively. No single telescope can do everything. Hubble, along with its many excellent features, provides fantastic images of distant galaxies with unprecedented clarity. However, the Earth-based Keck is needed to complement Hubble in collecting enough light to study the composition of the gas in those distant galaxies. In a similar way, the Constellation-X will enhance the high resolution imaging capability of Chandra by providing much greater collecting area and the higher spectral resolution needed to form a more complete picture of the X-ray Universe.

The Constellation-X will be a major step forward in significantly expanding our understanding of the Universe and will allow a tremendous increase in our knowledge of three fundamental areas:

- The Limits of Extreme Gravity: Black Holes and Active Galactic Nuclei
- The Life Cycles of Matter and Energy in the Universe
- The structure of the Universe: Galaxy Clusters and Dark Matter

X-ray astronomy is the preferred and, in many cases, the only way to study many of the questions within these broad areas. The Constellation-X brings the capability to perform the required observations.

Table 1.3-1 summarizes the top level capabilities of the Constellation-X required to meet its scientific objectives.

Table 1.3-1. The Constellation-X Science Requirements

Minimum Effective Area	15,000 cm^2 at1 keV 6,000 cm^2 at 6.4 keV 1,500 cm^2 at 40 keV
Minimum Telescope Angular Resolution	15 arc sec HPD from 0.25 to 10 keV 1 arc min above 10 keV
Minimum Spectral Resolving Power ($E/\Delta E$)	300 from 0.25 to 40 keV 3000 at 6 keV 10 at 40 keV
Band Pass	0.25 to 40 keV
Diameter Field of View	2.5 arc min < 10 keV 8 arc min > 10 keV
Mission Life	3 years minimum 5 years goal
Redundancy/Reliability	No one failure to result in loss of more than 33% of the mission science

The above science requirements led the GSFC /SAO team to define mission concepts and do trade studies for the Constellation-X. In 1998, teams from TRW and Ball Aerospace and Technologies Corporation, under the Cooperative Agreement Notice (CAN) with NASA, also performed a series of independent preliminary configuration trades. The studies considered the Constellation-X and payload requirements, launch vehicle capabilities, candidate orbits, and costs for the launch vehicles, spacecraft, instrument module and payload. This process generated the baseline configurations for each of TRW and Ball Aerospace and the GSFC/SAO Reference Configuration. All the CAN configurations are documented under separate reports.

2.0 Mission Systems Description

The Constellation-X Observatory is, in fact, a multi-satellite constellation of X-ray telescopes. Each satellite carries an equal portion of the large effective area and all of the satellites point at the same target at the same time to obtain total mission effective area. The satellites are placed in an orbit around the L2 Lagrange or Libration point on the earth-sun line and, in the course of a year, they travel around the Sun just as Earth does. The Constellation-X orbit around the L2 point is large enough that the Earth never masks the sun as seen by the satellites. The same side of each satellite is always pointing at the sun so power input is continuous and a stable thermal environment exists. Except for limits that may result from roll and pitch from nominal axis, observation time is unlimited.

Pointing and post facto aspect determination are sufficiently precise and accurate that data from all of the satellites can be superposed onto a common image or spectrum just as if a single telescope had been used. Data taken at different times but from the same target can also be superimposed on themselves or on previously taken data.

The following subsections summarize the Reference Mission designs and concepts in more detail.

2.1 Satellite Configuration

The GSFC/SAO satellite configuration shown in figure 2.1-1 is one of four satellites that are launched, two at a time, on an Atlas V or Delta IV. Each satellite as shown in figure 2.1-2 has separable instrument and spacecraft bus modules. The instrument module contains all of the telescope optics and detectors, as well as their metering structure. Each instrument module houses one Spectroscopy X-ray Telescope (SXT) with a 1.6-m diameter optic assembly and three Hard X-ray Telescopes (HXTs) with 40-cm diameter optics. An X-ray calorimeter is mounted at the focus of the SXT optic. CdZnTe detectors are mounted at the focus of each of the HXT optics. A cryostat with mechanical coolers provides milli-Kelvin operating temperatures for the X-ray calorimeter. Gratings mounted aft of the SXT optic disperse a portion of the incoming X-ray photons onto an array of CCDs positioned on a Rowland Circle, close to the calorimeter.

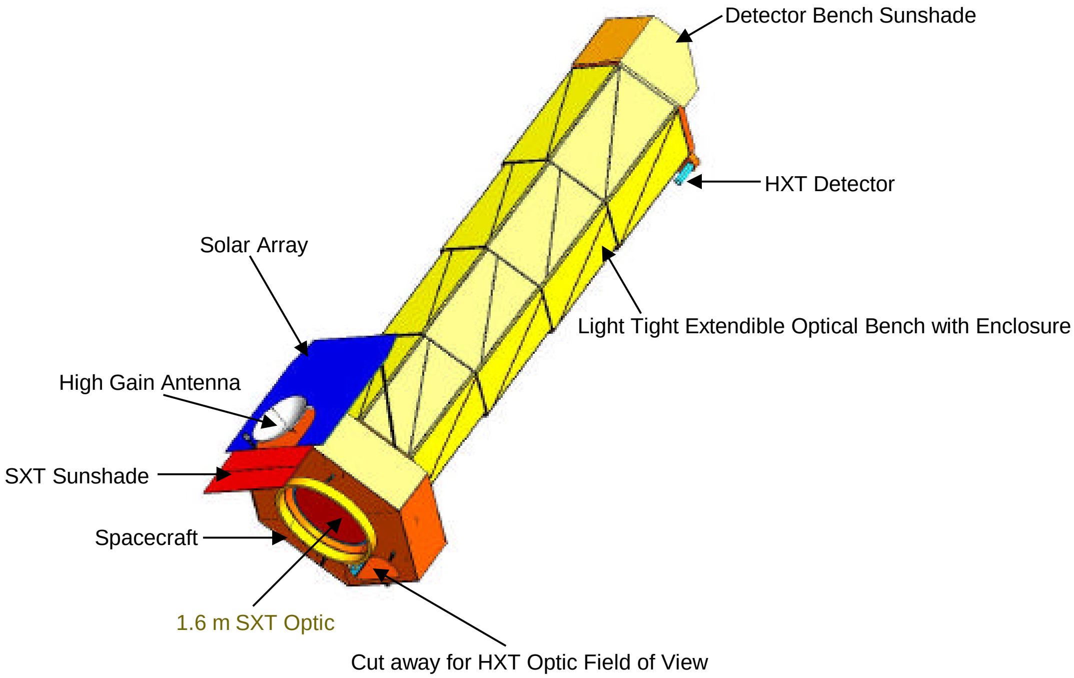


Figure 2.1-1. The Constellation-X Satellite Assembly

The detectors are deployed using an Extendible Optical Bench (EOB). This allows for the launch packaging of two satellites within a single fairing as shown in figure 2.2-3, yet provides the 10-meter focal length required for the telescope systems on-orbit. The EOB is a four-stage nested trusswork constructed out of a fiber composite material with a low coefficient of thermal expansion. It is deployed by a stepper motor/pulley/cable system. The detector end of the EOB has a relatively open view to space. This accommodates a passive thermal radiation design for the cryostat main shell and CCD detectors. The optics, a star tracker, and a gyro package mount to the same bench structure at the spacecraft bus end of the EOB.

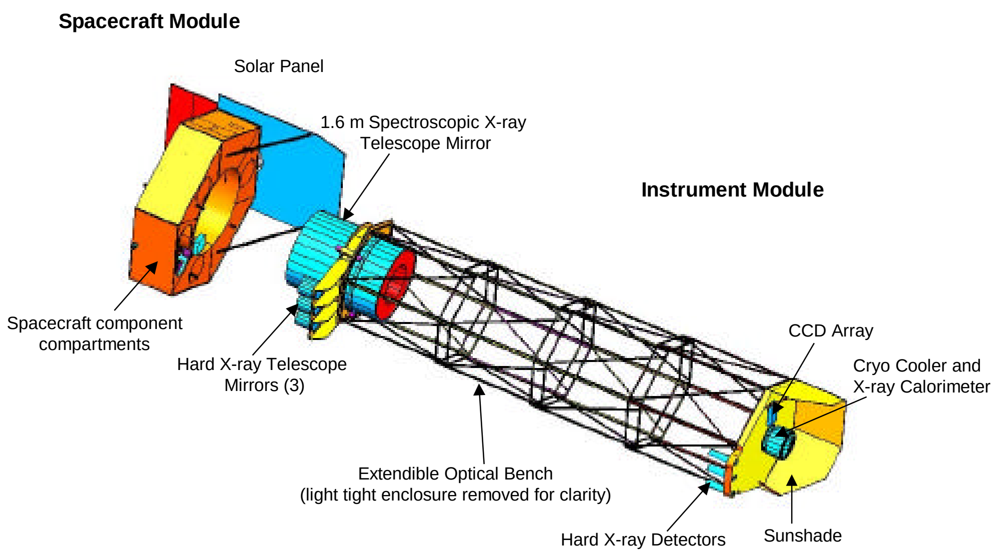


Figure 2.1-2. The Constellation-X Satellite Exploded View

The spacecraft bus module mounts kinematically to the optics bench and surrounds the large SXT optic assembly. One side of the satellite is always within 20 degrees of normal to the sun, allowing the solar panel to be fixed. A pointed high-gain antenna dish, located on the Earth-Sun side of the spacecraft bus module, is used for science data downlink. The spacecraft bus has a three-axis-stabilized system for attitude control and a hydrazine propulsion system to provide thrust for orbit insertion, as well as station keeping and momentum management.

The configuration design as shown in figure 2.1-2 separates the spacecraft bus and instrument modules to (1) minimize cost and complexity associated in the development, design, and qualification of the instruments, and (2) provide opportunity to incorporate a cost competitive off-the-shelf spacecraft bus. Total mass of each of the four observatories is about 2220 kg and is within the capabilities of an Atlas V or Delta-IV class vehicle for a dual launch.

2.2 Launch Vehicle Configuration

The Constellation-X satellites will be launched, two at a time, on Atlas V or Delta IV launch vehicles. These vehicles are part of the Evolved Expendable Launch Vehicle (EELV) program which is a planned evolution of the Atlas and Delta family of launch vehicles to capability in excess of the current Atlas and Delta family. Both the Atlas V and Delta IV launch vehicles are planned to be operational by FY03.

Considering payload mass, and volume; and preliminary vehicle performance, and cost estimates; the Atlas V Medium class vehicle using the longest fairing is currently projected as the most optimal vehicle for the Constellation-X. The specific Atlas 551 launch vehicle configuration (see Figure 2.2-1) would use five solid rockets with the Common Core Booster ™ stage. The fairing is 5.4 meters in diameter and 26.5 meters long. It encloses one Centaur Upper Stage engine and dual satellite payloads. The lower satellite of the dual manifest mounts to a Payload Adapter Fitting (PAF) and is surrounded by a standard Dual Payload Carrier. The upper satellite mounts to a PAF that is supported by the Dual Payload Carrier. Preliminary capabilities of the Atlas V Medium class using the 26.5 meter (87’) long fairing are shown in table 2.2-1 as well as launch configuration of the Constellation-X satellites in the fairing as shown in figures 2.2-2 and 2.2-3.

Launch Vehicle supplier feasibility studies are required to focus on specific performance capabilities within the EELV class capable of meeting the Constellation-X mission requirements. Further analysis will be performed to evaluate the full range of EELV class vehicles against performance, cost, and schedule requirements of the mission. Risk analysis will be performed in parallel with all studies.

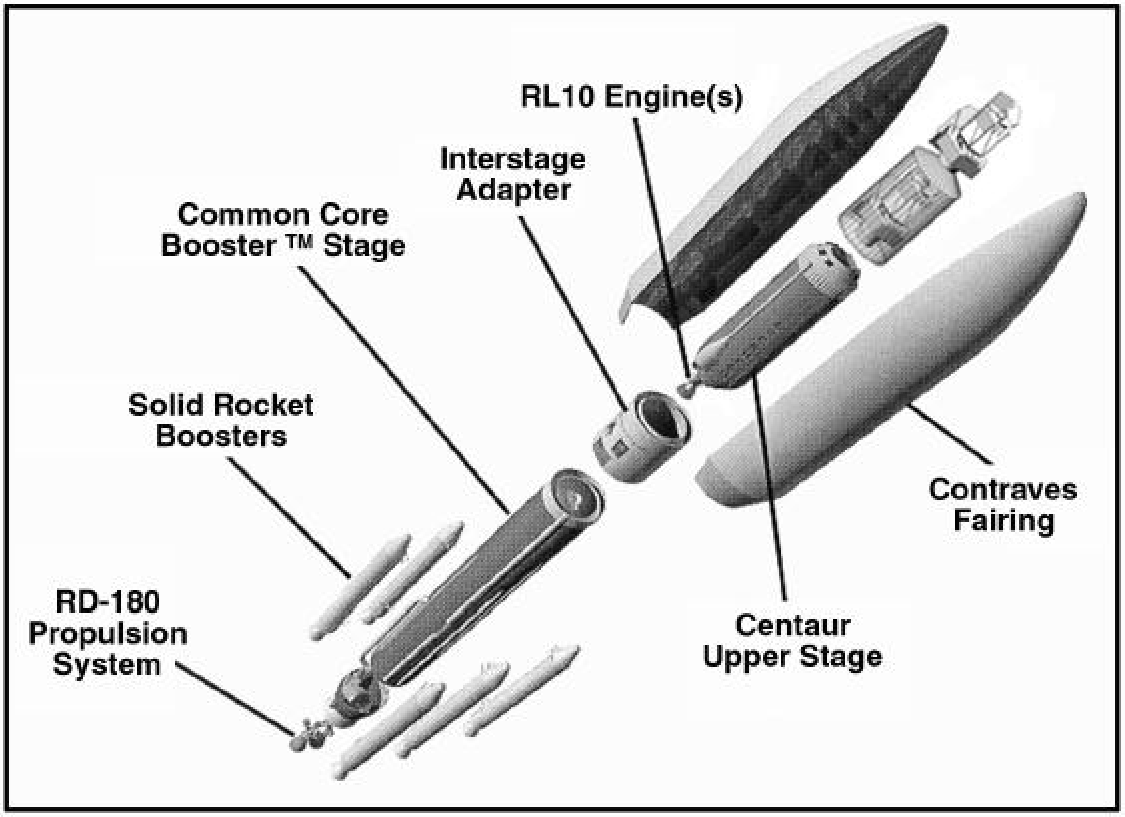


Figure 2.2-1. Atlas V – 500 Series (Medium) Launch Vehicle

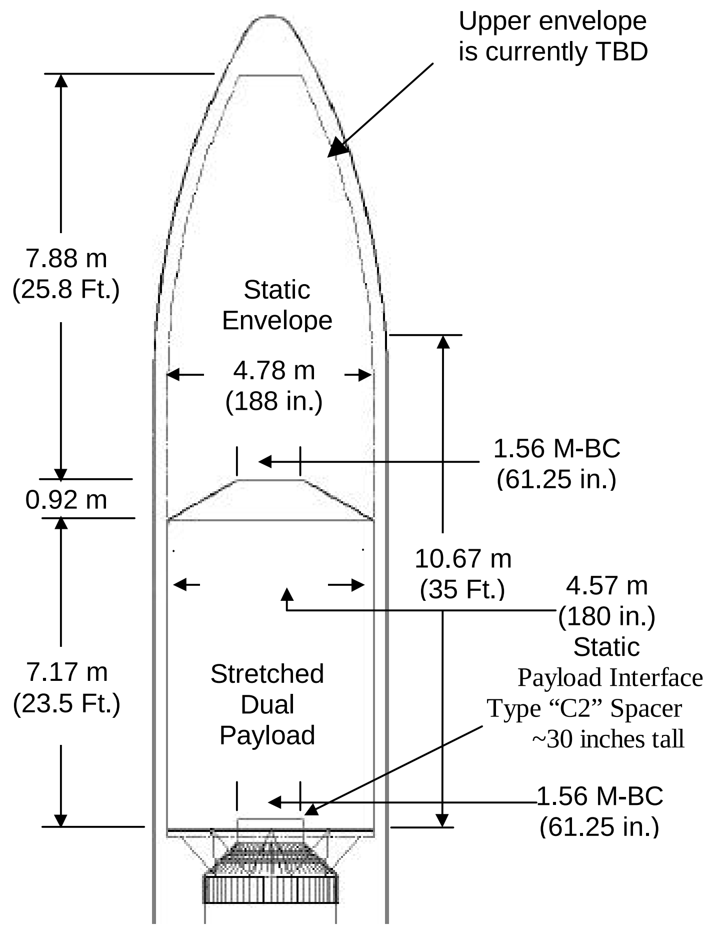


Figure 2.2-2. Figure Atlas V Long PLF 26.5 Meter (87')

Table 2.2-1. Atlas V551 Performance and The Constellation-X Reference Parameters

Atlas V 551 Performance	The Constellation-X Reference Parameters
C3 = -2.26	Final Orbit = L2
Long Payload Fairing (Ref figure 2.2-2)	S/C Volume = 4.675 m diameter x 6.575 m height
Performance Capability = 6,650 kg	Mass (with propellant) = 2,664 kg x 2 S/C = 5,328 kg
Confidence level = 3 sigma	

Note: Performance capability provided by Lockheed Martin and does not include the payload carrier mass of 1000 kg, payload spacers, adapters, separation systems, and other mission unique equipment. Analysis of these items must be performed in coordination with the KSC ELV MIB and the Launch Service Provider.

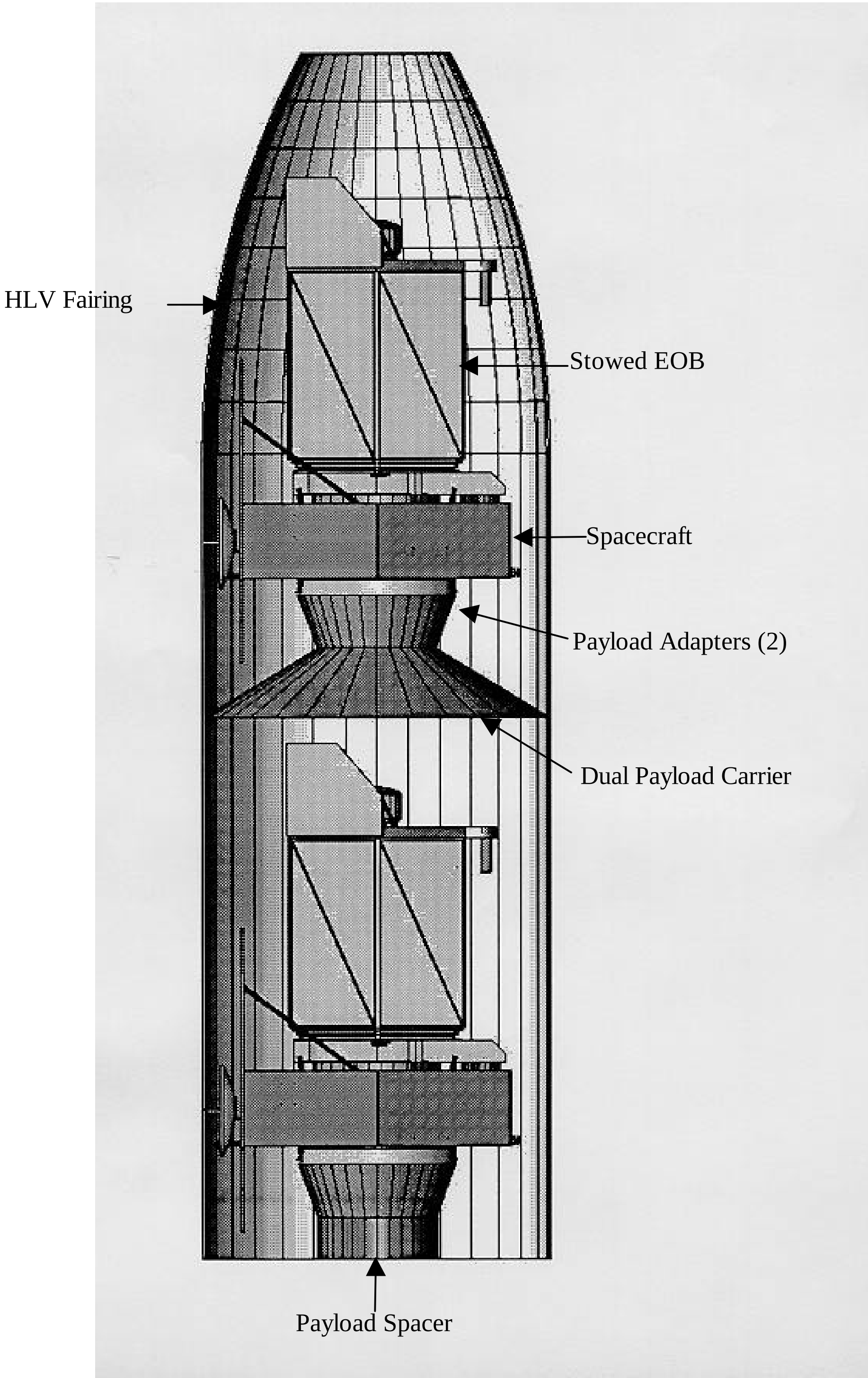


Figure 2.2-3. The Constellation-X Spacecraft in Atlas V Long 26.5 meter (87') Fairing: Dual Manifest Configuration

Note: The figure depicts two spacecrafts inside the Atlas V Long 26.5 meter (87') fairing in the dual payload configuration. Spacecraft to Fairing interface dimensions and tolerances to be evaluated as part of a future feasibility.

2.3 Mission Orbit

The Constellation-X orbit trade off studies have resulted in the selection of a Lissajous Orbit at the L2 - Sun-Earth Libration Point. This orbit has the fewest observing constraints, optimum thermal environment, the lowest radiation environment, and simplified spacecraft communications and operations.

For two celestial bodies in mutual revolution such as the Sun and the Earth, there are five points such that an object placed at any one of them will remain in the same place in the Ecliptic Plane, as shown in figure 2.3-1. These are known as Lagrange Points or Libration Points. There are three points L1, L2, and L3 on the line connecting the Sun and the Earth. There are two more points L4 and L5, which form equilateral triangles with the Sun and the Earth in the Ecliptic Plane. A satellite placed at Lagrange Point remains essentially in the same position relative to the Earth and the Sun. The L2 point is about 1.5 million km anti-sunward from the Earth.

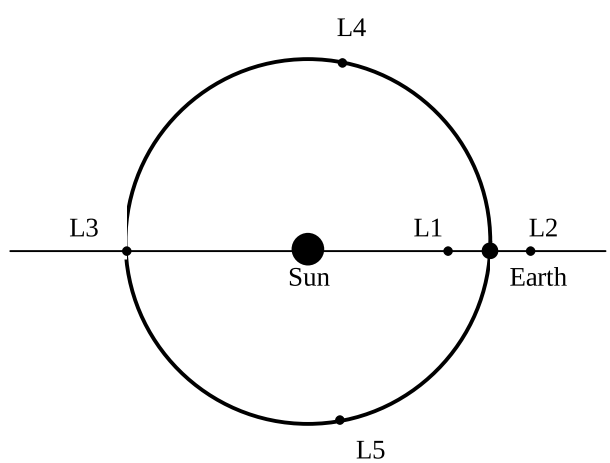


Figure 2.3-1. Lagrange Points of the Sun-Earth

A satellite placed **near** the Libration Point generally stays nearby with a quasi-periodic orbit in all the three reference planes. Such an orbit is known as a Lissajous Orbit. The Constellation-X satellites will revolve with approximately a six-month period in a Lissajous Orbit around L2, at a distance of approximately 300,000 km. This distance ensures that the satellite is always in sunlight. The satellites must also avoid collision in operation. This can be easily accomplished by small station keeping maneuvers that will be performed throughout the mission.

There are three methods to insert the satellite into the orbit. The first approach is by direct insertion. This approach consumes a lot of propulsion compared to others. The second approach uses Lunar Swingby. This approach saves propulsion, however limits the launch opportunities to once a lunar month. The third approach, baselined for the reference configuration, is to use phasing loops in Lunar Swingby. This approach increases the launch opportunities and further decreases the propulsion requirements. Table 2.3-1 shows the characteristics of this approach. Figure 2.3-2 shows the orbit trajectory with this approach.

Table 2.3-1. Launch and Orbit Insertion Characteristics

Launch Opportunities	~ 14 days/month
Daily Launch Window	20 minutes
Launch Vehicle C3*	-2.6 km^2/sec^2
Time from Launch to First Required Maneuver	~ 8 days
Time from Launch to L2	130 days

*C3 is double the combined potential and kinetic energy per unit mass at Thrust Termination Injection point of the launch vehicle (TTI). Smaller values of C3 yield a larger payload capability.
Note: The above characteristics are for Lunar Swingby with Phasing Loops Trajectory with 3 1/2 loops.

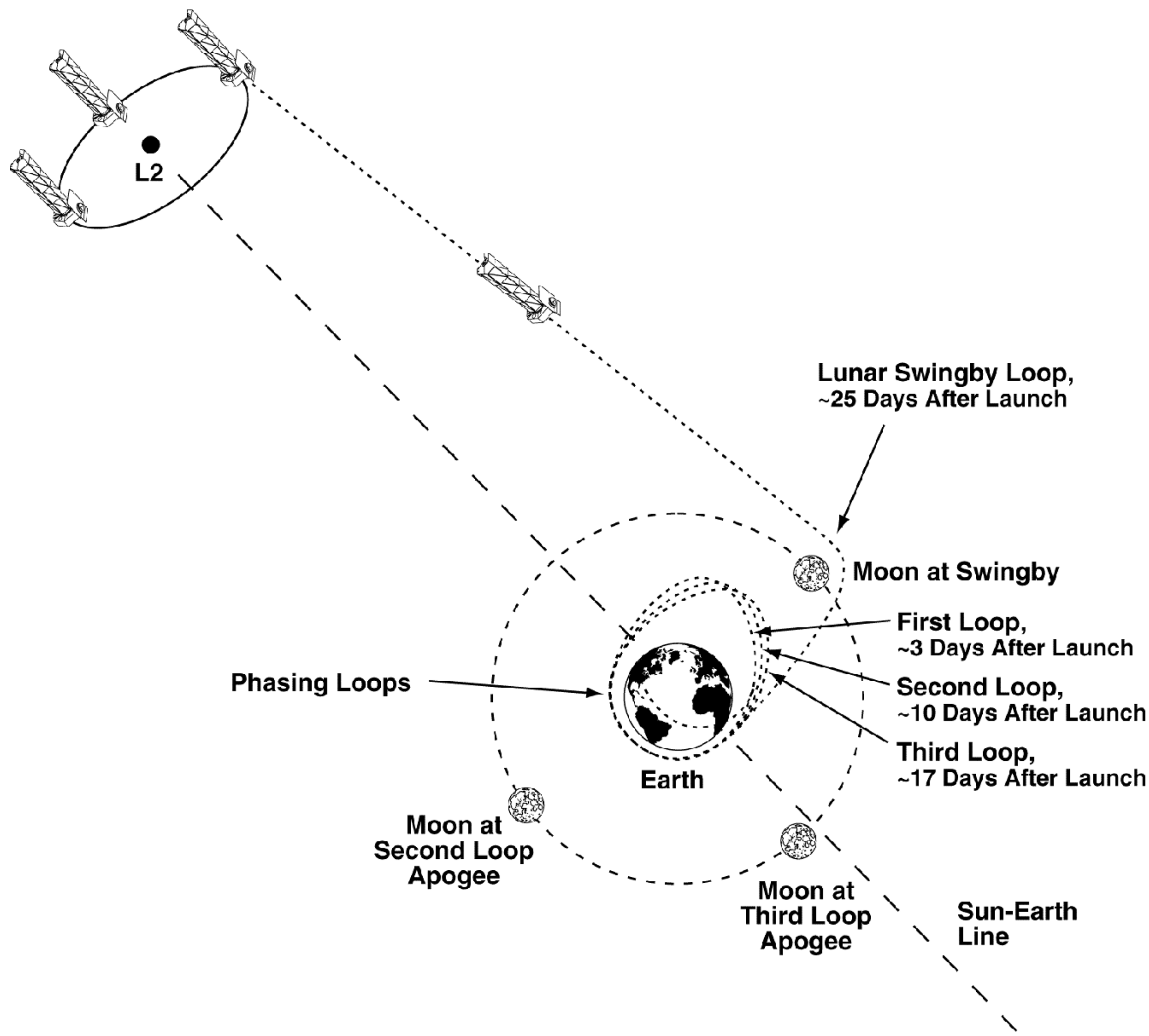


Figure 2.3-2. Trajectory with Phasing Loops and Lunar Swingby

The Constellation-X spacecraft will be placed in a highly eccentric phasing loop orbit by the launch vehicle. A maneuver of approximately 30 meters/second will be performed roughly centered on phasing loop perigee, using the spacecraft propulsion system, to send the spacecraft to a Lunar encounter. The timing and geometry of the Lunar encounter will be chosen to allow the spacecraft to be inserted into the L2 Lissajous orbit with little or no maneuver required from the spacecraft propulsion system.

The spacecraft propulsion system will be required to provide a velocity change estimated at 118 meters/second for a 5-year mission at L2; this amount of velocity change includes all the maneuvers including error corrections and L2 station keeping. The thrusters will need to be sized so that the finite burn losses for the 30-meter/second maneuver are within the limits of the spacecraft fuel capacity. The only relative navigation requirement for the constellation of spacecraft is collision avoidance. At L2, orbit determination will be done with range and range rate tracking data.

2.4 Flight System Properties

The following subsections document the technical resources of satellites necessary for the mission accomplishment.

2.4.1 Mass Summary

The mass estimate for each satellite and launch vehicle is shown in table 2.4-1. The mass of the scientific instrument module is estimated as 1308 kg, and the wet spacecraft bus is estimated at 774 kg.

Table 2.4-1. The Constellation-X Mass Estimate

Item	Mass in (kg)	Qty.	Total Mass (kg)
Instrument Module			
Mirror (1.6 m) with grating	753	1	753
CCD	20	1	20
HXT Optics (0.4 m)	29	3	87
HXT Detectors	11	3	33
Calorimeter with ADR	33	1	33
Cryo System	90	1	90
EOB	292	1	292
Subtotal			**1308**
Spacecraft Bus			
Structure	175	1	175
Mechanisms	7	1	7
Power	122	1	122
Thermal	17	1	17
Propulsion Hrdwr	35	1	35
Attitude Control	73	1	73
C&DH	7	1	7
Communications	38	1	38
Integration Mtrls	120	1	120
Propellent, etc.	180	1	180
Subtotal			**774**
Total Per Satellite			**2082**

Table 2.4-2. Launch Mass Estimate

Item	Mass in (kg)	Qty.	Total Mass (kg)
Satellites	2082	2	4164
Dual Payload Carrier	1000	1	1000
Spacer and Adapter	500	lot	500
Estimated Launch Mass			5664
Project Margin			986
Launch Mass (L2) Vehicle Performance			6650

2.4.2 Power Summary

The power estimate for each satellite is shown in table 2.4-3. These estimates are based on the latest information available and eventually subject to change. The total peak power estimate assumes all subsystems and instruments consume the peak power at the same time and is, therefore conservative.

Table 2.4-3. Power Estimate

Item	Average Power Watts	Peak Power Watts
Instrument Module		
X-ray Calorimeter with ADR	47	79
Gratings	14	14
HXT(Total three)	8	8
Thermal	329	379
Cryo-cooler	100	100
Subtotal	**498**	**580**
Spacecraft Bus		
Communications	10	134
C&DH	44	44
Attitude Control	160	240
Propulsion	41	90
EPS	36	36
Thermal	25	25
Subtotal	**316**	**569**
Total Per Satellite	**814**	**1149**

2.4.3 Command & Telemetry Summaries

The satellite collects science data at the average rate of 48 kbps.. It also collects housekeeping data at the average rate of 2 kbps. It processes command data at the average rate of 2 kbps.

Table 2.4-4. Average Data Rates

Description	Housekeeping (kbps)	Science (kbps)
X-ray Calorimeter		33.0
CCD		11.5
HXT(3)		3.5
Spacecraft	2.0	0
Total	**2.0**	**48.0**

3.0 Instrument Module

This section describes the Instrument Module of a single Constellation-X satellite. There are two major components with associated subsystems.

Spectroscopy X-ray Telescope (SXT)

- SXT Mirror Assembly
- Reflective Grating Assembly/CCD Detector Spectrometer
- X-ray Calorimeter/Cryo System

Hard X-ray Telescope (HXT)

- Multilayered X-ray Optics
- CdZnTe Detector

On each of the four satellites making up the Constellation X observatory, there are three instruments that together cover the X-ray energy range from 0.25 keV to 40 keV. Two of these instruments, the CCD/Gratings and the X-ray Calorimeter work with the SXT optics covering the energy band below 10 keV. The CCD and the Gratings work as one unit even though they are at two different physical locations. The gratings are located on the SXT between the mounting ring and the post-collimator (Figure 3.2-4). X-rays are dispersed off the gratings onto the CCD which is located at the spectroscopic focus of the SXT on the deployable detector bench (see Figure 3.2-1 depicting Rowland Circle). The X-ray calorimeter is located at the focus of the SXT, 10 meters from the node of the SXT. The X-ray calorimeter and its cryogenic cooling system are located on the deployable detector bench. The HXT, or Hard X-ray Telescope, really consists of three identical and independent units, each with its own optics and detector system. The energy range of the HXT covers from 6 keV to 40 keV. The three sets of HXT optics are mounted on the optics bench whereas the detectors are mounted on the detector bench with the detector systems of the other two instruments. The optics bench and the detector bench form the two ends of the Extendible Optical Bench (EOB).

The physical arrangement of the main elements in the Instrument Module is shown in figure 2.1-2 of the previous section. The Instrument Module is attached to the spacecraft bus through kinematic mounts. A key feature of the design is that the module can be integrated separately from the spacecraft and, in fact, with the appropriate Ground Support Equipment (GSE), much of its testing can also be accomplished independently. This approach yields programmatic and technical simplifications as well as cost reduction.

3.1 Spectroscopy X-ray Telescope Mirror Assembly (SXT)

It is possible to design a reflective system that will focus X-rays from a point source into a sharply defined point on the image plane and do so with acceptable efficiency. The SXT Mirror Assembly is a grazing incidence X-ray telescope characterized by moderate resolution, lightweight, large area, and comparatively broad bandwidth (range of energy response).

The baseline SXT Mirror Assembly consists of 70 concentric reflectors having a common focus. The front (entrance) element is parabolic in shape and the rear (exit) element is hyperbolic in shape. This arrangement is called a Wolter Type I mirror or, more commonly, just a Wolter I mirror. Figure 3.1-1 is a representative pictorial of a Wolter I mirror assembly concept.

The Mirror Assembly is kinematically mounted to the EOB. This minimizes deformations within the mirror assembly due to external forces. Figure 3.1-2 shows the kinematic mounts and figure 3.1-3 shows an exploded view of both the entire Instrument Module.

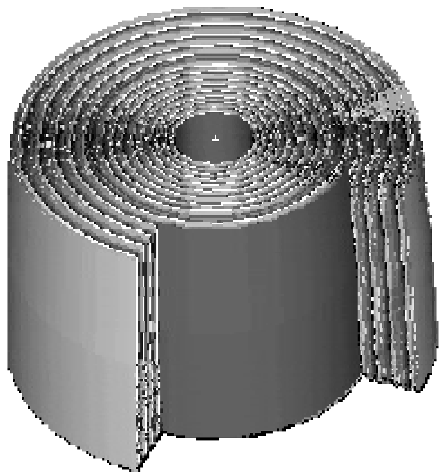


Figure 3.1-1. Wolter I Mirror Assembly Concept

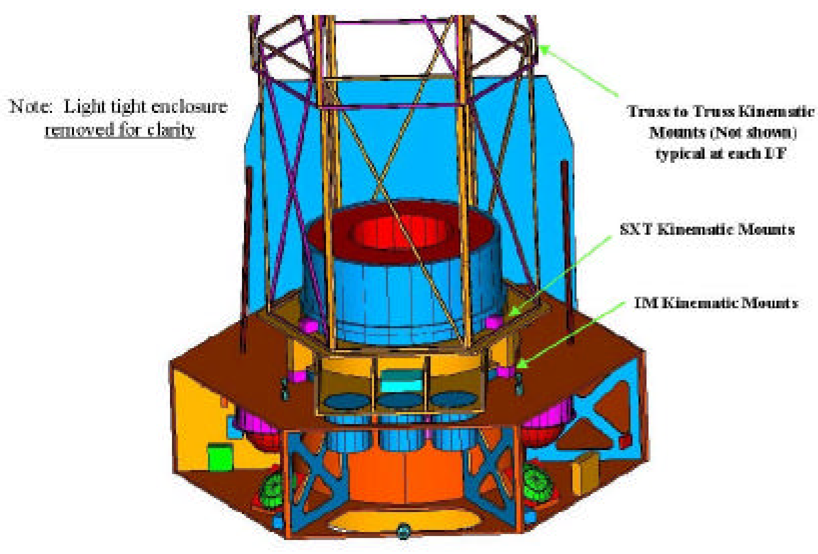
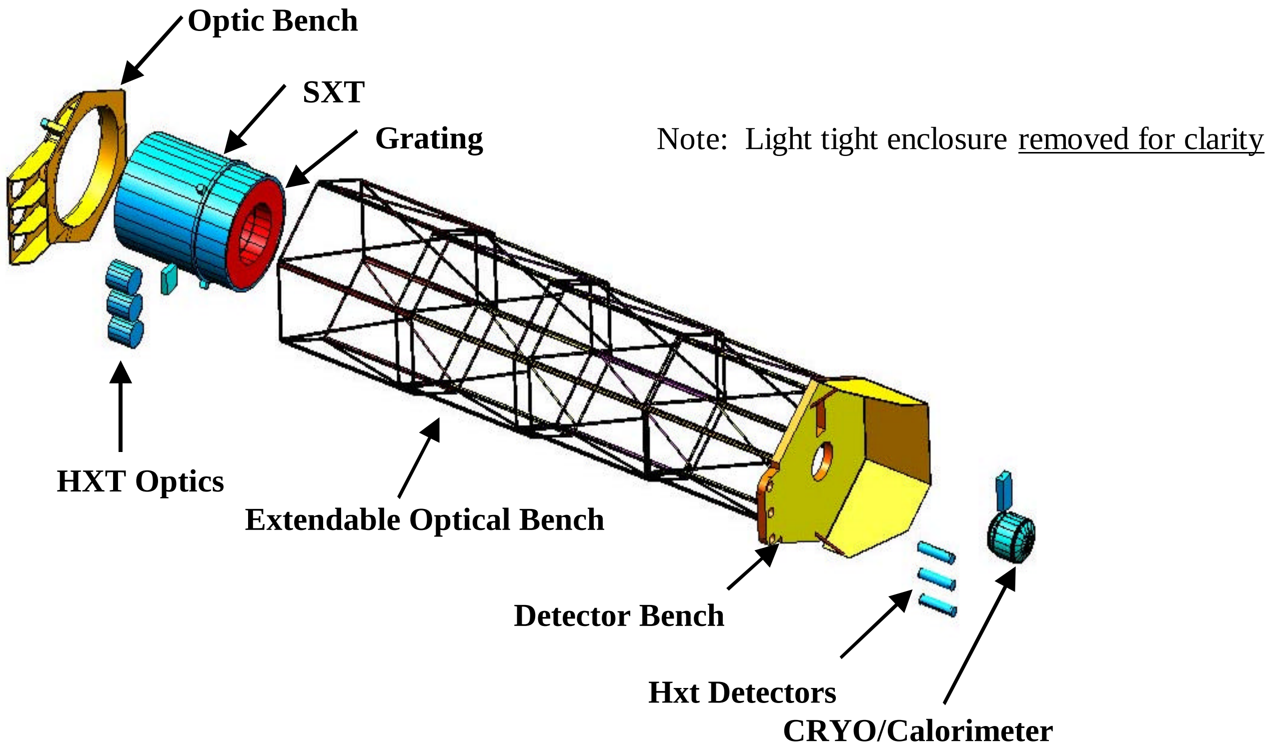


Figure 3.1-2. Kinematic Mounting



Exploded View (Top/Side) - IM

Figure 3.1-3. Instrument Module

3.1.2 SXT Size and Mass [1]

Figure 3.1-4 shows the envelope of the SXT Mirror Assembly and gratings. The envelope shown includes both the thermal precollimator and postcollimator as well as axial space allocated to the gratings.

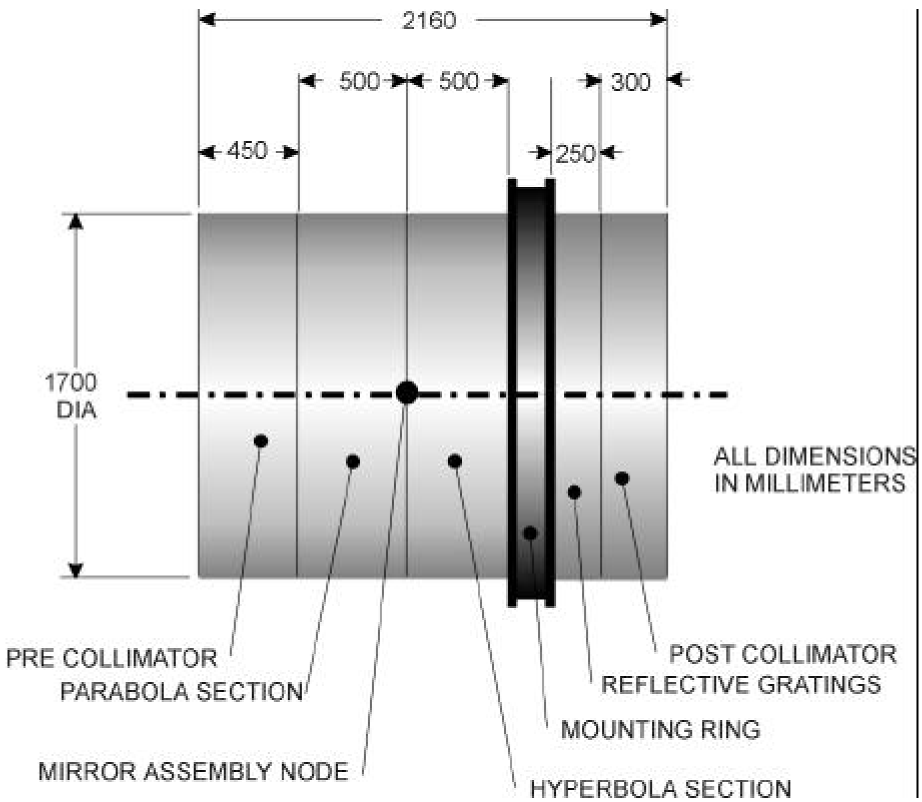


Figure 3.1-4. SXT Mirror Assembly

[1] At present, two different mirror technologies are currently being considered. Both use replication off precision mandrels to obtain optical figure and surface finish. One approach uses full shells of the same general type as the XMM mirror, but much lighter. The other approach uses segments which are then aligned together to approximate a full shell. Fortunately, the overall response characteristics by definition will be the same and the mechanical envelopes are expected to be very similar. The full shell mirror assembly is expected to weigh more than a segment mirror. This document uses the baseline mass for a full shell mirror.

The current estimated mass of the SXT Mirror Assembly including the gratings and the precollimators is (~753 kg). Table 3.1-1 shows the breakout of estimated mass.

Table 3.1-1. Mirror Assembly Mass Breakout

Item	**Mass in kg**		**Comment**
	Assembly	Item	
Optics (70 shells)	417.6	417.6	
Mirror structure and rear aperture plate	171.5		
Ribs		54.6	
Stiffening rings		37.2	
Mirror flanges		37.9	
Telescope interface		25.1	
Inner and outer cylinders		16.7	
Thermal control	47.9		
Precollimator		24.0	
Postcollimator		16.0	
Thermal control hardware		8.0	
Contamination Doors	20.6		
Front door		8.8	
Rear door		8.8	
Hinges and actuators		2.9	
Apertures and baffles	22.0		
X-ray baffles		7.3	
Front aperture plate		14.7	
Sub-total	679.6		
Gratings	73.5	73.5	
Total Mirror Assembly Mass	753.1		

3.1.3 Collecting Area Vs. Energy

Mirror assembly collection characteristics are defined by an area vs. energy curve. The curve for the Constellation-X is shown in figure 3.1-5. This figure shows the total area of four SXT Mirror Assemblies. The area of a single SXT Mirror Assembly is, therefore, one quarter of the value shown on the curve.

The curve shown is the total mirror curve only and does not include detector quantum efficiency or grating efficiency.

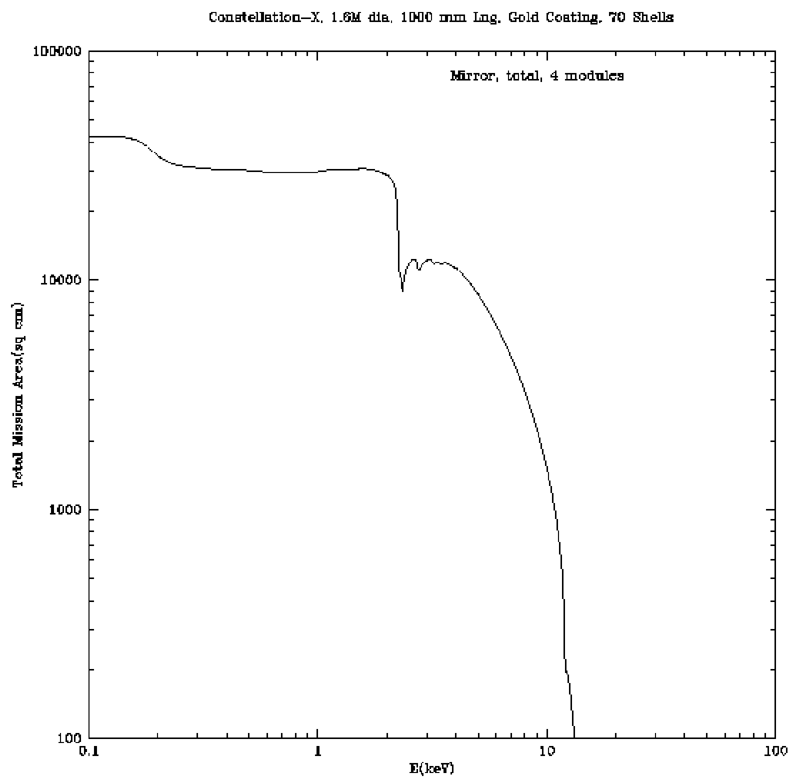


Figure 3.1-5. SXT Net Collecting Area vs. Energy

3.1.4 SXT Focal Length

The nominal focal length of the SXT mirror assembly is 10 meters. This is taken as the distance on the mirror assembly axis from the mirror assembly node to the on-axis focus. By convention, the SXT Mirror Axis is taken to be the geometric center line of the concentric mirror shells. This definition assumes that the individual shells were fabricated to close tolerance and that they have been assembled properly so that rays reaching each shell come to a common mirror focus. This is also the axis of best focus by which we mean that if we illuminate the mirror with parallel light, the sharpest spot will be obtained when the incoming light is parallel with the mirror axis. However, as a practical matter, for the Constellation-X mirror assembly, it is difficult to find the axis of best response with much precision. This is because we can change the mirror pointing by tens of arc seconds (see Figure 3.1-7) and not significantly affect its resolution or find the "sweet spot". Thus, it will be convenient to use the mirror geometric axis as the optical axis.

The node of the mirror is the intersection of the plane where the forward parabolas intersect the rear hyperbolas with the mirror center line. The node thus lies on the centerline and is in line with parabola-hyperbola intersection. An important property of the SXT mirror assembly is that it behaves as a thin lens and, therefore, rotation of the mirror about its node does not produce image motion – at least for small excursions – in our case up to 2 arc minutes or so. During initial alignment and assembly, individual mirrors may be moved forward or back along the axis to accommodate minor figure errors. The result of this is that the overall effective node of the mirror assembly is broadened along the mirror assembly axis, but the thin lens approximation is still valid.

Figure 3.1-6 illustrates the definition of focal length for both the SXT and the HXT. In the case of the HXT, the mirrors may be cone approximations of the parabolic and hyperbolic surfaces, but the definition of focal length is still the same. The SXT and HXT have the same 10 meter focal length and therefore their plate scales are all the same.

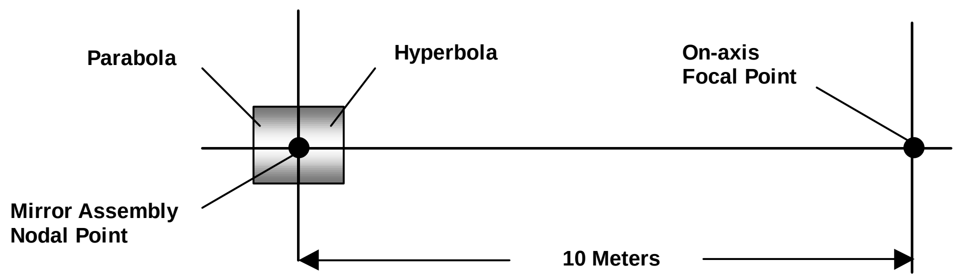


Figure 3.1-6. Definition of SXT and HXT Focal Length

The SXT Collecting Area vs. Energy curves shown in figure 3.1-5 are for an optical design having a focal length of 10 meters. The average plate scale is 48.5 microns per arc second. The mirror assembly half power diameter (HPD) is 15 arc seconds in angle so that on the focal plane, the response to a point source at infinity (due to the mirror assembly alone) is 0.73 millimeters HPD. This is often referred to as the point source spot size although in practice, there will be other effects that tend to spread the spot further. It is the best that we can do with the SXT Mirror Assembly under otherwise ideal conditions. The image produced by a point source at infinity on the image plane is called the Point Spread Function (PSF). In principle, for a fixed point source there is a slightly different PSF at each point in the image plane. Fortunately, on the Constellation-X, the differences are small and generally can be ignored except at the edges of the total field of view.

3.1.5 Resolution

Resolution is defined as the half power diameter of the image produced by an on-axis point source at infinity in the image plane of the SXT. For an ideal mirror assembly, it is the angular diameter centered on the center of the detected point spread function that includes one half of all of the incident X-rays. The SXT resolution of 15 arc seconds Half Power Diameter (HPD) is valid over the entire energy range of 0.25 keV to 10 keV. Note that resolution is defined by the response to a point source.

The response to the point source does not have to be symmetrical about some point and, in general, it is not. HPD is the diameter of the best fit circle in the image plane such that half of the detected photons fall within the circle and half lie outside it. HPD is a very useful definition that is easily understood and calculated with experimental or theoretical data.

Table 3.1-2 shows a representative error budget for a 15 arc second HPD mirror assembly made of full electroformed shells. The error budget components shown are for a complete 70 shell mirror assembly. The "M" values are from an alternate MSFC budget.

Table 3.1-2 is also an error budget for a segmented mirror. What is important is that both mirror budgets are consistent with the Constellation-X resolution requirements.

Table 3.1-2. Representative SXT Mirror Assembly Error Budget

Item	LEV 1	LEV 2	LEV 3	LEV 4	LEV 5	Comment
Overall	15.0					
Margin	2.2					
Initial On-Orbit		12.0				
Mirror module fabrication			11.2			M = 10.0
Mirror fabrication & metrology				10.0		M = 7.8
Mandrel					7.1	
Replication/release					7.1	
Assembly & alignment				5.0		M = 6.3
Deployment/launch shifts			4.1			
OB Deployment				2.0		M = 3.0
Mirror assembly launch shifts				3.6		
Alignment					2.0	M = 2.0
Optics					3.0	M = 3.0
Short term variations (during measuring)		4.1				M = 4.0
Mirror thermal			3.0			M = 2.0
OB thermal			2.0			M = 2.0
Vibration			2.0			M = 2.0
Long term variations		5.4				M = 3.6
Mirror optics instability			5.0			M = 3.0
Mirror/focal plane alignment instability			2.0			M = 2.0
Image reconstruction		5.6				M = 3.6
Aspect determination			5.0			M = 3.0
Mirror/focal plane dynamic errors			2.5			M = 2.0

3.1.6 Off-Axis Response

Figure 3.1-7 shows the SXT Mirror Assembly response with respect to off-axis angle. Curves are shown for the inner shell (shell 70) and for the outer shell (shell 1) and include vignetting effects. The upper set of curves shows the relative throughput of the mirror assembly versus off-axis angle. The lower set of curves defines the mirror resolution with respect to off-axis angle. These curves show geometric effects and are not energy dependent. The curves show geometric effects only.

The effects shown are only those due to off-axis viewing and must be combined with the on-axis mirror resolution to obtain overall resolution. Also the data are only for the inner and outer shells and therefore only define the extremes of mirror assembly performance.

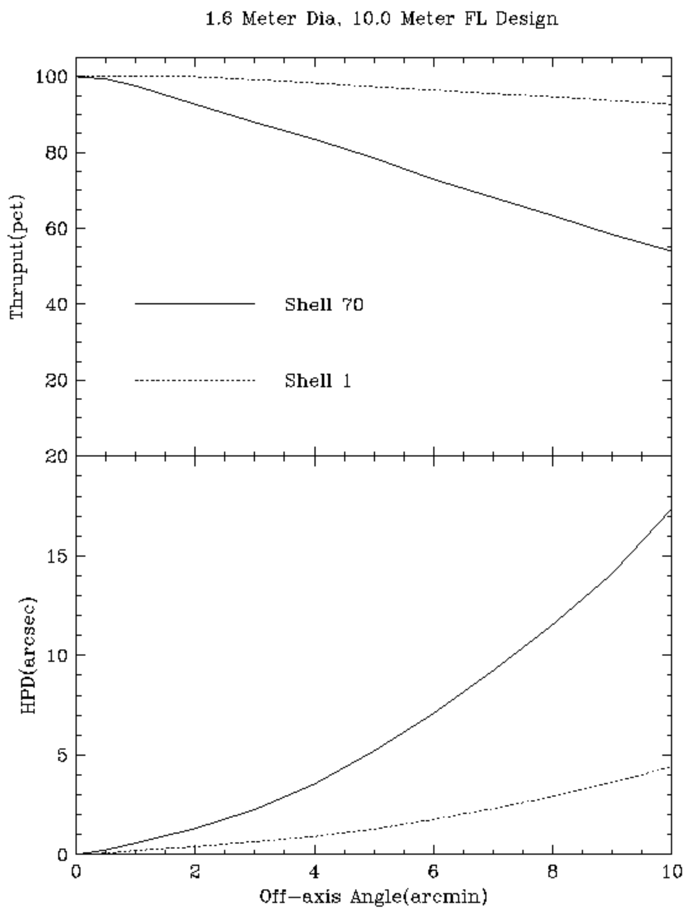


Figure 3.1-7. Off-Axis Response of SXT Mirrors

Figure 3.1-7 shows that the throughput, and therefore the field of view, of the mirror assembly falls off gradually with the effect being greatest for the outer shells. This means that the definition of the Field of View (FOV) of the Constellation-X mirror assembly is somewhat arbitrary. One definition comes from a constraint that the off-axis effect reduce the throughput by no more than 20 per cent and produce a resolution degradation of no more than 5 arc seconds. For Shell 1, this results in a field of view of about 10 arc minutes in diameter. Using this, or any similar definition, the overall mirror will have a slightly greater field of view when the other shells are added to the analysis. However, Shell 1 and its immediate neighbors dominate the low energy response of the SXT Mirror Assembly, so the value of 10 arc minutes diameter, while approximate, is both limiting and useful. (Provided the 20-5 example definition is acceptable.)

At the focal plane, 10 arc minutes is approximately 3.0 centimeters. System stability must be such that the detector focal plane (window, array, etc.) when aligned to the telescope will always stay within the field of view. Otherwise focal plane position adjustment of the detectors will be required. If the field of view becomes smaller, the stability and alignment requirement will become more stringent.

3.1.7 Temperature Control

The Constellation-X SXT Mirror assembly will almost certainly be aligned on the ground in a vertical orientation to minimize the effect of mirror element deflections due to gravity. This alignment will, of necessity, use visible or UV light. Elements will be placed in proper alignment and one way or another will be bonded into proper position. The elements themselves, whether full shells or segments, will be made to have the proper shape at room temperature (~20° C). Assembly and alignment will also take place at room temperature for both convenience and minimum cost.

Because the SXT Mirror Assembly is fabricated and aligned at room temperature, it is important that it be operated at room temperature on orbit. This coupled with tight temperature requirements on the reflective gratings means that a tight temperature control system for the SXT is required. The SXT thermal control system will use several approaches. These are:

- Interface heaters
- Thermal pre and postcollimators (actively controlled)
- Internal heaters (possibly)

3.1.7.1 General Thermal Requirements

Environmental conditions during mirror buildup and alignment will determine the overall temperature control requirements for the SXT Mirror Assembly. The operating temperature on orbit is essentially the same as the ambient temperature during mirror assembly buildup. This will nominally be room temperature. Trades exist between the material of the mirror reflector elements and the degree of control required. Requirements for a given design (including material specification) will limit overall temperature excursions as well as allowable gradients within the mirror assembly. These requirements are mainly driven by the overall 15 arc seconds HPD resolution requirement and the material of the mirror assembly. The format of the general requirements is:

- Mirror Assembly Temperature = 20 ° C ± 0.5° (TBR)

- Maximum Radial Gradient (outside to inside) = 4° C
- Maximum Axial Gradient (front to back) = 0.6 °C
- Maximum Diametral Gradient (side to side) = 1°C

These values are representative of the requirements for an assembly of nickel mirrors. Use of other materials will result in different requirements. In general, materials having lower coefficients of thermal expansion will have less stringent requirements.

3.1.7.2 Precollimator

Precollimators have been successfully used on *Einstein*, *ROSAT*, and now *Chandra* to control heat flow through the main mirror. A precollimator will be used on the Constellation-X SXT Mirror Assembly as well. On both Einstein and Chandra, the X-ray Mirror Assembly had a small number of elements and it was possible to construct aperture plates and align them with the mirror openings. This straightforward approach is not possible for the Constellation-X mirror assemblies because of the large number of mirror elements and the small space between adjacent elements. A concept for the Constellation-X precollimator is shown in figure 3.1-8.

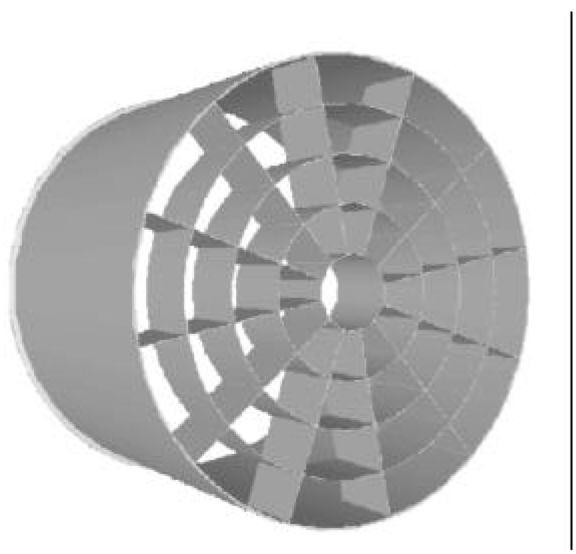


Figure 3.1-8. SXT Thermal Precollimator Concept

3.1.7.3 Postcollimator

The postcollimator provides thermal isolation between the exit surface of the SXT and the optical bench and focal plane and thus helps control both absolute mirror temperature and gradients (particularly axial gradients) within the mirror assembly.

The reflective gratings will be mounted to the exit (rear) face of the SXT and closely aligned to the mirror assembly itself. The gratings require tight temperature control and, like the mirror, are susceptible to axial gradients. The postcollimator design accommodates this tight temperature control. Viewing factors are more important here than with the precollimator because the vignetting (blocking) of the rays reflected off the gratings must be kept to a minimum. The baseline postcollimator configuration is similar to that of the precollimator.

3.2 CCD/Gratings Overview

The CCD camera and reflection grating spectrometer cover the energy range 0.25 keV to 2 keV (5 to 50 angstroms). For this energy range the resolving power of the CCD/Grating increases with decreasing energy. At the lowest energy and longest wavelength in this bandpass, (50 angstroms), the resolving power is 1000. The array of lightweight reflection gratings is mounted at grazing incidence to the beam, near the exit of the Spectroscopy X-ray Telescope (see Figure 3.2-4). The grating array covers only the outer 28 of the total 70 of the telescope shells and the gratings are spaced so as to “pick off” only approximately 57% of the light passing through. The remaining 43% of the photons from the outer shells pass undeflected to the calorimeter. The photons picked off by the gratings are dispersed to a strip CCD detectors offset in the dispersion direction at the telescope focal plane. The gratings are all mounted at the same incident angle with respect to the ray passing through the grating center, and they are positioned on a Rowland Circle which also contains the telescope focus and the CCD detectors (see Figure 3.2-1 and Table 3.2-1 which give the dispersion position of the wavelength of the photon).

3.2.1 CCD Accommodations

An array of CCDs is used as a readout device for the grating spectra. The positioning of the CCD and the grating assembly on the Rowland Circle is shown in figure 3.2-1. Information to accommodate the CCD detector is contained in table 3.2-1. The camera assembly dimensions in table 3.2-2 reflect outside dimensions. The outside length of 536 mm corresponds to the focal plane length of 446 mm; the outside width of 268 mm corresponds to the focal plane width of 89.925 mm. The depth dimension is 125 mm. These accommodation parameters assume that the CCD is passively cooled and do not reflect the possible requirement for a door with an associated mechanism, which depends on the outcome of the optical filter technology development. Figure 3.2-2 shows a block diagram of the CCD assembly. The CCD array consists of 6 CCDs for spectroscopy and 2 zero order CCDs for imaging as shown in figure 3.2-3. The layout reflects focal plane envelope dimensions, not package dimensions .

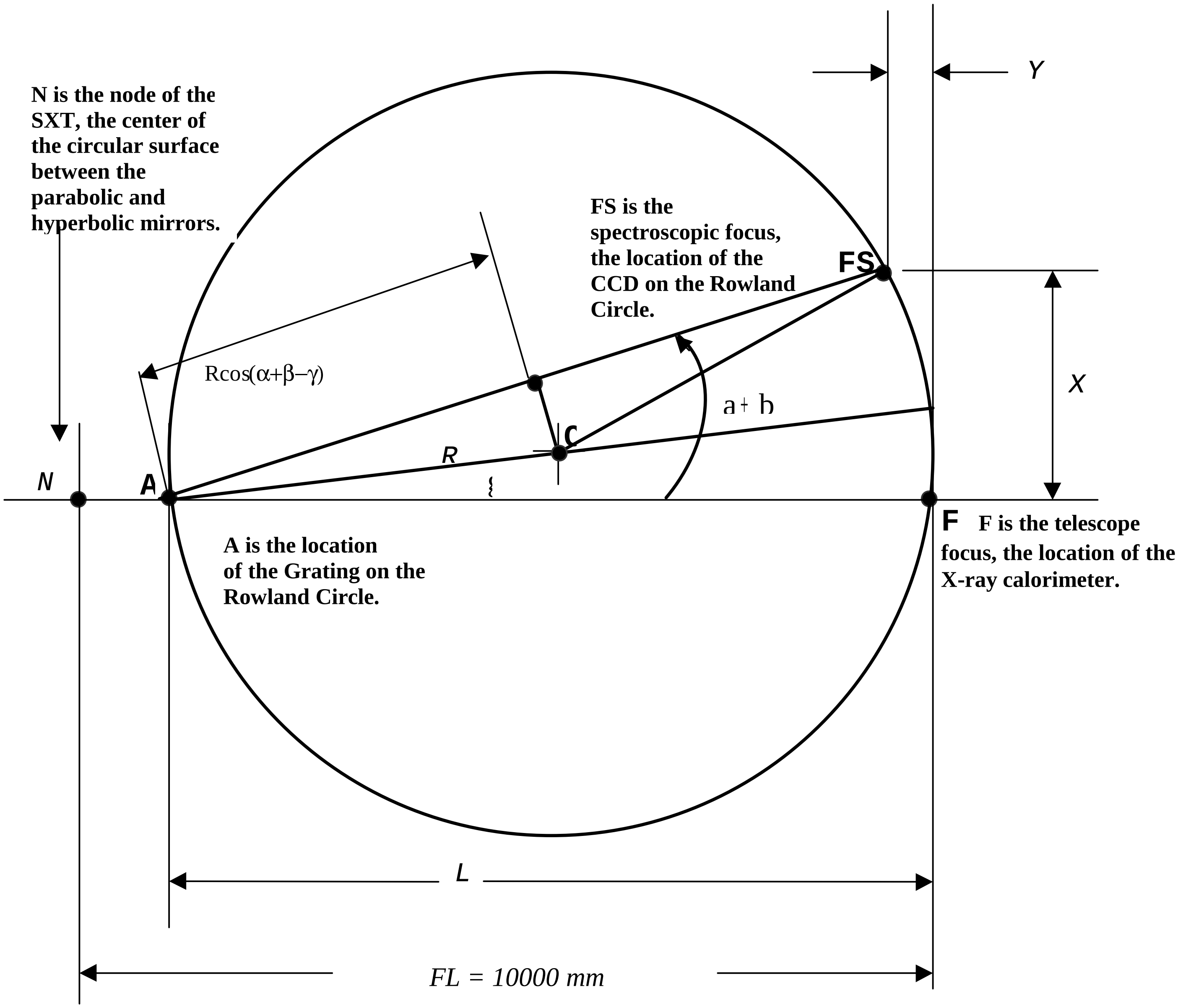


Figure 3.2-1. Rowland Circle

Table 3.2-1. Spectroscopic Displacements for CCD Camera

CCD Locations for 10.0m focal length **L = 9315 mm, R = 4660.97 mm**			
λ(Ång)	β(deg)	X(mm)	Y(mm)
0^{th} order	1.610	523.5	9.2
5.0	1.982	583.9	14.1
10.0	2.295	634.6	18.8
15.0	2.570	679.1	23.4
20.0	2.818	719.2	27.9
25.0	3.046	756.1	32.3
30.0	3.259	790.3	36.7
35.0	3.458	822.4	41.1
40.0	3.646	852.8	45.4
45.0	3.825	881.6	49.7
50.0	3.997	909.1	54.0

Table 3.2-2. CCD Accommodation Information

Subassembly	**Size (m)**	**Mass (kg)**	**Operating Power (watts)**	**Operating Temp. (^{o}C)**
Camera Assembly	0.536 x 0.268 x 0.125			
Array (focal plane)		0.6	0.12	-90 ± 10
Structure		3.6	------	-90 ± 10
Shielding		7.1	------	-90 ± 10
Analog Electronics	0.12 x 0 .1 x 0.3	2.4	5.35	$+10\pm30$
Digital Electronics	0.12x 0.1 x 0.3	2.4	5.95	$+10\pm30$
Power System	0.12 x 0.1 x 0.3	1.8	2.38	$+10\pm30$
Cables*		2.0		
Total		**19.9**	**13.80**	

* Cable mass is assumed to be approximately 10% of total mass”

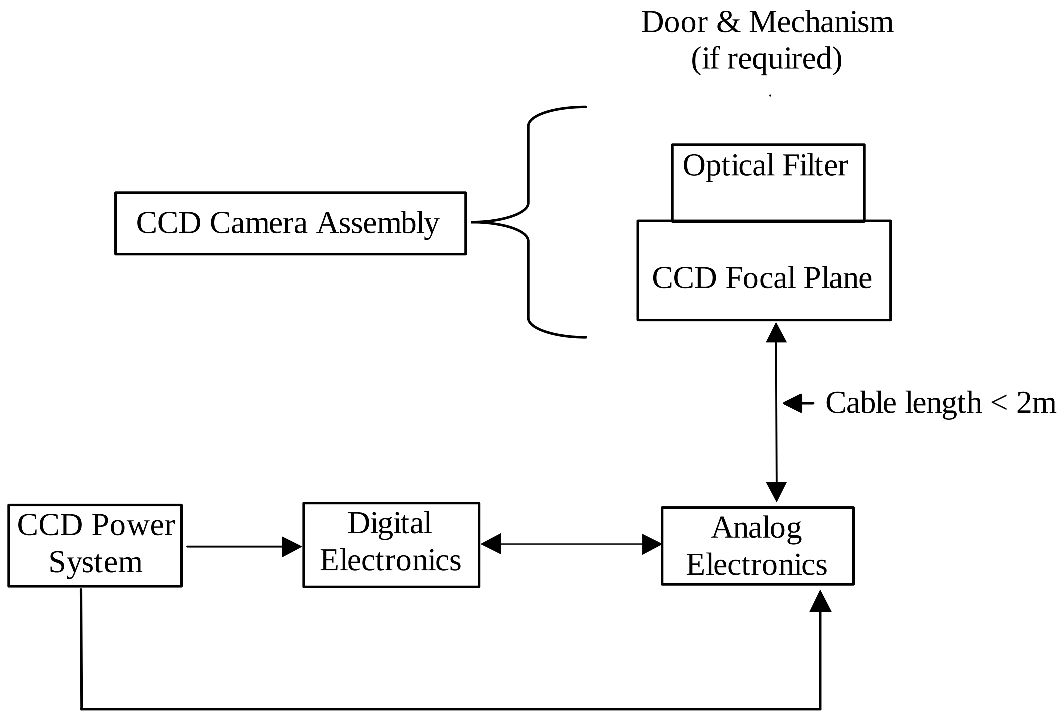


Figure 3.2-2. Block Diagram of the CCD Detector System

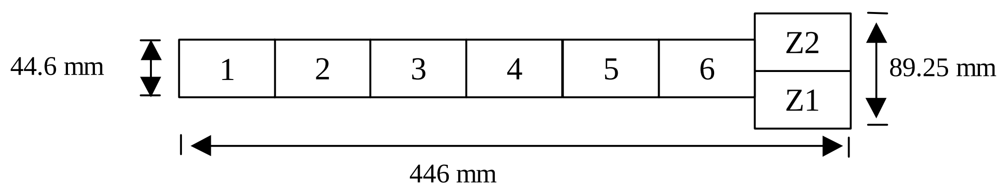


Figure 3.2-3. CCD Focal Plane Layout

3.2.2 Gratings

The gratings cover the outer 28 shells of the SXT. The accommodation requirements in table 3.2-3 assume a 1.6 meter diameter SXT.

Table 3.2-3. Grating Accommodation Requirements

Mass	73.5 kg
Outer Diameter	1.6 m
Inner Diameter	0.9327 m
Length	.25 m
Operating Temperature Range (^{o}C)	20 ±10
Temperature Gradient (along mirror axis)	TBD
Temperature Gradient (radial)	TBD
Alignment Requirement (x,y z,θ_x,θ_y,θ_z)	TBD
Stiffness of structure providing mounting interface with grating	TBD

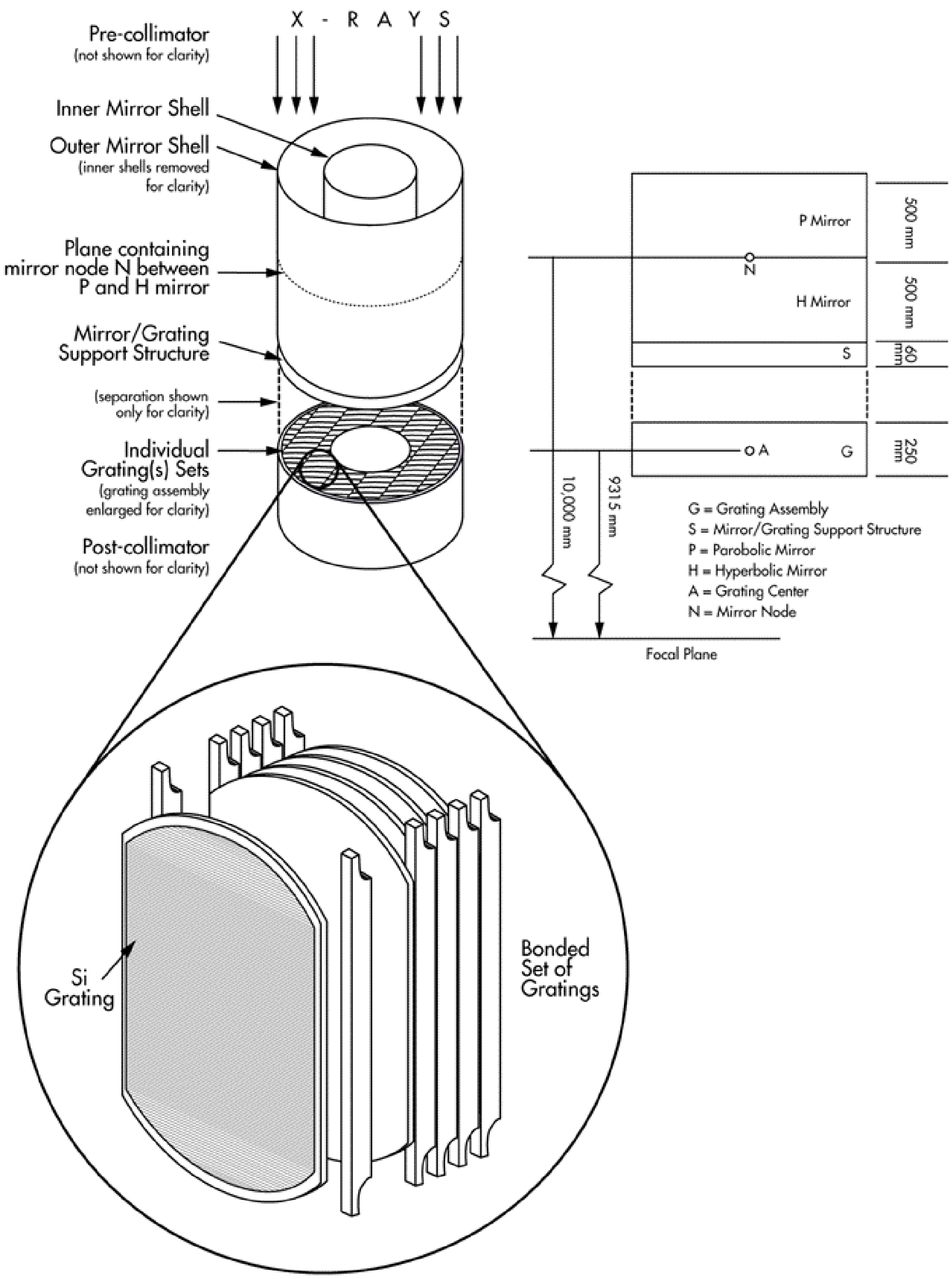


Figure 3.2-4. Grating Layout with Respect to SXT

3.2.3 Data Rate

The daily average science data rate estimated for the CCD (per satellite) is 10.5 kbps. The peak science rate is 300 kbps. The average housekeeping data rate is 1 kbps.

3.2.4 Special Accommodation Requirements: Sensitivities And Concerns For Gratings/CCD System

Table 3.2-4. Special Accommodation Requirements

Magnetic Cleanliness	TBD
EMI/EMC	TBD
Radiation	TBD
Micro-phonics	TBD
Contamination	TBD
Stray Light	TBD

3.2.5 CCD/Grating Performance Specifications

Table 3.2-5. CCD/Grating System Performance Specifications

Energy Resolution	Better than 0.05 A FWHM in first order
Beam Efficiency	$\geq$ 0.2 (TBR)
Mass	$\leq$100 kg (TBR)
Power	$\leq$75 watt (TBR)

3.3 X-ray Calorimeter Overview

Located at the focus of the SXT, the X-ray calorimeter detects with high quantum efficiency X-ray photons in the energy band from .3 keV to 10 keV. The X-ray photons are counted and time-tagged. The energy of these photons is measured with resolution of 2 eV.

The X-ray calorimeter can be thought of as a simple device in which the energy of an X-ray photon is deposited in an absorber and detected in the thermistor (see Figure 3.3-1). The X-ray, by hitting the absorber and knocking an electron loose, starts a process of thermalization. This electron gives up its kinetic energy to heat, which is measured when the thermistor comes to the same temperature as the absorber. The measured temperature increase corresponds to the energy of the incoming X-ray. The thermistor and absorber are connected to the heat sink through a weak thermal link. After a very short time, the thermistor returns to its normal operating temperature.

Several technologies for the thermistor are being considered. One is the super-conducting Transition Edge Sensor (TES) detector system, which is discussed in this section. The other is the semiconductor X-ray calorimeter that is discussed in Appendix A.

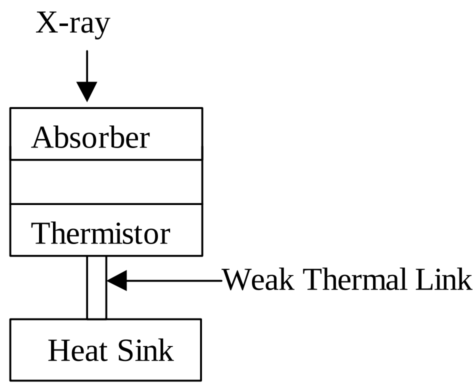


Figure 3.3-1. Functional Diagram of X-ray Calorimeter Detector

3.3.1 X-ray Calorimeter Accommodations

With the 10 m focal length of the SXT, the X-ray calorimeter's requirements of 5.0 arc-sec spatial resolution and 2.5 arc-min field of view can be achieved with a 32 by 32 pixel array of 0.24 mm by 0.24 mm pixels. A block diagram for the super-conducting Transition Edge Sensor (TES) array is shown in figure 3.3-2. At the window to each temperature region there is an optical filter or filters. These filters are thin metalized polymer films. In order to permit close proximity of the first-stage readout SQUIDs to the TES array, both will operate at the same temperature. The optimum operating temperature for the TES array is as close to absolute zero as possible and is estimated to be less than or equal to 50 mK. This operating temperature is one of the parameters varied in the ongoing technology development of the TES calorimeter and the ADR. The optimum operating temperature for the second-stage SQUID arrays will be between 50 mK to 4 K. In the implementation illustrated, each column of first-stage SQUIDs is multiplexed into a single second-stage SQUID series array. The SQUID arrays amplify the signal so that it is well matched to the “room temperature” electronics. The final operating temperature of the SQUID arrays will be determined as the system design of the TES, the ADR and cryo-cooler matures.

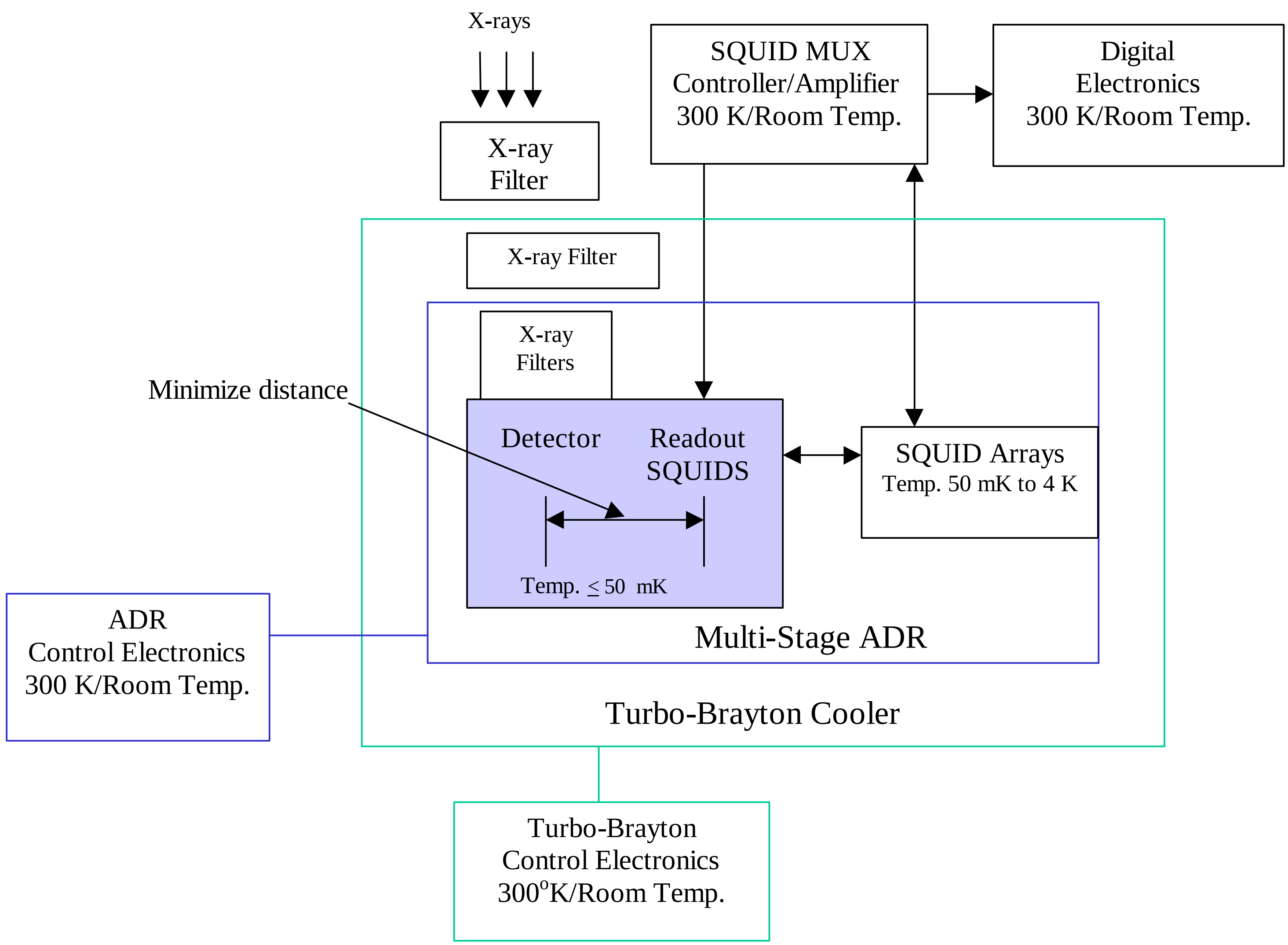


Figure 3.3-2. Block Diagram of X-ray Calorimeter Assembly

Table 3.3-1. Calorimeter Accommodation Assumptions and Requirements

Component	Mass (kg.)	Average Power (Watts)	Dimensions (cm)	Peak Power	Operating Temp. (^oK)	Survival Temp. (^oK)
Detector Assembly*	≤2.0 (TBR)	≤500 nW (TBR)	TBD	≤500 nW (TBR)	≤50 mK (TBR)	≤ 350 K (TBR)
Second-stage SQUID Arrays	≤2.0	≤ 10μW	TBD	≤ 10μW	≤4 K	430 K
Multistage ADR (including shielding)	7.0	TBD	TBD	TBD	TBD(Each stage will be at a diff. T)	TBD
X-ray Filters	1.0	TBD	TBD	TBD	TBD	TBD
SQUID MUX Controller and Amplifier	10.0	40 W (TBR)	TBD	TBD	TBD	TBD
Digital Electronics	7.0	40 W (TBR)	TBD	TBD	TBD	TBD
ADR Electronics	4.0	TBD	TBD	TBD	TBD	TBD
Total (Estimated)	33.0	TBD	TBD	TBD	TBD	TBD

* Detector Assembly contains detector and first-stage readout SQUIDS

Note: At the system level the X-ray calorimeter with ADR average power is carried at 47 watts.

3.3.2 Data Rate

The daily average science data rate for one X-ray Calorimeter instrument is 30 kbps. The peak science data rate per instrument is 1051 kbps. The average housekeeping data rate is 3kbps.

3.3.3 Special Accommodation Requirements: Sensitivities And Concerns

Table 3.3-2. Special Accommodation Requirements

Magnetic Cleanliness	TBD
EMI/EMC	TBD
Radiation	TBD
Micro-phonics/vibration	5 mG (TBR)
Contamination	TBD

3.3.4 X-ray Calorimeter Performance Specifications

Table 3.3-3. X-ray Calorimeter Performance Specifications

Energy Range	0.3 keV or lower to 10 keV or higher
Energy Resolution	2 eV FWHM over 1 keV to 5.9keV band
Quantum Efficiency	≥90% at 1 keV and 5.9 keV ≥50%over 0.3 keV to 10 keV
Spatial Resolution	≤ 5 arc sec
Field of View	≥ 2.5 arc min

3.4 Hard X-ray Telescope (HXT) System

On each of the four satellites there is a Hard X-ray Telescope system consisting of three independent and identical telescopes measuring X-rays in the energy band from 6 keV (or less than 6 keV) to 40 keV (or greater than 40 keV). Each telescope has its own optical and detector assembly. The detector assembly is integrated to the detector bench of the extendable optical bench and the optical assembly of the three HXTs is integrated to the optics bench. The optics bench is then integrated to the satellite bus. The field-of-view requirement for each of the three co-aligned telescopes is eight arc minutes with a spatial resolution of less than or equal to one arc minute. The energy resolution between 6 and 30 keV is less than or equal to 20 percent; for energies greater than 30 keV it is less than or equal to ten percent. Each of the three telescopes has a detector assembly in a cylindrical shape with a diameter of 13 cm and a length of 40 cm. The detector assembly contains the BGO shielding, at least one photo-multiplier tube and a low noise CdZnTe/ASIC detector.

A system schematic of the HXT is depicted in figure 3.4-1.

3.4.1 Hard X-ray Accommodation Assumptions and Requirements

The accommodations assumptions and requirements for the optical assembly are in table 3.4-1. The data in the table is for thermally formed glass shells. Other candidate materials for the HXT optic are epoxy replicated aluminum foils and replicated nickel shells. The accommodation requirements and assumptions for the components in the detector assembly are in table 3.4-2.

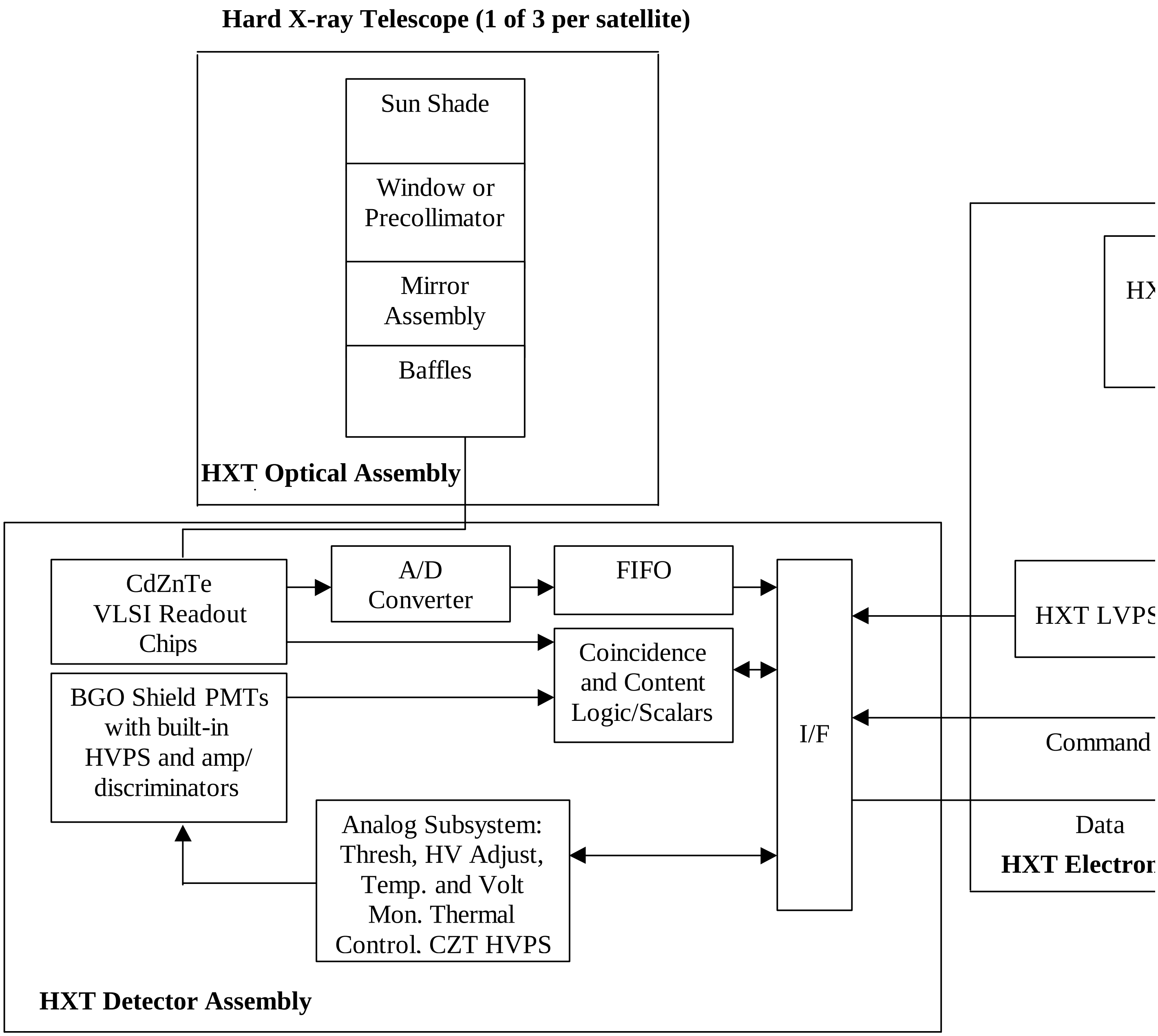


Figure 3.4-1. Hard X-ray Telescope Block Diagram

Table 3.4-1. Hard X-ray Optical Assembly

Number of Shells/HXT Assembly	149
HXT Assemblies/satellite	3
Mass (kg)/Assembly*	28.3
Length of Mirror Assy. (m)	0.5
Inner Diameter of Mirror Assy. (m)	0.07
Outer Diameter of Mirror Assembly (m)	0.4
Focal Length (m)	10.0
Bulk Temperature (°C)	20±5
Radial Temperature Gradient (°C)	TBD
Axial Temperature Gradient (°C/cm)	0.1
Heater power for optical assembly (watts)	30

Note: Optic Assembly Mass (kg)

Glass shells	25.6
Support Structure	2.7
Total	28.3

Table 3.4-2. Detector Assembly Components

Component	Power (Watts)	Operating Temp. (°C)	Survival Temp. (°C)	Mass (kg.)	Dimensions (cm)
Detector Assembly					
Detector CdZnTe w/VLSI	1.0	-10.0 to 0.0[1] with temp. stability of ± 2.5 °C over 2 hr.	-60. to 40.	0.5	2.5 x 2.5 (nominal) 3.0 x 3.0 (optimum)
PMTs per module	___	-50. to 30.[2]	-60. to 40.	.25	TBD
Shielding kg per telescope-3 HXTs per S/C)	-------	-50. to 30.[2]	-60. to 40.	6.0	TBD
HVPS,amplifier, discriminator for PMTs	0.1	-50. to 30.[2]	-60. to 40.	0.5	
Digital Electronics	0.33	-50. to 30.[2]	-60. to 40.	0.5	TBD
LVPS	0.8	-50. to 30.[2]	-60. to 40.	0.7	TBD
Analog Signal Circuitry	0.33	-50. to 30.[2]	-60. to 40.	0.3	TBD
Cables and Connectors	------	-50. to 30.[2]		0.5	N/A
Mech. Structure	--------	-50. to 30.[2]		1.8	N/A
Detector Assembly	2.56			11.0kg	Diameter: 13 cm; Length: 40 cm

Note 1 New results (7/99) indicate that it may be possible with little performance degradation to operate at room temperature (20^{o}C).
2 Temperature stability requirements of +/- 5.0 ^{o} C over 2 hours.

3.4.2 HXT Data Rate

The daily average science data rate for three HXTs on one satellite is 3 kbps. The peak science data rate for three HXTs on one satellite is 14 kbps. The average housekeeping data rate is 0.5 kbps.

3.4.3 Special Accommodation Requirements: Sensitivities And Concerns For HXT

Table 3.4-3. Special Accommodation Requirements for HXT

Magnetic Cleanliness	TBD
EMI/EMC	TBD
Radiation	TBD
Micro-phonics	TBD
Contamination	TBD

3.4.4 HXT Performance Specifications

Table 3.4-4. Hard X-ray Telescope System Performance Specification

Effective Area (per S/C)	375 cm^2
Focal Length	10 m
Spatial Resolution	$\leq$ 1 arc minute half power diameter
Energy Range	$\leq$ 6 keV to $\geq$ 40 keV
Field of View	$\geq$ 8 arc min over 6 to 40 keV band (TBR)
Energy Resolution	$\leq$20% FWHM over 6 to 30 keV band $\leq$ 10% FWHM above 30 keV

Table 3.4-5. Hard X-ray Telescope Optics Performance Specification

Effective Area (per telescope, per S/C)	$\geq$142.5 cm^2, $\geq$428 cm^2
Spatial Resolution	$\leq$ 1 arc min half power diameter
Energy Range	$\leq$ 6 keV to $\geq$ 40 keV
Field of View	$\geq$ 8 arc minute over 6 to 40 keV band (TBR)

Table 3.4-6. Hard X-ray Telescope Optics Position Sensitive Detector Performance Specifications

Energy Resolution	$\leq$20% FWHM over 6 to 30keV band , $\leq$10 % FWHM over 30 keV
Spatial Resolution	Nominally 15 arc seconds FWHM (telescope HPD must be over sampled by a factor of 4)
Energy Range	$\leq$6 keV to $\geq$ 40keV
Field of View	8 arc minute FWHM (TBR)
Quantum Efficiency	$\geq$ 90% over 6 to 40 keV band (TBR) (including K escape events)
Background	$\leq$ 2x10E-4 cts/cm2/sec/keV
Power (average including heaters)	TBD

3.5 Extendable Optical Bench (EOB)

The EOB subsystem as shown in figure 3.5-1 consists of the deployable metering truss structures and the associated mechanisms, an optical bench, a detector bench, detector bench sun shade, thermal enclosure, and electrical services. The EOB together with the instruments comprise the Instrument Module (IM). The following paragraphs describe the functional implementation, the hardware components, the interfaces, etc. to achieve the mission requirements.

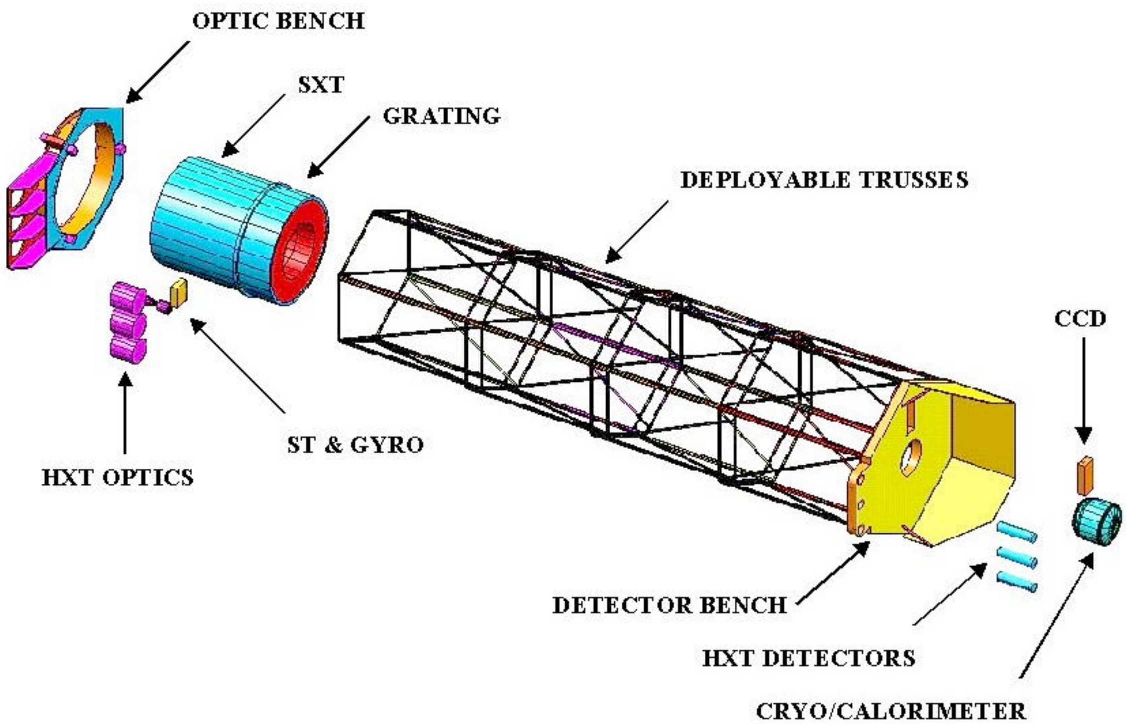


Figure 3.5-1. Instrument Module

3.5.1 Functional Description

The EOB provides for efficient packaging of the IM into the launch vehicle for the stowed configuration then deploys on-orbit to provide the required ten meter focal length for the SXT and HXT instruments. The EOB also provides a precise and highly stable platform that meets stringent position and stability requirements.

3.5.2 Hardware Description

The truss structures, optic and detector benches are constructed out of low coefficient of thermal expansion (CTE) composite materials with cyanate ester resins. To save mass, thermal MLI blankets are utilized as light tight closeouts around the trusses.

3.5.3 Performance Specifications

Table 3.5-1. Requirements (Mission and Derived)

Element	Parameter	Specification	Comments
Outer truss, optic and detector benches	Quasi-Static Design Loads	***Axial*** +10g/-1g (compression/tension) ***Lateral*** $\pm$ 4g	All combinations of axial and lateral load cases considered
EOB	Focal Length	10 meters	SXT and HXT instruments
EOB	Stray Light	1.0E+9 photons/cm^2/sec	Derived from Chandra Mission, Enclosed bench with"zero light leaks"
EOB	Deployment Accuracy	0.5 mm (focus) 0.5 mm (dispersion)	
EOB	Deployed Stability	0.2 mm (focus) 0.05 mm (dispersion)	Fiducial system anticipated to meet the dispersion requirement

3.5.4 Interfaces

The EOB interfaces with the following subsystems:

SXT/grating optic, HXT optics (3), HXT detectors (3), S/C structure, cryo/calorimeter, star trackers (2), gyros

3.5.5 Assumptions

A Fiducial system is baselined to provide high precision knowledge of SXT optic to calorimeter relative alignment.

A position mechanism is assumed to take out any potential errors in the focal length due to deployment or stability error.

3.6 The Cryogenic Subsystem

The cryogenic subsystem is a component of the Soft X-ray Telescope that actively maintains the X-ray Calorimeter at its operating temperature.

3.6.1 Functional Description

The cryogenic subsystem functions to control the X-ray Calorimeter detectors at sub-Kelvin temperatures. The system incorporates an adiabatic demagnetization refrigerator (ADR) and a turbo–Brayton cryocooler to continuously cool the X-ray Calorimeter by lifting detector and parasitic heat loads to a heat rejection interface with the spacecraft thermal bus. The X-ray

Calorimeter, ADR and elements of the cryocooler are contained within a dewar for thermal insulation from the spacecraft environment.

3.6.2 Hardware Description

The low temperature paramagnetic–salt stage of the multi–stage ADR serves as a high heat capacity thermal reservoir for the detectors. Active regulation of the reservoir's temperature achieves true continuous cooling. The remaining warmer stages are sequentially linked through heat switches and then cycled to cascade the heat to the relatively warm cryocooler interface. The final stage will reject heat slowly in order to minimize the peak heat load to the cryocooler, enabling the turbo–Brayton to maintain the lowest possible interface temperature.

The turbo–Brayton cryocooler employs very high speed miniature turbomachinery for the compression and expansion of gaseous helium in the implementation of the reverse Brayton thermodynamic cycle. The cryocooler is highly adaptable to a variety of design layouts since it is basically a continuous flow system made up of discrete, connected components that are relatively insensitive to their relative placement. The three major components are the compressor, expander and recuperator (counterflow heat exchanger).

Control of the ADR is accomplished through the ADR Control and Housekeeping Electonics (ACHE) associated with the microcalorimeter. The cryocooler is operated from its own power conversion electronics (PCE) which converts the 28 VDC power from the spacecraft bus to the 3 phase AC required to drive the compressor. The PCE also filters the conductive emissions that would otherwise feedback onto the power bus.

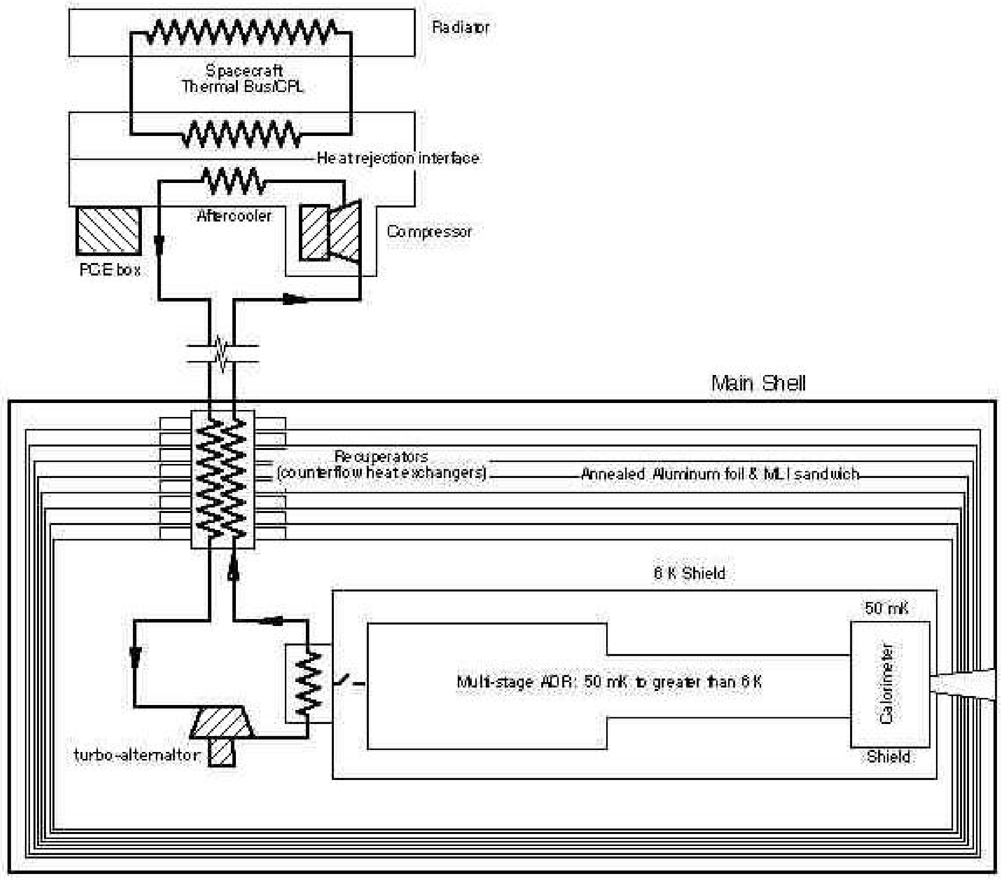


Figure 3.6-1. Cryogenic Subsystem Schematic (with radiator)

Table 3.6-1. Mass and Power Estimates

Element	Mass (kg)	Avg Power (W)	Peak Power (W)
Dewar	50	-	-
Cryocooler/PCE	39/1	100	100

3.6.3 Performance Specifications

Table 3.6-2. Performance Specifications

Element	Parameter	Specification	Comments
ADR	Cooling Power	10 µW @ 50 mK	Continuous cooling
	Heat Rejection	10-100 mW @ 6-10 K	Cyclic
Cryocooler	Cooling Power	5-100 mW	Dependent on heat Rejection temperature
	Heat Rejection	<100 W @ 220-300 K	

3.6.4 Interfaces

At the low-temperature continuous stage, the ADR will have a fixed uninterruptible thermal interface to the detectors, and a fixed interruptible thermal interface to the cryocooler cold tip. The ACHE provides the ADR interface to the X-ray Calorimeter and/or spacecraft power and C&DH buses.

Cryocooler power and C&DH is handled through the PCE to the corresponding spacecraft systems. Thermal interfaces are to the ADR through a heat switch and to the spacecraft at a heat rejection interface for heat removal to a "warm" radiator.

3.6.5 Assumptions

The current baseline design is being pursued with the most basic turbo-Brayton cooler. Further system thermal analysis will likely show that two compressors, a radiator or second turbo–alternator stage and, therefore, a second recuperator will be required to cool the X-ray Calorimeter.

4.0 Spacecraft Bus

The Spacecraft Bus Module will be engineered to interface with the separable independent Instrument Module and will provide Communications, Control and Data Handling, Power, Propulsion, Structure and Thermal Control. It also interfaces with the Dual Payload Adapter on the launch vehicle for insertion into elliptical orbit.

4.1 Communication Subsystem

The Communication Subsystem provides the communication uplink and downlink for the Constellation-X Satellites. It also provides turn around ranging for orbit determination by the ground stations. The following paragraphs describe the functional implementation, the hardware components, the interfaces, etc. needed to achieve the desired communication capabilities.

4.1.1 Functional Description

The ground station uplinks the commands to each satellite on a S-Band Carrier of 2096.000 Mhz (TBR). This is phase modulated with a Subcarrier of 16 Khz (TBR) frequency and Phase-Shift Keyed with command data in a PCM format at the rate of 2 kbps. The satellite is able to receive the signals in from all directions and decode the carrier and subcarrier to output commands to the C&DH Subsystem. The C&DH Subsystem also collects the housekeeping data and formats it in frames of engineering data at the rate of 2 kbps. The data is then Reed-Solomon encoded and subsequently Convolutionally encoded for error correction on the ground. The encoded data is phase modulated in the Communication Subsystem on an S-Band Carrier of 2250Mhz (TBR). The radio frequency is transmitted from the satellite and then received on the ground. Similarly, the science data at the rate of 1700kbps is formatted, encoded, and modulated on an X-Band Carrier of 8250Mhz (TBR). The X-Band frequency is transmitted from the satellite and received on the ground. The ground station also uplinks S-Band carrier modulated by a ranging signal. The satellite detects this signal and then transmits a turn around signal which is received by the ground station for orbit determination purpose.

4.1.2 Hardware Description

Figure 4.1-1 shows the overall communication links for the ground station and the satellite. Figure 4.1-2 shows the block diagram of the communication subsystem. The subsystem consists of two S-Band Hemispherical antennas to provide omni-directional coverage for the links. The S-Band RF signal is combined in a Hybrid and separated in a Diplexer as transmit and receive signals. A 5 Watt Transponder provides S-Band functions of transmitter, receiver, and tracker. The Transponder interfaces with C&DH Subsystem for Command and Engineering Telemetry. There is also a 20 Watt Power Amplifier to enable the return link to close when the satellite is at L2. A Band Reject Filter prevents out of band spurious signals generated in the power amplifier being transmitted to the ground.

The science data from the C&DH Subsystem is modulated in the 5-Watt X-Band transmitter and radiated from a 1.2 Meter Dish antenna in earth's view. The communication subsystem, considered for Four Spacecraft Configuration, has no redundancy at the component level.

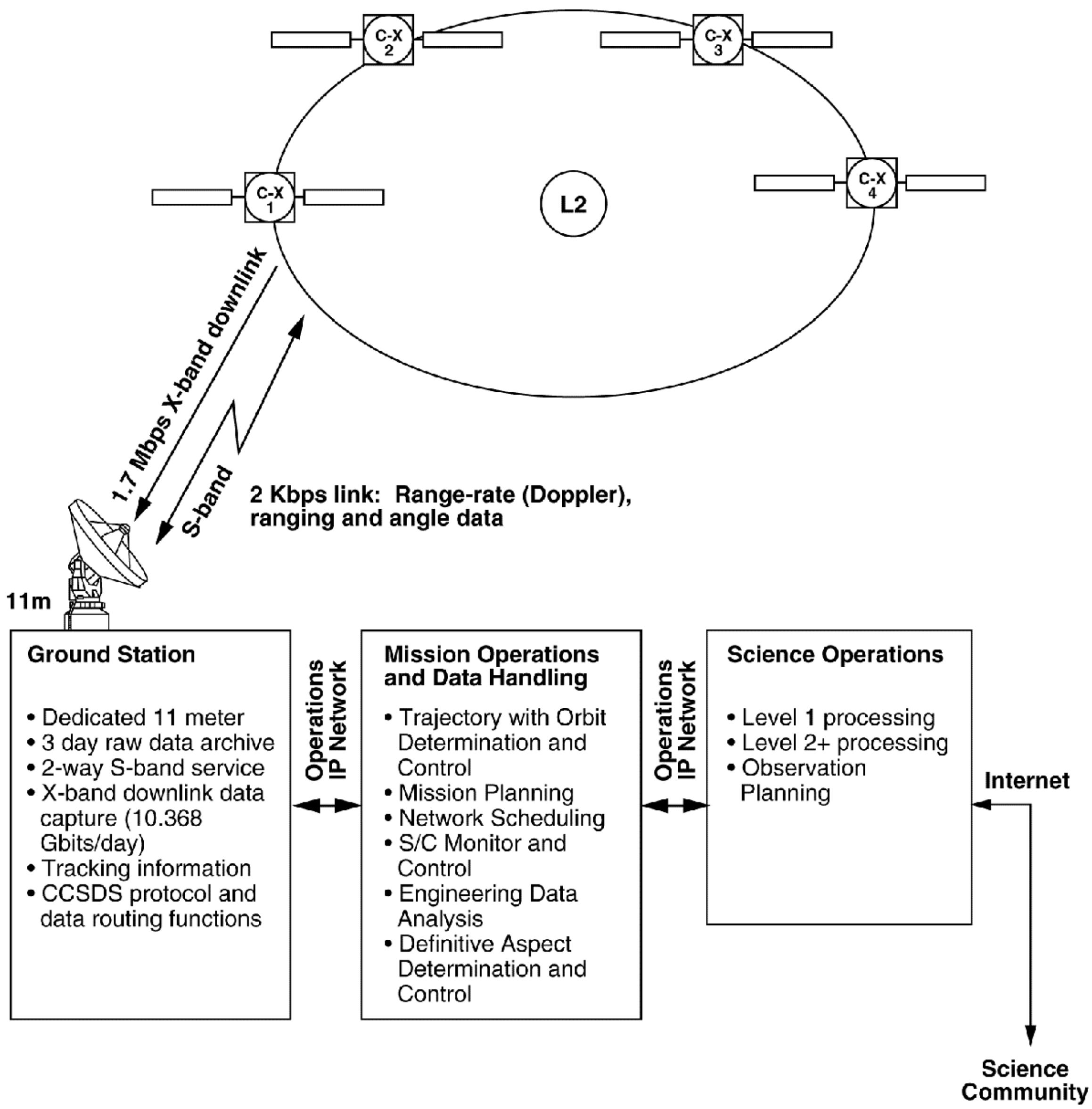


Figure 4.1-1. Mission Operations Block Diagram

The following table 4.1-1 itemizes physical characteristics of hardware components:

Table 4.1-1. Physical Characteristics of Hardware Components

Component	**Dimensions**	**Power**	**Mass**
	(L/W/H cm)	Watts Peak	Kgs
X/S-Band Hemi antenna(2)in house Build	15x15x15		8
Hybrid	5x5x5		1
Diplexer	10x5x5		1
Band Reject Filter	10x5x2.5		1
S-Band Transponder(L3//CXS-610)	19x13x7	26	4
S-Band 20 watt Power Amplifier	20x20x5	38	5
X-band High Gain Antenna(Dish)	1.2 meters dia.		13
X-Band Transmitter	20x22x8	70	5

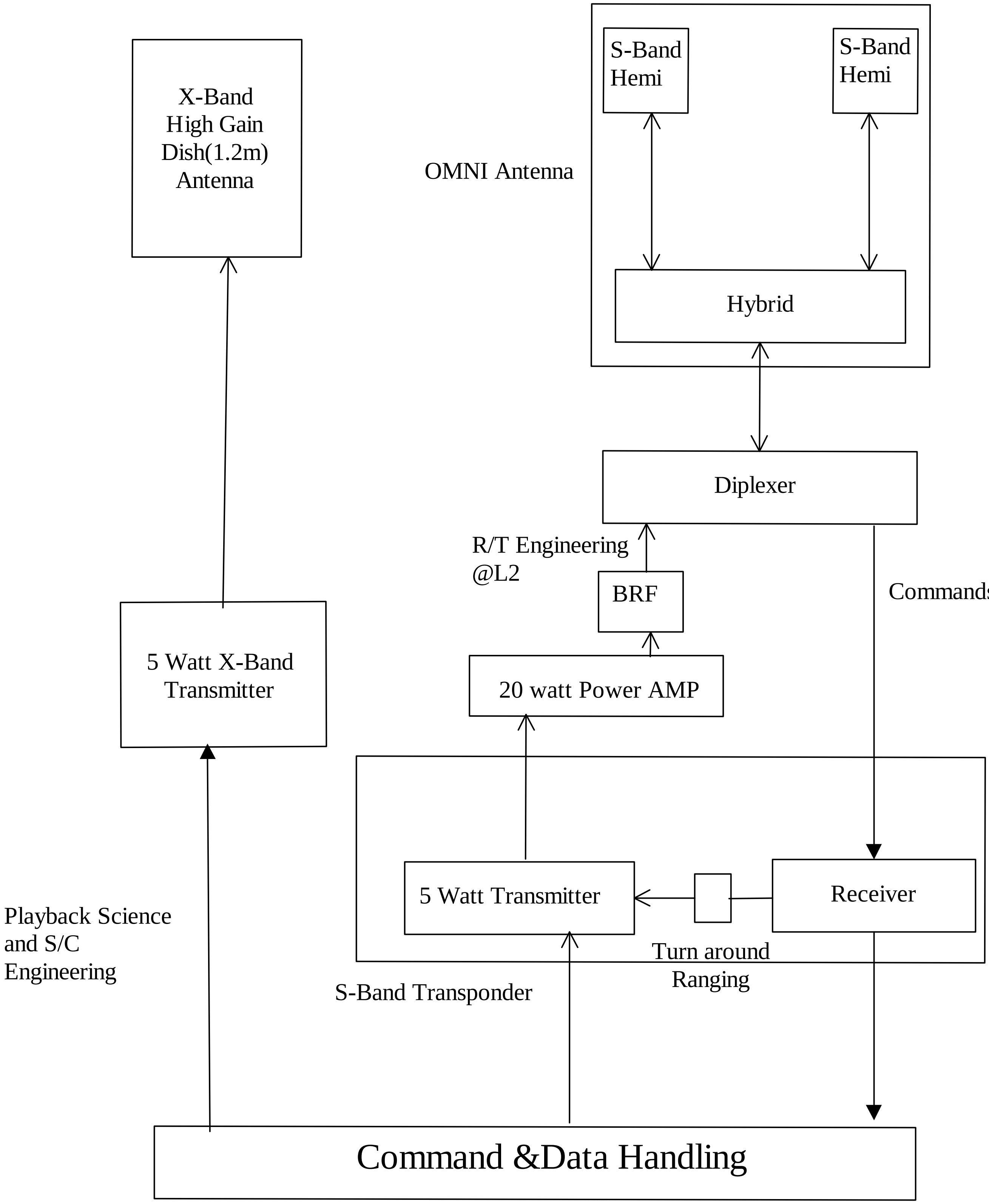


Figure 4.1-2. S/X-Band Communications Subsystem Block Diagram

4.1.3 Performance Specifications

The following table 4.1-2 shows the technical specifications for the communication subsystem:

Table 4.1-2. Technical Specifications for Communication Subsystem

Link	Parameter	Specification
S-Band Command Uplink	Carrier Frequency	2096.000Mhz
	Carrier Modulation	Phase
	Modulation Index	1.00 Radians Peak
	Subcarrier Frequency	16.00KHZ
	Subcarrier modulation	PSK
	Command Data	PCM
	Required Carrier/Noise	15 DB
	Command Data Rate	2 kbps
S-Band Telemetry Downlink	Carrier Frequency	2250.000Mhz
	Transmitter Power	20 Watts
	Carrier Modulation	Phase
	Telemetry Data Rate	2 kbps
	Modulation Index	1.2 Radians Peak
X-Band Telemetry Downlink	Carrier Frequency	8250.000Mhz
	Carrier Modulation	BPSK
	Transmitter Power	5 Watts
	Telemetry Data Rate	1700kbps
	Ground Antenna	11 meter Diameter Dish
	Satellite Antenna	1.2 meter Diameter Dish
Ranging Signal	Modulation	TBA
	Turnaround Ratio	TBA
All Links	Allowable Error Rate	1 error in 100,000

The detailed link margins are analyzed in Appendix B. The summary of the margins is given in tables 4.1-3 and 4.1-4. The links have adequate margins.

Table 4.1-3. S-Band Uplink Margin Summary

	IF SNR	Carrier Channel	Command Channel
	DB	DB	DB
Max.	-14.2	1.3	3.3
Min.	-13.5	2.0	4.0

Table 4.1-4. S-Band Downlink Margin Summary

Channel	Range	Telemetry Channel DB
IF SNR	Max.	-16.2
	Min.	-14.3
Carrier	Max.	.5
	Min.	2.4
PCM/PM	Max.	.8
	Min.	2.7

4.1.4 Interfaces

The Communication Subsystem interfaces with the ground system (which is compatible with the DSN System), C&DH Subsystem, and Power Subsystem. The RF Links communicate with the ground systems, while demodulated command and encoded telemetry interface with the C&DH Subsystem. Unregulated nominal 28 Volts DC power is supplied by the Power Subsystem.

4.2 The Command and Data Handling Subsystem

The Command Data Handling Subsystem processes the commands received from the ground, and the data from the satellite subsystems and the instruments. It also manages the time-keeping functions. The following paragraphs describe the functional implementation, the hardware components, and the interfaces of this subsystem.

4.2.1 Functional Description

The Command and Data Handling Subsystem receives the commands from the Communication Subsystem in Pulse Code Modulation format at 2kbps, and performs all decoding and validation functions such as the inversion detection and correction, the code-blocking, the satellite ID verification, and the checksum verification. It then distributes the commands to all software and hardware subsystems and the instruments.

The subsystem collects the housekeeping data from the satellite and the instruments. It then formats the data in frames of engineering data at the rate of 2kbps. The data is then first encoded with Reed-Solomon long code (255,233), and subsequently with 1/2 rate Convolutional Code for transmission in RF S-Band in real-time. It may also be optionally encoded with Pseudo-Random Code. The subsystem also monitors the housekeeping data and responds to out of limit conditions.

The subsystem collects science data from the instruments; and packetizes, and assembles them into CCSDS Transfer Frames with Reed-Solomon short code (255,245). The data is then stored on orbit in the recorder. When the recorder goes into dump mode for downlinking to the ground, the data is read, the errors are detected and corrected, and the short code is stripped. Corrected data is then first encoded with Reed-Solomon long code (255,233), and subsequently with 1/2 rate Convolutional Code for transmission in RF X-Band. The data may also be optionally encoded with Pseudo-Random Code.

The engineering and science data are also stored in the subsystem so that it can be dumped daily on schedule when the satellite is visible from the ground station.

The satellite receives UTC time from the ground and synchronizes its clock to UTC. It distributes the time and stamps the data with the synchronized time.

4.2.2 Hardware Description

The Constellation-X Command and Data Handling Subsystem consists of a single electronic box with four cards. They are as follows:

- Uplink/Downlink
- Instrument Interfaces
- Processor/Recorder
- Low Voltage Power Converter

The functions of the first three cards are described in section 4.2.1. The fourth card converts unregulated 28 Volts power from the Power Subsystem to the +5, and ±15 Volts regulated power required for its electronics.

Figure 4.2-1 shows a block diagram of the subsystem

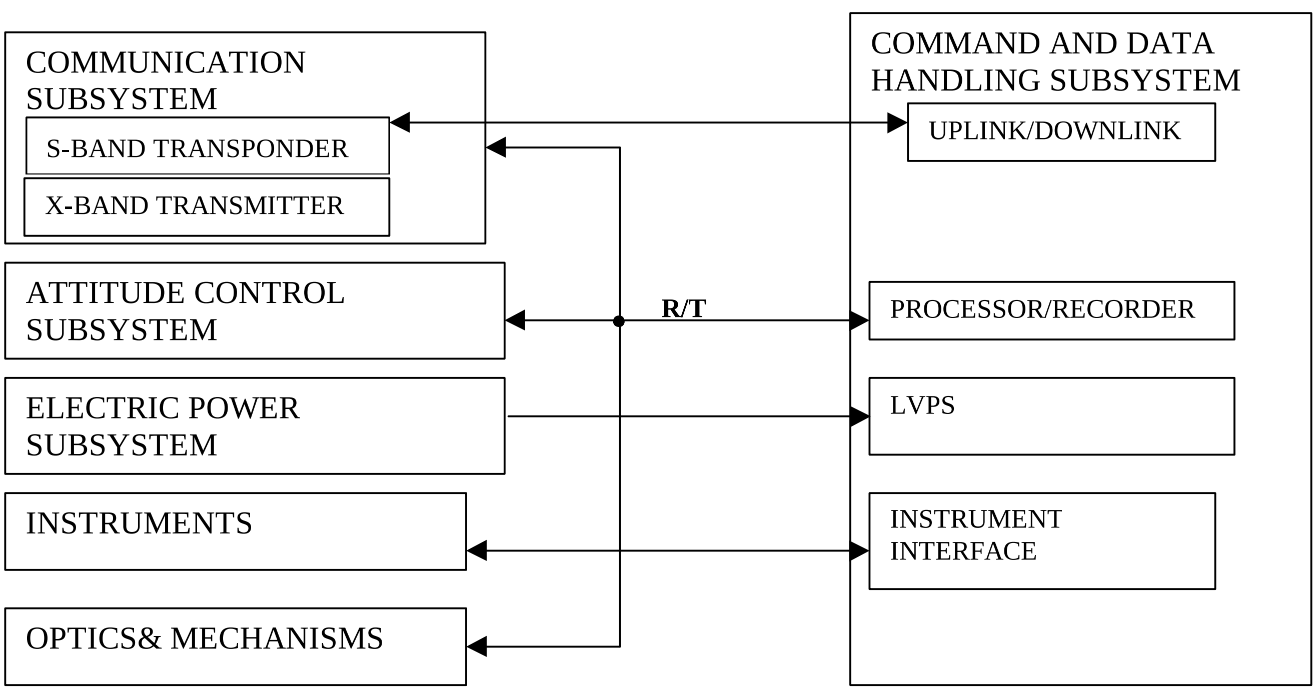


Figure 4.2-1. Command and Data Handling Subsystem

4.2.2.1 Mass Estimation

Component	Mass (kg) SOA 1998
Uplink/Downlink	0.9
Instrument Interface	1.0
Processor/Recorder	1.1
LVPC/Housekeeping	0.9
Chassis	2.5
Total	6.4

4.2.2.2 Basis of Estimate

State of Art (SOA) 1998 is based on MIDEX/MAP measured and/or estimated masses for electronics components with equivalent functions.

4.2.2.3 Power Estimation

Component	Power (watts) SOA 1998
Uplink/Downlink	12
Instrument Interface	12
Processor/Recorder	8
LVPC/Housekeeping	12
Total	44

4.2.2.4 Basis of Estimate

SOA 1998 is based on MIDEX/MAP measured and/or estimated power for electronics components with equivalent functions.

4.2.3 Performance Specifications

The following subsections cover data handling and command processing requirements for the four satellite configuration.

4.2.3.1 Data Rates

The following data rates are per spacecraft.

4.2.3.2 Data Ingest Rate

Spacecraft Level Requirement	Daily Average (kbits/s)	Peak (kbits/s)
Science	48	1365
Instrument Housekeeping	2	2
Spacecraft Housekeeping	2	2
Total	52	1369

Note: 2 kbps of Instrument Housekeeping data is over and above the housekeeping data included in 48 kbps. Future sizing of C&DH system will decrease the data by this 2 kbps.

4.2.3.3 Spacecraft Data Downlink Rate

The C&DH shall support X-band and S-band downlink at the following rates:

Downlink	Rate (kbits/s)
S-band (S/C housekeeping)	2
X-band	1700

4.2.3.4 Spacecraft Data Uplink Rate

The C&DH shall support an S-band uplink rate of 2 kbits/s and 150 bps.

4.2.3.5 End-to-End Data Rates and Storage

Mission	Description	Value	Units
	Science Data rate	192000	bits/s
Data Ingest	Assuming number of spacecraft	4	
	Science Data rate	48000	bits/s
	Instrument Housekeeping	2000	bits/s
	Spacecraft Housekeeping	2000	bits/s
	Data produced during 1 day	4492800000	bits
Data Storage	Data produced during 1 day	4492800000	bits
	with 1% Packet Header	4537728000	bits
	with 4% Short Reed Solomon	4719237120	bits
	RS(255,245)		
	Data stored for 2 days	9438474240	bits
X-band Downlink	Data stored for 1 day	4719237120	bits
	remove 4% Short Reed Solomon	4537728000	bits
	with 14% CADU overhead	5173009920	bits
	which includes S(255,223)		
	which has to go down a 1.7 Mbit/s downlink		
	which requires a pass time per day:	3042.947012	s

4.2.3.6 Processing Requirements

- Science Data Processing
  The C&DH shall perform instrument data processing involving the centering of the image on the CCD and the Fiducial Transfer System (FTS).

- Uplink Command Processing
  CCSDS Telecommand Protocol. Uplink command decoding, verification, distribution and execution.

- Downlink Data Processing
  CCSDS Telemetry Protocol. Implement Reed-Solomon, Cyclic Redundancy Code (CRC), Psuedo-Random (PS), ½ and ¼ Convolutional Encoding. Each can be bypass-able.

- Spacecraft Data Processing
  Housekeeping telemetry monitoring and out-of-limits response. Maintenance and distribution of spacecraft time (accurate to 10 ms). Correlate spacecraft time to ground time (accurate to 1 ms). Error Detection and Correction (EDAC) for data storage. Recorder management. The C&DH shall provide a hardware platform for Attitude Control System (ACS) processing.

4.2.3.7 Spacecraft Data Storage Requirement

The C&DH can store all 2 days worth of instrument and spacecraft data, at least 9.5 Gbits of data storage. It supports a nominal 1 ground contact per day, provides simultaneous record and playback capability and uses short code Reed-Solomon RS (255, 245) for error detection and correction on storage.

4.2.4 Interfaces

The Command and Data Handling Subsystem interfaces with the Communication Subsystem for the ground commands, and the engineering and science data downlinks. It also interfaces with all subsystems with transmit and reply buses for distribution of commands and collection of housekeeping data. Unregulated nominal 28 Volts DC Power is supplied by the Electrical Power Subsystem. The Attitude Control Subsystem supplies raw attitude subsystem data on transmit and reply buses to the Command and Data Handling Processor for attitude data processing.

4.3 The Attitude Control Subsystem

The Attitude Control Subsystem collects and processes on-orbit attitude information from the sensors, and provides it to the ground operations for further processing. It controls the attitude of the satellite, in all modes, autonomously as well as under ground control. It provides a safe attitude for the satellite under abnormal conditions. The following paragraphs describe the functional implementation, the hardware components, and the interfaces of this subsystem.

4.3.1 Functional Description

The launch vehicle is dual manifested and launches the satellites into highly elliptical orbits in the lunar plane. The ground system acquires the satellites and immediately commences Delta-V maneuvers to correct for launch dispersions, collision prevention, and safe attitude as necessary.

The Perigee of the orbit may be raised to prevent the satellites from reentering the earth. During all these phases, the attitude is controlled appropriately.

After initial satellite checkout, at appropriate time, Delta-V maneuvers are performed for the lunar swing by phasing loops. This will send the satellite to the L2 libration point. Again, the attitude is controlled appropriately.

As soon as possible, the satellite will be put in mission attitude and commanded to go into inertial hold attitude or slew to new position. It may be necessary to unload the momentum that is built-up during all these activities. The necessary control will be provided by the Attitude Control Subsystem. The attitude sensor data is telemetered to the ground for further processing to provide an aspect solution for the satellites.

The Attitude Control Subsystem also protects the satellite from depletion of the batteries, subsystem failures, etc. that will cause the loss of a satellite. It does so by placing the satellite in a safe attitude until the ground attains the control and prevents the loss. There is a separate and dedicated hardware for this function.

4.3.2 Hardware Description

Figure 4.3-1 Block Diagram for Attitude Control Subsystem, shows hardware components of the subsystem. The ACS uses three types of sensors. The first type is Sun Sensors. A total of eight Sun Sensors are placed at various locations to provide sufficient redundancy and coverage under all conditions. They are used to process attitude information during initial acquisition, maneuvers, and safe modes. They have independent hardware, separate from the C&DH Computer to process the information. The second type of sensor is the Star Tracker. This is the main attitude sensor, which is used for the mission attitude sensing. The tracker enables the satellite to have sufficient accuracy, knowledge, and stability for its attitude. The third sensor is the Inertial Reference Unit, which has gyros in it and computes the acceleration, etc. of the satellite to provide dynamic attitude information to the C&DH Computer. The sensors interface with Attitude Control Electronics, which also interfaces with the C&DH Computer with Transmit, and Receive Buses for transfer of commands, telemetry, and the attitude data.

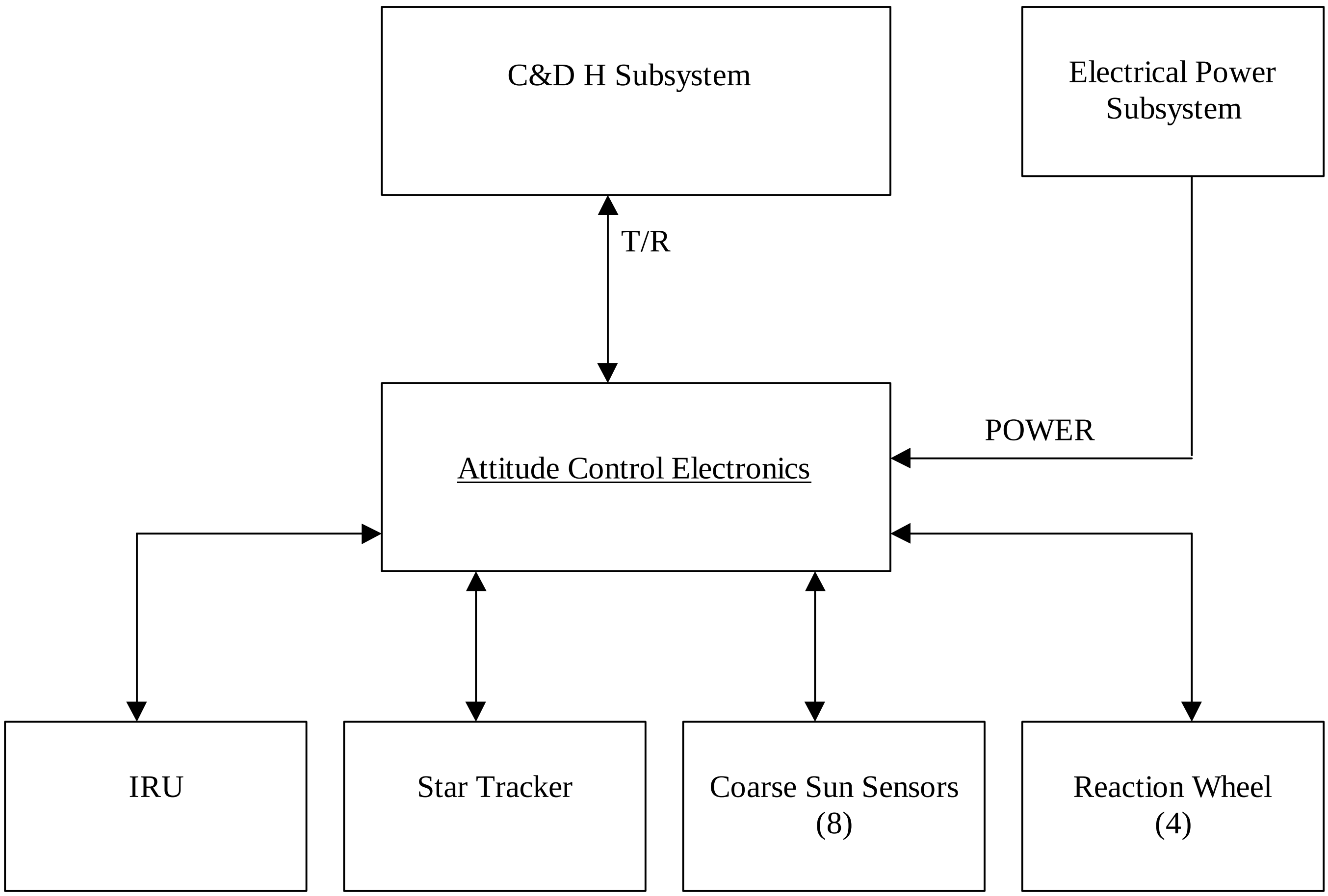


Figure 4.3-1. Block Diagram for Attitude Control System

The C&DH Computer processes the sensor data and commands four Reaction Wheels or eight Thrusters to control the attitude of the satellite.

Figure 4.3-2 shows the satellite attitude control loop, hardware and software functional allocation in a block diagram form.

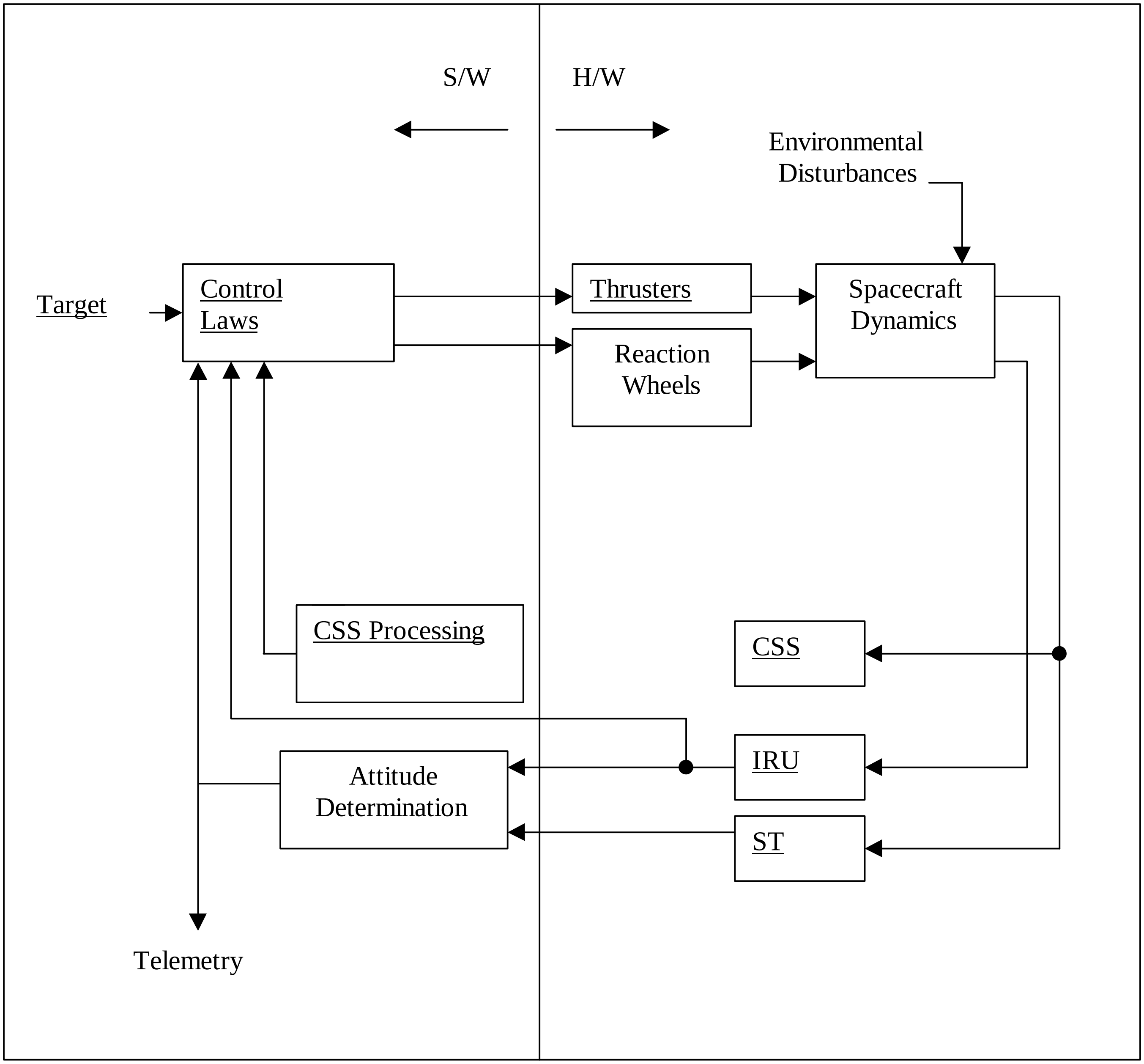


Figure 4.3-2. Functional Diagram for Attitude Control Systems

The following table 4.3-1 itemizes the physical characteristics of hardware components:

Table 4.3-1. Physical Characteristic of Hardware Components

Element	Make/Model	Qty	Mass, ea (kg)	Avg Power, ea (w)	Peak Power, ea (w)	Total Mass (kg)	Total Avg. Power (w)	Total Peak Power (w)
Star Tracker	Ball CT-602	1	6	10	10	6	10	10
Inertial Reference Unit	Litton SIRU	1	6	20	20	6	20	20
Coarse Sun Sensors	Adcole 11866	8	0.005	0	0	0.04	0	0
Reaction Wheels	Ithaco E-wheel	4	14	30	200	56	120	200
Attitude Control Elect.	in-house	1	5	10	10	5	10	10
ACS Totals						**73**	**160**	**240**

4.3.3 Performance Specifications

The following table 4.3-2 shows the performance specifications for the Attitude Control Subsystem:

Table 4.3-2. Performance Specifications for Attitude Control Subsystem

Description	Parameter	Specification	Notes
Pointing Range	Roll	± 20 Degrees	Max
	Pitch	± 20 Degrees	Max
	Yaw	± 180 Degrees	
Pointing Knowledge	Roll	60 Arcseconds	
3σ Accuracy	Pitch	5 Arcseconds	
	Yaw	5 Arcseconds	
Pointing Control	Roll	60 Arcseconds	
3σ Accuracy	Pitch	30 Arcseconds	
	Yaw	30 Arcseconds	
Pointing Stability	Pitch	0.6 Arcsec./Second	Max
	Yaw	0.6 Arcsec./Second	Max
Pointing Jitter	Roll	5 Arcseconds	Max
	Pitch	2 Arcseconds	Max
	Yaw	2 Arcseconds	Max
Slew Time Between Targets		1 Hour	Max
Sun Angle Range	Roll	45-135 Degrees	
Earth /Moon Angle	Roll	30-150 Degrees	

4.4 Electrical Power Subsystem

The Electrical Power Subsytem (EPS) provides conversion, generation, storage, control, and distribution of unregulated power for the operation of all the Constellation-X satellite subsystems and components. It is a Direct Energy Transfer (DET) system, which converts solar energy to electrical energy and provides it directly to all spacecraft loads at an unregulated voltage that varies from 22-32 V. Power balance, battery charge control, power safing, and ground power interfacing are all functional elements of this system. Power system telemetry is gathered, converted and provided to the Command and Data Handling System (C&DH) as part of the housekeeping data telemetry stream.

4.4.1 Functional Description

When in sunlight, solar power is gathered by a single rectangular Solar Array (SA) panel that is body mounted to the spacecraft, and is then converted to electrical power at a voltage determined by the battery (approximately 30V). This is applied to a main bus that supplies all loads with their operational power requirements at all times as well as providing charge to the battery. The main bus is routed through a power distribution unit that is split into two parts. The essential bus is an unswitched bus that provides power to essential spacecraft components such as the communications receiver, C&DH Subsystem, Attitude Control Subsystem (ACS) and survival heaters. The non-essential bus supplies switched power to the instrument as well as the operational heaters. Fault protection is provided to non-essential loads through the use of fuses or re-settable circuit breakers.

The Constellation-X energy storage will be provided by a single NiH battery and will be sufficient to meet the energy needs of the spacecraft during the launch ascent phase through the attainment of final orbit positioning. In addition the battery will provide power during peak load periods when solar array power is inadequate.

Battery charging and health will be maintained by the Power System Electronics (PSE). The PSE will monitor battery temperature, pressures, current and voltage and maintain state of charge through a combination of means. A Voltage/Temperature (VT) controller as well as an Ampere Hour Integrator (AHI) and Trickle Charge Controller (TCC) will work in unison to maintain an optimal battery state of health.

The PSE will also match array output power to load requirements through the operation of a full shunt system with Pulse Width Modulation (PWM) providing fine control. Excess current, beyond that required for spacecraft loads and battery charging, is shunted from the array through a series of parallel electronic switches several of which are pulsed at a rate and duty cycle that produces energy balance.

The PSE also provides load switching, fault protection, and safing functions. Electronic switches provide control of loads on the non-essential bus while short detection circuitry monitors these loads for over current conditions and removes power if such a condition is detected.

In the case of an anomalous condition where the battery state-of-charge drops below a certain level, or the battery voltage is low, the power system will act to remove non-essential loads and trigger safing actions until the condition is cleared.

4.4.2 Hardware Description

The Power Subsystem consists of the solar array, battery, and PSE, with the physical characteristics as shown in table 4.4-1.

4.4.2.1 Solar Array

The solar array will consist of one 6 meter square body mounted panel and provide 1400 W Beginning of Life (BOL) and 1200 W End of Life (EOL) power to support an 1100 W orbital average load. The panel will be made up of approximately 7500 Multijunction GaAS (25% bare cell efficiency) solar cells configured in 50 cell strings. Several strings will be bonded to an epoxy graphite substrate to form a module. These modules will then be mounted to a frame to form the total array. Standardized modules should reduce cost and schedule and are being used on several programs currently in development. Total panel mass including all cells, wiring, substrate and frame is estimated at 49 kg.

4.4.2.2 Battery

The Constellation-X energy storage will be provided by one 22 cell Single Pressure Vessel (SPV), 21AH, NiH battery. It is assumed the solar arrays will nearly always be in the sun so that minimizing battery capacity and thereby mass and volume will be possible. Nominal discharge voltage of this battery will be around 28V rising to 30V when charged. The battery will provide voltage, current, temperature and pressure information to the PSE to provide charge control and determine state-of- health. Battery assembly dimensions will be approximately 30cm x 30cm x 48cm with a mass of 21 kg.

The battery charging profile will consist of a full charge period where maximum charging current is supplied followed by a taper charge that occurs when the selected voltage limit is reached. A trickle charge will be applied when the AHI indicates the full recharge ratio has been reached. All these functions are controlled by the PSE.

4.4.2.3 PSE

The Constellation-X PSE will be based on the MAP design. It is a direct energy transfer (DET) system. Power is provided to the loads through switched and unswitched services. Battery charge control (AHI, CC, VT & TC) is achieved in software with some hardware backups. The system supports the following battery telemetry: current, voltage, half voltage, temperature and pressure. Battery relay and electronic load switching is done by the PSE (see Figure 4.4-1). Seven solar array segments are sequentially shunted as necessary with a PWM converter driving two other segments for fine power control. One solar array segment is unshunted and directly on the bus.

Additional engineering will have to be done to size the system for the Constellation-X load and configure switching, fusing and safing for its unique requirements. PSE dimensions are estimated at 60 x 27 x 40 cm with a mass of 52 kg. The PSE is designed to a functional rather than component level redundancy standard.

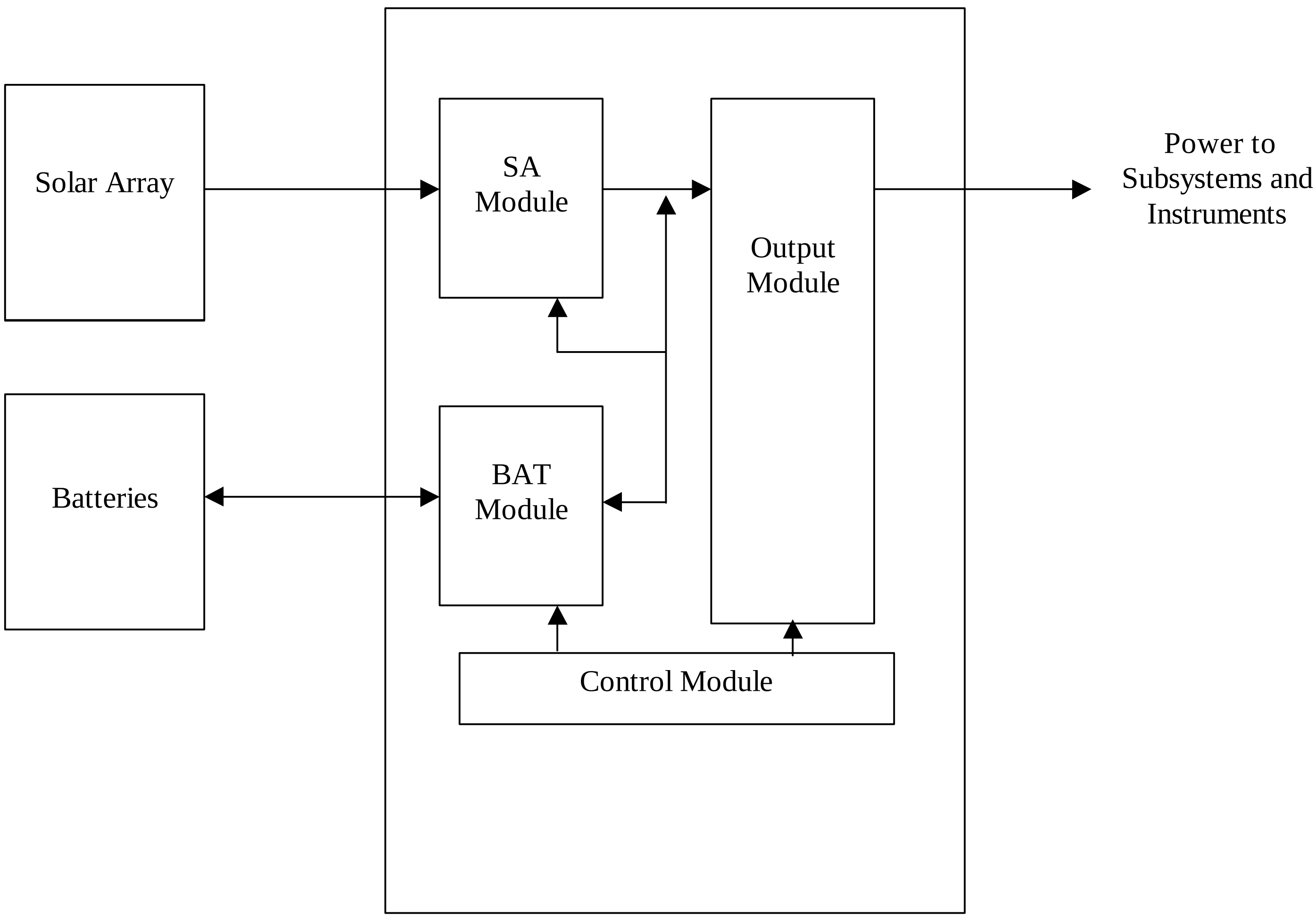


Figure 4.4-1. Power Supply Electronics

Table 4.4-1. Physical Characteristic of Hardware Components

Component	Dimensions	Power	Mass
Solar Array	6 Sq. Meters	N/A	49 kg.
Battery	30 X 30 X 48 cm	6 W	21 kg.
PSE	60 X 27 X 40 cm	30 W	52 kg.

4.4.3 Specifications/Performance

The following table 4.4-2 lists the specifications and performance of the Constellation-X Electrical Power System.

Table 4.4-2. Specifications and Performance of Electrical Power System

Parameter	Spacecraft Specification	EPS Performance
Bus V	24 – 34 V	22 – 35 V
EOL Load Capability	1100 W	1200 W
Design Life	5 Years	5 Years
Energy Storage	Support all ops	Support all ops*

* Limited duty cycle

4.4.4 Interfaces

The EPS interfaces to any spacecraft components that require unregulated bus voltage. This usually includes most components on the spacecraft except for the passive thermal control system. In addition, commands and telemetry is provided by an interface to the C&DH Subsystem.

4.5 The Propulsion Subsystem

This subsystem provides all the expendable propulsion requirements for five years of the Constellation-X satellite including acquisition, orbit correction, lunar phasing loops, lunar swingby maneuver, momentum unloading, etc. The subsystem utilizes mono-propellant hydrazine in a blow down mode.

4.5.1 Functional Description

The subsystem provides for the propulsion in the initial acquisition of the satellite by correcting for the tip-off, and the orbit dispersions of the launch vehicle. The necessary Delta-V for the lunar phasing and swingby maneuvers is also provided by the subsystem. As the satellite reaches the L2 point, the orbit insertion and corrections are done. The ACS subsystem acquires and tracks the stars using propulsions from the momentum wheels during the normal observation. When the wheels become saturated, the momentum is unloaded by the subsystem as a normal procedure. The ACS provides necessary control signals. The progress of the propulsion is monitored by the ground system by pressure, and temperature telemetry, which is provided to the C&DH Subsystem.

4.5.2 Hardware Description

The Constellation-X propulsion subsystem utilizes mono-propellant hydrazine chemical thrusters. All elements are COTS items. The system operates in a blow down mode, with the BOL pressure of 2757.6 KPa and the EOL pressure of 689.4 KPa. The dry mass of the propulsion subsystem is 35 kg, with the total propellant mass of 180 kg. Based on the initial spacecraft mass of 2285 kg, the propulsion subsystem provides up to 177 m/s ΔV. Shown in figure 4.5-1 is a schematic of the Constellation-X propulsion system.

Due to the volume constraints of the hex bus structure, three identical tanks are required to handle the total propellant load. The specification of the tank is listed in section 4.5.3. Four thrusters are mounted on both sides of the hex bus structure. Thrusters are to be canted outward from 5 to 10 degrees (TBR) in order to provide three axes attitude control and momentum unloading capabilities. Thrusters on either side can be fired in pairs to provide ΔV during orbit maneuvers.

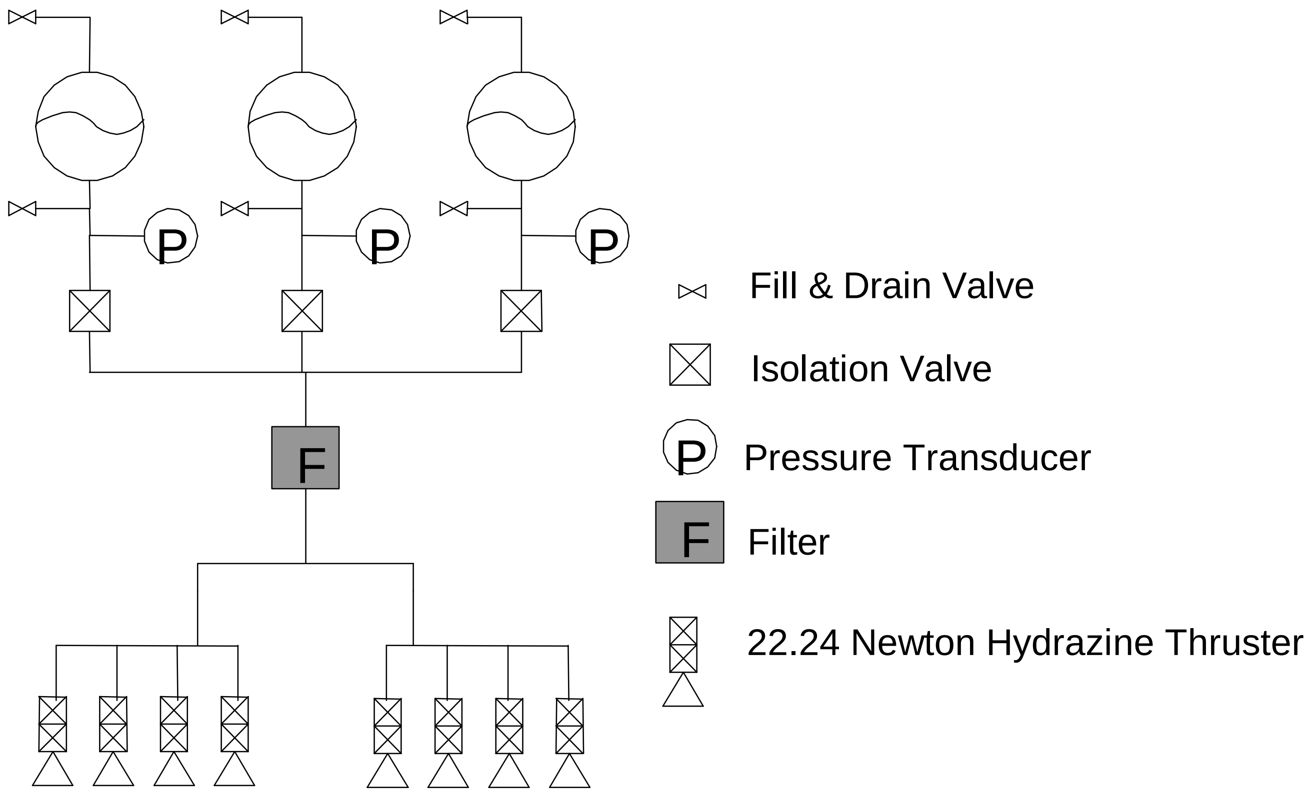


Figure 4.5-1. Schematic of the Propulsion System

4.5.3 Specification

Table 4.5-1. Propulsion Specification

Propulsion Component (PSI)	Size	Mass (kg)	Quantity	Total Mass (kg)
Hydrazine tank (PRIMEX)	55 cm O.D. sphere	5.7	3	17
22.24 Newtons thruster	7 cm O.D., 18 cm L.	0.7	8	5.6
Miniature Fill & Drain Valve (VACCO)	7 cm L., 1.4 cm O.D. Max.	0.03	6	0.18
Filter (VACCO)	8.4 cm L., 1.5 cm O.D. Max.	0.5	1	0.5
Pressure Transducer (GULTON)	6 cm L., 5 cm O.D.	0.3	3	0.9
Isolation Valve (VACCO)	5 cm x 7 cm x 8 cm	0.32	3	0.96
Miscellaneous Hardware + brackets	0.635 cm O.D. Tubing, etc.	5	N/A	9.9
Total				35

4.5.4 Performance

With a total propellant load of 180 kg, the Constellation-X propulsion subsystem can deliver up to 177 m/s of ΔV. Shown below is the table of 22.24 Newtons thruster performance specifications.

Table 4.5-2. Thruster Performance Specifications

Specific Impulse	220 sec.
Total Impulse	50681 Newtons-sec.
Minimum impulse bit	.33 Netwons-sec @ 1.69 KPa & 22 ms ON
Total pulses	12300 pulses

4.5.5 Interfaces

The propulsion subsystem will be actuated by the flight software in the C&DH Subsystem.

4.6 Flight Software Subsystem

The Flight Software resides in the processor card of the C&DH Subsystem and orchestrates the satellites for seamless functioning of the ground segment and the space segment including the instrument module.

4.6.1 Functional Description

The subsystem receives commands from the ground and processes them for distribution to other subsystems and instruments. It also collects engineering telemetry, formats it for ground stations to monitor health of the satellite, and collects science data for storage on tape recorder. It maintains and timestamps data with UTC time, synchronized accurately to the ground station and traceable to NIST. The subsystem monitors health and status data of the satellite and provides safe mode commands when needed.

The subsystem calibrates, determines, and controls the attitude of the satellite to achieve science and mission objectives. It provides onboard processing necessary for propulsion maneuvers.

4.6.2 Software Module Description

The Flight Software consists of the following modules:

- Executive Module
- Command Processor
- Data Processor
- Attitude Processor
- Health and Status Module

4.6.2.1 Executive

The processor will be designed with an operating system with limited capabilities compared to ground systems. The system will be able to handle multiple tasks in realtime. It will be designed to execute many different processes without conflicts and with sufficient time and memory margins.

4.6.2.2 Command Processor

The satellite receives uplinked commands from the ground either in realtime or stored command formats. The processor also generates commands onboard. The module checks the commands for validity and conflicts and distributes them to various subsystems and instruments. It maintains the execution status of the commands with counters and other status bits. It receives UTC time and synchronization commands from the ground and resets or adjusts gradually for drift as commanded by the ground. Scientific Instrument commanding associated with a target is usually specified by the ground via stored commands, although default macros may be used to reduce uplink data volume. The commanding will also support maintenance and calibration activities.

4.6.2.3 Data Processor

The C&DH Subsystem collects engineering data on request from the module. These requests are based on the different formats established during the design engineering phase. Similarly the science data is collected and formatted. The module controls both data flow into and out of solid state recorders based on the commands received from the ground segment. The ground system defines start and stop times of contacts. Based on these times and known configurations of antennas, the module establishes links, which are verified by the ground system. After successful data downlinking, the ground will request to overwrite the data in the solid state recorder. The module shall support both X and S band encoded and formatted data links simultaneously or individually.

4.6.2.4 Attitude Processor

The module has a minimum of three functions. The first function is the orbit determination. The ground system uplinks to the satellite parameters characterizing its orbit. This enables the satellite onboard processor to propagate the Earth, the Sun, and the Moon position and velocity with respect to the satellite. This information aids in the maneuvers and pointing of the satellite. The second function is attitude determination. This calculates the orientation of the satellite in real time based on calibrated attitude sensors such as star tracker, gyros, sun sensors, etc. This information is used in target acquisition and tracking, and maneuvers. The third function is control. The module supplies the link among the sensors, the reaction wheels, the propulsion subsystem, and the ground system to perform orbit and attitude maneuvers, attitude control, and momentum dumping.

4.6.2.5 Health and Status Module

The module monitors telemetry for violation of health and safety limits. If the behavior of the telemetry requires safe mode entry, it switches off the power to non-essential loads, puts the satellite in a safe attitude, and awaits instructions from the ground for failure recovery.

4.6.3 <u>Performance</u>

The executive shall perform seamlessly when all the tasks are run simultaneously at their maximum rate. The memory margin under the condition shall be 20%. Under the same conditions, all tasks shall be successfully initiated and completed with positive time margins.

The performance of the other modules shall be as given in the various hardware sections.

4.6.4 <u>Interfaces</u>

The Flight Software resides in the processor card of the C&DH Subsystem. It interfaces with the ground and space segment utilizing processor hardware interfaces.

4.7 Structure

The structural subsystem is illustrated in figure 4.7-1. It provides for the mounting and support of instrument and S/C components. The baseline configuration philosophically adopts a modular approach for an independent instrument module and S/C bus. The following paragraphs describe the functional implementation, the hardware components, and the interfaces of this subsystem.

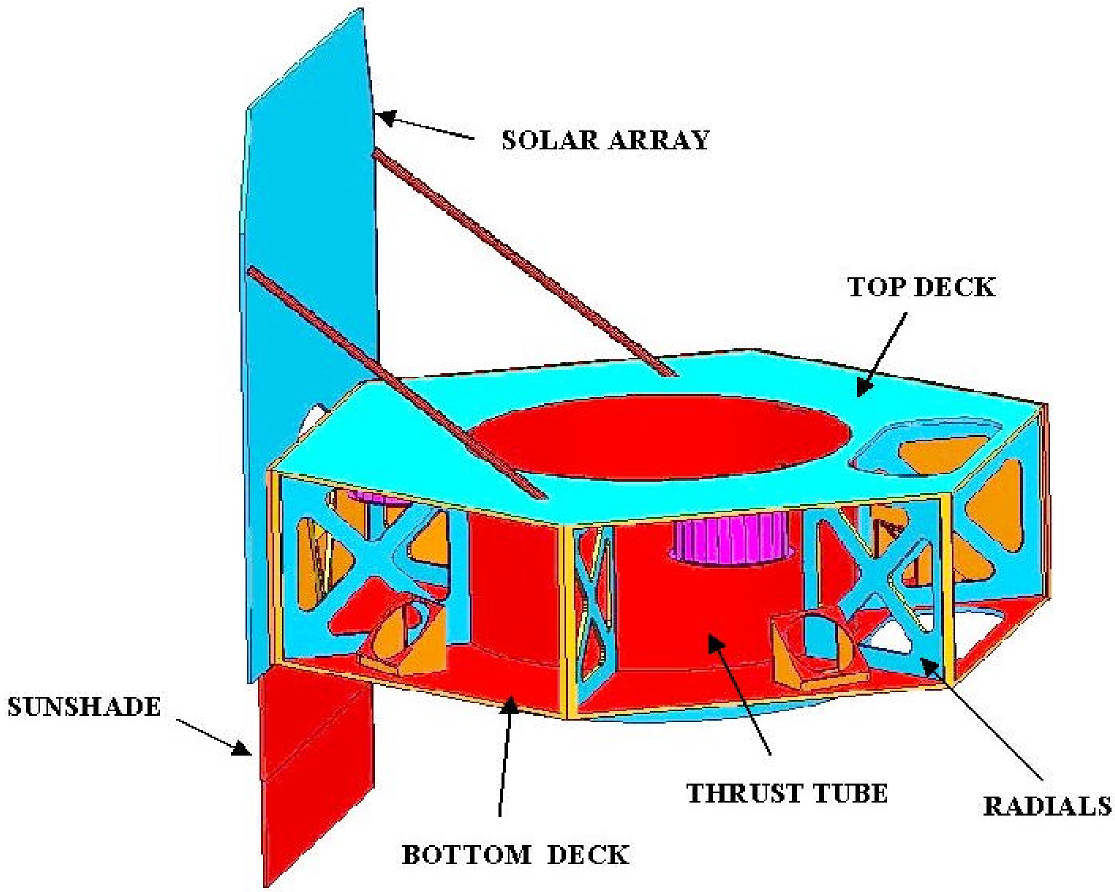


Figure 4.7-1. S/C Structure

4.7.1 Functional Description

The Structure provides several functions to the spacecraft. It provides a structural load path from the instrument module to the launch vehicle I/F to react loads developed during the launch environment. It also provides the mounting platform to the remaining spacecraft subsystem components as well.

A kinematic mounting scheme to the instrument module provides mechanical and thermal isolation for independent developments of the IM and bus structures. With this approach, the bus structure does not reside in the instrument optical path. This will provide a significantly more robust, cost effective system throughout all phases of the program.

4.7.2 Hardware Description

Figure 4.7-1 illustrates the baseline configuration. It is a metallic hex shaped structure using a combination of honeycomb panels, machined panels and fittings. Standard fastening techniques are utilized including bolted, hi-lok, and riveted joints. A central thrust tube provides the axial load path from the IM on top through a marman band payload attach fitting (PAF) at the launch vehicle I/F. Upper and lower decks are honeycomb with machined radial panels providing shear tie back to the thrust tube. Honeycomb panels close out the bays and provide the majority of real estate for locating S/C subsystem component electrical boxes.

Based on current power requirements, the baseline solar array is fixed and body mounted to one of the hex sides.

Mechanisms required include a linearly actuated high gain antenna and a contamination cover over the SXT optic.

A dual manifest launch configuration is the baseline.

4.7.3 Performance Specifications

Table 4.7-1. Requirements (Mission and Derived)

Element	Parameter	Specification	Comments
Hi Gain Antenna	FOV	Linear actuators used to provide 20^o pitch and roll coverage	
SXT Contamination Cover		TBD	
Structure	Quasi-Static Design Loads	***Axial*** +10g/-1g (compression/tension) ***Lateral*** $\pm$ 4g	All combinations of axial and lateral load cases considered
	Center of gravity	TBD	Referenced to LV sep. plane
	Center of Pressure	TBD	Referenced to LV sep. plane
	Inertias	Ixx = TBD Iyy = TBD Izz = TBD Ixy = TBD Ixz = TBD Iyz = TBD	Deployed (on-orbit) configuration
	GSE Handling	Support Observatory lift	Lift point capability must be compatible with mass and geometry for observatory configuration
	SXT/HXT Optic Shade	Provide shading at lower end of structure to accommodate 20^o pitch and roll coverage	

4.7.4 Interfaces

The structure interfaces with the following subsystems:

Attitude Control: reaction wheels (4), electronics, Command and Data Handling, Propulsion, Communication, Power, Thermal, Instrument, and Launch Vehicle.

4.7.5 Assumptions

The structure is compatible with either Delta IV or Atlas V dual manifest configurations. The baseline PAF currently does not exist; it represents a minor modification (mating diameter to the bus) to existing designs. This requirement is driven primarily by the 1.6 meter diameter SXT optic and its field of view requirements.

Although considered S/C components, the star trackers and gyros are mounted on the optic bench along with the instruments.

4.8 Thermal Control System

The Thermal Control System (TCS) ensures that spacecraft temperature requirements are met for all phases of the mission. Heaters, thermostats, insulation blankets, and heat pipes are used to maintain acceptable component temperature levels and structure gradients.

4.8.1 Functional Description

Heat generated by spacecraft components is transmitted to space via radiators on the no-sun sides of the spacecraft. These radiators are sized to maintain spacecraft temperatures without operational heater power on most components. Operational spacecraft power dissipation can be accommodated with less than 50% of available radiator area.

The spacecraft bus is thermally isolated from the HXT and SXT mirrors, since mirror temperatures are tightly controlled. The solar array runs much hotter than other spacecraft components, so it is also thermally isolated from the spacecraft structure.

4.8.2 Hardware Description

Thermal coatings and MLI (multi-layer insulation) blankets maintain most spacecraft components near room temperature. The only heaters needed in operational mode are on propulsion components, which require about 25 watts. Other components are protected during low power conditions by make-up heaters.

Low-conductivity mounts and MLI blankets are used to achieve spacecraft thermal isolation from instrument components and solar array.

5.0 Mission Operations

Figure 4.1-1 shows the mission operations in a block diagram.

5.1 Description

The ground support system software for the Constellation-X will be designed as a single system which will support GSE, I&T and operational functions, and will be built in an evolutionary manner to avoid duplication of functions and effort. The TLM and CMND functions will use COTS based products that will be capable of handling multiple simultaneous TLM streams for periods of time during I&T and launch and early orbit (during the first and second launches) where simultaneous communications may be required. CMND and TLM databases can then be easily combined as the spacecraft are assembled and tested. This software will have incorporated a state modeling engine which will allow unattended data collection, analysis, notification, and CMND response as needed to assist in reducing operations staff and retaining overall spacecraft knowledge. Commanding and command management functions should be uncomplicated and should be able to be accommodated by the COTS TLM/CMND system.

Trending of TLM and ancillary data will likewise be performed using a COTS product. This tool will have the capability to plot multiple telemetry parameters across multiple spacecraft (up to four). The data will be able to be aligned by time, event or environment. There will also be the capability to compare telemetry taken from I&T with post launch data. Capabilities required are available today and should be available as a COTS product by the time of launch.

Scheduling for the spacecraft should be limited to contact times for station acquisition during launch and early orbit, normal operations, and station keeping activities. This can be accomplished by almost any COTS scheduling product.

Level Zero Processing should not be required since the nominal plan is to routinely downlink one days worth of science TLM once per day, and will therefore not require assembly or time ordering. Data will be bent piped to the science data processing area for level 1 and above processing.

Current plans are for single shift operations for the constellation for normal routine operations. This would require the use of a single tracking station for coverage at L2. The four spacecraft will be contacted serially (1 hour each) every day to download science data, uplink CMNDs as needed and collect metric tracking data. There is also special tracking coverage for approximately 100 days after launch while the spacecraft is in transit to mission orbit and before, during and after station keeping maneuvers to maintain an L2 orbit. A trade study will need to be performed approximately 2 years prior to launch to determine whether to buy tracking services or a dedicated tracking dish.

5.2 Antenna Management

It is assumed at this point that there will be a 1.2 meter antenna onboard the S/C and that antenna maneuvers of up to 20 degrees will be required prior to and after TLM dumps when the S/C is at selected attitudes. These maneuvers will be preprogrammed to occur automatically with no ground intervention. This will require knowledge of the S/C position and attitude onboard.

5.3 Onboard Clock Maintenance

The satellite maintains a clock with UTC time accurate to 10 msecs, synchronized everyday by ground operations. The ground system will have a clock with UTC time accurate to 1 microsecond using a GPS receiver.

5.4 Performance Specifications

For routine operations, each spacecraft will require a one-hour station contact. It is assumed the spacecraft will be able to simultaneously receive real-time and recorder telemetry data while being able to command or receive tracking data. Four consecutive contacts must be scheduled contiguously, however the real-time monitoring and commanding system must be able to handle up to four contacts simultaneously. The set up time for the station is assumed to take not more than 20 minutes for the initial spacecraft contact and not more than 5 minutes between spacecraft contacts. Ninety-nine percent (TBR) of the onboard recorded science data collected must be recovered on the ground in order to achieve the overall 90% science data capture requirement. Post pass delivery of recorder data to the data center shall not exceed 24 hours (TBR) after the end of the last of the four passes each day. Orbit determination accuracy is best achieved with 30 minutes of tracking data per day. A minimum of 10 minutes is required during routine operations. One hour (TBR) per day per spacecraft is required during the transfer orbit, and 12 hours of tracking data for each station-keeping maneuver (4 hours before, 8 hours after).

5.5 Operations

Routine operations assume that each spacecraft will be contacted once per day for one hour each, and that the contacts will be taken back-to-back. This will allow for an 8 x 7 shift for routine operations. During each launch and early orbit phases of the mission, there will be full shift coverage until stability is achieved. During the transfer orbit and during orbit maintenance activities, staffing will be commensurate with planned activities. During these non-routine times, simultaneous spacecraft contacts may be scheduled, especially during the launch and early orbit phases. The mission is expected to span five years of operations with a nominal three-year overlapping period for all four spacecraft, and the first and fifth years supporting two spacecraft. Definitive orbit determination will be performed on the ground. Predictive and definitive results will be distributed to the science data processing and the control center. The data will be used for planning and up linking to control the spacecraft and to point the onboard antenna. Science data commanding is expected to occur at a frequency of approximately once per week, but should be able to be handled during routine contacts.

5.6 Interfaces

The primary interfaces for the ground system are with the tracking station, which will use CCSDS (or equivalent) over IP.

Acronyms

AC	Alternating Current
ACHE	ADR Control and Housekeeping Electronics
ACS	Attitude Control Subsystem
AGN	Active Galactic Nuclei
AMP	Amplifier
ADR	Adiabatic Demagnetizing Refrigerator
AHI	Ampere Hour Integrator
ASIC	Application-specific Integrated Circuit
BGO	Bismuth Germanium
BOL	Beginning of Life
BPSK	Binary Phase Shift Key
BRF	Bound Reject Filter
CADU	Channel Access Data Unit
CAN	Cooperative Agreement Notice
CCD	Charge-Coupled Device
CCSDS	Consultative Committee for Space Data Systems
C&DH	Command and Data Handling
CdZnTe	Cadmium Zinc Telluroid
CMND	command
COTS	Commercial Of f the Shelf
CRC	Cyclic Redundancy Code
CTE	Coefficient of Thermal Expansion
CZT	Cadmium Zinc Telluride
DB	Decibel
DC	Direct Current
DET	Direct Energy Transfer
DSN	Deep Space Network
EELV	Evolved Expendable Launch Vehicle
EDAC	Error Detection and Correction
ELV	Expendable Launch Vehicle
EMI/EMC	Electro-Magnetic Interference/Electro Magnetic Compatability
EOB	Extendible Optical Bench
EOL	End of Life
EPS	Electrical Power Subsystem
FIFO	First in First Out
FOV	Field of View

FTS	Fiducial Transfer System
FWHM	Full Width at Half Maximum
GSE	Ground Support Equipment
HPD	Half Power Diameter
HXT	Hard X-ray Telescope
HVPS	High Voltage Power Supply
I/F	Interface
I&T	Integration and Test
IM	Instrument Module
IP	Internet Protocol
kbps	Kilobits Per Second
Khz	Kilo Hertz
KSC	Kennedy Space Center
LVPC	Low Voltage Power Converter
LVPS	Low Voltage Power Supply
Mhz	Mega Hertz
MIB	Mission Integration Branch
MIDEX/MAP	Medium Explorer/Microwave Anisotropy Probe
MLI	Multilayer Insulation
ms	milliseconds
NIST	National Institute of Standards and Technology
PAF	Payload Adapter Fitting
PCE	Power Conversion Electronics
PCM	Pulse Code Modulation
PMT	Photo Multiplier Tube
PS	Psuedo-Random
PSE	Power System Electronics
PSF	Point Spread Function
psi	pounds per square inch
PSK	Pulse Shift Key
PWM	Pulse Width Modulation
RF	Radio Frequency
RS	Reed-Solomon
SA	Solar Array
S/C	Spacecraft

SOA	State of Art
SPV	Single Pressure Vessel
SXT	Spectroscopy X-ray Telescope
TBD	To Be Determined
TBR	To Be Resolved
TBS	To Be Supplied
TCC	Trickle Charg e Controller
TCS	Thermal Control System
TES	Transition Edge Sensor
TLM	Telemetry
TTI	Thrust Termination Injection point of the launch vehicle
UTC	Universal Time Code
UV	Ultra Violent
VDC	Volts Direct Currents
VLSI	Very Large Scale Integrated Circuit
VT	Voltage/Temperature
XMM	X-ray Multi-mirror Mission

Appendix - A

A.1 Grazing Incidence Optics

This appendix provides additional information regarding the theory of grazing incidence X-ray optics.

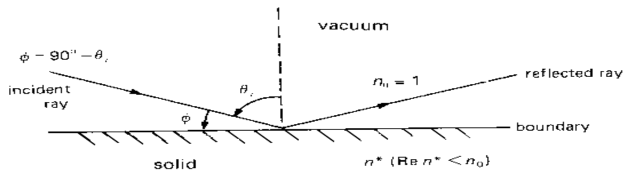
X-rays are photons and therefore follow the laws of optics for reflection and refraction although the indices are complicated expressions and vary as a function of X-ray energy and material absorption edges. However, it turns out that for X-rays in the Constellation-X bandwidth, the index of refraction of candidate mirror coating materials is less than unity. This means that for small grazing angles of incidence, these rays are at least partially reflected. Figure A-1 shows the main concepts as well as the complexity of the index of refraction.

As the angle of incidence becomes greater (grazing angle becomes less), fewer X-rays are absorbed in the reflector or, if it is thin, pass through it. The critical angle (maximum grazing angle for complete reflection) becomes smaller and smaller as the energy of the incident X-ray increases. However, as the grazing angle decreases, the projected area of the reflection surface also decreases and so there is a practical limit as to how shallow a grazing angle can be. This, in turn, limits the high energy response.

Figure A-2 shows an idealized response of a mirror as a function of grazing angle and energy.

Not all of the incident rays are reflected due to a variety of losses. The fraction of incident rays that are properly reflected is called the reflectivity of the reflecting surface. Reflectivity is a complicated function of coating material properties and the energy of the X-ray. Figure A-3 is a plot of reflectivity of gold and platinum for different grazing angles as a function of incident X-ray energy. These data only show relative effects. The top (and best) curve is for a grazing incidence angle of 30 arc minutes which is representative for the outer shells of the SXT. Note that the response is quite flat from about 1 eV to 20 keV or so.

Gold is the baseline coating material for the Constellation-X mirrors. It is a good coating material because it has a flat characteristic in the energy range of interest. It also is easily applied and has excellent stability. Platinum and Iridium have some attractive properties but may not be as suitable from either cost or technical considerations.



In the X-ray band the complex refractive index n^* is usually expressed as:

$$n^* = (1 - \delta) - i\beta,$$

with

$$\delta = \frac{1}{2\pi}\left(\frac{e^2}{m_e c^2}\right)\left(\frac{N_0 \rho}{A}\right) f_1 \lambda^2$$

and

$$\beta = \mu_a \lambda = \frac{1}{2\pi}\left(\frac{e^2}{m_e c^2}\right)\left(\frac{N_0 \rho}{A}\right) f_2 \lambda^2,$$

where

$$\frac{e^2}{m_e c^2} = r_e = \text{the classical electron radius},$$

N_0 = Avogadro's number,
ρ = density,
A = atomic weight,
λ = wavelength of the incident radiation,
f_1 = real part of the atomic scattering factor, and
f_2 = imaginary part of the atomic scattering factor.

$$f_1 = (\pi r_e h c)^{-1} \int_0^{\infty} \frac{E'^2 \mu_a(E')\, dE'}{E^2 - E'^2} + Z,$$

$$f_2 = \frac{\pi}{2} (\pi r_e h c)^{-1} E \mu_a,$$

where

Z = atomic number,
h = Planck's constant,
c = velocity of light,
E = incident photon energy, and
μ_a = atomic photoabsorption cross-section.

Figure A-1. X-ray Reflection (from Zombeck, *Handbook of Space Astronomy and Astrophysics*, 2nd Edition, Cambridge University Press, 1990)

Calculated specular reflectivity of an ideal surface as a function of normalized grazing angle ϕ/ϕ_c for various values of β/δ. (After Hendrick, *JOSA*, **47**, 165, 1957.)

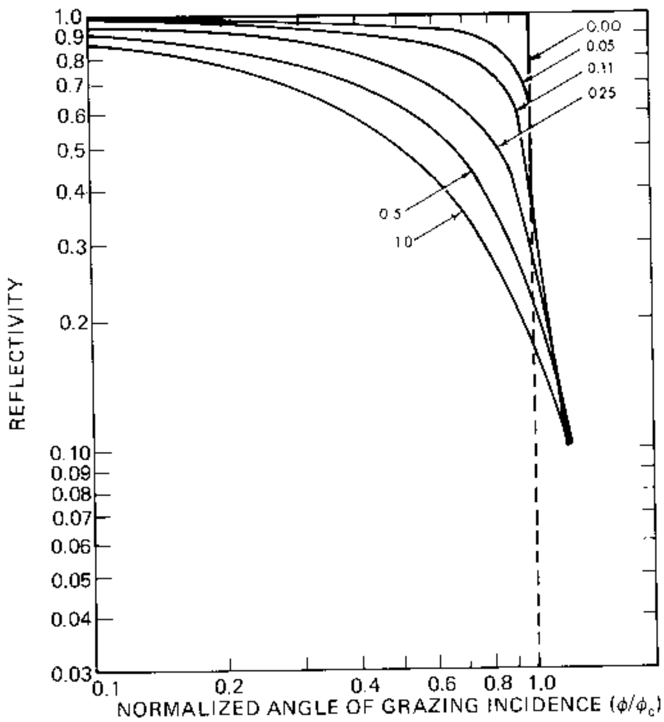


Figure A-2. Idealized Reflectivity (from Zombeck, *Handbook of Space Astronomy and Astrophysics*, 2nd Edition, Cambridge University Press, 1990)

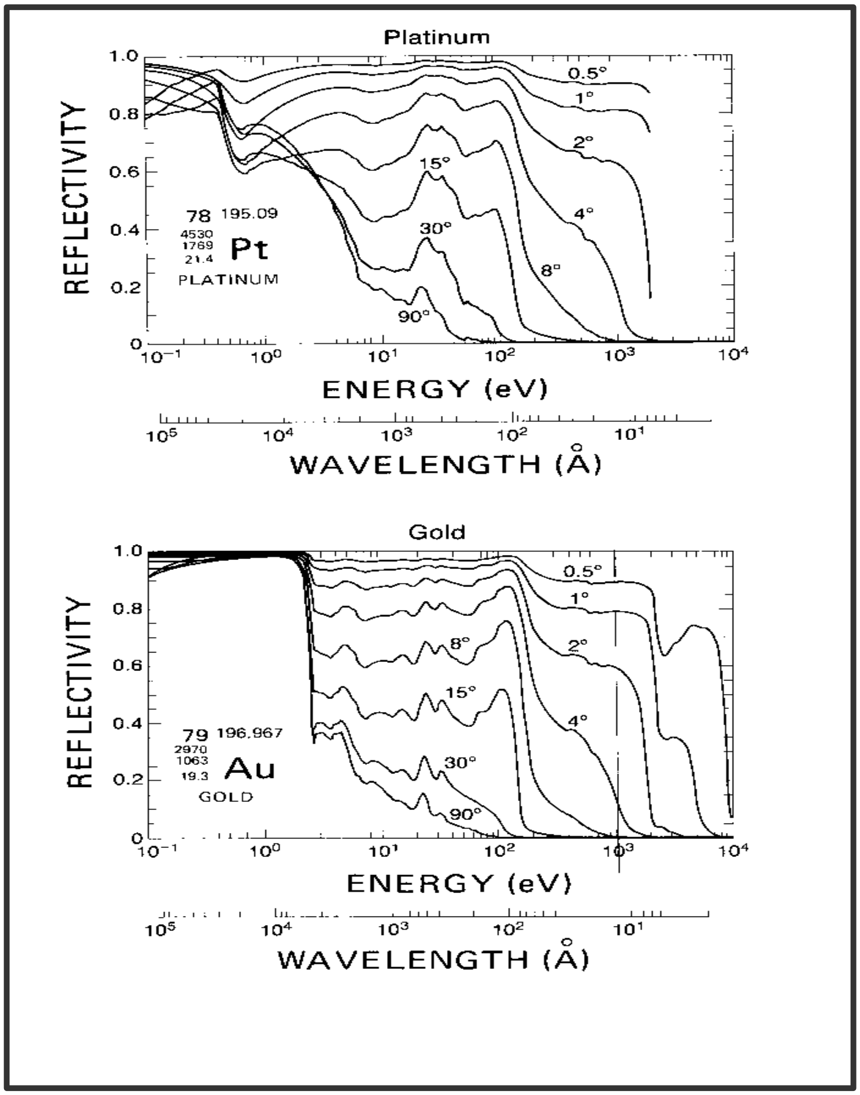


Figure A-3. Reflectivity of Gold and Platinum vs. Energy and Grazing Angle (from Zombeck, *Handbook of Space Astronomy and Astrophysics*, 2nd Edition, Cambridge University Press, 1990)

A.2 Description Semiconductor X-ray Calorimeter

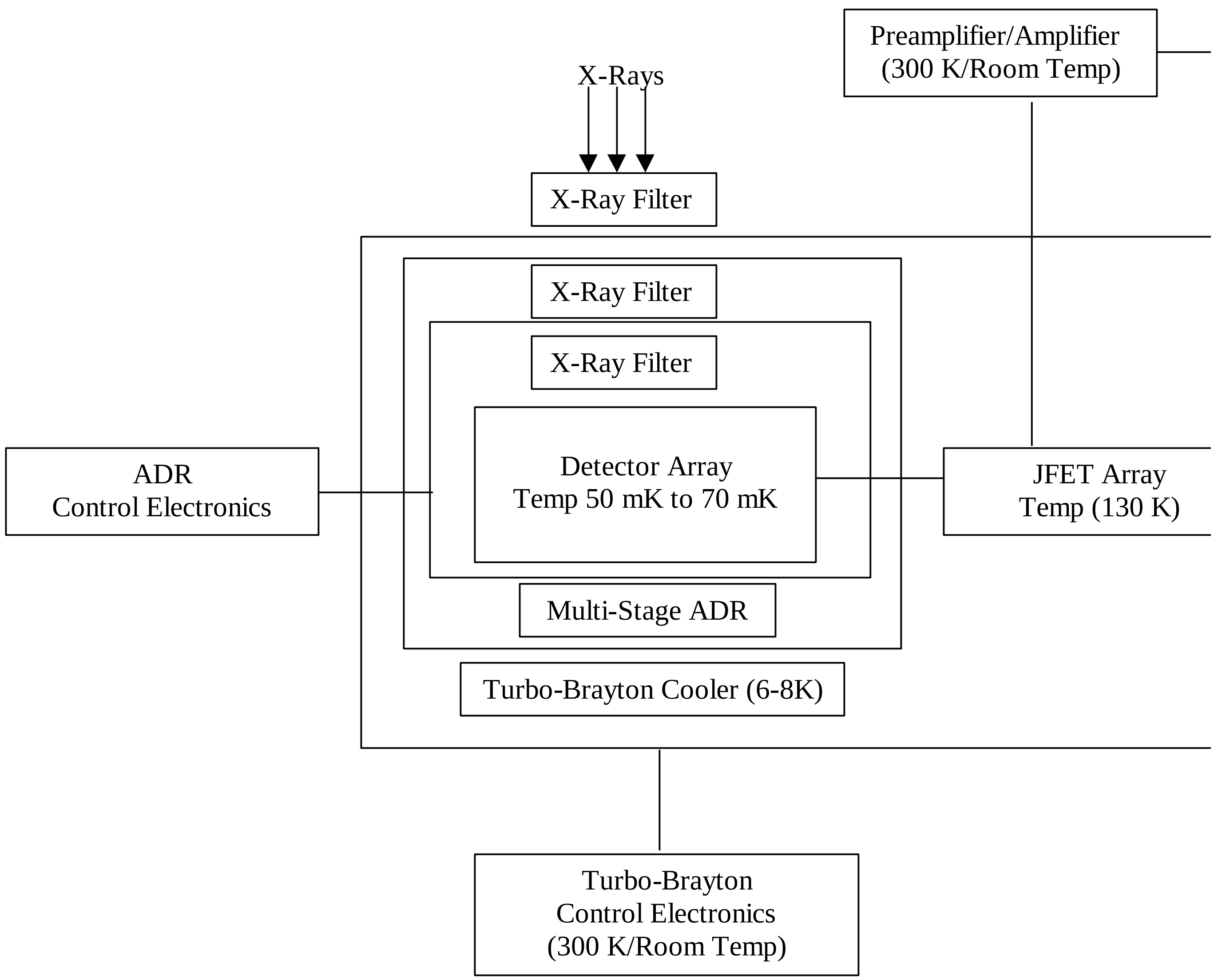


Figure A-4. Block Diagram for Semiconductor X-ray Calorimeter

Appendix - B

Communication Link Analysis

Table B-1. S/C System Temperature

```
TABLE S/C SYSTEM TEMP = 50 UPLINK        DATE & TIME:  6/14/99 15:18:23
THE CONSTELLATION-X
FREQUENCY - 2071.800 MHZ
GROUND ANTENNA - - - 11 METER
POWER -    2.0000 K WATTS
CONVOLUTIONAL ENCODING
```

PARAMETERS	UNITS	VALUES (MAX RNG: 1535514. KM 5.0 EL)	VALUES (MIN RNG: 1500000. KM 90.0 EL)	ESTIMATED TOLERANCES DB FAV	ESTIMATED TOLERANCES DB ADV
EFFECTIVE RADIATED POWER	DBM	106.0	106.0	1.0	-1.0
FREE SPACE DISPERSION LOSS	DB	-222.5	-222.3	.0	.0
ATMOSPHERIC LOSS	DB	-.5	.0	.0	.0
POLARIZATION LOSS	DB	-3.0	-3.0	.0	.0
SPACECRAFT ANTENNA GAIN	DBI	.0	.0	.0	.0
SPACECRAFT PASSIVE LOSS	DB	-1.0	-1.0	.1	-.1
MAXIMUM TOTAL RECEIVED POWER	DBM	-121.0	-120.3	1.0	-1.0
SPACECRAFT ANTENNA NULL DEPTH	DB	.0	.0	.0	.0
MINIMUM TOTAL RECEIVED POWER	DBM	-121.0	-120.3	1.0	-1.0
SYSTEM NOISE DENSITY	DBM/HZ	-171.6	-171.6	.0	.0
IF NOISE BANDWIDTH(3000.000 KHZ)	DB-HZ	64.8	64.8	.0	.0
IF NOISE POWER	DBM	-106.8	-106.8	.0	.0
IF SNR (MIN)	DB	-14.2	-13.5	1.0	-1.0
CARRIER CHANNEL					
CARRIER/TOTAL POWER	DB	-2.3	-2.3	.2	-.2
RECEIVED CARRIER POWER	DBM	-123.3	-122.6	1.0	-1.0
CARRIER LOOP NOISE BW(800. HZ)	DB-HZ	29.0	29.0	.0	.0
NOISE POWER	DBM	-142.6	-142.6	.0	.0
CARRIER/NOISE	DB	19.3	20.0	1.0	-1.0
REQUIRED CARRIER/NOISE	DB	15.0	15.0	.0	.0
AVAILABLE CARRIER MARGIN	DB	4.3	5.0	1.0	-1.0
REQUIRED PERFORMANCE MARGIN	DB	3.0	3.0	.0	.0
NET MARGIN	DB	1.3	2.0	1.0	-1.0
COMMAND CHANNEL (PCM/PSK/PM)					
COMMAND/TOTAL POWER(MI=1.00 RAD)	DB	-4.1	-4.1	.4	-.4
RECEIVED COMMAND POWER	DBM	-125.1	-124.4	1.1	-1.1
PREDETECTION (PSK) NOISE BW(32.000 KHZ)	DB-HZ	45.1	45.1	.0	.0
PREDETECTION (PSK) NOISE POWER	DB	-126.5	-126.5	.0	.0
PREDETECTION (PSK) SNR	DB	1.4	2.1	1.1	-1.1
COMMAND DATA RATE (2.000KBPS)	DB-BPS	33.0	33.0	.0	.0
AVAILABLE ENERGY PER BIT/NOISE DENSITY	DB	13.5	14.2	1.1	-1.1
DECODER DEGRADATION	DB	-2.0	-2.0	.0	.0
REQUIRED ENERGY PER BIT/NOISE DENSITY (BER=E-5)	DB	5.2	5.2	.0	.0
AVAILABLE COMMAND MARGIN	DB	6.3	7.0	1.1	-1.1
REQUIRED PERFORMANCE MARGIN	DB	3.0	3.0	.0	.0
NET MARGIN	DB	3.3	4.0	1.1	-1.1

1
UPLINK NET MARGIN SUMMARY TABLE

Table B-2. Downlink

1

TABLE A-1 DOWNLINK DATE & TIME: 6/14/99 14:56:55
THE CONSTELLATION-X
FREQUENCY - 2250.000 MHZ
GROUND ANTENNA - - - 11 METER
POWER - 20.00 WATTS
OMNI S/C ANTENNA
CONVOLUTIONAL RATE 1/2 AND R-S ENCODED

PARAMETERS	UNITS	VALUES (MAX RNG: 1535514. KM 5.0 EL)	VALUES (MIN RNG: 1500000. KM 90.0 EL)	ESTIMATED TOLERANCES DB FAV	ESTIMATED TOLERANCES DB ADV
TOTAL TRANSMITTER POWER	DBM	43.0	43.0	1.0	-1.0
SPACECRAFT PASSIVE LOSSES	DB	-1.0	-1.0	.1	-.1
SPACECRAFT ANTENNA GAIN	DBI	.0	.0	.0	.0
FREE SPACE DISPERSION LOSS	DB	-223.2	-223.0	.0	.0
ATMOSPHERIC LOSS	DB	-.5	.0	.0	.0
STDN ANTENNA GAIN (EFFECTIVE)	DBI	45.7	45.7	.0	.0
COMBINER LOSS	DB	-.5	-.5	.0	.0
POLARIZATION LOSS	DB	.0	.0	.0	.0
MAXIMUM TOTAL RECEIVED POWER	DBM	-136.5	-135.8	1.0	-1.0
SPACECRAFT ANTENNA NULL DEPTH	DB	.0	.0	.0	.0
MINIMUM TOTAL RECEIVED POWER	DBM	-136.5	-135.8	1.0	-1.0
SYSTEM NOISE DENSITY	DBM/HZ	-176.8	-178.0	.0	.0
IF NOISE BW(450.000 KHZ)	DB-HZ	56.5	56.5	.0	.0
IF SNR (MIN)	DB	-16.2	-14.3	1.0	-1.0
CARRIER CHANNEL					
CARRIER/TOTAL POWER	DB	-8.8	-8.8	.9	-.9
RECEIVED CARRIER POWER (MIN)	DBM	-145.3	-144.6	1.3	-1.3
CARRIER LOOP NOISE BANDWIDTH(20. HZ)	DB-HZ	13.0	13.0	.0	.0
NOISE POWER	DBM	-163.8	-165.0	.0	.0
CARRIER/NOISE	DB	18.5	20.4	1.3	-1.3
REQUIRED CARRIER/NOISE	DB	15.0	15.0	.0	.0
AVAILABLE CARRIER MARGIN	DB	3.5	5.4	1.3	-1.3
REQUIRED PERFORMANCE MARGIN	DB	3.0	3.0	.0	.0
NET MARGIN	DB	.5	2.4	1.3	-1.3
DATA CHANNEL (PCM/PM)					
DATA/TOTAL POWER(MI=1.20 RAD)	DB	-.6	-.6	.1	-.1
RECEIVED DATA POWER (MIN)	DBM	-137.1	-136.4	1.0	-1.0
INFORMATION RATE(2.000 KBPS)	DB-BPS	33.0	33.0	.0	.0
AVAILABLE SIGNAL/NOISE DENSITY	DB-HZ	39.7	41.6	1.0	-1.0
REQUIRED ENERGY PER BIT/NOISE DENSITY (BER= E-5)	DB	2.9	2.9	.0	.0
REQUIRED SIGNAL/NOISE DENSITY	DB-HZ	35.9	35.9	.0	.0
AVAILABLE SIGNAL MARGIN	DB	3.8	5.7	1.0	-1.0
REQUIRED PERFORMANCE MARGIN	DB	3.0	3.0	.0	.0
NET MARGIN	DB	.8	2.7	1.0	-1.0

Table B-3. Downlink Margin Calculation

GSFC C.L.A.S.S. ANALYSIS #1 DATE & TIME: 6/14/99 14:39:11 PERFORMED BY: R. VENTO
LINKID: THE CONSTELLATION-X

FREQUENCY: 8250.0 MHz RANGE: 1535514.0 km

MODULATION: BPSK
DATA RATE: 1700.000 kbps
CODING: RATE 1/2 WITH RS ENC. & RS INT.
BER: 1.00E-05

CLEAR SKY CONDITIONS
1.1 METER S/C ANTENNA

PARAMETER	VALUE	REMARKS
01. USER SPACECRAFT TRANSMITTER POWER - dBW	6.99	NOTE A; 5.0 WATTS
02. USER SPACECRAFT PASSIVE LOSS - dB	1.00	NOTE A
03. USER SPACECRAFT ANTENNA GAIN - dBi	37.72	NOTE A
04. USER SPACECRAFT POINTING LOSS - dB	.00	NOTE A
05. USER SPACECRAFT EIRP - dBWi	43.71	1 - 2 + 3 - 4
06. POLARIZATION LOSS - dB	.20	NOTE A
07. FREE SPACE LOSS - dB	234.50	NOTE B; ALT: 1529705.0 KM, EL: 5.0 DEG
08. ATMOSPHERIC LOSS - dB	.50	NOTE B
09. RAIN ATTENUATION - dB	.00	NOTE A
10. MULTIPATH LOSS - dB	.00	NOTE A
11. GROUND STATION ANTENNA GAIN - dBi	56.97	NOTE B; 11.0 M, EFF: 55.0%
12. GROUND STATION PASSIVE LOSS - dB	.00	NOTE A
13. GROUND STATION POINTING LOSS - dB	.00	NOTE A
14. SYSTEM NOISE TEMPERATURE - dB-DEGREES-K	23.01	NOTE A
14. GROUND STATION G/T - dB/DEGREES-K	33.96	11 - 12 - 13 - 14
16. BOLTZMANN'S CONSTANT - dBW/(Hz*K)	-228.60	CONSTANT
17. RECEIVED CARRIER TO NOISE DENSITY - dB/Hz	71.07	5 - 6 - 7 - 8 - 9 - 10 - 11 + 15 - 16
18. MODULATION LOSS - dB	.00	NOTE A
19. DATA RATE - dB-bps	62.30	NOTE A
20. DIFFERENTIAL ENCODING/DECODING LOSS - dB	.00	NOTE A
21. USER CONSTRAINT LOSS - dB	.00	NOTE A
22. RECEIVED Eb/No - dB	8.76	17 - 18 - 19 - 20 - 21
23. IMPLEMENTATION LOSS - dB	2.00	NOTE A
24. REQUIRED Eb/No - dB	2.90	NOTE A
25. REQUIRED PERFORMANCE MARGIN - dB	.00	NOTE A
26. MARGIN - dB	3.86	22 - 23 - 24 - 25

NOTE A: PARAMETER VALUE FROM USER PROJECT - SUBJECT TO CHANGE
NOTE B: FROM CLASS ANALYSIS IF COMPUTED