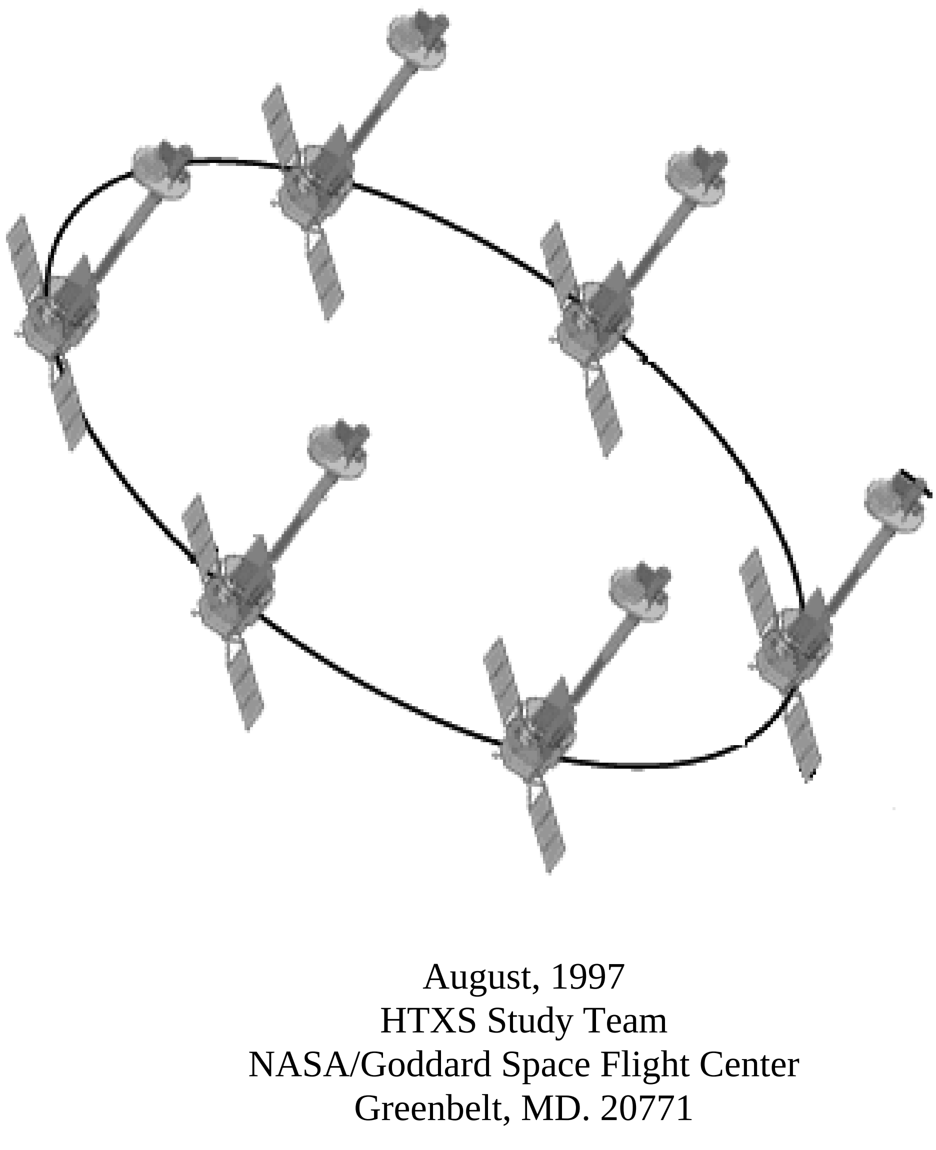
***H**igh*
***T**hroughput*
***X**-ray*
***S**pectroscopy*



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HTXS SPACECRAFT AND MISSION CONCEPT STUDY REPORT

TABLE OF CONTENTS

List of Figures and Tables

List of Acronyms

ACS	Attitude Control System
ADR	Adiabatic Demagnetization Refrigerator
CCD	Charged Coupled Device
C&DH	Command and Data Handling
CTE	Coefficient of Thermal Expansion
DSN	Deep Space Network
EOB	Extendible Optical Bench
EOL	End Of Life
GSFC	Goddard Space Flight Center
HTXS	High Throughput X Ray Spectroscopy
HXT	Hard X Ray Telescope
I&T	Integration and Test
LV	Launch Vehicle
MAP	Microwave Anisotropy Probe
MOC	Mission Operation Center
RF	Radio Frequency
SOC	Science Operation Center
S/C	Spacecraft
SSR	Solid State Recorder
SXT	Soft X Ray Telescope
XDS	X Ray Spectrometer (XRS) Detection System
WOTS	Wallops Orbital Tracking Station

INTRODUCTION

The scientific objective of the HTXS investigation is the study of life cycles of matter in the universe in processes where X-ray emission dominates the energy output or where the life cycles are otherwise inaccessible. It is the next generation X-ray observatory to follow ASTRO-E, AXAF, and XMM and is dedicated to high throughput observations at high spectral resolution from 0.25KeV to 40 KeV. This report describes a feasible mission concept to support the HTXS program. While there is no intent to establish a point design, the baseline described in this report was developed to show technical feasibility and provide a basis for cost, weight, and schedule estimates. The results represent a concept which satisfies all science requirements currently defined and indicates the mission can be accomplished within the scope of approximately $500 M (FY97 dollars). The baseline concept developed does not represent an optimized design. With limited resources, the study concentrated at this stage (pre-phase A) on developing a system approach which led to a reliable and robust engineering design and the identification of key technology developments which must be pursued for the mission to be viable. This report discusses some of the trade studies which have been made to establish the baseline concept. Final decisions in each trade study were based on simplicity, mass, and cost where appropriate.

In the instrument module configuration described, HTXS uses two sets of telescope systems with focal lengths of approximately 8.5meters. The low energy spectroscopy telescope will use highly-nested lightweight grazing incidence X-ray optics coupled to an array of micro-calorimeters which cover the 0.25 to 10 keV band plus a reflection grating/CCD system covering the 0.25 to 2 keV band. The calorimeter array is cryogenically cooled. At high energies, HTXS will employ three multilayer-coated grazing incidence optics coupled to a solid state CdZnTe or similar imaging spectrometer to cover the 6 to 40 keV band.

To meet the mission objectives, technology developments will be necessary in a few key areas including high resolution, lightweight X-ray optics; X-ray micro-calorimeters; reflection gratings; CCDs; advanced cryogenic coolers; hard X-ray focusing optics; and extendible optical benches with high stability. Details of the technology development hardware have been provided in the HTXS Technology Roadmap (February 28, 1997), and are not included in this document.

1.0 MISSION OVERVIEW

1.1 Observatory

The major mission requirement is the need for large photon collecting apertures for the X-ray telescopes. The necessary collecting area is obtained by using six identical, low cost, five year satellites that simultaneously view the same target, dwelling up to thirty hours on low intensity targets. Figure 1 depicts the observatory mission configuration and Table 1 summarizes the basic mission characteristics. The L2 Lagrange orbit has been selected to avoid interference from earth thermal variations and X-ray source occultations. The telescope bore sight is restricted to the region extending from 70^{o} to 110^{o} from the earth-sun line. This restriction minimizes thermal variations and permits the full sky to be surveyed every six months. Each spacecraft (S/C) will be kept within 300,000 km of the earth-sun line, thereby limiting the S/C antenna bore sight angle with the earth to be within $\pm 35^{o}$. Smaller orbits would require a more precise injection, which would require more orbit acquisition propellant. There is no requirement on the relative positions of the six S/C. These S/C will be sequentially launched within a 2 year period.

1.2 Orbit

The L2 Lagrange Point is about 1.5 million km anti-sunward from the earth. The HTXS observatory orbit will be a Lissajous pattern with a six month period. Table 2 summarizes three possible options for the trajectory to deliver each S/C to the L2 orbit. A Lunar Assist with 2.5 or 3.5 phasing loops was selected to minimize fuel requirements and maximize the time from launch until the first maneuver while providing a reasonable monthly launch window and arrival time at L2. Figure 2 shows this trajectory in a solar rotating coordinate frame. The number of loops depends on the epoch of the launch date and the launch vehicle velocity error. Phasing loops have heritage from GEOTAIL, WIND, Hiten, and MAP. Through the use of phasing loops and a Lunar Assist, the S/C will enter the L2 orbit with no final thruster firings. Once on orbit, stationkeeping thrusting will be performed frequently, as soon as the firing direction is clearly determined. Both earth and lunar eclipses may occur, however proper stationkeeping will avoid earth eclipses, and the battery will support the S/C during lunar eclipses. At the L2 orbit distance only lunar penumbras occur, with only about 10% reduction in sunlight. Stationkeeping thrusting will be coordinated with angular momentum management.

1.3 Spacecraft Concept

The S/C bus and instrument module described in detail in paragraphs 3.0 and 4.0 are the baseline design for the S/C. The baseline represents a systems approach considering development, integration, and simplicity. This ultimately reduces the cost significantly. The robustness of the design will provide flexibility as requirements change. The extendible optical bench (EOB), long life cryostat and accommodation of mirrors and detectors are the most challenging designs and have been given the most attention to date. A new electronically steered X-band phased array antenna system, being developed for the first New Millennium Mission, will meet the science data playback requirements. Other components are well within existing state-of-the-art designs. Preliminary estimates for mass and power are based on heritage, experience, and engineering judgment. They are not derived from a detailed design. These values are summarized in Tables 3 and 4. This baseline design meets all currently known requirements and provides sufficient detail to make a reasonable cost and schedule estimate.

Observatory Mission Configuration

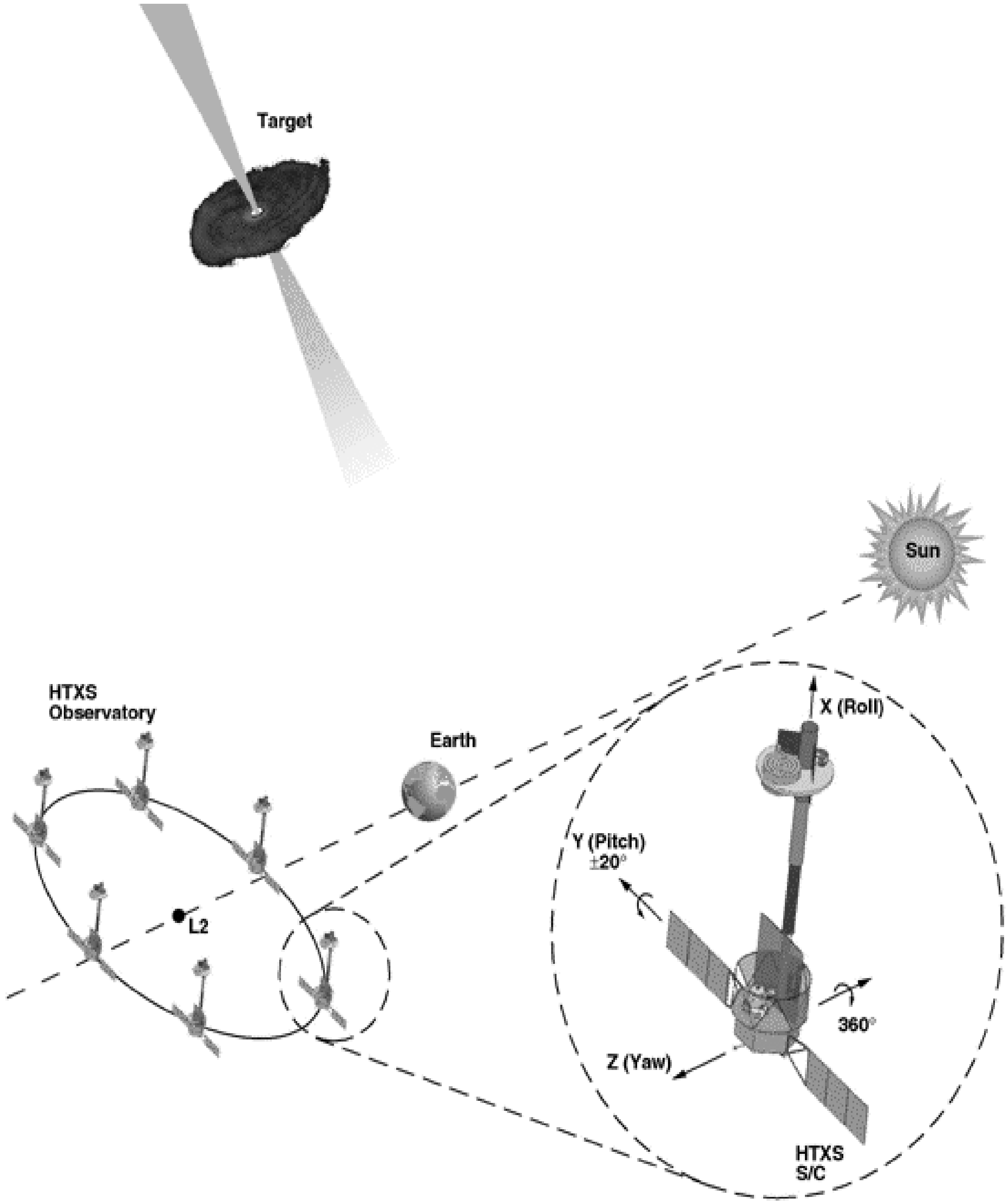

Figure 1

Baseline System Description

ITEM	HTXS
Mission Description	High Resolution Measurement of High Energy X Rays (0.25 to 40 KeV). Full sky coverage every six months
Observatory Configuration	Six Spacecraft (launched on 4 month centers)
Orbit	L2 Halo
Mission Life/Spacecraft Life	3 Years/5 Years
Launch Vehicle	Delta 7925H Class (3 stages)
Mass: Instrument Module / Spacecraft	940 Kg / 450 Kg
Power Generation Storage	 1400 Watts EOL / Two-Fixed Arrays Low Cycle Life, Low-Duty Cycle Battery
C&DH Science Data Rate Spacecraft Processor / Data Architecture Instrument Mass Memory Command Uploads	 20 kbps average, 500 Kbs maximum 32-bit R-3000/MIL-STD 1553 4 Gbit SSR, directly addressable. Autonomous operation, once per day uplink
RF Communications Playback Data Rate Antennas	 512 kbps Electronically Steerable High Gain X-band / Omni S-band/Medium gain S-band
Propulsion	84 kg Hydrazine
Attitude Control Type Pointing Knowledge	 3-axis stabilized, Inertial Ptg, Star Tracker, Sun Sensor, 3 reaction wheels, gyros and thrusters 2 arc seconds pitch & yaw, 20 arc seconds roll
Cryogenics	0.065^{o}K Three-Stage Cryogenic Cooler
Thermal	Passive radiators
Mechanical	Extendible Optical Bench of Composite Spacecraft Bus of Aluminum
Ground Station: Quantity/ Location/ Contacts	One in USA / one contact-per-day-per-spacecraft

Table 1

Transfer Option Summary

Parameter	**Direct Transfer**	**Lunar Assist without Phasing Loops**	**Lunar Assist with Phasing Loops**
Launch Opportunities	app. 27 days/month	1 day/month	app. 14 days/month
Daily Launch Window	5-10 minutes	5-10 minutes	20 minutes
Launch Vehicle C3* (TTI)	-0.70 km^2/sec^2	-1.90 km^2/sec^2	-2.60 km^2/sec^2
Time from TTI to First Required Maneuver	5-7 days	4-12 hours	app. 8 days
Time from TTI to L2	100 days	105 days	130 days
Phasing maneuver ΔV's	0 m/s	0 m/s	70 m/s
Post-TTI Error Correction Maneuver ΔV	5 m/s	75-100 m/s	5 m/s
Mid-course Maneuver ΔV	10 m/s	10 m/s	10 m/s
L2 Orbit Insertion ΔV	200 m/s	0 m/s	0 m/s
Station Keeping ΔV at L2 (5 years)	20 m/s	20 m/s	20 m/s
Total ΔV Budget for Spacecraft Propulsion System	235 m/s	130 m/s	105 m/s

TTI is Transfer Trajectory Injection

*C3 is double the combined potential and kinetic energy per unit mass at TTI. Smaller values of C3 yield a larger payload capability.

Table 2

Transfer Orbit Trajectory

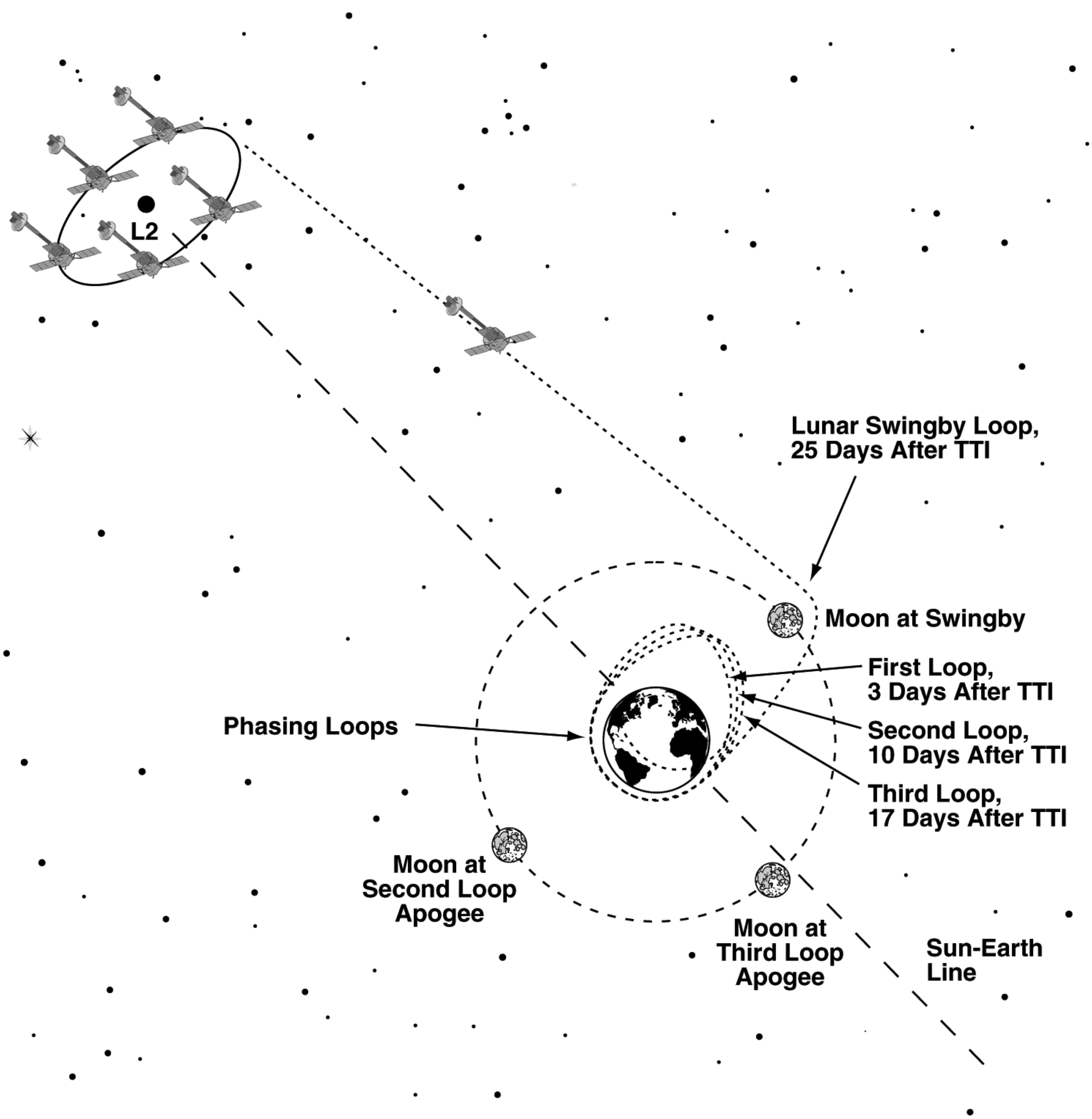

Figure 2

Estimated Mass Summary

ITEM	MASS (KG)
ACS	50
STRUCTURE	95
PROPULSION	105
COMM	30
C&DH	30
POWER	100
THERMAL	10
HARNESS	25
TOTAL S/C BUS	445
INSTRUMENT	
SXT	
Mirror & Housing	250
Pre-Collimator	20
Grating	50
CCD Detector	50
Cryostat	120
Calorimeter Electronics	27
HXT	
Mirrors (3)	98
Detectors (3)	28
Mirror Thermal Covers (3)	5
Payload Structure	
EOB	100
Elephant Truss Structure	40
Mirror Support Platform	30
Detector Support Platform	25
CCD Support Structure	7
Sun Shades	10
Thermal	15
Harness	45
Mechanisms	31
Total Instrument Module	**951**
Total Spacecraft	**1396**
LV Capability	**1536**
Margin on Estimates	**10%**
S/C Bus = 32% of Total	**Payload = 68% of Total**

Note: In general, mass estimates are made using engineering calculation or based on similar existing designs scaled as required

Table 3

Estimated On-Orbit Power Summary

ITEM	AVERAGE POWER* (WATTS)	PEAK POWER* (WATTS)
S/C BUS		
ACS	100	+50
COMM	12	+146
C&DH	25	
POWER	28	
THERMAL	25	
TOTAL S/C BUS	**190**	**+196**
INSTRUMENT		
SXT		
Mirror Heaters	315	
CCD	20	
Cryostat	110	+20
Calorimeter Electronics	83	
HXT		
Detectors (3)	25	
Mirror Heaters	20	
Total Instrument Module	**573**	**+20**
Total Spacecraft	**763**	**+216 (979)**
Total Power Available**	**1400**	**1400**
Margin on Estimates	**83%**	**43%**

Notes: * - Average power is the power used when not transmitting data to the earth, thermal recycling the ADR, or slewing between targets. These operational modes occur for less than one hour each day, require less than 200 watts additional power, and this peak power load is covered by the battery.

** - Defined at end of life (5 years).

In general, power estimates are made using engineering calculations or based on similar existing designs scaled as required.

Table 4

1.4 On-Orbit Calibration and Alignment

A complete calibration and alignment procedure will be performed after launch vehicle separation and deployment of the EOB. An alignment system will be used to move and co-align each detector relative to its mirror or grating. After this alignment the S/C star tracker will be used to orient the telescope bore sight to a series of known X-Ray targets. At each target the detectors will be calibrated and the SXT focus will be checked and adjusted. The star tracker bore sight will be calibrated relative to the telescope bore sight. Any bore sight error will be corrected with biases in the star tracker software.

The on-orbit alignment procedure can be repeated as frequently as necessary; it could even be an autonomous procedure performed while slewing between targets without interfering with the science data collection. The calibration procedure with known targets can also be repeated frequently.

1.5 Launch Vehicle

Launch vehicle (LV) trades were performed early in the study. One of the parameters which drove the packaging of the S/C was the 1.3M diameter SXT mirror. This represented the largest diameter mirror that will meet the science requirements based on projected technology development. As a result, six SXT mirrors are needed to achieve the total required aperture area for the mission. Selection of the L2 orbit as well as fairing diameter ruled out smaller LVs such as the Pegasus. Several mirrors could be packaged on a medium to large LV such as the Atlas II or Delta III reducing the number of LVs required from six to as little as two, but this option offers no redundancy in the event of LV failure. If one vehicle fails, the mission fails. With these considerations, the Delta II 7925H-10 was the launch vehicle baselined for this study. This vehicle will be phased out prior to the HTXS launch. However, a Delta IV or similar class launch vehicle most likely utilizing the same fairing will then be available at a significantly lower cost. The baseline LV is a three stage rocket which combines the existing first two stages and solid rocket motor strap-ons with an upgraded STAR 48V third stage. This third stage is being considered by a number of other payloads, and should be available in time for HTXS use. It has a swivel nozzle that will use the S/C gyros to control the nozzle orientation during third stage burn. Use of a spinning third stage would require significantly more stringent balancing of the S/C, a spin table, and yo-yo de-spin mechanism, thereby reducing the throw-weight.

The baseline has a 10 ft diameter composite two piece fairing and uses a 3712 payload attach fitting as the S/C interface. A smaller 9 ft. diameter fairing would give 65 kg more payload, but the larger fairing provides greater usable height and volume which more efficiently satisfies the requirements for packaging the instruments. This launch vehicle yields a throw-weight of approximately 1550 kg. Figure 3 illustrates the S/C stowed configuration.

Stowed S/C Configuration

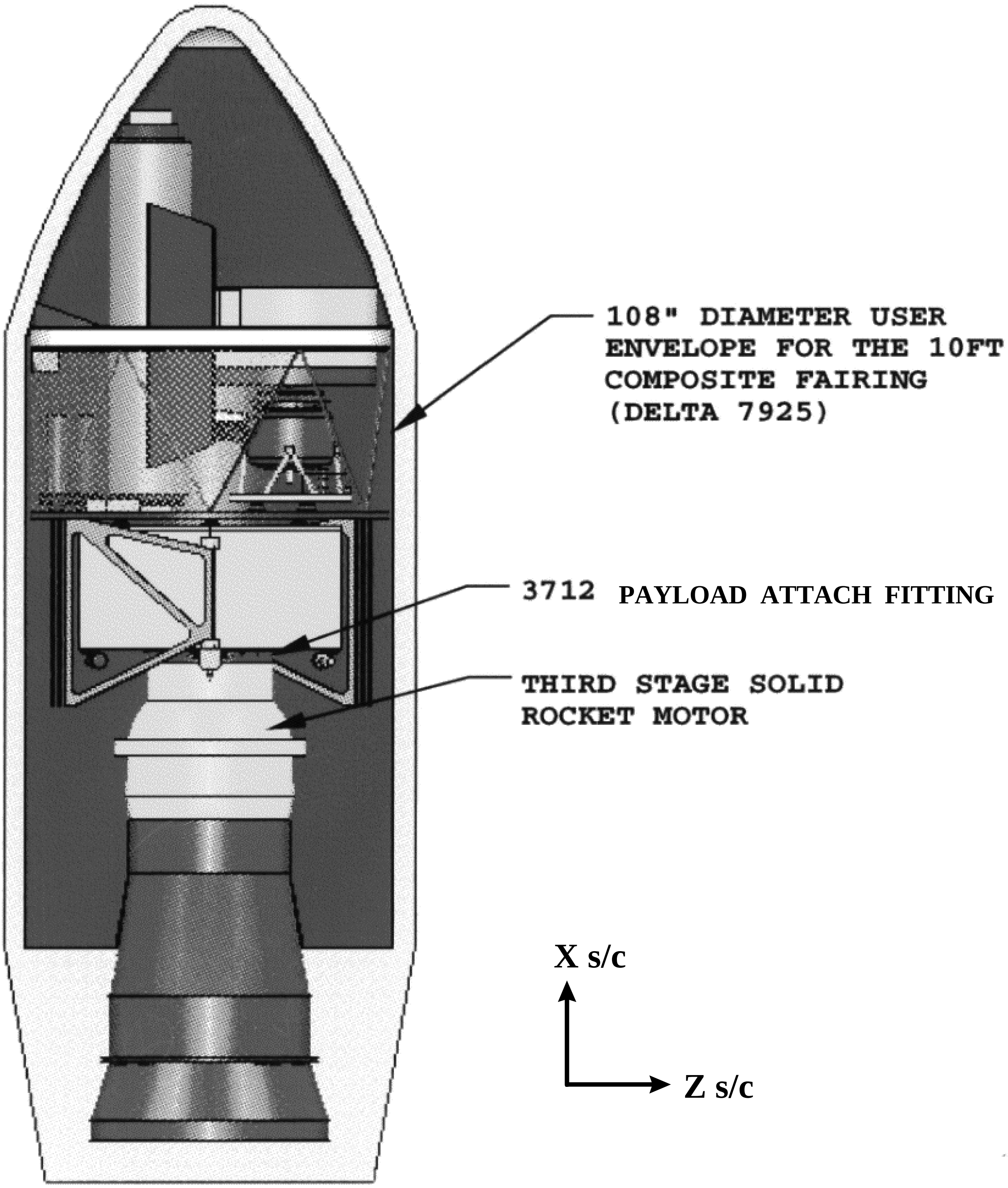

Figure 3

1.6 Ground System and Mission Operations

The RF ground station is comprised of a dedicated 11 meter antenna; feed; transmit and receive hardware; servo system; and the Operator Control Station. The ground station performs S-band tracking, telemetry, and command functions, as well as X-band telemetry functions. There may be simultaneous S-band and X-band transmission, restricted to single S/C operations. A computer workstation with software to provide multi-satellite pre-mission planning, automated pre-pass alignment, automatic system performance integrity analysis, signal routing assignments, remote control, and programming for post mission analysis and maintenance actions, is provided in the Operator Control station.

Figure 4 illustrates the baseline ground system architecture and its functional elements. The architecture consists of the RF ground station, Mission Operations Center (MOC), and the Science Operations Center (SOC). The RF ground station will be located at the MOC. Instrument science and housekeeping data will be downlinked using X-band. The S-band downlink will be used for spacecraft housekeeping telemetry, ranging, angle data and range-rate (Doppler). Commands will be transmitted via S-band.

Three options were studied to determine the most cost effective ground system. These options were use of the Deep Space Network (DSN), use of Wallops Orbiting Tracking Station (WOTS), and the procurement of a dedicated ground station. For a 5 year mission it proves to be more cost effective to procure a dedicated ground station. This decision made it possible to co-locate the ground station with mission operations and science operations, thereby reducing ground communications cost. Several approaches to integration and test (I&T) and preparation for flight operations were explored, with consideration given to minimizing operations personnel staffing while supporting I&T and the launching of 6 spacecraft.

To support normal operations there will be a one hour long contact with each S/C each day. Approximately 7 hours of contiguous support is required each day to provide this coverage. Commands will be transported from the MOC, and uplinked on S-band via the RF ground station. Real time telemetry will be transported to the MOC, via the ground station and ground network. Recorded science and spacecraft data dumped during a contact will be transferred to the MOC and SOC. It is assumed that all of the data processing functions will be automated, so actual delivery time will not affect the operations staffing. Ground communications and data transport will be done via commercial Internet protocols using a dedicated network. Ground communications traffic will include: transmission of spacecraft commands, telemetry, ground station remote control and status, tracking and acquisition data and products, and instrument related telemetry, commands and science data. With a ground network bandwidth of approximately 750 Kbps, and data transfer taking place while it is being received, science data will be available at the SOC within an hour of acquisition. The RF ground station will have data storage capability for network outages or SOC down time. The need for data pre-processing, i.e. Level 0 processing and protocol or format conversions, will depend on the data management and communications approach.

During launch, orbit insertion maneuvers, early on-orbit configuration and checkout, additional staffing will be required to support spacecraft initialization and checkout activities. Occasional 24 hour a day operations may be needed during the first two years. Based on studies done for the MAP mission, additional tracking, from DSN or WOTS, will also be required during the phasing loop period and after station keeping maneuvers.

Baseline Ground System Architecture

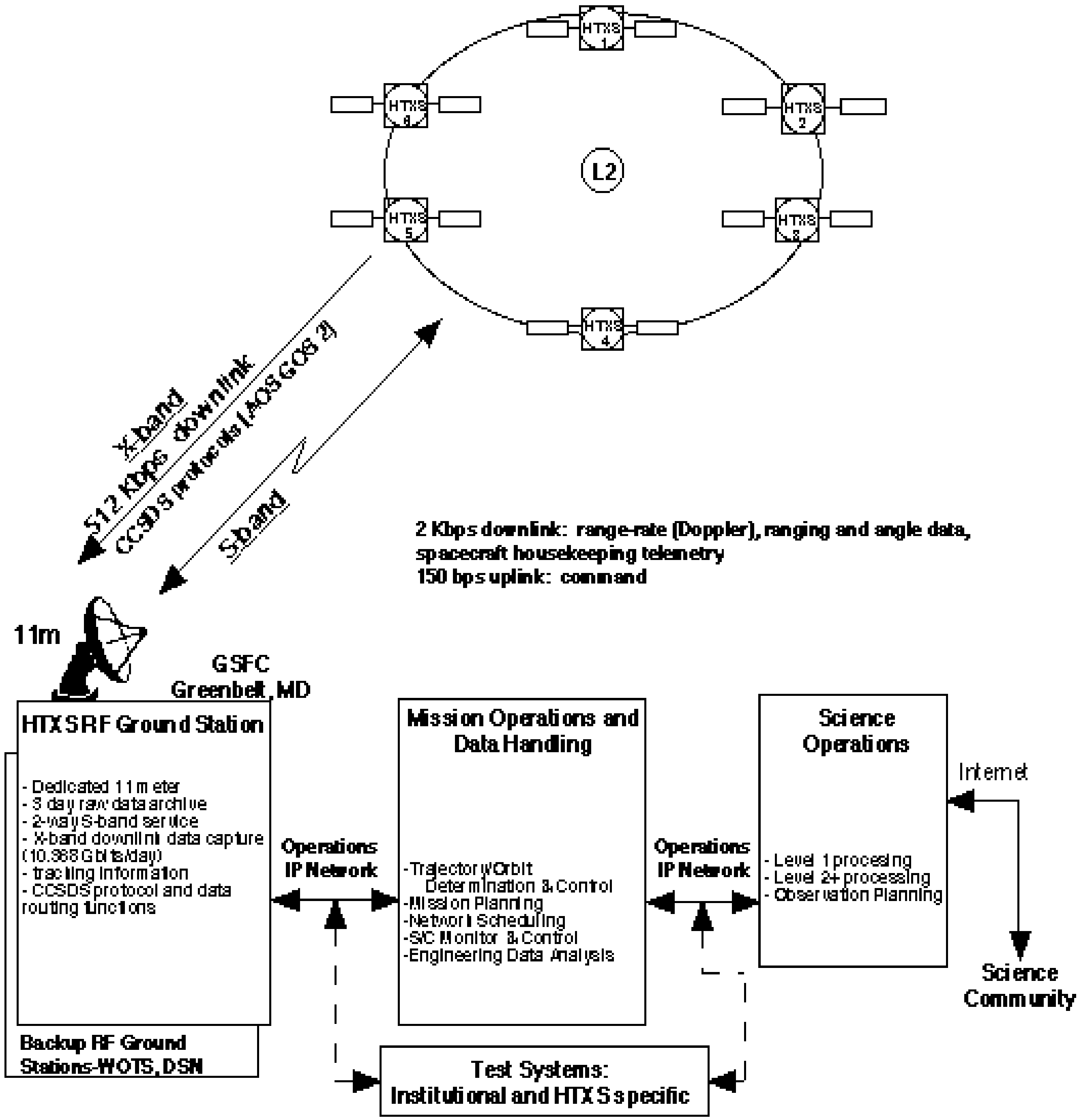

Figure 4

The estimated amount of contact time per day for each spacecraft will be adequate to support the tracking needs during the cruise out to L2 and during normal operations. GSFC flight dynamics experts will provide the mission analysis required for pre-launch spacecraft design and operations. This effort will include designing orbits around the L2 libration point, designing the transfer orbit or any other station acquisition maneuvers, performing station overage analysis, computing ΔV budgets, determining launch windows, examining launch vehicle dispersions, and providing orbit lifetime analysis.

For routine operations the console staff will consist of 3 people covering one 8 hour shift each day, 7 days per week. This staff will handle the daily operations activities, including: trajectory design, determination, and control, mission planning, clock maintenance, network scheduling, spacecraft control, spacecraft health and safety monitoring, and housekeeping data analysis. Onboard attitude control will be closed loop for normal operations, and definitive attitude determination will be done onboard and downlinked to the MOC. Science planning and command request inputs (including maneuver commands) will be provided about once per week. Spacecraft loads will be done once or twice per week for each spacecraft, assuming command memory is sized to support 7 days of normal operations.

2.0 GENERAL CONFIGURATION AND DESIGN PHILOSOPHY

The S/C in the deployed on-orbit configuration is shown in Figure 5. The required 8.5 m focal length is obtained by using a deployable telescope, which is necessary to take advantage of lower cost launch vehicles and their associated smaller fairings. This baseline deployable telescope utilizes existing technology. Four concentric tubes are extended by using a motor coupled to a system of cables and pulleys which lock the tubes into a series of kinematic joints. On-orbit alignment of the instruments is provided to allow for deployment and thermal variations, and material aging.

The S/C has been designed as two independent assemblies, an instrument module and a S/C bus. This separation permits the two assemblies to be both thermally and mechanically isolated from each other. Isolation of the instrument module from the S/C bus is necessary to provide stability for the highly sensitive instruments, and is obtained by using robust mechanical and thermal titanium kinematic mounts. The star tracker and gyros inertial reference system will be on the instrument module.

Completely separating the instrument module from the S/C bus allows them to be independently developed, each with its own vibration and thermal requirements. Each assembly can be separately tested and qualified, and the integration of the instrument module and S/C bus will then require minimal interaction/dependency between the two development/product teams.

S/C Deployed Configuration

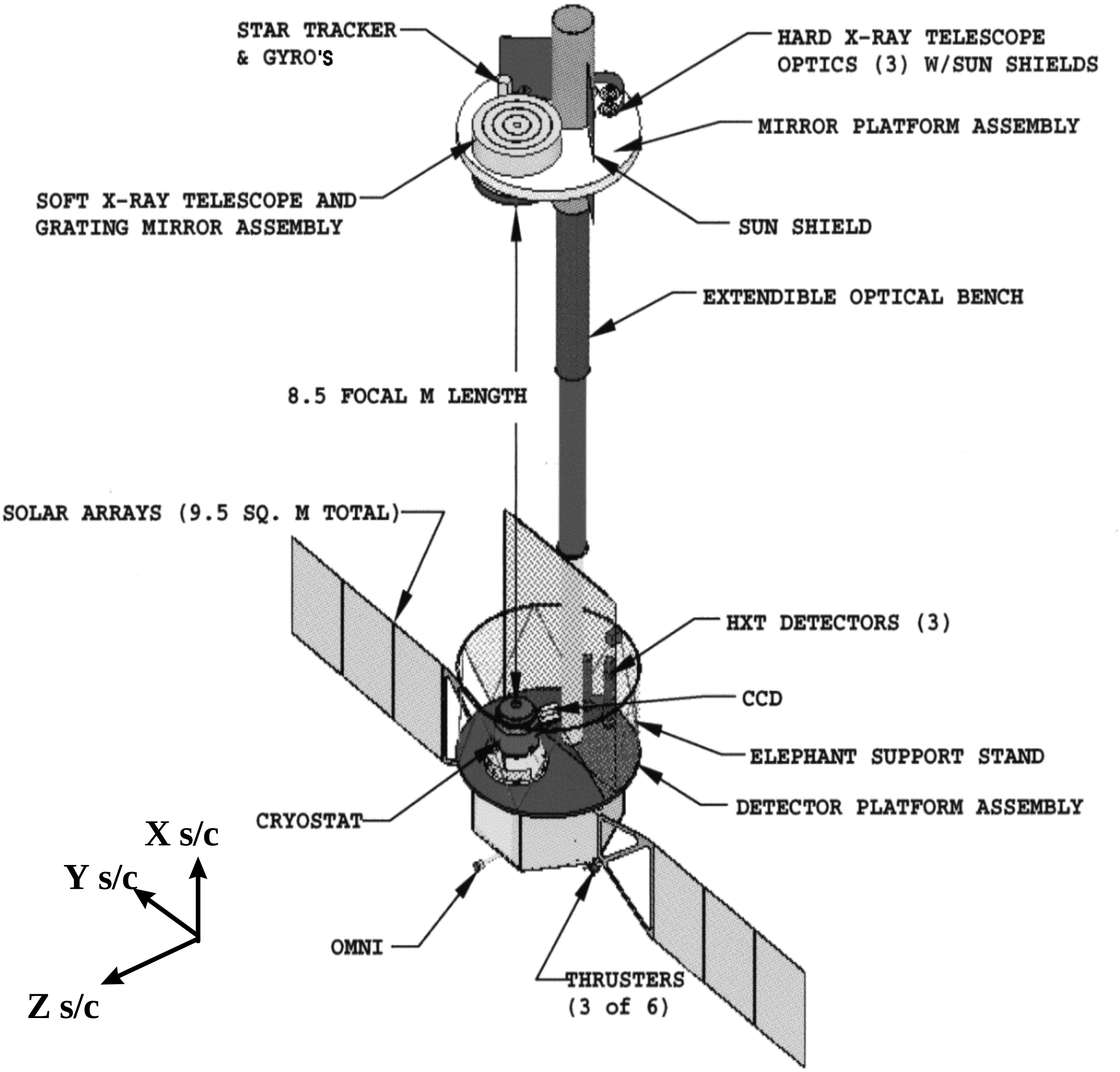

Figure 5

2.1 Instrument Module

The instrument module has the mirror assembly at one end of the EOB and the detector platform assembly at the other end. A key packaging trade which addressed the complexities associated with deploying the optics versus the detectors from the S/C bus was performed to determine the baseline configuration. Having the cryostat far away from the S/C bus enhances the thermal efficiency of the cryostat, but presents severe structural and thermal complexities. Large openings required for the mirrors made it very difficult to design an efficient S/C bus structure properly isolated from the IM. The constrained S/C bus mechanical configuration also resulted in poor load paths at the LV interface. Isolating the large SXT mirror from the non-isothermal S/C bus to maintain small thermal gradients across the mirror was another serious difficulty. In addition, it was necessary to place the mirrors very close to the LV pyro actuated separation bolts. These problems did not permit an efficient design that would maintain the co-alignment of the four mirrors without mounting them on individual two degree-of-freedom gimbals, with an associated optical alignment, measurement, and adjustment system for each mirror.

Although a deployed cryostat would weigh less, additional heater power would be required to maintain the cryostat mechanical compressor at an operable temperature. Some heater power would also be required for the various detectors, and their electronics, and for the translational alignment mechanisms for the detectors.

After considering all these problems the instrument module was baselined with the mirrors at the deployed end. This design is simple and far more efficient. The mirrors can be mounted directly on a rigid mirror assembly structure, and will maintain their co-alignment through launch, deployment, thermal variations, and aging. This eliminates the gimbals and their alignment system. There are no longer any exposed moving parts on the deployed mirror assembly (the gyros are hermetically sealed). The cryostat shell temperature is slightly higher when it is at the S/C bus end, which results in a cryostat that weighs slightly more. However, this configuration of the instrument module weighs less than the alternative, and it is baselined for this reason and its overall efficiency and simplicity.

2.2 S/C Bus

As a result of separating the instrument module from the S/C bus, the bus is of conventional design with no unique or costly requirements driven by any interfaces to the instrument. Therefore, this bus should not be a unique build and would be available commercially. Significant configuration considerations are:

- Two Omni antennas are placed on the lower deck to provide unobstructed spherical coverage. They are deployed after the solar arrays are unfolded.

- The high and medium gain antennas are mounted to the equipment panel on the sun side of the S/C.

- Six thrusters configured as two clusters of three each are located at the bottom deck. Two 5 Newton thrusters are aimed in the -X (anti-bore sight) direction, and four 1 Newton thrusters in the -Z (sun), and Z (anti-sun) directions.

- Solar arrays are designed with 9.5 m^2 of surface area populated by 5.5 mil Gallium Arsenide cells producing approximately 1400 watts end of life power.

3.0 INSTRUMENT MODULE DESCRIPTION

3.1 Mirror Assembly

The mirror assembly is comprised of a 15.2 cm thick low CTE composite honeycomb plate which supports the SXT mirror, three HXT mirrors, a star tracker, gyros, passive optical elements utilized for bore sight alignment, and thermal shields. The honeycomb plate also provides the interface for launch locks to the elephant support stand to transfer the loads generated during the launch environment down to the S/C bus rather than through the deployable tubes. The mirror assembly design concept provides a very rigid platform for the mirrors. They will be co-aligned on the ground during manufacturing and assembly, and the stability of the co-alignment will be demonstrated through vibration, thermal and deployment testing. Alignment on orbit will be maintained with all adjustments being done at the detector locations on the detector platform assembly.

The large SXT mirror carries a stringent temperature gradient requirement of less than 0.5°C across the 1.3 m diameter. A thermal design incorporating an electronic temperature controller and embedded heaters and sensors for the mirror and pre-collimator has been baselined. The HXT mirrors will be cold-biased and maintained within the desired 0 to 10 °C temperature range via an active heater control system.

3.2 Detector Platform Assembly

The detector platform assembly is comprised of a low CTE composite honeycomb plate which supports the cryostat, all detector alignment mechanisms, and the HXT detectors. It also provides structural interfaces to the deployable tubes, the elephant support stand, and the kinematic mounts to the S/C bus. The Grating/CCD detector is mounted to the cryostat main shell to maintain its relative position with respect to the grating and to take advantage of the greater range of translational adjustment capability for the cryostat which contains the SXT detector.

The CCD detector, which must operate at about -80°C, will be installed on a low-conductance bracket mechanically attached, but thermally isolated from the cryostat. The temperature of this unit will be maintained via a small temperature controller.

The HXT detectors, which are located on the sun side of the detector platform, are designed to operate at about -25^{o}C. These components do not dissipate large amounts of power, and a passive thermal design consisting of low α/ε radiators is baselined.

3.3 Extendible Optical Bench

3.3.1 General Overview

The EOB consists of four tubes, three of which extend with a pulley-cable system, with the fourth inner most tube fixed to the detector platform. The tubes ride on guide rails until they contact the latching devices, which have ball-to-cone interfaces. These interfaces ensure consistent, reliable deployment and provide a good load path from one tube to another. A harness cable pay-out reel will carry signal and power lines from the detector platform assembly through the interior of the tubes to the deployed mirror assembly platform.

3.3.2 Tube Structure

Tubes are fabricated from a low CTE composite honeycomb with an aluminum core. Face sheets will be made from K135 Ultra High Modulus carbon fiber with a cyanide-ester matrix. The fabric will be impregnated with Fiberite 954-2A cyanide-ester, two plies at 0-90^{o}, each ply being 0.2mm thick. Total tube honeycomb thickness is 12mm. Imbedded blocks will serve as hard point attachments for the pulleys, rails, and latch systems. The three outer tubes are each about 2.5m long; the innermost tube is 2.75m long. The four outer diameters are approximately, 300mm, 400mm, 500mm, and 600mm respectfully, and the tubes will have 250mm of overlap between them when fully deployed.

3.3.3 Guide Rails

The tubes will be guided during deployment by three rails per tube. There will be sufficient friction to keep positive tension on the pulley cables at all times.

3.3.4 Pulley System

The pulley system consists of three sets of pulleys on the ends of the tubes, three sets of cables which run serpentine fashion connecting the tubes, and a cable drum mechanism to wind up the cables. Deployment is actuated by means of a brushless d.c. motor with an integral reduction gear drive.

3.3.5 Stowed Latching System

The stowed tubes are kept firmly in place at the tube tops and bottoms by ball-cone interfaces that engage in the stowed configuration. The mirror assembly platform is latched to the elephant support stand during launch.

3.3.6 Deployed Latching System

If high pre-load is required, the latching system can consist of a high output paraffin actuator in concert with a stiff spring that draws the ball into the cone with a force of about 50 lbs. at each of the three tube interfaces. A simple passive system such as magnetic latches may be able to provide a sufficiently high pre-load.

3.3.7 Harness Pay-Out Reel

A simple reel with a retraction spring will be used to carry signal and power lines between the mirror assembly platform and the detector assembly platform through the tube interior.

3.4 Extendible Optical Bench Alignment System

The EOB Alignment System will be used on orbit after the EOB is deployed, and it may be used periodically thereafter as often as necessary to maintain the proper position of the detectors if they move due to thermal variation or aging. In the deployed configuration, the EOB alignment system will sense and adjust the alignment errors of the mirrors to their respective detectors. Detectors will be adjusted so that the optical axis of each telescope passes through the center of its respective detector. The alignment system uses an electro-optical device to sense angular and translational motion of the mirror platform with respect to the detector platform (with very high sensitivity and substantial range). A lens on the detector platform projects light through a pinhole off of an optically flat element on the mirror platform. A magnified image of the pinhole is formed at the detector platform on a CCD image sensor. The centroid of the image on the CCD sensor encodes relative platform movements. If the optically flat element is a mirror, the system senses angular motion; if it is a corner cube retroreflector, it senses translation. Translation and angle sensors are provided for each telescope as required. The sensed image displacements will be used to actuate the alignment mechanisms to change the detector location relative to its mirror optical axis, thereby registering the same X-Ray target on each detector. A laboratory demonstration of this concept with off-the-shelf components gave an angular range of +/-3 arc minutes and sensitivity of much less than one arc second. Translational range could be as large as +/- 5mm with 1 micron sensitivity. These values are more than adequate for the HTXS requirements.

3.5 Scattered Light Shield

Optical shielding will be used to prevent the detectors from receiving scattered light. The sensitivity of the detectors to scattered light must be quantified in order to select a suitable combination of shields, baffles, or a full light-tight “sock”.

3.6 Cryostat

The requirement for the cryostat is to provide a stable operating temperature of nominally 0.065K for the SXT detectors. The overall design of the cryostat (Figure 6) provides a 5 year lifetime limited only by the amount of expendable cryogen. To attain the 0.065K temperature, a three stage hybrid cooling system consisting of a mechanical cooler (Stage 1), an expendable liquid helium bath (Stage 2), and a sub-Kelvin Adiabatic Demagnetization Refrigerator (ADR) (Stage 3) is required. The ADR is a highly efficient magnetic low temperature producing device with no moving parts that acts as a heat pump between the 1.4K (2nd Stage) in the 0.065K (3rd Stage) temperatures. The baseline system has an estimated mass of 120 kg and an estimated average power of approximately 110W.

The ADR for HTXS will be an advanced version of the ADR employed in the Astro-E X-Ray Spectrometer Detection System (XDS) whose cooling cycle efficiency can be as large as 50% of the ideal Carnot cycle. The ADR requires a superconducting magnet which draws a significant current. The heat switch and the electrical leads to the magnet represent a parasitic heat load on the stored cryogen system. To minimize the parasitic head load to the stored cryogen, the XDS is developing superconducting current leads composed of high temperature superconducting materials and an improved heat switch. HTXS will require a further improved version of these superconducting leads, and also an improved He3/He4 diode heat switch. The sub-Kelvin cooling system will provide the detector array with a stable operating temperature of 65mK for approximately two days between thermal recycling. The recycling time will be less than 2 hours.

The baseline HTXS system concept replaces the XDS stored solid neon stage with a refrigerator that can cool down to about 8K. A miniature turbo-Brayton cycle cooler, potentially capable of meeting the HTXS requirements and essentially vibration free, is under development. A technology demonstration is planned for critical components of this cooler in 1997. If this technology demonstration is successful, the cooler could be ready for flight by the year 2000. The Origins program also has a requirement for a cooler that could be satisfied by the HTXS cryocooler. Other concepts will be considered as the technology matures and new systems are developed.

Cryostat Design

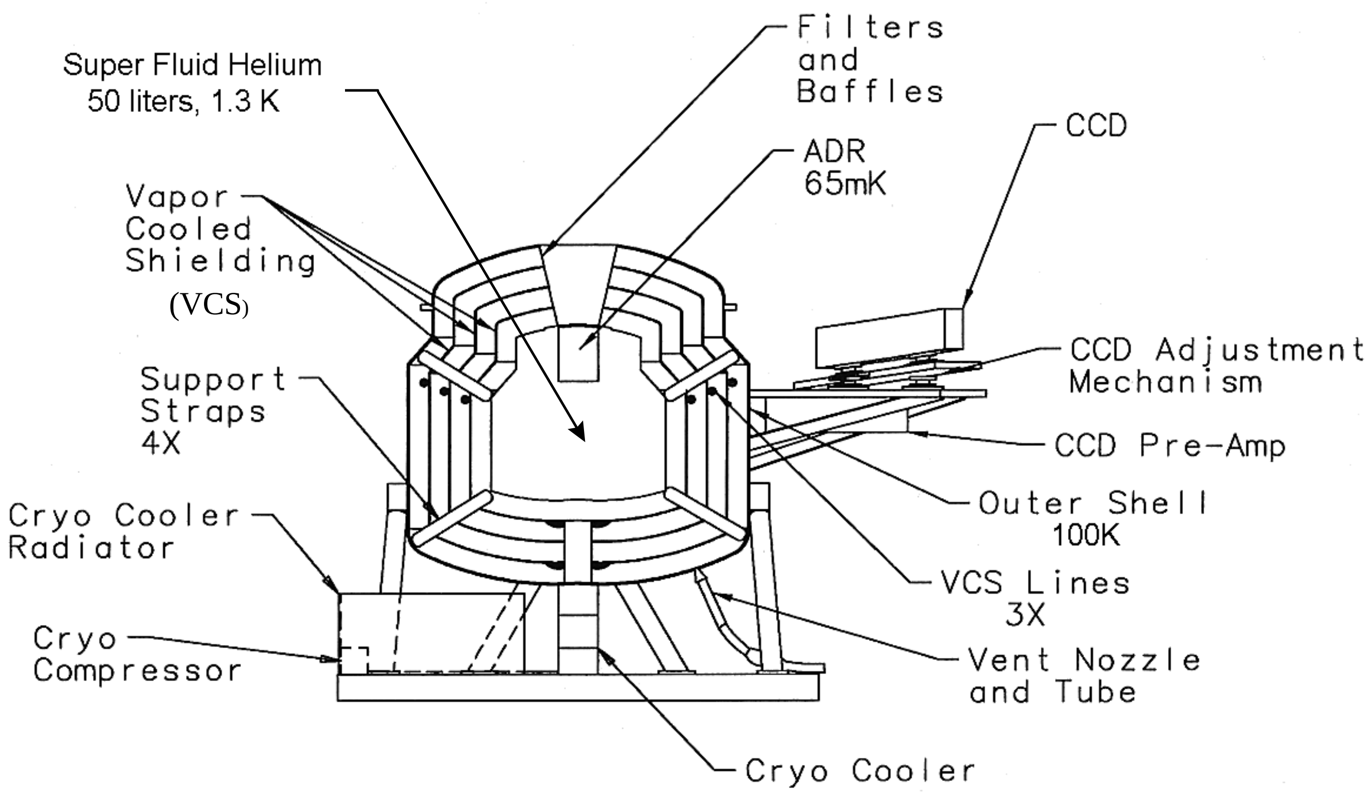

Figure 6

Preliminary calculations indicate that this hybrid system using the mechanical cooler to maintain the inner vapor shield to about 8K can reduce the required helium volume from 330 l to less than 50 l and the total mass from 240 kg to less than 120 kg. In order to meet mission lifetime requirements, the temperature of the cryostat shell should be about 100 K. Preliminary thermal analyses indicate that this requirement can be achieved by minimizing parasitic heat inputs. The cryostat will be conductively isolated from the detector platform by means of an ultra-low conductance composite structure similar to one now being developed for the WIRE spacecraft.

In close proximity to the cryostat mainshell is the cryocooler compressor which should be maintained at about 300 K. This device will dissipate approximately 50 Watts and therefore must be carefully isolated from the cryostat mainshell. Waste heat from the compressor will be conducted to a radiator panel that faces the anti-sun side of the spacecraft (See Figure 5).

4.0 SPACECRAFT BUS DESCRIPTION

4.1 Structure and Mechanisms

The baseline S/C bus structure is constructed of aluminum. There are no stringent thermal requirements. The primary load path will be through a central cylinder of sheet and stringer construction that distributes the launch loads on the instrument module to the Delta third stage. Honeycomb top and bottom decks, as well as equipment panels, provide access for the bus electronics mounted on the interior surface of the equipment panels.

Solar arrays and Omni antennas are permanently deployed after launch vehicle separation. When necessary, adjustments in translation and focus are employed for the cryostat (SXT detector) and Grating/CCD. The HXT will have only translation adjustability. Launch locks will be utilized for the cryostat adjustment mechanism.

4.2 Propulsion

The baseline propulsion system for the HTXS is a six thruster modular design having about 84 kg of hydrazine in two spherical 49.1 cm dia. tanks, with elastomeric diaphragms and propellant flow controls components. This system provides about 180 kN-s of total impulse to perform lunar phasing loop maneuvers, L2 orbit insertion and station keeping burns, 3-axes torquing, and accumulative disturbance torque unloading. The design provides no component or functional redundancy except as required by range safety.

4.3 Power

The 9.5 m^2 solar cell surface area is the maximum area that could be accommodated into the LV shroud. This area limitation requires the selection of more efficient Gallium Arsenide solar cells to obtain the necessary 1400 watts of EOL power.

A battery is required for power during launch, initial sun acquisition after LV separation, and long ΔV firings in the Lunar Assist phase when the solar arrays are not adequately sun-oriented. Battery power will also be required if any lunar eclipses occur during the transfer orbit or in the L2 orbit. There will be no earth eclipses.

In the normal on-orbit mode power requirements are essentially constant. There may be some variation in the thermal power used at different angles within the $\pm 20^{o}$ range of the solar array normal relative to the sun line, but this has been taken into account. Three peak power loads, which may all occur at the same time, are the daily data dump to the ground station requiring about 125 watts for one hour, the cryostat ADR requiring about 20 watts for up to two hours every few days, and slewing between targets when the ACS wheels require about 50 watts for periods of up to one hour. These peak power loads and the eclipse, launch, and initial acquisition loads can be handled by one battery of about 10 amp hours capacity.

4.4 Thermal Management

The L2 orbit provides a thermally benign environment. Unlike low-earth orbit missions, the L-2 orbit is not disturbed by frequent periods of earth shadow. Normal mission attitude also contributes to a highly-stable thermal environment since one side of the spacecraft always faces the sun within a 20° cone. As a consequence of the benign environment, the thermal design of the HTXS spacecraft bus will utilize inexpensive and reliable passive thermal control technologies. The exterior of the spacecraft will be covered with a combination of multi-layer insulation blankets and thermal radiator panels, each tailored to maintain spacecraft components within a safe operating temperature range.

4.5 Attitude Control System

The ACS on-orbit mode is a zero momentum, three axes stabilized system that uses one star tracker, three gyros, and three reaction wheels to keep the S/C rates low and the telescope bore sight close to the desired inertial orientation, and well within the field of view of all the detectors. Absolute inertial attitude knowledge will be about two arc seconds and control will be a small fraction of an arc minute, which should be quite satisfactory. The rate control requirement is to stay well within the 15 arc seconds spectrometer resolution for the several seconds integration time of the Grating/CCD detector. All of these requirements are easily achievable and have already been met on existing S/C.

A two axes sun sensor will be used for initial sun acquisition of the solar arrays, and provides a sun oriented safe hold mode.

A hydrazine propulsion system with six thrusters will provide thrusting capability for all ΔV maneuvers required for orbit acquisition and maintenance. It will also be used to unload the accumulated solar pressure disturbance torque. Orbit and torque disturbances from the cryogen effluent will be minimized by venting the effluent in equal and opposite directions.

After launch vehicle separation, or after the S/C is above 1000km (if separation occurs at a lower altitude), the solar arrays and antennas will be deployed and the S/C orientated to the sun. The EOB could also be deployed at this time, or at any future time (if there is any reason to delay deployment). The ACS will function properly either way.

For the several ΔV orbit acquisition maneuvers the S/C will be oriented with the anti-bore site axis in the thrust direction. This allows for a simple nozzle configuration and does not require any hydrazine components on the deployed section of the instrument module. During the Lunar Assist Phase some ΔV firings may be up to two hours long. Once past the Lunar Assist Phase further hydrazine usage for stationkeeping and momentum management will not require any S/C maneuvering that would place the solar array normal more than 20^{o} from the earth-sun line.

In normal operation the accumulated solar pressure disturbance torque may be unloaded in very small increments, thereby preventing a loss of target lock and disruption of data taking. Maximum slew rates during target transition will be about three degrees per minute, with a settling time of about fifteen minutes. For very large angle slews, small amounts of hydrazine could be used to substantially increase the slew rate.

4.6 Command & Data Handling

The maximum data rate for a bright source is less than 500 K bit/sec. Since the viewing of several bright sources in a row will seldom occur, the C&DH subsystem solid state recorder (SSR) has been sized at 4 gigabits to allow about two days storage at an average rate of 20 Kbs. A maximum daily playback time of one hour will transmit about 2 gigabits. Typical playback times will be shorter. Optimal downlink data rates are achieved by inclusion of a selectable rate convolutional coding chip, and built-in flexibility to select binary data rates. In addition, a Reed-Solomon encoder is used to reduce bit errors. Loss-less data compression techniques are utilized to yield up to a 3 dB increase in link margin. The C&DH will perform data management of the SSR in conjunction with ground operations.

4.7 RF Communications

Various studies were performed to select the baseline data playback system to unload the stored data. Included in the study were data rate, transmitter power, transmit frequency, spacecraft antenna gain, and ground station dish size. Heritage, size, weight, cost, complexity, link budget margin, compatibility, and frequency/bandwidth allocations were considered. As a result, the RF Communication Subsystem will utilize S-band for the forward link, and both S- and X-band for return links. Dual S-band Omni antennas provide full spherical coverage and are used for command reception and near Earth transmission. An RF switch allows the transmit path to change from the Omni antennas to the S-band medium gain antenna for improved link margins near L2. This allows full communication with the ground station from anywhere within the 600,000 km wide L2 orbit, and at any time, whether the S/C is locked on to an X Ray target, slewing between targets, or locked onto the sun. The medium gain antenna is a 4-element passive array being developed for the MAP mission. The S-band transponder performs turn-around ranging, receives commands, and transmits housekeeping telemetry. Instrument data will be transmitted via a high gain X-Band system. This X-band system design uses an exciter, controller, and includes a new phased array antenna currently under development. It will provide a minimum downlink data rate of 512 Kbps, QPSK (Quaternary Phase Shift Keying) modulated, to a dedicated ground station. The subsystem is compatible with DSN subnets for early mission phases and contingency modes. See Table 5 for a summary on link budgets.

Link Budget Summary

	S-Band Forward	S-Band Return	X-Band Return
Frequency (MHz)	2100	2290	8250
Data Rate (bps)	150	2K	512K
Transmit EIRP (dBm)	94	41.5	64.5
Path Loss (dB)	-222.8	-223.6	-237.6
Receive Antenna G/T (dB/K)	0	23	35
Req. SNR for BER 10E-5 (dB)	9.6	9.6	9.6
Coding Gain (dB)	0	7.1	7.1
Data Compression Gain (dB)	0	3	3
Margin (dB)	4.1	6	3.5

Table 5

5.0 DEVELOPMENT PHASE

5.1 Integration and Test

System level integration and testing of the instrument module and spacecraft will be performed at the GSFC. This activity requires a dedicated facility which includes a partial ground station and spacecraft simulator for command and data acquisition functions. Each major subsystem will be delivered to the I&T facility as a completely integrated and tested unit with all required documentation for handling, installation, and operation of the subsystem. Alignment of the mirrors and their respective grating and detectors will be done in the I&T phase of the program. The HTXS mirror/detector bore sight is established as the basic alignment reference. Co-alignment of the other instrument bore sight will follow. The star tracker bore sight will be aligned to the HTXS reference. Overall integration and test of the spacecraft and instrument module will complete this phase of the program.

5.2 Ground Calibration

Calibration of the six sets of mirrors with known sources is part of the I&T phase of the program and raises serious issues in both the technical and resource areas of the program. To accomplish the science objectives, the HTXS telescopes require large diameter mirrors and associated long focal lengths. Calibration of the mirrors by illuminating the full aperture is highly desirable but few facilities including the one at GSFC have the capacity to accommodate full aperture testing. Full aperture testing means traveling to a remote site incurring higher cost and scheduling difficulties. A less costly approach would be to partially illuminate the mirror in a smaller test facility like the one at the GSFC but this approach results in a poorer quality calibration. In view of these difficulties, a compromise plan would be to calibrate one set with full aperture illumination and employ partial testing of the other mirrors in a smaller facility and then correlate the data with that of the fully illuminated set. The final calibration plan has a significant effect on the cost, schedule, and the level of confidence assigned to the performance of the mirrors.

5.3 Cost Savings and Risk Reduction

Six spacecraft will be launched sequentially on 4 month centers. After the first spacecraft cost savings result from not having to repeat non-recurring costs such as observatory design, development and fabrication, software generation, fixturing, documentation, personnel training, and having an in-place I&T facility. Another major saving is achieved by building and operating an HTXS dedicated ground station. The loss of a spacecraft results in reduced performance rather than loss of the entire mission. This reduces the risk of using a single string philosophy for each spacecraft. Another major cost saving is achieved by close collaboration of the science and engineering teams.

6.0 FUTURE WORK

The design has only a 10% weight margin, which is insufficient at this stage of the program. Follow-on trades will consider S/C propulsion capability versus LV throw weight, EOB options, and instrument packaging. Trade studies will be performed to achieve a 20% weight margin. Finally, the study highlighted fabrication of the mirrors, detectors, EOB, and cryogenic subsystems as areas needing early and intensive development.

7.0 CONCLUSION

The concept of using six identical, low cost satellites to accomplish the HTXS Mission has been prepared. The concept meets all currently known scientific requirements and can be developed within the existing schedule and cost guidelines. This robust design focused on the L2 orbit, launch vehicle, instrument module and S/C bus in sufficient detail to show the concept to be feasible. Emphasized in the study were the packaging configuration, and the compatibility of the electrical, mechanical, and thermal interfaces.