FINAL REPORT

LARGE SPACE STRUCTURE EXPERIMENTS FOR AAP

VOLUME IV

FOCUSING X-RAY TELESCOPE
A LARGE SPACE STRUCTURE FOR X-RAY ASTRONOMY

REPORT NO. GDC-DCL67-009

20 November 1967

Contract NAS 8-18118

Prepared for

ADVANCED SYSTEMS OFFICE
MARSHALL SPACE FLIGHT CENTER
Huntsville, Alabama

Prepared by

CONVAIR DIVISION OF GENERAL DYNAMICS
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FOREWORD

The purpose of this report is to present the results of a study of "Large Space Structure Experiments for AAP" conducted by the Convair division of General Dynamics for the Marshall Space Flight Center, NASA. The study was performed during the interval 15 September 1966 to 15 September 1967, at a level of approximately $275,000.00, under Contract NAS 8-18118. The final report is published in five volumes as follows:

Volume I Technical Summary

This volume summarizes the results of the entire study.

Volume II Analysis and Evaluation of Space Structure Concepts

This volume presents the results of the analysis of the 40 space structure concepts analyzed during the first half of the study.

Volume III Crossed H Interferometer for Long Wave Radio Astronomy

This volume contains the design details of the crossed H interferometer that was one of the three concepts selected at mid-term for detailed analysis.

Volume IV Focusing X-Ray Telescope

This volume contains the design details of the focusing x-ray telescope that was one of the three concepts selected at mid-term for detailed analysis.

Volume V 100-Foot Parabolic Antenna

This volume contains the design details of the parabolic antenna that was one of the three concepts selected at mid-term for detailed analysis.

ACKNOWLEDGEMENT

The completeness of the program and program direction are due to the efforts of NASA Technical Program Manager, William T. Carey of MSFC and Norman Belasco of MSC. We wish to thank the members of the scientific community, particularly Drs. Riccardo Giacconi and Herbert Gursky; and those organizations which contributed to the scientific guidelines and engineering efforts. We also wish to thank the excellent staff of engineers that participated in this study.

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J. Rock	-	Electronics and Data Handling
M. Smith	-	Structural/Mechanical Design
H. Sturtevant	-	Reliability

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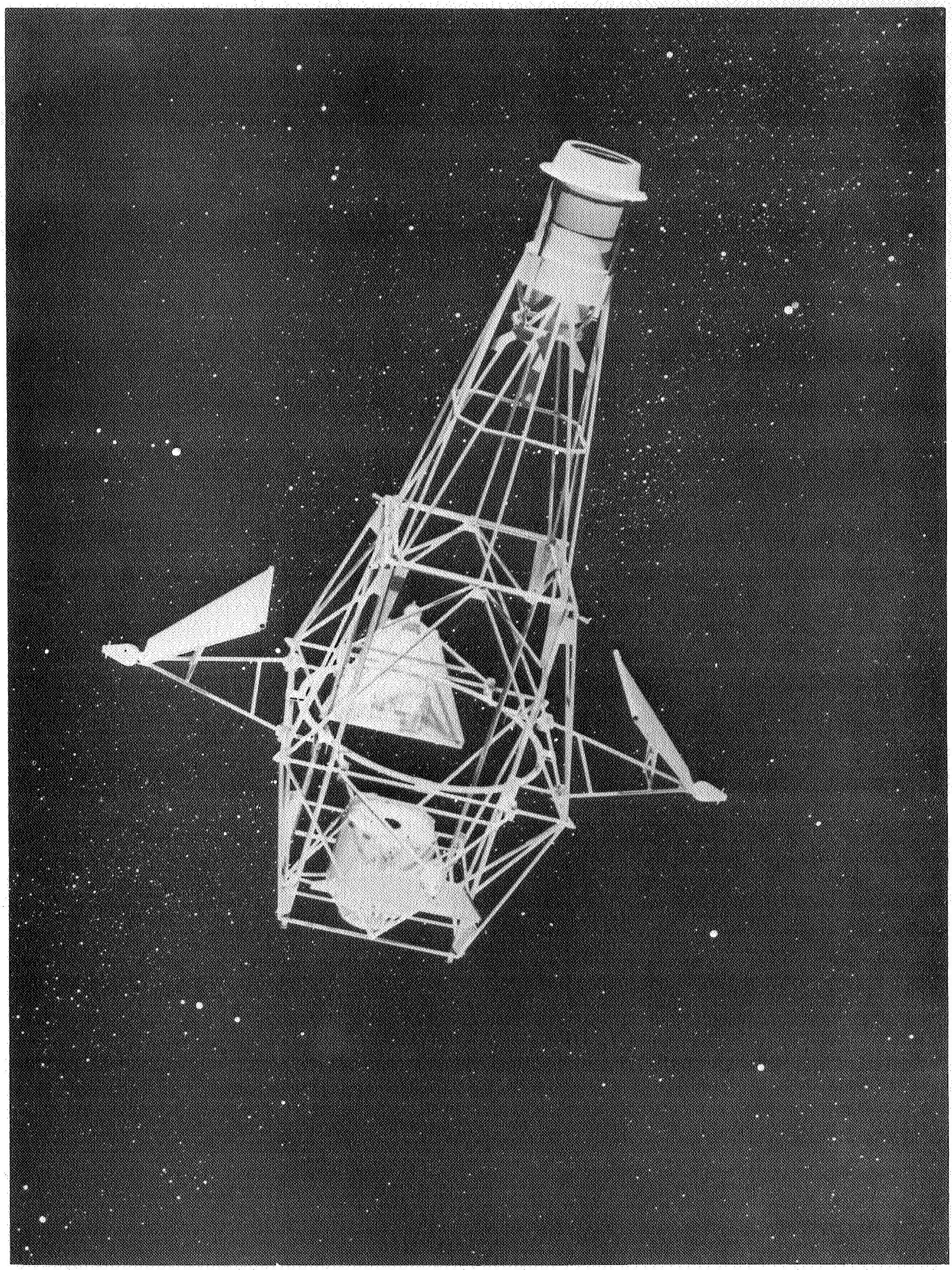


SECTION 1

INTRODUCTION

1.1 BACKGROUND. The purpose of this study has been to identify and define three large space structures experiments through which the following flight objectives could be accomplished: evaluate the role of man in the deployment, assembly, alignment, maintenance and repair of large structures in space; evaluate the performance and behavior of large structures in space from a technology viewpoint; and third, provide a useful space structure which can be used to fulfill a useful "user oriented" requirement such as a radio astronomy antenna or solar cell array.

The logical point of departure for a study such as this is to first determine the most promising areas of science and technology which will probably require large structures in space. In viewing the potential NASA missions throughout the next decade, one can conclude that some of the more prominent requirements will evolve from the areas of Astronomy, Communications, and to a lesser, but significant degree, from the requirements for solar cell arrays, micrometeoroid collectors and magnetometers.

With regard to Astronomy, regions of the electromagnetic spectrum which are of interest to astronomors begin with the very long radio waves and continue through the gamma ray region. Although not all astronomers necessarily agree on which areas of the spectrum should receive the highest priority, general concensus on those bands of particular interest and specific recommendations for future astronomy in space are found in "Space Research Directions for the Future, Part 2" (Woods-Hole Report). This document, together with other appropriate sources of literature, was used to guide this study in its relation to Astronomy. A few of the more important conclusions and recommendations contained in the Woods-Hole Report are summarized in Figure 1-1. The darkened area of the line at the top of the figure signifies those regions of the spectrum in which the atmospheric attenuation is greater than 10dB, and therefore, those regions which are essentially blacked-out from the Earth's surface. In these regions, astronomical observations are completely dependent on the ability to go into space. Accordingly, the Woods-Hole Report has made positive recommendations for space astronomy covering the entire spectrum with the exception of that region shown as radar astronomy on the figure which, according to that report, can be satisfied by ground-based observations.

Several of the bands of interest are particularly challenging to us who are interested in large space structures. First let's consider the very long wave (10 m and longer) region. Antennas designed to operate in this region have two dominant characteristics: Large physical dimensions and correspondingly

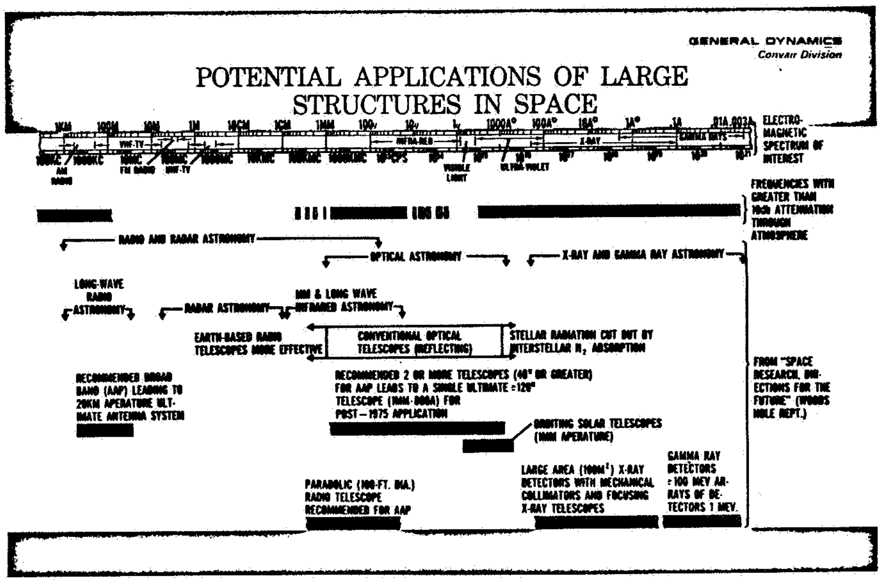


Figure 1-1. Potential Applications of Large Structures in Space

large allowable tolerances. The Woods-Hole Report recommended a "Broad Band Antenna System leading to a 20 km aperture ultimate antenna system". Although such an antenna would not be feasible during the AAP period, antenna types which may prove useful in the long wave region, and, therefore had to be accounted for in this study, include log periodics, rhombics, broadside arrays, and phased arrays. In the sub-millimeter region the primary useful antenna concept is a fairly large parabolic radio telescope with very stringent tolerance requirements. A great deal of emphasis was therefore placed in this region of interest. Additional regions of interest to astronomers which involve large structures are the X-ray and Gamma-ray ends of the spectrum. The Woods-Hole Report also contains specific large space structure requirements to support these Astronomical Programs. In summary, Convair was directed by NASA to include in this study detailed analysis on four types of space structures relating to Astronomy, longwave radio, sub-millimeter wave radio, X- and Gamma-Ray Astronomy. Although optical Astronomy, including infrared, ultraviolet, and visible regions of the spectrum, is extremely important to future space flight, Convair was directed by NASA not to include these types of structures in the study.

In summary, NASA directed that the concepts to be analyzed in the study be centered around those satisfying the following user-oriented applications requirements.

Longwave Radio Astronomy
Millimeter Wave Radio Astronomy
X- and Gamma-Ray Astronomy
Communications
Solar Cell Arrays
Magnetometers
Micrometeoroid Collectors

1.2 STUDY APPROACH. Figure 1-2 shows the major task areas to be accomplished during the study, as directed by NASA during the contract orientation. The study was essentially broken into two parts; the first half dealing with the analysis of a large number of candidate space structures concepts culminating in the selection of three which would then, during the second half, undergo more detailed analysis and design.

1.3 SUMMARY OF RESULTS. Tasks 1 and 2 resulted in the preliminary design and analysis of 40 candidate space structure concepts for flight in the 1970-75 time frame. Three of these structures were selected at the mid-term point of the study and were the subject of detailed preliminary design and analysis during the second half of the study. They are: a longwave radio astronomy antenna called a Crossed-H Interferometer, a Focusing X-Ray Telescope, and a 100 foot Aperture Parabolic Antenna. An over-all summary of the entire study can be found in Volume I. Analysis of the 40 candidate structures can be found in Volume II, and detail analysis of the three selected concepts is contained in Volumes III, IV, and V, respectively, as listed above.

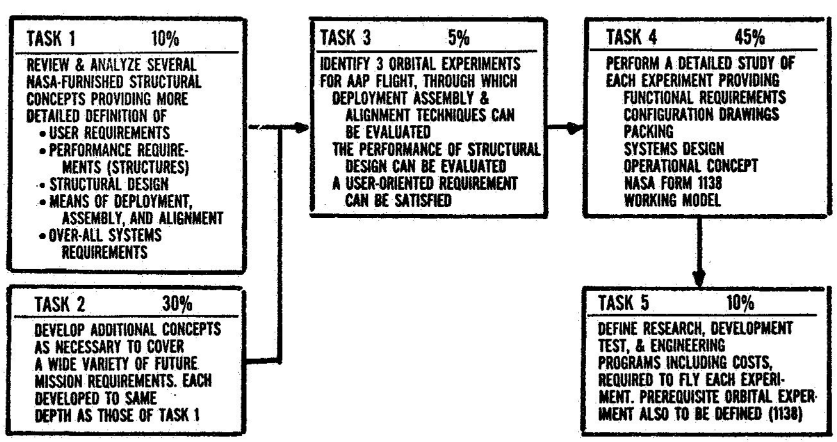


Figure 1-2. Large Space Structure Experiments for AAP

1.4 INTRODUCTION TO VOLUME IV. The approach utilized in the preliminary design and analysis of the X-Ray Telescope is depicted in Figure 1-3. The design includes a preliminary definition of the structure, subsystems, and mission instrumentation. The telescope mission was assumed to include detail source surveys of known x-ray areas of interest. Since the total observation time of each source varies considerably with differences in source intensity, type of measurements desired, orbit parameters, and source location, no attempt has been made to determine the actual number of sources surveyed each year.

The concept analysis included thermodynamics, stress, mass properties, and reliability, as well as functional and time line analysis.

1.4.1 Study Guidelines. The study of the X-Ray Telescope was guided by the following:

a. S-IB manned launch around 1975.

b. Circular 260 n. mi. or lower orbit at 28.5^{o} inclination or less.

c. Mission lifetime more than 3 years.

d. 20 to 40 inch maximum internal lens diameters.

e. Manned resupply/repair flights available as required (no more than four per year).

f. MSC 1968 to 1972 baseline astronaut data for orbital crew activities.

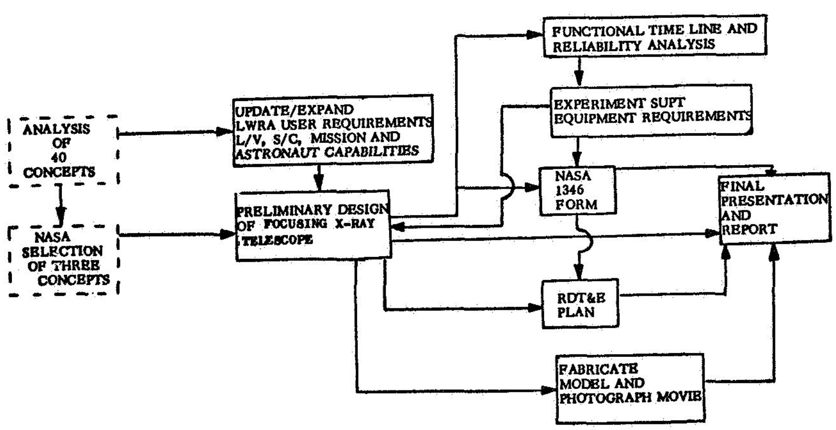


Figure 1-3. Technical Approach X-Ray Telescope

The study effort was directed to the investigation of the following areas with emphasis on their major engineering problems:

a. Structural design and analysis
b. Mechanical
c. Packaging
d. Materials
e. Subsystems
f. Alignment
g. Dynamic analysis
h. Thermal analysis
i. Determination of man's role in deployment, maintenance, and operation of the system.

1.4.2 Conclusions. The study results have yielded the following major conclusions:

a. To the extent of the analysis performed, it is feasible to design, develop, and deploy a grazing incidence X-ray telescope by 1974-75 which will resolve soft-x-ray sources to between 1 to 5 arc seconds accuracy.

b. An effective focusing x-ray telescope satellite can be packaged into, and orbited by, a manned S-IB with adequate reserve payload weight and volume for growth and additional (possibly unrelated) experiments.

c. Multiple (3 to 4) segment nested grazing incidence mirrors in the 20" to 40" maximum internal diameter range can be developed to the desired specifications for flights 5 years from go-ahead; 8 to 10 nested segments will extend the flight date 1-1/2 years.

d. The primary scheduled role of man in the x-ray system concept is in the control and backup of initial automatic deployment, EVA inspection and manual operation of the instrument turret, monitoring of ground controlled systems checkout, biannual resupply of expendables, replacement of short life components such as solar cells and attitude jets, and updating of vehicle capability by the addition of new or redesigned spacecraft or mission equipment.

e. In the area of unscheduled activities, man will increase mission reliability and operational lifetime by repair and replacement of failed or damaged components.

f. Suitable structures can be designed with adequate rigidity, deployment reliability, and thermal stability (or a suitable compensation system) for the 20 to 40 inch lens diameter range studied.

g. Lens diameters greater than 50 inches will require unconventional, higher expansion ratio structures for packaging in the manned Saturn launch vehicles. The maximum lens sizes compatible with unmanned payloads have not been determined (The above statements refer to lens focal length to diameter ratios of 8 to 1, or greater.)

SECTION 2

FLIGHT OBJECTIVES

The flight objectives of the focusing x-ray telescope are three fold as discussed below.

2.1 THE EVALUATION OF THE ROLE OF MAN. A primary flight objective of the x-ray telescope is to evaluate the role of man in the deployment, alignment, operation and maintenance of such a large space structure. To achieve this objective, consideration must be given to procedures and techniques and methods employed to obtain meaningful data as to man's effectivity in performing planned tasks. These procedures and techniques will vary from qualitative information obtained from the astronauts at the time of de-briefing to detailed biomedical data on each man. Standard biomedical data will be obtained on a real time basis to ensure the safety and well-being of the crew. This data is a part of the permanent flight record. Biomedical data will also be used to evaluate the effectiveness of man's participation based on heart rate, body temperature and metabolic expenditure. A record of all voice communications between astronauts and with the various ground stations, as well as a daily flight log can serve a similar useful purpose.

In addition to these essentially continuous "non-selective" records of astronaut performance, it will be necessary to plan in detail for specialized techniques for measuring man's effectivity. Photographic records of each critical EV activity for example, would be essential to establish the reason for any failure to accomplish an objective, and would also contribute to a modification of procedures and equipment for future flights. The value of such a photographic record, as well as some of the difficulties encountered, was clearly established in the Gemini program. Perhaps the most significant data which might be obtained from such an experiment, is an evaluation of the validity of the assumed time spans for task accomplishment. To ensure the success of each task that requires man's participation, procedures will be planned in detail and rehearsed with as much realism as can be achieved by means of neutral buoyancy or perhaps previous space flight experience such as orbital workshop simulation. This practice will verify that the procedure is valid and will familiarize the astronaut with the task. At the same time data will be obtained on the average time and metabolic expenditure to accomplish each task element. A comparison of this data with that obtained from the x-ray telescope experiment can provide invaluable information with respect to man's role, to guide planning for even more advanced missions.

Some of the typical measurements which will be taken during EVA periods are:

a. EKG
b. Oxygen consumption
c. Body temperature
d. Respiration rate
e. Suit pressure
f. Suit temperature
g. Suit oxygen concentration
h. Suit relative humidity

2.2 THE ADVANCEMENT OR STRUCTURES TECHNOLOGY. The ultimate flight objective in the structural area will be to perform the mission as designed with a structure assembly which is an extension of the state-of-the-art.

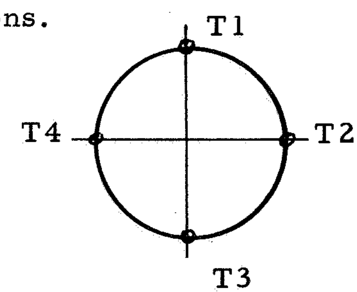
The problems of designing large x-ray telescope structures are similar to those confronting optical telescope designers -- the structure is required to remain perfectly aligned between imaging components separated by large distances. The base x-ray telescope has a 30 ft. focal length.

The problems of rigidity are further complicated by the focal length shortening required for packaging into the launch configuration. In many designs, the structural rigidity requirements conflict with thermal structure stability -- a shell structure is one example. Shell structures inherently exhibit maximum structural stiffness in bending, shear, and torsion; however, the continuous skin shells have maximum sun shadowing which generates high differential temperatures resulting in larger thermally induced distortions than a similar size open truss structure. The performance of the structure in orbit will be evaluated against the design predictions by instrumentation to obtain the orbit temperatures, stresses and distortion.

Some of the specific parameters to be measured are:

a. Optical axis (truss centerline) displacement and rotation.

b. Four quadrant temperatures on the primary truss tubes with at least six varying orientations.



c. Stress levels.

d. Imaging turret and lens gimbal bearing friction.

e. Mirror temperature, four-quadrant temperatures on all segments.

f. Image capsule internal temperature distributions.

The optical axis distortions will be compensated by the lens alignment system. The various temperature and stress levels will be measured by conventional thermocouples and strain gage instrumentation. The operating friction of the bearings of interest can be monitored by measuring the drive motor power with wattmeters.

The optical axis distortions and tube temperature will provide the data necessary to evaluate the performance of the thermal coatings against time in orbit --this area may well be critical aspect for long missions if the vehicle design relies on the maintenance of specific properties.

2.3 SCIENTIFIC OBJECTIVES. The third flight objective is to provide a large x-ray telescope capable of accomplishing a wide variety of scientific observations in the soft x-ray region. Such a scientific instrument is envisioned as being equivalent to a major astronomical facility providing one set of focusing optics capable of accommodating a wide variety of different scientific instruments at the focal plane. Such a multipurpose facility could be manned during the initial setup and checkout, and thereafter periodically manned for refurbishment, maintenance and repair during its operational lifetime. Although the telescope is capable of accomplishing almost any x-ray observation likely to be required in the middle 1970's, specific objectives which resulted in design goals are listed below:

a. Study the solar flares and other transient events, active centers, quiet corona and quiet chromosphere.

b. Search for weaker discrete sources (10^{-3} to 10^{-6} times the strength of those observed so far).

c. Precise determination (to one minute or better) of the location of the discrete sources in order to make possible identification of the x-ray sources with optical or radio objects.

d. Study of the structure of the discrete x-ray sources with a resolution of better than five seconds or establishing an upper limit of this order of magnitude for their size.

e. Study of the spectral distribution of the radiation from the various discrete sources; search for emission lines, absorption edges, and the long wavelength cutoff expected from interstellar absorption.

f. Search for polarization of the X radiation from discrete sources.

g. Directional and spectral study of the diffuse radiation, with the aim of establishing its galactic or extragalactic origin, and investigation of the properties of the media in which it arises and through which it passes.

h. Search for time variation of both long and short duration in the intensity of the discrete sources.

The observation of strong x-ray sources beyond the solar system has been one of the outstanding discoveries of space astronomy to date. In five years, the observations have progressed from the first evidence of a localized flux to the detection of over 20 discrete sources in the wavelength region between 1 to 10Å. In comparison it took radio astronomy a dozen years to progress from Jansky's original discovery to the detection of the radio source Cygnus A.

Much stronger galactic x-ray sources have been found than were anticipated prior to observation, and sources of at least two basic types appear to be present. The mechanism of x-ray generation in these sources is not yet known, nor is it known what messages concerning the universe they hold. It is expected that x-ray astronomy will make important contributions to an understanding of high energy cosmic precesses, the generation of fast electrons and possibly cosmic rays, the composition and density of gaseous matter between stars and galaxies, and possibly the nature of highly condensed matter states, such as the hypothetical neutron star. If neutron stars are found, interesting and unique checks of the gravitational aspects of relativity theory may be expected.

By measuring x-ray background outside the Earth's atmosphere, useful information for determining the density and distribution of gas within this galaxy may be obtained. The transparency of interstellar gas to x-rays increases with decreasing wavelength. There is, in any given direction, a threshold wavelength below which x-rays can be transmitted over a given path. The wavelength threshold indicates the density of gas along the path of transmission. Our galaxy is transparent up to about 6 to 10Å. From 10 to 15Å absorption by interstellar gas increases rapidly. Measurements of background x-ray intensities at about 50Å should provide a good measure of the distribution and density of gas in this galaxy.

Several "weak" x-ray sources have been discovered, and it is clear that many more are indicated by signals near the background level. An increase in sensitivity and resolution should suffice to reveal these sources clearly. The majority of x-ray sources are still unidentified with optical or radio objects.

Table 2-1 is a tabulation of most observed x-ray sources, with their respective locations (Reference 1 and up-dated).

Table 2-1. X-Ray Sources

	Intensity (counts/ cm^2/sec.)	Right* Ascension h	Min.	Declination* Deg.	Min.
Cyg XR-1	0.91	19	57	34	-
Cyg XR-2	0.8	21	45	39	-
Cyg XR-3 (Cyg A)	0.40	19	58	40	37
Cyg Loop (2 sources)	0.4	20	48	30	48
Cyg XR-4	0.6	21	18	44	-
Tau XR-1 (Crab nebula)	2.7	5	32	22	-
Sco XR-1	18.7	16	17	-15	31
Sco XR-2	1.4	17	8	-36	24
Sco XR-3	1.1	17	24	-44	18
Oph XR-1	1.3	17	32	-20	42
Sgr XR-1	1.6	17	54	-29	12
Sgr XR-2	1.5	18	10	-17	-
Ser XR-1	0.7	18	45	5	18
M87 (Virgo)	0.6	12	29	12	37
Cas A	0.5	23	22	58	34
Leo XR-1	0.5	9	33	8	36

*The uncertainty of position for any of these x-ray sources is approximately 1.5 degrees except for Tau XR-1 and Sco XR-1.

The x-ray flux in the 1 to 10Å region from the strongest of the x-ray sources is about one-tenth of that from the quiet Sun.

Most, if not all, of the observed x-ray sources seem to be within the galaxy. The use of satellite telescopes, which will provide longer observing times, will almost certainly bring a large number of weaker galactic sources within range of study and permit the observation of strong extragalactic sources. A diffuse x-ray flux, which may be the integrated effect of the contributions of all external galaxies, has been observed.

There have been explanations of these x-ray sources. Among the suggested source mechanisms are:

a. Bremsstrahlung from a hot plasma.

b. The inverse Compton effect in which energetic electrons scatter starlight photons.

c. Synchrotron radiation produced by energetic electrons spiraling in a magnetic field.

d. Thermal emission of a Planck spectrum from the surface of a hot compact, neutron star.

2.3.1 Stellar X-Ray Astronomy Goals. A description of the kinds of measurements required will justify the design of the large x-ray telescope described herein.

What has not yet been achieved is an understanding of which of the many possible physical processes gives rise to the x-rays observed, of the mechanisms of energy injection which can maintain the high kinetic temperatures required in the sources, of the nature of the astronomical objects being observed, and of their place in cosmology. The fact that Tau XR-1 is associated with the Crab Nebula is not in contradiction with this remark since at most two of the galactic sources are associated with supernova remnants and, therefore a different explanation must be found for the great majority of the sources (Reference 3).

There are, in general, two different approaches to a deeper understanding of the nature of the x-ray sources;

One approach is to identify their visual and/or radio counterpart, thereby permitting the full use of visual or radio observational techniques to study these objects. Also, if peculiar visible objects or class of objects with known properties could be shown to be associated with the x-ray sources, an immediate insight into the properties of x-ray sources in general might be gained. While this is certainly true in principle for most of the celestial x-ray objects, it appears that the energy radiated in x-ray is greater than the energy emitted in radio and visible. Particular objects (Sco X-1) emit 1000 times more energy in x-rays than in visible. Thus, it may not a priori be entirely reasonable to expect to deduce which mechanism produces the observed x-rays from consideration of the visible and radio components of the electromagnetic spectrum which are only a minute spill-over of energy from the powerful x-ray generating processes involved and whose characteristics may be weakly dependent on the nature of these processes.

A second approach is the study of the properties of the x-ray objects, through detailed examination of the emitted x-ray spectra. Detection of characteristic lines or absorption edges, Doppler Shift, continua, etc. will permit identification of the physical process which is taking place at the source, as well as provide a considerable amount of information on the properties of the astronomical body. Study of the polarization of the observed radiation and study of the structure of the extended sources (such as Tau XR-1) can yield direct information on the x-ray emitting mechanisms as well as further understanding of the energy inputs involved.

To study the x-ray sources through detailed inspection of their x-ray spectrum, improved measurements are necessary for the identification of the visual and radio counterparts as well. The great majority of the surveys up to 1965 yielded a precision of location determination of approximately 0.5°. It turns out that the visual objects which correspond to the x-ray sources observed are of 13th magnitude or fainter. There are approximately one hundred 13th magnitude objects per square degree in the sky, and this number increases as the objects become fainter. Thus, to unequivocally establish an identification, a precision of better than about 1/1000 of a square degree is necessary. This corresponds to a few min. of arc angular precision.

The Giacconi group in 1965 performed a precise location of the source Sco X-1, coming up with a combined area of uncertainty of only four square min., which is about 1/1000 square degree. This area of the sky was quickly related to a visible object appearing as a blue star of about 13th magnitude.

2.3.2 <u>Solar X-Ray Astronomy Goals.</u> Solar investigations differ from stellar investigations only in the angular size and intensity of the solar disk and radiation. Hence, detailed studies of the sun and its corona are possible to a degree not obtainable for any other star. Because the disk of the Sun can be resolved, single features such as sunspots, prominences, granules, and flares can be observed, and spectral scans across the disk from center to limb can be made with a resolving telescope, which in turn give the temperature and density variations with depth in the photosphere. The solar atmosphere can be used as a standard against which theories of stellar atmospheres and spectral line formation can be tested, because the Sun is a stable, main sequence star. Every new piece of information about the Sun contributes to our comprehension of the stars.

The extreme ultraviolet and x-ray spectra ($\lambda < 500Å$) are a particularly important part of solar radiation for observing and studying (Reference 1):

a. Solar activity such as flares and other transient events.

b. Active regions of the chromosphere and corona.

c. The quiet corona against the disk.

As in other spectral regions, future requirements in wavelengths <500Å are for high spatial resolution. A resolution of 5 arc sec. or better will permit study of the chromospheric network structure, general features of active regions, and flares. There is a need for higher resolution (1 arc sec. or better) to permit observations and study of individual spicules, granules, elements of the network structure, prominences, flare nuclei and microstructure of the corona. These shortlived features also require high time resolution.

The requirement for high spatial and time resolution goes beyond the capacity of either AOSO or ATM and indicates the need for more advanced systems associated either with AAP or with large observatories using man for servicing. The instruments proposed for post-AOSO observations in the XUV and x-ray region of the spectrum include the present telescope with grazing-incidence primary optics coupled with normal or grazing spectrometers or crystal spectrometers. In the spectral region $500 > \lambda \gtrsim 170$Å, either a normal incidence primary or a grazing-incidence primary with a grazing-incidence grating spectrometer have approximately the same over-all efficiency. The lower effective area of the grazing-incidence primary being balanced by the higher reflectivity. At wavelengths of less than 170Å, however, the grazing incidence primary must be used, since the normal incidence reflection efficiency deteriorates very rapidly.

<u>Flare Activity.</u> Spectral lines of high ionization potential are important, with lower ionization stages needed to complete the structure. X-ray lines and continua should be emphasized. Time-history of spectrum at wavelengths shorter than 20Å during active events is important.

High spatial resolution with medium time resolution (≤ 1 min.) for studies of the history of chromosphere and corona at time of the flare (with spectral resolution $\lambda/\Delta\lambda = 10^2$). Lower spatial resolution with high time resolution ($\Delta t \leq 10$ sec.) for detection and analysis of rapid events at flare time. Bursts of hard x-rays observations with high time resolution (< 1 sec.) at $\lambda < 0.5$Å might be combined with observations having high space resolution at $\lambda \sim 10$Å, $\Delta\lambda \sim 5$Å. Most important are observations in the 1-40Å region with both high spatial and temporal resolution.

<u>Active Centers.</u> Observations are needed upon which to base a model of the distribution of electron density and temperature in the active chromosphere and corona, and to study variations of the model with time. This includes identification and analyses of small, high-temperature regions ("hot spots"). Time resolution should be adequate to study the development of active centers.

The most important spectral lines for this study are the Fe coronal lines in the interval 170-360Å, which permit study of the variation in the degree of ionization from Fe IX to XVI. The chromospheric lines of 0 I-0 V, which show the variation in chromospheric excitation, and the coronal lines of 0 VI, 0 VII, and 0 VIII are also extremely important. Because of its high ionization potential, the C VI line at 33Å is also useful for identifying very hot regions. Spectral resolution of 10^3 and spatial resolution of 10 arc sec. are needed. Time resolution of minutes to hours is sufficient for the slow variations.

It is possible that the highest possible resolution will reveal slowly varying hot spots of scale 1 arc sec. Once that resolution is available, only stable pointing and longer integration times will be needed. Very high spectral resolution is desirable to resolve the profiles of a few lines, such as C VI at 33Å.

For studies of active regions, only relative intensities are necessary, since the chief concern is with the changes in relative ionization.

It is especially important to study active regions in the poorly known interval $1 < \lambda < 20$Å with $\Delta\lambda = 0.01$Å, angular resolution of 5 arc sec. or better and $\Delta t \sim 1$ - 150 sec. The energetic radiation in this region is particularly sensitive to very high temperatures.

Quiet Corona. Spectral and spectroheliographic studies of the same lines mentioned above under Active Centers will help to reveal the nature of the faint corona outside the active regions. It is of particular interest to connect the quiet corona structure with the chromospheric network and the spicule bushes, to determine the nature of the chromosphere-corona interface.

Important spectral lines for this study are the Fe series mentioned above; also Ne VII (465Å) and Ne VIII (780Å); Mg IX (368Å) and Mg X (610Å); and Si X (254Å, 272Å); Si XI (303Å); Si XII (499Å, 521Å). Comparisons of the intensities of the resonance lines with those of subordinate lines in the 100Å region provide a good measure of the electron temperature. (Departure of the resonance line doublets of the Ne VIII-Mg X-Si XII sequence from the 2:1 ratio is a sensitive measure of optical depth).

Spectral resolution of about 10^3 is needed, or better for strongly blended lines. Resolution of 5 arc sec. will determine the gross correlation of the network structure with that of the corona, but 1-2 arc sec. is necessary to reveal the details of the interface.

Quiet Chromosphere. Most of the important lines are at longer wavelengths, but attention should be called to the lines He II 304Å and 256Å. There is a need for spectral observations of the over-all and detailed distribution of these lines, to determine the height variation of temperature and density of the chromosphere as well as the general structure. Low-noise observations are necessary to search for a possible uniform hot chromosphere over the center of the network cells in addition to the hot elements at the edges. Reasonable time resolution may be obtained by restricting observations to selected parts of the disk.

Spectral lines of interest include the He II 304Å and 256Å lines, the He II continuum, the C and 0 chromospheric ions, and isoelectronic sequences such as C II-Ne III-0 IV. Spectral resolution of 10^3 or better is needed.

Spatial resolution of 5 arc sec. near the limb, with less near the center of the disk, will resolve the network; 1 arc sec. over an area of 30 square sec. will permit study of the detailed dynamic structure.

Time resolution should be less than the chromospheric oscillation period (300 sec. for 10 arc sec. resolution), but high spatial resolution (1 arc sec.) will also require higher time resolution (10 sec.) because smaller chromospheric structures vary more rapidly than large ones.

2.4 PAST X-RAY ASTRONOMY ACCOMPLISHMENTS. The earliest x-ray studies in space were carried out with detectors scanning the Sun. The early measurements gave upper limits on the x-ray fluxes in space; Friedman found an upper limit of 10^{-8} ergs/cm^2/sec/Å for the influx of x-rays from beyond the solar system (Reference 4).

The first x-rays detected from sources outside the solar system were found in an experiment of Giacconi, Gursky, Paolini, and Rossi. They flew a rocket from the White Sands Missile Range containing uncollimated thin-window Geiger counters with some 60 cm^2 of sensitive area. Discrimination against energetic particles was achieved with an anticoincidence scintillator. The windows of the Geiger counters had thicknesses corresponding to 1.7 and 7 mg per cm^2. Their transmission, together with that of the filling gas, gave a band of sensitivity for x-rays of wavelengths between 2 and 8Å. This rocket experiment detected a soft x-ray source from a direction near the galactic center. This very intense source was later determined to be in the constellation Scorpio. There was also an indication of a second source in the neighborhood of Cygnus.

Further x-ray measurements were reported in 1963 by Gursky, Giacconi, Paolini, and Rossi. In these flights the Geiger-counter windows were made of beryllium 0.002 in. thick. These counters were supplemented by sodium iodide and anthracene scintillation counters which were intended to measure the more energetic x-ray and any electrons which might be present. The flights confirmed the presence of the source in Scorpio, reinforced the evidence for a source in Cygnus, and suggested the presence of a third source in the general direction of the Crab Nebula.

Further evidence of celestial x-rays was given by Fisher and Meyerott. Their analysis suggested that a multitude of x-ray sources is present in the sky, but that these sources are not statistically well established.

Bowyer, Byram, Chubb, and Friedman flew an instrumented rocket in April 1963 which carried proportional counters containing 65 cm^2 of sensitive area having a field of view of 10^o at half-maximum sensitivity. These counters had beryllium windows 0.0005 inch thick, and were sensitive to x-rays from 1.5 to 8Å. The source in Scorpio was confirmed and its position located at RA 16^h15^m, decl -15^o, the uncertainty in the position was stated to be about 2^o, and the angular diameter was less than 5^o. This flight also located a source in the direction of the Crab Nebula with a strength only one-eighth as great as that of the Scorpio source.

An extremely important advance was made by Bowyer, Byram, Chubb and Friedman on July 7, 1964, when they launched a stabilized Aerobee rocket guided to point Geiger counters with 114 cm^2 area at the Moon during the critical 5 minute phase in which the Moon was occulting the central portion of the Crab Nebula. There were two counters having Mylar windows, coated with 60Å of Nichrome, one 0.001 in. thick and the other 0.00025 in. thick. The difference in counting rates between the counters was expected to indicate something about the spectral distribution of the x-ray in the low energy range. Both counters recorded essentially the same number of counts, which led Bowyer et al. to conclude that the x-rays from the Crab Nebula were concentrated below 5Å. However, Friedman reported at the Symposium on Relativistic Astrophysics held at Austin, Texas, in December 1964 that it had rained on the day of the flight until nearly flight time and the moisture had apparently degraded the performance of the lower energy counter sufficiently to distort the results. Consequently, the data obtained from the July 1964 flight are open to question. This flight gave the very important result that the angular width of the x-ray source in the Crab Nebula was about 1 arc min. This indicated a diameter of about 1 light-year.

In August 1964, Giacconi et al. flew another rocket carrying a number of different counters. This flight indicated a source in Sagittarius, in addition to the well determined Scorpio source.

In February 1965, Oda, Clark, Garmire, Wada, Giacconi, Gursky, and

Waters reported some important data on the angular sizes of the x-ray sources in Scorpio and Sagittarius. They used an ingenious x-ray collimation system consisting of two grids of parallel wires. These wires were separated by slightly less than one wire diameter. The two grids were mounted one behind the other 1.5 in. apart. As the collimator scans across a point source, the shadow of the front set of wires will fall alternately on the back wires and on the intervals between the back wires. Hence a point source gives a modulated signal, whereas an extended source gives a more nearly continuous signal.

Oda et al. concluded that the Scorpio source extended in a direction approximately parallel to the galactic plane definitely less than 30 arc min. and probably less than 8 arc min. They were also able to conclude that the Scorpio source did not extend more than 1° perpendicular to the galactic plane. They further concluded that the Sagittarius source was either extended in space or consisted of more than one point source. The x-ray source is spread over a region more than 30 arc min. in diameter.

The number of known x-ray sources was considerably extended as a result of two flights of Bowyer, Byram, Chubb, and Friedman, one June 16, 1964, and the other on November 25, 1964. Geiger counters were mounted facing outward through the skin of an unguided Aerobee rocket. Aluminum-honeycomb collimators were used limiting the field of view to 8.4° at half-maximum transmission. The rolling and precession of the rocket caused Geiger counters to scan a large portion of the sky. The counters were sensitive to x-rays in the range 1 to 15Å. The effective area for x-ray detection was 906 cm^2. These flights detected eight new x-ray sources in addition to Tau XR-1 and Sco XR-1.

Little information is yet available about the presence of higher energy x-rays from these sources. In July 1964, Clark flew a balloon carrying an x-ray detector in the form of a scintillation counter with a sodium iodide crystal of 97 cm^2 in area and 1 mm thick. This detector was collimated to provide a field of view of 16° in one direction and 55° in the other. Clark detected x-rays in three energy channels between 15 and 60 keV.

There have been several more rocket flights, mainly by the MIT group, the Fisher Lockheed group, the Friedman NRL group, the Bowyer Catholic University of America group, the GSFC group, and the A.S.&E. group, during the last two years. The NRL group has reported evidence for the existence of a measurable x-ray flux associated with the extragalactic objects Cyg A and M-87 in the constellation of Virgo. A survey by the Lockheed group of the Cygnus region in 1964 failed to show evidence of a source in the direction of Cygnus A. Preliminary results from a rocket experiment, flown last October by the Giacconi group at AS&E, set a severe upper limit on the x-ray flux observed from Cyg A. The GSFC group has reported tentative evidence for a source of large angular size (a few degrees) in the direction of the Coma Cluster. No verification of the existence of this source has yet been obtained.

The most notable of recent investigations was perhaps the identification of Sco X-1 performed by the Giacconi group in a collaborative effort by a group of scientists at AS&E, MIT, Mount Wilson and Palomar Observatories and Tokyo University (Reference 3). The rocket flight was carried out by the AS&E-MIT group on March 8, 1966. The experiment used a refined version of the modulation collimator as described above, originally proposed by Oda. It was concluded that the angular size of Sco X-1 could not exceed 20 arc sec. The combined area of uncertainty was only four square minutes, or about 1/1000 square degree. An optical counterpart was very quickly identified by the Tokyo Observatory and confirmed by the Mount Wilson and Palomar Observatories, as a blue star of 13th magnitude or so. During the same flight, the x-ray source in the Crab Nebula (Tau XR-1) was observed to have an angular size of about 100 arc sec. The center of that source appears to occur within 20 arc sec. of the visible and radio center of the Nebula (Reference 3).

A diffuse background of soft x-rays also of celestial origin has been observed with intensity of about 10 photons/cm^2 -sec-ster. This background of unexplained origin has been tentatively attributed to the collective effect of a number of weaker sources not yet resolved.

2.5 CURRENT AND PLANNED PROGRAMS IN X-RAY ASTRONOMY.

Having delved into past accomplishments, it is appropriate to look at some of the planned activities, to gain a better insight into the justification for the large imaging x-ray telescope.

Among the current and planned programs falls a newly announced NASA Small Astronomy Satellites (SAS) program. The first spacecraft is scheduled to be launched in 1969. The objective of the program is to map x-ray stellar sources within and outside our galaxy. This is under the management of Goddard Space Flight Center, to be launched with a Scout booster.

Although man may be phased into the program as early as 1969, NASA also sees a continuing need for smaller automated observatories and explorers throughout the next decade and beyond (Reference 25). The continuation and expansion of the Explorer satellite program will permit the active participation of university scientists. The x-ray astronomy satellite is a comparatively simple satellite which will carry a complement of x-ray detectors to obtain an all-sky map of x-ray sources including those an order of magnitude or more fainter than those observable from a short-lived rocket and rough spectral curves for the brighter sources. Plans are also being formulated for small spinning and pointed satellites for surveys in the ultraviolet, x-ray, and gamma ray wavelengths.

In the NASA OSSA Prospectus (Reference 25) it is proposed that the present OSC and Solar Explorer programs be continued with minor improvements made to the spacecraft. During the early part of the 1971-1974 period, NASA envisions a need to study the hard components of electromagnetic radiation from about 1 KeV to the highest energy photons observable (Reference 1). During the early part of that time period, the Sun will still be relatively active. This activity is manifested in part by the emission of hard X-radiation.

Between the automated observatories of the 1960's and the very large observatories of the 1980's, man will be phased into the astronomy program to an increasing degree. He makes possible the use of film as a rapid, compact, high-resolution detector and data storage device. He can play an increasing role in repairing or replacing failed parts to enhance reliability and in exchanging auxiliary equipment to increase instrument versatility. He will be phased into the program in a major way with the launch of the solar Apollo Telescope Mount (ATM). The approved ATM has instrumentation for high spatial and spectral resolution studies in the ultraviolet and x-ray regions. It will study solar x-rays, particularly those emitted during flare conditions, which appear to have their sources in highly localized regions.

Three ATM's may be employed for stellar and galactic observations. The first ATM is proposed to have two x-ray grazing incidence telescopes. Dr. Riccardo Giacconi of American Science and Engineering, Inc., Cambridge, Mass., is principal investigator for an x-ray spectrographic telescope. This experiment proposes to obtain x-ray photographs of flares with a spatial resolution of about 2.5 arc sec. and to simultaneously record spectrally disbursed emissions over the range of two to eight Å with a resolution of a fraction of an Å. An electronic flare detector, in addition to the main telescope, will give the astronaut a visual indication of flare buildup. This will allow him to select active regions and photograph the flares in the early x-ray rise periods.

The NASA Goddard Space Flight Center's x-ray telescope uses a different technique to get information on impending flare buildup. Principal investigator for the Goddard experiment is James Milligan. This experiment will use a glancing incidence x-ray and extreme ultraviolet telescope with a resolution capable of recording the solar x-ray distribution in the 3 to 100Å wavelength region. The information will be recorded on ultraviolet sensitive 35 mm roll film. There are also two proportional counters to monitor the total solar x-ray flux in the spectral regions of two to eight and eight to 20Å. This data will be pulse height sorted and recorded on tape.

After the ATM and the automated observatories, man-assisted and maintained observatories with both longer lifetimes and more accurate pointing capabilities should be phased into both the solar and stellar programs. The large focusing telescope design presented in this report is intended for these purposes. Either as part of a national astronomical observatory or by itself it will consist of instrumentation to conduct a wide variety of astronomical observations. This major instrument has been strongly recommended by the 1965 NAS Woods Hole Summer Study, the President's Science Advisory committee and others. Comparing x-ray astronomy with radio astronomy, it is likely that at the time of this telescope or a complete National Space Observatory, a number of x-ray sources will be known with moderately good information on their spatial structure and spectra, but that to understand the physical characteristics of these sources and in particular the source of their enormous energies, high resolution information will be imperative.

SECTION 3

SCIENTIFIC REQUIREMENTS AND PERFORMANCE

To arrive at large space structures that fulfill the third main objective of the entire study, namely that of providing a scientifically useful device in earth orbit, an extensive investigation was undertaken in the first half of the study to come up with scientific user requirements. These scientific user requirements were then utilized both as evaluation criteria for NASA supplied concepts and as design goals for developing those new concepts that were eventually selected by NASA for further detail design study during the latter half of the study. Several prominent members of the scientific community were instrumental in devising these design goals. Convair would like to acknowledge the unselfish and invaluable assistance, at no consulting cost to NASA or Convair, of the following members of the scientific community who were concerned with x- and gamma-ray astronomy:

Dr. C. Stuart Bowyer, University of California
Dr. Giovanni G. Fazio, Smithsonian Astrophysical Laboratory - Harvard
Dr. Carl E. Fichtel, Goddard Space Flight Center
Dr. Phillip C. Fisher, Lockheed Research Laboratories
Mr. Kenneth J. Frost, Goddard Space Flight Center
Dr. Riccardo Giacconi, American Science and Engineering, Inc.
Dr. Herbert Gursky, American Science and Engineering, Inc.
Dr. Henry F. Helmken, Smithsonian Astrophysical Laboratory - Harvard
Dr. Laurence E. Peterson, University of California
Dr. Herbert W. Schnopper, Massachusetts Institute of Technology
Dr. P. Vandenbout, Columbia University
Dr. John R. Waters, American Science and Engineering, Inc.
Mr. T. E. Wing, Columbia University

3.1 DESIGN GOALS. To accomplish the observations required to fulfill the scientific objectives discussed in detail in Section 2.3, the following design goals were established and utilized during the preliminary design of the x-ray telescope:

Lifetime:	At least a year
Orbit:	260 n. mi. or lower circular, 28.5° or less, inclination
Field of View or Beam-width:	10 - 30 arc min.
Angular Resolution:	2 arc sec. or better - depends on jitter rate and number of exposures per sec.
Pointing Capability:	5 arc min. off-set
Lock-on Accuracy:	Within a 1 arc min. square

Jitter:	The jitter must not exceed the angular resolution desired.
Sensors or Detectors:	Image intensifier plus film or electronic imaging tube. Spectrometers. Polarimeters.
Bandwidth:	~ 2 Å to 300 Å.
Spectral Resolution:	$\tau/\Delta\tau$ = 100 to 1,000 or better
Collecting Area:	200 cm^2 minimum effective collecting area.
Lock-on time:	Seconds to several hours.
Data:	5-10 x 10^6 bits per orbit for scientific information.
Structural Tolerance:	Paraboloidal - hyperboloidal reflecting surfaces must be precision machined and the blank or a coating, such as electrodeposited nickel or flame sprayed nickel oxide, optically polished to reflect grazing incidence x-rays down to at least 2 Å wavelength. The alignment of the imaging mirrors with the focal point must be kept so that the image remains entirely on the detector sensing area.
Physical Deployed Dimensions:	Up to 10 ft. diameter by up to 100 ft. long.

The low orbit altitude and inclination requirement stems from the necessity of remaining clear of the earth's radiation belts. This is to cut down background noise to a tolerable level for studying very weak stellar and galactic sources.

The term minimum effective collecting area refers to the projected frontal area of the parabolic part of the glancing incidence mirror system, multiplied by the glancing incidence reflection efficiency. This efficiency increases with the decreasing diameter of the smaller nested mirrors, since they have larger F/D. Thus, the effective collecting area is very much dependent on the field of view, the number of concentrically arranged mirrors, the mirror (and/or mirror coating) material, and the focal length to objective diameter ratio (F/D).

3.2 TYPICAL MISSION PROFILE. The principal investigators will dictate the mission requirements. It is probable that the order of events in a typical mission profile may take this form:

a. Command lock-on to within ± 0.5 arc min. of known coordinate spot in the sky - accomplished by guidance system using on-board computer and star trackers.

b. Visual telescope (slaved to the x-ray telescope) identification of target area. Electronic comparison of visual telescope image with desired coordinates holds the x-ray telescope to within a field of 1 arc min. by 1 arc min. centered on the desired coordinates.

c. The imaging, spectrometry and polarimetry investigations of the target take place, while the automatic attitude control system holds the telescope pointed with the 1 arc min. by 1 arc min. field centered on the desired coordinates. The desired image resolution of 2 arc sec. (or whatever resolution the lens system can achieve) is obtained using an image motion compensator to correct for drift.

   This step may entail on the order of one to 60 min. for imaging, from 10 to 150 min. per line for spectrometry, 60 to 1,000 min. for the entire spectrum using a slitless spectrometer suggested by A. S. & E. and 5 to 60 min. for polarimetry, assuming a source on the order of 10^{-4} the intensity of SCOX-1 (SCOX-1 has an intensity of approximately 18 counts/cm^2-sec.) and an x-ray telescope on the order of 30 - 40 in. mirror diameter with two to eight tiers.

d. A traverse in pointing direction takes place and steps 1-3 are repeated for another x-ray source or interesting area of the sky. Most sources will be within $\pm$ 20° of the galactic equator. The Milky Way will provide a large number of viewing opportunities.

e. The data is stored in the spacecraft and transmitted to ground upon passing a ground station. If film is used for imaging, the exposed film will periodically be retrieved and resupplied by the astronaut.

f. The data is analyzed on the ground.

3.3 <u>DETECTION SYSTEMS.</u> No attempt has been made to make a complete list of typical experiment equipment. The individual scientific principal investigators will specify the particular type of research or observation and detection instrumentation that they wish to employ. The discussion below serves to point out the typical type of equipment that has received attention during the design of the telescope system, and also to show some projected resulting performance of these telescope/detection instrument combinations. Other instrumentation would not change the basic design of the telescope.

There are three primary detection systems for use with the imaging x-ray telescope, for which alternative instrumentation is suggested:

a. Imaging system - either film or electronic (with image motion compensation).

b. Spectrometer system - either a slitless spectrometer as discussed by A. S. & E. (Reference 26) or a focusing crystal or concave diffraction grating type device.

c. Polarization system - a light element crystal scattering polarizer, or a Borrman Effect crystal device.

Preliminary instrument parameters are shown in Table 3-1. An equipment block diagram is displayed in Figure 3-1.

3.3.1 Imaging Instruments. For effective imaging to 2 arc sec. resolution, all jitter and pointing stability total motion must be kept within a 2 arc sec. amplitude during that amount of time designated for each exposure. Assume a total jitter rate of 2 arc sec./sec., and that the image must be formed from information obtained once per sec. to achieve the 2 arc sec. resolution required. Further, assuming a 1 m diameter telescope (with

Table 3-1. X-ray Telescope Typical Equipment Parameters

Instrument:	Dimensions:	Weight of Assembly:	Power Requirements:	Remarks:
Image Intensifier	Disk 10 cm dia. + 1 ft^3 electronics	30 lb.	20-40 w	Converts every tenth x-ray photon to 10^5 visible photon
Image Motion Compensator (IMC)	1 ft^3 electronics	20 lb	20 w	Provides electronic motion compensation
Vidicon/ Orthicon Astracon	1 ft long by 1 ft dia.	20 lb	40 w	
Optical Telescope	10"-12" dia. x 6'-10' + 3 ft electronics	100 lb	100 w	Provides optical correlation images and signals for image motion compensator - uses II, Vidicon and computer
Slitless Spectrometer	1/4" x dia lens (frame)	10-30 lb	Used with II, IMC Vidicon	2-300 Å with several gratings. Ruled gratings placed behind lens assembly. Only for point sources. $\lambda/\Delta\lambda \simeq 100$ resolution
Focusing Crystal Spectrometer	Up to 60" x 60" x 6"	40 lb	20 w	2 Å - 25 Å operating wavelength range $\lambda/\Delta\lambda \simeq 1000$ resolution
Concave Grating Spectrometer	2m x 11/2m x 15cm	40 lb	20 w	Extended Objects $\lambda/\Delta\lambda \simeq 1000$ resolution
Solid or Liquid H_2 Polarimeter	40 cm long wedge x 10 cm thick and prop. counters along sides	100 lb	10-20w	Liquid He cooling system
Camera and Film	2 x 2 x 3 ft	400-500 lb	10 w intermittent	Includes 00 lbs of film. Film drive and spool change.
Sun Shield Membrane	1/4" x dia lens	10-20 lb		Must be removed for non-solar observations. Mylar or beryllium - .25 or .50 mil

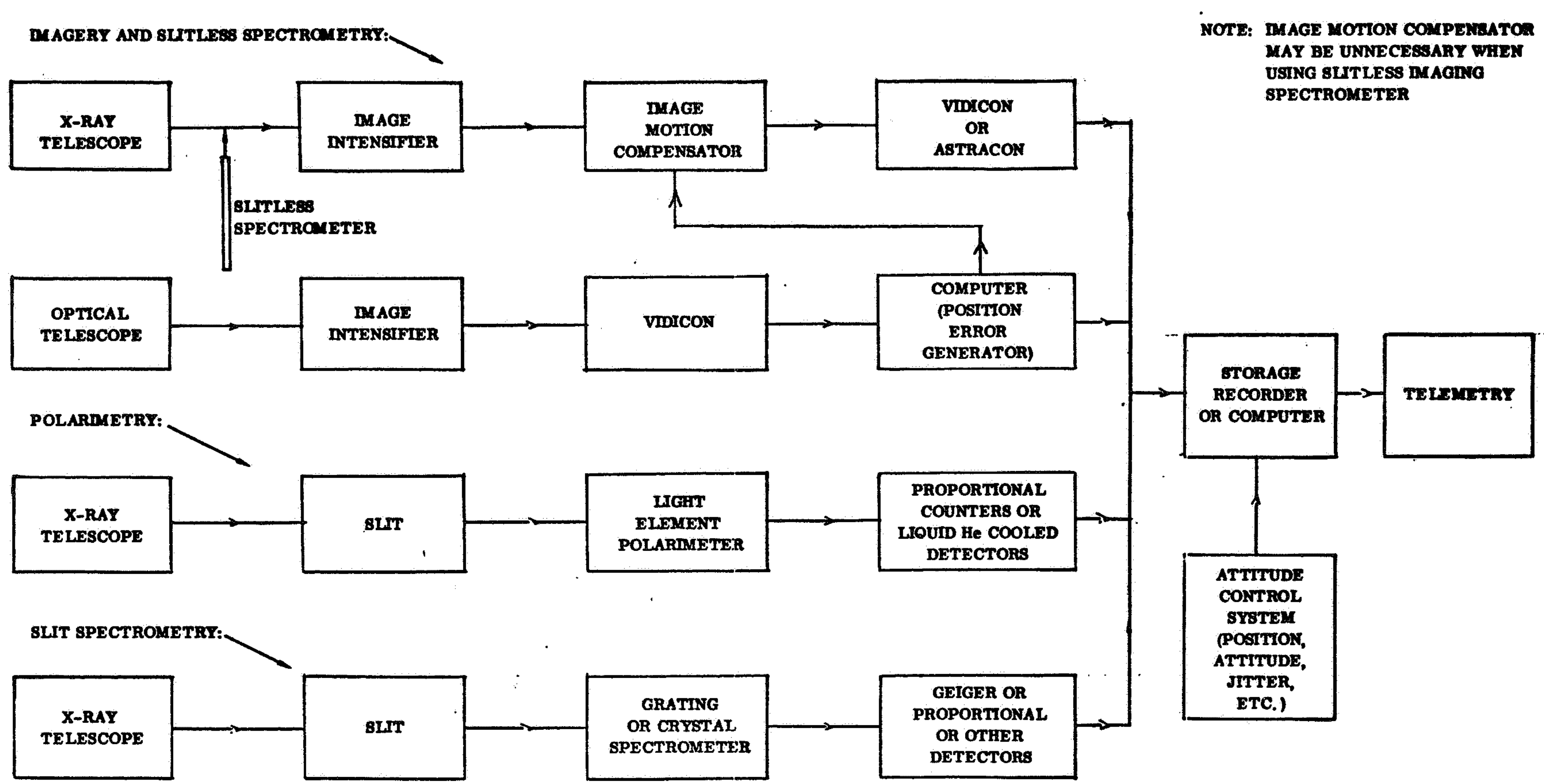


Figure 3-1. Research Equipment Block Diagram

an effective collecting area of 750 cm^2) and the desire to obtain the image of an x-ray source 10^{-4} SCO X-1 in intensity, calculations show that at least 150 sec. would be required to sensitize even the best x-ray film (need 10^7 photon/cm^2) to get even one dot somewhere on the image. Thus, for film, a source 150 times stronger than SCO X-1 is needed if a jitter rate or drift of 2 arc sec./sec. is allowed. This shows the explicit need for image motion compensation, since the pointing accuracy of the telescope is required to be only within a 1 arc min. by 1 arc min. square. Image motion compensation can be avoided only at the cost of decreasing the resolution requirements to that of the pointing requirement.

The use of an image intensifier would allow some relaxing of this requirement, in that under the same conditions, the same image could be obtained in 60 sec. of time rather than 150. Again, if film was used at the rate of one frame per sec., the film usage rate would become prohibitive. (The ATM will employ film but has pointing accuracy within a square 2.5 arc sec. by 2.5 arc sec., which is also the resulting resolution.)

Instead of either of the above alone, it was suggested by Drs. Giacconi and Gursky that an image motion compensator be introduced behind the image intensifier. This would electronically compensate for the movement of the telescope back and forth within this 1 arc min. box, in order that the film or electronic imaging tube not be affected by any other motion than the allowed 2 arc sec. jitter. Such a device could be built deriving its cues from the optical telescope image. It is further suggested to completely depart from film to using an electronic imaging tube as recording medium. It is probable that such a tube with up to one half hour "memory" exists - called an "astracon".

Thus for imaging, the recommendation is to use a system as shown in Figure 3-1 (ref.), where the x-ray scanner would be either some type of vidicon tube or an "Astracon", or possibly film plus a line scanner. There is little doubt that such an Image Intensifier - Image Motion Compensator - Electronic Imaging Tube System would be feasible before 1974 and that it would be the better imaging system. The size, weight and power requirements of the image intensifier are on the order of those of a video system.

3.3.2 Spectrometers.

Slitless Spectrometer. (Reference 26)
This is an A.S. & E. suggested device, consisting of a fine wire grid placed immediately behind the grazing incidence lens assembly. The grating itself blocks about 50% of the incident energy that has been collected by the lenses, and about 40% of that total power is diffracted. Due to the resulting efficiency of 10%-20%, calculations show that, for the telescope of 1 m diameter

assumed above, the lock-on time for a 10Å entire spectrum with 0.025Å resolution would approach 10^4 sec. for a source the strength of about 10^{-4} that of SCO X-1. The sensor end would utilize either film or electronic imaging in conjunction with an image intensifier.

Focusing Crystal Spectrometer.

This particular type of device has already been flown in rocket flights, using a pair of crystals, each refracting the x-ray energy of a particular range on to a photo multiplier tube. It is a very high resolution device, and is used to study one spectral line at a time. Because of the requirement that only one line per crystal can be registered on the scanning equipment for each integration time, the available incoming energy is either split up on to several crystals, or to one line at a time with one crystal; both alternatives imply long integration (lock-on) times. For example: with the telescope and source assumptions above, the integration time per line is approximately 600 sec., using an image intensifier and film or electronic imaging devices to register the output beam energy from the Bragg crystals. The entire instrument would be placed a ft. or two behind the focal point, where a collimating slit must be positioned. The energy is refracted from the crystal on to the single counters (photo multiplier tubes, e.g.) and different lines are obtained by rotating the crystal. The entire crystal and mechanism package is on the order of 8 x 8 x 3 in. for solar x-ray spectrometry, and will probably be at least three to four times that size for the sensitive stellar work that this large telescope will be capable of.

Slit Type Concave Grating Spectrometer.

Compared to the Slitless Spectrometer above, this instrument has the advantages of better efficiency and higher dispersion. It would also be capable of providing spectra of extended sources, whereas the slitless spectrometer would only be able to handle point source spectra. However, the instrument would require a large distance (on the order of one meter) between the slit (located at the focal point) and the diffraction grating. It would also require a long bent film plate or image intensifier to register the spectrum.

3.3.3 Polarimeters. Polarimetry can be accomplished using a) a scattering crystal of a light element, b) the Borrman Effect, or c) the Compton Effect.

In a scattering polarimeter, depending on a judicious scattering of the incoming photons in differing directions depending on their polarization, a light element crystal is used. This instrument has the advantage of being largely frequency independent, so that one crystal may suffice for the entire bandwidth under investigation.

One configuration that was suggested was the use of a solid or liquid hydrogen scattering crystal, cooled by liquid helium. The entire apparatus is placed behind a slit at the focal plane, and the crystal is surrounded by counters on all four lateral sides. These detectors could be, for instance, liquid helium cooled germanium detectors, or proportional counters. The size of the crystal would be on the order of 30-40 cm long, wedge shaped, and perhaps 10 cm thick.

The Borrman Effect depends on the selective transmission of energy along planar paths in selected crystals. The problem here is that each crystal is very selective as to wavelength, and two orthogonal crystals are required for each wavelength to ascertain the polarization components in two directions.

The Compton Effect utilizes a carbon block, and would entail extremely long counting times.

Table 3-2. Measurement Requirements/Performance Comparison.

Parameter:	Goals:	Obtained by ATM	Achievable by Large X-Ray Telescope	Comments: (pertain to large telescope)
Field of View	10 - 30 arc min.	40 arc min.	30' arc	A matter of trade-offs
Resolution	1 arc sec. or better	2.5-15 arc sec.	1 - 5 arc sec.	1966 Technology: 5 arc sec.
Pointing Capability	5 arc min. offset	± 16 arc min. offset	5 arc.min. offset	X-ray source may not coincide with visible object
Lock-on Accuracy	Within 1 min. by 1 min. arc	± 2 arc min.	1 min. by 1 arc min.	Image motion compensation within this area
Jitter	< 1 arc sec./sec.	≤ 2.5 arc sec.	≤ 1 arc sec./sec.	Structural and thermal vibrations
Lock-on Time	1-360 min.	100 sec.	Up to 60 min.	Can be designed for up to 360 min. by use of larger momentum wheels
Bandwidth	2Å - 300Å	2 - 8Å 3-100Å	2-300Å	2-25Å most important up to 300Å for solar observation
Sprectral Resolution	$\lambda/\Delta\lambda$ = 100 to 1000	0.1Å	$\lambda/\Delta\lambda$ = 1000	Focusing crystal or concave grating spectrom. (100 with slitless spectrum.)
Polarization	Yes		Yes	Cryogenically cooled detectors and crystal
Projected Hori-tal Collecting Area	200-2000 cm^2		50-750 cm^2	Depends on the number of concentrically arranged elements.

3.4 LARGE X-RAY TELESCOPE PERFORMANCE. The Convair design has responded to those design goals set forth in section 3.1 above. The scientific requirements, as portrayed there can be easily compared in Table 3-2 to those obtained by the large grazing incidence telescope. Some values predicted for the ATM are included for illustrative purposes as described in References 27, 28 and 29.

The telescope/detection equipment combined performance is the key to the attractiveness of larger and larger collecting areas. By collecting larger amounts of arriving energy with the lens assembly and focusing this onto a relatively small detector surface area, a significant improvement in signal-to-noise ratio is achieved. The latter application permits the use of dispersive techniques for high-resolution spectroscopic measurements, and affords improved possibilities for polarization measurements.

It is appropriate at this point to show some calculations of typical observation integration times to see the advantages realized with the size order telescope proposed here.

3.4.1 Electronic Imaging. For most cases in which imaging systems are used in practice the lower limit of the intensity of a discrete source that can be detected by an x-ray telescope is set by the requirement that a significant number of photons be detected within an image resolution element. This can be understood by considering that the area of a resolution element in the focal plane is many orders of magnitude smaller than the area of collection. The part of the background that depends on the area of the detector element, such as that produced by cosmic rays, will therefore be reduced by a correspoinding factor when compared with detectors with only mechanical collimation, and can be further reduced by increasing the angular resolution of the device. The diffuse cosmic flux is much weaker than the background from other factors. In view of these considerations, the effective sensitivity of a focusing telescope is proportional to A_t, the product of the area of collection and the time available for observation. The sensitivity of mechanically collimated detectors, which is determined by the signal-to-noise ratio, is, on the other hand, proportional to $(A_t)^{1/2}$ (Reference 2).

It is possible to use either film or some type of vidicon with a built-in memory storage for image recording. For discussing the capabilities of the large telescope assume an electronic imaging system utilizing an image intensifier. Then, making the conservative assumptions that:

a. The image intensifier is only 10% efficient,

b. 10 counts are needed in one resolution element for statistical assurance that it is in fact an image point, and further that the total energy from a point source is focused on one resolution element, the imaging time necessary for a point source to register in one resolution element can be shown versus source intensity. Figure 3-2 is plotted for effective collecting areas of 50, 100, 300 and 500 cm^2. The effective collecting area is here defined as the total frontal projected collecting area, multiplied by the realistic double reflection efficiency at any particular wavelength for the specific mirror or coating material. From Figure 3-2 it is obvious that a point source with the strength of SCO X-1 demands only 1/100 to 1/10 of a sec. imaging time, while one 10^{-4} times the intensity of SCO X-1 would require approximately 100 to 1000 sec. (1.66 to 16.6 min.) of imaging time. This points up the need for telescopes with large effective collecting areas. The lower of the above integration times would require a telescope with a diameter on the order of one meter, with several concentrically arranged parabolic - hyperbolic collecting mirrors. (The number of tiers depends on the focal length, field of view, material, wavelength, etc.)

The projected frontal areas of the telscopes under investigation here for a focal ratio of ten, are from 50 to 529 cm^2 when considering 20 to 40 in. internal diameter of the largest mirror, and one to three concentrially arranged mirrors.

Projected Frontal Areas at F/D = 10:

	20 in.	30 in.	40 in.
1 mirror	50 cm^2	114 cm^2	202 cm^2
2 mirrors	95	213	378
3 mirrors	130	297	529

These areas have been computed, on the basis of the geometry shown in Figure 3-3, from projected frontal area (Reference 5).

$$\pi (y_o + \alpha L)^2 - \pi y_o^2$$

$$= \pi(y_o^2 + 2 \alpha L y_o + \alpha^2 L^2) - \pi y_o^2$$

for α small, ignore $\alpha^2 L^2$; thus area = $2 \pi \alpha y_o L$, where

$4 \alpha = y_o/F$ so that projected frontal collecting area per lens in the

nested lens assembly is $\dfrac{\pi y_o^2 L}{2F}$.

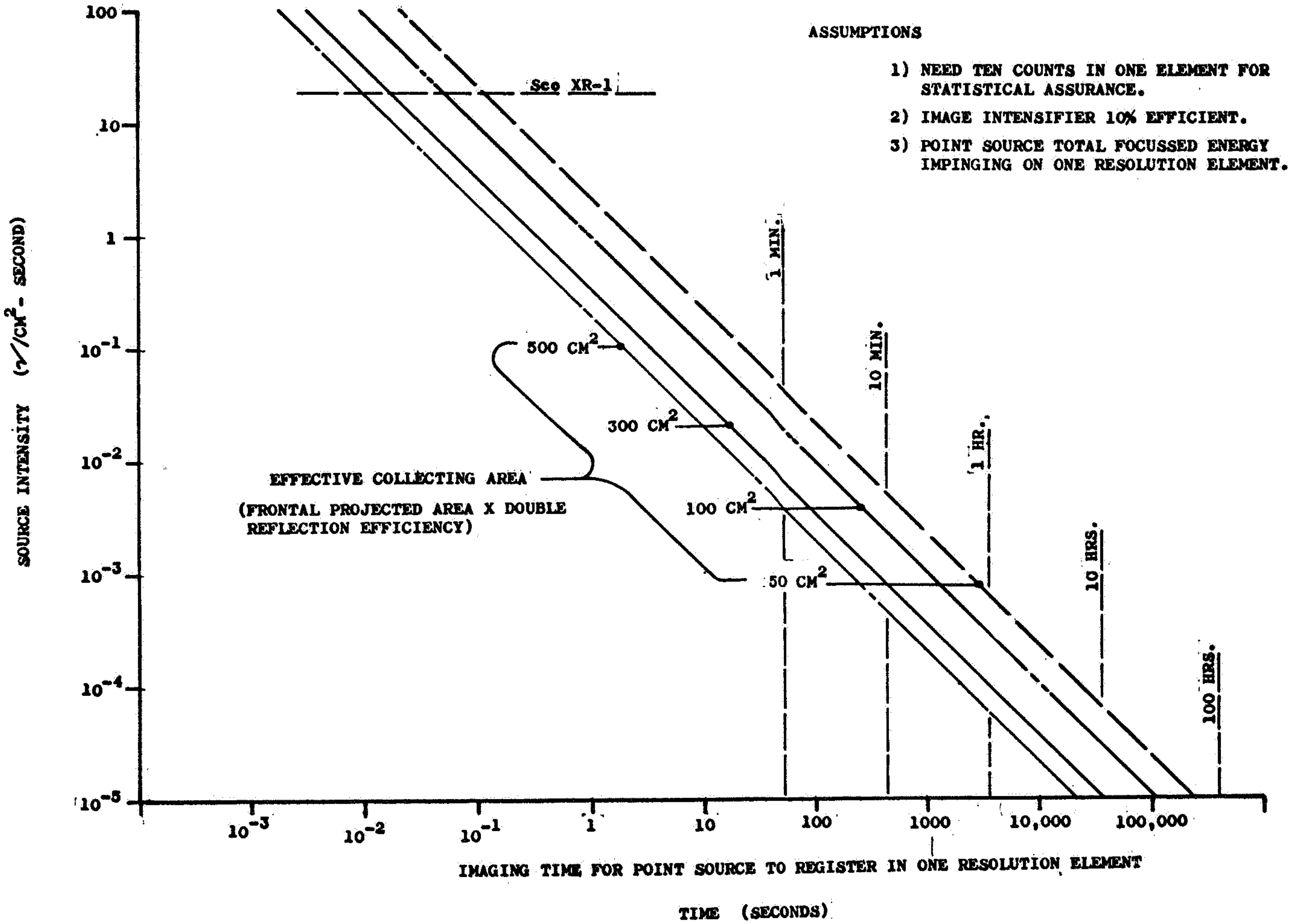


Figure 3-2. Imaging - Using Electronic Imaging Intensifier

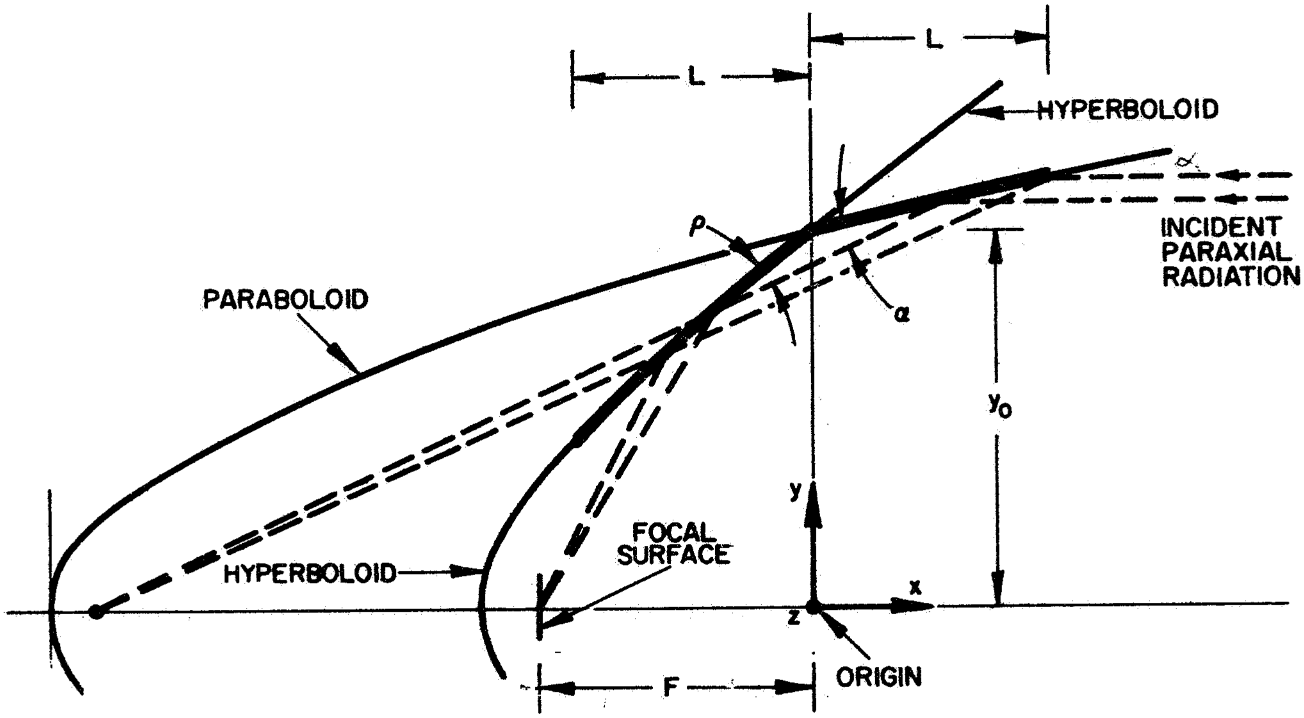


Figure 3-3. Schematic Cross-Section of an X-Ray Telescope

To proceed from the projected frontal areas to the effective collecting areas, it is necessary to know the realistic efficiency of the focusing system as shown by Figure 3-4, when working with a 30 arc min. field of view, the maximum grazing incidence angle occurs for the largest mirror since it has the lowest F/D ratio of the concentrically arranged mirrors and is 58 arc min. for F/D = 10 and 51 arc min. for F/D = 12. The other concentric mirrors all have larger F/D numbers and hence smaller incidence angles, which will improve the reflection efficiency.

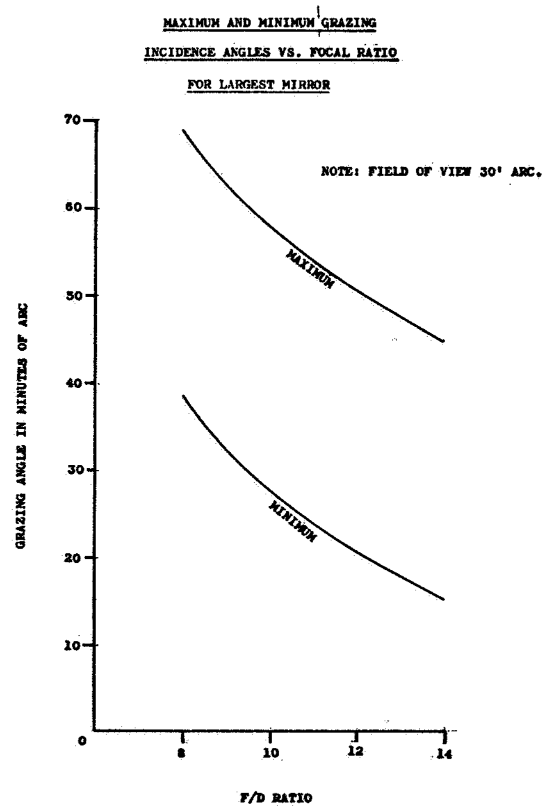


Figure 3-4. Grazing Incidence Angle

Assuming for the moment that a nickel coating reflecting surface is applied to the substrate mirrors we have the theoretical double reflection efficiency plotted in Figure 3-5 for various grazing incidence angles (Reference 5). The double reflection efficiency decreases sharply with increasing grazing incidence angle. The reflection efficiency curves for nickel have a sharp absorption edge at approximately 13Å, but pick up again rapidly after that and remain fairly flat up to a couple of hundred Angstrom. Realizing now that the double reflection efficiency is not constant versus wavelength, for illustrative purposes use an average efficiency for the 58 arc min. grazing incidence angle mentioned above and those significantly smaller grazing incidence angles which will pertain to the inner concentric mirrors. Further degrading this theoretical average double reflection efficiency for practical purposes we arrive at an assumed realistic average double reflection efficiency of 0.5. From the example, the effective collecting areas are on the order of 50 to 265 sq. cm for two tiers. With this result return to Figure 3-2 (ref.) to find that the 40 in. diameter three tier telescope will require on the order of 6 min. imaging time for a point source 10^{-4} the intensity of SCO X-1. This is an effective reminder that to avoid excessive image integration time it will be necessary to have either several tiers or a very large diameter telescope.

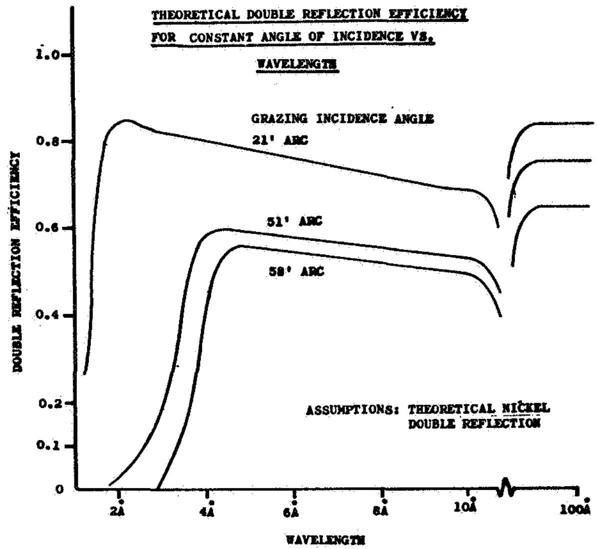


Figure 3-5. Reflection Efficiency

3.4.2 <u>Spectrometry and Polarimetry</u>. Equally important in the astronomy investigations, the source intensity versus wavelength and the polarization of the emanating energy will demand a considerable portion of the orbital research to be performed by the x-ray telescope. Although specification of the actual experiment equipment to be accommodated by the facility is up to the various principal investigators in the scientific community, for a point of discussion return to the hypothetical experiment equipment outlined in Table 3-1 (ref.). To form a better idea, the typical equipment block diagram is as illustrated in Figure 3-1 (ref.).

In utilizing the image forming slitless spectrometer, as suggested by American Science and Engineering, Inc., the following assumptions were made.

a. The grating blocks 50% of incoming flux.

b. Approximately 40% of the remaining energy becomes diffracted flux.

Since the spectrum will be spread in two directions over the image plane, this leaves 10% of the incoming flux available for one spectrum. A reasonable bandwidth to be handled with one grating is about 10Å. Hence, with an achievable resolution $\lambda/\Delta\lambda$ of approximately 100, each line will receive an average of 0.001 of the incoming flux. Each line occupies a width of about 1800/200 = 9 one arc sec. resolution elements on the face of the image plane, which results in about 10^{-4} of incoming energy into one resolution element. Assuming that the image intensifier is 10% efficient and needs 5 counts in one element for statistical significance, the useful flux is about 2×10^{-6} of incoming flux to register part of a spectral line in one resolution element.

Taking an effective collecting area of about 300 cm^2, for a source intensity of 0.1 photon/cm^2 - sec. (0.5×10^{-2} SCO X-1) the effective flux registered in one resolution element on the image intensifier is 6×10^{-5} counts/sec. Hence an integration time of some 1.5×10^4 sec. would be required for the entire spectrum for this experiment.

When considering a focusing crystal spectrometer or a concave grating spectrometer, the very high resolution capability must be taken into account. It is probable that an integration time on the order of 600 sec. per spectral line may suffice for a source 10^{-4} SCO X-1.

The above calculations may be overly conservative, and should be regarded as being correct only to within an order of magnitude. The liquid or solid light element polarization instrumentation suggested in Table 3-1 (ref.) will also require quite long integration times for statistically significant measurements.

SECTION 4

TELESCOPE DESIGN

4.1 CONFIGURATION DESCRIPTION. The selected configuration for the nominal 30 inch aperture x-ray telescope is shown in Figures 4-1 and 4-2. The deployed structure provides a focal length to diameter ratio of 12. The same structure design can be used for up to a 40 inch aperture with focal length to diameter ratios of 10.

The telescope structure consists of two open truss frame assemblies. The truss members are 2 inch diameter titanium tubes with fixed ends. The deployed 40 foot structure tapers from a maximum depth of 13.3 feet to 8.7 feet on the afte end and 40 inches at the forward (lens) end. The truss dimensions provide maximum geometric stiffness within the launch vehicle payload volume. The open nature of the structure was selected for the following reasons:

a. Minimum shadowing and thus minimum thermal distortions

b. Maximum EVA access to all assemblies

c. Minimum cross-section area (minimum aero drag)

d. Minimum weight

The open truss design requires shielding of the imaging equipment to minimize background radiation. This is accomplished by shrouding in front of and behind (not shown) the lens and by a narrow collimating tube attached to the image capsule.

The truss tube size was selected for astronaut hand grip compatability rather than stress or dynamic stiffness requirements which are very low. The ratio of deployed to stowed structure length is 2:1. Therefore, a simple rail guided slider extension system is used for maximum reliability; the sliding truss sections are postively locked together after extension, thus eliminating all joint slack.

The satellite electrical power is generated by four nonoriented 35 sq. ft. solar cell arrays which are cantilevered outboard of the main truss on auxiliary frames. This arrangement reduces structure shadowing and tends to equalize the vehicle 3 axes mass moments of inertia, decreasing gravity gradient torques. The attitude control system uses three inertia wheels for fine pointing and four 3 unit H_2O_2 0.5 lb. jet thrusters for coarse maneuvering and wheel desaturation. The navigation system primary sensors are star trackers (2), a third star tracker is provided as a standby instrument and is shielded from the space environment with a protective cover. A solar sensor and an inertial unit of three strapped down displacement gyros provide additional orientation reference during primary star occultation or tracker acquisition periods.

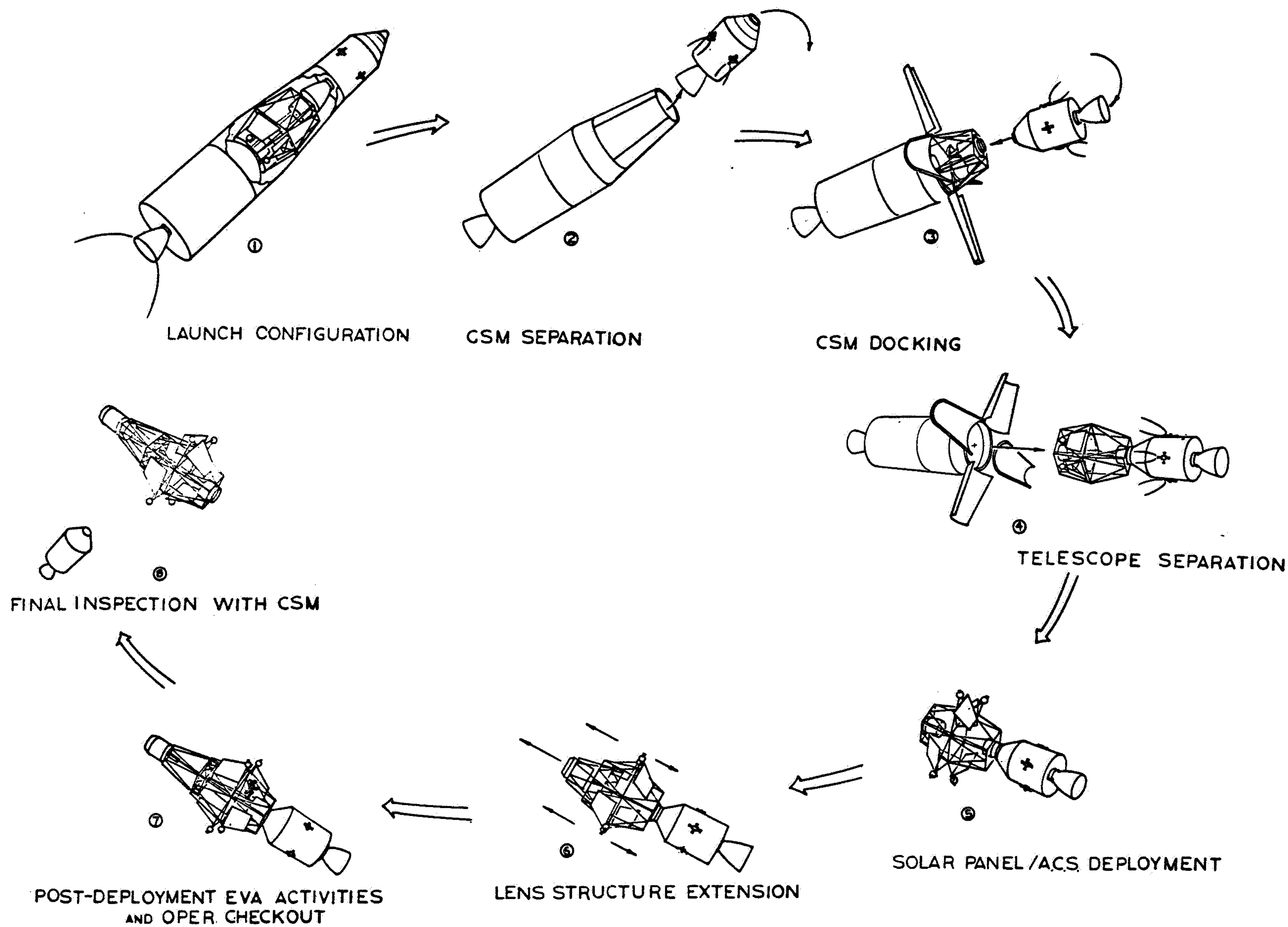


Figure 4-1. X-Ray Telescope Orbital Deployment

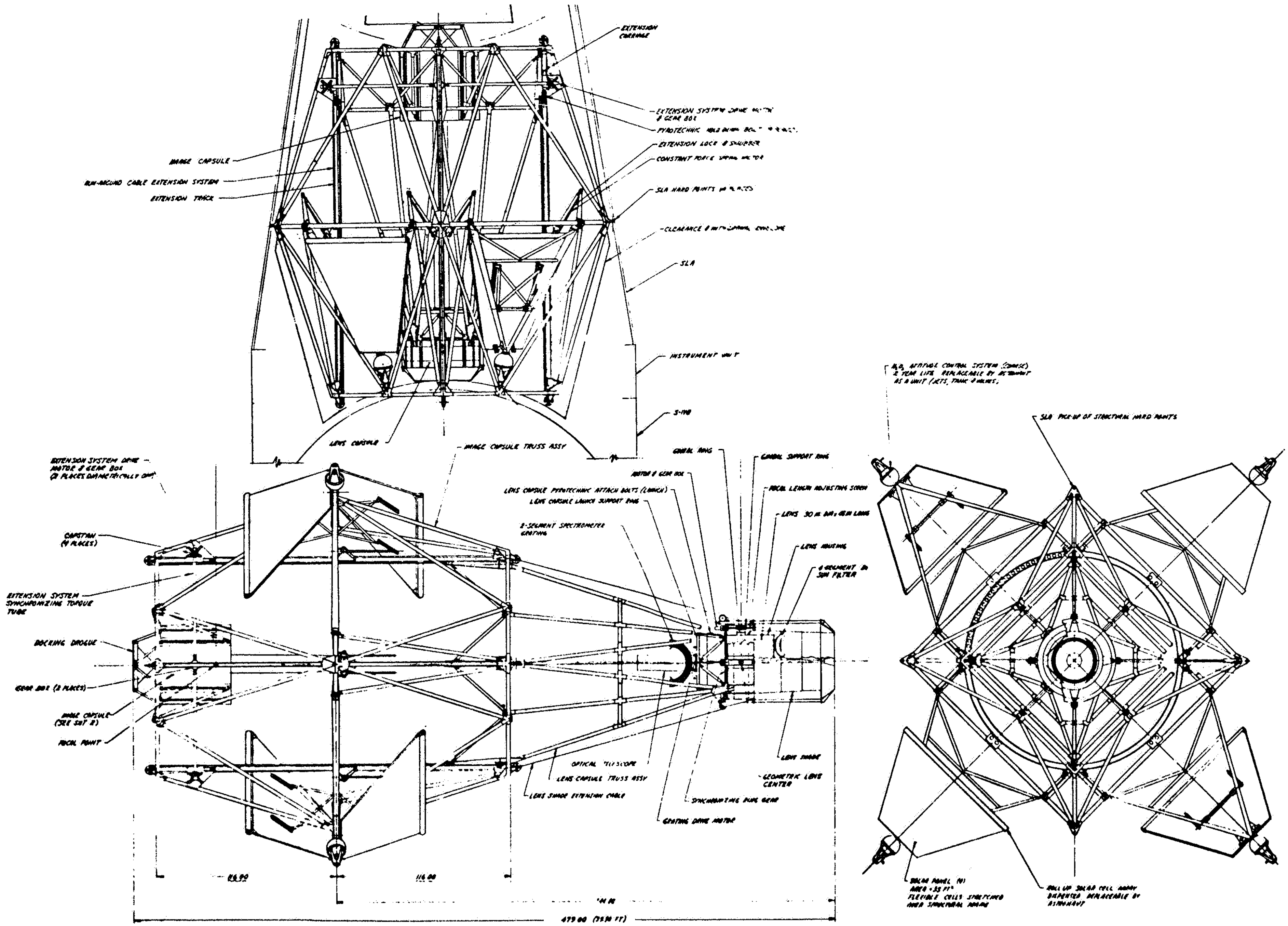


Figure 4-2. Grazing Incidence X-Ray Telescope, Nominal 30 Inch Lens

The preliminary design analysis indicates that thermal distortions of the structure will occasionally exceed critical tolerances, therefore, an optical alignment system is included which consists of laser - reflector equipment capable of measuring changes in the lens axis with respect to the image plane of less than 10 arc seconds. The error signals measured are used to drive the gimballed x-ray lens to a corrected position. The error signals are also relayed to the navigation system to compensate for the changes in effective optical axis. The aft end of the satellite contains an insulated, unpressurized shell referred to as the image capsule. The shell provides micro-meteoroid protection and thermal control for the subsystem equipment. The primary x-ray mission equipment consists of instrumentation for image, spectrometry, and polarization measurements of x-ray stellar and solar sources.

An alternate mode of operation would be to operate the x-ray telescope in conjunction with a manned space station, in this case the telescope would be docked to the station during periods of maintenance, repair, etc. and would operate undocked during the times that scientific observations were being made. If the telescope were used in such a way, some redesign of the docking assembly and provision of rendezvous and docking equipment may be required.

4.1.1 Generalized Structural Requirements. The generalized structural requirements for the orbital x-ray telescope are as follows:

	Design Criteria	Requirements
a.	Size: Stowed Deployed	Payload Envelope Lens Diameter, Lens Focal Length
b.	Weight	Booster Capability, Lens & Equipment
c.	Strength	Launch, Orbit, EVA
d.	Stiffness	Observation, Maneuvering, Docking
e.	Subsystem Volume	Lens Size, Mission Environment
f.	Mechanisms	Deployment, Subsystems
g.	Thermal Response	Observation/Lens, Tolerances
h.	Environment	Meteoroid, Radiation
i.	EVA Provisions	Mission Time, Equipment Reliability, Resupply

The structure size requirements were established by NASA, i.e., lens sizes between 20 and 40 inch diameter and thus, a nominal 30 inch aperture was selected for the study. This range is representative of the next logical instrument size which could be expected to fly in the 1974 to 1975 time period.

The orbit parameters of 260 n. mi., at an inclination of 28 1/2° or less, resulted from a consensus of interested x-ray astronomers, inclinations greater than 28 1/2° have undesirable effects on the observations because of electromagnetic interferences. The uprated S-IB manned launch vehicle has a net payload capability of approximately 6700 lb. to a 28 1/2° 260 n. mi. orbit. Utilizing a Hohmann transfer trajectory; this represents the total telescope maximum allowable weight if launched manned. If launched unmanned, however, a payload capability of 29, 300 lb. is available with a Hohmann transfer, or 24, 400 lb. direct.

Preliminary equipment lists of all synthesized on-board systems established the equipment (image capsule) volume requirements of 50 cu. ft. The preliminary vehicle loading summary is shown on Figure 4-3. The remaining criteria could not be compared until specific concepts were evaluated.

CONDITION	OTHER VEHICLE INTERFACE	STRUCTURE CONFIGURATION	"G" UNITS		FT-LBS ROTATION
			AXIAL	TRANS	
TRANSPORTATION	A/C, TRUCK, RAIL	RETRACTED	+ 1.0	2.5	-
HOISTING (GROUND)		"	+ 2.0	1.0	-
LAUNCH	SIB-SLA	"	+ 4.9 - 1.7	.65	-
DOCKING a. initial b. orbital c. orbital	 CSM-SLA CSM SPACE STATION	 RETRACTED EXTENDED EXTENDED	 (2233#) - 1.0 -1.0	 (3118#) 1.0	 8134
MANEUVERING a. b. c.	 NONE CSM SPACE STATION	 EXTENDED EXTENDED EXTENDED	 ±.0002 ± .15		 4.0 1200
ASTRONAUT IMPACT	350 LB MAN INCL EQUIP	EXTENDED			

* USING CSM 100 LB THRUST ENGINES

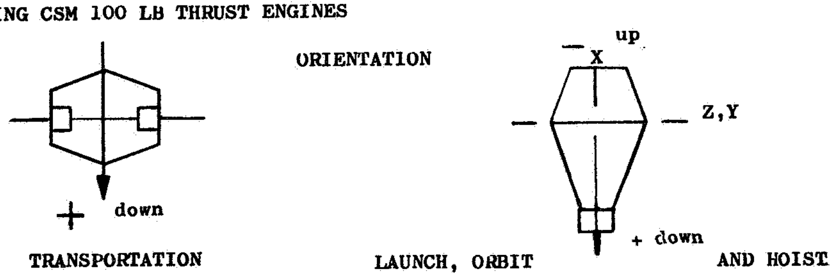


Figure 4-3. Load Summary X-Ray Telescope

4.1.2 Concepts. All known applicable structural concepts were considered for the telescope primary structure. Some of the most promising concepts are shown on Figure 4-4. The simplest and least expensive expandable structure consists of multiple telescoping tubes (piston and cylinder). This design was evaluated in the earlier phase of this study for a 10-foot diameter lens and was found to lack sufficient structural/dynamic stiffness. The oscillations of the lens assembly resulting from attitude control jet forces could not be damped to acceptable levels for observation between ACS firing. The jets were required at short intervals because of the high gravity gradient torques inherent in the long slender structure. This analysis indicated that the mass moments of inertia should be equalized about all 3 axes as much as possible. One solution to achieve increased inertia and stiffness is to deploy transverse booms supporting the ACS jets and solar panels to increase the roll moment of inertia, and add wire bracing to the main boom using the transverse booms as spreader bars. This design, while unrefined did produce acceptable results for the large diameter lens and may still be the most practical solution for a 100-foot configuration.

The lens size range specified by NASA (20" - 40") greatly simplifies the structural design since with nominal focal lengths of 12 times the lens diameter, based on the outer segment, the focal length ranges from 240 to 480 inches. Since the manned booster envelope is approximately 230 inches long, the required structure expansion ratio is approximately 2:1.

The low expansion ratio permits the use of relatively conventional structures such as the open truss. The open truss is a sound and basic geometric approach to minimizing the primary structural problem of thermal distortion.

[illegible]		[illegible]	[illegible]	[illegible]	[illegible]
1. TELESCOPING TUBES		10/1	POOR	FAIR	GOOD
2. HINGED TRUSS		5/1	GOOD	GOOD	GOOD
3. SCISSOR TRUSS		10-14/1	GOOD	GOOD	FAIR
4. FULL-DIA. TELESCOPING TUBE OR TRUSS		5/1	GOOD	FAIR	EXCELLENT
5. WIRE-BRACED TRUSS		—	GOOD	GOOD	FAIR
6. TAPERED TELESCOPING TRUSS		2/1	EXCELLENT	EXCELLENT	EXCELLENT
7. TAPERED TELESCOPING SHELL		2/1	EXCELLENT	FAIR	EXCELLENT

Figure 4-4. Telescope Structure Concepts

Initial designs were based on constant depth telescoping trusses for maximum simplicity and a minimum of moving parts. One redundant feature of these designs was the launch support structure, which contributed little to the basic orbital structural rigidity. See Figure 4-5.

An outgrowth of the foregoing was the tapered truss, "A"frame geometry, which incorporated the primary structure into the launch support structure. An example of this design is shown in Figure 4-6, which also includes a thermal shroud. The shroud has not been established as being necessary. The symmetrical configuration requires the image capsule and lens assembly to telescope approximately equal distances.

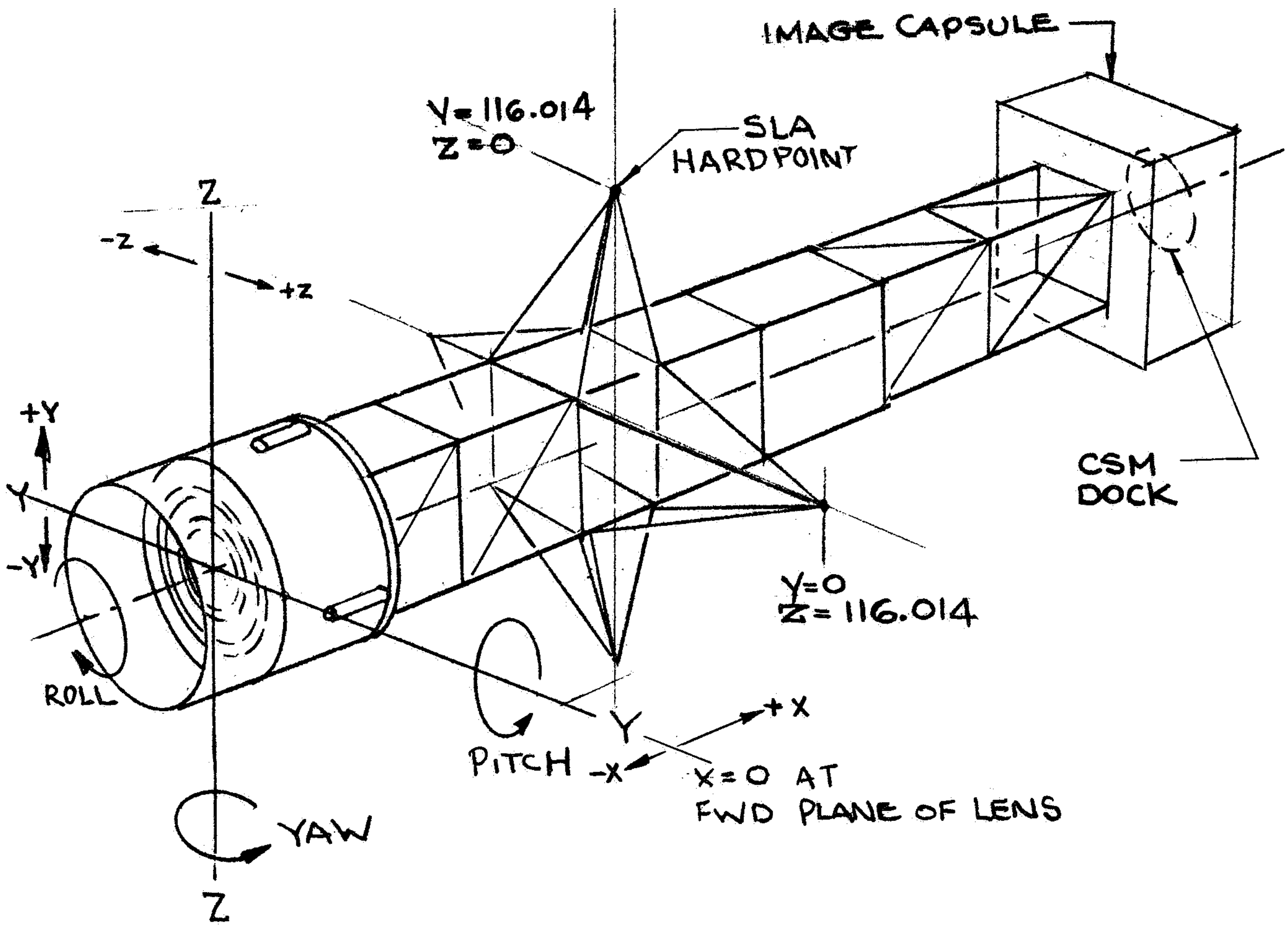


Figure 4-5. Constant Section Truss

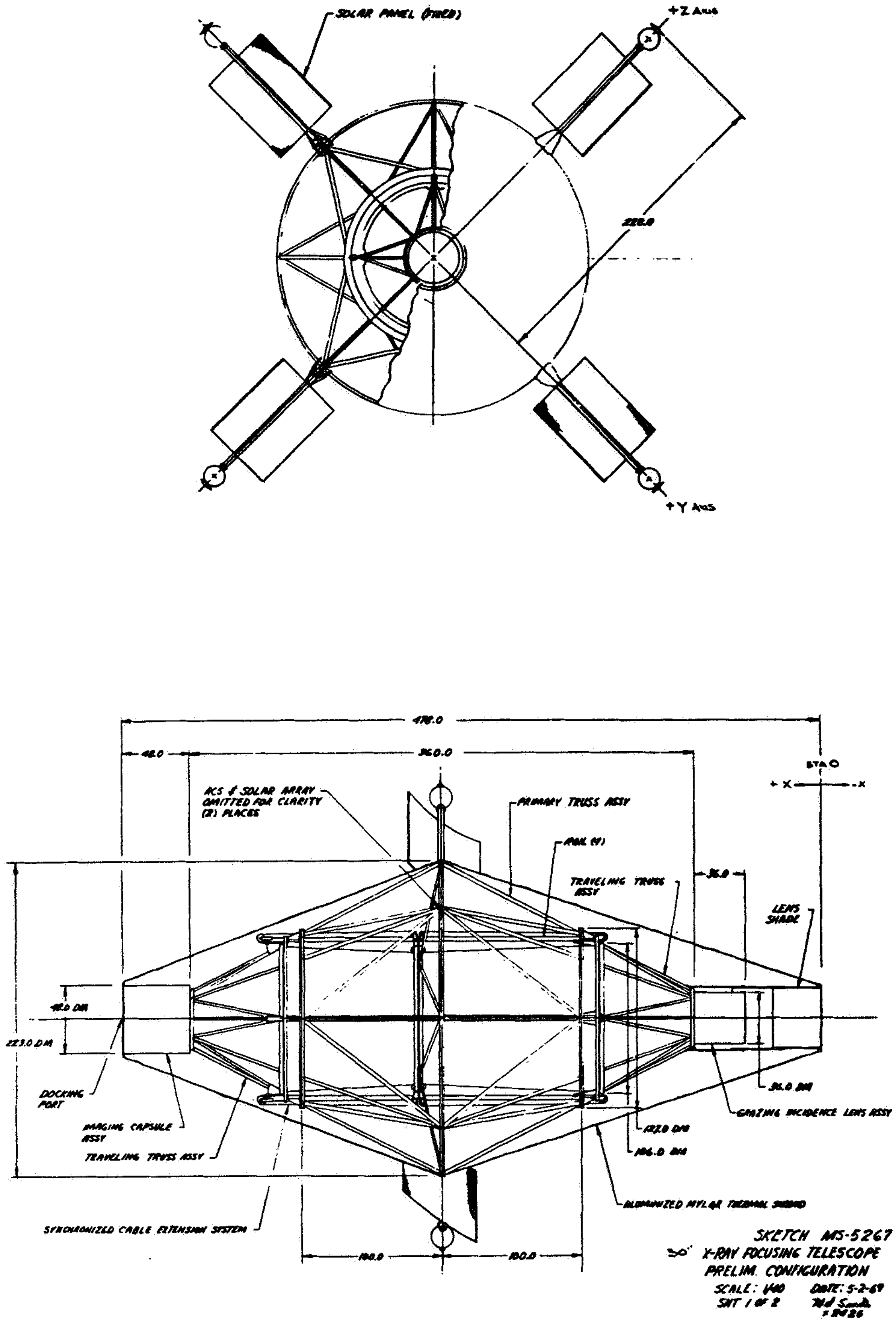


Figure 4-6. X-Ray Focusing Telescope, Preliminary Configuration

Further configuration analysis evolved a simpler single telescoping octagonal structure. A 1/20 scale wood model was built of this configuration. Figures 4-7 and 4-8 are photographs of the model stowed and deployed. The octagonal cross section was used to achieve maximum symmetry and equalize solar heating at all sun angles with the outer thermal shroud of superinsulation. When the thermal shroud requirement was deleted, the near circular geometry was not required and an analysis of 3, 4, and 8-sided trusses (main section only) was performed. Figure 4-9 compares the various geometries. The three-sided truss has the maximum stiffness (bending) and the least members; however, the existing (4) SLA support points can not be picked up without a complicated and heavy transition truss work. The square geometry is the best compromise of rigidity and simplicity and, therefore, was selected for the primary configuration.

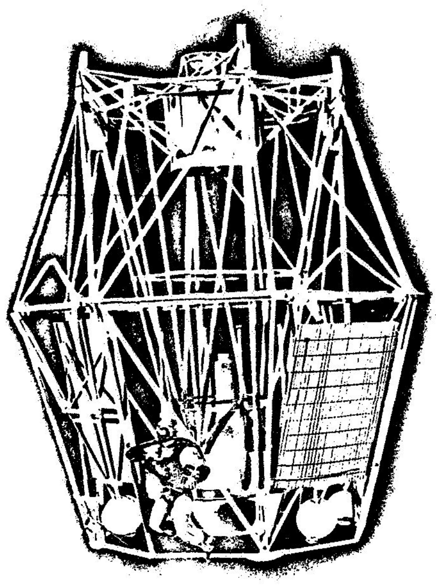


Figure 4-7. Stowed Configuration

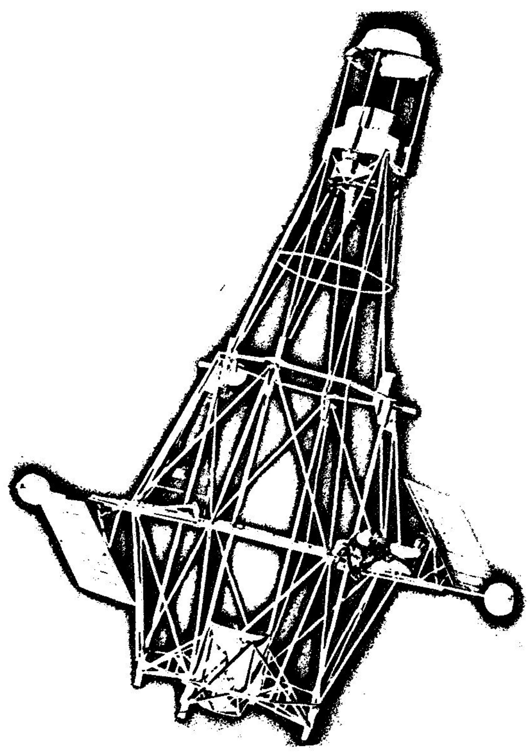
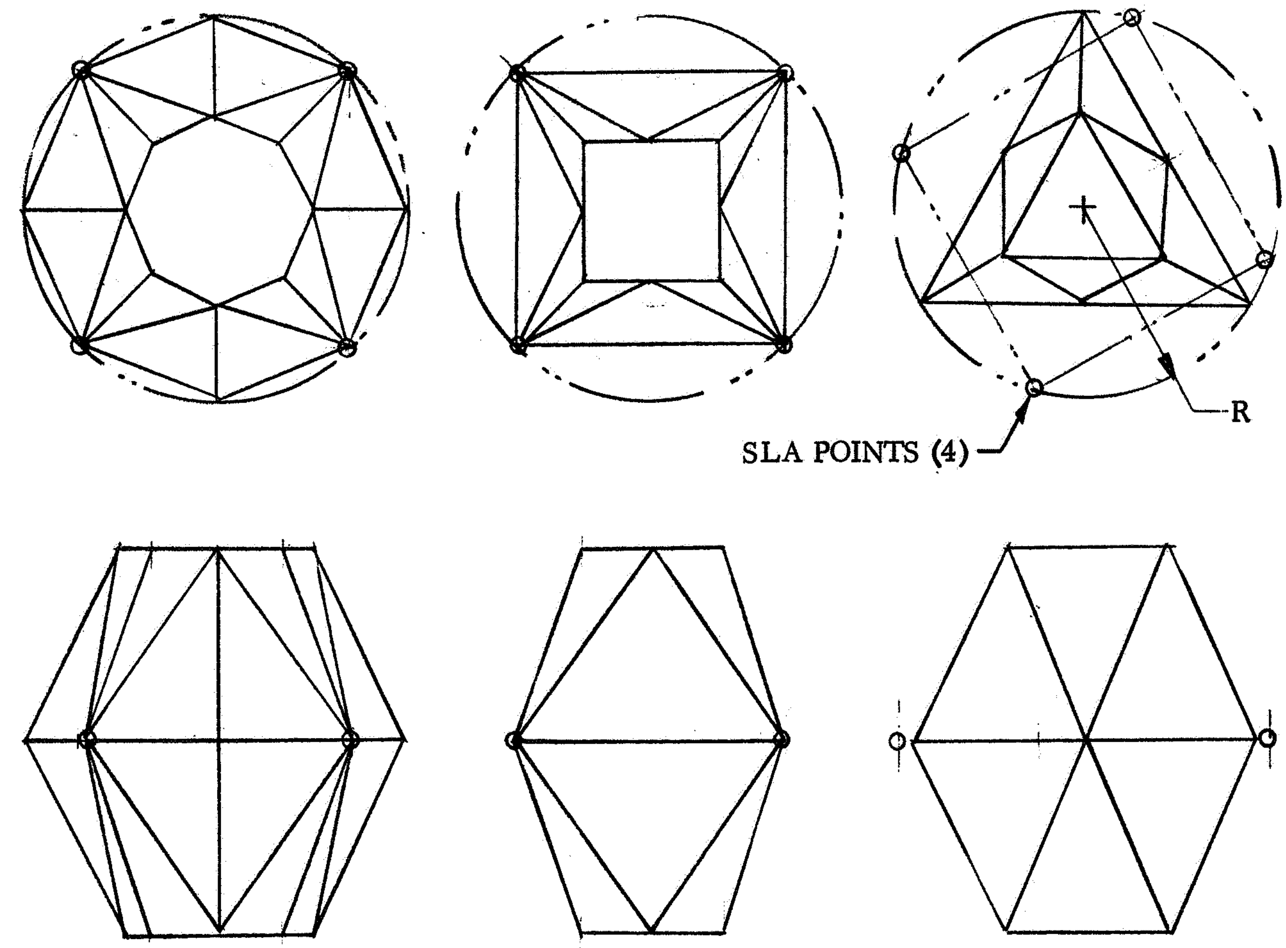


Figure 4-8. Deployed Configuration



LONG MEMBERS	32	24	18
JOINTS	64	48	36
LENGTH-INCHES	4330	3280	1230
INERTIA	$4AR^2$	$2AR^2$	$1.5AR^2$
INERTIA/WEIGHT	.64	.50	1.0

Figure 4-9. Lower Truss Geometry

4.1.3 Selected Configuration General Arrangement. The X-Ray Telescope structure is geometrically configured around a nominal 30 in. lens diameter and a 30 ft. focal length. The primary structure is an open truss frame which provides the prescribed distance between the lens and focal point. This same truss frame distributes the boost phase loads into the Spacecraft Launch Adapter (SLA).

A single telescoping extension converts the telescope from the launch to the deployed configuration. The only other extendables are four "A" frame structures which contain the ACS and the solar panels.

The image capsule, which contains the receiving and processing instruments at the focal point is located at one extreme end of the deployed configuration. This capsule also contains elements of such subsystems as power, ACS, telemetry, navigation and command and control. The lens assembly is controllable for remote focal length adjustments and is suspended within two-axis control gimbals to provide compensating control in the event of supporting structure distortions.

The basic truss frame structure must function as a structural link between the lens and the focal point while minimizing or controlling the deviations from the theoretical optimum geometry resulting from structural distortions. Thus, full advantage is taken of the available maximum width and length dimensions within the confines of the SLA to produce a very open and large type of truss. This also helps to minimize shadowing from such appendages as the solar panels and unrelated experiment packages.

The mechanical functions of deployment are generally single mode and independent rather than series dependent. Once deployed, the structure and appendages are permanently fixed. Subsequently, only the mission instruments (including the lens assembly) have mechanical functions. All mechanical functions have the prime movers designed for permitting astronaut back-up assist in event of malfunction. EVA is not required of the crew during deployment, except in the event of malfunctions. The crew is utilized in the EVA mode following the deployment to remove boost phase restraints within the image capsule.

EVA is used extensively for resupply and/or refurbishment and places a very significant design requirement on mechanical devices relative to crew replacement. This is probably one of the more important factors that has become evident in the course of evolving this large space structure.

The final configuration evolved from several basic geometric cross sections, but generally the four point supports within the SLA dictated figures of four or eight sides in cross section. Shell type structures of circular cross section are adaptable to the same basic configuration and appear attractive in many respects, but time permitted only cursory investigation in view of the considerable increase in structural weight. It is also considered that thermal investigation in more detail would show that the design might be enhanced by a thermal shroud covering the entire truss structure. The design is amenable to such a shroud although it portends other undesirable aspects. The increased aerodynamic drag would require increased propellant, the EVA must be conducted in an artificially lighted jungle and accessibility to components would be significantly impaired.

The ACS/Solar panel components are deployed at maximum distances from the vehicle's centerline to more nearly produce equal mass moments of inertia about all axes. The solar panels are canted, fixed and provide useable power regardless of target aspect. Recent and projected advances in solar cell technology are reflected in the panel design so as to provide sufficient power, minimum shadowing, yet have capability for replacement by the crew.

4.1.3.1 Primary Structure. The most severe constraint placed upon the primary structure is that of minimizing the deviations from the theoretical ray geometry between the widely separated lens and focal plane. This dictates a structure possessing high stiffness with minimum deflections due to thermal gradients. Table 4-I shows characteristics of candidate materials. It appears that beryllium (or Lockally) have by far the most desired characteristics. Beryllium alloy was used for the initial analysis of the primary structure. The extremely high stiffness, desirable from the telescope aspects, was determined to present hazards to the crew. Such hazards exist where collision with a member occurs and wherein the kinetic energy of collision is not absorbed by member deflection within the material yield stress. The characteristic brittle fracture of the thin wall beryllium tube presents potential puncture damage to the astronaut's spacesuit. Other drawbacks to beryllium use lie in its relatively high cost and lack of industry familiarity with fabrication. A further discussion of materials is presented in Section 4.1.3.2.

Table 4-I. Truss Candidate Materials

Material	Density ρ (Lb/In^3)	Yield Stress (Psi	Modulus E (10^6 Psi)	Thermal Conductivity K (Btu/Hr/Ft^2/^{o}F/Ft)	Coefficient of Expansion $\alpha \times 10^6$ (In/In/^{o}F)	$K/\alpha \times 10^{-6}$	Specific Stiffness $E/\rho \times 10^{-6}$	$K/\alpha \times E/\rho \times 10^{-12}$	Equal Weight Impact Coefficient $FY^2/E/\rho$
Beryllium	.067	50000	42.0	87.0	6.4	13.6	627	8520	890
Lockalloy	.076	40000	29.0	123.0	9.2	13.4	382	5100	735
7075 AL	.101	70000	10.3	90	12.9	6.9	102	725	4750
6AL4V Titanium	.160	120000	16.0	3.8	4.8	.79	100	79	5600
Elgiloy	.284	----	29.5	6.6	5.8	1.13	104	118	-----
Invar (39% Ni.) (Magnetic)	.290	40000	21.4	6.3	.28	22.5	74	1660	258
Low Exp. Nickel	.291	40000	21.0	7.8	1.1	7.1	72	510	262

Two large separable assemblies constitute the truss structure. The basic geometric cross section is a square. The square cross section offered the most direct approach, uses a minimum of members and cluster fittings and is well suited to the four-point pickup in the SLA. Also, as compared to an eight-sided cross section, considerable volume is available within the SLA between the flats of the square and the shroud for adding additional experiments.

The truss members are tubular, thin wall and generally 2.00 in. diameter. The base ring in the plane of the pickup points is 3.00 in. diameter. The 2.00 in. diameter was selected from several basic parameters, but more rigorous analysis is needed to optimize the diameter and wall thickness. The selection parameters were:

a. Structure is excellent "hand-hold" if diameter of tube is judiciously selected.

b. Stiffness - large diameter

c. Weight - proper cross-over of D/t. Small diameter.

d. Thermal gradient - small diameter, thick wall.

In consideration of cost, the cluster fittings at the tube junctions are precision cast aluminum. The primary attachment means utilize brazing, with bonding used as a secondary method. The lower strength of bonded joints is offset by improved structural damping coefficient.

The main truss assembly is the larger in diameter and is that which attaches to the SLA hard points. The forward truss assembly supports the lens capsule and is telescoped within and secured to the main assembly by eight pyrotechnic bolts (four places) at launch. Subsequent to deployment, the lens capsule assembly is secured to the main truss by 12 connectors (8 places).

The lens capsule truss assembly resembles a truncated cone with the end terminated at a 50 in. diameter monocoque cylinder which is eighteen in. long. The tubular members cluster at 4 points on the aft perimeter of the cylinder and are attached by brazing or bonding to form a part of the basic truss assembly. A circular ring is suspended by another tubular truss just aft of the above cylinder for purposes of supporting the lens assembly and shroud during the boost phase. This truss picks up machined fittings at the four cluster points.

Truss Extension System. This extension system accomplishes two functions: the extension of the lens and its supporting truss assembly, and extension of the lens shade. The first function is a direct action of a cable run-around system, the second function derives its necessary force input by virtue of the extension action of the first function. The lens truss assembly extends approximately 16 ft. from the stowed position and the lens shade travels approximately 40 in.

The lens truss assembly is guided by four carriages mounted on the aft corners of the truss. These carriages contain rollers of Teflon composition which run against four longitudinal rails within the primary or image capsule truss assembly. The carriages are held to each rail by Teflon slippers on the back flange of the rail.

Deployment is provided by an electro-mechanical extension system consisting of the components shown in Figure 4-10. The drive system consists of two reversible 0.25 HP DC motors driving four capstan gear boxes which are interconnected for synchronization and redundancy by four torque type shafts. A 5/32 in. diameter flexible cable attaches to the forward end of the extension carriage, routes through a sheave at the forward end of the rail, then back through the capstan, through an aft sheave and again attaches to the extension carriage (aft end). This cable system is duplicated at all four rails

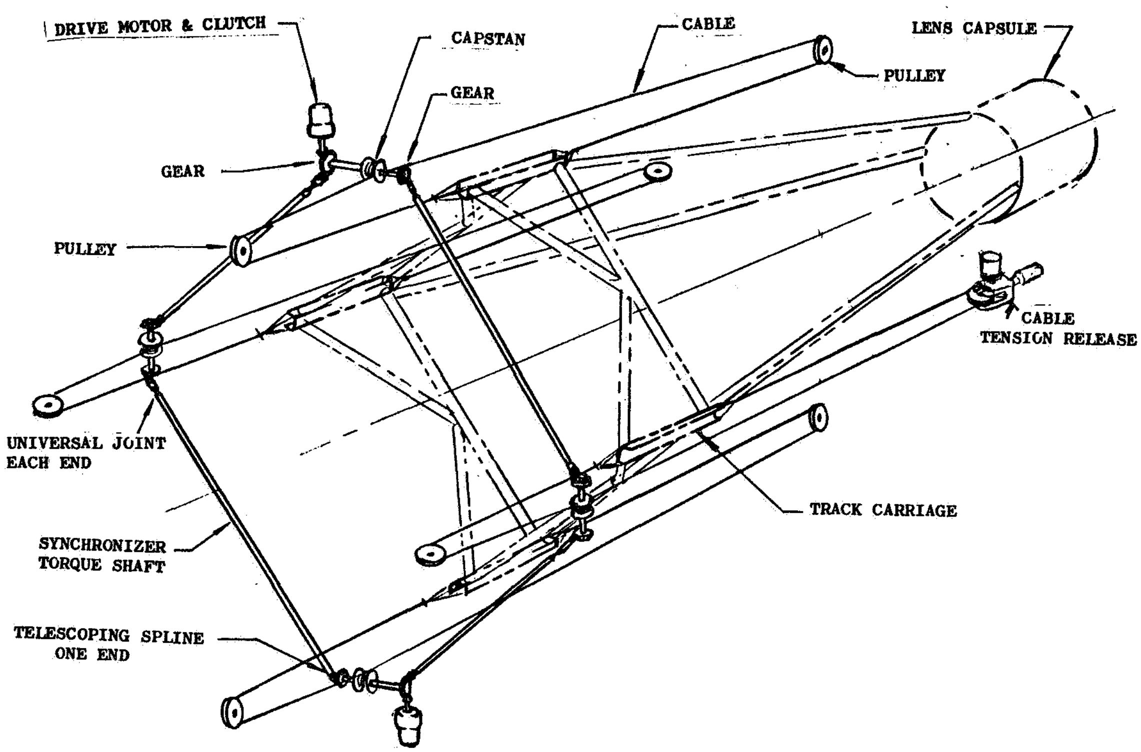


Figure 4-10. Electro-Mechanical Extension System

Rotation of the capstan causes each cable to traverse, hence extension occurs. Action can be stopped at any point within the extension cycle or the cycle can be reversed. Once full extension has occurred, locks are automatically engaged and reversal is no longer possible. The locks consist of a primary engagement of lock tongs with automatic secondary cinch-up by a pyrotechnic ram to remove all joint free play.

The extension cycle is under full control of the crew from within the CM. Deployment against a 1 g longitudinal resisting load would take 5 minutes.

The run-around cable system was selected for its simplicity and reliability. Both drive motors are energized during the extension cycle; however, should one fail, the other has ample margin to deploy the truss. In the event both motors should fail, or a circuit power failure should occur, an EV crewman can complete the extension by engaging a portable power tool into auxiliary gear box receptacles. The drive clutches of the prime motors are electrically actuated for engagement and hence, would fail in the disengage position.

Subsequent to the extension and lock-up function, a pyrotechnically energized mechanism releases one of the cable sheaves in each of the four cable systems. This prevents structural distortions axially as a result of the preload and differential thermal expansion of the cables and truss.

4.1.3.2 Materials. The basic factors affecting material selection of the primary truss structure are:

a. Maximum Stiffness

b. Minimum Weight

c. Maximum Damping Coefficient

d. Minimum Cost (Material)

e. Minimum Fabrication Cost

f. Maximum Impact Resistance (EVA)

g. Maximum Resistance to Thermal Distortion

The above criteria are meaningless without limits since one property must be sacrificed to gain another. With the benefit of the initial analyses in each technical discipline many limits may be derived.

Thermal and dynamic structure distortions summarized from Sections 5. 1. 4. 2 and 5. 4 are shown in Figure 4-11. The dynamic distortions of the beryllium structure analyzed are insignificant (. 371 arc sec. for the jets, and 3.12×10^{-3} arc sec. for the wheels) while the thermal distortions of the pin-jointed model are serious, and require an active lens alignment system. The thermal distortions must be reduced to approximately 10 arc sec. to eliminate the requirement for the lens alignment system. Assume that the structure stiffness coefficient could be reduced by a factor of $\frac{10}{.371} = 27$ for the jets, and $\frac{10}{3.12 \times 10^{-3}} = 3200$ for the wheels.

It is conservative to assume that the material elastic modulus (42×10^6 for beryllium) can be reduced by at least 10 while still maintaining acceptable jitter during jet firing, reductions greater than this may cause problems during other phases of operation, particularly when the vehicle is docked to the CSM and the CSM uses the 100 lb. thrust attitude engines. All materials of interest have sufficiently high elastic modulus to virtually eliminate dynamic problems. This results from the extreme rigidity of the selected geometry.

Structure weight can not be considered a primary constraint since the total truss weight of 337 lb. represents only 5% of the launch vehicle capability, ($\approx$6700 lb). Damping coefficients can be neglected providing the vibratory amplitudes are below acceptable limits. Truss material manufacturing costs are not strong influences since the structure cost represents a minor percentage of the total experiment cost. Space structures comprised of long slender members present an astronaut impact problem. The required tube weight is generally directly proportional to elastic modulus times density and inversely proportional to the square of the yield stress for a given impact energy. Assume an astronaut of mass M is moving at velocity V_1 and strikes a tube in mid-span. Setting the kinetic energy of the moving astronaut equal to the potential energy of the deflected tube

$$.5MV^2 = .5PX. \tag{1}$$

The deflection of a fixed end beam of length L under a center point load equals

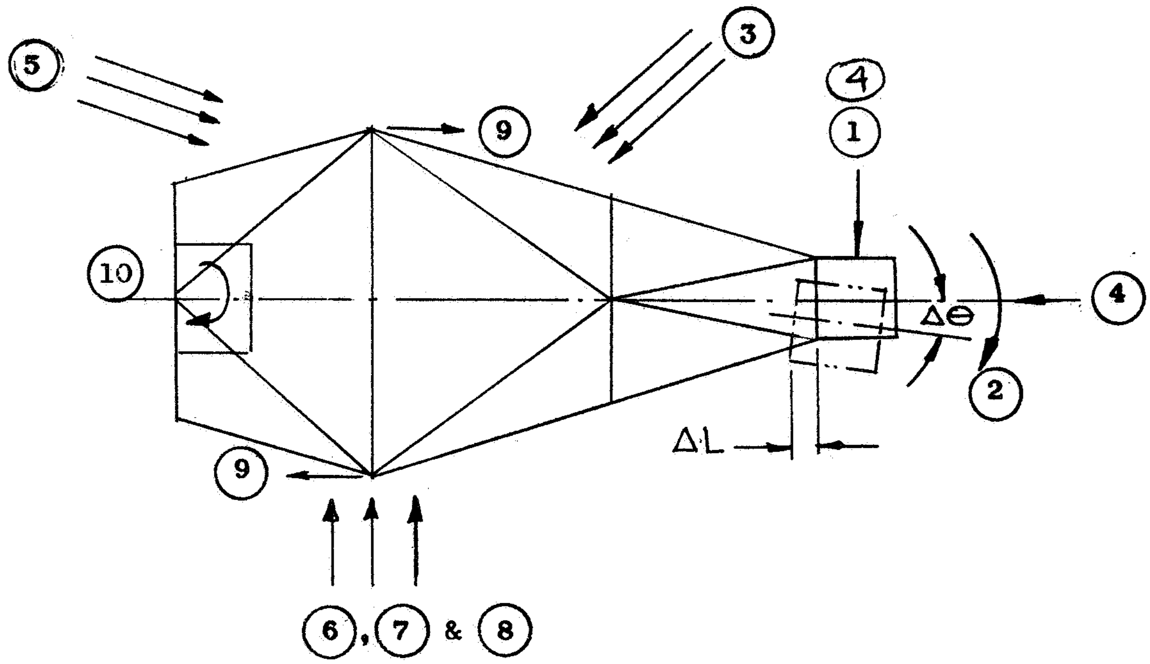
$$X \text{ (ft.)} = \frac{1}{12 \times 192} \frac{P^2L^3}{E1}, \tag{2}$$

$$MV^2 = \frac{1}{12 \times 192} \frac{P^2L^3}{E1}, \tag{3}$$

$$\text{and } P = V \sqrt{\frac{ME1}{L^3} \; 192 \times 12} \tag{4}$$

For fixed tubes, the maximum bending moment = 1/8 PL, (5)

$$Fy = \frac{my}{I} = \frac{PL}{8\pi R^2 t} \tag{6}$$



Note: Structure 1st mode bending frequency ≈ 20 - 25 cps.
Based on 40" lens size, octagonal truss geometry.
Tube joints pinned or fixed, as specified.

Condition	Load	Configuration	Material	ΔL Inches	ΔΘ Arc-secs.
1	400 lb.	Open truss-fixed	Beryllium tube 2. 5" dia. . 020 wall	---	103
2	400in. lb.	Open truss-fixed	"	---	. 052
3	45^{o} sun	Open truss-fixed	"	---	1. 9
4	400 lb.	Open truss-fixed	"	---	210
5	30^{o} sun	Open truss-pin	"	. 23	54
6	90^{o} sun	Closed conical shell structure	Alum-polished skins	. 236	156
7	"	"	" - white paint	. 756	500
8	"	"	Beryllium - white paint	. 415	276
9	ACS jets	Open truss-fixed	Beryllium tube	---	. 371
10	ACS inertia wheels	Open truss-fixed	Beryllium tube	---	3.12×10^{-3}

Figure 4-11. Structure Distortion Summary

Combining (4) and (6),

The critical $MV^2 = \dfrac{.087\ Fy^2\ RtL}{E}$ (7)

For the 2.5" beryllium tubes 75" long $\dfrac{250}{32.2} = 7.75$

The maximum $MV = \dfrac{.087\ (50000)^2\ 1.25 \times .020 \times 75}{42 \times 10^6}$

The critical velocity for beryllium = $\left(\dfrac{.094 \times 10^2}{7.75}\right)^{\frac{1}{2}} = 1.1$ ft/sec.

For aluminum the corresponding velocity is $1.1 \times \dfrac{700000^2}{500000^2} \times \dfrac{42}{10} = 9$ ft/sec.

The weight ratio of any two materials relative to their impact resistance is:

$$\frac{Wt_1}{Wt_2} = \frac{K\rho_1\ E_1 Fy_2^{\,2}}{\rho_2\ E_2 Fy_1^{\,2}}$$

Beryllium is $\dfrac{.067 \times 42 \times 10^6}{.100 \times 10 \times 10^6}\ \dfrac{\times\ 70000^2}{50000^2} = 5.5$ times as heavy as aluminum designed to the same energy. The foregoing indicates that beryllium is a poor choice of material if impacts are expected. It is not known, and would require at least neutral buoyancy simulation to determine, if the 1.1 fps impact from a 250 lb. astronaut is realistic.

The inverse of the above can be considered a value of the impact resistance of equal weight structure $\left(\dfrac{\sigma y^2}{E\rho}\right)$. Based on this parameter, the material with the highest impact resistance is titanium (5600), aluminum (4900) and beryllium (890). This criteria shows beryllium to be a poor choice for long slender members subject to impact.

The material property of thermal distortion resistance is by far the more significant design factor. Two aspects of this problem were analyzed. First, the analysis considered a fixed octagonal geometry structure model and the distortions resulting from circumferential thermal gradients. End moments required to maintain zero slope as the tube bends were applied to the total 3 dimensional truss and total structure distortions calculated.

This condition is not the critical aspect and the distortion produced was approximately .19 arc sec. The most significant truss thermal distortions were calculated using the quad truss geometry with a 3-dimensional pin jointed model. The thermal bowing effect of the tubes was ignored and only the change in length ($\alpha\Delta TL$) considered. This analysis is conservative because of the pin joint assumption; however, the displacement is at least one order of magnitude too high. Since this condition is a direct result of thermal length changes caused by uneven heating of the unsymmetrical truss, the material choice should be based on minimum thermal expansion coefficient. Of the prime spacecraft structural materials titanium has the lowest α; 4.8 x 10^{-6}, unfortunately, however, even titanium only reduces the thermal distortion 8 to 130 arc sec. which is still 10 times greater than permissible without a lens alignment system. This assumes that the total distortion is directly proportional to the $\alpha \Delta TL$ length change. Owing to the complexity of the truss, this may not be exactly correct.

This further assumes that the average absolute temperature of the titanium members would be the same as the beryllium members analyzed. This assumption is correct for tube circumferential differential temperatures $\approx 50^{\circ}$F with identical surface coatings.

A search for other possible structural tube materials with expansion coefficients approximately 0.5 x 10^{-6} (0.1 of titanium) yielded Invar, low expansion nickel, and tungsten alloys. These alloys are heavy, expensive, difficult to produce, and are very poor spring materials at reduced temperatures. Non-metallic structural materials like the glass reinforced plastics offer no improvement as these composites have higher expansion coefficients than titanium.

The penalty for using titanium in place of beryllium on the total satellite weight is not significant. Specifically, the total beryllium tube weight in the octagonal configured truss is 183 lb. Replacing the titanium of equal dimensions would produce a total tube weight of 436 lb. (253 lb. increase). The corresponding weight penalty for Invar is 587 lb. Based on the structural stiffness requirements for observational distortions and astronaut impact, the 2.5 in. tube diameter can be decreased. Tube diameters down to 1.5 in. offer improved space-suited astronaut handholds; furthermore, the reduction from 25 to 1.5 would reduce the tube circumferential temperature differential by a factor of 3.35 (the ratio of diameters squared).

In summary, titanium and possibly Invar appear to be the best material choices for a truss type telescope structure. The associated weight penalties of these higher density materials can be minimized by reducing the tube diameters since both titanium and Invar have higher impact resistance than beryllium, based on equal tube dimensions.

Titanium was selected for use in the baseline weight, cost and manufacturing analyses of the study instead of Invar, because insufficient time was available for a complete analysis of the effects of a highly magnetic material for the structure; possible interference with instrumentation, increased orbital magnetic torques, etc.

Another material possibility may be an adaptation of the composite wire mesh currently being developed by Convair for NASA for use in storable tubular extendable booms. The circumferential wire material is high conductivity beryllium copper; the longitudinal wires are low expansion Elgiloy. These mesh tubes have proven significant reduction in tube ΔT's. High tube ΔT's are not desirable, but none of the analysis data indicates that bowing contributes significantly to distortions of the optical axis in the fixed end truss configuration studied.

Detailed comparative studies of tube materials should include mesh, composites, and possibly non-circular cross-sections as it may be possible to minimize the distortions sufficiently to eliminate the lens alignment system.

4.1.3.3 Lens Group Assembly. The X-Ray Telescope lens group assembly is separable from the truss assembly at one of the gimbal axes, and consists of the following subassemblies: (See Figure 4-12)

a. Lens Shroud Assembly
   Sun Filter Assembly

b. Gimbal Ring Assembly

c. Sun Shade Assembly

d. Reflecting Lens Assembly
   Optical telescope
   Alignment System Reflector Pkg.

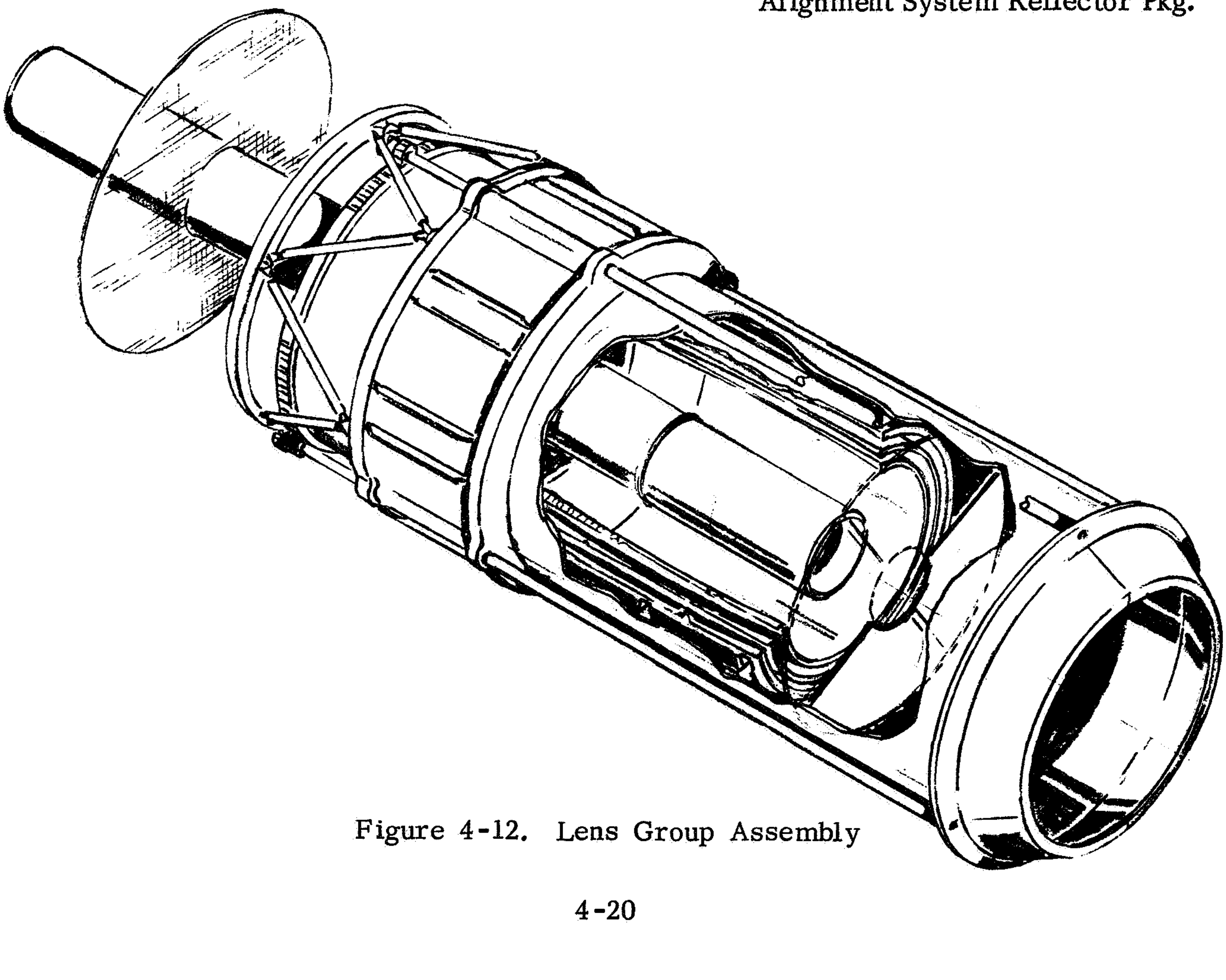


Figure 4-12. Lens Group Assembly

Lens Shroud Assembly - The mirror assembly is a 45 in. diameter cylinder of monocoque construction and is 46 inches long. The reflecting lens assembly is directly suspended within this cylinder. The shroud assembly contains trunnion points which serve as one axis of the two axis gimbal system. During launch, the aft end of this ass y is held by four pyrotechnic bolts to a support ring which is suspended from the lens support truss assembly. Thus, the weight of the reflector lens assembly, telescope and associated equipment are not borne by the gimbals during the boost phase. Each end of the focal length adjust screw is mounted with Belleville washers to take thrust loads and further reduce dynamic boost loads from being transferred through the gimbals.

A four segment sun filter is mounted on the forward end of the shroud assembly. The filter element consists of 1/4 mil thick beryllium sheet and covers only the annular lens entry slit. Each of the four filter elements is retained along the full length of its outside edge by a reinforcing u-shaped frame. A flat sheet metal beryllium copper spring constitutes the hinge. At launch the filter covers the lens slit being held in place by the sun shade conical frame. Upon deploying the sun shade, the filter segments will erect. Actuation of the filter is by a small motor operating a cable drum. The cable actuates cams, depressing the spring hinges and thus covering the lens entry slits. The design is such that in event of failure in the mechanism, the filters will tend to return to the open position. The cable is routed through a pyrotechnic cable cutter which may be fired to release all filters for failures which may occur in the motor or gear box while in the filter position.

Gimbal Ring Assembly - The lens shroud assembly is suspended within the gimbal ring. The gimbal ring is in turn suspended on the other orthogonal axis by trunnion mount to a traveling support ring, located in the structural end cylinder of the lens support truss assembly. The support ring is moved longitudinally ± five inches from the theoretical prime focal distance by four drive screws located equidistant around the ring. Recirculating ball nuts are integral with the gimbal support rings. The drive screws are driven by gear head motors at two diametrically opposite screws, (see Figure 4-13). The other two screws are capable of taking a portable plug-in power tool to permit operation by the astronaut. The gear head drive motors are easily accessible and replaceable by the EV crew. It is presently assumed that only an initial setting of the focal length will be required.

The gimbal motors constitute an area of design requiring considerable investigation. Since design of this system is fairly active, no attempt has been made to optimize it. The primary complication arises in making this installation serviceable by the crew. Most gimbal angle corrections will be at angles on the order of arc-sec. Stepping motors in conjunction with reduction units of negligible back-lash, such as the Harmonic drive, offer precise control; however, this installation has problems due to location of the controller required for the stepping motor and its need for controlled temperatures. The gimbal bearings will likely require a lifetime equal to the lifetime of the satellite, as they would be extremely difficult to replace.

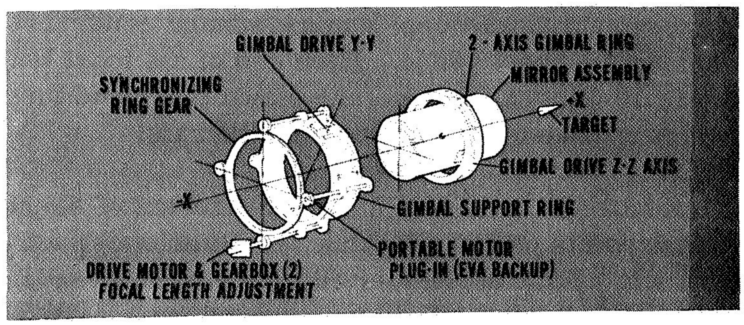


Figure 4-13. Lens Adjustment Mechanism

Distortions of the structure from any extraneous source result in the focal point wandering (relative to image capsule). The image receiving equipment has a limited motion compensating capability, not sufficient for the gross structural deflections. The gimbal system of the lens is coupled to an error detecting laser system to correct for the structural distortions. This alignment system, once activated, is continuous and automatic. The laser beam source is mounted on the image capsule while the beam is split and reflected by two mirrors on the lens assembly to the error sensors on the image capsule.

Lens Sun Shade Assembly - The third primary assembly in the x-ray lens package is the shroud assembly. The function of the lens shade is to prevent off-axis solar radiation from impinging on the exposed mirror elements. The lens assembly is screened from the target direction by a collapsible sun shade. This shade is extended by four parallel tubes operating in unison as the lens capsule truss assembly extends. Cable actuated drums rotate cog wheels which drive the tubes as shown by Figure 4-14. Snap locks secure the tubes at full travel. The shade is of a woven, impregnated fabric and has only the single extension cycle. The geometry prevents direct solar heating of the mirror forward edges when the telescope axis is pointed 40^{o} away from the sun. This angle is arbitrarily selected since no detailed target to sun angles have been calculated on the basis of an over-all observational mission profile. If it is shown that the 40^{o} cone about the sun is of interest, the shade tube can be lengthened to reduce the angle to at least 20^{o} without requiring unusual design concepts. Similar solar shielding is required to protect the aft end of the mirror. Since adequate structure exists to attach the rear shield it was not considered a problem and not investigated or shown on the drawings. The telescope is designed for direct solar observations using the 1/4 mil beryllium filter which is opaque to the IR heating, but relatively transparent to the solar x-radiation of interest.

The filter is not sufficiently transparent to the low energy stellar x-ray source radiation and therefore, it is presently assumed that stellar observations can not be conducted within the 40° half angle cone around the sun. Although detailed data has not been generated, the anticipated number of sources outside this cone will be large enough, so that the observational program can easily be sequenced to avoid observations in this area of thermal uncertainty.

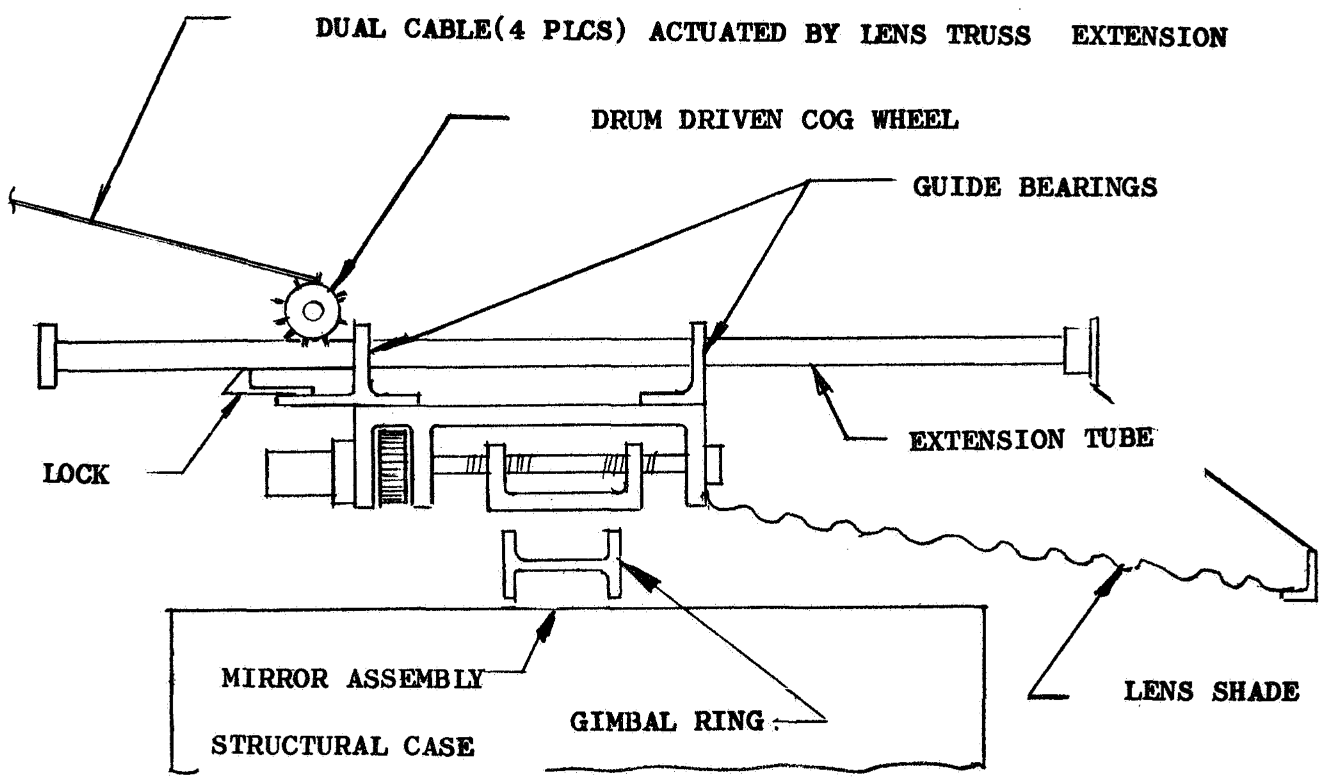


Figure 4-14. Lens Shade Extension Mechanism

Reflecting Lens Assembly - The principal element in a focusing x-ray telescope is the lens system, and focusing x-ray energy in the short (2 to 300 Å) wavelength spectrum to extremely high resolution (1-2 arc sec.) required precise tolerance control during manufacturing and control of the lens while in orbit. The lens concept consists of single or multiple confocal parabolic and hyperbolic surfaces which are designed to turn the paraxial x-rays into a focal point.

The theory and design of the mirror surfaces is discussed in detail in Reference 5 and will not be covered here except as applicable. American Science and Engineering, Inc., provided Convair with the general mirror surface tolerances as shown below to obtain the image resolution desired by the scientific community.

a. Optical surface finish - 1.0 micro in.

b. Out of roundness - 5 micro in.

c. Concentricity between nested segments .0015 (lens resolution x focal length).

d. Maximum deviation from theoretical contour $\pm$ .715 micro in. per in. of figure length.

e. Radius dimension $\pm$.001 in.

f. Focal length matching between segments $\pm$ 1% of focal length.

g. Angular misalignment between segments $\pm$ 30' arc.

Perkin Elmer Co. and Diffraction Limited Co. are currently fabricating 13 in. and 9 in. diameter grazing incidence mirrors for ATM. The tolerances for small mirrors are at least equal to those required of the larger lenses, and some dimensions such as concentricity, theoretical contour deviation and focal length matching are more precise as they are functions of size.

Convair has discussed the problems of fabricating 20 - 40 in. diameter lenses with American Science and Engineering and Perkin Elmer Co. and the conclusion is made that the larger lenses, possibly up to 5 feet diameter are feasible and, aside from the obvious increase in figuring and polishing time and cost, can be fabricated to the necessary dimensions with state-of-the-art technology. The problems of orbit distortions are covered in detail in Section 4.1.3.3 and 5.2.2. Figure 4-15 shows the basic components of the reflecting lens assembly which was used as a baseline for the performance of this study.

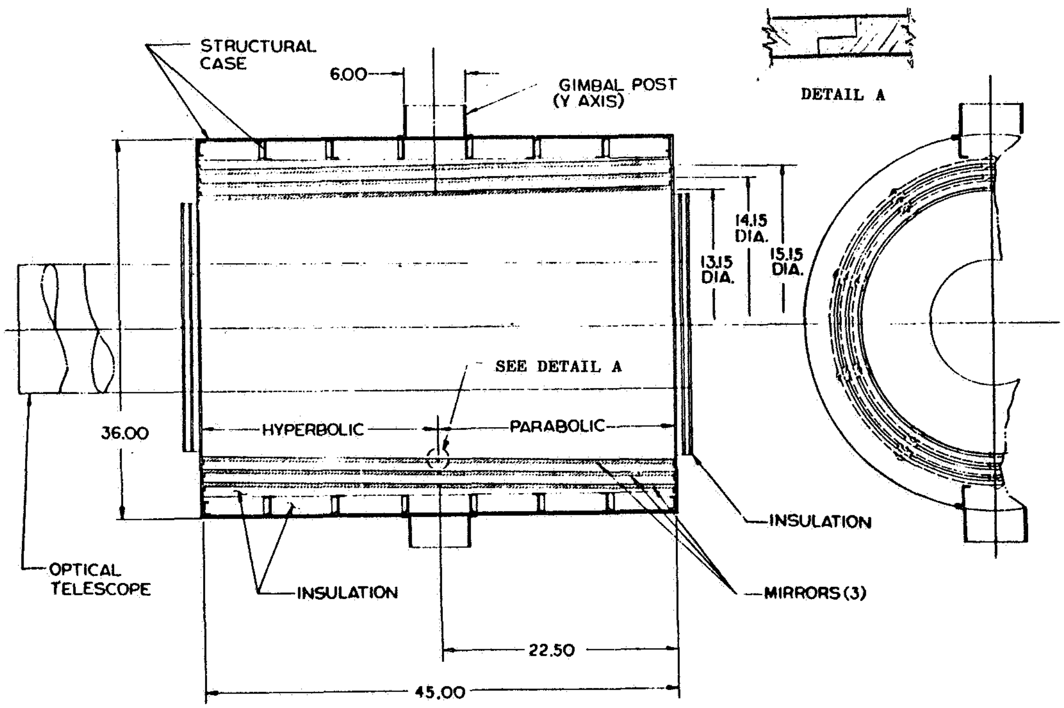


Figure 4-15. Reflecting Lens Assembly

To determine the effect of the specified lens size range on the over-all configuration, several parametric studies were conducted. When varying lens diameters, it is necessary to vary the focal length in proportion to maintain constant grazing angles and lens reflection efficiency. The effect of incidence angle on reflection efficiency is shown in Table 4-2 assuming constant focal length of 375 in.

Table 4-2. Single Reflection Efficiency at 2Å

Diameter (In.)	Grazing Angle (Arc Min.)	Reflection Efficiency (Nickel)
20	23	92%
30	34	40%
40	45	10%

The optical geometry of double reflection grazing incidence lenses which is shown in Figure 3-3, Section 3.5.1, is defined in Reference 5.

Figure 4-16 shows the effective collecting area as a function of focal length to diameter ratio for a 30 in. mirror. Figure 4-17 shows effect of F/D on double reflection efficiency.

With a single lens, there is a long spread between the optimum F/D ratio for 2Å and the optimum for 4Å. The nested multiple mirror configurations compensate for this, however, since the inside segments are progressively smaller, but the focal length is a constant. If the outer mirror focal length and diameter are sized for maximum collecting area at 4Å, an inside segment with a diameter 45% of the outer mirror would have optimum geometry for the 2Å wavelength.

To simplify the evaluation of lens sizes from 20 in. to 40 in. for 1 to 3 segments, the following assumptions and approximations are made: Refer to Figure 3-3.

a. Lens Radius - Y_o to thickness ratio = 30:1, this is considerably lighter than convention optical practice of 6:1, however, the low dynamic loading and zero g environment justifies drastic stiffness reductions.

b. Optical surface area $= 2\pi Y_o (2 Y_o)$ segment.

c. Minimum spacing between adjacent segments tg = t_{shell} = ts.

Y_{o1} = outer segment optical radius.

Y_{o2} = 2nd segment optical radius = $Y_{o1} - 2t_s$

Y_{o3} = 3rd segment optical = $Y_{o1} - 4t_s$

d. Reference 5 single segment collecting

$$\text{Area} = \frac{\pi Y_o^2 L}{2F} \qquad \text{if } L = Y_o \text{ and } F = \text{Focal Length}$$

$$\text{Rearranging, area} = \frac{\pi Y_o^2}{4\left(\frac{F}{D}\right)} \quad \text{where } D = 2 Y_o$$

Therefore, for F/D = 10, the collecting area = $.0785\ Y_o^2$ for one segment assuming constant focal length for the three outer segments.

The 2 segment collecting area $= \frac{\pi}{40} (Y_{o1}^2 + Y_{o2}^2) = .147\ Y_o^2$.

Similarly three segment area = $.205\ Y_o^2$.

The optical surface area $A_o = 4\pi Y_o^2$ (1 Seg). Area two segment $= 4\pi Y_o [Y_o + .933\ Y_o]_2 = 24.2\ Y_o^2$. Area three segment $= 4\pi Y_o [Y_o + .933\ Y_o + .866]$ equals $34.8\ Y_o^2$.

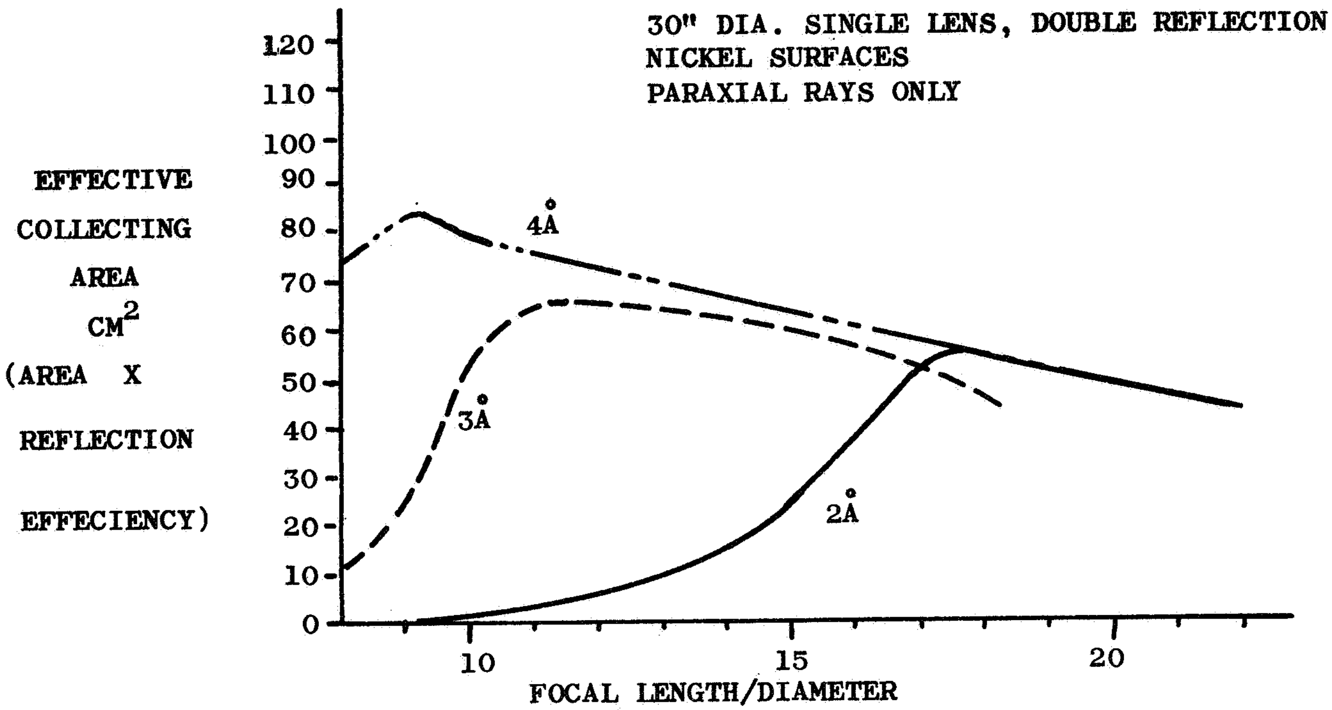


Figure 4-16. Theoretical Effective Lens Area

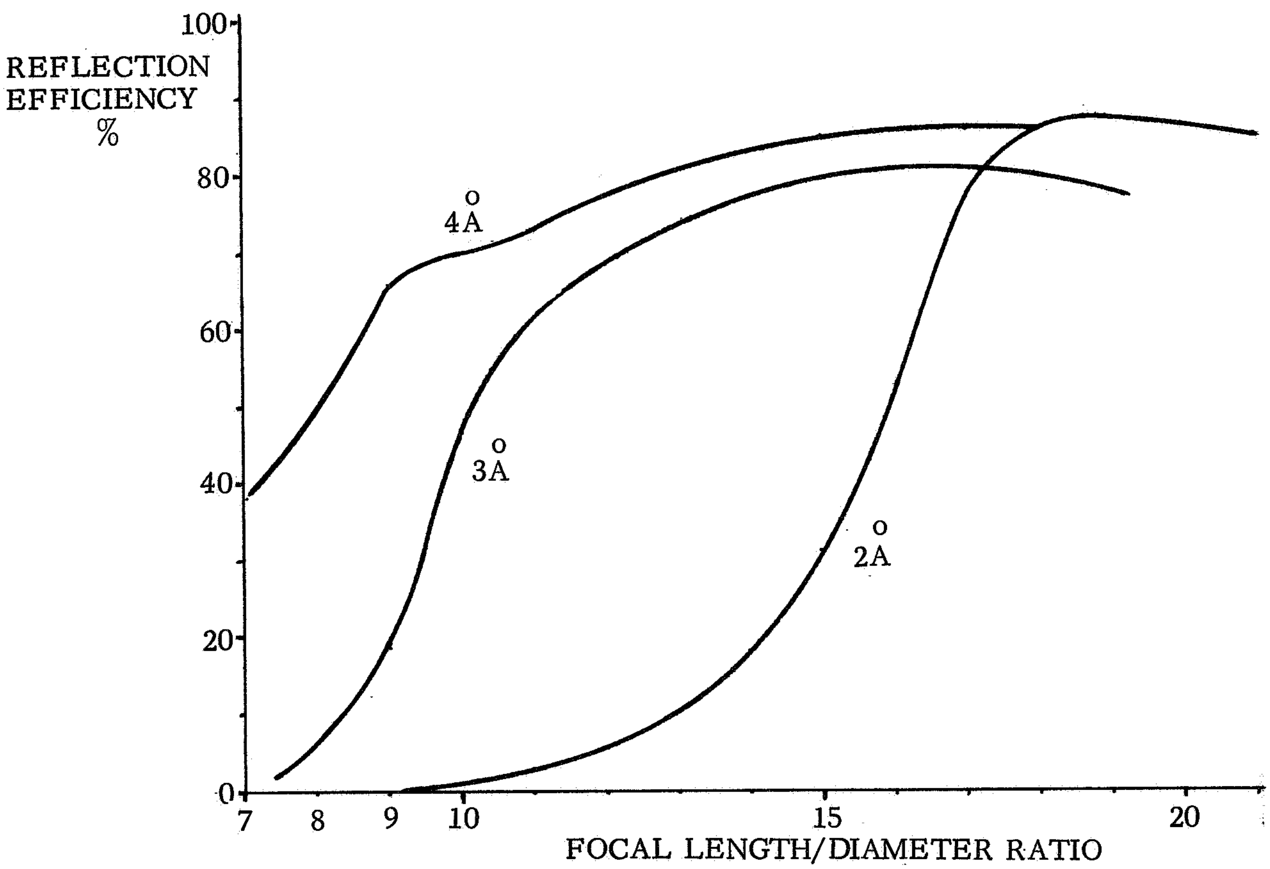


Figure 4-17. Theoretical Double Reflection Efficiency

The results obtained for the various lenses are shown in Table 4-3.

Table 4-3. Lens Characteristics - 20 To 40 Inch

Dia. In.	Shell t~in.	No. of Segments	Frontal Collecting Area $in.^2$	cm^2	Optical Surface Area $in.^2$	$ft.^2$	Lens Weight lb. Lockalloy
20	.25	1	7.85	50.5	1256	8.7	23.6
	.25	2	14.7	95	2420	16.8	45.5
	.25	3	20.5	130	3480	24.2	65.3
30	.375	1	17.7	114	2820	19.6	79.5
	.375	2	33.0	213	5450	37.8	154
	.375	3	46.0	297	7750	54.0	219
40	.50	1	31.4	202	5030	34.9	189
	.50	2	58.6	378	9700	67.5	364
	.50	3	82.0	529	13800	95.5	517

A first iteration of parametric cost vs. size was performed based on the following additional assumptions:

a. Blank forming (form and machining to .005 oversize)
   Material cost ~ \$200/lb.
   Forming & Machining ~ \$50/ft.^{2}

b. Kanigen coating deposition ~ \$100/ft.^{2}

c. Polishing cost based on estimates of 3 in. diameter AS & E and 13 in. ATM
   $\text{cost} = \$10,000 \left[Ao_1^{1.3} + Ao_2^{1.3} + Ao_3^{1.3} \right] = (\$10,000 \times \text{optical surface area}^{1.3})$

d. Assembly cost - 1 segment has no close tolerance mounting
   2 segment - assume 2 man months @ \$10/hr. = \$3,380.
   3 segments - 6 man months = \$10,080.

Although these assumptions are very gross, they nevertheless are believed to yield basic trends. A summary of the cost derived from the foregoing assumptions are shown in Table 4-4.

Table 4-4. Approximate Grazing Lens Unit Costs

		Thousand of Dollars				
Dia.	No. of Segments	Blank Forming	Machine + Coating	Polishing	Mounting	Total
20	1	4.72	1.31	167.5		173.53
	2	9.10	2.52	319.5	3.36	334.48
	3	13.06	3.64	454.5	10.08	481.28
30	1	15.90	2.92	480.0		498.85
	2	30.80	5.67	918.0	3.36	957.83
	3	43.80	8.10	1293.0	10.08	1354.98
40	1	37.80	5.24	1000.0		1043.04
	2	72.80	10.10	1920.0	3.36	2006.26
	3	103.40	14.30	2680.0	10.08	2807.78

The lens cost estimates were plotted against lens area to illustrate the effect of size on cost. In Figure 4-18, the lens performance (collecting area) compared against lens cost only shows a small difference, (in dollars per cm^2 of aperture).

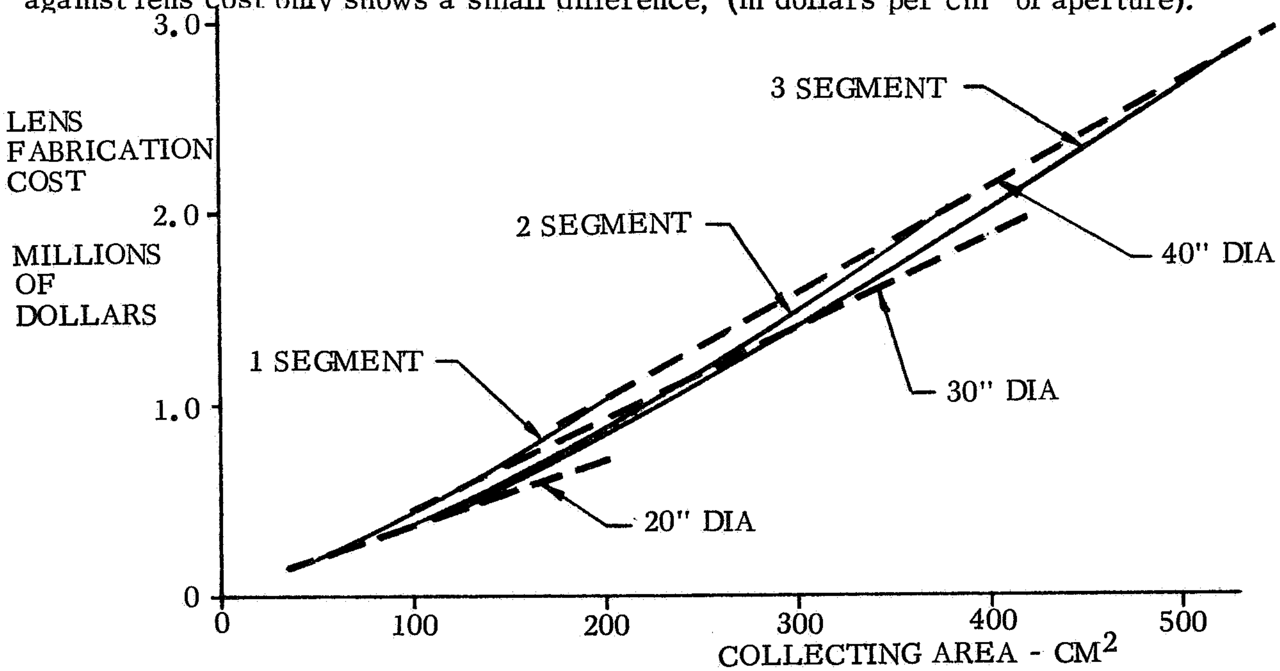


Figure 4-18. Lens Cost

Gross estimates of total system unit hardware costs were made. These are listed in Table 4-5. It is based on the assumption that the structure is not changed by number of segments and that the actual size range is small enough not to affect the major systems.

Table 4-5. Unit Hardware Cost Estimate - Thousand Dollars

System	Lens Diameter (in.) 20	30	40
Basic Structure	408	464	520
Attitude Control & Guidance	1340	1340	1340
Command Data and TLM	150	150	150
Electrical Power	380	380	380
Scientific Instrument	800	800	800
Total	3078	3134	3190

The above totals were added to the lens cost from Table 4-4, and divided by the respective collecting areas to show cost per cm^2 of area. See Figure 4-19. These results show that lens area can be obtained at minimum unit cost in the larger sizes. The trend of reduced unit collecting area cost would be expected to continue only until the lens costs becomes high with respect to the total. It appears that this total relative cost becomes nearly constant with multisegment lenses over 50 in. dia. This is based on the simple assumptions used, since it is not the intent of this study to rate the scientific value of the larger more sensitive instruments.

Subsequent cost analyses indicate that the 30 in. - 3 segment mirror estimate (\$1. 35 million) is approximately 70% of the later value (\$2. 33 million) and the additional costs estimated for the 30 in. (\$3. 134 million) are also approximately 70% of the latest value (\$4. 53 million).

This validates the basic estimate, however, there are approximately \$3. 09 million additional recurring costs per article which were not included originally (assembly, integration, sustaining engineering, etc.). This makes the lens cost a less significant portion of the cost per cm^2 and effectively drives the curve out beyond the 50 in. lenses. That is, the unit cost of the total system per cm^2 of collecting area will be minimum with multisegment lenses possibly as large as 60 in. diameter.

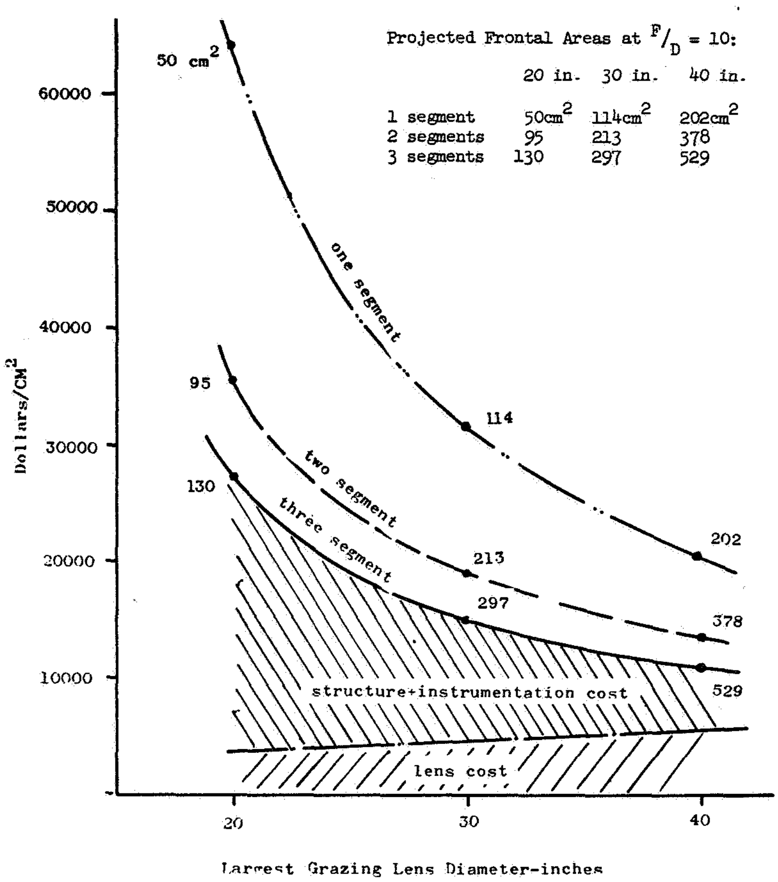


Figure 4-19. Relative Unit Hardware Cost

Mirror Materials - The evaluation of various lens materials for the x-ray telescope has been conducted primarily from a feasibility aspect, i.e., what materials will be suitable in these size lenses. Material optimization would require more specific mission definition to ascertain which wave length region is of primary interest since in x-ray optics the reflection efficiency is a function of the material properties. Current x-ray lenses are being manufactured of fused silica, beryllium, 440 stainless steel, electro formed nickel, and aluminum. Surface coating techniques can be used to add a particular material which has desired reflection characteristics and ease of polishing and figuring. One attractive coating presently used is kanogen, an amorphous metal coating of nickel-nickel-phosphorous composition that can be applied to a wide range of base materials at low temperatures.

The Speedring Company is optimistic that development of a beryllium blank for a large mirror - up to 60 in. internal diameter could be done in about two years. They indicated that tooling is the biggest problem. Perkin Elmer Company has also expressed confidence that the 60 in. mirror with a kanogen coating, could be available in two years.

Fused silica is an attractive material choice for many optical components primarily because of its low coefficient of expansion (6×10^{-7} per ^{o}C). (Reference 10) Fused silica has been proposed to MSFC for the 13 in. ATM x-ray telescope.

If a kanogen coating is used, the beryllium or base metal blanks must be formed within a few mils of the geometrical figure since the coating is only 5 mils thick. While the basic dimensional requirements of the lens are believed completely feasible before 1970, in sizes up to 5 ft. diameter, the desired resolution of 1 arc-sec. is considered to be beyond the present state-of-the-art. Perkin Elmer Company is guaranteeing 5 arc sec. and working toward 2 arc sec. on a "best effort basis" for the 9 in. ID ATM mirror. Neglecting cost and manufacturing problems, some of the material properties of beryllium are particularly attractive. Table 4-6 lists the properties of typical lens materials.

For purposes of material evaluation, some simplifying assumptions were made relative to geometry. First, the mirrors may be considered cylindrical as the surfaces converge at half angles less than 1^{o}. Secondly, thin wall shell equations may be applied as the mirror diameter to thickness ratios considered are greater than 10:1. The dynamic response of the various materials is best shown by the stiffness to weight parameter E/ρ. A conservative example of mirror distortions is shown in Section 5. 1. 4. 1. The analysis assumed a 30 in. diameter mirror of 0. 18 in. thickness, the actual minimum thickness considered for design was 0. 38 in. The ratio of the point loaded to uniform loaded coefficients for ring deflection is shown in Reference 13 as $\frac{.07439}{.01153} = 6.48$. It is concluded that the lens, like the truss, is unaffected by the dynamic control forces during observation, and that the dynamic distortions of the mirror will be within tolerance regardless of the material.

Material	Density ρ (Lb/In^3)	Yield Stress (Psi)	Modulus E (10^6 Psi)	Thermal Conductivity K (Btu/Hr/Ft^2/°F/Ft)	Coefficient of Expansion $\alpha \times 10^6$ (In/In/°F)	$K/\alpha \times 10^{-6}$	Specific Stiffness $E/\rho \times 10^{-6}$	$K/\alpha \times E/\rho \times 10^{-12}$	Equal Weight Impact Coefficient $Fy^2/E/\rho$
Beryllium	.067	50000	42.0	87.0	6.4	13.6	627	8520	890
Lockaloy	.076	40000	29.0	123.0	9.2	13.4	382	5100	735
Fused Silica	.079	1000	10.5	.77	0.30	2.56	133	340	1.2
7075 AL	.101	70000	10.3	90	12.9	6.9	102	725	4750
Nickel	.321	20000	30.0	34.4	7.4	4.65	93.5	435	40
440 CRES	.280	60000	29.0	14	5.6	2.5	104	260	445

Table 4-6. Mirror Candidate Materials

Eliminating the dynamic problems leads to the conclusion that the principal lens material property selection criterion should be thermal stability. The four most critical lens tolerances are:

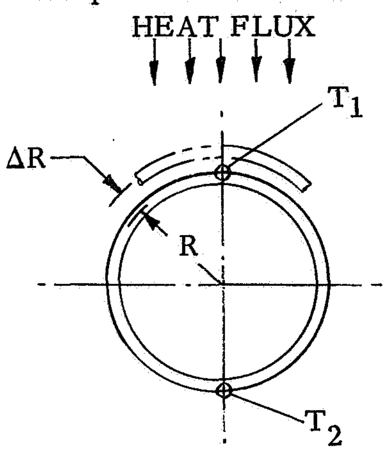
a. Radius $\pm$ 0.001 in.

b. Out of roundness 0.5 microinches.

c. Concentricity = 0.0015 in.

d. Longitudinal contour = 5/7 microinch/in.

The radius growth or shrinkage can be compensated for in manufacturing if the flight temperature is known.



$\Delta R = \alpha \Delta TR$ Uniform heating

ΔR max. = $1/2\alpha\Delta TR$ with linear uniform distribution

On 15 in. radius, the increase in T which will cause a 0.001 in. change in radius

$$\frac{.001}{6.4 \times 10^{-6} \times 15} = 10.45^{\circ}.$$

If the segment is heated unsymmetrically, the total circumferential growth is only 1/2, and therefore, requires a ΔT of 20° to effect a 0.001 radial growth. This change is large and can probably be reduced considerably. The effect of ΔT on out of roundness and concentricity is probably far more serious.

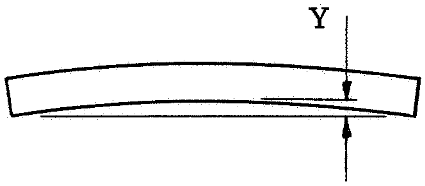
As shown previously, the temperature across the lens (from an infinite point radiation source) = $\Delta T = K \dfrac{D^2}{tk}$. (Reference 11)

and longitudinal displacements = $Y = \dfrac{\alpha \Delta T L^2}{D4}$. (Reference 12)

For the 30 in. diameter, 45 in. long, 1/4 in. thick geometry, the equations become:

$$\Delta T = \frac{(30)^2}{.25K} \quad \frac{(450 \times .05 \times 1)}{(96)}$$

$$\Delta T = \frac{844}{K}$$



$$Y = \alpha \Delta T \quad \frac{(45^2)}{(4 \times 30)} = 16.8\ \alpha \Delta T$$

Or

$$Y = 16.8 \times 844 \quad \frac{(\alpha)}{(K)} = 14150 \quad \frac{\alpha}{K}$$

Considering the maximum deviation from theoretical contour tolerance of 5 microinch/7 in. length, assume half of the total tolerance is allowed for thermal tolerance equals 5/7 x 45 = 32.2 microinches/2 = 16.1 microinch. Under solar heating the materials would have the following deviations:

	ΔT^oF	Y(microinches)
Beryllium	$844/87 = 9.7^o$	1000
Fused Silica	$844/.77 = 1090^o$	5500
Aluminum	$844/90 = 9.35^o$	2000
Titanium	$844/3.8 = 222^o$	17000

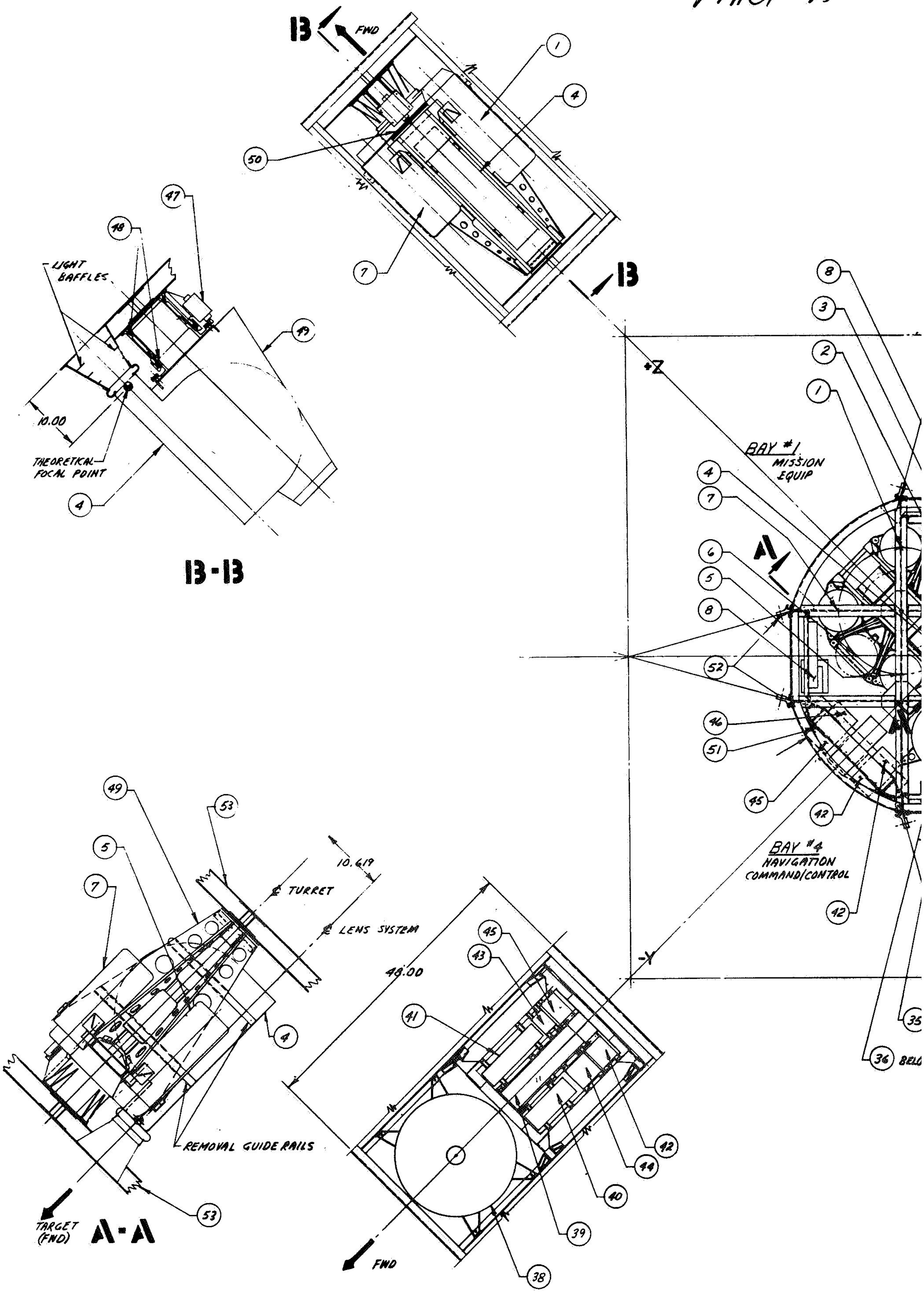
The foregoing distortions are obviously unsatisfactory. Beryllium the lowest, is some 1000/16.1 = 62 times greater than allowed; i. e., the maximum permissible for beryllium $\Delta T = \dfrac{16.1 \times 10^{-6}}{16.8 \times 6.4 \times 10^{-6}} = 0.15^o$.

For fused silica, this ΔT maximum becomes $0.15 \times \dfrac{6.4}{.3} = 3.2^o$.

This value is becoming reasonable, however, as shown above, the conductivity of fused silica is too low to permit distribution of heating due to solar radiation. The implication is clear that an active thermal control system may well be required, possibly using thermostat controlled louver doors similar to the image capsule system.

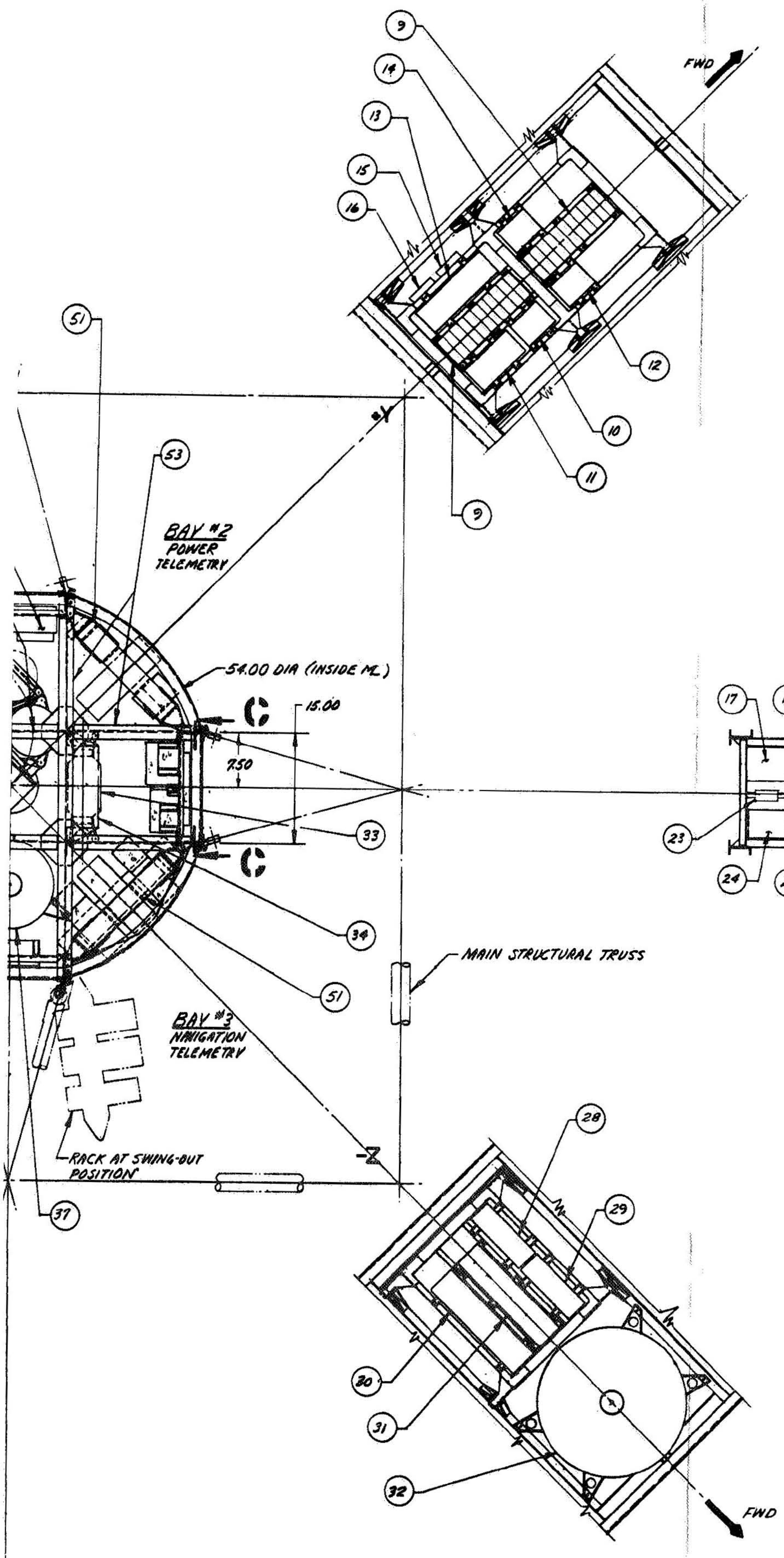
4.1.3.4 Image Capsule Assembly. The image capsule shown in Figure 4-20 is the prime reference base for the orbiting vehicle. It contains the mission scientific receiving equipment, the navigating equipment, power conditioning, command/control and telemetering equipment and docking provisions. The capsule is temperature controlled, but is not pressurized.

PART A



FOLDOUT FRAME 1

PART B

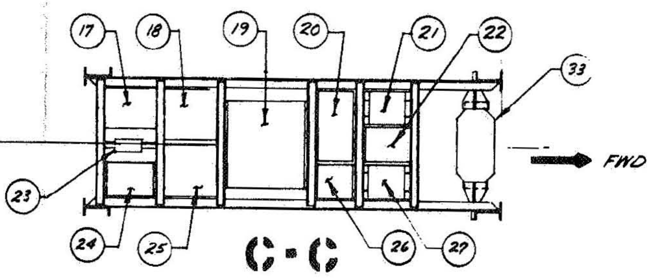


1. INSTRUMENT STATION #1 - IMAGE SIGNATURE - IMAGE INTENSIFIER, MOTION COMPENSATOR, ASTRACHON, ELECTRONICS.
2. INSTRUMENT STATION #2 - EMPTY, RESERVED FOR "ADD-ON".
3. INSTRUMENT STATION #3 - POLARIMETRY SLIT, H_2 DETECTOR WITH He DEWAR, 4 PHOTOMULTIPLIERS, ELECTRONICS.
4. INSTRUMENT STATION #4 - SPECTROMETRY SLIT, CRYSTAL SPECTROMETER, PHOTOMULTIPLIERS, ELECTRONICS.
5. INSTRUMENT STATION #5 - SAME AS #1
6. INSTRUMENT STATION #6 - EMPTY
7. INSTRUMENT STATION #7 - SAME AS #3
8. X-RAY EQUIPMENT ELECTRONICS. TURRET ACTUATOR CONTROLLER

9. BATTERY - 20 AMP-HR, 535 W-HR, 26.8v
10. BATTERY CHARGER
11. REGULATOR
12. VOLTAGE BOOSTER
13. MAIN POWER DISTRIBUTOR
14. AUXILIARY POWER DISTRIBUTOR
15. VOLTAGE LIMITER
16. LOW VOLTAGE SENSOR

17. TIME GENERATOR
18. DIGITAL LOGIC ASSEMBLER
19. TAPE RECORDER; SERIAL-TO-PARALLEL, PARALLEL TO SERIAL CONVERTER
20. TRANSMITTER & ANTENNA
21. ANALOG TO DIGITAL CONVERTER
22. PRE-MODULATION PROCESSOR
23. PRE-MODULATION FILTERS
24. SIGNAL CONDITIONERS; BUFFER STORAGE, PARALLEL GATES, ETC.
25. PCM MULTIPLIER
26. BUFFER STORAGE
27. INPUT SELECTOR

28. SENSOR SIGNAL PROCESSOR (PITCH & YAW)
29. SENSOR SIGNAL PROCESSOR (ROLL)
30. STAR TRACKER CONTROLLER
31. STAR TRACKER SIGNAL PROCESSOR



32. INERTIA WHEEL - YAW AXIS
33. INERTIAL PLATFORM
34. DIGITIZER LOGIC UNIT
35. FINE WHEEL & JET CONTROLLER (PITCH & YAW)
36. FINE WHEEL & JET CONTROLLER (ROLL)
37. INERTIA WHEEL - ROLL AXIS
38. INERTIA WHEEL - PITCH AXIS
39. COMPUTER
40. HIGH TORQUE CONTROLLER
41. PHASOLVER

42. COMMAND RECEIVERS
43. SIGNAL COMBINER
44. TONE DECODER
45. PCM DECODER
46. COMMAND PROGRAMMER

47. TURRET ACTUATOR
48. TURRET BEARINGS
49. TURRET SUPERSTRUCTURE
50. SPRING-LOADED, ANTI-BACK LASH SPLIT GEAR, SELF LUBE SIMITEX.
51. EQUIPMENT SWING-OUT RACK
52. TRUSS PICK-UP FITTINGS & DOOR TRACKS
53. BULKHEAD CROSS BEAMS

NOTE: IMAGE CAPSULE IS SHOWN WITH OUTER BULKHEAD SKINS OMITTED, INSULATION OMITTED, DOORS & TEMPERATURE CONTROL LOUVRES OMITTED AND EXTERNAL MOUNTED EQUIPMENT OMITTED. SEE SHT #2 FOR EXTERIOR

Figure 4-20. Image Capsule

Geometrical Arrangement - The basic capsule diameter is 54 in., the length is 48 in. Insulation is added outboard of these basic dimensions along with four full length access doors having integral temperature controlled louvers. (See Figure 4-21)

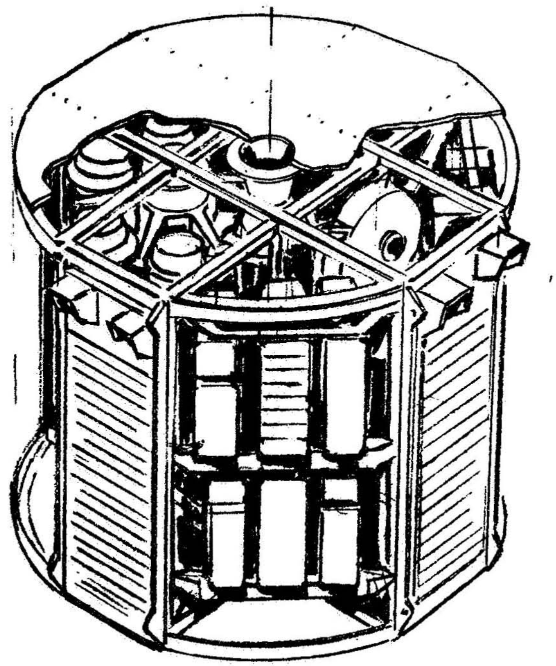


Figure 4-21. Image Capsule

The structural arrangement consists of cross and vertical beams, walls, and tension straps. Double cruciform beams, three in. deep and spread 15 in. apart from bulkheads at the forward and aft ends of the module. Side wall beams join the top and bottom beams with a shear web spanning the 15 in. separation of the vertical beams. The bulkheads have external metal webs. Diagonal tension straps interconnect the forward and aft bulkheads at three of the four apexes formed by the double cruciform. The fourth apex is the pivot axis for the x-ray receiving equipment turret. The primary construction material is 2024 aluminum alloy.

The tabular truss attaching the image capsule to the main truss assembly picks up the eight vertical beams at their juncture with the bulkhead beams. (See Figure 4-22)

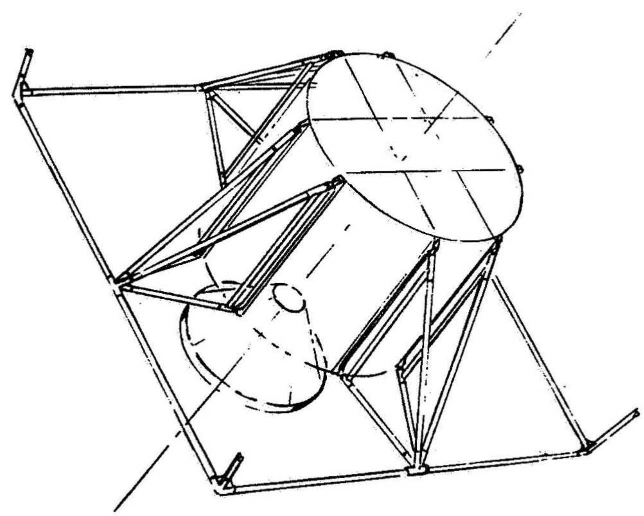


Figure 4-22. Support Truss

The double cruciform shape essentially creates an eight bay geometry. Four of these bays are flat sided at the 15 in. width between beams. The other four bays are on the 54 in. diameter and are enclosed with the access doors. All internal equipment and walls are accessible through the access doors by a space suited astronaut. The equipment mounting arrangement is designed especially to provide this accessibility. Most equipment is immediately inside the access doors on swing-out racks. Thus, both sides of the equipment are immediately at hand. The swingout feature permits access to additional equipment further inside and also allows approach to internal equipment from more than one access door. This design approach creates some difficulty in harness wiring, but offers considerable latitude in terms of servicing. The four access bays designated Bay I, Bay II, Bay III and Bay IV feature particular subsystem equipment. Much refinement is required to resolve the trade-offs of weight and balance, optimum operating characteristics, heat balance, service priorities and volumetric efficiency which are beyond the scope of this study.

Bay I contains the x-ray receiving instruments mounted on a turret. Related electronic equipment is mounted on two walls formed by the separated vertical beams. The turret is located at an apex of the cruciform beams and spans between the forward and aft bulkheads. During the boost phase and prior to operational checkout, the turret structure is secured to both bulkheads. Following the deployment phase, the turret boost restraints are disconnected from the aft bulkhead by the astronaut.

The turret contains seven instrument stations. Station 1 contains an image signature package consisting of the equipment discussed in Section 4.9. Station 2 is left empty for purposes of an "add-on" instrument at a later date by resupply. Station 3 is a polarization measuring instrument. The instrument of Station 4 is a crystal spectrometer. Station 5 duplicates 1, Station 6 is left empty and Station 7 duplicates Station 3.

Bay II is the power section. Most identifiable equipment, including the batteries are mounted on a swing out rack. Access to the wall-mounted electronic accessory equipment of Bay I is gained through this bay by swinging out the rack. Access through this bay can be made to the telemetry equipment mounted on the wall between Bays II and III, and to the inertial platform mounted on the forward bulkhead cross beams, and the navigation panel. The latter two items are located between Bays II and III. It may be advisable to remove the batteries from the swingout rack because of their weight and the associated heavy cabling that must be swung around the pivot. Available space exists on the cross beams of the forward bulkhead. This bay contains considerable unused volume.

Bay III primarily contains navigation equipment. The yaw inertia wheel (84 ft/lb/sec) is mounted in the forward portion of the bay and navigation equipment is located on the swingout rack. Access through this bay can be made to the wall-mounted telemetry equipment between Bays II and III. Also, accessible and located between Bays II and III, are the inertial platform and the navigation panel. The roll inertia wheel is located on the cross beams of the forward bulkhead between Bays III and IV and is accessible from Bays III and IV. Certain wall mounted navigation equipment is mounted between Bays III and IV and accessible from either bay.

Bay IV contains navigation and command/control equipment. The forward portion of the bay is spanned by an 84 ft/lb/sec pitch inertia wheel. The aft portion has the equipment mounted on a swingout rack. With the rack swung out, access is gained to other equipment between Bays III and IV as previously mentioned. In addition, access is gained to electronic equipment mounted on the perimeter wall between Bays IV and I.

Instrument Turret - The use of a rotating turret to place multiple instrument packages in the prime focal plane makes certain assumptions with regard to axial alignment accuracies. The proposed design utilizes several complete instrument packages; i.e., the imaging package, the spectroscopy package, etc. Each package will have the components located (on the ground) with respect to the other elements to the required tolerance. The instrument package is then attached to the turret which fixes the package location with respect to the prime focus. The only tolerance or variable inherent in the design is the axial turret bearing tolerance and this can be reduced to less than 0.010 in. by preloading the bearing against its forward reference plane. Some of the current x-ray lenses in production are being built to specified focal lengths within 0.25% of focal length, this value is 0.90 in. for the nominal telescope design. The lens assembly design contains provisions for adjusting focal length after deployment, however, it is not anticipated that the focal length would be changed after initial adjustment. The axial growth of the structure from thermal heating may change the focal length 0.5 in. Considering the previous variables, the small turret bearing tolerance is probably insignificant. The rotational positioning accuracy of the turret is one arc minute (46×10^{-6} revolutions) which is adequate for any package considered here.

The instrument turret rotates through 315 degrees. By limiting the rotation to less than 350 degrees, it is possible to wire all units to stationary mounted accessory equipment, thereby eliminating slip rings. Station 4 is at receiving position during launch. As other stations are indexed to the receiving position, the station opposite is placed at the access door for servicing or removal. The turret is not connected at the aft bulkhead so that any distortion of the structure has a minimum effect on the turret. It is driven by an actuator incorporating a special application of the basic harmonic drive principal by combining the functions of a prime mover (electric motor) with a speed reducer in a single compact and mechanically simple unit. In this special application, the wave generation is performed by a rotating magnetic field. As a result, the inertia, and wear of a high speed rotor are avoided. The prime mover is a stepping motor incorporating a stator assembly having salient poles that are sequentially energized in diametrically opposite pairs giving precise resolution. The position indication is derived from an encoder driven directly by the ring gear. The actuator and encoder are mounted on the stationary base of the turret and are immediately accessible at the access door. Attachments are provided for unit replacement by the astronaut. The actuator drives a 13. 5 in. pitch diameter spring-loaded, anti-backlash, split ring gear located just above the turret bearings and under the turret equipment mounting base. Due to the relatively large inertia of the turret and equipment, the actuator stepping motor must be limited to under 200 steps per second resulting in station to next station positioning time of 13. 5 seconds minimum.

The main rotation bearings of the turret are eight in. inside diameter. The current design is based on these bearings sharing the launch induced loads in conjuntion with the points of attachment of turret to the aft bulkhead. These bearings may be subject to brinelling, especially for metal balls. It may be necessary to completely relieve the launch loads from the bearings to insure long life since they are not replaceable. However, the present design considers non-metallic balls such as teflon in a composition. The resilient nature of such a bearing eliminates brinelling. Three bearings are used in the turret. Two bearings back-to-back (or double row) take all of the longitudinal thrust. All three bearings support radial loads. The double row bearings and the slip bearing are axial and separated by seven in. In the operating mode, any torsional eccentricities or distortion of the capsule structure will not impose loads onto the turret rotational mechanisms. Securing the bearings axially at the double row bearing removes the influence by manufacturing or assembly tolerances, hence will not have built-in stresses and is essentially a no-play combination. This will insure repeatability, smooth operation and long life for the turret installation.

The true image point is 10 in. inside the forward face of the forward bulkhead. A truncated cone structure is located along the 10 in. length. At its aft end, a teflon coated annulus-shaped spring seals the face of each instrument as it is rotated into receiving position. The imaging equipment volumes used in the image capsule design (base line) and the maximum available volumes are shown on Figure 4-23. The image capsule design may be modified to accommodate larger equipment packages.

Image Capsule Access Doors - These doors feature provisions to enhance the servicing of the module by the astronaut. The doors are not load carrying members of the primary structure. The door is single piece, molded fiberglass and extends approximately the full length of the capsule (48 in.). Cross bulkheads add stiffness. Temperature control louvers are mounted in the door for approximately three-quarters of its length. The beaded edges are teflon coated and slide in teflon

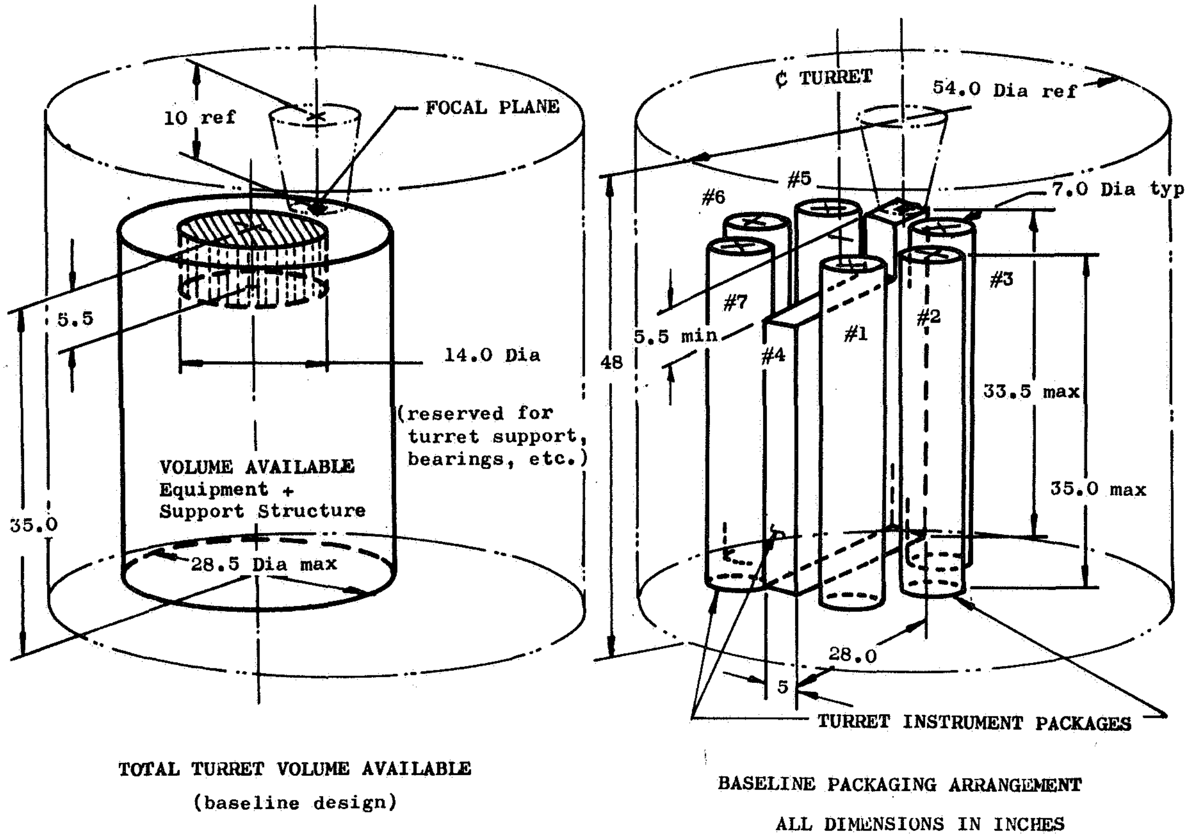


Figure 4-23. Instrumentation Turret Packaging Volumes

coated ways of the strut pick-up longerons. These longerons are fiberglass with local bosses for strut attachment at forward and aft ends and thermally isolate the image capsule from the truss structure. Imbedded metallic inserts (aluminum) locally reinforce the bosses at high stress areas.

Each door has foot stirrups at the forward end allowing the astronaut to mount the door. Lugs along each side receive a restraint harness. The aft right hand corner of each door has a plug-in receptacle for an astronaut portable rotary power tool. This is approximately at the waist level of the astronaut. The receptacle contains a reducer, torque limiting clutch, a 90° bevel gear drive, torque shaft and a capstan. The capstan is located on the interior of the door. Energizing the power tool causes the capstan to traverse along a cable stretched full length of the capsule. Stops prevent the door opening more than 37 in. The astronaut can position himself at any height, the portable tool serves as a brake. Hand latches at the aft end of the capsule secure the door in the closed position, but are primarily needed only during the boost phase. Manual opening may be possible if sufficient force is exerted to overcome the edge friction and the mechanical drag of the capstan and reducer.

The temperature control louvers are actuated by bimetallic devices to regulate the position of the louvers in response to internal heat loads, allowing radiation to space.

Insulation - The forward and aft bulkheads and the flat sections between the designated bays are covered externally with separated layers of aluminized mylar. Certain equipment such as the star trackers and antenna project through this insulation.

4.1.3.5 Docking Assembly. The x-ray telescope is equipped with provisions to dock with an Apollo CSM for the following conditions:

a. Initial stabilized telescope separation from the SLA.

b. Attachment to and stabilization by CSM for final orbit corrections.

c. (ΔV) if required, and for control and monitoring of telescope deployment by CSM crew.

d. Attachment to CSM for scheduled post deployment and operational checkout EVA tasks.

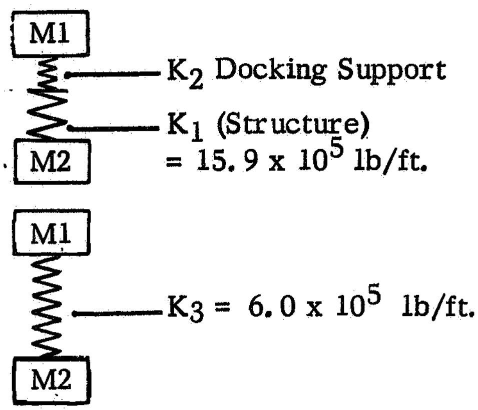
e. Attachment to CSM for scheduled resupply and refurbishment and any unscheduled repair or replacement.

The initial separation from the launch vehicle may be accomplished without docking by spring loaded or jet ejection of the stowed x-ray package; however, the satellite has no attitude control until the solar/ACS booms are deployed and unsymmetrical ejection loads might result in tumbling rates high enough to damage or jam the booms during deployment.

Requirements - The design requirements for the docking assembly are established by the initial docking on the packaged satellite, at which time the light weight x-ray structure is loaded by the momentum of the CSM and rigidly reacted by the 20,000 lb. spent S-IVB stage. The initial beryllium truss designs exhibited excessive stiffness and required that the LEM docking adaptor be soft mounted on the image capsule, (See Section 5.1.1). The axial compressive stiffness of the truss structure is reduced by a factor of 3.3 by using 2 in. titanium tubes in place of the 2.5 in. beryllium tubes (maintaining constant wall thickness). The titanium structure stiffness is then 15.9×10^5 lb./ft. This value is still more than 2.5 times as stiff as required to permit the CSM final velocity of 0.15 ft/sec defined in the docking specification without over stressing the structure.

Structural/Mechanical - The necessary reduction in stiffness for docking is achieved - by incorporating spring loaded, friction damping snubbers into the 8 struts supporting the docking cone on the image capsule. See Figure 4-24.

The strut stiffness model is shown below:



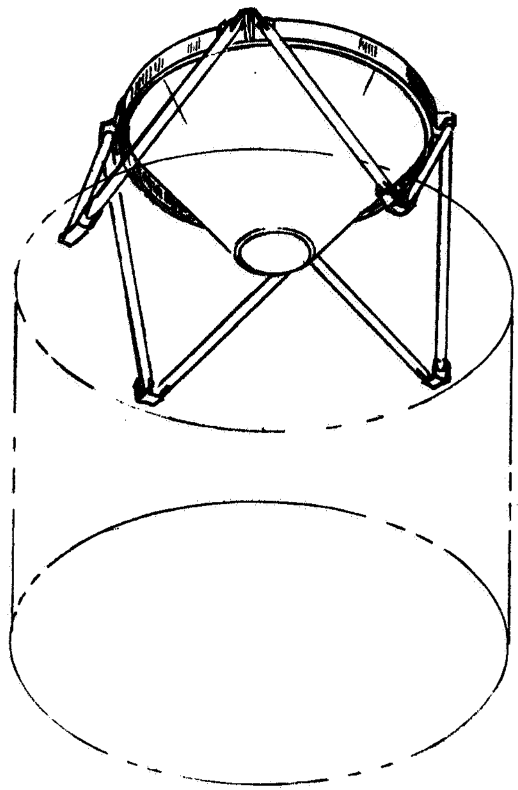


Figure 4-24. Docking Cone

Setting the deflection (Δ) under impact of the compound spring equal to the single spring (ideal structure)

$$\Delta_1 + \Delta_2 = \Delta_3$$

If the resisting force of the two systems is equal

$$P = \Delta_3 K_3 = \Delta_1 K_1 = \Delta_2 K_2$$

Solving for K_2; $K_2 = K_3 \left(\frac{\Delta 1}{\Delta 2} + 1 \right)$

$$\text{or } K_2 = \frac{K_3}{\left(1 - \frac{K_3}{K_1}\right)} = 9.65 \times 10^5 \text{ lb/ft.}$$

Therefore each strut is required to have a spring constant of 10,000 lb/in., which is a spring system sufficiently stiff to permit attitude control of the docked vehicles without excessive vibrations. It is preferable to a dashpot design which may require a clamp-up mechanism to provide a solid attachment during maneuvering; i.e., an additional active system.

4.1.3.6 <u>Solar Panel/ACS Support Boom Assembly.</u> The telescope solar panel/ ACS boom assembly consists of four "A" frame tube trusses which are hinged on the primary SLA support frame. The ACS jet modules (including tanks, regulators, etc.) are mounted on the apex of the booms. The solar cell panels are hinged to the "A" frame by an auxiliary brace which rotates the solar panels 45^{0} from the deployed "A" frame which is normal to the telescope centerline. The solar panels and ACS jets are boom mounted to increase the vehicle roll moment of inertia, reduce structure shadowing and increase the ACS jet moment arm. (See Figure 4-25)

<u>Deployment Mechanism</u> - The initial boom deployment system design utilized a single operating constant force spring powered system with hydraulic snubbers. The original intent was to provide the simplest, most reliable system for automatic deployment of the panels and ACS jets. Recent analysis of EVA maintainability and replaceability point to the support booms as a problem area, because the assemblies, as designed deploy the ACS jets and solar panels to positions which are relatively inaccessible to the EVA astronaut. However, a well stabilized AMU used as a platform would permit the astronaut to hover precisely stabilized while making repairs and/or replacements. The foregoing "Humming Bird" represents a solution to the problems of EVA locomotion and stabilization around and on large structures; however, the concept also involves risks to the astronaut and for this reason an alternate method of panel and jet module replacement has been considered here.

Since the telescope attitude control system is inoperative during resupply and refurbishment, it is possible to design the boom assemblies to be partially retracted to permit EVA access to the solar panels and jet modules by an astronaut secured to the primary truss. The proposed retraction system is a reversible run-around cable system consisting of electric motors with reduction gear boxes, which drive cable capstans similar to the primary lens deployment system.

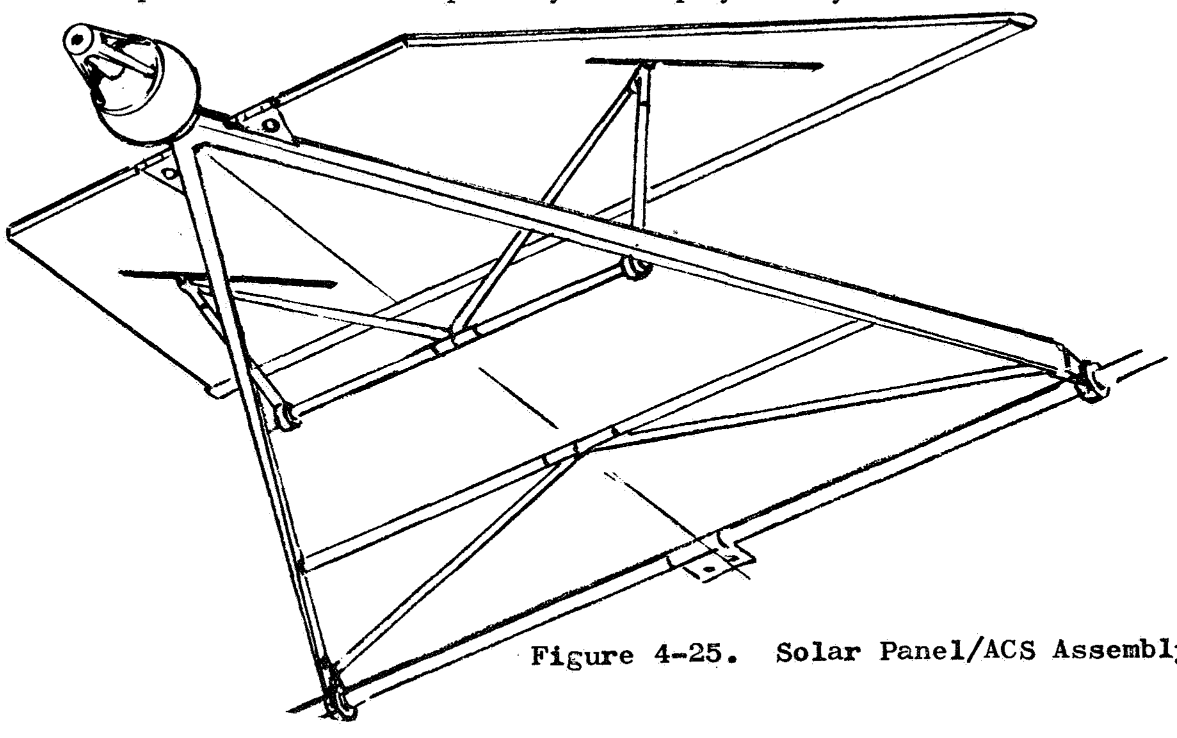


Figure 4-25. Solar Panel/ACS Assembly

After manually releasing the extension locks, the EVA astronaut remotely energizes the boom retraction system for each boom individually as required. It may be possible to eliminate the extension locks depending on cable tension to prevent boom oscillations, thereby simplifying the astronauts task. With a fully reversible system on the booms, the astronaut can retract the panels as necessary to bring them within reach of a secure work position. The astronaut will be required to work on the inside of the primary truss to change the two inward facing solar panels.

The solar area can be increased approximately 20% by increasing the panel dimensions. Additional increased area can be provided by adding solar cell panels on the back (with an appropriate air space) of the present array. The back side panels will be slightly shadowed by the truss work in some sun orientations, but the open nature of the truss prevents serious shadowing. This method of increasing area is preferred for mechanical simplicity to alternate designs using side hinging or folding panels because the panel shadowing of the primary structure is not increased.

EVA Replacement Features - In addition to the retractable boom mechanism discussed previously, there are several other EVA design considerations incorporated into the solar cell/ACS assembly. The solar cell panels are designed with roll up capability by mounting the rigid cells on a flexible cell substrate or using thin film flexible cells. The design simplifies two major problems - reduced storage space requirements on the resupply vehicle and ease of handling for the EVA crewman. Since the deployed panels are 76 in. x 96 in., it is not desirable to require the astronaut to locomote from the SM to the telescope booms with a large panel. Furthermore, the possibility of damage to the cells is minimized by the protective "window shade" canister.

The solar cell canister size for an entire panel is approximately 10 in. in diameter and 8 ft. long. While this is considerably easier to handle than the deployed flat panel, it may be desirable to reduce the packaged size further. The canister length may be halved by adding a structural longeron member along the centerline of the panel frame. This requires that two additional solar panel edges be secured, but reduces the resupply vehicle stowage and EVA handling and transportation problems considerably. The weight penalty for this modification would be negligible. The new canister will be secured to the original container and the cells unrolled over the existing cells. The end and free edges are then secured and the required electrical connections changed. The overlay philosophy assumes that the new active cell performance will not be affected by the decrease in backside radiation. If this is not satisfactory, the original cell panel will be unsecured on the bottom and edges and retracted into the container, the new cells will then be deployed and connected. The "overlay" philosophy is also used on the ACS jet modules. To simplify the resupply or replacement, the astronaut disconnects the electrical connections to the expended or failed unit and secures the new module on the outboard end of the old unit. Two opposing overcenter locking levers provide tension on a ring clamp which secures the flange halves, as shown by Figure 4-26.

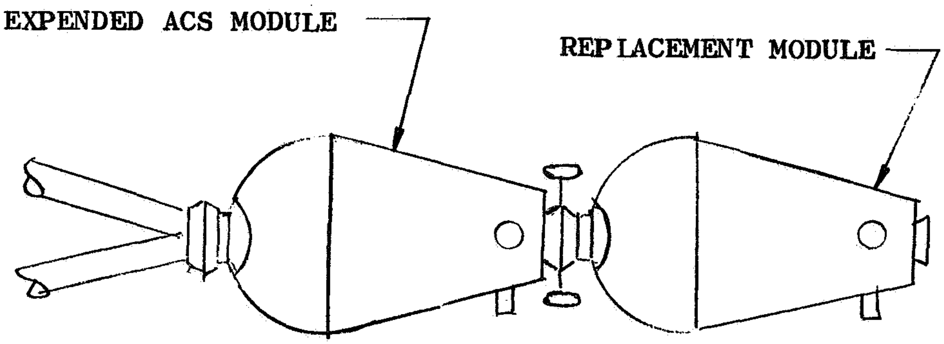


Figure 4-26. "Add-On" EVA Feature

4.1.3.7 Launch Support and Separation System. In addition to the four primary SLA reaction points, the x-ray telescope design includes provisions to support the following assemblies during the boost phase:

a. Forward Truss

b. ACS/Solar Panel Booms

c. Grazing Incidence Lens Segments

d. Image Capsule Equipment Turret

Launch Loads - The Saturn IB launch vehicle accelerations from Reference 5 are listed below:

	Acceleration (g units) Maximum
Lift Off	1.3
1st Stage Burnout	4.15
2nd Stage Burnout	3.1

The acoustic noise time history, sound pressure level, power spectral density and acceleration vs. frequency are shown in Volume II, Appendix B.

Launch Support - The packaged x-ray telescope is securely restrained by the hold downs on the forward truss, ACS/solar panel booms, lens segments, and image capsule turret. The existing structure is adequate to permit normal handling, transportation or hoisting operations without damage.

Separation Systems - The four separation systems required for the telescope experiment utilize redundant pyrotechnic devices for the primary release mode. The separation command will be provided by the CSM crew; no provision for ground release is required as all separation and deployment functions are performed when the CSM is docked to the experiment. For maximum reliability and safety all separation signals will be transmitted from the CSM to the experiment using the umbilical which is connected by the CSM crew subsequent to the initial docking. This hard wire connector will simplify the command system and reduce the possibility of premature separation.

SLA to Experiment - The stowed experiment is supported within the SLA on the four standard LEM pyrotechnic attachment fittings. This is the only structural interface between the experiment and the launch vehicle. As mentioned previously, the telescope truss geometry was designed to integrate the launch support and structure with the deployed orbital structure, and therefore, no additional reinforcement or auxiliary bracing is required to withstand launch loads.

When the proper orbit has been achieved, the CSM will separate from the SLA, turn around and dock on the telescope after the adapter petals are deployed. When the docking maneuver, cinch up and umbilical connections are complete, the CSM will command the release of the primary hold down bolts and extract the experiment from the adapter. No additional jettison springs or propulsion devices are necessary for the separation as the CSM attitude engines provide adequate thrust and orientation control for the maneuver. The separation includes disconnecting umbilicals between the experiment and the adapter.

Lens Support Structure Extension - The sliding lens support truss is clamped to the primary truss by four pyrotechnic bolts, (one on each track) through machined tension splice fittings. One side of the bath tub fittings are located on the lower end of the track slider, the other side is attached to tracks adjacent to the extension system drive support plates. Consideration has been given to the possible requirement for additional transverse structural supports on the lens assembly during the boost phase, because the lens is cantilevered approximately 13 ft. below the launch hold-down fittings.

The maximum axial loading is 4.1 g at first stage burnout. The critical transverse load consists of approximately 0.5 g steady state acceleration and superimposed random vibratory inputs of small amplitude ($\approx$0.07 in. in the 5-16 cps range, the range of the structure natural first mode frequency).

An analysis of the critical launch load indicates additional forward truss supports are not required because of the high structural rigidity and the random nature of the applied load. A more detailed analysis of the structure subjected to the launch vibration specification will be required before precise structural response can be determined. Additional supports in this area are undesirable as they would require additional separation devices and retraction mechanisms or EVA participation in the initial phase of deployment.

Lens Extension - In the packaged launch configuration, the mirror assembly is positioned approximately 5 in. aft of the mirror operating position relative to the forward truss and secured to a support ring by four pyrotechnic bolts. The support ring is strut mounted on the forward end of the gimbal support cylinder. This provision is included to isolate the lens gimbals from launch loads. The telescope gimbal system is a high precision assembly requiring angular positioning and read-out to less than 10 arc sec. accuracy, therefore, it is necessary to unload the gimbals during launch. Although the present mirror support ring does not include attachment for the individual mirror elements, the feasibility of accomplishing this should be examined in further design studies. The present design requires a mirror suspension system which will transmit the axial launch loads from the inner elements outboard to the support ring. While it appears that this is feasible, a more desirable design would include additional launch supports to react the axial inertia of each mirror segment individually. The difficulty of automating this function is in removing the supports from the lens path after separation. This task can be accomplished by EVA if required.

Image Capsule - The recommended image capsule design includes structural restraints on the equipment turret, the restraints react the inertia of the equipment and the turret during launch and minimize the danger of brinelling the turret bearings. The turret bearings are a critical aspect of the telescope design philosophy as they are not replaceable and must be designed to operate for up to 4 years. The observational mission would be seriously impaired if the turret were disabled since the imaging equipment could not be interchanged. The turret restraints removal is difficult to automate and the risk of interference is high, therefore, EVA was selected to remove them. This also permits manned inspection of the image capsule to guarantee that no interferences or loose wires exist.

Consideration has been given to including a clutch in the turret mechanism to permit the astronaut to rotate the turret during his inspection of the capsule. This procedure while not mandatory is desirable due to the large number of instruments packaged in the image capsule and their proximity to the rotating turret.

4.2 SUBSYSTEM DESIGN.

The following subsystems were synthesized for the x-ray telescope spacecraft in order to develop a realistic predesign regarding over-all weight, volume and power requirements in addition to investigating subsystem feasibility.

The scientific guidelines, subsystem concepts, and hardware sizes for the telescope instrumentation and optical drift correction systems were developed through consultation with American Science and Engineering, Inc.

The other subsystem concepts were developed by Convair.

4.2.1 Lens Alignment System. The requirement for lens alignment was established by analysis of the truss structure distortions resulting from orbital solar heating. The redundancy of the truss structure and the large combination of sun angles during observations complicates the determination of precise distortion values, and without detailed mission observational profiles, the worst case sun angles can not be determined. Inspection of the structure geometry shows that maximum strut absolute temperature differentials occur when the sun is oriented 18^{o} from the telescope centerline. The struts pointed away from the sun cool to $330^{o}R$, while the opposing struts stabilize at $490^{o}R$ (based on silver-plated tubing). A high emissivity white paint reduces the maximum to $350^{o}R$, thereby reducing the total possible excursion to 20^{o} (160^{o} at subsolar point). The probability of observing at the 18^{o} sun angle is quite remote, particularly considering the wide range of source locations. Therefore, a sun angle of 30^{o} to the telescope centerline was selected as a design criteria, with the telescope at the subsolar point in the orbit. This assumes a longer lens shade than shown to provide additional shielding. In any event, a necessity is indicated for either: reduction of the distortions by thermal control techniques, or a method of compensating for the distortions.

4.2.1.1 Reduction of Thermal Gradients. It appears that use of the "heat pipe" thermal control could drastically reduce the temperature gradients in the truss structure with a very small weight penalty. The "heat pipe" system involves the use of a fluid such as Freon or alcohol in a closed container (pipe) with a "wick" on the internal container surfaces to assure that a fluid film covers the surfaces. The remaining volume in the container is filled with the vapor of the fluid; other gases are excluded. In operation, the heated areas evaporate fluid which causes condensation in cooler areas. This provides a rapid heat transfer system since the evaporation of the fluid absorbs heat and the condensation evolves heat. The process is continuous since the capillary flow in the wick continually returns the fluid to the surface from which it was evaporated. In representative examples, thermal gradients can be reduced to around 2% of their original temperature difference.

The heat pipe system was invented at Los Alamos and has recently been receiving considerable attention. An excellent general discussion of heat pipes was presented by Dr. Samuel Katzoff at the AIAA Thermophysics Specialists Conference in New Orleans in April of this year (1967 - Paper No. 67-310). Heat pipe systems are currently being investigated within the Aerospace Industry and specifically by General Dynamics Convair and General Atomic.

The application of the heat pipe thermal control system to the x-ray telescope is envisioned as follows: The ends of the titanium tubes in the truss structure would be interconnected forming one continuous chamber sealed from the outside. Thus, each truss structure would be a heat pipe system. The inner surfaces of the tubes and the connectors would be covered with a very lightweight mesh screen, plastic or other material suitably bonded to provide the capillary attraction. The system would be evacuated and the proper amount of a working fluid would be introduced. It is impractical to interconnect the two truss assemblies; i.e., the forward lens truss and the primary truss, so there would be two heat pipe systems involved. The elimination of the temperature gradients in each structure eliminates the distortions in the positioning of the image even though a small temperature differential existed between the two structures. In actuality, the thermal gradient can not be completely eliminated, but the reduction to about 2% of the uncompensated value might eliminate the necessity for an active alignment system. Until a detailed analysis of the use of a "heat pipe" system is made, the necessity for an active lens realignment system is assumed.

4.2.1.2 Compensation Techniques. Consideration was given to increasing the size of the image plane equipment (orthicon, image intensifier, etc.) to compensate for image shifts. This increases the image scan and associated telemetry requirements in proportion to the area increase.

A more significant problem resulting from changes in the lens reference coordinates is the effect on the vehicle pointing accuracy. The preliminary mission design criteria require the navigation system to point and hold the telescope to any given celestial coordinates within a 1 arc sec. error circle. This limits the field of the image motion compensator to reasonable values.

Distortions of the truss structure change the lens optical axis with respect to the star trackers adding to the tracker error. The anticipated structure distortions alone exceed the 0.5 sec. accuracy pointing requirement.

For the above reasons, a lens compensation system is required and several concepts were considered. The alignment accuracy required is approximated by subtracting the star tracker tolerance of $\pm$ 15-22 arc sec. (from Reference 6 and 7) from the $\pm$ 30 arc sec. pointing goal, or $\pm$ 8-15 arc sec. tolerance.

Regardless of the lens alignment system used, it is necessary to provide Z & Y axis error signals to the navigation computer to account for changes in lens centerline with respect to star tracker reference coordinates (Φ). (See Figure 4-27)

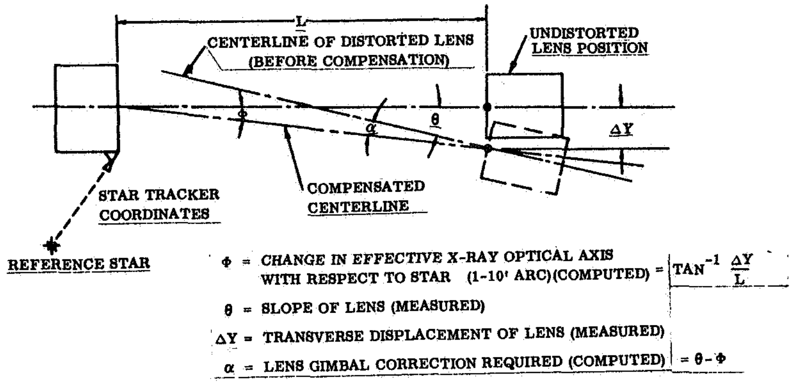


Figure 4-27. Lens Alignment Geometry

4.2.1.3 Equipment. Several system concepts have been considered for determining and compensating for the lens misalignment, one of the most promising is a laser electro-optical assembly.

The assembly consists of two mirrors, one plane and one corner cube reflector mounted on the lens capsule, a laser beam splitter, and two photo detector sensors mounted on the image capsule.

The plane mirror rotates a relfected light beam, an angle twice the mirror rotation. Mirror translation does not affect this beam. A corner cube reflector always returns the light beam parallel to the incident beam, and small angular rotations of the reflector have no effect on the beam. Lateral translation of the reflector by X displaces the return beam by 2X. Thus, the combined corner reflector and plan mirror can independently measure both the structure and lens assembly rotations and translations necessary to realign the optical axis with the image capsule.

Gallium Arsenide Infrared Lasers are a good choice to minimize system electrical power requirements. Power will be reduced further by pulsing the laser approximately once every second. The laser filament power of 5 watts must be supplied continuously to eliminate heat up time, but the additional operating power of 20 watts is reduced to 2 watts average by a maximum 10% duty cycle factor. The precise pulse timing would be established by the response time of the gimbal system in order to close the loop and eliminate "hunting".

The alignment system discussed here utilizes state-of-the-art equipment; lasers, sensors, electronics, etc., no long development articles are anticipated. Convair optical laboratory estimates a working model could be operating in 2 months and a flight hardware prototype ready for testing 6 months after go-ahead. A schematic of the proposed alignment system is shown in Figure 4-28.

4.2.2 Electrical Power and Distribution System. The electrical power subsystem for the x-ray telescope consists of an omnidirectional color cell array, batteries, conditioning, and distribution equipment. The system block diagram and a summary of the components are shown in Figure 4-29.

4.2.2.1 Power Profile. The power requirements for the spacecraft are based on a viewing time of up to 100% and an average ground station contact of 5 minutes per orbit. The sunlight time of the 28.5° orbit will average about 65%.

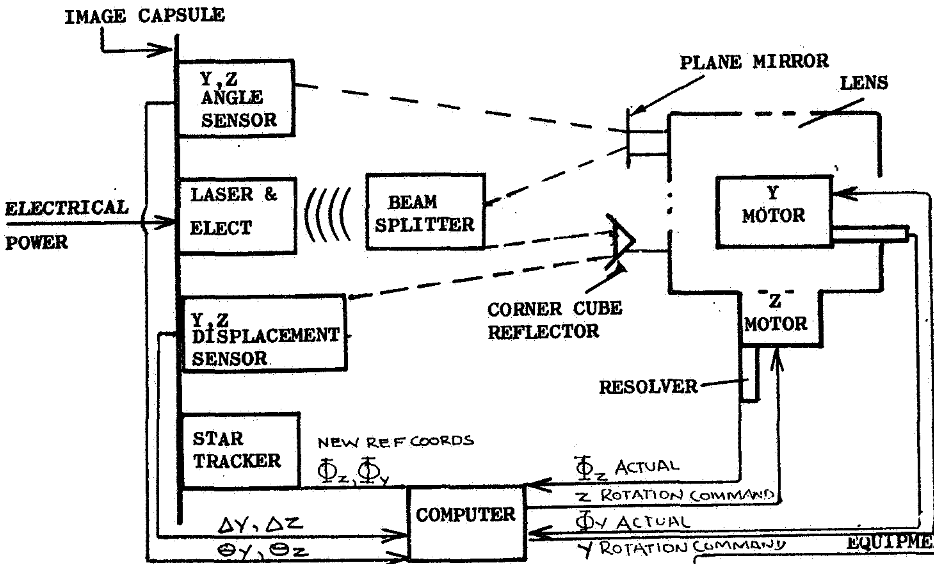
The average power to supply the various systems will be about 260 watts and the peak power may be as high as 450 watts. A sample power profile is shown in Figure 4-30.

A battery source takes care of peak power requirements and dark portions of orbit. The solar array supplies the required average power plus the power lost in the battery conversion. The solar cell power will only be supplied during the sunlight time or 65% of orbit. The solar array power requirement is:

$$P(\text{solar}) = \frac{P(\text{ave}) + L(\text{watt})}{.65} = \frac{260 + 20}{.65} = 430 \text{ watt/orbit}$$

4.2.2.2 Solar Cell System. The solar array is designed to meet the load requirements under the various constraints of the orbital conditions and to replenish the battery in the 63-minute light period. Typical characterisitics of available solar cells are summarized as follows:

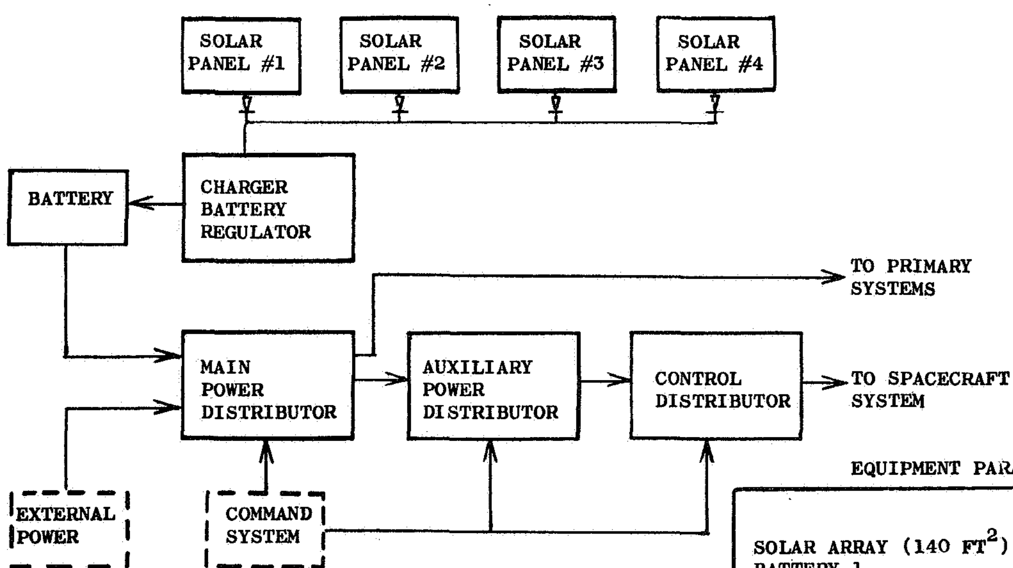
	1967			1972	
	Conventional Silicon Cells	Dendritic Silicon Cells	Thin-Film Non-Silicon Cells	Projected Performance Thin-Film Cells	X-Ray Design Values
Watts/lb (gross area)	6.5	12.0	20.0		
Watts/sq. ft. (net area)	9.8	8.3	4.5	15	13.5
Cell Conversion Efficiency	12.0%	10.5%	5.0%		
Area Utilization Efficiency	90.0%	95.0%	75.0%	95	92.5
Space Exposure Efficiency(2 yr)	50.0%	60.0%	80.0%		
Radiation Resistance	Fair	Fair	Good		



EQUIPMENT PARAMETERS

	IN^3	LBS	WATTS
LASER & ELEC	63	5	7
BEAM SPLITTER	8	1	
SENSOR	14	1	
SENSOR ELEC	96	5	2

Figure 4-28. Lens Alignment System X-Ray



EQUIPMENT PARAMETERS

	IN^3	LBS
SOLAR ARRAY (140 FT^2)	13000	140
BATTERY 1	1500	36
BATTERY 2	1500	36
BATTERY 3	1500	36
REGULATOR	100	5
BATTERY CHARGER	80	3
MAIN PWR DISTR	146	5
AUX PWR DISTR	55	3
VOLTAGE BOOSTER	64	5
VOLTAGE LIMITER	8	1
LOW VOLTAGE SENSOR	8	1

Figure 4-29. Electrical Power And Distribution System

SYSTEM	TOTAL COMPONENT POWER - watts	PEAK POWER * watts
NAVIGATION	241	226
TELESCOPE INSTRUMENTATION	130	70
DATA, TLM, AND COMMUNICATIONS	90	59
ATTITUDE CONTROL	57	57
OPTICAL DRIFT CORRECTION	24	24
COMMAND	15	12
LENS ALIGNMENT	9	9
	566	467

* Peak power equals total component power minus redundant items, the peak power is never achieved, as many components do not operate continuously. The average power drain is 57% of peak power.

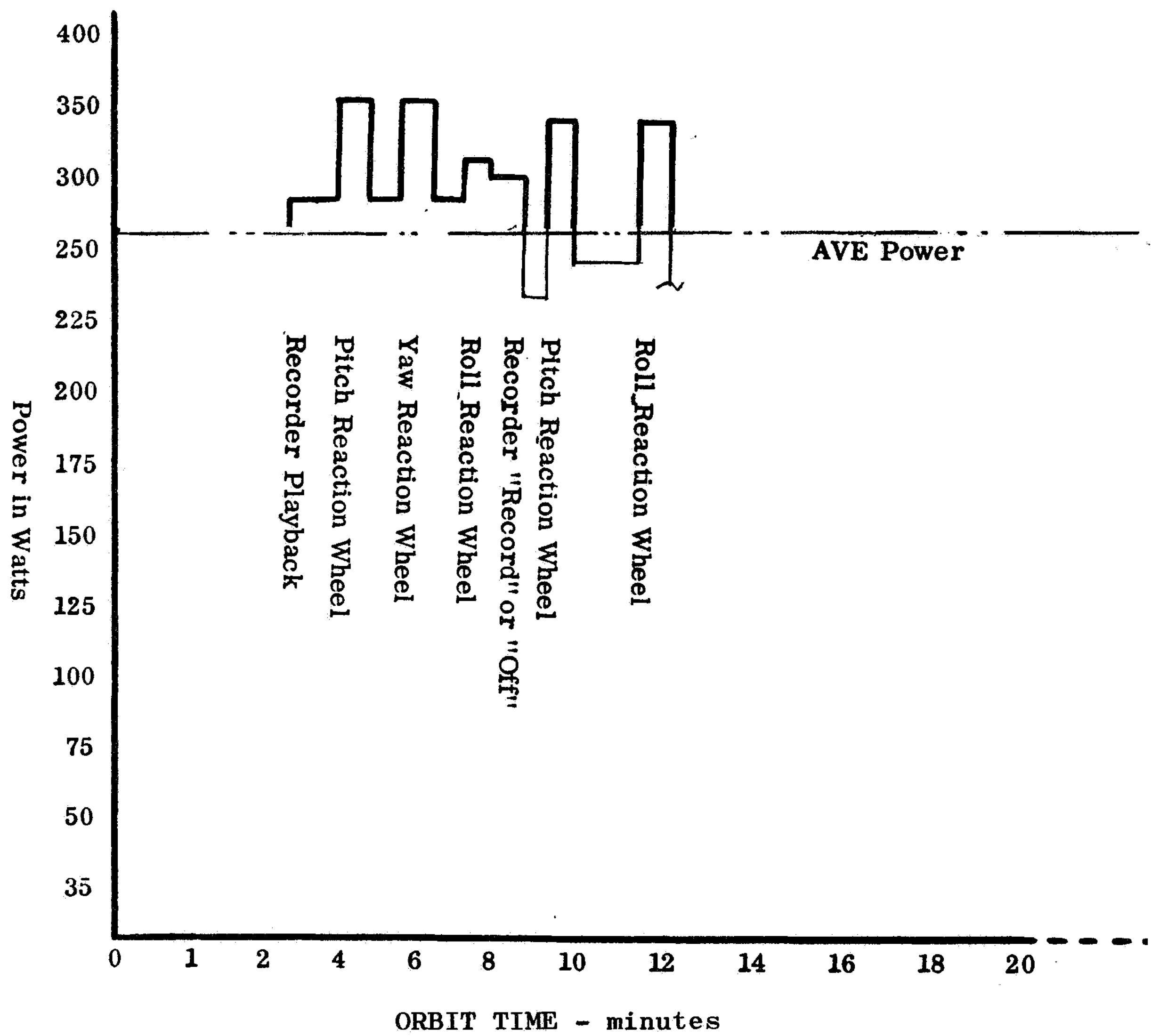


Figure 4-30. Electrical Power Summary

The solar cell performance parameters used to size the x-ray telescope array are slightly more conservative than industry-anticipated performance for the 1972 time period. The roll-up array is capable of being designed with flexible cells or rigid cells on a flexible substrate. The thin film cells are more suitable for the two years without replacement assumption because of reduced degradation in the space environment and high resistance to handling damage.

The solar array design shown has not been optimized for final area configuration since this is dependent upon a correspondingly high degree of optimization within all systems affecting the power generation. The basic concept, however, possesses sufficient design flexibility to cope with increased power requirements by moderate changes to the solar panel or support truss geometry.

Of these, the conventional solar cells are most readily available and are the only ones with demonstrated flight performance. The cells are mounted on four flat panels arranged for random orientation relative to the sun. The 140 sq. ft. array provides 13.5 watts/sq. ft. for a total capability of approximately 1.9 KW. However, the configuration is arranged for a long-term average aspect ratio of $1/\pi$ or 600 watts. A sufficient margin is allowed for less-than-average aspect ratio or radiation degradation.

Energy Storage - Batteries are generally used in spacecraft for storing electrical energy. Typical characteristics of appropriate systems are shown below:

Battery Type	Cell Volts	W-Hr. Per Lb.	W-Hr. Cu. In.		Shelf Life
Nickel-Cadmium	1.1	15	1.7	a. 25% Drain 10,000	5 year
Silver-Cadmium	1.08	25	1.5	a. 25% Drain 6,000 b. 50% Drain 2,000	3 year
Silver-Zinc (Sealed)	1.5	50	2.5	a. 25% Drain 1,000 b. 50% Drain 400	2 year
Zinc-Oxygen	1.3	70	5.0	Primary	5 year

For the x-ray telescope application, the nickel-cadmium type of battery offers the advantages of good energy density, adequate cycle life and good shelf life. Battery sizing is based on providing 350 watts for the duration of the orbital dark period, as follows:

Battery Discharge = (350 watts) x (0. 5 hours)
= 175 watt-hours

At a 30% depth of drain, the required capacity is: $\frac{175}{0.3}$ = 583 watt-hours

Battery characteristics are:

1. Capacity - 583 watt-hours
2. Rating - 20 amp. hours
3. Weight - 36 pounds

Three such batteries are included to enhance system reliability through redundancy.

4.2.2.3 Power Conditioning And Distribution. A control system will turn the solar array charging current "off" and "on". Voltage monitors will switch the charge current from the selected level to the trickle charge level with a fixed time return. Temperature sensing devices reduce charging rate when battery temperature approaches a dangerous value. Failure protection allows reduced operation on one battery or on solar array only.

Regulation - Pulse width modulation techniques provide regulated D-C voltage (± 1/2%) at efficiencies in excess of 90%. If required, a boost feature can be added to extend the range of input voltage over which the unit provides good regulation.

Chargers - Several techniques are available for controlling the state of charge of spacecraft batteries. Currently, the optimum systems relate to nickel-cadmium batteries which have been used most extensively in space programs to date. The near-optimum Amp-Gate charging system was developed by Mallory Company for nickel-cadmium batteries. This device is connected to the cell terminals so that conduction starts when the end-of-charge voltage is reached. Current flow causes a temperature rise in the diode which decreases its resistance, thereby allowing more current to flow through the diode. Within a few minutes almost all of the current is flowing through the diode while a small trickle is flowing through the cell. The cell is then protected from excessive charging rates. It is anticipated that this technique will be available for the x-ray telescope.

Distributors - The distribution of electrical power is controlled by means of relay switching. Appropriate division of loads is made for experiment and support subsystems.

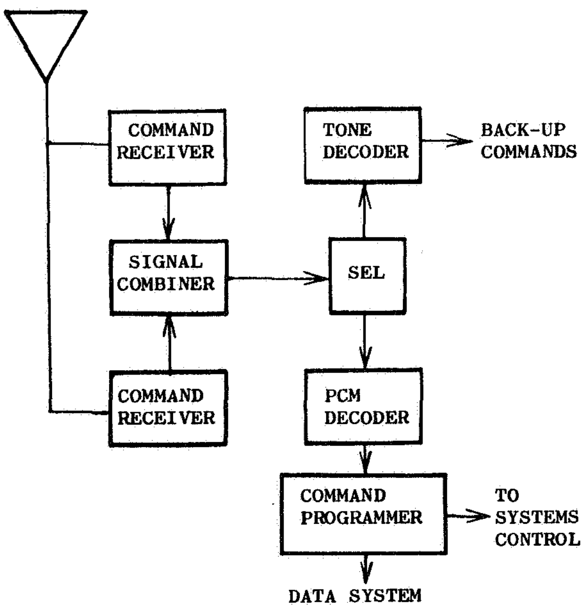
4.2.3 Command Systems. Many operating functions of the telescope will be remote controlled from another spacecraft or a ground station. Many of the telescope deployment operations described in Section 4.2 will be controlled by remote commands. After the vehicle is deployed, orbit and attitude commands will position vehicle. Pointing commands will locate x-ray sources with the telescope. Sun shields, lens covers, gratings, crystals and spectrometry devices will also have command controls. Power to various systems and/or devices will use radio control signals. Modes of operation of data storage equipment and multiplexing equipment will be switched by command codes.

The main functions to be controlled by the command system include the following groups: external relays, internal relays, guidance computer, central timing up-dating, test messages, and reset command. These groups have several individual functions to be addressed.

About 100 remote command functions will be used to control and operate the spacecraft and associated equipment. A block diagram of the vehicle command control system and equipment list are shown in Figure 4-31.

The received signal will be demodulated and decoded; the commands and instruction words with the necessary synchronization signals will be fed to the proper spacecraft systems. A command verification will be provided to ensure command reliability.

Specifications - The command system receives digital data from the ground station, verified it, determines the system and function to which it is addressed, and routes it to the function. The command system consists of two command receivers, a signal selector, a detector-decoder, and interface and buffer storage, relay drivers, and a relay assembly.



EQUIPMENT PARAMETERS

	°C	IN^3	LBS	WATTS
COMM RCVRS (2)	+65°	85	4	4
SIGNAL CONDITION	-20°	15	1	1
TONE DECODER	"	40	1.5	2
PCM DECODER	"	38	2	3
COMMAND PROGRAM	"	70	5	5
HARNESSES, BRKTS			5	

Figure 4-31. Command System

A digital command format, compatible with the ground station network, will be demodulated from the FSK modulated carrier. A pseudo-random noise code will establish bit synchronization with a phase-lock loop. A decoder programmer will route the command instructions to the various sytems and functions.

The programmed codes will operate relays, reload or up-date computer instructions, up-date timing generator with time words, reset relay groups, and test system.

The received command word will be checked and returned on telemetry link for verification. If errors occur, retransmission will be requested. If verification is made, an execute command is transmitted from grou nd to spacecraft.

A tone decoder provides for back-up and emergency operation.

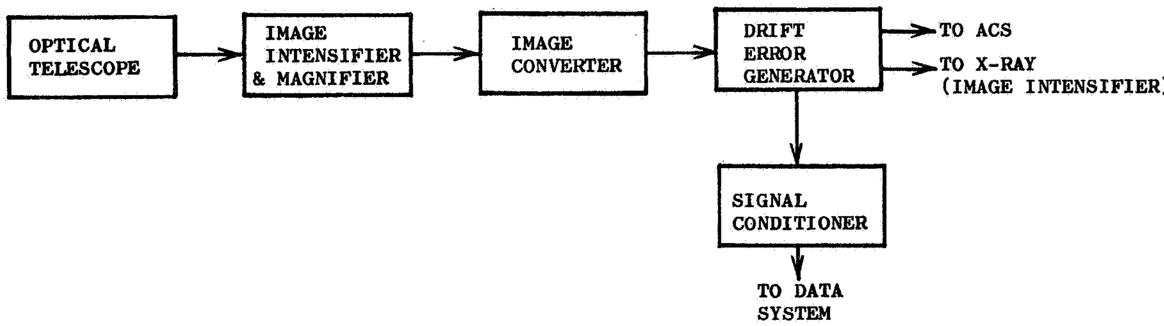
Discrete bit signals will be generated and transmitted on the telemetry link to give position verification of execution.

4.2.4 Optical Drift Error Signal System. To simplify the imaging of x-ray sources to an accuracy of one arc sec. an image motion compensating device is considered desirable to maintain the image forming photons in the correct positions ($\pm$ 0.5 arc sec.) while the image is generated on the film or vidicon.

The drift correction system uses a star tracking optical telescope, which is coaligned to the x-ray mirror assembly. By observing a star field, the vehicle drift is determined and generates error correction signals used in the image motion compensator. The system block diagram and typical equipment parameters are shown in Figure 4-32.

4.2.4.1 Function. The drift tracking system will be capable of generating signals that will maintain an image accuracy of one arc sec. during the image integration time although the x-ray telescope drift rates may be as high as five arc sec. per sec. The vehicle and x-ray telescope pointing coordinates will be stabilized in pitch and yaw within $\pm$ 30 arc sec. and roll will be maintained to $\pm$ 5 arc min. accuracy. With the roll of the vehicle maintained within 10 arc min. the image error from roll alone would be well within tolerance.

Since the x-ray source to be investigated can not be assumed to have a visible counterpart, or its visible counterpart is faint, brighter nearby stars must be used for guidance. Further, since the telescope is coaligned with the x-ray telescope, not gimballed, a field of view is selected which assures that a star of some minimum brightness will be visible during pointing to sky coordinates. A plot of star population per square degree (see Figure 4-33) shows that a field of view of 2° will gurantee the presence of one or more stars of the tenth apparent magnitude for the worst case of looking toward the galactic poles, and at least one of eighth magnitude when looking at the galactic equator.



EQUIPMENT PARAMETERS

	IN^3	LBS	WATTS
OPTICAL TELESCOPE	13600	200	
IMAGE INTEN & MAG	250	20	10
IMAGE CONV (VIDICON)	300	20	10
DRIFT ERROR CONV	30	2	2
INTERFACE UNIT	32	2	2

Figure 4-32. Optical Drift Correction System

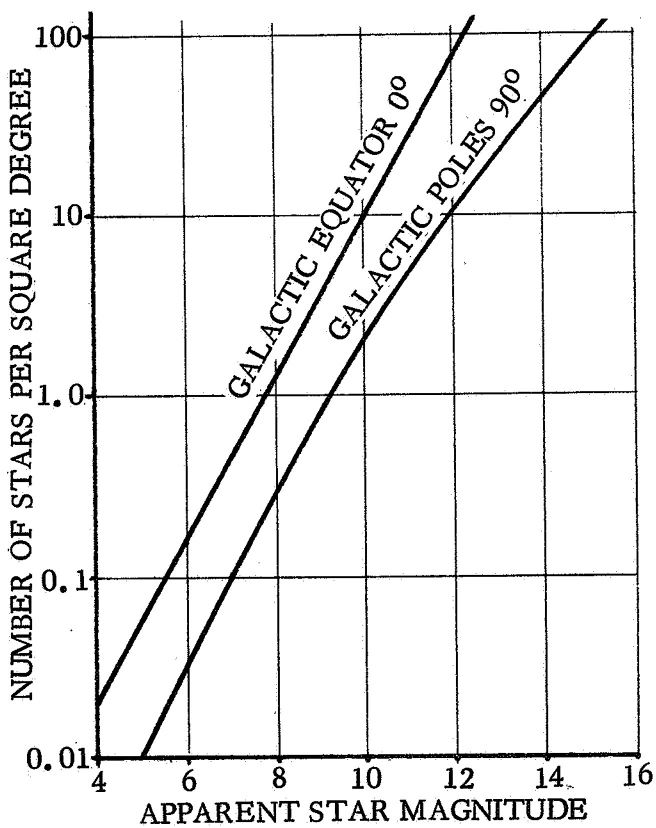


Figure 4-33. Star Population Density

The location of one or more stars will be recorded by the optical system vidicon at vehicle lock-on time; any changes in the reference star positions will be converted to positioning signals by the error generator and relayed to the x-ray image motion compensator to maintain the x-ray image within one arc sec. of original position at lock-on. A maximum drift correction of one arc min. will be required for the drift error system.

4.2.4.2 Specifications. The optical drift tracking system uses an optical telescope with a field of view of 2^o. An image intensifier and scan magnifier will assure less than one arc sec. position determining accuracy.

A typical vidicon could have a 1000 line scan at 5 times per sec. To obtain a one arc sec. resolution accuracy the maximum vidicon field would be 17 min. Therefore, a 2^o telescope field requires a magnification of about $\frac{2 \times 60}{17} = 7$ times to obtain a resolution less than one arc. sec.

The image intensifier positions the star location on the vidicon screen. The coarse scan will be used without the magnifier to locate the star coordinates with respect to the vehicle coordinates. Signals will be generated to position the star in the center of the vidicon scan field. Once the star is centered on the vidicon it will remain within the vidicon field since the vehicle pointing is held within one arc min. An automatic control will switch the magnifier out when the reference star is not within the vidicon field.

A computer will determine the star coordinates and compute the offset values required to position the star in the center of the vidicon field view. It will also determine the mode of the magnifier. The guidance control computer may be used for this function.

The signals generated for the x-ray image position compensator will be entered into the data format for telemetering to a ground station and later checked for accuracy, when compared to the other pointing reference readout.

4.2.5 Navigation System. The necessity of avoiding equipment duplication and unnecessary software results in the location of all sensor and computer functions in one integrated system. This navigation system is discussed here. A major portion is ground based.

4.2.5.1 Function. The functions which the satellite integrated sensor and computer system performs are minimized to improve reliability. Ground based sensor and computer facilities are more easily maintained, and functions will be performed on the ground where possible.

A basic disadvantage of ground basing is the quantity of information which must be sent up to the satellite. Further, continuous contact is not available. For this particular experiment, the information which must be transmitted is relatively small, and continuous satellite contact is not required.

Approximately once each orbit, the vehicle will be provided with the coordinates of the next source. If it is necessary to shift one or both of the star tracker's reference stars, the information for performing this change will also be transferred. The vehicle attitude control system will then maneuver to the source, lock-on, and conduct the observation. Experimental results will be sent down and the next source pointing data will be sent up. Since it may not always be convenient to transfer this information between each observation, the vehicle system will be capable of storing data for several observations.

The ground based functions include:

a. Orbit determination - estimation of orbit.

b. Orbit integration - continuous estimate of satellite orbital position, velocity, and altitude.

c. Programming - selection of sources and providing source coordinates to satellite.

d. Star tracker pointing - data necessary for acquisition of a different stellar reference.

e. Control of drag velocity loss correction.

The vehicle based functions include:

a. Calculation of vehicle rotational rates from star tracker position data.

b. Autopilot control of each channel.

c. Logic for inertia wheel unloading.

d. Failure detection logic.

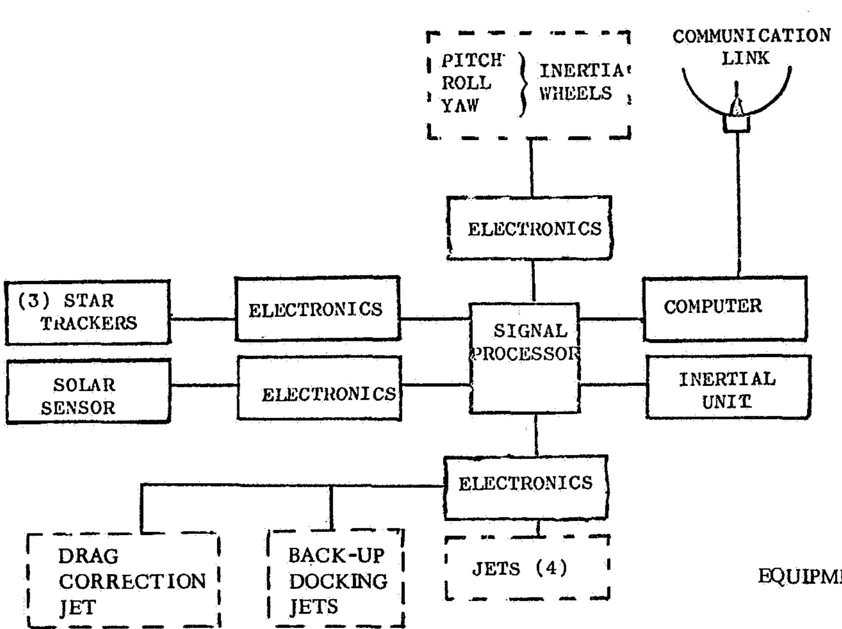
e. Switching redundant jet valves.

4.2.5.2 Specification. Since the vehicle is to be pointed to $\pm$ 0.5 arc min., star trackers must be utilized as the sensors. The vehicle orientation must be determined by the computer from star tracker data well within the position specification of $\pm$ 0.5 arc min.

An additional orientation reference is necessary to provide stabilization while a star tracker is being switched between stellar references. The tolerance on it will be determined by the star tracker operation rather than vehicle pointing.

The computer will have a small load and will be relatively small physically. However, a considerable amount of redundancy will be used for longer life.

4.2.5.3 Equipment. The equipment list and block diagram for sensor instrumentation and computing is shown in Figure 4-34, and power and weight estimates are included. An additional star tracker is included for redundancy, so there are three. The additional orientation reference will be an inertial unit. Further, a solar sensor is included as an aid to initial orientation and for vehicle stabilization should two star trackers fail.



EQUIPMENT PARAMETERS

	IN^3	LBS	WATTS
INERTIAL UNIT		30	100
STAR TRACKER (3)	3000	30	45
SOLAR SENSOR	10	1	1
COMPUTER	700	35	75
SIGNAL PROCESSOR	700	18	10
PHASOLVER	210	78	2
SENSOR SIGNAL PROCESSOR(P&Y)	300	10	5
SENSOR SIGNAL PROCESSOR(ROLL)	180	6	3

Figure 4-34. Navigation System

Preliminary specifications for the Star Trackers and Computer:

STAR TRACKER (Kollsman solid state tracker)

General

Solid state electronics.

Two-position gimbals, torque motors respond to digital commands from external computer.

Point telescope within 120^{o} cone angle.

System Size	Tracker	Electronics
Volume	<1000 in.^{3}	<100 in.^{3}
	<10 lb.	<5 lb.

Power	<15 watts
Reliability	Designed for one-year operation in space.

Performance	
Accuracy	15 arc sec. per axis (2σ).
Recognition	Acquire and track stars of +1. 1 silicon magnitude and brighter.
Acquisition Rate	0. 5^{o}/sec. (acquire star for apparent velocity).
Sun/Earth Impingement	Unimpaired accuracy: as close 25^{o} to sun; 1^{o} to earth.

Environmental	
Temperature	0^{o}F to 150^{o}F
Acceleration	11. 3 g (all axes) duration 4. 5 min. /axis

DIGITAL COMPUTER (Preliminary, similar to IBM 4π)

General

General purpose, stored program, binary operated airborne computer.
Memory: random access, parallel readout.
Parallel processing.
Redundancy: parts redundancy.
Arithmetic: parallel processing

Size	
Volume	≤ 700 in.^{3}
Weight	$\leq$ 35 lb.

Power	≤ 75 watts (see notes)
Reliability	≥ 5000 hours, MTBF (see note 2)

Performance

Word Length: (may be decreased) See Note a.
Instruction: (32 bit instruction words)
Data: orbit data word

Instruction Times: (further speed-up with current micro-circuits)

Add/Subtract	<500 μsec.
Multiply	$<$ 10 μsec.
Divide	$<$ 15 μsec.

Memory

Access Time 1. 0μsec.

Notes:

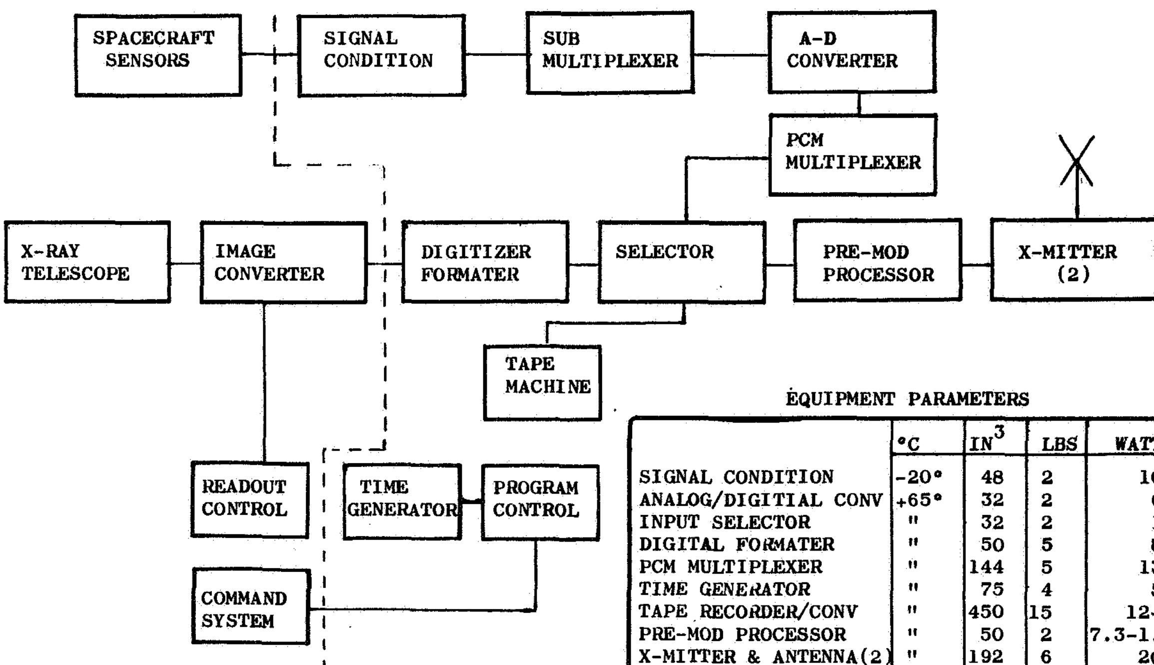
a. Computer specification covering speed, capacity, word length, and power should be finalized only after total computational tasks have been defined; also required accuracy and operating periods.

b. It is assumed that during some periods, the computer can be shut off.

c. Power indicated is an average estimate made by evaluating Saturn, Apollo, IBM4111 and micro-electronic 1224 computers. This estimate should decrease at a future date.

4.2.6 Data, TLM And Communications Systems. The information collected by the focusing x-ray telescope spacecraft will be furnished to the ground for use in scientific studies. To avoid the risk of losing the information by storing the data and retrieving physically, a telemetry system is planned. Immediate results of the measurements are then possible so that corrections or alterations in the operations can be tried.

The types of information to be collected lends itself more readily to a digital type data system. Therefore, a PCM telemetry system is considered for this vehicle. A block diagram of the system and equipment list with sizing parameters is shown in Figure 4-35.



EQUIPMENT PARAMETERS

	°C	IN^3	LBS	WATTS
SIGNAL CONDITION	-20°	48	2	10
ANALOG/DIGITIAL CONV	+65°	32	2	6
INPUT SELECTOR	"	32	2	1
DIGITAL FORMATER	"	50	5	8
PCM MULTIPLEXER	"	144	5	13
TIME GENERATOR	"	75	4	5
TAPE RECORDER/CONV	"	450	15	12-7
PRE-MOD PROCESSOR	"	50	2	7.3-1.5
X-MITTER & ANTENNA(2)	"	192	6	20
MULTIPLIER	"	48	2.5	5
PRE-MODUL FILTERS	"	7	.5	1
PROGRAMMED SELECTOR	"	60	2	2
HARNESSES, BRKTS			10	

Figure 4-35. Data, TLM & Communications

Assumptions for Data Collection:

Locating source and determing structural nature -

a. Maximum time 60 min. (1 min. minimum)

b. Image resolution to 1 arc sec.

c. One source in field

d. Maximum sourse size: 2 arc min.

e. Eight levels of intensity (3 data bits)

Measuring intensity vs. wavelength -

a. Bandwidth 2 Å - 60 Å (Possibly 3000 Å)

b. Resolution 0.25 Å (240 band measurements)

c. Intensity levels (10 to 5×10^4 counts) (14 bit word)

d. Ten bands per orbit (Slitless spectrometer)

e. Maximum time per band 150 min. (10 min. minimum)

Determining Polarization -

a. Maximum time per test 60 min. (5 min. minimum)

4.2.6.1 Data Handling Parametric Studies. First analysis indicated that the data load for the x-ray astronomy telexcope could be rather large.

A calculation was made assuming the use of a memory tube of some type (such as an Astracon) for accumulating the x-ray image and using a scanning device to convert the image to a telemetry signal for storage and transmission to ground stations.

A resolution of one sec. in a 30 min. field requires at least 1800 lines per frame assuming 8 levels of intensity (3 bits).

Bits per frame $= 3 \times 1800 \times 1800 = 9.72 \times 10^6$ bits/frame.

A frame per min.

$$\frac{9.72 \times 10^6}{60} = 162 \text{ kbps}$$

Additional data such as: drift error, position coordinates and such, could increase rate by possibly 10% or about 178 kbps.

Since ground coverage is limited to about 5 - 10 min. per orbit, it is necessary to store data and transmit to ground station at an increased rate of approximately $\left(\frac{90}{5} = 18\right)$ - say 16 to 1.

The maximum data handling rate of magnetic tape machines is between 10^6 and 1.5×10^6 bits per sec. This is either by using a continuous data envelope on one track or by using parallel digital on multiple tracks. The over-all difference is not appreciable.

Record rate for a 16 to 1 speed up is $\frac{1.2 \times 10^6}{16} = 75 \times 10^3$ bits per sec.

This is too low a rate for an image scan of once per min. Therefore, the data load must be reduced or the storage capacity increased. The scan interval could be reduced to once per 2.5 min. which reduces the data load so that one tape recorder would record data for one orbit and playback during a six min. ground station coverage.

Another solution would be to use 3 tape recorders to record sequentially and then playback to ground station simultaneously. This method requires a wider transmission bandwidth because of the larger data load.

<u>Data Reduction</u> - Methods can be used to reduce the data load when only a certain part or parts of the image is of interest. It is necessary now to label the scan positions of interest. When all points of the scan were sequential the points of interest are positioned in time from the starting point. When unwanted points are eliminated, the time pattern has discontinuties.

Because of the discontinuties of time, an additional 22 bits (for a resolution of 1800) per point of image must be used. The field of interest must be less than 1/12 of the total field to effect a data saving (3 bit resolution). See Figure 4-36.

1 min. x 1 min. section = 86.4 K bits per frame; 37.5:1 saving.

Considerable saving could be obtained by retaining only the slices of the image containing the points of interest. In this case, only 11 additional bits are required for each scan line having points of interest (0.2% increase in bits). Saving is almost directly proportional to the slice width. (1 min. slice; 30:1 saving)

The data compression requires additional equipment. A buffer storage is required to compress the data and generate an evenly time sequenced data rate. The data must be arranged in a multiplexed format with the necessary information for retrieval and reconstruction.

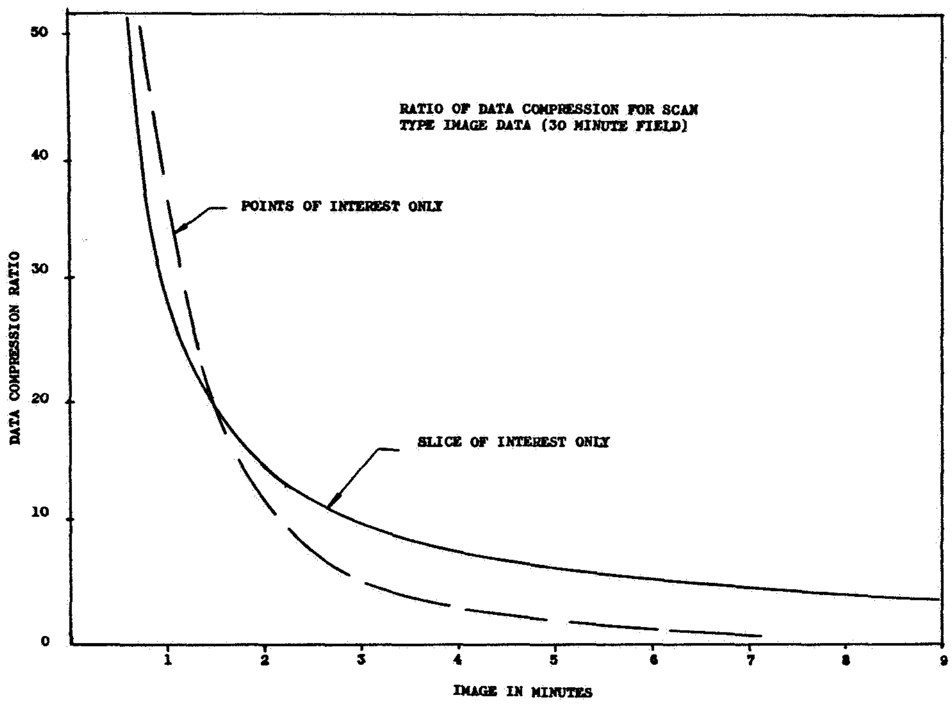


Figure 4-36. Compressed Image Data

A device for recognizing the area of interest in the image field is necessary. This analyzer determines whether the image point and its location should be retained. The image point value (intensity) and its location must be held until the analyzer makes a decision. If the decision is "yes", the values are transferred to the buffer storage. The buffer storage accumulates the image information in digital form for one frame scan. These values are transferred under control of multiplexer to the magnetic tape.

Locating Source and Mapping Structural Nature - Telescope is pointed within ± 0.5 min. of source. Use star-tracker to determine error to ± 15 arc sec. Image compensator used to correct image position to 1 arc sec. per sec.

An x-ray image intensifier tube (phospher plate) converts the x-ray photons to visible light photons. An accumulator/storage tube will be used to integrate the image for a period of time. A storage time of 45 to 60 min. may be desirable. This probably can not be done with electronic scanned tubes because the smearing degrades the resolution.

Therefore, it may be necessary to scan the tube at short time intervals (25 or 30 sec.) or use film as a collector. As calculated above, one full frame would consist of about 9.75×10^6 kilobits. Assuming a field of interest of about 2 arc min. the data can be reduced to 650 kilobits per scan. With a scan every 30 sec. for a 60 min. test, $\frac{650 \times 10^3 \text{ bits}}{0.5 \text{ min.}} \times 60 \text{ min.} = 7.8 \times 10^6$ bits per orbit would result.

Assuming an average 6 min ground coverage, the playback rate is:

$$7.8 \times 10^6 = 260 \text{ kilobits per sec.}$$

The housekeeping data would not be recorded (or only at a very low rate) and would be transmitted on a separate subcarrier while over a ground station.

An intensity monitor would be used to control the scan rate and the mapping time. With high intensity, the scan rate would increase to prevent over-exposure and loss of intensity (gray level) resolution. The high intensity sources would require less time for mapping and, therefore, the data load would be maintained at a reasonable level. The tape recorder capability would be increased or a buffer storage would be used so that only one scan out of a number of scans (depending on intensity) would be recorded.

If film was used, this same monitor would prevent saturation. With film, the image would be collected for the time necessary to sensitize the film. A frame per sec. would be necessary to map some sources. The exposure time would be recorded on film and other data, such as position of telescope and image compensator, would be recorded on tape for transmission.

At time of this flight, it will probably be feasible to use a satellite communications network for continuous coverage. Therefore, the results will not depend upon the reliability of the magnetic tape machine which is probably the weakest link in the system.

Calibration of the image compensator will be required at regular intervals.

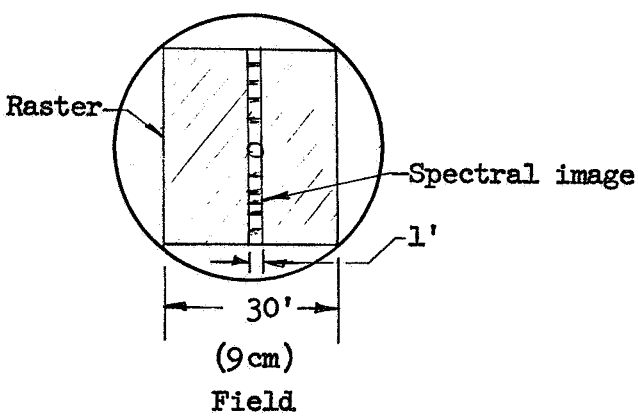
Measuring Intensity Vs. Wavelength - Measurements of spectral distribution will be made with the slitless spectrometer, a focusing crystal (Bragg) spectrometer, and a concave grating spectrometer. Only one type of measurement will be made at a time.

The slitless spectrometer uses different gratings to cover the x-ray spectrum. One grating will be used to cover the range from two to approximately 10 Å. The integration time varies according to the source strength and should be in the range from 150 - 1000 min. For a source strength of 1.8×10^{-3} photon/sec., the lock-on time to obtain 0.025 Å resolution in a 10 Å spectrum would approach 10,000 sec. Since this integration time is greater than an orbit time, the x-ray source would be in a direction perpendicular to the plane of orbit, or else the spectral bar image must be stored during occultation time. This also requires blocking out the imaging equipment during occultation so that extraneous sources would not contribute to the measurement.

Since it is difficult to store the electronic energy such a long periof of time, the spectral image will be scanned at a short interval (5 min.) and the recreated image will be integrated at a ground based station.

Computations for a 20,000 line per cm grating shows a resolution of 0.025 Å and an image size extending 4 cm on each side of a center point. For a spectral band of 10 Å, the image scan would require 400 lines per 4 cm or 10 lines per mm. Therefore, the same scan resolution (20 lines per mm) used with the image definition phase would suffice for the slitless spectrometer.

The image plane are of interest in the spectrometer test is greater in the vertical direction than the image definition measurement, but only a thin strip of the horizontal direction is required. Assuming a telescope stability of one arc min., a spectral image strip one min. wide should recover the spectral intensity points. See diagram:



A resolution of 20 lines per mm gives 1800 lines in vertical for the 30 min. field which is somewhat greater than the 8 cm required for the 0.025 Å spectral resolution. In the horizontal direction, 60 points would cover the 1 min. area allowed for drift. The total number of image points would be 60 x 1800 = 108,000. When digitized to 32 levels (5 bits), the data rate would be 540,000 bits per scan. The additional data required to give coordinates, time, operating modes, etc. probably requires a total rate of 600 kilobits per scan. For a scan rate of once per min., the data rate would then be 10 kilobits per sec. Allowing a speed up of 16 to 1 for playing back stored data over a ground station the playback data rate would be 160 Kbps.

After completion of the 2 Å to 10 Å measurement, a different grating will be installed for the spectral measurement in the 10 Å to 25 Å region.

Focusing Crystal Spectrometer Measurement - The grating and imaging electronics are moved out of the focal point by rotating the turret and the crystal spectrometer and pulse counter are positioned. Spectral lines of 0.1 Å resolution will be measured between 2 and 10 Å. These measurements will probably take from 10 to 150 min., each using the electron counter to record the intensity over some time interval. The data readout for these measurements could be controlled by the counter load as well as time interval. For high intensities, the counter will fill fast and will be read out before an overflow occurs. The time will be read out with each data value. The data rate for this measurement will probably vary between 50 bits and 50, 000 bits per sec. The rates are within the capability of the data system required for the image definition phase.

Polarimeter Measurements - The polarimeter will be moved into position to replace the spectrometers. The polarimeter measures polarization in four directions by using four separate pulse counters. The measurement time will probably be long and the data rate low. The data system provided to handle the other measurements will be capable of handling the data from these measurements.

4.2.6.2 Data Transmission. Using a data transmission rate of 260 kilobits per sec. as determined above and bit error probability (P_e^b) of one bit in 100 the transmitter power requirements are obtained.

Assume $P_e^b = 1 \times 10^{-2}$

$E/(N/B) = 4.3$ db signed energy from

$P_e^b = 1/2 \left[1 - \operatorname{erf} \sqrt{E/(N/B)} \right]$ or from curves

Data rate $R = \frac{1}{T} = 260$ kbps

$T = E/(N/B) - S/(N/B)$

$T = 1 \times 10^{-3} = 3.85 \times 10^{-6}$ sec.

$T = 55.85$ db

$S/(N/B) = 4.3 + 55.85 = 60$ db

Free-space loss (2000 mi. slant range)

$L_s = 32.45 + 20 \log 2200 + 20 \log 3200$

$32.45 + 66.84 + 70.10 = 169.39$ db

Transmitter power of 10 watts

3 watts in sidebands

$P_s = 10 \log 3000 = 34.77$ dbm

Antenna gain = 0 db

Receiver noise level $S_n = -186.6$ dbm/cps (40°K) -173.4 dbm/cycle KTB

Antenna $G_R = 44.0$ db

Miscellaneous loss 8 db

$$S/(N/B) = 35 + 44.00 + 175 - 8.00 = 76 \text{ dbm/cps}$$

$$P_s + G_R - KTB - Ls - Lm$$

Operating margin = 76 - 60 = 16 db

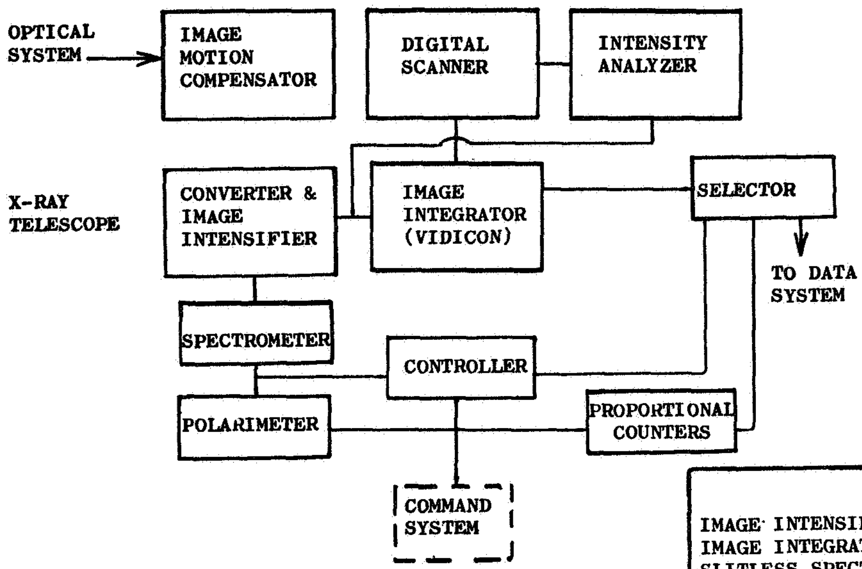
4.2.6.3 Data Storage. For a 260 mi. orbit, the average transmission time to a ground station is about 5 to 6 min. per orbit. Therefore, the data transmission rate must have about a 16 to 1 increase over the acquisition rate to transmit the data accumulated during a complete orbit. Previously, the transmission data rate was determined to be approximately 260 Kbps. A magnetic tape machine could be used to store the data during the uncovered part of orbit and playback when over a ground station. The total storage capacity is about 8 million bits. This could be handled by one tape machine, with a second machine as back-up.

Special Equipment - In compressing the amount of scan data, special equipment will be necessary. A sensing device is needed to determine the intensity of an area of the scan field. The device determines whether the point in the scan field should be retained as data or should be discarded. Special multiplying equipment may be necessary to rearrange the discontinuities in time sequence of retained data values into a continuous time multiplex for transmission.

Timing Equipment - The timing system will be used as the basic clock for the data handling system. It provides the synchronization signals, and will code and store time information. The system will time-correlate stored data. The time generator will be reset or up-dated by a command or time word sent on the command link.

4.2.7 X-Ray Telescope Scientific Instrumentation System. The telescope instrumentation is the primary mission equipment used to fulfill the scientific flight objectives. The required measurements and performance goals are discussed in Section 3.0 This section summarizes the equipment in the same format as the vehicle oriented subsystems. The simplified system block diagram and equipment physcial specifications are shown by Figure 4-37. All telescope instrumentation (except some auxiliary electronics packages) is mounted on the image capsule turret as each of the various measurements (imaging, spectrographic, and polarization) must be performed at the telescope focal plane.

4.2.7.1 Imaging. With the turret position aligned to the imaging station and the telescope pointed to an x-ray source within one arc min., x-ray photons are converted to visible photons by an x-ray image intensifier tube. This tube consists of an image screen, a phosphor screen coupled to a photocathode stage, and an electron intensifier. See Figure 4-38. The tube also includes beam forming and focusing elements. A beam positioning device will be used in the second stage for image motion compensation using converted error signals from drift components detected by the optical telescope system.



EQUIPMENT PARAMETERS

	IN^3	LBS	WATTS
IMAGE INTENSIFIER	2000	20	20
IMAGE INTEGRATOR(VIDICON)	2000	20	20
SLITLESS SPECTROMETER	-	30	-
CRYSTAL SPECTROMETER	20000	40	20
GRATING SPECTRO (SLIT)	29000	40	20
POLARIMETER	2000	50	20
SUN SHIELD	-	20	-
CONTROLLER	150	5	10
IMAGE MOTION COMPENSATOR	2000	20	20

Figure 4-37. Telescope Instrumentation System

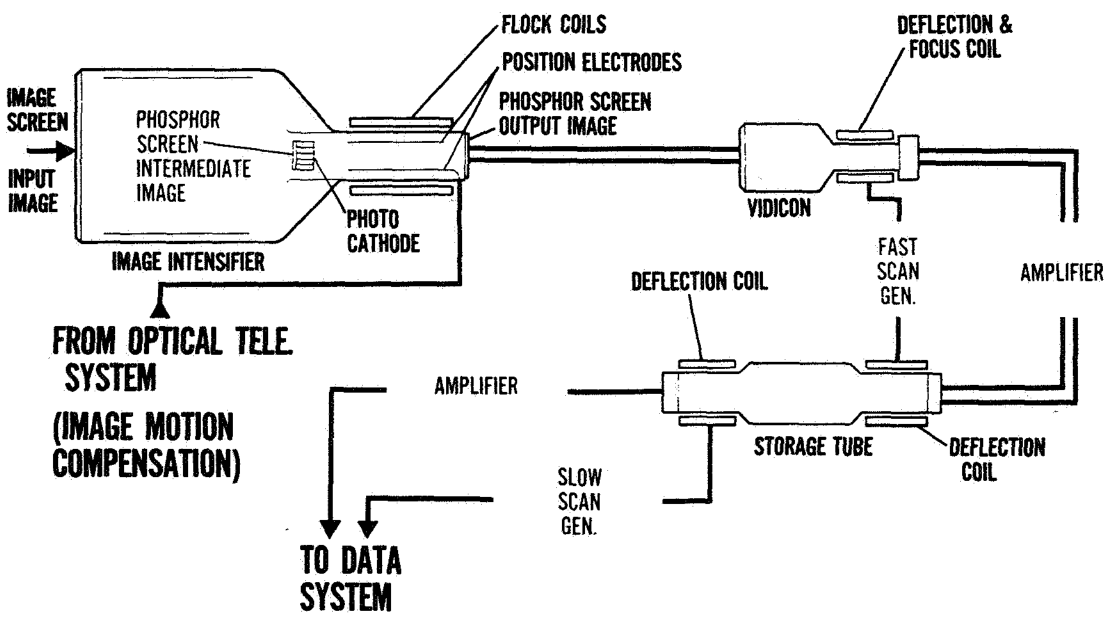


Figure 4-38. X-Ray Imaging Instrumentation

The output image of the image intensifier will be coupled to a vidicon storage tube since long integration times of 10 to 30 min. can not be accomplished with the image intensifier. The image will be read into the storage tube with a fast scan (one per sec.) and the readout will take place as necessary, or upon reaching maximum memory (one per min. to one per 30 min.). The scan rate may be controlled by an intensity monitor.

The actual source images of interest will be much less than the x-ray telescope field of view and the amount of data required to telemeter an image will, therefore, be reduced considerably by making the output image a small part of the telescope field or by telemetering only those elements that registered incoming photons.

4.2.7.2 Spectroscopy. The slitless spectrometer uses the same image intensifier as used for imaging. The image motion compensator may be turned off. The vidicon and/or storage tube will record the spectral line image. The total image would be 0.5 cm on each side of center or a total of 1 cm. The image size will be adjusted to fit the imaging screen.

Another turret position will move a focusing crystal spectrometer instrument into alignment with the x-ray telescope. A collimating slit must be placed at the focal point. One or more crystals will be used behind the slit to focus x-ray energy of a particular range on to detectors. Only one spectral line at a time can be registered for any one crystal. Different lines are obtained by rotating the crystal, or moving the detectors, or both.

Another turret position will move the concave grating spectrometer instrumentation into alignment with the x-ray telescope. A collimating slit is located at the focal point and the concave types diffraction grating would occupy a space relatively far behind the slit. The diffracted spectral lines will be registered on a long bent surface or on counters.

4.2.7.3 Polarimetry. The turret position containing the polarimeter instrumentation will have a collimating slit at the focal point; a scattering crystal, and perhaps proportional counters on the four lateral sides. A light element crystal will be used to scatter the incoming photons in differing directions depending on their polarization. This instrument can be largely frequency-independent and, therefore, one crystal can be used for the entire bandwidth to be investigated.

4.2.8 Attitude Control System. The attitude control system selected for the x-ray telescope utilizes inertia wheels for precision pointing control during source observations. Hydrogen peroxide jets unload the wheels and correct for aerodynamic drag velocity loss. There is a separate open loop N_2 system which will be used to stabilize the vehicle for docking should the primary ACS system fail. Pointing specifications require star trackers for the attitude sensors. The integrated sensor instrumentation and computer system (navigation) are discussed in Section 4.2.5. The attitude control system as designed satisfies the mission design criteria and performs all operational functions.

With respect to reliability, the weakest link is the inertia wheel system, further reliability studies may indicate the desirability of providing a redundant set.

The total weight of the 2 year attitude control system, including the actuator system and the navigation system, will be 570 lb, and the navigation system 191 lb.

4.2.8.1 Functions and Specifications. The vehicle attitude control pointing accuracy specifications are $\pm$ 0.5 arc min. and $\pm$ 1.0 arc sec. per sec. drift rate. One arc sec. per sec. is 4.83×10^{-6} radians per sec. The jitter specifications are $\pm$ 1.0 arc sec. Jitter refers to the vehicle non rigid characteristics, which include structural vibrations, load deformations, and thermal distortions.

Attitude control of a rigid body satellite is sensor limited, assuming no detrimental restrictions on actuator selection. Since the pointing specifications are well within the capability of state-of-the-art sensors, and the actuators are adequate, there is no problem in meeting the desired specifications. The ACS predesign involves many considerations and trade-offs.

Long term orbital reliability is a questionable area on any new system comprised of part or all new components. However, the risks of obtaining the long mission durations can be considerably reduced with man in the system to effect emergency procedures and make repairs. The proposed telescope design includes an emergency back-up ACS system which would be controlled visually by a rendezvousing CSM crew bypassing the telescope inertial reference system. The back-up ACS uses highly reliable cold N_2 gas for fuel and is capable of stabilizing the telescope to permit the CSM to dock in the event of a primary ACS failure which resulted in high vehicle tumbling rates. The back-up system is only required if both the H_2O_2 coarse jet system and the fine inertia wheel system fail, or the inertial reference system fails to provide the proper stabilization signals.

Structural vibrations are examined in Section 5.1.4 and it is shown that the jitter specifications are met.

The attitude control system maneuvers the telescope to a source, and locks on. It then limits cycles about the source, maintaining the desired tolerance of $\pm$ 0.5 arc min. while correcting for the environment disturbances. Aerodynamic drag causes significant orbital velocity losses at 260 n. mi., and the attitude control system is required to correct for this loss.

There is no exact specification on orbit trajectory. Therefore, the ACS does not have to provide velocity corrections (except for drag loss), small vehicle velocity changes due to attitude control operations will not degrade the mission.

The impulse functions which the actuation devices must provide include:

a. Environmental disturbances

b. Manuevering

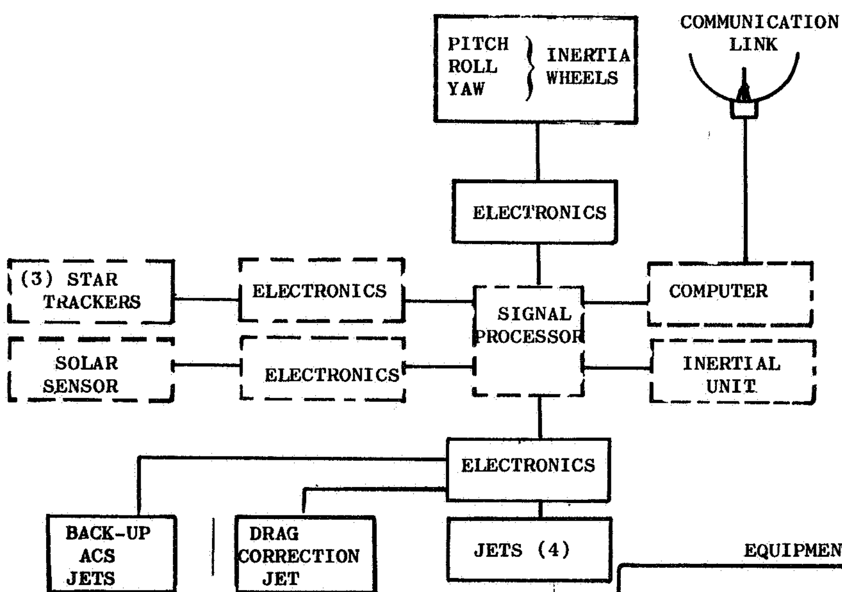
c. Limit cycle operation

d. Drag velocity loss correction

e. Back-up ACS for docking

These functions necessitate control of three axes rotation and capability of introducing a linear velocity change in the desired direction.

4.2.8.2 Selected System Concept. The selected system uses H_2O_2 jets and inertia wheels to provide coarse and fine attitude control. Drag velocity correction is provided by a single H_2O_2 jet. Star trackers are the sensors. A simple back-up ACS for docking uses N_2 propellant. The considerations involved in these selections are now discussed. See Figure 4-39.



EQUIPMENT PARAMETERS

	IN	LBS	WATTS
ATTITUDE CONTROL JETS		328	
" " WHEELS	5300	185	33.5
DRAG CORRECTION JETS	32		
BACK-UP ACS JETS	30		
MISC ELECTRONICS	1340	29	24

Figure 4-39. Attitude Control System

The ACS actuators are selected for the nominal mission, which has a source lock-on for one orbit, no thermal blanket, and a 1975 launch. Maximum vehicle surface area would be increased by 250% if a complete thermal blanket is introduced into the design. Atmospheric density varies with solar conditions, the density is minimum in the year 1975. Maximum air density at 260 n. mi. is twenty times greater than the minimum.

The attitude control actuator selection is dominated by two conflicting requirements. The close pointing specifications dictate small rotational torques, while the total mission impulse is large, due principally to the environmental gravity gradient torque and long mission time.

The usual micro thrusters would meet the pointing specifications, but the total propellant weight would be high. Electric propulsion microthrusters, which have low propellant consumption, were not considered because they are still in the development phase. The momentum storage devices are required to satisfy the pointing specifications and are state-of-the-art, additionally, they conserve propellant since a substantial portion of the rotational impulse requirements are cyclic. Jet reliability is improved by decreasing jet operation frequency. Therefore, momentum storage devices are selected to control the vehicle during source lock-on. Since the position specification is $\pm$ 0. 5 arc min., only one set is required.

The OAO position specification is on the order of arc sec. much more severe than the x-ray telescope; this required both coarse and fine wheels which are not necessary here.

Jets provide rotational control about all three axes, their primary use being momentum device unloading since the jets are not fired during source lock-on. There are four groups of three jets each, each group having a single propellant tank.

The vehicle will rotate to point the drag velocity thruster in the direction required for the correction.

The system sizing of the momentum storage devices were determined for the environment disturbances compensation, maneuvering, and limit cycle operation functions.

Significant environmental disturbances are gravity gradient and aerodynamic drag, the maximum torques are 0. 0403 and 0. 000343 ft-lbs., respectively.

In limit cycle operation, one must ensure that the momentum device can meet the specifications. A conservative estimate of the minimum momentum change is one-thousandth of the rated capacity. This is 0. 084 ft. -lb. -sec., and the corresponding velocity change is 3.28×10^{-6} radian per sec. This conservative estimate of velocity error is slightly less than the specification. The time elapsed in drifting through the position pointing specification at this velocity would be 89 sec. Therefore, the limit cycle operation will be within the pointing specifications.

For a one orbit inertial hold (i.e., source lock-on), the momentum storage devices must provide capacities of 36.6 ft-lb-sec in pitch and 76.3 ft-lb-sec in yaw due to gravity gradient. They must provide 0.6 ft-lb-sec in either pitch or yaw due to drag. Roll torque and momentum capacity will be smaller.

Maneuvering to new sources is assumed to be at a velocity of one radian per hour, requiring a momentum capacity of 7.2 ft-lb-sec in both pitch and yaw. The torque must be 0.12 ft-lb in order for the manuevering momentum exchange to occur in sixty sec.

The requirements of the momentum device is shown below. Since the vehicle is symmetric, a device may at any given time be controlling either the pitch channel or the yaw channel, the roll requirements are smaller. It would take 525 sec. for the capacity to be exchanged using the required torque. It may be desirable to specify a high torque to shorten this time.

Momentum Device, Pitch/Yaw

	Momentum ft-lb-sec		Torque lb-sec
	76.3	(gravity gradient)	0.0403
	0.6	(aerodynamic drag)	0.0003
	7.1	(maneuvering)	0.12
Total	84.0		0.16

These requirements are best supplied by inertia reaction wheels. Control moment gyros are more efficient for larger momentum requirements. The physical characteristics of the inertia wheels are listed below:

Inertia Wheel Characteristics

	Roll	Pitch or Yaw (each wheel)
Momentum, ft-lb-sec	10.	84.
Torque, ft-lb	0.02	0.16
Weight, lb.	25.	80.
Maximum Power, watts	35.	150
Average Power, watts	3.5	15.

The attitude control impulse requirements are determined as follows. Since the nominal vehicle operation is in inertial hold, the aerodynamic torque is largely cyclic. The pitch gravity gradient is cyclic and the yaw gravity gradient is a steady bias. It is assumed that the average yaw torque is 0.707 of maximum and 50% is included for roll, other motions, and contingencies. Thus, the jets must supply 4.44×10^5 ft-lb-sec each year, and noting that the arm is 19 ft., this is 2.34×10^4 lb-sec each year.

The aerodynamic drag results in an impulse of 2170 lb-sec each year. If uncorrected, the satellite velocity loss would be 38. 8 ft/sec each year and the orbit altitude would decrease more than 10 n. mi. Thus, for the nominal mission, about 10% of the impulse requirement is required for drag correction.

The weight of the system which meets these requirements is highly dependent on the propellant selection and the mission duration. Propellants evaluated were N_2 and H_2O_2.

The system weight vs. time is shown on Figure 4-40. The nominal SIB payload capacity is 6700 lb., and the satellite weight, excluding the attitude control system, is 2125 lb. Either propellant will result in a total satellite weight within the payload capability. However, the use of N_2 substantially decreases the weight available for additional payloads. The effects of resupply on over-all weight is observable from the Figure.

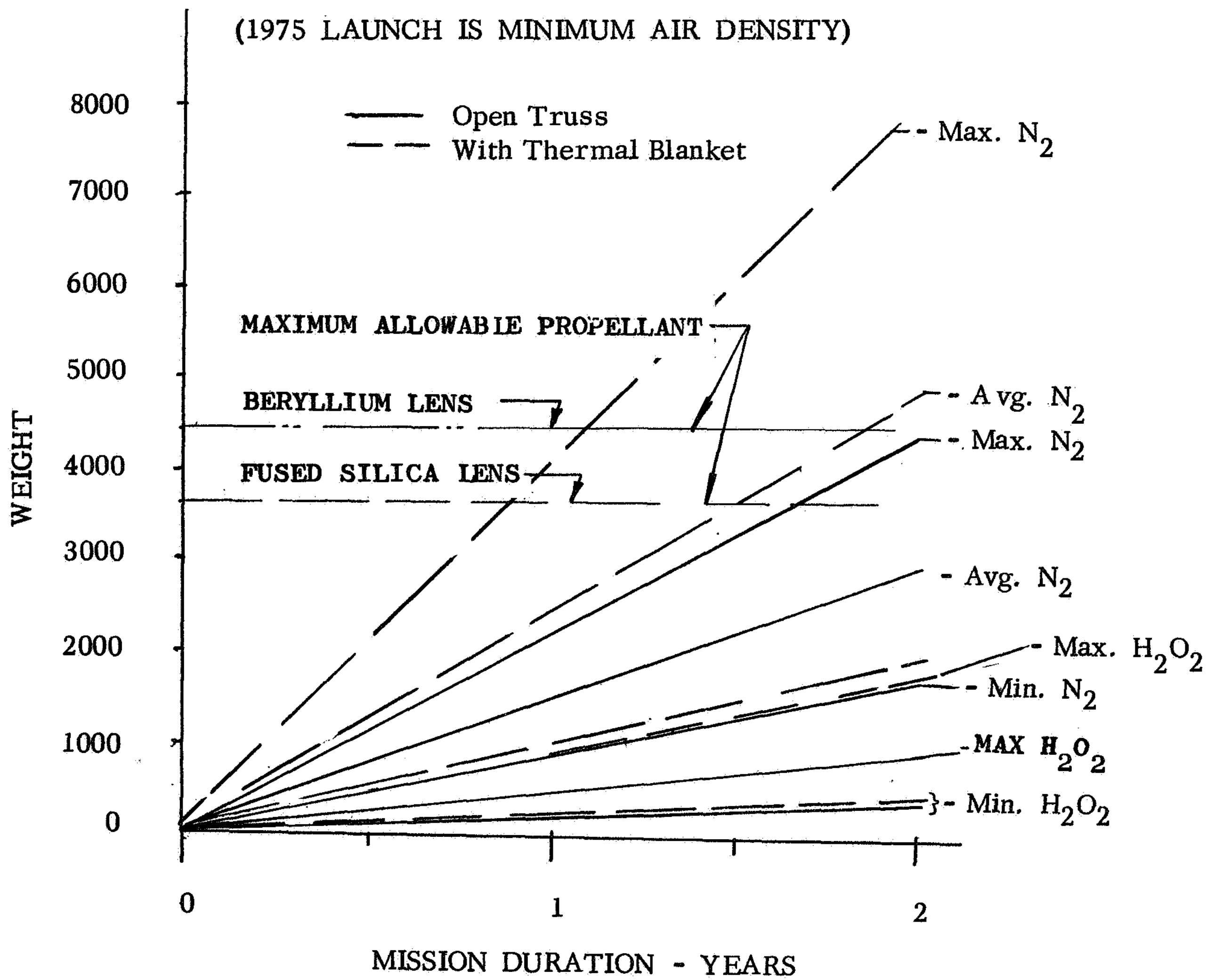


Figure 4-40. Variation Of Propellant Vs. Mission Duration

Propellant weight is increased for launch years past 1975, for which the air density will be higher, the weight advantages of H_2O_2 are more striking for later launches.

Hydrogen peroxide is selected as the propellant on the basis of these weight considerations. The thrust level of the jets is 0.5 lb. for the twelve used for attitude control and 10 lb. for the single jet used for velocity correction.

The jet reliability is subject to the following facts. Since valves are the dominant failure cause, each jet will have redundant valves and a selector to increase reliability and prevent catastrophic valve open failures. The reliability of this particular vehicle is greatly enhanced by the absence of restrictive velocity specifications. Consequently, if one of the two jets providing a couple fails, the vehicle can still be rotated and the associated velocity change is not catastrophic. The major use of the attitude control jets is wheel unloading. A jet pair will, therefore, be operated several hundred times a year. Failure of the drag correction jet would result in orbital altitude decrease. This jet will be operated perhaps ten times a year.

The back-up attitude control which stabilizes the vehicle for docking is a minimal system consisting of N_2 jets and an RF receiver. When the CSM approaches the tumbling vehicle, the crew will control the back-up system open loop, using visual observations.

The attitude control actuator system shown in the table below is for a one year mission. Section 5.3 lists the ACS weight for the nominal 2-year resupply mission.

Attitude Control Actuator System, One-Year Resupply

Function	Hardware	Weight(lb.)
Attitude Control	Twleve jets in four groups of three, H_2O_2, T = 0.5 lbs.	171
Attitude Control	Three inertia wheels, P/y 84 ft-lb-sec, 0.16 ft-lb. Roll 10 ft-lb-sec, 0.02 ft-lb.	185
Drag Velocity Correction	One jet, H_2O_2, T = 10 lb.	16
Back-up ACS for Docking	Six jets, N_2, T = 0.5 lb.	20
	Total Weight	392

Operational conditions other than nominal will be briefly discussed, noting their effects on the actuator selection. These conditions are: different launch years, use of thermal blanket, and source lock-on for five hours. Nominal one orbit lock-on is 90 minutes. The launch year and blanket conditions can increase aerodynamic drag and torque up to a factor of 50. The changes in wheel capacity will be an increase of up to 30 ft-lb-sec, but does not constitute a problem. The effects on drag velocity correction have already been discussed.

A cursory analysis of the effects of increasing vehicle lock-on time capability from the 90 minute design proposed to 5 hours resulted in the following conclusions (assuming source occultation is by proper source selection):

1. The pitch and yaw momentum capacity must be increased from 84 to 336 ft-lb-sec.

2. The roll momentum capacity must be increased from 10 to 40 lt-lb-sec.

3. The ACS system weight will increase approximately 205 lb.

4. The electrical power required will increase forcing a solar panel size/weight increase.

5. The pitch and yaw wheels increase in diameter from 21 in. to 28 in.

6. The image capsule diameter increased from 54 in. to 64 in. which increased shadowing and decreases astronaut access to interior.

7. The larger pitch and yaw momentum requirements may dictate a shift to control moment gyros for optimum performance.

SECTION 5

ANALYSIS

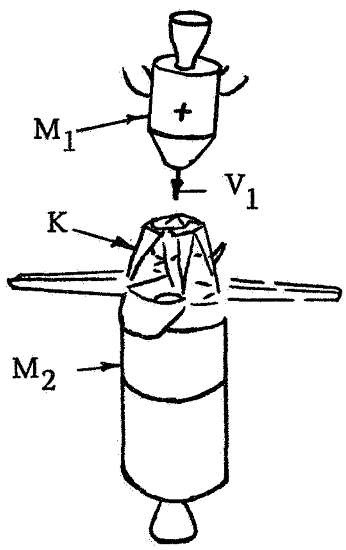
5.1 DYNAMICS. The dynamic behavior of the designed structure is satisfactory. The deflections of the lens and the structure assembly are within tolerance. Separation and deployment are simple controllable operations for this vehicle and no problems are anticipated.

The practice of using very conservative analytical techniques to bound magnitudes of dynamic response has been used extensively in this study. This approach is particularly desirable when configurations are only defined to the feasibility demonstration level. As the design details become more firmly established, it will be appropriate to use more exact analysis tools. Rigid body mass parameters, control system characteristics and elastic response and structural feedback should all be integrated into a detailed mathematical simulation. In support of this simulation and loads evaluation, it will be necessary to compute the three dimensional model properties.

Docking, separation and deployment loading can be determined using model response methods; however, the complexity of the docking probe-drogue mechanism interaction makes the ultimate use of a hybrid (i.e., docking hardware coupled to simulated vehicles) simulation desirable. Limits on pointing and tracking performance can be established from a combination of simulation and model response analysis.

5.1.1 DOCKING. The dynamic analysis of initial docking drogue support truss requires shock attenuation additions. The necessary change in stiffness is feasible and can be introduced in redesign. Orbital docking dynamics will then be satisfactory by a wide margin.

Initial docking occurs while the telescope is attached to the SLA and SIVB. Prior to contact, the CSM at mass M_1 has a velocity of V_1, relative to the telescope, SLA and SIVB which are mass M_2. From conservation of linear momentum, $M_1F_1 = (M_1 + M_2) V_2$. The energy change will be $\Delta E = 1/2 (M_1V_1^2) - 1/2 \left[(M_1 + M_2) V_2^2\right]$. Using the momentum relation:



$$\Delta E = (1/2) M_1V_1^2 - \frac{(1/2) M_1^2V_1^2}{M_1 + M_2} =$$

$$- (1/2) M_1V_1^2 \left(\frac{1}{1 + \frac{M_1}{M_2}}\right)$$

The minus sign indicates an energy loss which must ultimately be dissipated in the docked assemblage of structures.

The docking mechanism will absorb some part of this energy prior to latching, but the remainder will appear for a time as strain energy in the telescope truss structure. If the docking mechanism has reduced the initial velocity by some ΔV at latching, the equation for ΔE is still valid except that V_1 is now the original V_1 minus ΔV and the truss structure will be stressed by some load L_1 as an initial condition.

Assuming a spring constant for the structure of K, the change in stored energy due to a deflection X is:

$$PE = L_1 X + (1/2) KX^2 \text{ which is also equal to } \Delta E.$$

The total load will be $L_T = L_1 + KX$. Solving the energy expression for X and substituting into the expression for L_T yields:

$$L_T = \sqrt{L_1^2 + V_1^2 K \left(\frac{M_1}{1 + M_1/M_2}\right)}$$

This expression is used to obtain the parameter $V_1^2 K$ as a function of the initial load L_1. The limit load value, L_T, is 3000 lb., M_1 is 685 slugs, and M_2 is 800 slugs. The relationship is shown in Figure 5-1.

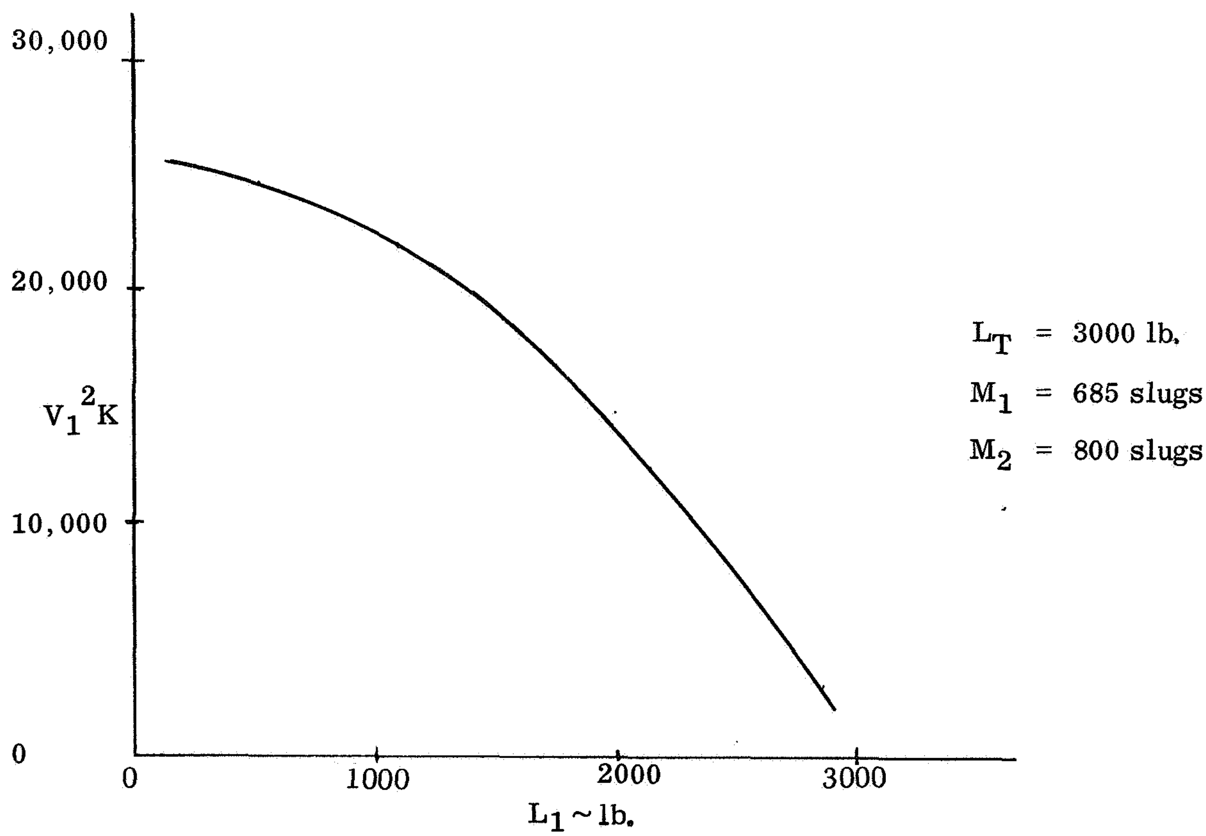


Figure 5-1. Parameter (V_1^2 K) Versus L_1, Initial Load

The maximum possible value of V_1 which would not cause damage is obtained. It will be associated with zero initial load, for which $V_1^2 K$ is 24,500. The value of K for the original beryllium structure as designed is 52.3×10^5 lb/ft, restricting V_1 to 0.069 ft/sec.

The velocity subsequent to latching is expected to be as high as 0.15 ft/sec, and the initial load will be greater than zero. Therefore, the current design will fail in initial docking.

Damage will be prevented by reducing the structural stiffness to more efficiently absorb the energy. Determination of the redesign is obtained from the Figure using conditions of 0.15 ft/sec for V_1 and 2000 lb for L_1. This results in $K = 6.0 \times 10^5$ lb per ft, which is a factor of approximately 2.5 below the current titanium design.

This reduction in stiffness could be achieved via a general reduction in the satellite stiffness, if such a change did not compromise other areas of performance. The area which is most likely to be degraded to unacceptable level is dynamic vibration with respect to the jitter specification. This is investigated in Section 5.1.4.2, and a conservative estimate yielded a safety factor of four. The proposed approach is to provide local reductions in stiffness at the docking drogue support truss. It is clearly feasible to obtain the necessary changes.

The orbital docking loads will be considerably less than for initial docking because the satellite is not restrained by the SLA and SIVB. Consequently, once the initial docking dynamics are acceptable, orbital docking loads may be neglected.

5.1.2 Separation. The satellite will be separated from the SIVB by the CSM. This procedure provides positive human control for the separation operation, which should eliminate the possibility of problems. Therefore, satellite separation is not an operation posing design feasibility questions.

5.1.3 Deployment. Deployment of this vehicle entails two simple operations. The solar panel arrays and ACS support boom assemblies are extended and the vehicle is expanded to its full length. No problems are anticipated.

5.1.4 Structural Dynamics. The dynamic response of the lens and of the structure assembly are of primary concern in operating within the design criteria. Examination of the operation uses simple conservative analyses to estimate upper bounds on the dynamic motion. These estimated motions are smaller than the requirements.

The solar panels and the ACS support boom assembly will be required to have enough strength to preclude coupling problems. Since the jets will not be fired during source observations, no difficulties are anticipated.

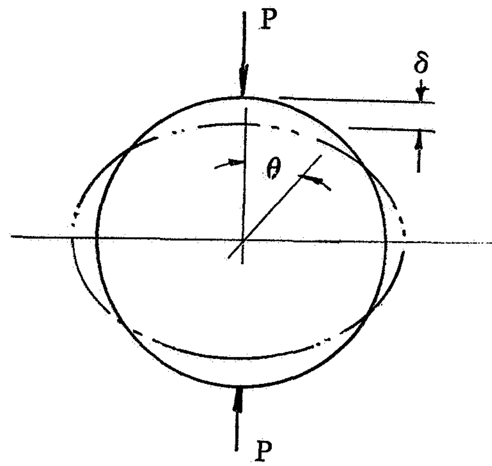
5.1.4.1 Lens. To bound the vibration response of the x-ray lens elements to control perturbations, simplifying, but conservative assumptions, have been made:

a. The element is, for small vibration amplitudes, unrestrained by the edge support system.

b. Any loads are reacted into the element in a manner which will excite maximum response.

The outermost cylinder of the lens assembly was selected for analysis since it will have the largest response to an impulsive disturbance. A close approximation to the first radial vibration mode of a cylinder is the deformation shape assumed under static point loading: The deformation is distributed around the cylinder according to $\delta \cos 2\theta$.

From shell theory: $\delta = \dfrac{.15Pa^3}{4DL}$



$$D = \frac{Eh^3}{12\,(1-\gamma^2)}$$

P = load lb.

a = radius = 15.0 in

1 = cylinder half length = 15.0 in.

E = Young's Modulus = 42.0×10^6 lb/in^2

h = thickness = 0.18 in.

γ = Poisson's ratio

The force P moves through the distance 2δ. Thence, the equivalent spring constant at the point of load application is $K = P/2\delta$ or 667 lb/in.

For a mode of the prescribed shape the equivalent mass M (considering a unit deflection at the load point) is 0.5 of the actual mass. From the equivalent or generalized spring rate and mass the frequency of vibration is $f = 1/2\pi\sqrt{K/M} = 38$ cps.

If a step input of load is assumed, the total response of the element will be $\delta_T = 2.0\delta$ or $2.0\ P/K$. The load P is due to inertia reaction to the lateral component of acceleration experienced as the satellite undergoes rotational acceleration. (i.e., $\delta_T = \ddot{Y}$ mass of element/K, or

$$\frac{17 \text{ lb.}\ddot{Y}}{2 \times 386.4 \times 667 \text{ lb/in.}} = \delta_T).$$

Since it is required that $\delta_T < 5.0 \times 10^{-6}$ it follows that $\ddot{Y} < .075$ in/sec^2 or that with an arm to the cg of 260 in., $\ddot{\alpha} < 2.9 \times 10^{-4}$ rad/sec^2.

Using a moment of inertia of 25.6×10^3 slug ft^2 the step torque permitted is less than 88 in. lb., however, the maximum inertia wheel torque will be 1.8 in. lb.

It is concluded then that vibratory response of the x-ray lens elements will be well below the allowable 5.0×10^{-6} inches.

5.1.4.2 Structure Assembly. As with the lens element analysis the procedure followed is intended to bound the response rather than to define it in detail by a more rigorous mathematical model. The anticipated vibration mode shape and frequency is the free-free satellite mode. A cantilever mode will have lower natural frequency and display larger responses to disturbances, thus, yielding conservative results.

The structure was assumed cantilevered with the heavier lens capsule end mass free and the truss structure mass distributed uniformly along the satellite length, this is a valid assumption since the remaining mass is almost entirely located within the image capsule (fixed end).

From a detailed structural analysis of the beryllium satellite frame, it was determined that the cantilevered truss had a bending stiffness, as measured at its end, of 2.34 x 10^4 lb/ft. A generalized mass of 24.3 slugs was computed for a unit deflection at the free end, giving rise to a frequency of 15.6 cps. The free-free configuration will have a frequency of 20 to 25 cps. This relative proximity of the satellite first mode frequency and the lens element frequency (38 cps) implies that some amplification factor, greater than 1, but less than 2, should be used in computing lens element response to control torques. Two disturbance sources have been considered:

a. Two (0.5) lb. reaction motors firing as a pair to create a pure couple.

b. Inertia wheels torquing at a maximum of 0.16 ft-lb.

In the first case the motors act on a 19.0 ft. arm to produce a torque of 2 x 0.5 x 19 = 19 ft-lb. This level of torque will induce an end to end vibratory angular deflection of $1.03 \times 10^{-4\circ}$ (using an amplification factor of 2.0). The comparable deflection for the second case is $8.67 \times 10^{-7\circ}$. These results are 0.371 and 0.00312 arc seconds respectively.

Comparisons of these values with the jitter specification requiring amplitudes less than 1 arc sec. clearly indicate the dynamic response of the structure is satisfactory.

5.2 THERMODYNAMICS. The influence of the orbital thermal environment was investigated for the major assemblies of the x-ray telescope; structure, lens and image capsule. Temperature transients resulting from passage through the earth's shadow and spacecraft re-orientation affects the design of each of these assemblies, while non-uniform heating primarily affects the structure and lens designs.

Orbital Environment. Transient heating in the 260 n. mi. altitude circular orbit primarily depends on the percent of exposure to the sun. Initial studies were made on the basis of a 50° inclination which was later changed to 28.5° inclination. The lower inclination angle results in a maximum sun exposure of about 81%. This change would effect only the image capsule analysis and would result in reduced louver area. This percent is a function of the angle between the earth-sun vector and orbit plane. For the 50° inclination, this angle, η, can be between 0° and 73.5° (inclination angle plus angle between earth's equator and ecliptic plane) depending on the time of year and the right ascension of the ascending node. The minimum percent sun occurs at 0° angle between earth-sun vector and orbit plane and is computed to be about 62%.

The minimum angle η at which 100% sun exposure occurs is computed to be 68.5° which is less than the maximum possible of 73.5°. Thus, the 100% sun condition can occur for 50^{o} inclination at least twice a year and sometimes 3 or 4 times a year, depending on the right ascension of the ascending node.

Yearly average heating rates can be obtained from the above orbital conditions and are: solar, Q_S = 442.4 BTU/ft.^{2} hr.; albedo, Q_A = 177 BTU/ft.^{2} hr. (based on a reflectance of 0.4); and earth thermal, Q_T = 66.4 BTU/ft.^{2} hr. In early July, solar and albedo heat rates are about 3.5% less, and in early January about 3.3% more than the average values.

The heat balance on the spacecraft depends on both the geometric factor between the vehicle surface and the radiant heat source, and on the thermal radiation properties of the surface. In the following analysis of the thermal behavior of the major spacecraft assemblies, limiting cases have been studied to establish extremes of orbital environment to which they will be exposed.

5.2.1 Structure.

Tubular Structure. The spacecraft structure is composed of tubular elements joined to provide a framework to which the other assemblies may be attached. Radiation heating of these members will cause both a temperature gradient to be established across the diameter of the tube and an increase in the average tube temperature. The gradient results in non-uniform expansion of the leading face and the trailing face and, thus, thermal distortion, i.e. bending. An increase in average temperature results in a finite lengthening of the members which causes deflection of the structure as a unit.

Temperature gradients and temperatures can be controlled by the choice of materials and the choice of thermal radiation properties for the surface. They are calculated by

$$\Delta T_{max} = \frac{D^2}{4tk} \alpha_S J_S$$

where

ΔT_{max} = diametrical temperature gradient

D = tube diameter

t = material thickness

k = material thermal conductivity

α_S = surface solar absorptance

J_S = solar constant, 442.4 BTU/ft.^{2} hr.

This formula is based on the tube axis at 90^{o} to the spacecraft-sun vector and that the albedo and earth thermal can be neglected. In most cases consideration of albedo and earth thermal radiation would be in opposition to the solar heating and result in lower gradients. Only beyond 90^{o} past the sub-solar point is this not true.

However, in this region the albedo becomes zero and the thermal is small, growing to a maximum of about 20 BTU/ft.^{2} hr. before the spacecraft enters the umbra.

Temperature gradients were calculated for aluminum and beryllium tubes varying the wall thickness and the surface treatment. Figures 5-2 and 5-3 show the results. Silver plating reduces the gradient by about three times contrasted to white paint. Decreasing the tube diameter from 3.5 in. to 2.0 in. yields another factor of about 3 reduction in gradient. Since the gradient is an inverse function of the wall thickness, diminishing benefits are quickly reached.

Typical deflections resulting from the thermal gradients were computed for a 2 in. diameter tube, for various lengths assuming a wall thickness of 0.02 inc. The governing equation is

$$\delta = \frac{\epsilon \ell^2 \Delta T}{2D}$$

where

δ = tip deflection

ϵ = coefficient of thermal expansion

ℓ = member length

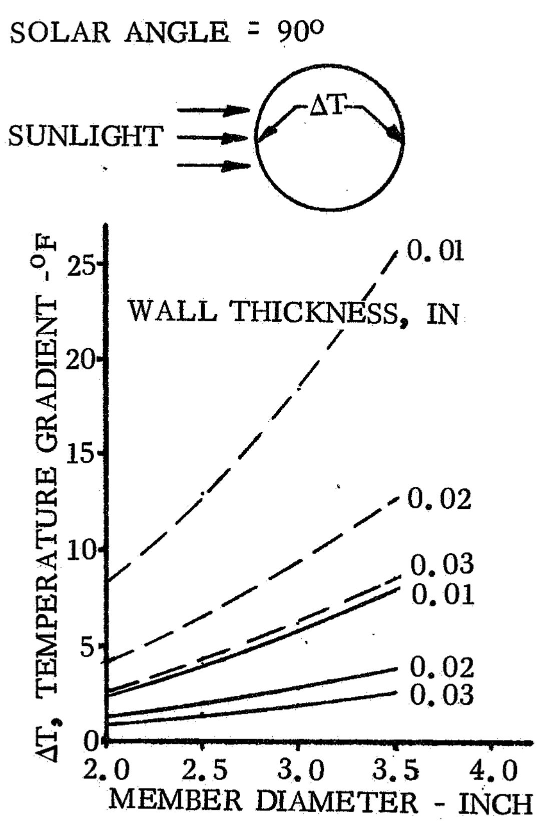


Figure 5-2. Temperature Gradient in Aluminum Members

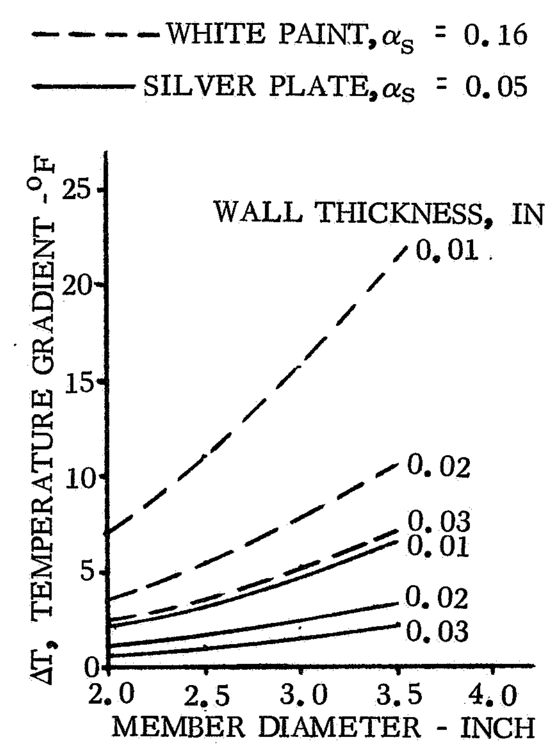


Figure 5-3. Temperature Gradient in Beryllium Members

This formula assumes that the member is unrestrained and, thus, able to deflect freely. The unrestrained values are approximately 5 times greater than actual since the truss members have fixed ends. Figure 5-4 gives the computed values for aluminum and beryllium solid tubes and for copper-invar wire mesh tube having no seam and the properties shown. Although, about one order of magnitude improvement results when the mesh is used, it cannot be considered structurally adequate for this application until detailed analyses of EVA loads are evaluated for mesh tubes. The variation of deflection with member diameter is given in Figure 5-5.

Since the structural members are not aligned each one will be oriented at a different attitude with respect to the sun. The effect of other orientations on the temperature gradient was evaluated by assuming that as the solar angle decreased from 90° the projected area exposed to earth thermal radiation increased until the full value of 66 BTU/ft.^{2} hr. was reached when the solar angle became 0°. Furthermore, it was assumed that these fluxes were both incident on the same face of the tube and additive; this could occur at an orbital position 90° away from the subsolar point. The gradient variation computed is shown in Figure 5-6. Corresponding average tube temperatures are given in Figure 5-7 and were calculated by assuming that the actual temperature gradient was small. The apparent benefit of white paint in minimizing both the gradient variation and temperature variation is a consequence of this particular orbital position and tube orientation. In the earth's shadow the white paint causes larger gradients and a greater variation with angle than would the silver plate. At the subsolar point the white paint could again provide benefits in minimizing the temperature gradient variation. A transient heating analysis incorporating a variety of orientations with respect to the sun and earth would be necessary to compare the performance of different surface treatments and provide criteria for thermal radiation property selection.

Time required to warm-up a tubular member upon emerging from the earth's shadow and time to cool-down upon entering the shadow were calculated. A 2.5 in. diameter beryllium tube painted white was used in the analysis. Temperature just prior to emerging from the shadow was computed to be 321°R. The orientation was taken to be a solar angle of 90° when exposed to the sunlight, thus resulting in the largest heating rates and the longest warm-up time. For this condition the calculated time is 7.37 min.

The reverse of this condition was taken for calculation of the cool-down time. For this case the initial temperature was 470°R upon entering the earth's shadow. The continually reducing radiation heat loss with reduced tube temperature results in a long cool-down time of about 24.3 min. This is shorter than the maximum shadow time of about 33 min., and complete cool-down can result.

Shell Structure. An alternate structure design consisting of closed conical shells was studied to evaluate the maximum temperature gradient across the diameter. As with the individual tubes, the maximum gradient would occur when the spacecraft axis was 90° to the sun-spacecraft vector and 0° to the earth-spacecraft vector.

For this analysis, the opposing surfaces of the shell were idealized as finite flat plates having a radiation interchange factor of 0.2. Radiation heat transfer is the only significant mode due to the long high resistance conduction path through the

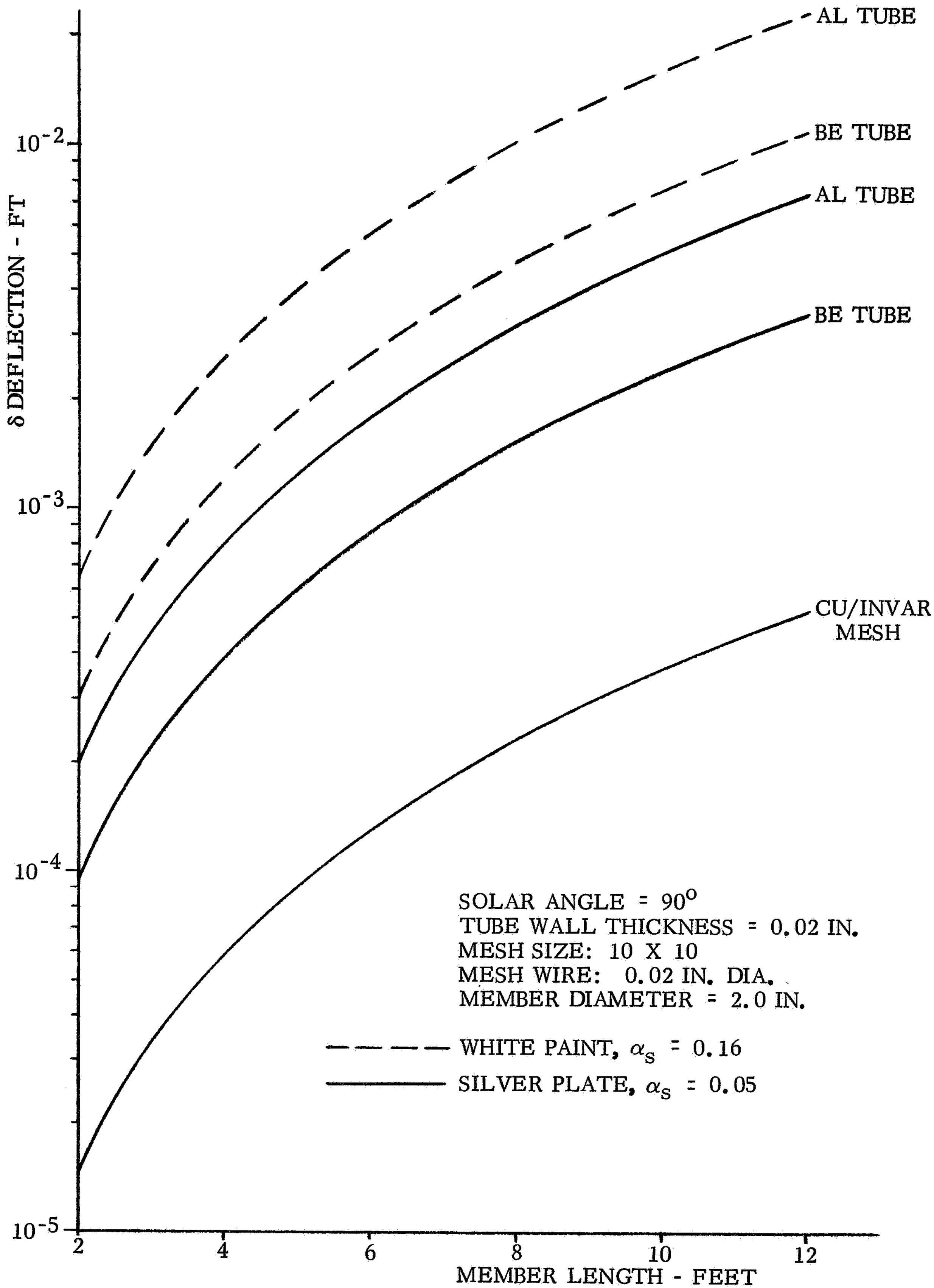


Figure 5-4. Deflection of Unrestrained Members

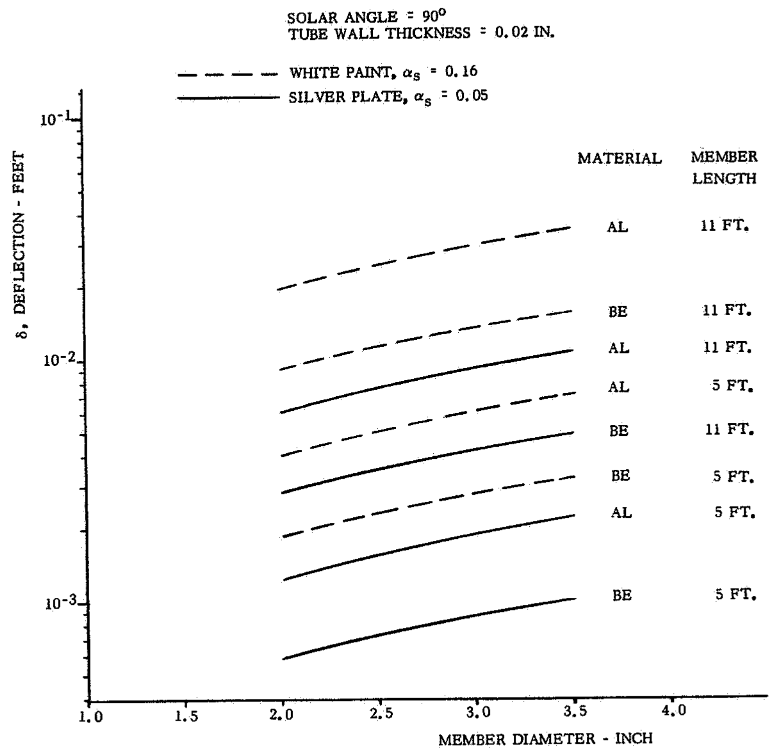


Figure 5-5. Effect of Diameter on Deflection of Solid Tube Unrestrained Members

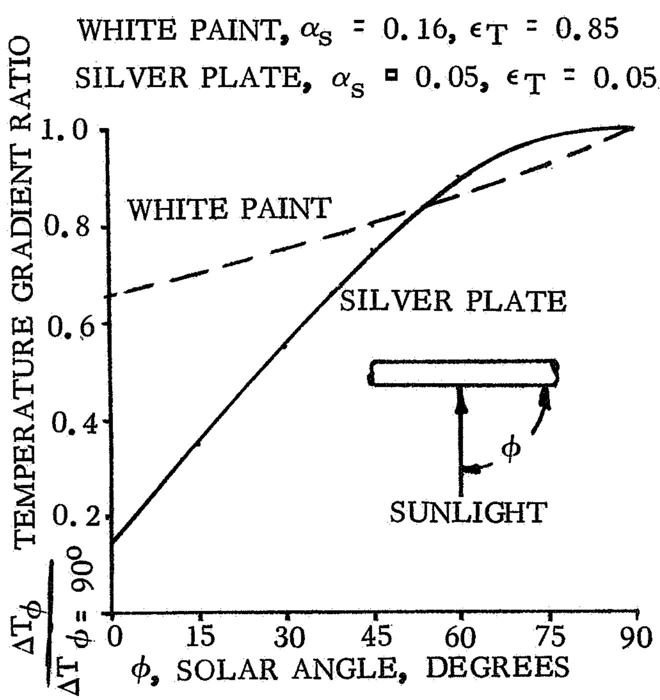


Figure 5-6. Temperature Gradient With Solar Angle

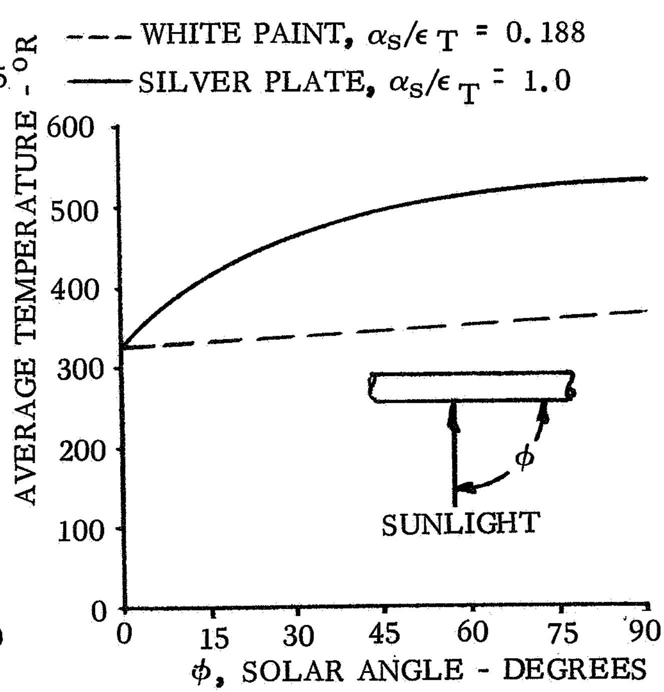


Figure 5-7. Average Tube Temperature Variation With Solar Angle.

shell. The inside surfaces were considered painted black having an emittance of 0.97. Two different surface treatments were considered for the exterior: polished aluminum (α_S = 0.245 and ϵ = 0.05) and white paint (α_S = 0.186 and ϵ = 0.92). Resulting steady state temperatures and temperature gradients are shown below.

Shell Structure Temperatures

Solar Angle = 90°

Black Paint Interior

Surface Treatment	Sunlight Side	Shade Side	ΔT
Polished Aluminum	913°R	862°R	51°R
White Paint	456°R	296°R	160°R

5.2.2 Lens. The lens assembly is sensitive to thermal distortion effecting segment radius growth, circularity, and concentricity. Each of these possible effects derive from temperature gradients along or around the lens segment or temperature excursions which cause temperature differences between segments. Thermal control of the lens assembly is required to do the following:

a. Reduce temperature gradients.

b. Reduce temperature excursions between illuminated and dark portions of the shadowed orbit.

c. Maintain fairly constant temperature as orbital heating rates change from 62% sun to 100% sun condition..

Items a. and b. can be accomplished with external insulation which is in turn covered by a shield having appropriate surface treatment and thermal capacity. In addition, the internal components must be thermally coupled by affording high conduction and radiation heat transfer paths. High conductivity material (copper, aluminum, tungsten, beryllium) should be used to minimize temperature gradients. An estimate of insulation requirements was made by calculating the relative heat transfer through the insulation compared to heat transfer within an internal cylindrical aluminum structural housing. A multi-layered aluminized mylar insulation, Superfloc, developed by Convair for space applications was considered due to its extremely low apparent thermal conductivity, $k = 2.6 \times 10^{-5}$ BTU/ft.-hr-°R.

The heat transfer comparison was made by calculating the thermal resistance to heat being admitted through the insulation and a similar thermal resistance to heat transfer along the aluminum structure. For side heating, the projected area of the insulation was used to calculate its thermal resistance while the thickness was varied. This was compared to circumferential conduction within the structure using one-fourth the circumference as the conduction path while varying the thickness. For end heating, similar calculations were made using the lens cylinder end area and the complete cylinder length.

The insulation thermal resistance was set at 100 times that of the structure thermal resistance, and corresponding insulation and structure thicknesses were computed.

Figure 5-8 shows the results. This data means that for each 100 hundred degree temperature gradient through the insulation, for a given thickness, there will be only 1^{o} temperature gradient through the structure, at the corresponding thickness.

The maximum anticipated ΔT which may be imposed on the insulation can be obtained by comparing the maximum steady state surface temperatures when in the sun to those in the shade. These are given in below for both white paint and polished aluminum.

	Maximum Steady State Surface Temperatures		
	In Sun	In Shade	ΔT
White Paint	$470^{o}R$	$292^{o}R$	$178^{o}R$
Aluminum	$1060^{o}R$	$659^{o}R$	$400^{o}R$

This indicates the advantage of using low α_S/ϵ coatings, such as white paint, to minimize the temperature excursions. Applying this gradient, $\Delta T = 178^{o}R$, with Figure 5-8 the insulation thickness necessary to limit gradients within the structure to 1^{o} can be computed. If the structure were 0.02 thick, then about 1.9 in. would be required for the side and about 0.5 in. on the ends.

The use of Superfloc insulation may also be effective in controlling the absolute temperature of the lens assembly within close limits. Initial calculations show that the total heat loss through the insulation while in the shade is less than 1 BTU/hr., based on the insulation thicknesses, structure thickness and temperature gradients given above. This could result in only a few degrees temperature change of the lens assembly. A detailed thermal analysis employing transient calculation techniques, internal radiation/conduction heat transfer and including losses through insulation penetrations and structure attachment points needs to be done before a completely passive thermal control system can be evaluated. If resulting temperature extremes are too great using just insulation and surface coatings, then active control systems such as heaters or louvers would have to be considered.

5.2.3 Image Capsule. Incident heating throughout the orbit was evaluated for 4.5 ft. diameter, 4.0 ft. length image capsule cylinder. This was done by applying the average yearly heat rates of solar, albedo and earth thermal in combination with geometric factors for the convex portion and the ends. In addition, it was assumed that no significant shadowing would occur from other elements of the spacecraft. Both the hottest case,

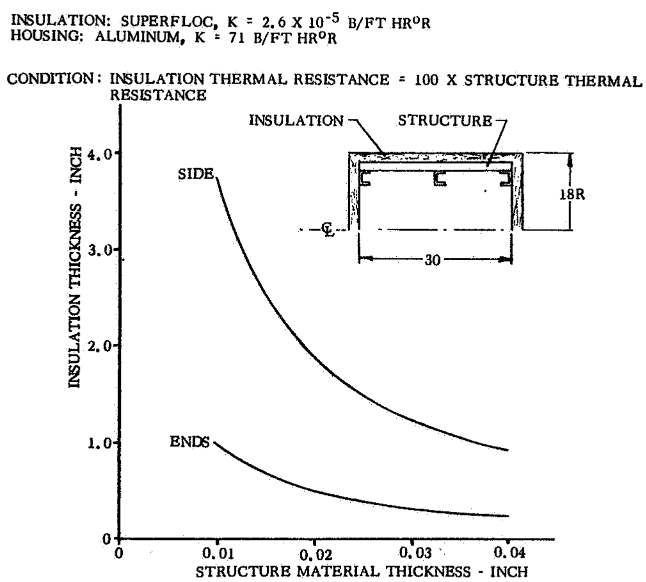


Figure 5-8. Insulation of Lens Assembly

100% sun, and the coldest case, maximum shadow, were calculated and are shown in Figure 5-9 and 5-10, respectively.

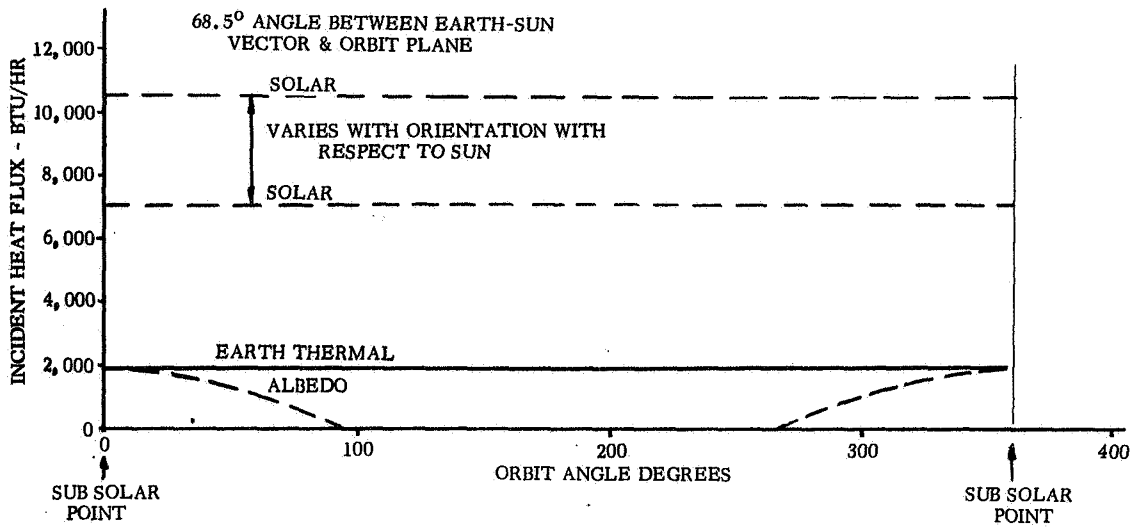


Figure 5-9. Incident Heat Flux to Image Capsule - 100% Sun

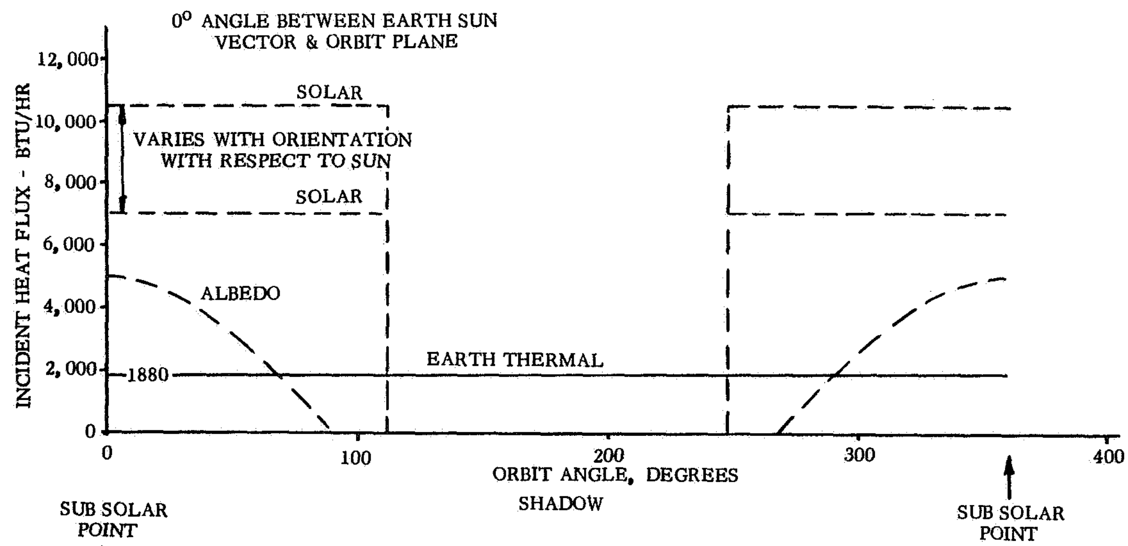


Figure 5-10. Incident Heat Flux to Image Capsule - Maximum Shadow

Variation of the total incident albedo and thermal radiation with orientation was for a cylinder of L/D ≃ 0.9 at 260 n. mi. altitude. An appreciable orientation effect was obtained for the total incident solar radiation as shown. Orbital average heating conditions were calculated for these cases and are given below. Heating for the maximum sun exposure of 81% with a 28.5° inclination will be about 25% lower then the 100% exposure hot case.

Average Orbital Heating of Image Capsule	Hottest Case	Coldest Case
Incident Solar	10,500 BTU/hr.	4,360 BTU/hr.
Incident Thermal	1,880	1,880
Incident Albedo	606	1,570

By using the average orbital heating, the average image capsule temperature was calculated for a range of α_S/ϵ values and is shown in Figure 5-11. These results are based on no-power dissipation since the power level was not precisely known. Internal power dissipation slightly increases the temperature level and reduces the temperature excursions. Typical examples are shown below.

Image Capsule Average Temperature With Power Dissipated

	Hot Case		Cold Case	
	100 Watts	175 Watts	100 Watts	175 Watts
α_S/ϵ	0.75	0.75	1.07	1.07
T_{ave}	63°F	72°F	45°F	21°F
ΔT due to power	13°F	22°F	59°F	35°F

To limit temperature excursions during passage through the earth's shadow, surface coatings with a very low emittance, ϵ, are required. Polished aluminum has a low ϵ (0.045) but the α_S/ϵ ratio is too high (5.45) to minimize the temperature excursions between the hot case and the cold case. A lower value of α_S/ϵ ratio may be obtained with only a small variation in the percent of polished aluminum used, as shown in Figure 5-12.

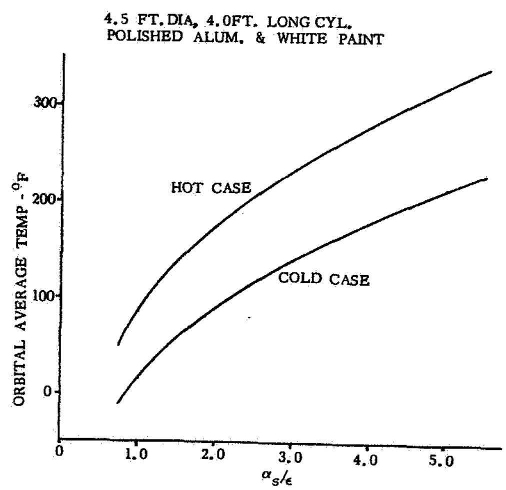


Figure 5-11. Average Orbital Temp.

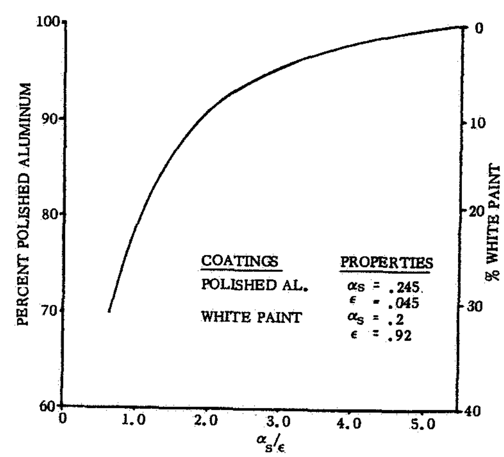


Figure 5-12. Surface Coatings Radiation Properties

Maintenance of internal equipment case temperatures of $70^{\circ}F \pm 5^{\circ}$ may be difficult to achieve depending on individual component sizes and power dissipations. The $\pm 5^{\circ}F$ is not presently required by the equipment considered. As previously shown the average temperatures could be designed for $70^{\circ}F \pm 45^{\circ}$ with no active thermal control system. An α_S/ϵ of about 1.10 would be required for this case and could be obtained with about 80% polished aluminum and 20% white paint.

An active system using heaters does not appear practical for this application. With this system, the hot, 100% sun, case would require an α_S/ϵ of about 0.95 to limit the average orbital temperature to $75^{\circ}F$. During the cold case, heater power on the order of about 350 watts would be necessary to maintain the average orbital temperature at the operating range.

The use of louvers to control temperatures appears feasible. An α_S/ϵ of about 1.6 would maintain an orbital average temperature of about $65^{\circ}F$ during the cold case. This would be obtained with 88% polished aluminum on the image capsule surface. Additional heat rejection is required for the hot case and an α_S/ϵ ratio of about 0.95 is again necessary. This can be obtained with a louver coverage of about 30%, i.e. 27 sq. ft. The louver area could be decreased if non-operating temperatures were allowed to exceed $75^{\circ}F$ and if operation of the spacecraft were limited to orbits having a maximum of 81% sun time, for example, the reduction would be about 25%.

The effect of the image capsule structure consisting of a double skin shell, 0.75 in. overall thickness, on the thermal balance was investigated. This design has foam between the skins for micrometeroid protection. Internal power dissipation with consideration of component duty cycle requires that the internal equipment be thermally connected to the external surface. The use of foam between the skins would not interfere with the thermal performance as long as the stiffeners joining the inner and outer skins provide good conduction paths. About 2.5×10^{-3} ft.^{2} of conduction area would be required for each sq. ft. of surface area. Bonding compounds should be used to limit contact resistance at the joints between the stiffeners and the skins.

Actual equipment operating temperatures depend on a detailed radiation/conduction analysis of the heat flow between internal components and the image capsule structure. This type of analysis would employ transient techniques and depend on specific information as to component arrangement, heat output, surface conditioning, mounting, etc., and could not be realistically modeled at this time.

5.3 MASS PROPERTIES. This section summarizes the mass properties of the x-ray telescope spacecraft design with titanium primary structure and a 30 in. maximum lens diameter. The total vehicle weight with a lightweight beryllium mirror assembly is 2700 lb. This weight is expected to increase to 2570 lb. with a fused silica or low coefficient glass mirror assembly. The nature of this study has not permitted an optimization of materials, therefore, the variation is shown to illustrate the effect of the lens material selection on the total vehicle. The total weights include a contingency and growth factor of 10% as shown in Table 5-1.

The most recent Saturn IB payload information received from NASA/MSFC in February 1967 indicates that for a 260 n. mi. altitude and a 28.5° inclination, using a Hohmann transfer trajectory, the increase in payload capability is approximately 4000 lb. Therefore, without a Hohmann transfer, this design could be placed in the desired orbit with a 14 day CSM; with a Hohmann transfer, a CSM of more than 45 days and less than 90 days could be used.

Table 5-1. Weight Summary

Nominal 30 in. diameter
3 mirror system, F/D = 12

	UNIT WEIGHT		SUBTOTALS	
STRUCTURE & MECHANISMS			661	
Truss	297			
Solar Panel Support Structure	33			
Lens Support Structure	16			
Lens Shade	16			
Lens Aligning Mechanism	83			
Lens Capsule Shell	41			
Image Capsule Shell	75			
Docking Structure (LEM)	50			
Deployment Mechanism	50			
REFLECTING LENS ASSEMBLY			292	(1085)
Mirror Assembly	162	(955)		
Optical Telescope/Optical Orthicon & Imc Generating Element	100			
Slitless Spectrometer Grating & Mechanism	30			
ATTITUDE CONTROL (2 Year Mission)			575	
Reaction Wheels	185			
Primary Jet System	328			
Drag Velocity Correction System	32			
Back-up ACS Docking System	30			
COMMAND/CONTROL			11.5	
Command Receivers	2			
Signal Combiner	1			
PCM Decoder	2			
Command Programmer	5			
Tone Decoder	1.5			
TELEMETRY			44.0	
Signal Conditioners	2			
Analog to Digital Converter	1			
Input Selector	1			
Digital Logic Assembler	5			
PCM Multiplexer	5			
Time Generator	2			
Tape Recorder w/series to parrallel, parallel to series Converter	10			
Pre-modulation Processor	2			
Transmitter and Antenna	3			
Buffer Storage	2.5			
Pre-modulation Filters	.5			
Harnesses and Brackets	10			
NAVIGATION			191	
Star Trackers (3)	30			
Signal Processor	18			
Phasolver	7.5			
Signal Processor (Pitch & Yaw)	10			
Sensor Processor (Roll)	6			
Wheel & Jet Controller (Pitch & Yaw)	12.5			
Wheel & Jet Controller (Roll)	11			
Computer	35			
Digitizer Logic Unit	30			
Inertial Unit	30			
Solar Sensor	1.0			

Table 5-1. Weight Summary, Contd.

Nominal 30 in. diameter
3 mirror system, F/D = 12

	UNIT WEIGHT	SUBTOTALS
POWER SYSTEM		304
Solar Arrays	173	
Batteries	108	
Battery Charger & Regulator	8	
Main Power Distributor	5	
Auxiliary Power Distributor	3	
Voltage Booster	5	
Voltage Limiter	1	
Low Voltage Sensor	1	
LENS ALIGNMENT ERROR SENSING		15.5
He Laser & Electronics	5	
Beam Splitter	1	
Sensor	4	
Sensor Electronics	5	
Sensor Shroud Tube	.5	
SCIENTIFIC INSTRUMENTATION		
Instrument Package A (2 provided)	120	
Image Intensifier		
Image Motion Compensator		
X-ray Imaging Orthicon		
Electronics		
Instrument Package B (2 provided)	40	
Slit		
Crystal Spectrometer		
Photomultipliers		
Electronics		
Instrument Package C (1 provided)	200	
Slit		
H2 Polarimeter/Cryogenic System		
Photomultipliers		
Electronics		
Sub Total	2,454	3,247
Contingency & Growth (10%)	246	323
Total Vehicle	2,700(1)	3,571(2)

(1) Beryllium Mirror Assembly

(2) Fused Silica Mirror Assembly

5.3.1 Vehicle Inertias. The first iteration vehicle mass moments of inertia were computed for the beryllium octagonal truss structure with a 40 in. lens diameter. These were:

25600 Slug $ft.^2$	Pitch and Yaw
3000 Slug $ft.^2$	Roll

The attitude control system environmental torques and propellant requirements shown previously were based on these values. Mass moments of inertia were calculated for the final vehicle configuration with the titanium square section truss and a 30 in. maximum lens diameter. These were:

13800 Slug $ft.^2$	Pitch and Yaw
2500 Slug $ft.^2$	Roll

The study schedule did not permit a final attitude control system iteration; therefore, the propellant weights shown are conservative for the 30 in. design.

5.4 STRESS ANALYSIS. Two programs for the structural analysis of the x-ray telescope truss by use of a digital computer were applied. (Both of these have been in use at Convair division for more than three years). These programs are based upon analyses of the well-known displacement type. (Reference 23)

5.4.1 The Analysis Models. Two analysis models were used: one assumed completely rigid joints and the other assumed pinned joints.

A portion of the stiff jointed model was analyzed, by use of the principle of symmetry and anti-symmetry. (The octagonal configuartion was considered.) All truss memebers were taken to be 2.5 in. OD by 0.020 beryllium tubes, excepting the mutually parallel memebers which guide the traveling truss assembly; these were 3 in. OD by 0.030 tubes. The housing for the lens was regarded as very rigid, and the cylindrical element was replaced by plate elements. Output from this program consists of (1) deflections at each joint: three translations in the direction of three orthogonal axes, and three rotations about these axes; (2) internal loads, or stress resultants: an axial load, two transverse shears in the direction of and two bending moments about the principal axes of inertia, and torsion about the centroidal axis.

A planar truss model, considered to be a simplified version of the actual structure was devised, and was analyzed by the pinjointed program. Output here consists only of translations at the joints and axial loads in the members.

5.4.2 Loading Conditions. Four loading conditions were applied to the stiff jointed model; all were in extended configuration. There were:

a. Unit shear lead

b. Unit moment

c. 45^o Thermal Gradient

d. Docking

Loading Condition a. consisted of a 400 lb. load in the X direction, while Loading

Condition b. was a 400 in. lb. moment about Y. Both were applied at the node marked "28" in Figure 5-13. These loads were applied to obtain influence coefficients for the calculation of natural frequencies. Loading Condition d. consisted of 1 g transverse and axial loads applied at the grazing incidence lens assembly; no damping was considered.

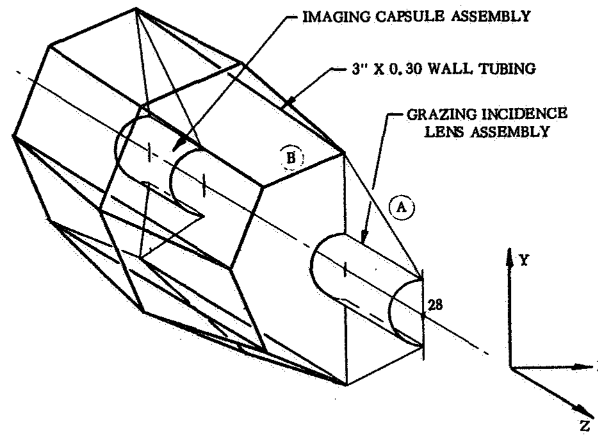


Figure 5-13. Simplified Telescope Structure

Loading Condition c. is shown in Figure 5-14. The premise was that while the members could be at a steady state temperature, each member would exhibit a thermal gradient through its thickness as shown in Figure 5-15.

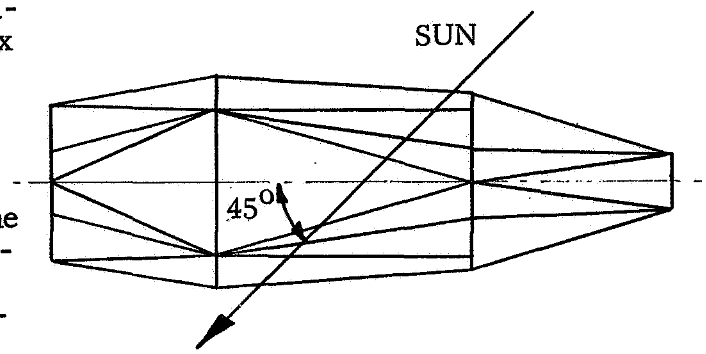


Figure 5-14. Loading Condition c.

Such a gradient induces a moment, constant over the tube length (see Appendix III) given by

$$M_{T_x} = \frac{1}{8} (T_U - T_L) (r_o^4 - r_i^4)$$

Accordingly, M_{T_x} was computed for the truss members. These were then reversed, summed at each node, and applied as a pseudo external force vector.

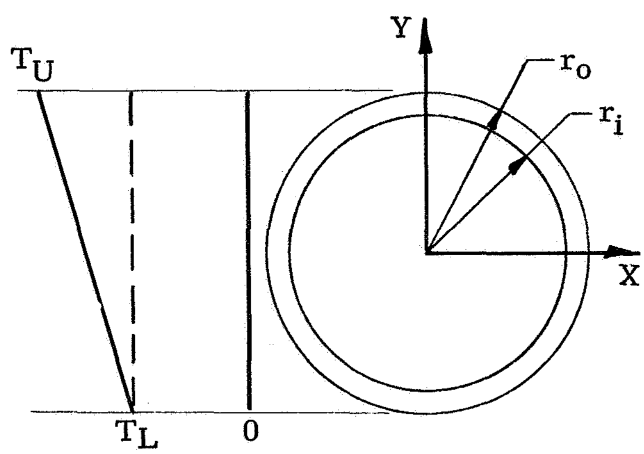


Figure 5-15. Tube Thermal Gradient

It is of interest to note that

$$M_{T_x} \doteq 4.4 \text{ lb. in.}/^{o}\text{F for } B_e$$

$$M_{T_x} \doteq 2.0 \text{ lb. in.}/^{o}\text{F for A}$$

A single loading condition was applied to the pin jointed model shown in Figure 5-16.

In this case, it was postulated that each member would reach an equilibrium temperature based upon the orientation of its centroidal axis with respect to the sun's line of action. With an average temperature known for each member, the program proceeds in the standard manner for thermal stress analysis: fixing forces (= $\Delta E \alpha \Delta T$) for each member are calculated, reversed, summed over the nodes, and applied as an external force vector. This constitutes a relaxation, and displacements are found, from

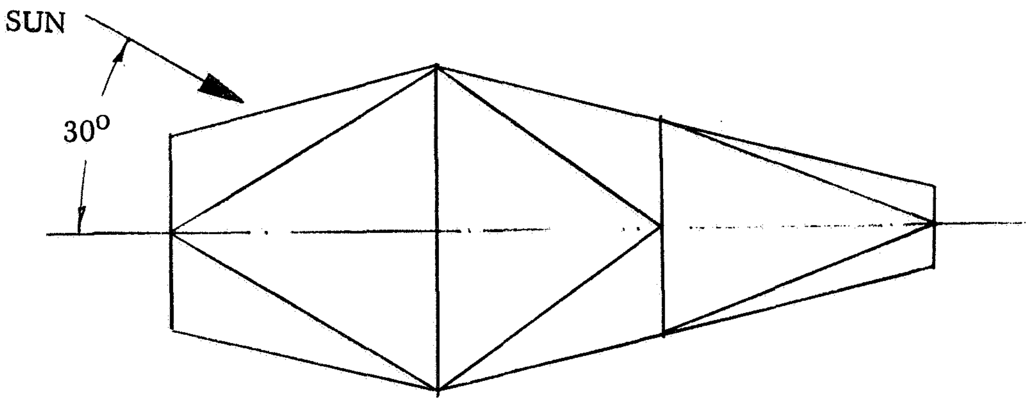


Figure 5-16. Pin Jointed Analysis Model

which actual element internal loads may be computed.

5.4.3 <u>The Results of the Analysis.</u> Since the first two loading conditions on the fixed joint model are unit loadings, they are not considered further in this section.

First consider internal loads, which are of less importance than the deflections. Scanning output shows that in general member loads are higher by two or more orders of magnitude than the loads under the thermal condition. Search shows the highest absolute axial load value is 628 lb. and the highest compressive load value is 592 lb. (These are in truss members indicated by a circled A on Figure 5-13.

The compressive loaded member has an area of 0.156 in.^{2}. The compressive stress then is 3800 psi. The member is 99 in. long, and its moment of inertia is 0.12. Assuming pinned ends, which gives the lowest value for the buckling load, P_{cr}

$$P_{cr} = \frac{\pi^2 EI}{{}^2} \simeq \frac{0.12 \times 4.2 \times 10^7 \ \pi^2}{10^4} \simeq 5,000 \text{ lb.}$$

The maximum bending moment is found in a member of the type indicated by a circled B on Figure 5-13. The bending stress is

$$\sigma_B = \frac{Mc}{I} = \frac{1,395 \times 1.25}{0.12} = 14,500 \text{ psi}$$

The stress levels shown by the preceding calculations are low enough to be neglected, and these are ten to one hundred times as high as the average stresses in the structure.

The displacements shown in Figure 5-17 typify the behavior of the grazing incidence lens assembly (top) and the imaging capsule assembly during Loading Condition c.

$$\theta_1 = \frac{3.95573 - 2.18410}{20} \times 10^{-5} = 0.885 \times 10^{-6} \text{ rad.}$$

$$\theta_2 = \frac{0.818664 - 0.817709}{32} \times 10^{-5} = 0.001 \times 10^{-6} \text{ rad.}$$

$$\theta_3 = \frac{2.1841 - 0.8187}{415} \times 10^{-5} = 0.033 \times 10^{-6} \text{ rad.}$$

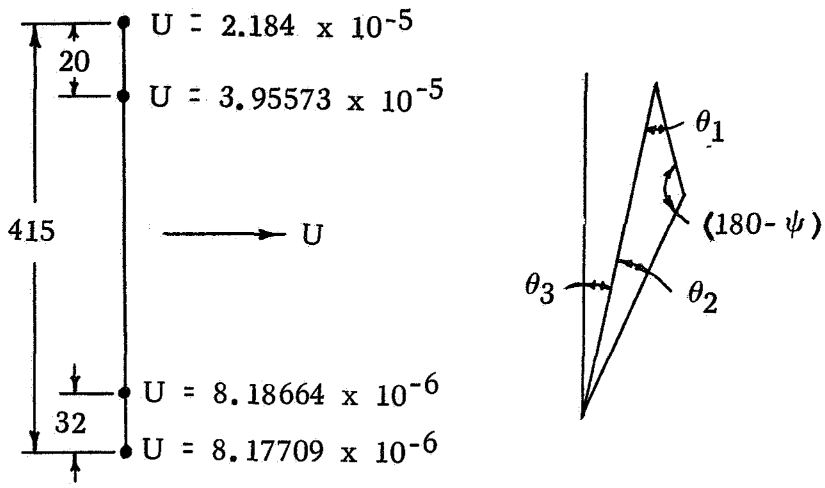


Figure 5-17. Displacement During Loading Condition C

The divergence angle is measured by

$$\psi = (0.885 + 0.001 + 0.033) \times 10^{-6} = 0.92 \times 10^{-6} \text{ rad.}$$

$$\psi = 0.19 \text{ sec.}$$

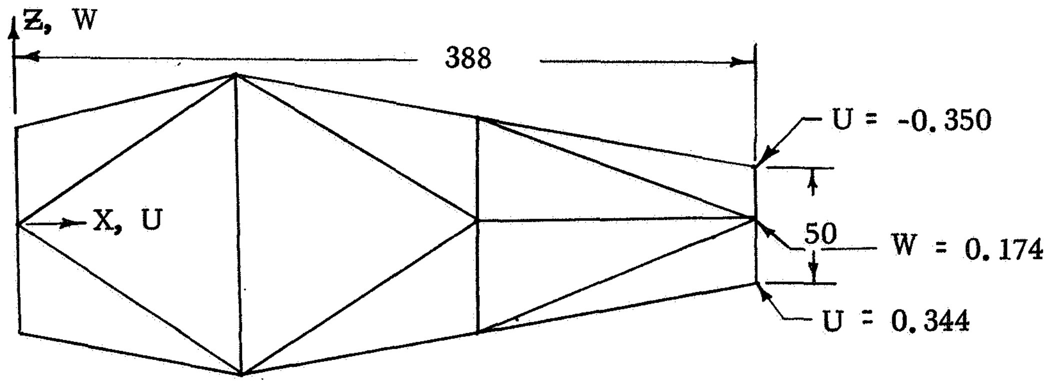
Figure 5-18 shows the deflections in the titanium pin jointed structure.

Schematically, the misalignment angle, $\psi = \theta + \theta_2$

$$\psi = \frac{0.174}{388} + \frac{0.006}{50} \doteq (.511 + .12) \times 10^{-3} \text{ rad.}$$

$$\psi = .631 \times 10^{-3} \text{ rad.} = 130 \text{ arc sec.}$$

Comparison of the divergence angles calculated above strongly indicates that the thermal moment mechanism is so much less effective than the thermal axial growth mechanism that it may be disregarded for structures such as the orbiting telescope. (This situation has been found to prevail for various antenna dishes, in orbit and on the ground, which have been analyzed at Convair division.



Schematically

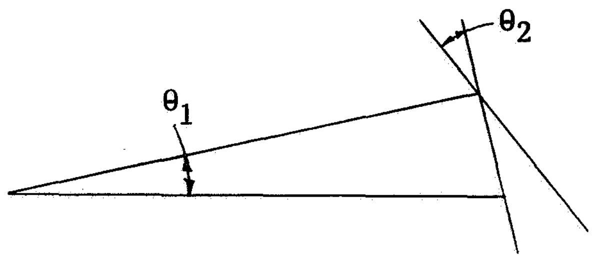


Figure 5-18. Pin Jointed Structure Deflections

In cases such as arise here where a structure is built entirely of one material, scaling laws can be stated which permit the calculation of deflections and internal loads if all material were changed (Section 5.4.5). These laws are given where subscripts B and A indicate beryllium and aluminum respectively.

5.4.4 Scaling Laws. In the case of mechanical loads, if a structure of material "i" has been analyzed, and the deflection vector for material "j" is

$$\left\{\gamma_j\right\} = \frac{E_i}{E_j} \left\{\gamma_i\right\}$$

The internal loads on each element are alike whether material i or j is used. If thermal loads were applied, to find the deflections for material j,

$$\left\{\gamma_j\right\} = \frac{\alpha_j}{\alpha_i} \left\{\gamma_i\right\}$$

To find the internal loads in any element, $\left\{Pa\right\}_j$ in the structure of material j, compute the thermal load vector for the member as made of material i

$$\left\{P_t^a\right\}_i = A\, E_i\, \alpha_i\, (\Delta T)$$

Subtract this from the final load vector for the element, from the computer run, this gives the elastic loading on the element,

$$\{P_e^a\}_i = \{P^a\}_i - \{P_t^a\}_i$$

Find the elastic loading on the element of material j,

$$\{P_e^a\}_j = \frac{\alpha j}{\alpha i} \{P_e^a\}_i$$

Compute

$$\{P_t^a\}_j = AE_j \alpha_m (\Delta T)$$

then the final load is

$$\{P^a\}_j = \{P_e^a\}_j + \{P_t^a\}_j$$

Note that in the preceding discussion it was assumed that the temperatures of all members were alike, independent of material.

Also note if all elements are of the same cross-sectional area and a pin jointed analysis was run, a change of area, from A_I to A_{II} affects the deflection vectors,

$$\{\gamma_{II}\} = \frac{A_I}{A_{II}} \{\gamma_I\}$$

5.5 RELIABILITY.

5.5.1 Scientific Mission Reliability. As defined by NASA the x-ray telescope has three primary flight objectives as were brought forth earlier in this report. Thus by definition the x-ray telescope is a manned space flight experiment. It is of interest however to look at a reliability analysis as though the telescope were an unmanned satellite and to compare the results. The first two flight objectives of the telescope could not be fulfilled by flying the telescope in an unmanned mode, however, the astronomy mission, could, to a large degree still be accomplished. The results of a fairly brief analysis from this viewpoint is presented.

5.5.1.1 Reliability in Design. To assure operation over a maximum portion of the total time in orbit, the x-ray telescope must be designed for long unattended operating periods. Although every effort has been made to design the spacecraft with sufficient maintainability to recover from most failure modes by EVA, frequent failures will cause prolonged times in a failed state prior to the accomplishment of the repair mission. The prolonged down times result in fewer observations being taken in a given time period, seriously reducing the effectiveness of the system. Also, certain failure modes preclude the possibility of restoration to normal operation, regardless of maintainability features.

Parts and components used for these satellites will be mostly within the art for the 1970-75 period. No large-scale component development for this program is anticipated. Reliability must be achieved by careful minimization of critical failure modes. Certain failure modes affecting the operation of the attitude control system (ACS) are most critical for the system. Any mode that causes uncontrollable thrusting from one or more reaction jets results in loss of attitude stability of the satellite. In that case, a CM could not dock to the telescope, unless the telescope can be stabilized by remote control. Possible failure modes resulting in uncontrolled thrusting include an open failure of a thruster valve or a spurious command to thrust from a jet controller. Such a command could be generated in the ACS electronic control subsystem or in the navigation system. The valve open failures can be virtually eliminated by redundant design. One solution would require the addition of an additional shut off valve between each thruster chamber and the normal control valve. The shut off valve could be energized by the spacecraft computer after sensing higher than normal vehicle accelerations.

The tube crimper-cutters used on Gemini can also be used to completely sever and seal the propellant tank from the thruster in the event of a valve open failure.

The back-up docking ACS is particularly important since the resupply crew can repair most failures but they are first required to dock the CSM to the telescope. The back-up system is not coupled to the x-ray navigation system and would be actuated by the rendezvousing CSM crew from a safe distance.

Part and component redundancies are being employed in design to reduce system failure rates. Some redundancy is inherent among the attitude control thrusters. Solar arrays are designed for over capacity, so that the electrical system can tolerate a number of cell failures. It is certain that detailed design of electronic subsystems, including instrumentation, communications, navigation and control, will probably be available for use in most applications in this satellite. Not only does their small size and low cost make redundancy attractive, but also single-chip reliability is continuously increasing as experience is gained in quality improvement.

Finally, wide operating tolerances are employed whenever feasible. In this manner the system remains operable in spite of normal drift of part values. For example, an image position restorer is provided in the experiment packages, to relax the tolerance requirements for the lens drive and control subsystem. In turn, the lens angle can be changed to relax tolerance requirements on the ACS and navigation system. Initially and periodically these subsystems will be adjusted for maximum accuracy or precision. However, the image can still be focussed even after some degradation of the performance of these subsystems.

5.5.1.2 Unmanned System Reliability. An initial reliability prediction was performed for a completely unattended satellite, so that the effects of astronaut participation on probability of mission success could be evaluated. The prediction includes the effects of launch into orbit but does not include the launch system, upper stage or ground systems. The results are summarized for a two-year period by subsystem and by mission phase in Table 5-2. The figures can be applied only to the third flight

Table 5-2. X-ray Telescope - Cummulative Unmanned Reliability

SUBSYSTEM	DEPLOYMENT	3 MOS.	6 MOS.	1 YEAR	2 YEARS
Electrical Power	.999947	.9952	.9905	.981	.960
X-ray Mirror System & Experimental Equipment	.9983	.937	.878	.773	.595
Data, TLM, & Comm. & Housekeeping	~1.0	.99990	.99979	.99920	.9968
Attitude Control	.99970	.9901	.980	.959	.912
Command & Control	~1.0	.999962	.99986	.99947	.9978
Deployment	.9989	.9989	.9989	.9989	.9989
Total Spacecraft	.9969	.922	.851	.725	.517

objective of the structure (scientific mission) since by definition the vehicle requires man to accomplish the other two flight objectives.

Structural components are not included, since sufficient margins of safety are incorporated to assure their integrity for the duration of the mission. Component failure rates are based primarily on those in the Failure Rate Data Handbook (FARADA), published by the U. S. Navy, FMSAEG, Corona, California. Electrical power component data were taken from the paper, "An Evaluation and Comparison of Power Systems for Long Duration Manned Space Vehicles", by J. G. Krisilas and H. J. Killian of the Aerospace Corp., presented at the Intersociety Energy Conversion Engineering Conference of AIAA, ASME, IEEE AND AICE, 26 to 28 September 1966. Data for most of the ACS and navigation components were derived from the G. E. report. "Satellite Orientation with OAO Developed Control Componets".

The failure rates were adjusted to ground laboratory conditions as required, to make them equivalent to rates in orbital environment. The ground rates were divided by two as a conservative allowance for improvement in state-of-the-art reliability. For the orbital launch phase the adjusted ground rates were multiplied by 80, in accordance with MIL-STD-756A. Most electrical and electronic components were assumed to contain part and circuit redundancy, the effect of which is equivalent to redundancy at the component level. The results of redundancy on the reliability analysis is an "aging" effect, due to the mixed exponential distribution of system times between failures. The effect simulates not only redundancy, but system degradation due to drift of operating parameters.

The predictions in Table 5-2 indicate that the major portion of system unreliability is in the x-ray mirror system and experiment equipment. Volume of the proposed design would not allow complete redundancy at the component level. The experimental package turret has room for two each of Packages "A" and "C", which are provided. These packages are complex, implying relatively high failure rates. The laser alignment subsystem, being non-redundant, has relatively low reliability. Finally, the mechanical drives in the system, some of which require precise control, significantly reduce system reliability. These drives inclued the turret, grating, sun shield, and lens drive assemblies.

One factor which was not considered in the prediction is the degradation of the grating due to impact of particles on its surface. Evaluation of this problem requires a definition of the amount of degradation that is tolerable. However, in detailed design, means of countering the problem must be considered, such as protected excess surface to be exposed in steps as the previously exposed surface becomes excessively degraded.

The attitude control system reliability is reduced by the inertia wheels which are not redundant in the proposed design. Adequate volume exists in the image capsule to provide a redundant set of wheels, the weight penalty being approximately 200 lb. To an extent the function of a failed wheel could be assumed by the reaction subsystem. However, satellite pointing accuracy would be marginal at best, and then only with high propellant rates.

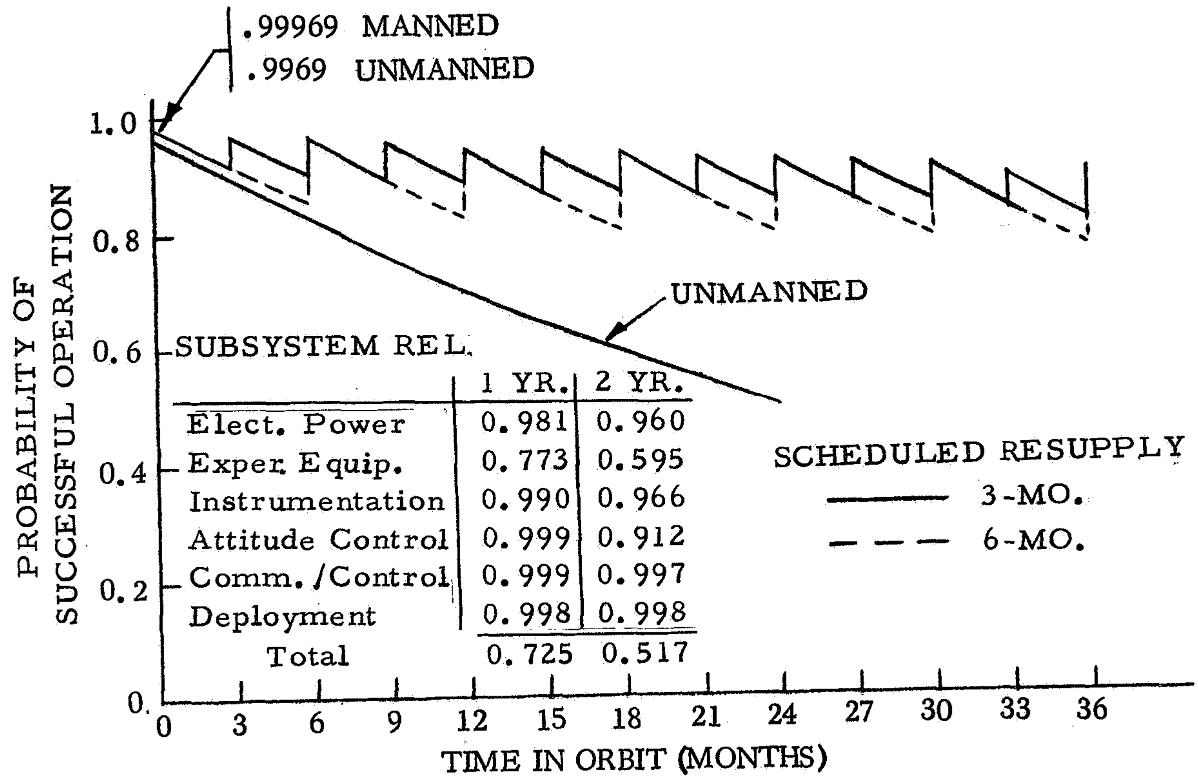
5.5.2 Man's Impact on Mission Reliability. Man contributes in a very valuable way to the mission by maintaining and up-dating the telescope instrumentation as well as any facility subsystems. Man extends the useful life time and improves the scientific quality of the telescope through the addition of new and more sophisticated modules.

Design criteria include features for maintainability in an EVA environment, such as modules for replaceable units, large knobs and handles for clamping or making adjustments, handholds and belt attachments for astronaut restraint and access doors to components. Safety provisions include flexible antennas and soft coatings on sharp corners to avoid snagging or tearing a pressure suit or astronaut tether.

Since the design is not sufficiently detailed to determine all componet failures which can be corrected by maintenance, it is assumed for analysis that the astronauts will repair 90% of all failures occurring between each rendezvous period. Such repairs include actions to restore the system to its peak operating condition, whether or not complete system failure has occurred. Failures resulting in the loss of satellite attitude stability would preclude posibility of repair, unless stability can first be restored by remote control.

Figure 5-19 depicts the contribution of man to the system reliability at various times during a three-year stay in orbit for the satellite. The effects of scheduled manned visits very three months, six months and one year are compared with satellite reliability over a two-year period. All repair schedules are begun with manned checkout and repair at the beginning of the satellite mission. Since the figure consists of plots of probability vs. time, the area under the probability curve between any two times is expected time that the satellite is operable. The area between the curve and the horizontal line for probability of one is, therefore, expected down time. The figure shows that more frequent visits result in less satellite inoperable time because of the greater number of recovery peaks.

Reduction in expected down time with increasing frequency of scheduled repair missions is shown in greater detail in Figure 5-20. This figures shows, for a given repair schedule, cumulative expected satellite down times for comparison with the totally unmanned case. Since expected down time is represented by the area above the reliability curve, its value during satellite orbital time $t_2 - t_1$ is



SUBSYSTEM REL.	1 YR.	2 YR.
Elect. Power	0.981	0.960
Exper. Equip.	0.773	0.595
Instrumentation	0.990	0.966
Attitude Control	0.999	0.912
Comm. /Control	0.999	0.997
Deployment	0.998	0.998
Total	0.725	0.517

Figure 5-19. Probability Of Operation At Given Time

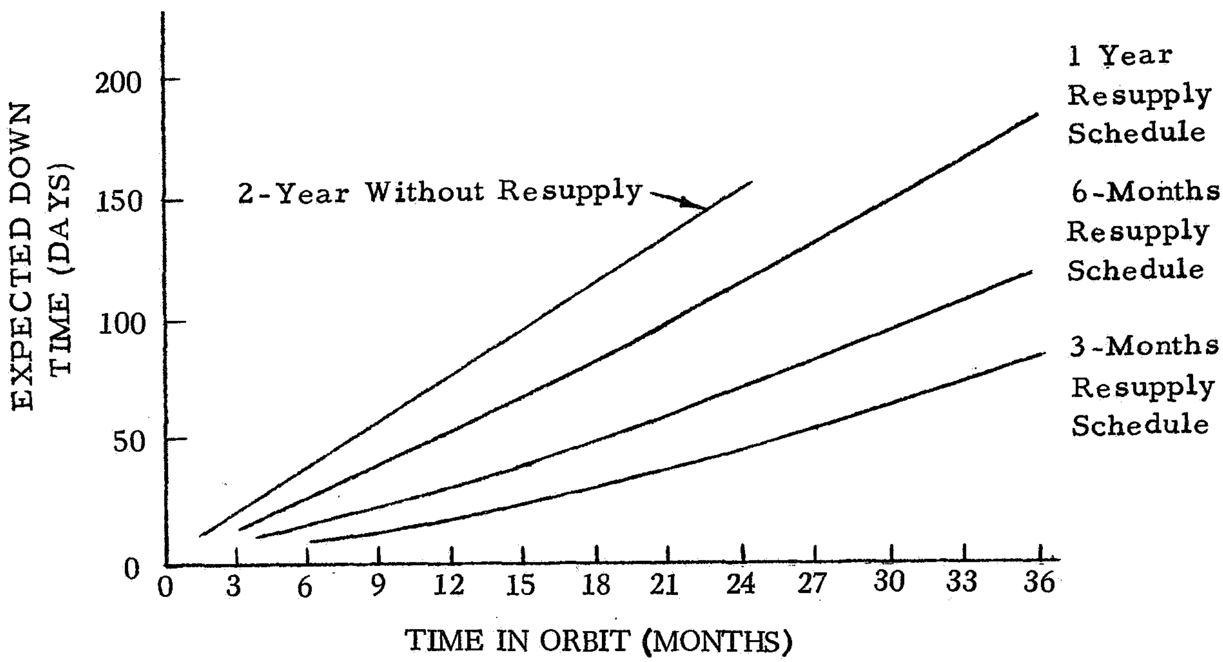


Figure 5-20. Expected Down Time

$$E(t_d) = \int_{t_1}^{t_2} \left[1 - R(t)\right] dt$$

where R(t) is the system reliability function. Due to discontinuities of R (t) at periods of potential repair, the integral is evaluated for each period between repairs. The resulting down times are then accumulated. With the many component redundancies the actual R(t) is a function which requires lengthy calculations to integrate. However, it can be approximated reasonably between t_1 and t_2 as $R(t) = P_s e^{- \ t}$, where   is the average system failure rate during the period $(t_2 - t_1)$ and P_s is the probability of system operation at time t_1. Since R(t) is not continuous over a period longer than $(t_2 - t_1)$, zero can be substituted for t_1 as the lower limit and $(t_2 - t_1)$ for t_2 as the upper limit of the integral.

Thus the expected down time per internal between rendezvous periods is expressed as

$$E(t_d) = \int_{o}^{t_2 - t_1} \left[1 - P_s e^{-\lambda t}\right] dt = t_2 - t_1 - \frac{P_s \left(1 - e^{-\lambda(t_2 - t_1)}\right)}{\lambda}$$

For the purposes of the study all systems were assumed to be of equal value to the overall mission. While this allows a gross estimate of the effect of man on system reliability the shortcomings must be recognized.

A more accurate reliability assessment of the system would require assigning a criticality number to each system. The critical number might represent what percent of the mission the system must function i.e., the ACS, and electrical power criticality would be 100% since the mission would be terminated until the system was repaired. Other subsystems such as the sunshade mechanism for example, might only be required 10% of the mission. Once the criticality values were assigned (assuming the failure rates known) the total vehicle would then be "flown" by "Monte Carlo" computer simulation, a large number of flights - perhaps 100. The time to critical system failure could then be analyzed. Man's ability to extend the mission effectiveness could better be evaluated as the specific failures would be identified. In most cases the EVA astronaut would be able to return the critical systems to 100% operating specifications - whereas the simplified analysis model used 90%.

In summary the reliability analysis assumptions used to generate Figures 5-15 and 5-16 are:

a. 1970-75 state-of-the-art parts and components.

b. Complimnetary operating tolerances (Image motion compensator relaxes lens drive and control tolerances - lens drive tolerances relax ACS etc.)

c. Part and component redundancies reduce system failure rates.

d. Repair capability is 90% of peak operating system.

SECTION 6

CREW SYSTEMS

As discussed earlier in this report, the x-ray telescope has three primary flight objectives to fulfill. One has to do with man's capability in space ("EVA"), another has to do with space structures technology, and thirdly the provision of an operational astronomical instrument in space. Man's role in the x-ray telescope thus plays two concurrent roles: He is the subject of the experiment (that is to say, while doing EVA he is learning to do EVA, evaluating procedures and equipment) and concurrently the astronaut is performing operational maintenance and repair type activities. Thus man satisfies both major objectives concurrently. Either of two approaches could have been pursued: Man could have been utilized to do virtually everything (even in some cases where automating would have been easy) or man could have been used only where it appeared to be optimum through a man vs. automation tradeoff and best judgment. The latter approach was used.

The basic "man" philosophy has been that man (including the launch of manned vehicles) will be available for resupply (repair) to support the telescope as required.

6.1 CREW SYSTEMS CAPABILITIES. The description of an extravehicular astronaut's performance capabilities as a function of the equipment available for EVA operations is covered in the NASA/MSC "Baseline Astronaut for 1968-1972" document reproduced in Volume II of this report. In addition, coordination meetings with Mr. N. Belasco of MSC and with the NAA/S&ID study of Extra Vehicular Engineering Activities have helped to establish crew capabilities. Two presentation/meetings were held at MSC. At the mid-term meeting, astronauts M. Collins, G. Cernan, and O. Garriott provided a critique of the Convair approach. The orbiting x-ray telescope described in Section 4.0 was designed for compatibility with the baseline astronaut capability and therefore does not require additional or unusual capabilities.

The scheduled operational phases of the mission require various degrees of manned participation; deployment, checkout, resupply and refurbishment have been analyzed in sufficient detail to ascertain that the design is compatible with the requirements of the EVA astronaut.

The initial EVA philosophy used to generate the proposed design is summarized by the following:

a. CSM docked at image capsule for all EVA activities.

b. Utilization of a 50 ft. tether at all times.

c. EVA locomotion by hand using truss structure, handrails as required, and hand gun as alternate.

d. Equipment large enough to be unwieldy or impair astronaut's mobility would be transferred by "clothesline".

e. All equipment to be replaced or secured by gross clamping motions, no nut or bolt attachments.

f. No pneumatic or hydraulic lines would be disassembled or assembled. ACS units to be separate integral modules requiring only structural and electrical interface mating.

g. Life support provided by PLSS.

h. Automate all functions where practical. Do not use man except where necessary.

i. Design all automatic systems with manual over-ride capability for backup where practical.

The tubular truss design of the telescope structure provides the EVA astronaut virtually unlimited handholds and accessibility to all equipment external to the image capsule. In addition, the open nature of the structure allows the EVA astronaut considerable mobility while remaining in visual contact with the CSM crew. This is an important safety factor which permits the rescue crewman to remain in the CSM and observe the EVA crew.

The initial structure design was based on the use of beryllium for lightweight and stiffness, but further analysis proved beryllium to be a poor choice for the long slender tubes primarily from an astronaut safety aspect; the beryllium has low impact resistance because of its relatively low yield stress and high elastic modules.

Titanium appears to be the best overall tube material, particularly as it exhibits excellent impact resistance. The tube size proposed is capable of resisting astronaut loads far in excess of anticipated.

Early in the study phase, it became readily apparent that the manned interfaces must be evaluated for each system to arrive at compatible compromises. Some of the features which were included as design constraints to assist the EVA astronaut in his required task are:

a. Astronaut command retractable solar cell/ACS booms instead of single deploying structure.

b. Astronaut controlled (reversible) primary truss extension system to prevent damage during extension from unforeseen causes.

c. Roll-up solar panels to simplify stowing, transporting, and installing new panels by EVA.

d. Add-on ACS modules - eliminates problem of removal, stowage, etc.

e. Built-in restraints for astronaut on the 4 image capsule doors providing them with an ideal work platform for access to all the internal systems. See Figure 6-1.

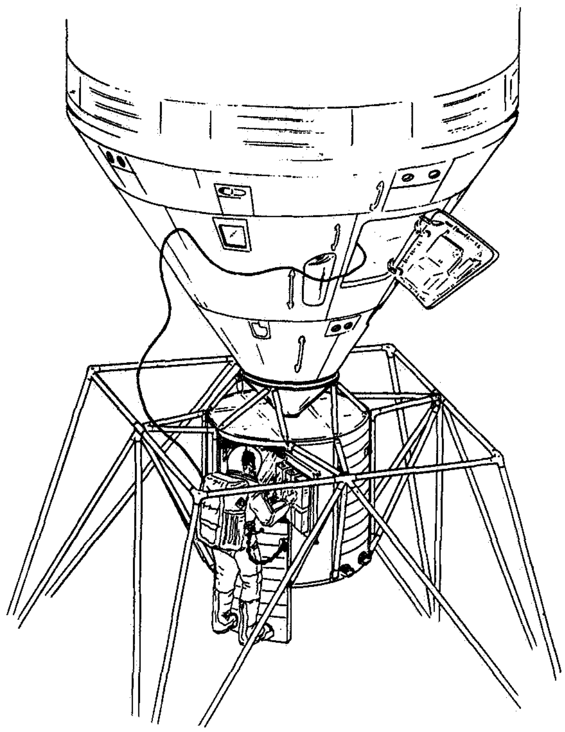


Figure 6-1. EVA Image Capsule Maintenance

6.2 OBJECTIVES OF EVA. The objectives of planned EVA on the x-ray telescope are four-fold.

a. Develop confidence in man's ability to perform the tasks necessary to support a useful scientific mission.

b. Achieve the highest deployment reliability to guarantee the effective initiation of the scientific mission.

c. Extend the useful lifetime of the system beyond the normal component and subsystem lifetimes by repair, replacement, and resupply of critical items.

d. Update the experiment capability with the latest equipment designs to deter obsolescence.

The following paragraphs describe astronaut participation in the x-ray telescope mission.

6.2.1 Deployment.

a. Predeployment status check (IVA).

b. Initiate and control deployment of solar cell frames and ACS booms and check lock indicators (IVA).

c. Test solar panel and ACS operation (IVA).

d. Initiate and control extension of lens capsule (IVA) (manned observation of the extension can assure safe deployment by stopping operation if binding occurs--then requires EVA to correct malfunction).

e. Separate lens capsule from launch support ring (IVA) (the optimum focal length adjustment may require the CSM crew to monitor the image screen, with a repeater scope, while setting the lens focal length within the $\pm$ 5-in. adjustment).

f. Subsequent to proper deployment - when lens extension indicates locked (8 places), release extension system cable tension (electro-pyrotechnic 4 places) IVA.

g. Check lens shade tubes locked (4 places) IVA.

h. Egress CSM, locomote to image capsule, open all four access doors, remove launch supports on turret and inspect for wire bundle interferences, etc. Free wheel turret by hand full travel (270°) to verify clearances, energize master turret switch (EVA).

i. Cycle turret with local control, switch to auto control, and observe ground control operation (EVA).

Operational Checkout

a. Actuate spectrometer grating and observe, slew lens gimbal motors and observe correction made by lens alignment system - check visual alignment indicators - return to CSM (EVA).

b. Monitor ground-controlled systems check of all x-ray systems (EVA).

c. Compare x-ray pointing coordinates with Apollo navigation system (IVA).

6.2.2 Resupply. Tentative scheduled resupply of ACS expendables and solar panels is two years after launch. Resupply prior to this would be for purposes of updating any equipment or replacement due to life of less than two years or failures and is considered unscheduled resupply.

EVA tasks on resupply missions are as follows:

a. Installation of equipment in image capsule. The turret is designed for 7 stations. Three are occupied by primary packages A, B and C capable of four separate observations. Two more stations contain duplications of packages A and C. Two stations are therefore available for addition on resupply. Add-on is preferred to replacement; however, all packages are removable by EVA astronaut and will fit into CM for earth return, if desired. For failure analysis, etc., sufficient volume and strength exists on the structure to stow expended equipment.

c. The ACS and solar panels are replaceable by the astronaut. Each ACS unit contains jets, valves and tanks, has a single mechanical attachment collar and an electrical connector. The solar panels are flexible roll-out type contained in a cylindrical dispenser. The astronaut secures himself to the ACS deployment frame to replace the ACS units or solar panel.

c. Battery replacement will likely occur coincident with solar panel replacement. These batteries are located in a sector of the image capsule.

6.2.3 Refurbishment. Refurbishment consists of replacement of certain items as per the normal resupply except additional equipment will be replaced to prolong the useful life of the facility. Also, new and probably different types of mission equipment will be added to broaden the scope of the facility investigations. The total number or specific items to be replaced are not completely known. Those identified here can be considered typical. EVA will be required extensively to perform some of these tasks:

a. Image capsule mission equipment.

b. Image capsule operational systems (momentum wheels, turret motor/gear box, laser receiver system, electronic modules, batteries, star tracker, etc.).

c. Lens capsule operational systems (focal length drive motors, gimbal motors, spectrometer grating and/or motors, lens shade patching, telescope electronics package laser beam generator package, filter discus, etc.).

d. ACS units.

e. Solar panel units.

6.3 CREW TASKS AND TIME LINE ANALYSIS. A summary of planned crew activities is contained in the following tables, 6-I and 6-II. These tables, based on preliminary functional analysis of mission operations, reflect the general requirements for crew participation in the initial deployment operations and resupply operations. The remaining operation of refurbishment was only analyzed briefly because of study time limits. Task analysis work sheets were initiated, but not refined (see appendix). The refurbishment tasks consist of the previously outlined resupply tasks, and in addition include replacement of such items as noted in Section 6.2.3. The scope of analysis to determine the exact items likely to require replacement on refurbishment is extensive and goes beyond the time budget of this study. It is estimated that the orbital refurbishment task will require a minimum of double the time and effort necessary for the resupply mission.

Table 6-1. Time Line Analysis
X-ray Telescope Initial Deployment to Operational Status

Final orbit altitude achieved (260 n.mi.). Crew is in CM in spacesuits. CSM has transpositioned, SLA panels have deployed and CSM is docked onto Image Capsule drogue. This is point of beginning.

EVENT NO.	EVENT AND TASKS	EVENT TIME (MIN.)	ACCUM TIME (MIN.)	EVA ACCUM TIME (MIN.)
1	Time starts at point of beginning	0.0	0.0	
2	Crewman equalizes pressure in drogue. Removes forward hatch and stows. Latches the 12 manual latches. Connects electrical umbilical to X-Ray Telescope system	30.0	30.0	
3	Crew initiates separation of pyro bolts at 4 SLA hard points. CSM maneuvered to extract telescope from SLA.	5.0	35.0	
4	CSM maneuvered to correct orbit parameters and gain separation distance from S-IVB.	30.0	65.0	
5	Orbit circularized (assume occurs at half orbit period). This period is also used to conduct spacecraft housekeeping and biomedical checks. Shirtsleeve atmosphere.	45.0	105.0	
6	All recording systems checked relative to strain gauge data, temperature, ICSD, etc.	15.0	120.0	
7	Status for go is established through critique with ground control.	5.0	125.0	
8	Crew initiates signal to release #1 ACS/solar frame. Pyro bolt release. Elect indicator.	0.1	125.1	
9	Crew operates momentary switch to deploy. Other crew members observe deployment via windows and/or periscope. Frame rotates 103° and locks. Solar panel is automatically unlocked with extension of frame and is automatically signaled to extend within last quarter of frame travel. Solar panel extends and locks. Frame extension time 25 sec. and panel extension time 10 sec. with a 5 sec. overlap. Total = 30 sec. Switches indicate lock-up of both.	0.5	125.6	
10	Crew observes deployment. Verifies deployment in conjunc-with lights. Decision to proceed.	0.5	126.1	
11	#2 ACS/solar frame deployed, observed	1.0	127.1	
12	#3 ACS/solar frame deployed, observed	1.0	128.1	
13	#4 ACS/solar frame deployed, observed	1.0	129.1	
14	Crew critique. Re-check of all systems. Decision to proceed.	5.0	134.1	
15	Signal energizing X-Ray Telescope power generating system. Output of each solar panel monitored. Electrical characteristics at buses monitored.	15.0	149.1	

Table 6-1. Time Line Analysis (cont'd.)

EVENT NO.	EVENT AND TASKS	EVENT TIME (MIN.)	ACCUM TIME (MIN.)	EVA ACCUM TIME (MIN.)
16	Individual signals to fire each ACS jet. Verification that each operates, (CSM readout). Note: Attitude control is still maintained by CSM. Telescope ACS system is <u>not</u> fully activated.	15.0	164.1	
17	Critique by crew. Ground critique. Decision making. Go ahead.	5.0	169.1	
18	Signal to release lens lateral restraints. Pyro tape cutters sever tapes (or cables). Tapes retract into cases. Indicator lights verify completion.	0.4	169.5	
19	Crew initiates signal to release lens capsule truss carriage hold-down bots. Pyro release. Signal indication of release.	0.5	170.0	
20	Crew initiates signal to truss extension motor system. Run-around cable system extends truss. Lens shade is extended as part of cycle.	5.0	175.0	
21	Two-stage locks engage at end of stroke. Second stage pyro locks automatically initiated; extension system terminated. Electrical indication of lock..	0.1	175.1	
22	Crew initiates signal to release cable tension by retraction of pulley. Pyro or solenoid actuation. Strain gauges on tracks indicate release.	0.5	175.6	
23	Crew initiates signal to energize lens capsule separation bolts. Indicator lights show release.	0.4	176.0	
24	Crew initiates signal to advance lens capsule to mid-position. Capsule travels to position. Encoder relays position information.	1.0	177.0	
25	Crew initiates individual override signals to both axis of gimbals for trial run. Readout system also checked for data output.	5.0	182.0	
26	Crew critique. Rest period. Meal.	90.0	272.0	
27	Crew dons spacesuits, suit checkout, PLSS check, etc.	120.0	392.0	
28	Depressurize CM	4.0	396.0	
29	One crew member opens hatch and exits CM (Hatch left open)	30.0	426.0	34.0
30	Crew member travels to image capsule (tether). Bay #1	5.0	431.0	39.0
31	EV astronaut engages feet in door stirrups and engages waist restraint onto door.	4.0	435.0	43.0
32	Astronaut plugs portable power tool into door receptacle and retracts door. (Approximately 3/4 open)	1.0	436.0	44.0
33	Rest period. Flood lights in capsule on.	2.0	438.0	46.0

Table 6-1. Time Line Analysis (cont'd.)

EVENT NO.	EVENT AND TASKS	EVENT TIME (MIN.)	ACCUM TIME (MIN.)	EVA ACCUM TIME (MIN.)
34	EV astronaut actuates switch on interior panel energizing the laser alignment system. Beam impinges on target. A search mode is assumed to obtain target. Gimbal motors can be given override commands by astronaut. Astronaut must detach himself from door to make visual check on gross miss of target.	10.0	448.0	56.0
35	Rest period	2.0	450.0	58.0
37	Completion of laser alignment. Ground check.	5.0	470.0	63.0
38	EV astronaut checks for clearances around turret. All harnesses checked and any losse items secured.	2.0	472.0	65.0
39	EV astronaut disengages launch restraint pins at top of turret. Duplex thrust bearing glange cinched up. Drive motor clutch is checked for disengagement. (Actually dis-engaged prior to launch.)	10.0	482.0	75.0
41	Rest period	2.0	484.0	77.0
42	EV astronaut slowly hand rotates free-wheeling turret. Turret elec power is off. All scientific packages checked for securing to turret and wire harness receptacles engaged.	5.0	489.0	82.0
43	EV astronaut engages drive clutch. Switches on turret power and rotates turret through all seven stations. This is override control. Position indicators note turret position. Scientific instruments are "off". Limit switches prevent over-travel.	5.0	489.0	87.0
44	Turret rotation control switched to ground control and cycling repeated. EV astronaut stands by as observer. Turret left at Station #1 (image signature). Rest for EV astronaut.	7.0	496.0	94.0
45	EV astrnaut actuates power tool and closes door.	1.0	497.0	99.0
46	Astronaut disengages from door and returns to CM.	9.0	506.0	108.0
47	Hatch is closed. CM pressurized.	30.0	536.0	138.0
48	Astronauts doff space suits. PLSS regenerates, etc.	60.0	596.0	
49	Repressurize CM	5.0	601.0	
50	Complete check of all systems, strain gauges, temperature sensors, etc. Ground to spacecraft critique.	60.0	661.0	
51	CSM ACS switched "off". X-Ray ACS system "on". Inertial platform "on".	1.0	662.0	
52	Attitude control system check.	15.0	677.0	
53	Sleep, personal time, hygiene, awake, eat.	480.0	1157.0	

Table 6-1. Time Line Analysis (cont'd.)

EVENT NO.	EVENT AND TASKS	EVENT TIME (MIN.)	ACCUM TIME (MIN.)	EVA ACCUM TIME (MIN.)
54	All systems check especially CM housekeeping. Ground to spacecraft critique.	60.0	1217.0	
55	Attitude of spacecraft altered relative to sun and laser alignment system monitored to reveal compensation capability. Repeated minimum 3 times to gain assurance of performance.	45.0	1262.0	
56	All X-ray systems energized, CSM ACS and navigation "off". Telescope inertial platform "on". Scientific equipment "on".	5.0	1267.0	
57	Ground control sends up-date information to navigation system and commands pointing to particular sky coordinates. (Prime X-ray reference source.) Telescope slows and locks on.	5.0	1272.0	
58	Optical telescope locks on guide star. Verified at ground station.	1.0	1273.0	
59	Scientific instrument package at Station #1 energized by ground control. Held on target for approximately 24 min. Verifies instrument is receiving, IMC operating, control system compensation. (Image signature) Pick-up and communication time.	30.0	1303.0	
60	Station #3 commanded into receive position. Held 24 min and pick-up and communications time.	30.0	1333.0	
61	Auxiliary tasks, housekeeping, systems checks until target is again available.	30.0	1363.0	
62	Station #4 commanded into receive position. Held 24 min and communications and pick-up time.	30.0	1393.0	
63	Station #5 commanded into receive position. Held 24 min and communications and pick-up time.	30.0	1423.0	
64	Auxiliary tasks, housekeeping, systems check until target is again available. Eat.	30.0	1453.0	
65	Station #7 commanded into receive position. Held 24 min and communication and pick-up time.	30.0	1483.0	
66	Check of all systems. Prepare for separation. Telescope stabilized by reference to inertial platform. All systems check-off.	30.0	1513.0	
67	Space suits donned. Suit systems checked.	120.0	1633.0	
68	Docking manual latches disengaged. Pressure hatch installed.	60.0	1693.0	
69	CSM undocks from x-ray telescope. CSM is flown around telescope. Movies, stills, close observation. Formation at approximately 200 feet distance.	90.0	1783.0	

Table 6-1. Time Line Analysis (cont'd.)

EVENT NO.	EVENT AND TASKS	EVENT TIME (MIN.)	ACCUM TIME (MIN.)	EVA ACCUM TIME (MIN.)
70	Rest, eat, exercise.	60.0	1843.0	
71	Ground control re-energizes scientific equipment. Reference x-ray source picked up. Instruments cycled through operational check similar to before except all accomplished in 2 orbits. Turret final cycle is at Station #1. CSM flies formation during this check.	180.0	1923.0	
72	Telescope commanded to new source.	5.0	1928.0	
73	CSM flys formation for 8 orbits (12 hours). This completes normal preparation to operational status.	720.0	2648.0	

Total time to operational status: 1 day, 20 hrs., 8 min.

Accumulated EVA time: 2 hrs., 18 min.

Table 6-2. Time Line Analysis - Resupply

Rendezvous with telescope has been achieved and CSM has docked onto the x-ray telescope image capsule drogue. This is point of beginning. Following EVA tasks include approximately 2 min. rest for each 10 min. of moderate to strenuous effort.

EVENT NO.	EVENT AND TASKS	EVENT TIME (MIN.)	ACCUM TIME (MIN.)	EVA ACCUM TIME (MIN.)
1	Point of beginning. Crew in spacesuits.	0.0	0.0	
2	Crewman equalizes pressure in drogue. Removes forward hatch and stows. Latches the 12 manual latches. Connects electrical umbilical to x-ray telescope system. Telescope systems de-energized.	34.0	34.0	
3	CM depressurized. Hatch opened. Tether rigged, etc.	64.0	98.0	
4	Astronaut "B" exits CM. Maneuvers to SM. Erects conveyor line and transfers units to main truss center frame.	46.0	114.0	46.0
5	Retraction and installation of #1 ACS unit.	34.0	178.0	80.0
6	Retraction and installation of #2 ACS unit. Return to CM.	34.0	212.0	114.0
7	CM hatch closed. CM pressurized.	35.0	247.0	
8	Space suits doffed. PLSS regenerated. All systems check, biomed checks. Crew and ground critique.	60.0	307.0	
9	Eat, personal hygiene, housekeeping, biomed.	90.0	397.0	
10	Space suits donned. Suit and PLSS check-out.	120.0	517.0	
11	CM depressurized. EVA hatch opened. Tether rigged, etc.	64.0	581.0	
12	Astronaut "C" exits. Maneuvers to SM. ACS units #3 and 4 attached to conveyor and conveyed to truss center and secured.	42.0	623.0	156.0
13	Retraction and installation of #3 ACS unit.	34.0	657.0	190.0
14	Retraction and installation of #4 ACS unit. Astronaut "C" returns to SM.	39.0	696.0	229.0
15	Astronaut "C" attaches #1 and #2 solar panel units to conveyor. Returns to CM.	32.0	728.0	261.0
16	Hatch closed. CM pressurized.	35.0	763.0	
17	Space suits doffed. PLSS regenerated. All systems check, biomed checks. Crew and ground critique.	60.0	823.0	
18	Eat, personal hygiene, housekeeping.	60.0	883.0	
19	Sleep.	480.0	1363.0	

Table 6-2. Time Line Analysis - Resupply (cont'd.)

EVENT NO.	EVENT AND TASKS	EVENT TIME (MIN.)	ACCUM TIME (MIN.)	EVA ACCUM TIME (MIN.)
20	Eat, personal hygiene, housekeeping, biomed checks.	60.0	1423.0	
21	Space suits donned. Suit and PLSS check-out.	120.0	1543.0	
22	CM depressurized. EVA hatch opened. Tether rigged, etc.	64.0	1607.0	
23	Astronaut "A" exits CM. Maneuvers to #1 solar panel area. Installs #1 solar panel.	50.0	1657.0	311.0
24	Extension of #1 ACS/solar panel frame.	12.0	1669.0	323.0
25	Installation of #2 solar panel.	50.0	1719.0	373.0
26	Extension of #2 ACS/solar panel frame. Returns to CM.	28.0	1747.0	401.0
27	CM hatch closed. CM pressurized.	35.0	1782.0	
28	Space suits doffed. PLSS regenerated. All systems check, biomed checks. Crew and ground critique.	60.0	1842.0	
29	Eat, personal hygiene, housekeeping biomed.	90.0	1932.0	
30	Space suits donned. Suit and PLSS check-out.	120.0	2052.0	
31	CM depressurized. EVA hatch opened. Tether rigged.	64.0	2116.0	
32	Astronaut "B" exits and maneuvers to SM, removes and attaches solar panels #3 and #4 and conveys to center of main truss.	38.0	2154.0	439.0
33	Installation of solar panel #3.	50.0	2204.0	489.0
34	Extension of #3 ACS/solar panel frame.	12.0	2216.0	501.0
35	Installation of solar panel #4.	50.0	2266.0	551.0
36	Extension of #4 ACS/solar panel frame. Astronaut "B" returns to CM.	18.0	2284.0	569.0
37	CM side hatch closed. CM pressurized.	35.0	2319.0	
38	Space suits doffed. PLSS regenerated. All systems check, biomed checks. Crew and ground critique.	60.0	2379.0	
39	Eat, personal hygiene, housekeeping, biomed.	60.0	2439.0	
40	Operational check-out of power system.	30.0	2469.0	
41	Operational check-out of ACS system. Telescope ACS "on", CM ACS "off".	30.0	2499.0	
42	Telescope ACS "off". CM ACS "on".	1.0	2500.0	
43	Sleep.	480.0	2980.0	
44	Eat, personal hygiene, housekeeping, biomed.	60.0	3040.0	

Table 6-2. Time Line Analysis - Resupply (cont'd.)

EVENT NO.	EVENT AND TASKS	EVENT TIME (MIN.)	ACCUM TIME (MIN.)	EVA ACCUM TIME (MIN.)
45	Space suits donned. Suit and PLSS check-out.	120.0	3160.0	
46	CM depressurized. EVA hatch opened. Tether rigged.	64.0	3224.0	
47	Astronaut "C" exits and maneuvers to SM, removes propulsion unit, straps to self. Maneuvers to image capsule and secures unit to truss.	24.0	3248.0	593.0
48	Old propulsion unit removed, new unit installed, old unit secured to truss. Returns to CM.	50.0	3298.0	643.0
49	CM hatch closed. CM pressurized.	34.0	3332.0	
50	Suits doffed. PLSS regenerated. All systems checked. Biomed checks. Crew and ground critique.	60.0	3392.0	
51	Eat, personal hygiene, housekeeping, biomed.	90.0	3482.0	
52	Space suits donned. Suit and PLSS check-out.	120.0	3602.0	
53	CM depressurized. EVA hatch opened. Tether rigged.	64.0	3666.0	
54	Astronaut "A" exits and maneuvers to SM, removes and attaches Station #3 scientific package to conveyor. Maneuvers to main truss center frame and alters location for accessibility to image capsule. Bay I. Hauls cargo to capsule and secures.	44.0	3710.0	687.0
55	"A" secures self to Bay I elevator door, engages portable power tool and opens door, exposing turret area.	10.0	3720.0	697.0
56	Turret Station #3 scientific package removed and temporarily secured to nearby truss.	14.0	3734.0	711.0
57	Turret Station #3 scientific package installed. Operates turret for clearance check. Turret left at #4 Station. Door re-positioned.	32.0	3766.0	743.0
58	Astronaut "A" disengages from door. Removes old package from temporary stowage on strut and returns with package to CM (or SM).	25.0	3791.0	768.0
59	CM side hatch closed. CM pressurized.	35.0	3826.0	
60	Space suits doffed. PLSS regerated.	60.0	3886.0	
61	Eat, personal hygiene, housekeeping, biomed.	60.0	3946.0	
62	Sleep.	480.0	4426.0	
63	Eat, personal hygiene, housekeeping, biomed.	60.0	4486.0	
64	Space suits donned. Suit and PLSS check-out.	120.0	4606.0	
65	CM depressurized. EVA hatch opened. Tether rigged.	64.0	4670.0	
66	Astronaut "B" exits and maneuvers to SM, removes and attaches Station #4 scientific package to conveyer. Maneuvers to image capsule Bay #I. Hauls cargo to capsule and secures.	44.0	4714.0	812.0

Table 6-2. Time Line Analysis - Resupply (cont'd.)

EVENT NO.	EVENT AND TASKS	EVENT TIME (MIN.)	ACCUM TIME (MIN.)	EVA ACCUM TIME (MIN.)
67	"B" secures self to Bay I elevator door, engages portable power tool and opens door, exposing turret area.	10.0	4724.0	822.0
68	Turret Station #4 scientific package removed and temporarily secured to nearby truss.	14.0	4738.0	836.0
69	Turret Station #4 scientific package installed. Operates turret for clearance check. Turret left at #5 Station. Door repositioned.	32.0	4770.0	868.0
70	Astronaut "B" disengages from door. Removes old package from temporary stowage and returns with package to CM (or SM).	25.0	4795.0	893.0
71	CM side hatch closed. CM pressurized.	35.0	4830.0	
72	Space suits doffed. PLSS regenerated.	160.0	4890.0	
73	Eat, personal hygiene, housekeeping, biomed.	90.0	4980.0	
74	Space suits donned. Suit and PLSS check-out.	120.0	5100.0	
75	CM depressurized. EVA hatch opened. Tether rigged.	64.0	5164.0	
76	Astronaut "C" maneuvers to SM, removes and attaches Station #5 scientific package to conveyor. Maneuvers to image capsule Bay I. Hauls cargo to capsule and secures.	44.0	5208.0	937.0
77	"C" secures self to Bay I door, engages portable power tool and opens door, exposing turret area.	10.0	5218.0	947.0
78	Turret Station #5 scientific package removed and temporarily secured to nearby truss.	14.0	5232.0	961.0
79	Turret Station #5 scientific package installed. Operates turret for clearance check. Door repositioned.	32.0	5264.0	993.0
80	Astronaut "C" disengages from door. Removes old package from temporary stowage and returns package to CM (or SM).	25.0	5289.0	1018.0
81	Astronaut "C" maneuvers to terminal end of conveyor line, detaches and returns to SM. Retracts conveyor line anchor device and secures. Returns to CM.	34.0	5323.0	1052.0
82	CM side hatch closed. CM pressurized.	34.0	5357.0	
83	Suits doffed. PLSS regenerated	60.0	5417.0	
84	Eat, personal hygiene, housekeeping, biomed.	60.0	5477.0	
85	Sleep.	480.0	5957.0	
86	Eat, personal hygiene, housekeeping, biomed.	60.0	6017.0	
87	Mission instrument checkout. All x-ray telescope systems "on". CSM ACS "off".	5.0	6022.0	

Table 6-2. Time Line Analysis - Resupply (cont'd.)

EVENT NO.	EVENT AND TASKS	EVENT TIME (MIN)	ACCUM TIME (MIN)	EVA ACCUM TIME (MIN)
88	Ground command to lock-on prime reference source.	5.0	6027.0	
89	Turret Station #1 commanded into receiving position and "on". Stored data.	15.0	6042.0	
90	Turret Station #2 commanded into receiving position and "on". Stored data.	15.0	6057.0	
91	Turret Station #3 commanded into receiving position and "on". Stored data.	15.0	6087.0	
92	Target not available. Verification from ground station that turret Stations #1, 2 and 3 receiving.	30.0	6117.0	
93	Turret Station #4 commanded into receiving position and "on". Stored data.	30.0	6147.0	
94	Turret Station #5 commanded into receiving position and "on". Stored data.	30.0	6177.0	
95	Target not available. Verification from ground station that turret Stations #4 and #5 receiving.	30.0	6207.0	
96	Turret Station #6 commanded into receiving position and "on". Stored data.	15.0	6222.0	
97	Turret Station #7 commanded into receiving position and "on". Stored data.	15.0	6237.0	
98	Critique and verification from ground that Stations #6 and #7 are receiving.	45.0	6282.0	
99	Eat, personal hygiene, housekeeping, biomed.	90.0	6372.0	
100	Preparation for CSM separation from telescope.	30.0	6402.0	
101	Space suits donned. Suit systems check.	120.0	6522.0	
102	Docking locks and electrical umbilical disconnected. Pressure hatch installed.	60.0	6582.0	
103	CSM undocked. Crew maneuvers CSM in fly-around inspection. Photographic coverage.	90.0	6672.0	
104	Detached operational check. CSM crew on standby as telescope is commanded from ground inputs to cycle all instruments through operation and verify systems operational. Assumed accomplished in approximately two orbits.	18.0	6852.0	
105	Mission terminated.			

Total time to operational status:	4 days	18 hours	12 minutes
Accumulated EVA time outside of CM		17 hours	32 minutes

NOTES:

(1) See Appendix II for detailed work sheet of the resupply mission.

(2) See Figure 6-2 for summary bar graph of the resupply mission.

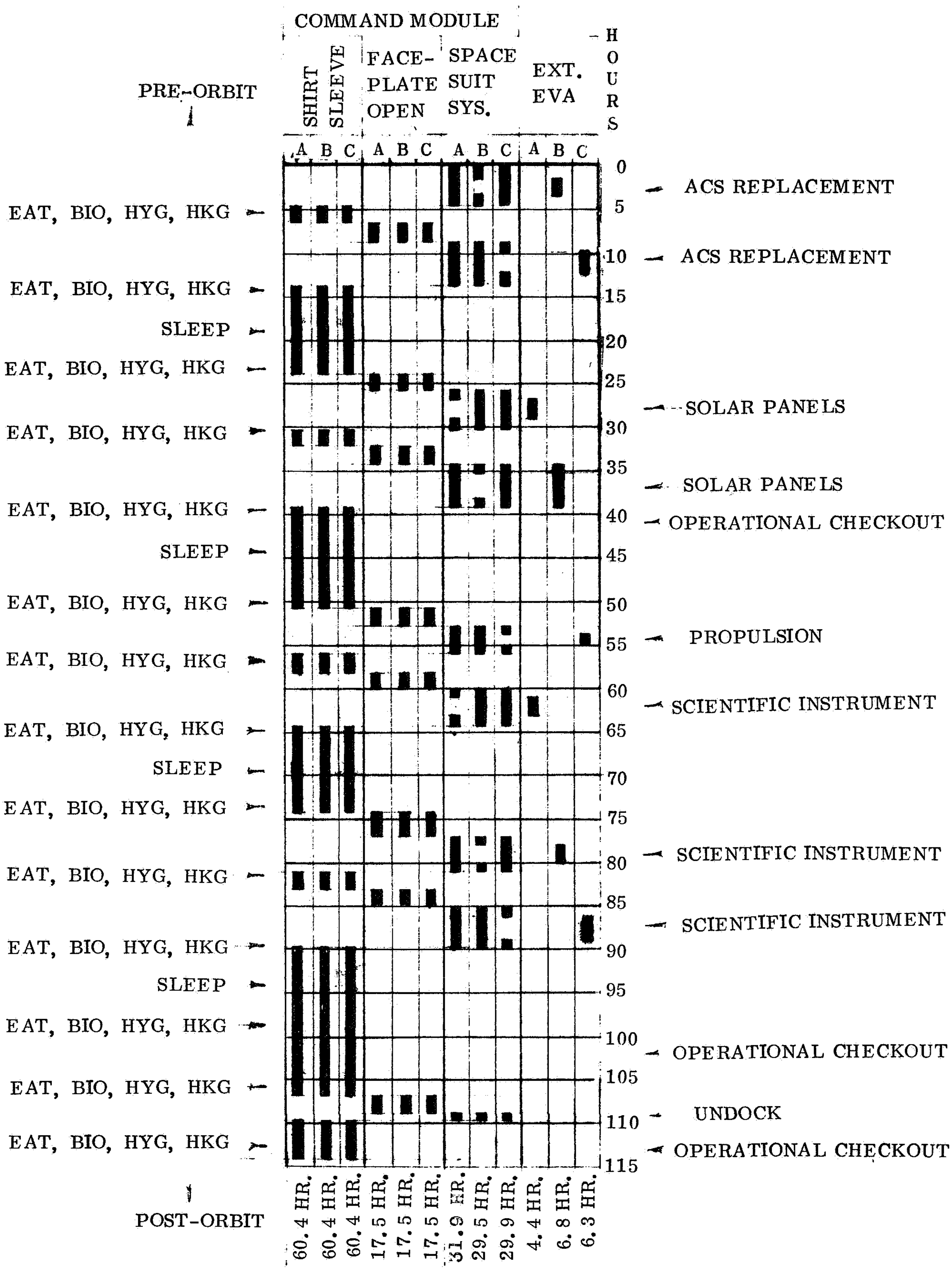


Figure 6-2. Resupply Mission Summary.

6.4 ASTRONAUT SUPPORT EQUIPMENT REQUIREMENTS. In addition to the basic pressure garment assembly, certain equipment is necessary to support the EVA astronaut in accomplishing the inspection, repair, and/or replacement of damaged components and refurbishment of the x-ray telescope. Table 6-3 summarizes the estimated astronaut primary equipment requirements.

The (PLSS) will supply pressurized oxygen to the pressure garment assembly (PGA), clean and cool the expired oxygen, circulate cool liquid in the liquid-cooled garment (LGG) and function as a transceiver for the astronaut bioinstrumentation and communication. The operating time of this unit is 5 hr. at 1200 Btu/hr.; 3.25 hr. at 1600 Btu/hr.; and 2 hr. at 2000 Btu/hr. The oxygen and water systems may be recharged in either the Airlock Module or Command Module.

The use of a life support umbilical instead of the PLSS is preferred, but the current Apollo CM does not have the necessary umbilical provisions. It is recommended the CM be modified to allow the use of EVA umbilicals which will provide oxygen, H_2O, biomedical communications and serve as a safety tether. The umbilical will afford the astronaut more dexterity, mobility, and access to more confined areas.

The worksite anchors consist of a waist restraint which provides astronaut torso stability in any given position while leaving both hands free for work. This restraint is an assembly consisting of a waist harness with two controllable flexible legs approximately 45 in. long. The application of this equipment at the worksite will aid the astronaut in maintaining body position with a minimum of energy expenditure. It has the advantage when compared to flexible waist tethers, of maintaining the body in a specific position with respect to the worksite and of freeing both hands for any required work tasks. The variable waist restraint is usually used in conjunction with foot restraints which are "built into" the vehicle or attached to the structure in the area desired.

The PLSS communications system provides voice communication between the EV astronaut, the CM pilot, and his backup safety man. A reliable communication system is vital for mission success and the safety of the astronauts.

The low earth orbit (260 n.mi.) of the telescope average sixty min. of daylight and thirty min. of night (dark) per orbit. EVA is limited to daylight, the refurbishment tasks may require more than 12 days to accomplish, therefore illumination is assumed to be required on the x-ray telescope structure. EVA schedule will be planned to take maximum advantage of daylight. Other lighting will consist of helmet-mounted and chest mounted lights and some other portable lights. The helmet-mounted lights are preferred to minimize tool or limb interference with the illumination of the tasks.

Table 6-3. Astronaut Support Equipment

EQUIPMENT	ESTIMATED QUANTITY REQ'D.	INSTALLED OR STOWED LOCATION	POWERED OR RECHARGED BY	REMARKS
1. Portable Life Support System	3	CM	CM (SM)	Contains communication system
2. Work Site Anchors				
Fixed	4 Sets	Image Capsule		Built in Dutch shoes on access doors
Portable Dutch Shoes	2 Sets	CM		Clamp on truss tubing
Portable Waist Restraints	2 Sets	CM		
3. Illumination System				
Fixed	3 Systems	Telescope	Telescope EPS	Image capsule, overall truss, & fwd. lens
Portable	4	CM	CM	2 helmet & 2 chest mounted lamps Rechargeable battery type
4. Rotary Reactionless Tool-Portable	2	CM	CM	Torque unit for powering failed systems, Removing fasteners, etc. (rechargeable)
5. Hand Held Maneuvering Unit	3	CM	CM	Safety item - hand locomotion is primary mode
6. 50 Ft. Tether	3	CM		Nylon tape
7. Tools, Misc.		Worksite		Tools req'd. for scheduled equipment changes will be incorporated into the design or attached at the worksite.
8. Tools, Emerg. Repair		CM or Telescope		Certain tools will be carried for emergency repair; such as tube splicing clamps, etc. (unscheduled usage).

The x-ray telescope design philosophy assumes that small conventional tools generally present more problems than they solve, therefore the design of the vehicle and equipment is intended to eliminate the need for an astronaut to carry tools wherever possible.

Many concepts for modular replacement, and quick disconnecting fasteners are available, and additional concepts have been predesigned to the extent that the use of wrenches and screwdriver type tools can be eliminated. Certain module replacement that can be done most effectively with a tool will have the tool designed integral or at least attached at the work site to reduce the astronauts portable inventory. Special emergency tools for replacing damaged structure, tubing, and possibly wiring would be carried in the CM.

The use of an AMU was evaluated for the telescope mission and rejected. All EVA is performed with the CM docked to the telescope and there are no requirements for astronaut locomotion which cannot be satisfied by hand locomotion. Early solar panel designs which required an AMU to change panels were revised to retract to the predeployed position for servicing. In the interests of simplicity and safety it is believed that AMU should not be used unless a firm requirement exists, this design does not have such a requirement.

SECTION 7

RESEARCH, DEVELOPMENT, TEST AND ENGINEERING

7.1 INTRODUCTION. This chapter presents the RDT&E plan for the x-ray telescope documenting the Task 5, approved study work statement items.

The RDT&E plan provides a focal point for the planning documents and information necessary to define all steps required to achieve a functioning orbital telescope as part of the AAP. Definition is required in sufficient detail to support NASA resource allocations between the various alternatives to identify requirements for manpower, research development and test facilities, and to define schedule interactions and budgetary planning data to achieve, for AAP, maximum utilization of resources.

The RDT&E plan provides:

a. Work breakdown structure.

b. Prerequisite orbital experiments.

c. Research, manufacturing, test, and support plans.

d. Schedule.

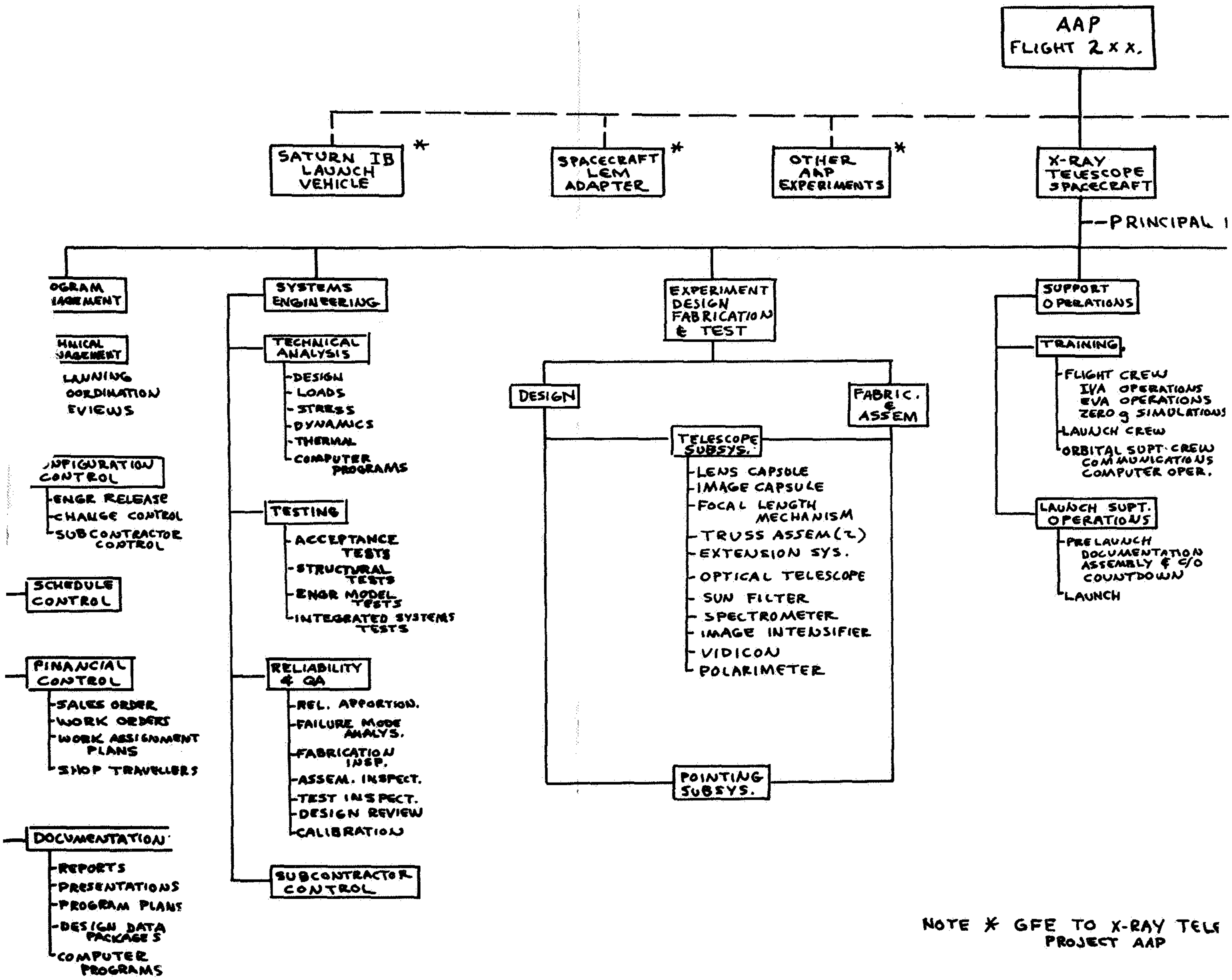
e. Cost analysis.

The work breakdown structure incorporates the system elements and identifies the tasks associated with each. Based on guidelines established in the preceding design chapters, in the system specification and the supplementary research, manufacturing, test and support plans of this chapter, the schedule and cost data (keyed to the work breakdown structure) have been generated.

The level of definition throughout the RDT&E plan is selected to be commensurate with the planning indicated in the design chapters, and tailored to subsequent completion of NASA Form 1346 for the x-ray telescope.

7.2 WORK BREAKDOWN STRUCTURE. A work breakdown structure for the x-ray telescope is shown in Figure 7-1. Development of the telescope is assumed to be within the AAP so that the other major segments required for flight are GFE to the telescope project. Incorporation of other experiments into the same launch vehicle, while shown as potentially feasible, is also assumed to be GFE. In addition to basic hardware the following elements are shown:

PART B



FOLDOUT FRAME 1

PART A

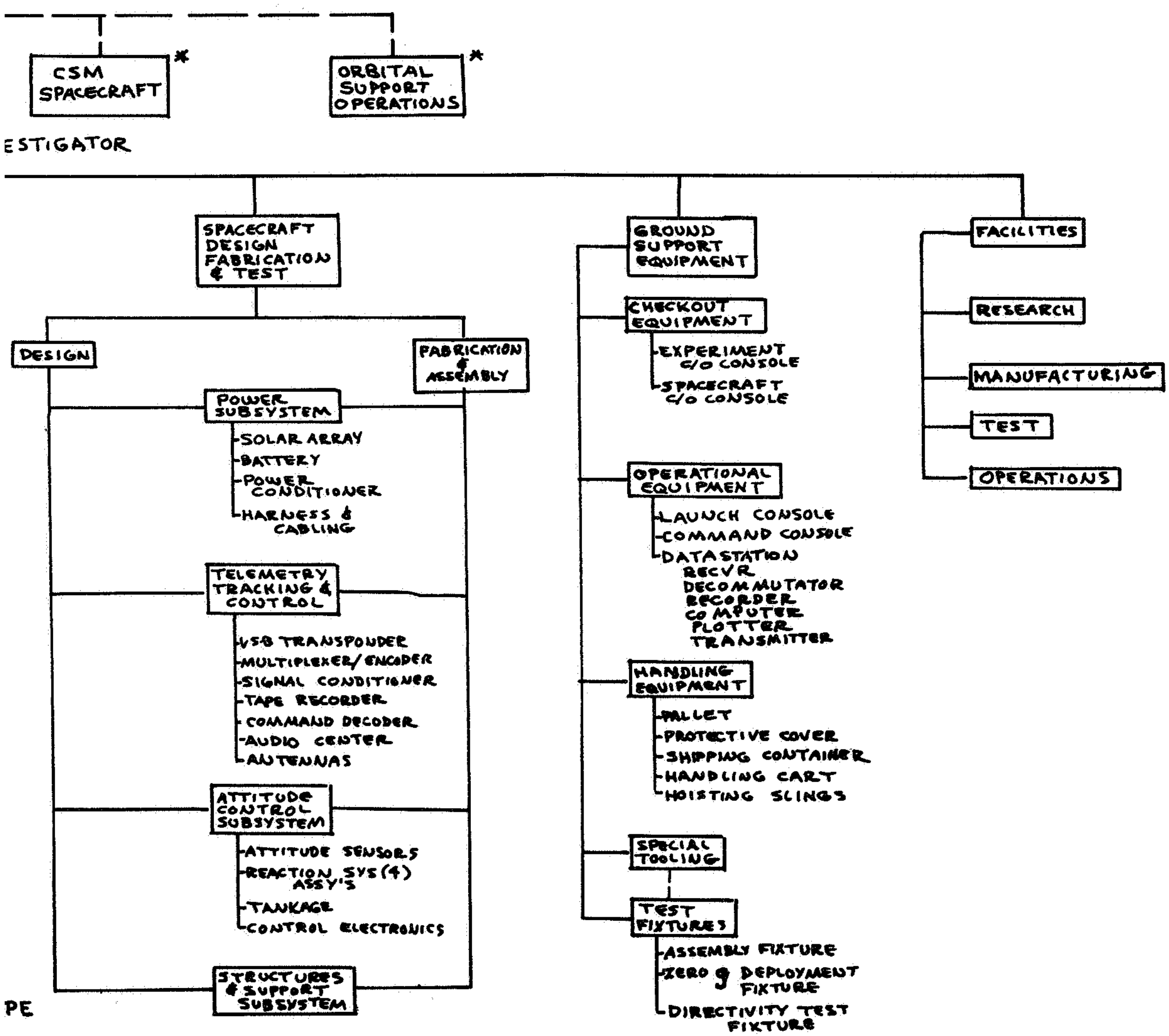


Figure 7-1. Work Breakdown Structure - Focusing X-ray Telescope

FOLDOUT FRAME 2

a. Program Management. This element includes all of the contractor's program office activities in general categories, of technical management, control and documentation.

b. Systems Engineering. Two work areas are defined--contractor control and subcontractor control. In the first, systems engineering performs technical analyses and initiates design requirements, reliability standards are established; verification of design is achieved through an integrated test plan. Subcontractor control is achieved through selection, surveillance and product inspection.

c. Support Operations. This element includes those auxiliary operations necessary to the accomplishment of the telescope mission--training of personnel, preparation of procedures, participation in pre-launch and launch activities.

7.2.1 Aerospace Equipment. The vehicle-borne equipment is shown under two different classifications. First, the telescope equipment is shown with associated design, fabrication and assembly tasks. The mirror assembly, adjustments and associated components are included in this group. Second, the vehicle equipment is similarly defined. This equipment is used in support of the orbital telescope and typically includes telemetry, tracking, command and power subsystems.

7.2.2 Ground Support Equipment. Ground support equipment (GSE) typically includes checkout equipment, operational equipment, handling equipment and other special equipment. To the greatest extent practicable, fixtures will be used both for tooling/ assembly and test.

7.2.3 Facilities. While the four standard types of facilities are listed, no new facility items are currently identifiable

7.2.3.1 Zero-g Deployment Facility. The structure deployment will be accomplished upward in a vertical position with pulley and track systems to accommodate the appropriate counterweights. The vertical deployment will require 45 ft. of overhead clearance which is not considered a new facility requirement.

7.3 PREREQUISITE ORBITAL EXPERIMENTS. The analysis of the x-ray telescope has not established firm requirements for prerequisite orbital experiments, however, several experiments have been identified which would contribute useful data to the design of the telescope structure and the planning of astronaut capabilities.

7.3.1 Astronaut Locomotion Loads.

a. Purpose - To determine actual loads on typical long tube lightweight spacecraft structures resulting from astronaut locomotion and cargo transfer. Results will be used to generate realistic structural design load limits.

b. Equipment - A structural assembly is used in conjunction with the S-IVB Workshop. This assembly is erected and hard-mounted to the external surface of the S-IVB. It consists of an open framework of structural members of various shapes and lengths (extruded, flat and cylindrical sections). Typical handholds and tether attachment points are included to facilitate normal astronaut activities.

c. Procedure - The astronaut assembles the structure in sections within the workshop. These sections are then transported and attached to the external surface of the workshop and the assembly is completed as part of EVA. Signal and power cables are attached and the instrumentation cable is attached. The astronaut then performs usual maneuvers about the experimental structure. Upon completion of the experiment the data tape is transported or transmitted to the appropriate ground facility for reduction and evaluation.

d. Measurements - The following measurements are made:

1. Optical deflections/strain gage

2. Shock (10 to 50 G accelerometers)

3. Triaxial accelerations (0 to 10 G accelerometers)

4. Temperatures

7.3.2 Clothesline Supply.

a. Purpose - To demonstrate the practicability of supplying a remotely positioned EVA astronaut with replacement modules as required from the CSM or S-IVB workshop through the use of a "loop" clothesline.

b. Equipment - Basic hardware consists of a loop of line under tension between two sheaves and a traveller to which items are attached for transport, or a similar device possibly consisting of a single line with a powered traveller which is controlled to traverse the line either to or from the work area from the cargo area. The precise design of the cargo handling device is not important at this time, only that a test of this type be performed.

c. Procedure - An EVA astronaut moves to a remote position with the clothesline assembly attached by tether to the CSM. When one sheave is fastened to the structure, the second astronaut draws in the other sheave by retracting the tether. The loop of line feeds out from a spring-loaded take-up spool in the traveller. The clothesline is then used for transferring materials.

d. Measurements - Some of the parameters that can be adjusted and evaluated are:

1. Line tension (tiedown loads on structure).

2. Damping in the take-up spool.

3. Effect of vehicle ACS maneuvering on the line tension while materials are being transferred.

4. Time to install, reposition, retract and return the line to storage.

7.3.3 Equipment Replacement.

a. Purpose - To demonstrate the astronaut's capability to replace equipment in the image capsule of the x-ray telescope.

b. Equipment - Hard mockup or structural test model of the image capsule with typical components installed.

c. Procedure - The test model is transported to orbit with an S-IVB Workshop or CSM. In orbit the model is secured to the external surface of the parent spacecraft. The astronaut then performs the procedural activities for opening the capsule, removing modules, replacing modules and closing the capsule.

d. Measurements - Qualitative measurements are made, as follows:

1. Accessibility

2. Astronaut maneuverability

3. Equipment replacement times.

7.4 RESEARCH PLAN. The design of the majority of the x-ray telescope subsystems are based on current state-of-the-art technology; these are:

a. Structure

b. Navigation

c. TLM, Data and Communications

d. Attitude Control

e. Command

f. Lens Alignment

g. Drift Error Compensation.

Although the electrical power system components are current state-of-the-art, the performance of the proposed solar cell array (13.5 watts/ft.^{2}) is based on industry anticipated estimates for 1972. The feasibility of the design is not affected by this performance projection, however, as the array area may be increased to perform the mission using 1967 solar cell specific power.

The two proposed subsystems which do not fall as precisely into the present day technology are the mirror assembly and the research instrumentation. Since grazing incidence mirrors larger than 9 in. internal diameter have not been manufactured to date, the multisegment 30 in. nominal mirror system proposed represents an extension of current capabilities. It is conceded by the leading designers and producers of focusing x-ray optics that the 20 to 40 in. aperture range of interest here is well within the limits of current manufacturing capability. The problem of multiple nested segments, perhaps as many as 10, involves new technology, but this technology is not a "high risk" area as a suitable 2 or 3 segment mirror can be manufactured--the technology question is how large a diameter and how many segments are optimum or practical.

The further study areas relative to the mirror assembly should include:

a. Diameter and number of segment tradeoff analysis.

b. Material analysis

    1. Reflectivity

    2. Specular characteristics

    3. Grain size

    4. Dimensional stability

c. Structures analysis

d. Optical tolerance analysis

e. System, fabrication and test techniques

f. Image energy distribution.

The remaining subsystem of interest is the primary mission equipment, imaging, spectrometers, and polarimeters.

In all cases equipment currently exists which will perform the desired mission to varying degress of effectiveness. The x-ray telescope proposed for post 1973 flight includes assumed equipment capabilities which are an extension of the current capabilities--consequently these are considered development items as opposed to pure research areas.

The instrumentation system components and performance have been synthesized by consultation with the majority of U. S. x-ray astronomers and consequently are considered realistic in terms of physical size and performance.

Specific areas of further study relative to the instrumentation should include

a. Long memory high resolution imaging orthicons.

b. Improved packaging of high $\lambda/d\lambda$ spectrometers.

c. More efficient polarimeters to reduce imaging times.

d. Development of compact light weight cryogenic cooling systems to permit use of liquid hydrogen polarimeters.

While not specifically required, research in the following areas will contribute to performance, cost, or schedule improvement for the x-ray telescope.

a. Passive techniques for minimizing large space structure distortions. The "heat pipe" system is an example but one which may be too sensitive to minor micrometeorite punctures.

b. Compatability of telescope with S-IVB Workshop or space station--this may permit the efficient use of film which would solve some instrumentation problems.

c. Grazing incidence lens performance degradation in space environment.

d. Imaging equipment shielding requirements--definition of local anticoincidence shielding for equipment and shielding requirements of optical path for the open telescope structure.

e. Mechanisms, motors, and equipment designs for long term space use (up to four years).

f. Development of structural materials with zero coefficients of thermal expansion.

7.5 MANUFACTURING PLAN. Manufacturing processes required to fabricate components of the x-ray telescope are essentially those processes and techniques used extensively by airframe and aerospace manufacturers. The principal approach followed in the construction of the structure is to erect it in a vertical, sub-assembly fashion. The basic tubular cage form (20 ft. x 40 ft.) consists of an image capsule truss assembly and an extendable lens capsule truss assembly with solar panels mounted to swingout "A" frames. The vertical fixturing concept serves as both assembly tooling and operational checkout facility. Prevention of tolerance build-up through close control of each element of fabrication is the key factor in the assembly operation.

A "pilot line" concept applied to manufacturing the x-ray telescope experiment will utilize some sub-assembly positions. Assemblies will be complete and flow to the final assembly position when the complexity, manufacturing processes or test requirements indicate this to be advantageous.

The x-ray telescope experiment manufacturing sequence (Figure 7-2) depicts an appropriate manufacturing breakdown and sequence of manufacturing events. Some events, such as structural frame assembly, x-ray telescope assembly, and checkout and test, are performed in the same manufacturing position and area without assembly movement. Other events will occur in various manufacturing areas due to the difference in the nature of the manufacturing processes involved.

7.5.1 Fabrication Procedures. Detail parts are fabricated in shop areas performing similar tasks.

Special Programs Pilot Line type planning is used. This type planning provides an original copy for the planning files and carbon copies for Materials, Dispatching, Quality Control and the Shop. Planning paper is prepared by Operations Planning to call out the material, define fabrication procedures, and establish inspection check points. All verification of inspection operations is recorded on the planning paper, which become permanent records.

Identification of parts, including material identification stamping, is accomplished as early in the fabrication cycle as possible. The part reflects the same identification number carried on the planning paper.

Maximum use of skilled machinists, employing standard tools and machine setups minimizes special tooling requirements and manufacturing costs. Optical techniques and a laser measurement system are used for linear alignment, parallelism, and perpendicularity of parts and assemblies to further reduce the need for special tooling.

7.5.2 Detail Fabrication. The image capsule truss assembly is fabricated of thin wall titanium tubing mechanically joined or bonded to socket type cluster fittings. Four built-up extruded titanium rails are included for pull-cable extension of the lens capsule truss structure fixtured per optical alignment means. Included are:

a. Image Capsule.

b. Extension Rails (4).

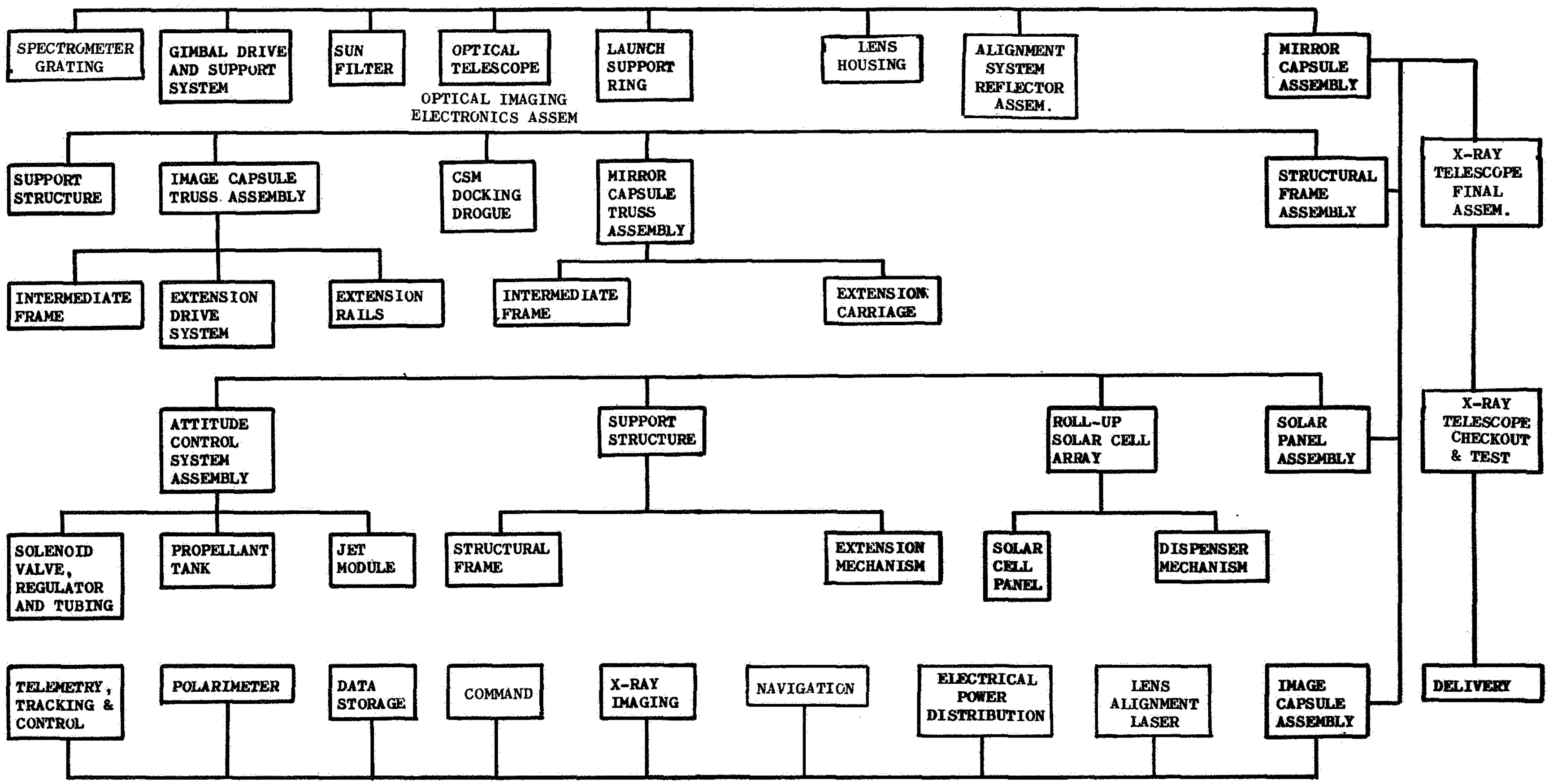


Figure 7-2. X-RAY TELESCOPE MANUFACTURING SEQUENCE

c. Intermediate Frame.

d. Extension Drive System.

e. Docking Drogue (GFE)

The mirror capsule truss assembly is also fabricated of thin wall titanium tubing and joined in like manner of the image capsule truss. Slider extension carriages are integral with the structure. The lens capsule and optical telescope will be simulated for assembly purpose, however, the capsule launch support ring and gimbal drive system are included with the following:

a. Lens Housing (simulated).

b. Launch Support Ring.

c. Optical Telescope (simulated).

d. Gimbal Drive and Support System.

e. Extension Carriages (4).

The image capsule is constructed of standard aluminum members with integral truss fittings and sliding exterior panels. The structural body is approximately 4.5 ft x 4 ft. containing equipment superstructure and platforms to be fabricated. Assembly build-up uses a conventional form assembly fixture with controlled interface points for the docking drogue and truss support fittings. Drilling operations are accomplished with coordinated drill plates and minimal machine jig boring. Insulation panels as well as the internal equipment are not installed at this level, however, mock-up units will suffice for mounting provisions and check fit.

The extension rails consist of titanium extruded sections requiring a minimum of mill facing. A surface coating of Teflon (or similar) material and conventional cam rollers are required for glide carriage action. Drilling of attach point holes in addition to mounting the associated extension drive system bracketry constitutes sub-assembly operations.

The intermediate frame contained in the image capsule truss assembly is approximately twelve ft. diameter of conventional inner/outer ring and web design. Rings of "T" and angle section (titanium) lend themselves to normal roll forming followed by finish machined turning. Titanium web sections require standard blank and form lightening holes. A table form fixture accommodates assembly control of the riveted structure.

Extension carriages are primarily composed of machined details of a channel section with over-ride flanges for track retention. Numerical tape control is applied for consistent milling operations of approximately 4 ft. sections.

7.5.3 Final Assembly. Upon completion of the image and mirror capsule structure assemblies, the structures are mated (interconnect structure) and supported on a fixture to be used during experiment final assembly, checkout and shipping. Consideration of the use of the actual launch support for this fixture eliminates transfer of the assembly later. Mechanical and electrical components as depicted in the manufacturing sequence are installed.

Image capsule truss assembly and test requires special attention to fabrication and assembly methods. Precise control of critical interface areas is required to assure frictionless and unhampered movement during extension and retraction. This is best accomplished by constructing all truss section assemblies in a single special fixture having the stability and tolerances of a "tooling dock". Reference points are established to enable adjustment of rollers and gears so that only minor adjustment, if any, is needed at final assembly.

7.5.4 Material Handling and Packaging. Material handling and packaging is controlled by National Aerospace Standards (NAS). These standards establish the methods, materials, and devices to be used throughout the procurement, receiving, manufacturing and shipping phases of the program. Supplier packaging standards (NAS) applied to all procurement initiating documentation ensure that material is packaged for damage-free delivery to the plant, and the material is packaged so that maximum use of supplier packaging is made during receiving, receiving inspection, and storage functions.

In-plant handling and packaging standards are used as material flows from receiving, through receiving inspection, stores and manufacturing cycles. Manufactured detail parts are similarily protected during fabrication and temporary storage. A handling and shipping device is used for in-plant movement and shipment of the telescope in its packaged configuration.

Preparation for delivery instructions provide protection against damage and degradation during shipment to destination. For delivery of telescope to the destination the handling/shipping device is secured to a skidded base suitable for handling by crane or fork-lift truck. The telescope is shrouded with a barrier material to exclude dirt or other foreign contaminants. Container sides, ends and top are provided to protect against damage. The solar panel arrays, explosive hardware, battery, and miscellaneous hardware are packaged and shipped in separate containers. Containers are marked in accordance with MIL-STD-129, including hazardous warnings and shipping piece numbers, to ensure proper handling and ready identification at destination.

7.5.5 Make or Buy. Make-or-buy policy and directives result from review of customer policies, government regulations, the technical and functional requirements of systems and subsystems, and the evaluation of capabilities available in industry. Coordination is continually maintained with the Small Business Liaison Officer so that small businesses may be given an equitable opportunity to participate in the program. Further, Make or Buy Administration participates with the Material Department in reviewing products and capabilities presented by outside vendors to ensure recognition and knowledge of alternative industry sources.

The make-or-buy structure makes full use of products of reputable manufacturers regularly engaged in commercial production of equipment and components required for this program.

7.5.6 Tooling. Major assembly buildup is conceived to be erected in a vertical dock type fixture (Figure 7-3) with hard points controlled by tooling gages and applied optics. Lens and image capsules will be simulated for alignment purpose and interface patterns controlled with master gages.

Assembly tools for the image capsule, solar panel frames, extension carriages, intermediate frame, etc., are of conventional type. Locators provide positioning and securing of parts in conjunction with coordinated drilling operations.

Numerous dimensional control type tools are expected to be applied throughout detail fabrication to minimize tolerance buildup. Examples are: bonding fixture, checking gages, and templates, drill plates, Go/No Go tools and machine tapes.

7.5.7 Facilities. Facilities to fabricate and assemble this configuration consist primarily of standard sheet metal and machine shop plant equipment. Existing equipment would be utilized for such typical operations as cutting, roll forming, drilling, and machining. Numerically controlled 3-axis single mills could be used advantageously to final machine the precision cast cluster fittings. The predominant structural material is titanium, but part size and material thicknesses would not exceed the rigidity and overall capabilities of a major contractor's equipment.

Structural assembly requires high bay factory space with an overhead bridge crane. Depending on assembly sequence and tooling break points, a truss height up to approximately 42 ft. is required.

If the titanium tube members are bonded to socket fittings as now planned, an autoclave or some similar bonding facility would be required. Autoclave size is governed by tooling break points and final design, but it should be possible to plan this operation to fit the typical 8 to 10 ft. diameter autoclaves available in most plants.

An environmentally controlled assembly area equivalent to Federal Standard 209 class 100,000 clean rooms is required for image capsule assembly. Fabrication and assembly of electronic components would also be performed in a clean area. Fabrication of electronic components, utilizing integrated circuits, thin film, and other state-of-the-art design, is not expected to exceed established capabilities.

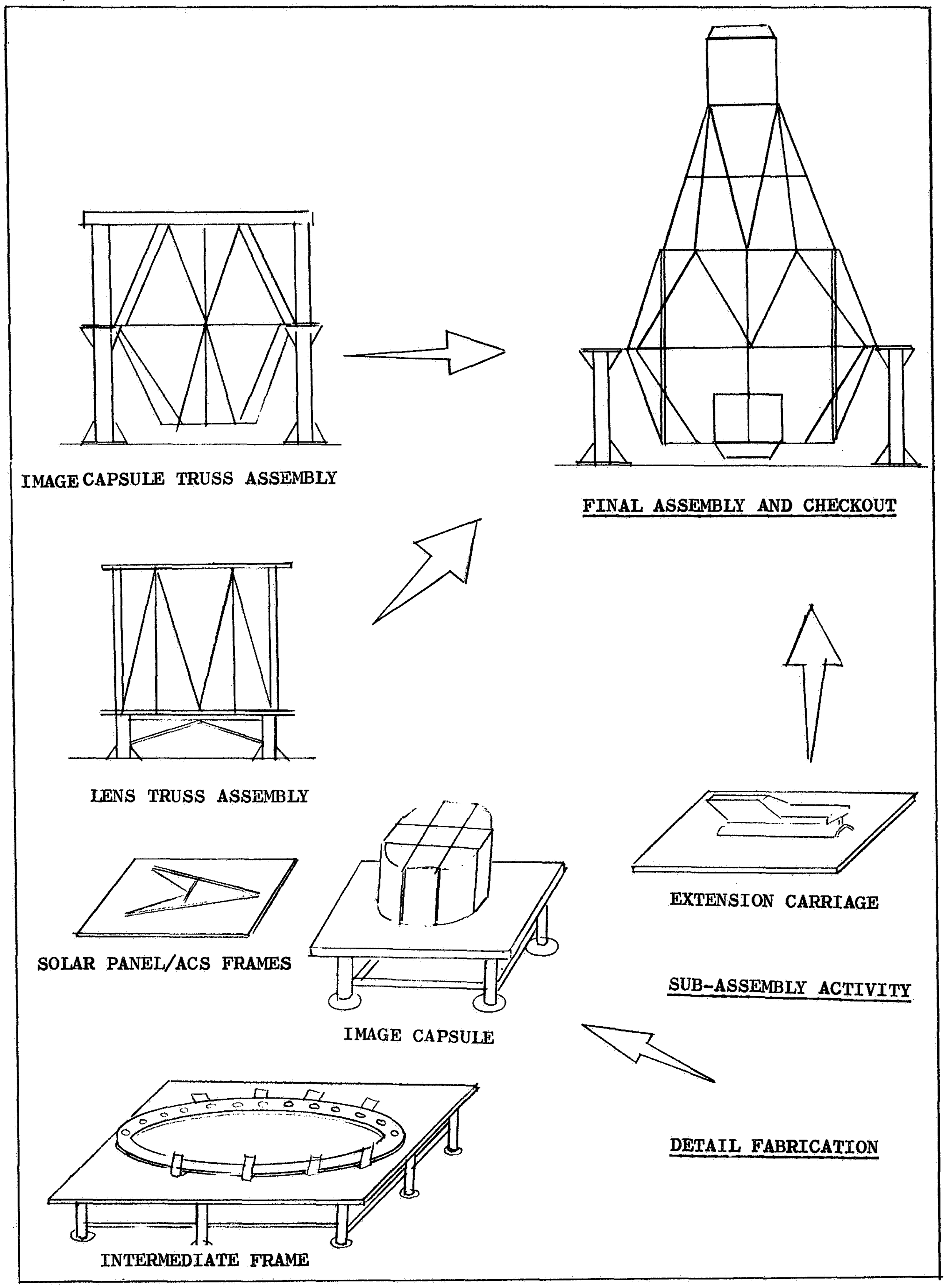


FIGURE 7-3 VERTICAL FIXTURING CONCEPT

7.6 TEST PLAN. A test program is conducted to provide design reliability, and quality assurance such that the equipment will survive the pre-launch, launch, and space environments with no malfunctions or deteriorating effects; deployment can be effected with the planned amount of EVA; and that the structure and equipment will operate within specified tolerances. Testing is divided into the following three types of tests:

a. Development tests.

b. Qualification tests.

c. Acceptance tests.

Component, subsystem, and system level testing is performed on the following test articles:

a. Subsystem test articles.

b. Engineering model (or prototype).

c. Flight article.

Table 7-I identifies the proposed tests.

7.6.1 Development Tests. Development tests are conducted on the components and subsystems in the prototype stages in order to evaluate the suitability of the units for use in the flight article.

7.6.1.1 Component Tests. Some of the basic components that are subjected to development type of tests include the following:

a. Digital computer.

b. Logic unit.

c. Optical telescope.

d. X-ray imaging orthicon.

e. Polarimeter.

f. Extension and gimballing motors.

g. Thermal control louvers.

Table 7-1. TEST PLAN SUMMARY- X-RAY TELESCOPE

TYPE OF TESTS / TEST ARTICLES	DEVELOPMENT TESTS						QUALIFICATION & ACCEPTANCE TESTS																			
	STRESS ANALYSIS	STRUCTURAL INTEGRITY STATIC & DYNAMIC LOADING	THERMODYNAMIC EVALUATION	FUNCTIONAL TESTING ELECTRICAL & MECH.	FAILURE MODE ANALYSIS	MAINTENANCE CAPABILITIES	EXAMINATION OF PRODUCT ■	FUNCTIONAL TEST SATISFACTORY PERF. ■	TEMPERATURE, HUMIDITY, ALTITUDE	VIBRATION: PACKAGED, SHIPPING & BOOST	ACCELERATION TEST: BOOST PHASE	SHOCK TEST: HANDLING & PACKAGED	ENVIRONMENTS: PACKAGED & RAIN	SAND & DUST	SALT ATMOSPHERE	MOISTURE & FUNGUS RESISTANCE	THERMAL VACUUM	SOLAR RADIATION	VIBRATION-DEPLOYED SINE/RANDOM	DEPLOYMENT TESTS ZERO g SIMULATION ■	RF TESTING	EMI TESTING	ACOUSTICAL NOISE TESTS	LIFE TESTS	STATIC LOADING LIMIT	PROOF TESTS ■
COMPONENTS																										
STAR TRACKERS & COMPUTER				X	X		X	X	X								X	X	X	X	X	X	X	X		X
OPTICAL TELESCOPE			X	X			X	X	X	X	X		X				X	X	X	X						X
SLITLESS SPECTROMETER			X	X			X	X				X		X			X	X	X	X						X
H LASER & ELECTRONICS				X			X	X		X	X				X		X	X		X	X	X	X			X
EQUIPMENT TURRET		X	X	X			X	X		X	X	X				X	X	X		X	X	X	X			X
RECEIVERS, TRANSMITTER & ANTENNA							X	X				X					X	X		X	X	X	X			X
TONE DECODER							X	X		X	X						X	X		X	X	X	X			X
DIGITAL LOGIC ASSEMBLER							X	X	X								X	X		X	X	X	X	X		X
TAPE RECODER							X	X	X			X					X	X	X	X						X
SOLAR ARRAYS & BATTERIES							X	X		X	X		X	X	X	X	X	X	X	X				X		X
X-RAY IMAGING ORTHICON			X	X			X	X	X								X	X	X	X				X		X
POLARIMETER			X	X	X		X	X	X	X							X	X	X	X				X		X
THERMAL CONTROL LOUVERS	X	X	X				X	X	X								X	X	X	X					X	X
DEPLOYMENT & GIMBALING MOTORS				X		X	X	X	X			X		X	X		X	X	X	X				X		X
PRESSURE REGULATORS, FLOW VALVES, JETS				X		X	X	X	X			X		X	X		X	X	X	X						X
SUB-SYSTEMS																										
LENS ALIGNMENT SYSTEM		X		X			X	X									X			X						X
ELECTRICAL POWER SYSTEM				X			X	X																		X
COMMAND/DATA, TLM & COMMUNICATION				X			X	X																		X
OPTICAL DRIFT ERROR SYSTEM				X			X	X																		X
NAVIGATION/ATTITUDE CONTROL SYSTEM		X		X			X	X												X						X
TELESCOPE INSTRUMENTATION				X			X	X																		X
SYSTEMS/ASSEMBLIES																										
LENS ASSEM-X-RAY TELESCOPE			X			X	X	X												X						X
IMAGE CAPSULE ASSEM.			X			X	X	X												X						X
IMAGE CAPSULE TRUSS & DOCKING ASSEM.	X	X			X	X	X	X																	X	X
LENS CAPSULE TRUSS ASSEM. & SHROUD.	X	X			X	X	X	X																	X	X
SOLAR CELL PANEL/ACS ASSEM.				X		X	X	X												X						X
EXTENSION/DEPLOYMENT SYSTEM				X		X	X	X																	X	X

Stress analysis, structural integrity, static, and dynamic loading tests are performed on the structural components such as truss members and deployment mechanisms. These tests verify the basic design limit loads and the stiffness, weight, and deflection characteristics of the specimens.

Thermodynamic evaluation tests are performed on various electrical and mechanical components to demonstrate the performance under temperature, load and input power conditions. Thermodynamic evaluation is made of such components as the polarimeter, extension motors, and attitude control jets.

Functional testing is performed on all electrical, mechanical, and structural components to verify satisfactory operation of the test specimen for final design requirements.

Each component to be tested is evaluated for maintenance capabilities while in the space environment as well as for ground conditions. Maintenance of faulty equipment is required by EVA using the following replacement techniques.

a. Additive.

b. Replacement.

The above replacement techniques are expected to be required on components as shown below.

	Replacement Techniques.	
ADDITIVE:	SOLAR CELL PANELS	
	SPECTROMETER	
	ACS MODULES:	JET
		CONTROL VALVES
		REGULATORS
	BATTERIES	
	ELECTRONIC PACKAGES (SPACE PERMITTING)	
REPLACEMENT:	TELESCOPE INSTRUMENTATION	
	GIMBALING MOTOR (OUTER GIMBAL)	

7.6.1.2 Subsystems Tests. Initial subsystem tests are performed on prototype and engineering test models. These tests are of an evaluative nature and are also used to check out procedures to be employed in subsequent testing of the flight articles. The major subsystems to be evaluated include the following:

a. Attitude control.

b. Power.

c. Guidance.

d. Data link electronics.

e. Data storage.

f. Lens alignment system.

g. Optical drift error system.

h. Telescope instrumentation.

Full-scale tests are conducted on the above.

The subsystem assemblies are mounted on low-frequency exciters through load transducers and sensitive low-frequency accelerometers are attached to various components and structural members of the subsystem assembly. Vibration forces are introduced and failure modes are defined from the acceleration response. While employing the low-frequency, lightweight exciters, damping is determined with the logarithmic decrement technique from response decay records obtained after the exciter armature circuit is opened. Data obtained will be integrated into calculation for the complete full-scale structure.

Functional operating capabilities tests are performed on the subsystem assemblies in the following areas:

a. Electrical tests.

b. Mechanical tests.

c. Assembly tests.

d. Deployment tests.

The above tests are performed on each subsystem as an independent unit, wherever possible.

Each electrical and mechanical subsystem is operated and the performance monitored under ambient conditions. Where applicable, the subsystems are assembled in steps, demonstrating the proper action during deployment at each phase of assembly.

Subsystems are deployed under room ambient temperature, simulating zero-g conditions by suspension from overhead members with compliant systems attached to the most massive specimen sections. Deployment force and accelerations are measured. Overloads and out-of-tolerance conditions are applied to determine subsystem responses and recovery capability.

7.6.1.3 System Tests. Upon completion of the individual subsystem tests, the subsystems are integrated into functional system/assemblies. Testing of the systems is conducted to show that the simultaneous operation of all system equipment does not produce improper operation of each subsystem or component. The major assemblies to be evaluated include the following:

a. Lens Capsule Assembly.

b. Image Capsule Assembly.

c. Extension Deployment System.

d. Lens Capsule Truss Assembly.

e. Image Capsule Truss Assembly.

f. ACS/Solar Panel Assembly.

The objective of these tests is to demonstrate the capability of each system to withstand the launch and flight environmental conditions.

Structural vibration tests demonstrate the structural integrity of each system. The tests consist of vibration in three axes at design g-levels which are in excess of those anticipated during the launch and flight phases.

Dynamic stability testing consists of an operational test of the ACS, unrestrained, mounted on air-bearings and demonstrates the response of the ACS to the planned program missions. The flight configuration electronic components installed for the test are slaved to receive external stimuli simulating the flight environment during actual mission. A closed loop RF system is used to receive signals from the test package.

The ACS is monitored for stability, maneuverability, and orientation.

Integrated systems operational testing is performed on each complete system with all flight equipment installed. System functional testing exercises all electronic components, valves, switches, etc.; through a typical flight mission.

Weight and balancing each system with its full complement of flight equipment is conducted to optimize the CG location and to minimize disturbing torques during launch.

7.6.2 Qualification Tests. Qualification tests are performed or verified on each component of the complete experiment. Procured components previously qualified by the vendor to AAP requirements are accepted only after the vendor's test data is reviewed by the cognizant engineering design group.

Qualification tests are performed to formalized test procedures and are witnessed by quality control inspection and verified by the customer.

Detailed test procedures covering each component to be qualified are written by the contractor and submitted to the customer for written approval before initiation of the test program. The test procedures include operating requirements, testing tolerances, proof cycles, step-by-step sequence of testing events, schematics of each proposed test setup, block diagrams of proposed instrumentation, and data sheets.

Qualification tests are conducted mostly at the component level. Testing is conducted under various environmental conditions and, as a minimum, include temperature, vibration, acceleration, thermal vacuum, solar radiation, deployment tests, electrical tests, and proof tests.

7.6.2.1 Component Qualification Testing. The objective of these tests is to demonstrate each components capability to perform a required function of the complete system. Each component is functionally operated under various environmental conditions over and above the ground and flight conditions, such as increased temperatures, accelerations, shocks, and extended life tests. Component qualification testing can be divided into three groups:

a. Electrical components.

b. Mechanical components.

c. Structural components.

Qualification testing of electrical components may utilize more than one experiment component as support equipment to perform operational functions during tests. Instrumentation equipment monitoring operating parameters is calibrated prior to the start of testing. During use, calibration surveillance and service continues with the "valid decal" requirement.

Some of the basic electrical components that are subjected to Qualification Tests are as follows:

a. Star tracker
b. Solar sensor
c. Digital computer
d. Optical telescope
e. Logic unit
f. Telemetry receiver & transmitter
g. Laser and electronics
h. TLM antennas
i. ACS control electronics
j. Extension motors
k. Batteries & solar cell panels
l. Polarimeter
m. X-ray imaging orthicon

Qualification of mechanical components is performed to determine the physical characteristics of the components as well as the operating capabilities. The strength of the design and thermal gradient characteristics is considered.

All mechanical/electrical components require holding jigs and mounting fixtures for vibration, acceleration, and shock testing. All the fixtures simulate the normal mounting attachments of the in-flight conditions. Design, fabrication, and evaluation of all fixtures is performed by the test lab. For components requiring deployment tests a suspension system for zero-g balanced-force environment is used. The zero-g environment is simulated by suspension from overhead members by compliant systems attached to the most massive specimen sections.

Some of the basic mechanical components that are subjected to qualification tests are as follows:

a. Lens capsule extension.

b. Lens gimbaling mechanism.

c. Extension carriage mechanism.

d. Jets and inertia wheels

Qualification of structural components is performed on specimens such as truss members and extension mechanisms. Dynamic and static limit loading is performed to verify the structural integrity of each component. Meteoroid impact is considered as to the deterioration effects on each component. Life impulse tests are performed to verify reliability requirements. Failures during these types of tests may warrant design modifications prior to successful completion of qualification testing.

7.6.2.2 Subsystem and System Qualification. After completion of the individual component tests, the qualified components are assembled into their respective subsystem and system configurations. Subsystem and system level testing is then performed to verify satisfactory functional operation of all system equipment.

Some examples of system qualification testing follow.

The test specimen, packaged in the launch configuration, is subjected to translational vibration test conditions simulating the predicted launch environment. This test demonstrates the structural integrity of the packaged assembly.

The full-scale engineering model in the packaged configuration with all appropriate equipment, is activated in the operating deployment mode. This is done under simulated zero-g conditions. At full deployment, the truss members and joints are effectively stress free. Full camera coverage is used during deployment operations and functional instrumentation is monitored.

After the full-scale engineering model has been deployed in the simulated zero-g environment, the distortion of the RF reflective surfaces are measured.

7.6.3 Acceptance Tests. Acceptance testing is performed on the flight article. Acceptance tests are basically functional tests performed to formal procedures approved by the customer and requiring quality control inspection witnessing and sell-off to the customer's in-plant representative.

Acceptance tests environments never exceed the operating design levels encountered during ground and launch conditions. Acceptance tests to be performed on the flight article are selected from the sequence of tests conducted during qualification testing.

7.6.4 Test Facilities. The facilities described in the following paragraphs represent only the major items that are used in direct support of the program. Omitted are secondary support facilities such as laboratory and bench type test equipment which the typical aerospace contractor would normally have available. Facility requirements are predicated on a production of three spacecraft of the x-ray telescope configuration. Manufacturing fixtures and equipment serve a dual purpose in that they are, in many cases, used also during the testing phases of the fabrication and development program.

7.6.4.1 Component Test Facilities. Testing at the component level requires environmental chambers, vibration equipment and a vacuum chamber with a cryogenic shroud and solar radiation capability. Shielded enclosures are also used for conducting electromagnetic interference tests.

7.6.4.2 Subsystem Test Facilities. Subsystem test programs utilize all of the equipment previously identified for component testing and such additional equipment as structural loading facilities and a large vacuum chamber. The latter must be large enough to accommodate a representative segment of the telescope structure that is activated from a packaged configuration to deployment. This operation is conducted while the chamber is providing a simulated deep space environment, complete with cryogenic shroud and solar radiation and the telescope is deployed by means of a zero "g" fixture. In addition, the selected contractor should have at his facility, or have access to, a large swimming pool. This pool is used by potential astronauts in conducting EVA studies on the x-ray telescope structure while experiencing a zero "g" or neutral buoyant conditions. These typical activities include assembly and repair of structural members, and routine adjustments and inspections.

7.6.4.3 System Test Facilities. During the system checkout and evaluation program, whenever practical, manufacturing fixtures serve a double purpose in that they also serve as test fixtures.

Testing programs are conducted in the environmental controlled assembly area to maintain desired tolerance requirements. Component and subsystem level test equipment meets the needs of the systems test with the exception of the final RF test program.

7.7 SUPPORT PLAN. The x-ray telescope support plan summarizes the general requirements for all activities performed on the spacecraft subsequent to sell-off of the flight unit at the contractor's facility. It also defines the necessary support for these activities. Typically, included are:

a. Personnel training.

b. Pre-launch activities.

c. Range documentation.

d. Launch site operations.

e. Orbital operations.

7.7.1 PERSONNEL TRAINING. Special training is required to develop the necessary skills in mission-oriented tasks. In addition to the detailed training program for the astronauts, some training is required for both the launch crew (in countdown procedures and spacecraft checkout) and the orbital support crew (communications and data handling).

The elements of the astronaut training program are shown in Figure 7-4. It is assumed that at least two 3-man crews, all electrical engineers who have received astronaut training, will be given the special training simultaneously.

The first phase would be an intensive refresher course in x-ray astronomy theory. Overlapping this is the indoctrination in use of equipment and hardware associated with the telescope concept. The study of malfunction detection, analysis, and repair techniques may be most effective if protracted over a period of four months, spanning the time in which the various phases of the mission are simulated. The safety procedures portion of the program will concentrate on the EVA aspects, analyzing the conceivable hazards associated with the inspection, adjustment, and repair tasks

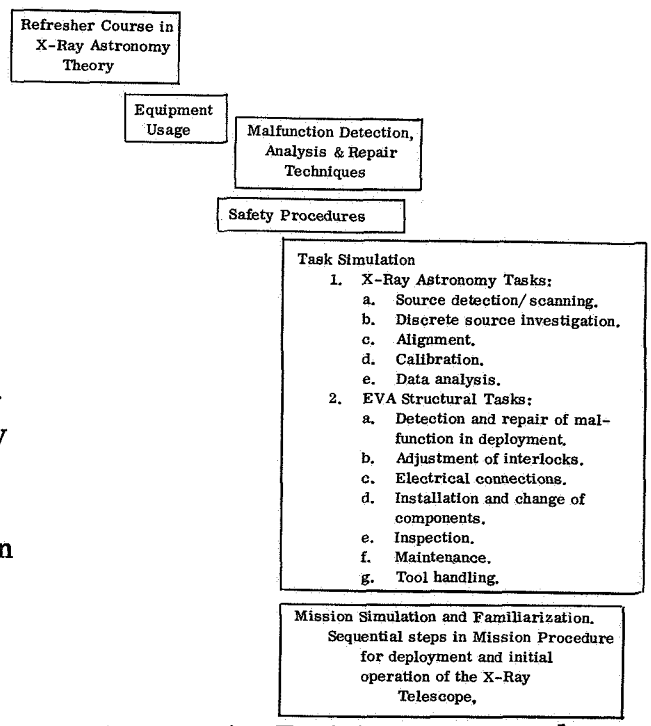


Figure 7-4. Training program plan.

for each of the telescopes, how these might be avoided, and the corrective actions to be taken. Certain emergency conditions will then be simulated in an underwater facility.

Simulation is divided into four major sections: x-ray tasks, EV tasks, inspection and checking, and mission familiarization. It may be desirable to rotate the astronaut-trainees in groups of three (the selected crews for a given mission) through the simulation course for most efficient use of the simulators. Representative tasks are indicated in the accompanying chart for each facility.

The x-ray astronomy training facility will require consoles with associated controls, displays, and recorders. It will be necessary for these to be integrated with the orbital flight simulator (in the latter part of the training program when simulated missions are run).

At the present time, underwater weightless simulation appears to be the best available method to employ for training in EVA tasks involving relatively large structures. Although the effects of hydrodynamic drag cannot be eliminated, they may be minimized by slow movements. A practical limit on the size of an underwater test facility that provides maximum safety to the test subjects, overhead cranes, underwater viewing, accessibility and other desired features will also limit the size of the test structure that can be immersed. A 30 ft. depth would appear to be a reasonable compromise to permit the handling of full-scale mockups of the telescope. Thirty ft. is also the break point in diving where the test subject can come up without a decompression pause.

Inspection and mechanical-electrical checking, as well as mission familiarization, require an orbital flight simulator with instrumentation for tolerance checks. If inspection of the telescope package, deployment and surface prove to be most effectively accomplished by an unmanned maneuvering unit, a simulated, or model RMU will be required with man in the control loop.

A technical orientation and training session is required to familiarize range and orbital support crews with their mission-oriented tasks. This session includes presentations by the experimenter, the contractor and personnel from the orbital range. The principal investigator or his representative describes the experiment, its objectives, and the data gathering requirements. A range representative presents ground-station capabilities, range operating techniques and planned level of range support. The contractor discusses the spacecraft design capabilities, command control requirements, and the planned orbital operations philosophy.

The final phase of the training program will require thorough familiarization with the various steps of the mission in sequence, with the exception of EVA. Required training facilities will include an orbital mission simulator and the x-ray astronomy console.

7.7.2 Prelaunch Activities.

7.7.2.1 Handling and Shipping Operations. The x-ray telescope spacecraft is shipped completely assembled, with experiment equipment installed, but in the stowed condition from the contractor's facility to the Marshall Space Flight Center by government air freight. Special handling is designated. Subsequent to MSFC operations the spacecraft is again air-lifted to the launch site (KSC/AFETR). Other transportation is by specially designed flat-bed trailer assembly.

The spacecraft is transported at all times in a specially designed shipping container. Solar cell arrays, batteries and operational test equipment are shipped separately and handled as delicate instruments. Installation hardware is packed in suitable containers and secured in the appropriate shipping containers.

7.7.2.2 MSFC Operations. The spacecraft and ground support equipment is shipped to MSFC for MSFN network compatibility tests and Saturn IB fit checks.

a. MSFN Compatibility Test. The spacecraft is removed from the shipping container, visually inspected and subjected to a standardized checkout of electrical, command, telemetry and experiment subsystems. Test equipment, procedures and operations are identical to those used at the launch site. MSFC establishes the requirements for the compatibility tests and directs their performance.

b. Saturn IB Fit-Check. The x-ray telescope spacecraft system, LEM adaptor system and the CSM are mated together to determine their mechanical structural compatibility. Static weight and balance measurements and CG trimming are accomplished at this time.

Following all tests, the spacecraft is re-installed in the shipping container and transported to KSC/AFETR for launch operations.

7.7.3 Range Documentation. The basic documentation requirements of the MSFN stations are listed below.

a. Support and Instrumentation Requirements Document. This document specifies requirements for facilities, data processing logistics, telemetry, and instrumentation support. It also furnishes a brief description of the spacecraft and provides detailed information on the characteristics of the telemetry and command systems.

b. Operations Plans. The operations plan is prepared by NASA to specify requirements for injection and early orbital support. The contractor inputs to this document include the launch operations test plan, the spacecraft operations notebook and nominal command schedule.

The launch operations test plan pertains primarily to the launch range operations and requirements. It specified procedural and equipment interfaces and provides an operations schedule from the time of shipment of the launch site to orbital injection. However, it also furnishes information pertinent to orbital operations such as vehicle frequency utilization, telemetry formats, predicted launch mark events and processing requirements for orbital data.

The spacecraft operations notebook includes subsystem descriptions, command control operating instructions, an orbital operations philosophy, and housekeeping data analysis. The command control instructions present a detailed analysis of spacecraft response to each of the possible commands under normal and abnormal conditions. It also defines and explains all cautions to be observed during operation. The orbital operations philosophy is a detailed discussion of the theory and reasoning used as a basis for the command schedule. The purpose is to provide the MSFN spacecraft controllers with guidelines so they may utilize the maximum capability of the spacecraft without endangering its operational life. This allows flexibility and rapid response to experimenter requests. Definition and analysis of all housekeeping data and their predicted limits are included in the operations notebook. The housekeeping data critical to spacecraft and experiment health are designated as red-line or go/no-go functions and their permissible limits defined.

The spacecraft nominal command schedule provides predicted look angles, acquisition times, and sequence of commands to be transmitted for each ground station.

c. Spacecraft Telemetry Test Tape. A voice annotated magnetic tape of the telemetry composite video signal recorded during simulated normal and abnormal orbital acquisitions will be provided to the range for operational training and data processing tests. A script of the tape contents noting time and duration of commands will also be furnished.

d. Range Compatibility. Spacecraft-to-ground station compatibility may be verified by arrangements with the MSFN Satellite Operations Center for the applicable NASA stations to acquire and operate the spacecraft in orbit. The telemetry test tape could also be used for compatibility checks.

e. Range Orientation Session. A brief technical orientation and training session is required. This session includes presentations by the experimenter, the contractor and personnel from the orbital range. The principal investigator or his representative describes the experiment, its objectives, and the data gathering requirements. A range representative presents ground-station capabilities, range operating techniques and planned level of range support. The contractor discusses the spacecraft design capabilities, command control requirements, and the planned orbital operations philosophy.

f. Test Evaluation Report. A test flight report will be prepared for the x-ray telescope spacecraft system. This report includes evaluation of the propulsion module through the period of orbital injection. This portion of the report consists of a quick-look diagnostic evaluation of MSFN- and KSC supplied data. Evaluation of the spacecraft performance throughout the mission will be based on MSFN-supplied reduced data. It includes sufficient detail to evaluate overall performance against predicted criteria for each of the subsystems.

7.7.4 Launch Site Operations. At the launch site the spacecraft undergoes inspection, checkout, Saturn IB assembly and launch operations.

7.7.4.1 Sequence of Operations. A summary of the sequence of events is given in Figure 7-5. Shipping and receiving of all x-ray telescope equipment is accomplished per published shipping instructions listing all deliverable items with inspection and storage instructions of each. These functions are accomplished by the organization and facilities established to support previous AAP operations.

A clean, environmentally-controlled area is provided for storage, assembly, and test of the satellite and experiment equipment.

Engineering confidence tests are performed on all spacecraft subsystems to verify proper operation after shipment and prior to experiment interface checks. These tests are accomplished by cycling the spacecraft through its operating modes while recording and verifying data per the telescope spacecraft functional checklist. If an anomaly is noted, the particular subsystem is tested per the appropriate section of the telescope spacecraft subsystems acceptance test.

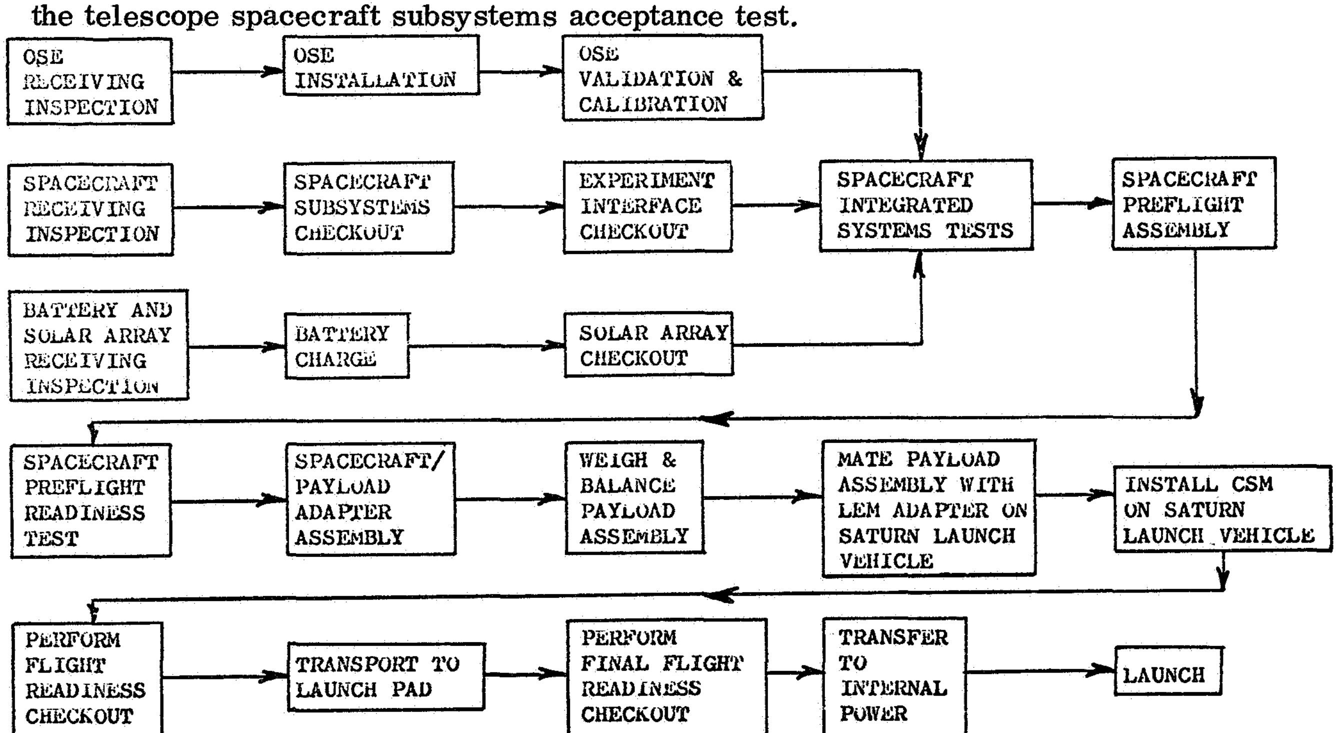


Figure 7-5. Launch Site Operations Flow Chart.

In addition to these confidence checks, two formal engineering procedures are performed: (1) solar array electrical test and, (2) electrical storage battery charge procedure.

Engineering confidence tests verify the mechanical and electrical interfaces between the spacecraft support subsystems and the experiment payload. Interface tests are performed with both the flight and the backup subsystems.

Electrical interfaces are checked in all operating modes. These tests verify proper response of the spacecraft and experiment payload to external commands and to internal logic signals. Data from these tests is reviewed for quality and compared to data from the telescope spacecraft acceptance test.

Upon completion of interface checks the spacecraft is assembled to prepare for mating with its payload adapter assembly. The assembly procedure includes:

a. Thorough cleaning of the spacecraft/adapter interface.

b. Installing the flight battery.

c. Verifying proper mating of each spacecraft electrical connector.

d. Installing equipment status panels.

e. Installing solar panels.

A formal engineering systems test is performed after preflight assembly and prior to mating with the payload adapter assembly. This test duplicates, as closely as the fully assembled configuration will permit, the telescope spacecraft systems acceptance test performed at the factory. The data is analyzed to verify proper operation of all subsystems and compared with like data recorded during launch site confidence tests and the factory systems acceptance test.

After completing the preflight readiness test the weight and balance of the assembled payload is checked to verify launch configuration. The payload is then installed in the LEM adapter and mounted atop the Saturn launch vehicle.

The flight readiness checkout procedure includes four sections: (1) Spacecraft system checks, (2) Experiment system checks, (3) Installation, checkout, and interface verification, and (4) Final spacecraft flight preparation. The subsystems checks again verify proper response of the spacecraft and experiment payload command control logic to external commands and internal logic signals as well as providing data for qualitative analysis of the telemetry and electrical power systems.

At the launch pad final flight readiness checkouts are performed in conjunction with the launch vehicle countdown. Shortly before launch the telescope spacecraft is transferred to internal power to sustain standby loads until final orbit is achieved.

7.7.4.2 Launch Site Procedures. All operations at the launch site are performed under the surveillance of MSFC quality control personnel to ensure maintenance of system configuration and performance integrity. An engineering log of all launch site activities is maintained by the systems engineer. Operations are divided into two categories: (1) formal engineering tests, and (2) engineering confidence tests. The formal engineering tests are conducted by cognizant design engineers in accordance with published procedures specifying step-by-step sequential operations and witnessed by a quality control inspector. The engineering confidence tests are also conducted by cognizant design engineers in accordance with functional checklists specifying required data and monitored by a quality control inspector. Table 7-2 lists typical launch site procedures applicable to the telescope spacecraft and the purpose of each.

7.7.4.3 Launch Site Facilities. The Saturn IB uses launch Complex 34 and 37 of the Kennedy Space Center. Complex 34 has one pad while Complex 37 has two (Pad A and B). At present only Complex 34 and Complex 37B will be used. Figure 7-6 shows an illustration of the layout of Complex 37.

Complex 37 launch pad is 47 ft. square and 35 ft. tall, with a 33 ft. diameter opening in the center to channel the engine exhaust gases onto the deflector below. The pad is also used as a base upon which to mount support equipment, such as the tower containing electrical, pneumatic and propellant loading umbilicals.

The umbilical tower, which stands alongside the launch pad, is 268 ft. high and designed so that as much as 52 ft. of structure could be added should additional height be required. It supports umbilical swing arms which carry the electrical, environment-control, pneumatic, and propellant vehicle servicing lines during pre-launch preparation and launch. An automatic ground control station (AGSC) is located partly in the base of the umbilical tower and partly below the pad surface under the tower base.

Table 7-2. Spacecraft launch site procedures.

TITLE	PURPOSE
Spacecraft Shipping Instructions	Provide listing of all deliverable items with instructions for shipping and storage.
Test Equipment Validation	Provide instructions for system qualification of special test equipment. (Standard test equipment will be calibrated by local calibration or precision equipment laboratory.)
Spacecraft Subsystems Check list	Engineering confidence test for support systems.
Solar Array Electrical Test	Test of open circuit voltage and short circuit current of each series string with a fixed illumination source. Also checks forward and reverse impedance of isolation diodes.
Battery Charge	Provides instructions for formation charge, standard full charge or partial charge of the spacecraft electrical storage battery.
Spacecraft Final Assembly	Prepares spacecraft for mating with final stage.
Preflight Readiness	Final spacecraft systems test.
Flight Readiness	Performs functional subsystem tests on launch pad and provides instructions for final flight preparations.

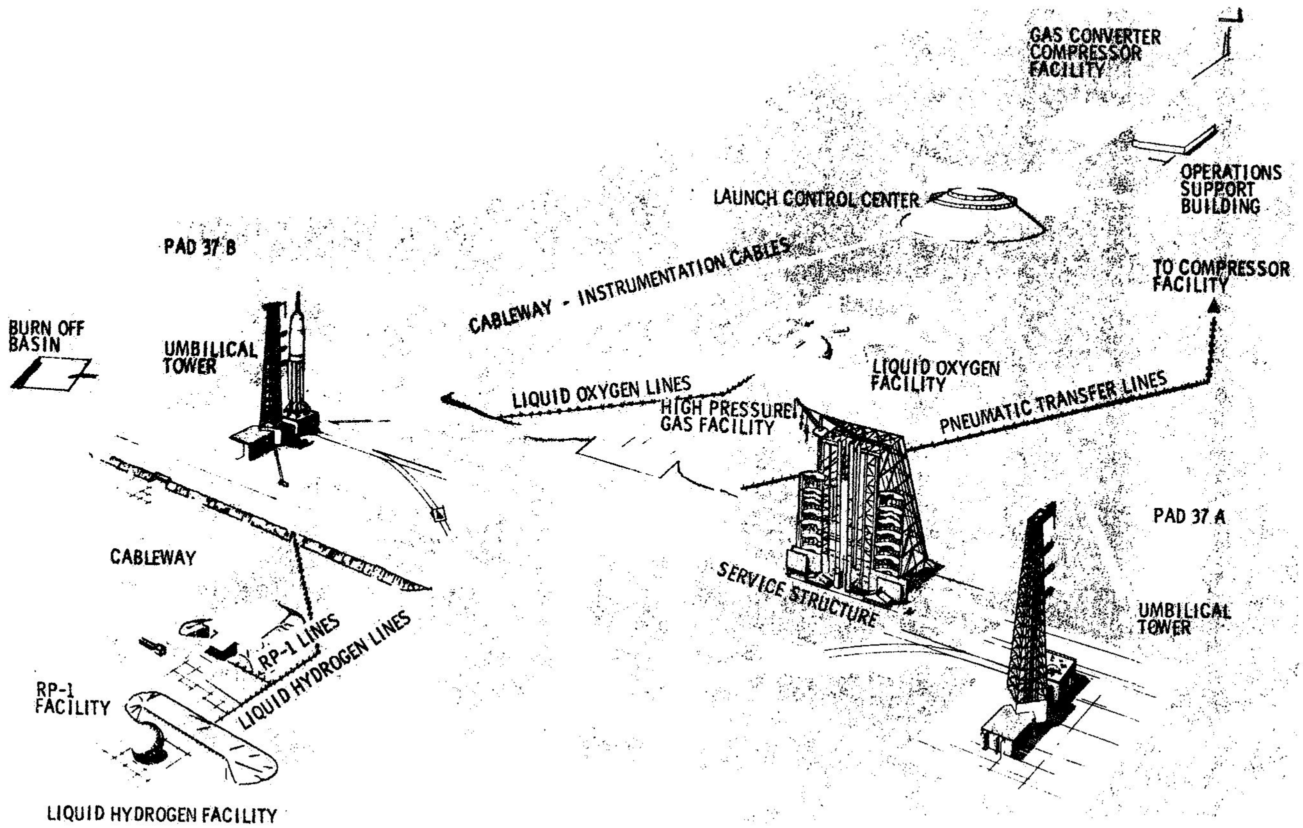


Figure 7-6. Saturn IB Launch Complex 37

It houses the equipment and serves as a terminal and distribution point for measuring and checkout equipment, electrical power control circuitry for propellant and high-pressure gas system, pneumatic control equipment, and piping systems connecting the vehicle and its ground support equipment. A generator room there provides AC and DC power. A cable tunnel under the pad connects the AGCS room to the cableway extending to the launch control center.

The launch service structure is 300 ft. high. It is designed so that 30 ft. of structure can be added if needed. A derrick mounted atop of the launch service structure lifts all stages and spacecraft during the erection of the launch vehicle. The service structure is equipped with a clamshell "silo" to enclose the vehicle on the pad for hurricane protection. Adjustable work platforms give access to all parts of the vehicle during launch preparation.

Complex 37 is equipped with a central high-pressure gas system to supply gaseous nitrogen and helium. Liquid nitrogen is stored in a 35,000 gallon insulated tank. A gaseous-nitrogen storage tank is supplied by vaporizers in the liquid-nitrogen storage tank. The gaseous helium storage supply is charged by compressors from supplies of helium brought in at lower pressures.

The blockhouse, housing the Launch Control Center (LCC), is a circular dome of reinforced concrete. From it can be controlled all prelaunch and launch activities. It can checkout the complete vehicle automatically with an RCA 110A computer system. The Launch Control Center is approximately 1150 ft. from the launch pad and has visual contact via periscopes and by television.

In addition to the above facilities, the Kennedy Space Center has pulse radar systems, infrared systems, metric optics (ballistic cameras, cinetheodolite, fixed cameras), documentary photography, missile impact location system (underwater sound detection and location) and telemetry (FM/FM, PAM/FM/FM, PDM/FM, Inflight Television, PCM/FM) for tracking and data collection.

7.7.5 Mission Operations. Flight mission operations are controlled by NASA/MSFC and MSC.

7.7.5.1 Boost and Orbit Injection. Definition of the flight sequence from liftoff through injection of the CSM/x-ray telescope spacecraft payload into orbit is the responsibility of NASA/MSFC and will result from trajectory shaping and optimization for the specific payload involved.

7.7.5.2 Orbit Operations. The astronaut participation in the deployment, checkout, and resupply phases of operations is covered in detail in Section 6.2.

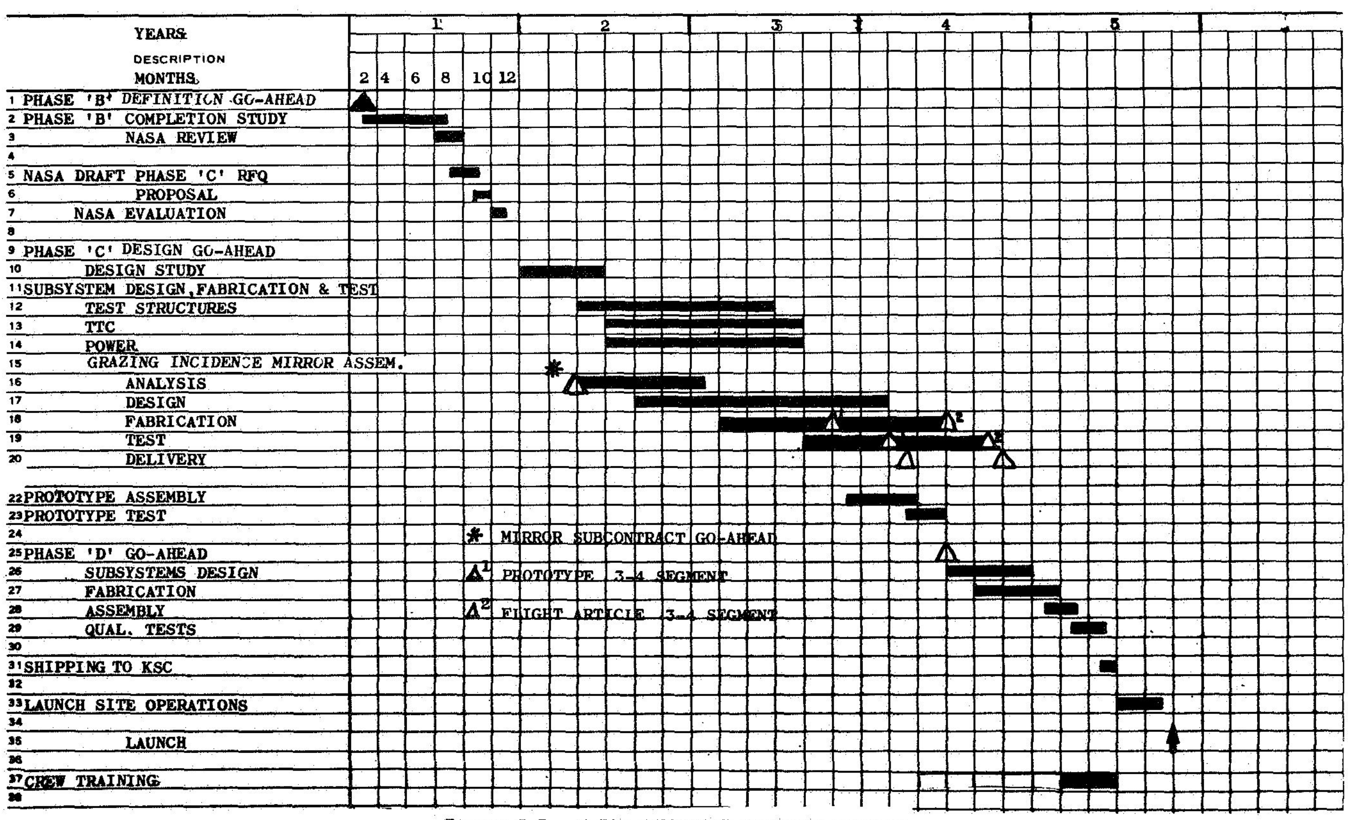


Figure 7-7. X-RAY TELESCOPE SCHEDULE SUMMARY

7.8 SCHEDULE. The program schedule for the x-ray telescope spacecraft is shown in Figure 7-7. The schedule indicates that a 3 to 4 segment nominal 30 in. aperture telescope can be launched in slightly less than five years after Phase "B" definition go-ahead. This program would be extended approximately 1.5 years if an 8 to 10 segment nominal 30 in. mirror assembly is selected.

7.9 COST ANALYSIS.

7.9.1 Introduction and Ground Rules.

The funding requirements for the focusing x-ray telescope experiment are presented herein. Included are non-recurring cost, recurring cost, vehicle support, and facility costs.

The cost estimating task for this telescope system was carried out in accordance with guidelines provided by MSFC as outlined below:

a. Cost estimates are to be developed for the x-ray telescope in accordance and compatible with, the general format outlined in "Cost Estimation for Future Programs" (ASO 12 May 1967).

b. The level of detail of the costs will be dictated by the level of design definition attained and the time allocated to this portion of the task.

c. Cost estimates are to be made for the space structure and support hardware, and x-ray telescope space vehicle system integration only. Costs will not be included for the launch vehicle, Apollo CSM or other spacecraft, mission support, mission operations, AAP payload integration, or subsequent flights for rendezvous and/or experiment refurbishment.

d. The program to be costed will include one flight vehicle and one complete prototype, with no backup flight article.

e. In addition to total system costs, funding requirements will be presented in accordance with NASA Phased Project Planning Guidelines. These fiscal year cost estimates will include Phase B (Definition), Phase C (Design), and Phase D (Development).

f. In addition to total dollars, the system cost estimate is to include a breakout of labor in man years and material costs.

Further ground rules which were used in this analysis include the following:

a. 1967 dollars were used throughout both for labor and material.

b. Present manufacturing and test facilities are assumed adequate and available for the conduct of this program with the exception of these unique or new facilities included in the cost estimate.

c. Fully developed and flight qualified hardware components were utilized wherever possible.

d. The program is assumed to be a normally paced (not crash) development program. Labor costs are based on a single shift operation.

e. NASA in-house costs are excluded.

7.9.2 Cost Estimating Procedure. In general, costs were estimated for 1) nonrecurring or development phase of the effort; 2) recurring cost or flight hardware unit cost; 3) vehicle support equipment (GSE and STE) and 4) facilities.

Nonrecurring Cost. The nonrecurring cost includes all research, development, design, analysis and test including all development hardware necessary to fabricate and fly the operational experiment.

The nonrecurring cost as defined herein includes only those costs up to the point where fabrication of the flight article is initiated. The nonrecurring cost is added to the recurring (unit) cost to obtain total program cost. This procedure was used to give an incremental unit cost for the flight unit in a one vehicle program.

The system and subsystem definition and the development plans were reviewed and analyzed to determine general task requirements at the major subsystem level. Manpower requirements were then estimated for these tasks and for overall system integration tasks.

The development plan and the test plan provided the basis for defining the development and test units, other test hardware, and associated tooling. Costs were then developed for both material and labor.

Composite labor rates were applied to manpower requirements and summed with the materials cost and overhead rates for the total nonrecurring cost estimate.

Recurring Cost. The recurring cost includes the incremental unit cost of the experiment, test operations and checkout, and spares.

The design definition at the system and subsystem level was reviewed and analyzed and a major component list prepared. Costs were then estimated for purchased items. In the case of manufactured items, material and labor costs for fabrication and assembly including the appropriate tooling, quality control, etc. were estimated. System/subsystem test and checkout labor was estimated based on the complexity of the experiment and appropriate costs developed from available cost analogs. Appropriate factory overhead, material burden, and G&A overhead factors were then added to provide total incremental unit cost.

Spares are estimated on a highly aggregated basis based on a percentage factor of total estimated unit cost.

Vehicle Support. Cost estimates were made both for GSE and Special Test Equipment (STE) at a relatively high level of aggregation because of the lack of design definition of the equipment in this category.

Facilities. The design definition and the manufacturing and test plan for the x-ray telescope were examined to determine the requirements for new or unique and unusual facilities in each of the areas of manufacturing, test, and operations, and served as the basis of the cost estimates for these required facilities.

Cost Uncertainties. The confidence limits of the cost estimates presented in this report are believed to be compatible with the level of definition of the subsystems, components and with the development plan available at the time of the cost estimating effort. These estimates are to be regarded as area estimates based on varying degrees of definition. In some areas, such as some of the experiment instrumentation, only cost allowances were made. These areas are discussed and identified below. Cost estimates in more detail as well as greater confidence will be developed at the completion of the Phase B (Definition) studies of the Phase Project Planning cycle.

The cost estimates presented are believed to be representative for this program in view of the present state of definition. Upon more detailed analysis some of the cost factors may prove sensitive to further definition and to design changes. Within this context the cost may be expected to be in the tolerance range of -10% to + 30% of this estimate.

Table 7-3. Cost Summary.

	COST (MILLIONS OF DOLLARS)	
NONRECURRING COST		30.310
DESIGN & DEVELOPMENT	28.720	
GSE/STE	.995	
FACILITIES	.595	
RECURRING COST (UNIT COST)		9.950
TOTAL PROGRAM COST		40.260

7.9.3 X-ray Telescope Cost. The results of the cost analysis of the focusing x-ray telescope are summarized in Table 7-3 with the detailed cost breakouts presented in Tables 7-4 through 7-6 and discussed below. Funding requirements by fiscal year for Phases B, C and D are presented in Table 7-7.

The nonrecurring cost of this astronomical facility is presently estimated to be $28.7M for the total program design and development. The recurring cost is $9.95M for the fabrication of a single flight article. The total program is estimated to cost $40.3M including the vehicle support and facilities. The principal uncertainties and assumptions are discussed below.

Hardware costs were developed on the basis of purchased components and materials and on manufactured items. Labor estimates were made for the overall system tasks and the design, analysis, test, integration, and fabrication of the "make" items. The total cost estimate also includes design, development, test, and qualification of certain components (in addition to the structure), such as the flight control logic unit (including the inertial unit) and the mirror alignment system. The remainder of the components are considered purchased items although some will have an associated nonrecurring cost.

The costs presented herein were based on one complete flight article, one complete prototype (with certain exceptions) plus single item components for use in subsystem and "bits-and-pieces" testing. Some of the prototype equipment will not be of the flight configuration.

The most notable item in this category is the primary x-ray mirror assembly. Because of the high cost as well as the long fabrication and grinding and polishing time required for these mirrors, it is expected that only one set will be procured to the full optical figure required for the flight article. Because of the high solar cell cost, dummy panels will be utilized on the prototype.

The components selected for costing purposes are current off-the-shelf, space qualified units wherever possible. The components were selected for performance that either met or exceeded the capabilities requirements. In cases where no qualified components were available, costs were based on equipment with excess capability. This was done to avoid incurring a large development and/or qualification costs at the component level which would completely override any recurring hardware savings which might be associated with a lower performance requirement. In the case of the star trackers, costs were based on trackers with a 15 arc sec. capability.

Tracking, telemetry, and command system components and the data management/data storage system components were estimated on the basis of similar systems in the Apollo CSM Telecommunication System even though certain elements of the Apollo equipment may have excess capability. In some cases a lower cost estimate was used to allow for the fact that only portions of the electronic system "black-boxes" were required.

Table 7-4. Nonrecurring Cost

	Cost (millions of dollars)					
	Engineering Design, Development and Analysis	Tooling	Development And Test Hardware		Test	Total
			Labor	Material		
Structure	0.230 (7.2)	0.040 (1.5)	0.185 (8.2)	0.140	0.240 (9.0)	0.835
Electrical Power System	0.200 (6.2)	0.015 (0.5)	0.030 (1.2)	0.940	0.190 (7.0)	1.375
Stability and Control	0.900 (28.8)	0.025 (1.0)	0.110 (5.0)	3.135	0.465 (18.2)	4.635
Telecommunications/Data	0.665 (21.2)	0.010 (0.5)	0.100 (4.5)	2.510	0.405 (15.1)	3.690
Experiment Instrumentation	0.665 (21.2)	0.035 (1.2)	0.115 (4.0)	10.255	0.470 (18.2)	11.540
Vehicle System (Assembly and Integration)	0.250 (8.0)	0.040 (1.5)	0.130 (5.8)	0.060	1.145 (41.5)	1.625
Tech Data/Manuals						0.595 (8.8)
System Engineering/ Management						3.720 (113.2)
Training						0.630 (22.4)
Travel						0.075
TOTAL	2.910	0.165	0.670	17.040	2.975	28.720

(Labor Requirements, in man years, are shown in parenthesis)

Table 7-5. Recurring Cost

	Cost (millions of dollars)				
	Fabrication And Assembly		Test And Checkout	Spares	Total
	Labor	Material			
Structure	0.167 (6.5)	0.080			0.247
Electrical Power System	0.055 (2.0)	0.680			0.735
Stability and Control	0.115 (4.1)	0.910			1.025
Telecommunication/Data	0.100 (3.8)	1.350			1.450
Experiment Instrumentation	0.108 (3.0)	3.300			3.408
Vehicle System (Assembly and Integration)	0.040 (1.8)	--	0.185	0.560	0.785
Sustaining Engineering					0.550 (18.0)
System Management/System Integration/Training					1.215 (39.0)
Launch Support					0.485 (15.0)
Travel					0.050
TOTAL					9.950

(Labor Requirements, in man years, are shown in parenthesis)

Table 7-6. Vehicle Support and Facilities

	Cost (millions of dollars)
Vehicle Support	
GSE	
Launch Site Support	$0.365
In-Plant Support	0.490
Total	0.855
STE	0.290
Facilities	
Manufacturing	0.500
Test	0.095
Operational	--
Total	$0.595

Table 7-7. Funding Requirements - Phased Project Planning

	FY0	FY1	FY2	FY3	FY4	Total
PHASE B COMPLETION	0.200					0.2
Phase C & D		5.610	11.415	13.830	9.405	40.260

Certain of the component packages are not expected to be "off-the-shelf" in the time period of interest. The package which must be developed includes the flight control programmer/logic unit for the attitude control system, thermal control systems, the mirror alignment system, and the x-ray image sensing instrumentation.

The greatest cost uncertainty lies in the area of the imaging instrumentation (including spectroscopy and polarimetry). This cost analysis includes only gross cost allowances for experiment instrumentation (both hardware unit cost and development). Lack of sufficiently detailed definition at this time precluded costing of the individual equipment packages. The cost of the total experiment systems including the mirrors and the image instrumentation will undoubtedly be very high. The cost of the total program is therefore expected to be very sensitive to these experiment systems costs. The cost estimate for the mirrors is based on discussions with the Perkin Elmer Corporation which is currently fabricating a smaller, similar type of mirror for the ATM program. This estimate is based on a program that has a high probability of utilizing beryllium although the choice of this material over fused silica or "zero expansion" glass will require further system analysis study. The program is not expected, however, to be overly sensitive to this material choice. The total cost of a nominal program to design develop, fabricate and test a baseline mirror configuration consisting of a 30 in. diameter three mirror systems is estimated to be \$6. 80M. This includes 1) a Phase B definition cost of \$200K, 2) a nonrecurring design, analysis, and prototype test article (not polished to final optical figure) cost of \$3. 07M, 3) a facility cost of \$500K, 4) a test program cost of \$750K, and 5) a flight article cost of \$2. 33M.

A similar program for a 30 in. nominal 8-10 segment mirror assembly would cost an additional \$3. 0M.

Equally uncertain are the costs involved in the image experiments themselves. These experiments include three types of spectrometers, an image intensifier system, and a polarimeter. It is believed that the feasibility of these concepts have been demonstrated; however, not at the size (and in the package) required for this system. As a result, the costs of these experiments are highly speculative.

The structure appears to be relatively straightforward from the cost viewpoint. The structural concept estimated herein is based on titanium tubing in the primary truss structure. One alternative being considered is the use of beryllium tubes which would cause a substantial increase in the material and fabrication cost of the structural portions of this experiment.

There appears to be no unusual facility requirements for the fabrication and test of this x-ray telescope. The only uncertainty appears to be possible requirement for new facilities for mirror fabrication.

SECTION 8

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APPENDIX I

NASA FORM 1346

Contained herein is the completed NASA form 1346 which summarizes the proposed x-ray telescope. Although the actual scientific experiments which will be flown are not yet determined, a "Hypothetical Observation Program" was conceived and utilized as a design guide. Presentation of this model program is included in this report for reference only.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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EXPERIMENT PROPOSAL

FOR

MANNED SPACE FLIGHT

TITLE X-RAY TELESCOPE

(Confine to total of 30 letters, numerals, spaces, punctuation marks, etc.)

____
EXPERIMENT NUMBER

PRINCIPAL INVESTIGATOR ______________________________

(Signature) *(Date)*

PRINCIPAL ADMINISTRATOR ______________________________

(Signature) *(Date)*

NASA FORM 1346 JAN 67 REPLACES NASA FORMS 1138, 1138A, B, AND C, WHICH ARE OBSOLETE.

THIS DOCUMENT PROVIDES THE FORMAT TO BE FOLLOWED BY THE INVESTIGATOR OR PROPOSING INSTITUTION, WITH SUPPORT AS REQUIRED FROM THE NASA SPONSORING PROGRAM OFFICE, WHEN SUBMITTING PROPOSED EXPERIMENTS FOR MANNED SPACE FLIGHT TO NASA FOR REVIEW AND ACCEPTANCE. INFORMATION REQUESTED SHOULD BE COMPLETED AS ACCURATELY AND WITH AS MUCH DETAIL AS POSSIBLE, SINCE THIS DOCUMENT WILL PROVIDE THE PRIMARY DATA FOR AN EVALUATION OF EXPERIMENT MERIT AND DETERMINATION OF EXPERIMENT COMPATIBILITY TO A MANNED SPACE MISSION. ALSO, THE TECHNICAL ENGINEERING, AND OPERATIONAL INFORMATION (SECTIONS II, III, AND IV) CONTAINED IN THIS PROPOSAL, WHEN UPDATED, WILL CONSTITUTE THE EXPERIMENT DESCRIPTIVE INFORMATION PORTION OF THE EXPERIMENT IMPLEMENTATION PLAN, TO BE PREPARED BY A NASA CENTER BEFORE FINAL APPROVAL OF THE EXPERIMENT CAN BE AUTHORIZED.

FOR MONITORING PURPOSES, PLEASE NUMBER THE PAGES OF YOUR PROPOSAL PROGRESSIVELY FROM THE FIRST TO THE LAST AS FOLLOWS: PAGE 1 OF N PAGES, PAGE 2 OF N PAGES . . ., PAGE N OF N PAGES.

SECTION I - ADMINISTRATIVE /BIOGRAPHICAL

1. APPLICANT INSTITUTION

Name of Applicant Institution	Type of Institution ☐ Government ☐ Non-Profit ☐ University ☐ Industrial ☐ Other

Address	Telephone

Name of Principal Administrator Responsible for Experiment	Title

2. PRINCIPAL INVESTIGATOR

Name of Principal Investigator	Title

Mailing Address	Telephone

Biographical Sketch:
Brief summary of education, experience and professional qualifications.

3. OTHER INVESTIGATORS

NAMES	MAILING ADDRESSES	TITLES OR POSITIONS

NASA FORM 1346 Jan 67

4. RESEARCH SUPPORT

List all other funded and proposed research support of the principal investigator. Include support for this project received from own organization. Amounts shown should reflect total funds awarded over the entire grant periods indicated in the final column.

a. NATIONAL AERONAUTICS AND SPACE ADMINISTRATION SUPPORT

GRANT/CONTRACT NUMBER	TITLE OF PROJECT	APPROXIMATE PERCENT TIME/ EFFORT ON PROJECT	TOTAL AMOUNT ($)	TOTAL PERIOD OF SUPPORT WITH DATES

b. ALL OTHER RESEARCH SUPPORT

SOURCE AND GRANT/ CONTRACT NUMBER (If designated)	TITLE OF PROJECT	APPROXIMATE PERCENT TIME/ EFFORT ON PROJECT	TOTAL AMOUNT ($)	TOTAL PERIOD OF SUPPORT WITH DATES

5. PRINCIPAL INVESTIGATOR'S ROLE IN RELATION TO THIS EXPERIMENT
(Include percent of time to be spent on this project)

6. RESPONSIBILITIES OF OTHER KEY PERSONNEL
(Include percent of time to be spent on this project)

SECTION II - TECHNICAL INFORMATION

1. OBJECTIVES

The flight objectives of the X-Ray Telescope are to:

a. Evaluate the role of man in the deployment, assembly, alignment, maintenance refurbishment and repair of large structures in low earth orbit. The effectiveness of the astronaut in performing his scheduled tasks will be recorded and evaluated by photography, biomedical sensors, task time records and by flight crew debriefing. The data generated during this mission will provide an improved astronaut capability baseline from which more advanced missions can be planned. The ease or difficulty of the various EVA functions of transporting, handling or replacing components can be reiterated in the hardware design cycle resulting in further improvements or extensions of the state-of-the-art of future spacecraft.

b. Advance the state-of-the-art in space structures by evaluation of the structure performance. Conventional instrumentation techniques employing strain gages, thermocouples, and photography will be used. The x-ray telescope subsystems proposed will also contribute directly to the structure evaluation. Specifically the lens alignment system gimbal angle errors can be recorded to develop a time-phased history of the total truss (optical axis) distortions. The short term orbital effects of sunlight vs dark distortions as well as the long term effects of the environmental degradation of thermal coatings will be resolved directly by the lens gimbal angle. Other data of interest is the degradation of the continuously operating mechanical system components, particularly bearings. The image capsule turret and lens gimbal drive motor power monitored intermittently will record changes in bearing friction.

c. Provide a useful astronomical facility in the soft x-ray spectrum capable of satisfying a majority of the x-ray scientific community measurement requirements. Although the actual scientific experiments which are proposed to fly on the telescope are yet to be determined, a "hypothetical scientific observation program" was conceived and utilized as a design goal throughout the conceptual design phase. Maximum flexibility has been built into the hypothetical observation program (and therefore into the design of the telescope) and it is anticipated that the basic configuration is capable of performing most if not all, of the focusing x-ray observations in the 2 to 300 Å wavelength spectrum. A description of the observational program is as follows:

(1) Lock on to any celestial coordinates within ± 30 arc min. (approximately one orbit, 90 min.) before the attitude control reaction wheels must be desaturated.

(2) Perform imaging with resolution of 2 to 5 arc sec. on source of interest using image intensifier, image motion compensator and orthicon. Store image data electronically, (image motion compensator reduces drift rates to less than 1 arc sec./sec.); two imaging packages available.

(3) Perform coarse spectral measurements ($\lambda/d\lambda=100$) with slitless spectrometer grating and imaging package.

(4) Perform precise spectral measurements ($\lambda/d\lambda=1000$) by rotating instrument turret to focusing crystal spectrometer - record and store electronically.

(5) Perform polarization measurements, record, and store electronically. Two polarimeter packages available.

(6) Perform measurements with the undefined equipment in the two spare stations on the turret.

(7) At the completion of measurements on the first source, desaturate inertia wheels and repoint to second source.

(8) Transmit observational data to ground station on command on passing suitable ground station.

The use of electronic data recording was selected for the baseline design primarily for the following reasons:

(a) Electronic data handling techniques permit a larger amount of information to be collected than could be collected using film because of the large volume required for long term film storage.

(b) The observation data is available to the astronomer conducting the observations within 1.5 hr. after the measurement is made, depending on ground station location.

(c) The almost immediate evaluation of the measurement data permits detection of equipment failures or problems which may be corrected by changing to redundant packages by ground control or at least providing proper spares for the next manned rendezvous with the telescope. If film (without on-board developing and scanning) is used many measurements may be wasted if the experiment packages develop defects which are otherwise undetectable from the ground.

The specific scientific objectives of this facility in solar and stellar observational astronomy are to (Reference 1):

(a) Search for weak discrete source 10^{-3} to 10^{-6} times the strength of those observed so far.

(b) Determine precisely (within arc sec.) the location of the discrete sources in order to correlate identification of the x-ray source with optical or radio objects.

(c) Study the structure of the sources with a resolution of 1-5 arc sec..

(d) Study the spectral distribution of the radiation from the various sources.

(e) Search for polarization of the radiation from the sources.

(f) Study the diffuse radiation with the aim of establishing its origin and investigating the properties of the media in which it arises and through which it passes.

(g) Search for time variations in the intensity of the discrete sources.

(h) Study of solar activity, flare and other transient events, active regions of the chromosphere and corona, and the quiet corona.

2. SIGNIFICANCE

Man's role in the initial deployment and checkout, malfunction, repair and scheduled refurbishment of the x-ray telescope will expand and advance man's capabilities in support of future large space structures.

The deployment and operation of the telescope will contribute technological advancements to structures and mechanisms which will be used in future space systems; e.g., extendable trusses, accurate thermal control or compensation methods for large lightweight structures requiring precise alignment independent of environmental influences, mechanisms and motors for long term space operation.

In addition to demonstrating the feasibility of a large space structure and man's ability to provide the necessary support in the facility, the telescope will contribute significantly to x-ray astronomy science by making measurements that can only be made from space, providing greater sensitivity than presently available, and greater resolution for mapping x-ray sources and determining their positions.

3. DISCIPLINARY RELATIONSHIP

a. History of Related Work

The Gemini flight program has provided the primary basis for the assumed x-ray telescope EVA tasks. The Gemini astronauts proved man's ability to maneuver outside the parent spacecraft and perform basic functions which were representative of those required on more complex missions. The x-ray telescope EVA flight objective requires an extrapolation of current capability. In addition to Gemini flight experience, the proposed x-ray telescope tasks are based to a degree, on the USAF C-131 zero g experiments and neutral buoyancy simulation. The EVA requirements for the mission have been developed under the guidance of the Manned Spacecraft Center Crew Systems personnel and in accordance with the MSC document the "Baseline Astronaut Capabilities in the 1968-72 Time Period".

The second flight objective extending the state-of-the-art of space structures by the performance of a large orbital structure has no direct precedent since no structures of similar size and requirements have been flown. The design of the x-ray telescope structure utilizes state-of-the-art technology and can be fabricated and deployed in space with a high degree of confidence. The primary new technology area concerns the ability to either eliminate, control or compensate for the environmental thermal distortions which cause excessive optical axis misalignment. Nine and 13 in. focusing x-ray mirror systems are currently being developed for the Apollo Telescope Mount. While the nominal 30 in. mirror proposed is about three times as large as the current systems, it is still small enough to be developed with current material and fabrication technology. The desired resolution of 2 to 5 arc sec. is state-of-the-art for the smaller mirrors and therefore is theoretically easier to achieve with the 30 in. mirror.

The history of work relating to the scientific flight objective of telescope is extremely short; in five years the x-ray observations have progressed from the first evidence of a localized flux to the detection of about two dozen discreet sources (0.5 - 15 Å). The observations to date have been primarily time limited by the rocket-borne instruments and resolution limited by the nature of the instruments, generally uncollimated detectors with fields of view in degrees. The sources observed are divided fairly evenly into those which are relatively isolated and fully resolved or densely clustered. The latter are expected to be resolved by improved resolution.

There appears to be two classes of sources: those accompanied by radio or optical emission, and those which remain as yet unidentified with any radio or optical object. The following is a list of observed x-ray sources and their respective locations.

Source	Intensity (counts/ cm^2/sec.)	Right Ascension* h	Right Ascension* Min.	Declination* Deg.	Declination* Min.
Cyg XR-1	0.91	19	57	34	-
Cyg XR-2	0.8	21	45	39	-
Cyg XR-3 (Cyg A)	0.40	19	58	40	37
Cyg Loop (2 sources)	0.4	20	48	30	48
Cyg XR-4	0.6	21	18	44	-
Tau XR-1 (Crab nebula)	2.7	5	32	22	-
Sco XR-1	18.7	16	17	-15	31
Sco XR-2	1.4	17	8	-36	24
Sco XR-3	1.1	17	24	-44	18
Oph XR-1	1.3	17	32	-20	42
Sgr XR-1	1.6	17	54	-29	12
Sgr XR-2	1.5	18	10	-17	-
Ser XR-1	0.7	18	45	5	18
M87 (Virgo)	0.6	12	29	12	37
Cas A	0.5	23	22	58	34
Leo XR-1	0.5	9	33	8	36

*The uncertainty of position for any of these x-ray sources is approximately 1.5° except for Tau XR-1 and SCO XR-1.

b. Present Development in the Field

The development of EVA capabilities since the flight of Gemini 12 has progressed by aircraft zero g experiment and underwater simulation. While these synthetic techniques cannot substitute for actual space environment flight experience, good correlation factors are being developed which will be extremely useful in the design and astronaut training areas. The optimum degree of man participation vs automation is a different and complex problem. In the design of any orbital system, many of the feasibility and trade-off areas can be resolved by simulation experiments. The validity of the simulations will be proved in future flights.

The Apollo mission will contribute adequate basis for substantiating man's role in the x-ray telescope in sufficient time to be useful in the latter design phases.

The telescope design proposed utilizes a thin wall titanium truss structure which is current technology. The proposed structure is not the ultimate for several reasons, the most significant of which is thermal distortion. The Saturn booster capability permits almost unlimited choice of structure concepts and materials; however, none of the designs evaluated in the first phase exhibited satisfactory thermal distortion characteristics. Titanium was selected as a compromise material which, with an active alignment system, will perform the mission within tolerances.

The development of new structural materials with nearly zero thermal expansion coefficients would provide improvements in system reliability for the x-ray telescope. Such a material would be extremely useful in all large space structures, particularly telescopes where high resolution demands precise control of widely separated components. In the scientific development area there are several programs which relate directly to the x-ray telescope.

Among the current and planned programs falls a newly announced NASA Small Astronomy Satellites (SAS) program. The first spacecraft is scheduled to be launched in 1969. The objective of the program is to map x-ray stellar sources within and outside our galaxy. This is under the management of Goddard Space Flight Center, to be launched with a Scout booster.

Although man may be phased into the program as early as 1969, NASA also sees a continuing need for smaller automated observatories and explorers throughout the next decade and beyond

The continuation and expansion of the Explorer satellite program will permit the active participation of university scientists. The x-ray astronomy satellite is a comparatively simple satellite which will carry a complement of x-ray detectors to obtain an all-sky map of x-ray sources, including those of about an order of magnitude fainter than those observable from a short-lived rocket and rough spectral curves for the brighter sources.

Plans are also being formulated for small spinning and pointed satellites for surveys in the ultraviolet, x-ray and gamma-ray wave lengths.

In the NASA OSSA Prospectus it is proposed that the present OSO and Solar Explorer programs be continued with minor improvements made to the spacecraft. During the early part of the 1971-74 period, NASA envisions a need to study the hard components of electromagnetic radiation from about 1 KeV to the highest energy photons observable. During the early part of that time period, the sun will still be relatively active. This activity is manifested in part by the emission of hard x-radiation.

Between the automated observatories of the 1960's and the very large observatories of the 1980's, man will be phased into the astronomy program to an increasing degree. He makes possible the use of film as a rapid, compact, high resolution detector and data storage device. He can play an increasing role in repairing or replacing failed parts to enhance reliability and in exchanging auxiliary equipment to increase instrument versatility. He will be phased into the program in a major way with the launch of the solar Apollo Telescope Mount (ATM). The approved ATM has instrumentation for high spatial and spectral resolution studies in the ultraviolet and x-ray regions. It will study solar x-rays, particularly those emitted during flare conditions, which appear to have their sources in highly localized regions.

Three ATM's may be employed for stellar and galactic observations. The first ATM will have two x-ray grazing incidence telescope. Dr. Riccardo Giacconi of American Science and Engineering, Inc., is principal investigator for an x-ray spectrographic telescope. This experiment proposes to obtain x-ray photographs of flares with a spatial resolution of about 2.5 arc sec. and to simultaneously record spectrally disbursed emissions over the range of 2-8 Å with a resolution of a fraction of an angstrom. An electronic flare detector, in addition to the main telescope, will give the astronaut a visual indication of flare build-up. This will allow him to select active regions and photograph the flares in the early x-ray rise periods.

The NASA Goddard Space Flight Center's x-ray telescope uses a different technique to get information on impending flare build-up. Principal investigator for the Goddard experiment is James Milligan. This experiment will use a grazing incidence x-ray and extreme ultraviolet telescope with a resolution capable of recording the solar x-ray distribution in the 3-100 Å wavelength region. The information will be recorded on ultraviolet sensitive 35 mm roll film. There are also two proportional counters to monitor the total solar x-ray flux in the spectral regions of 2-8 and 8-20 Å. This data will be pulse-height sorted and recorded on tape.

After the ATM and the automated observatories, man-assisted and -maintained observatories with both longer lifetimes and more accurate pointing capabilities, should be phased into both the solar and stellar programs. The large focusing telescope design presented in this report is intended for these purposes. Either as part of a national astronomical observatory or by itself, it will consist of instrumentation to conduct a wide variety of astronomical observations. This major instrument has been strongly recommended by the 1965 NAS Woods Hole Summer Study, the President's Science Advisory Committee and others. Comparing x-ray astronomy with radio astronomy, it is likely that at the time this telescope or a complete national space observatory is flown, a number of x-ray sources will be known with moderately good information on their spatial structure and spectra, but that to understand the physical characteristics of these sources and in particular the source of their enormous energies, high resolution information will be imperative.

The proposed x-ray telescope appears to be a logical instrument to accomplish soft x-ray observations in the mid-1970's. The telescope may be used in conjunction with a space station which could provide the necessary maintenance and support.

4. EXPERIMENT APPROACH

a. Experiment Concept

The x-ray telescope utilizes the technique of focusing the x-ray energy by means of double reflection at low angles of incidence, making imaging possible.

Imaging x-ray optics have already been used successfully for solar observations. Such telescopes can now be utilized for the observation of cosmic sources. In principle, they can produce images of cosmic x-ray sources with angular resolutions comparable to what can be achieved in visible light. In addition, they can focus a parallel beam of x-rays impinging on a large collecting area on to a small target area, where they can then be detected with a much improved signal-to-noise ratio. The latter application permits the use of dispersive techniques for high-resolution spectroscopic measurements, and affords improved possibilities for polarization measurements. The x-ray reflecting telescope should prove to be a powerful tool for the study of galactic and extragalactic sources at wavelengths greater than 2Å.

The optical devices for focusing x-rays consist of reflectors on which x-rays impinge at small angles and are totally externally reflected. Total reflection optics are essentially achromatic at wavelengths longer than a certain cutoff value that depends on the atomic number of the reflecting material. The lowest cutoff wavelength that can be achieved at present is 2 or 3Å. This limit can probably be extended to about 1Å. Reflection optics do not substantially alter the spectral shape and the state of polarization of the impinging radiation. Two or more reflections are necessary to remove first-order aberrations of a focusing system. The reflecting surfaces of the objective under consideration here are a paraboloid and a hyperboloid, after Wolter's original ideas.

For most cases in which imaging systems are used in practice, the lower limit on the intensity of a discrete source that can be detected by an x-ray telescope is set, not by the signal-to-noise ratio, but rather by the requirement that a significant number of photons be detected within an image resolution element. This is understandable since the area of a resolution element in the focal plane is many orders of magnitude smaller than the area of collection. The part of the background that depends on the area of the detector element, such as that produced by cosmic rays, will therefore be reduced by a corresponding factor when compared with detectors with only mechanical collimation, and can be further reduced by increasing the angular resolution of the device. The diffuse cosmic x-ray flux background is much weaker than the background from other factors. In view of these considerations, the effective sensitivity of a focusing telescope is proportional to A_t, the product of the area of collection and the time available for observation. The sensitivity of mechanically collimated detectors, which is determined by the signal-to-noise ratio, is, on the other hand porportional to $(A_t)1/2$.

The nominal 30 in. x-ray telescope is shown deployed in orbit with the Apollo CSM docked in the model photograph Figure I-1. The normal scientific mission would be conducted without the CSM. The basic telescope structure consists of two telescoping truss assemblies fabricated with thin walled titanium tubes attached to precision cast spider socket fittings. The 40 ft. deployed truss assembly tapers from a maximum depth of 13.3 ft. to 8.7 ft. on the aft end, and 40 in. on the forward end. This geometry provides maximum structural stiffness consistent with the booster launch envelope.

Preliminary structure concepts evolved octagonal structure geometries to provide maximum solar heating symmetry for a fully shrouded vehicle. The cross section was changed to a square for simplicity when thermal distortion analysis showed little improvement in the structure distortions with a full shroud. The open structure has the further advantages of maximum EVA access, minimum weight, and minimum aerodynamic drag.

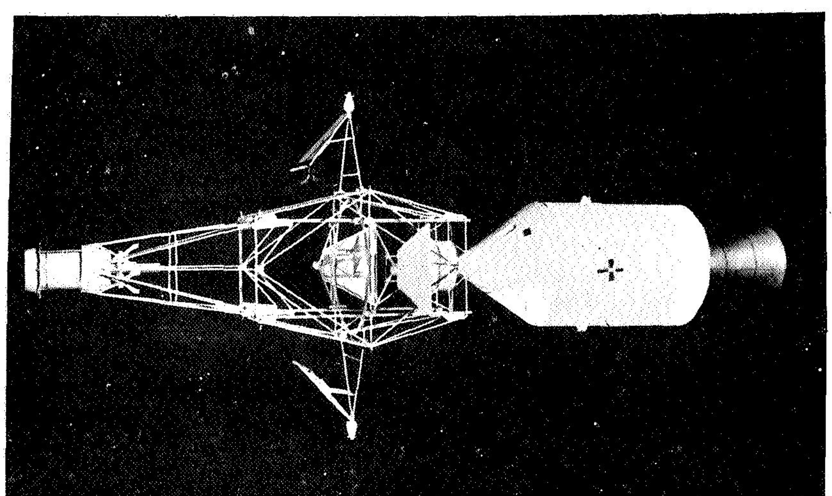


Figure I-1. Orbital Telescope Configuration with CSM Docked

The 2 in. diameter tube size was selected for astronaut hand grip compatibility rather than stress requirements, which are very low. The ratio of deployed to stowed structure length is ≈ 2/1 permitting a simple rail guided slider extension system for maximum reliability. The sliding truss sections are positively locked together after extension eliminating all joint slack.

The selection of titanium for the primary structure was based principally on fabrication, astronaut impact, and thermal expansion considerations. Further analysis is required to derive the optimum material which will likely be selected primarily on its thermal stability.

The Saturn I-B booster has ample payload capability to permit large weight increases in the proposed telescope.

The deployed structure is shown on the in-board profile Figure I-2.

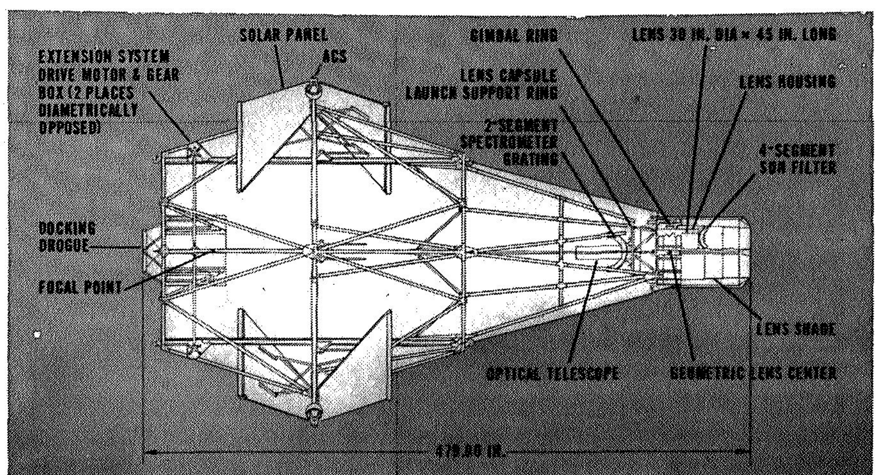


Figure I-2. Deployed Configuration

Truss Extension System

The forward lens truss assembly, secured inside the primary truss during launch, is extended to the deployed position by an electromechanical cable run-around system. Forty in. long guide sliders on the lens truss ride on four parallel rails which are attached to the primary truss. The track sliders maintain the parallel orientation between truss assemblies. The critical alignment between assemblies is adjusted by the lens alignment systems.

The extension system primary activation power is provided by 2 1/4 hp electric motors which, through reducing gear boxes and synchronizing torque shafts, rotate four cable capstans. The cable system is pretensioned and reversible; i.e., if a failure occurs, or appears evident to the CSM crew controlling deployment the operation may be stopped or reversed until the malfunction is corrected. The drive motors are redundant and in the event that both fail the gear boxes have provisions for EVA portable drill motor plug-in to deploy the lens. At the fully extended position the two trusses are rigidly locked together at 8 places and the four forward cable guide sheaves are shifted aft to the cable tension. The high cable system reliability approaches 100% with the astronaut back-up. The lens shade is automatically extended by cables during the final truss extension, the truss extension motors are sized for additional load.

Lens Group Assembly

The x-ray telescope lens group assembly is separable from the truss assembly at one of the gimbal axes, and consists of the following subassemblies:

(1) Lens Shroud Assembly
    - Sun Filter Assembly

(2) Gimbal Ring Assembly

(3) Sun Shade Assembly

(4) Reflecting Lens Assembly
    - Optical Telescope
    - Alignment System Reflector Package

Lens Shroud Assembly

The lens shroud assembly consists of a traveling structural support ring approximately 50 in. in diameter and 18 in. long. The ring contains trunnion points for the outer Y axis lens gimbal, hold down points for the lens launch tie down, jack screws for adjusting the focal length, and supports for the sun shade and extension tubes.

A segmented hinged sun filter is mounted on the forward end of the ring. The filter consists of four ammular segments of 0.5 mil beryllium sheet to shield the lens and imaging equipment from the solar UV radiation during solar observations. The sun filter segments are spring loaded to fall open. The filter closing mechanism is a small electric motor and cable driven. Pyrotechnic cable cutters would be used to open the filter if the drive motor or gear box malfunctioned.

Gimbal Ring Assembly

The gimbal ring assembly is another part of the lens alignment or optical axis adjustment system.

The inner gimbal assembly provides X axis trunnions for the mirror assembly. Gimbal power for both axes is provided by stepping motors in conjunction with negligible backlash reduction gearing, like the harmonic drives. High accuracy encoders are required to sense the lens rotation angles as this data is required by the navigation system since changes in lens rotation constitute a shift in the vehicle optical axis, with respect to the star trackers and inertia wheels. The inner gimbal motors are extremely difficult to replace by EVA and will probably be required to operate for the entire x-ray telescope mission lifetime. See Figure I-3.

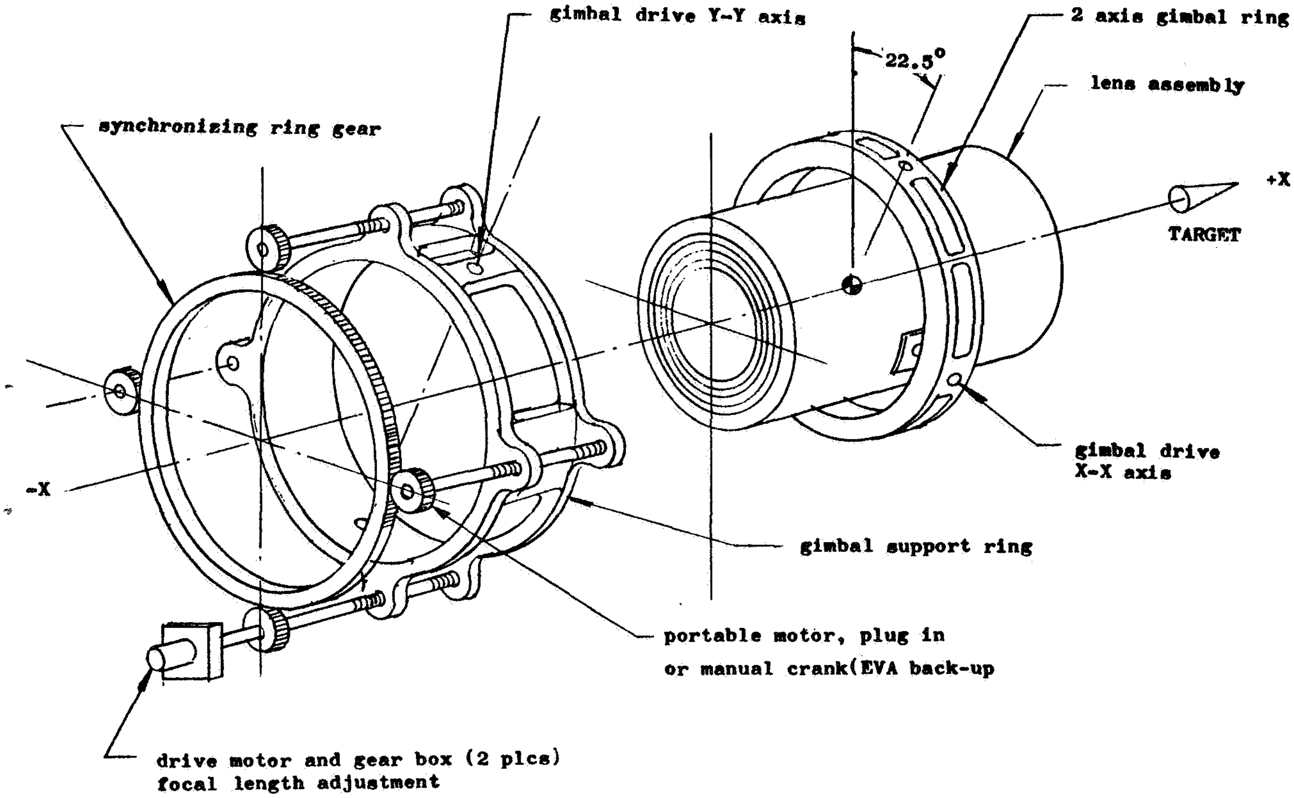


Figure I-3. Lens Adjustment Mechanism

Lens Sun Shade Assembly

The sun shade assembly is provided to prevent off-axis solar radiation from striking the lens. The assembly is basically a tube of woven, impregnated fabric which is collapsed during launch and extended by the force of the lens truss extension. The fabric is supported by a forward metallic ring attached to four tubes which are in turn supported by the lens shroud ring. The geometry is sufficient, with additional internal baffle rings, to shield the lens from direct solar heating when the telescope is pointed greater than 40° from the sun. A similar shade would be required behind the lens but this will not require collapsing and would be attached to the lens truss. The forward shade can be designed for shielding as close as 20° to the sun if a complete mission analysis determined this was desirable. It is anticipated that sufficient stellar sources will exist to permit more selectivity in viewing angles with respect to the sun.

Reflecting Lens Assembly

The principal structural element in the x-ray telescope is the reflecting mirror system. The grazing incidence mirrors are formed of parabolic-hyperbolic surfaces which barely converge from a cylinder. The effective aperture of a grazing incidence mirror is extremely small compared to its overall surface area because of the necessity to reflect the incoming x-rays at very small angles below 1° in the shorter wave lengths 2 to 10 A. Increased aperture area is best achieved by multiple confocal mirror segments, the multiple design also tends to balance the reflection efficiencies throughout a wider range, as each mirror has a different focal length to diameter ratio and a correspondingly different efficiency due to the change in grazing angle.

American Science and Engineering, Inc. provided Convair with the general mirror surface tolerances which would be required to meet the scientific community's measurement objectives. These are listed below.

Reflecting Mirror Tolerance Summary

(1) Optical surface finish - 1.0 micro in.
(2) Out of roundness - 5 micro in.
(3) Concentricity between nested segments 0.0015 (lens resolution x focal length)
(4) Maximum deviation from theoretical contour 0.715 micro in. per inch of figure length
(5) Radius dimension 0.001 in.
(6) Focal length matching between segments 1% of focal length
(7) Angular misalignment between segments arc sec.

The beryllium three mirror assembly used in the final vehicle sizing is shown in Figure I-4.

Mirror Materials

A cursory evaluation of lens materials has been conducted primarily from a feasibility aspect. No attempt can be made to optimize mirror materials until a mission is defined down to the primary spectrum of interest, vehicle to sun critentations, etc. Current x-ray mirrors are being manufactured of fused silica, beryllium, 440 stainless steel, electroformed nickel, and aluminum. Surface coatings, such as Kanogen, are used to add a material which has more desirable reflection characteristics or is easier to polish or figure than the base material.

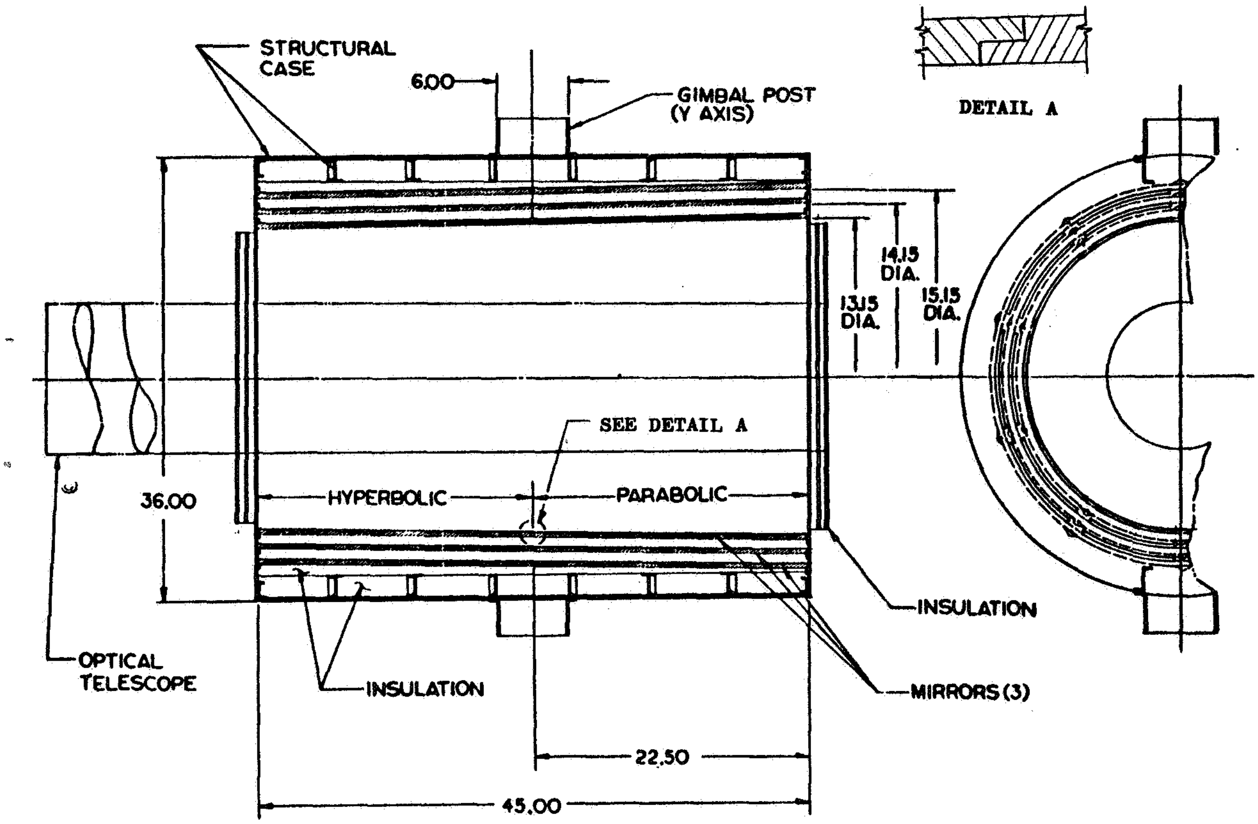


Figure I-4. Reflecting Lens Assembly

The mirror material selection is quite sensitive to the thermal environment and detailed thermal transient analyses of various designs will be necessary to calculate the exact temperature excursions of the mirrors. If the increased weight can be neglected, which appears true for the sizes up to 40 in., the low coefficient glasses are ideal for the fluctuating thermal environment.

Image Capsule Assembly

The image capsule is the prime reference base for the orbiting vehicle. It contains the mission scientific receiving equipment, the navigating equipment, power conditioning, command/control and telemetering equipment and docking provisions. The capsule is temperature controlled, but is not pressurized.

Geometrical Arrangement

The basic capsule diameter is 54 in. and the length is 48 in.. Insulation is added outboard of these basic dimensions along with four full length access doors having integral temperature controlled louvers.

The structural arrangement consists of cross and vertical beams, walls, and tension straps. Double cruciform beams, three in. deep and spread 15 in. apart form bulkheads at the forward and aft ends of the module. Post members connect the forward and aft bulkheads at three of the four apexes formed by the double cruciform. The fourth apex is the pivot axis for the x-ray receiving equipment turret. The primary construction material is 2024 aluminum alloy. All internal equipment and walls are accessible through the access doors by a space suited astronaut. Most equipment is immediately inside the access doors on swing-out racks. Thus both sides of the equipment is immediately at hand.

The capsule contains four access bays - I, II, III and IV. Bay I contains the x-ray receiving equipment mounted on a turret which has seven stations, two imaging, two polarimetry, one crystal spectrometer, and two empty stations for unassigned equipment. The turret rotates each station to the focal plane of the primary x-ray mirror. The volume available for instrumentation is shown in Figure I-5.

The image capsule access doors are louvered thermal control doors with built in foot restraints to permit the EVA astronaut to secure himself to the door, open it by portable electric drill motor and he is then securely anchored to a good work site for access into the image capsule.

Docking Assembly

The telescope satellite is equipped with a LEM docking cone supported by shock mounts for docking with an Apollo CSM. Docking is required for the initial extraction of the telescope from the SLA and post deployment docking for servicing and/or repair.

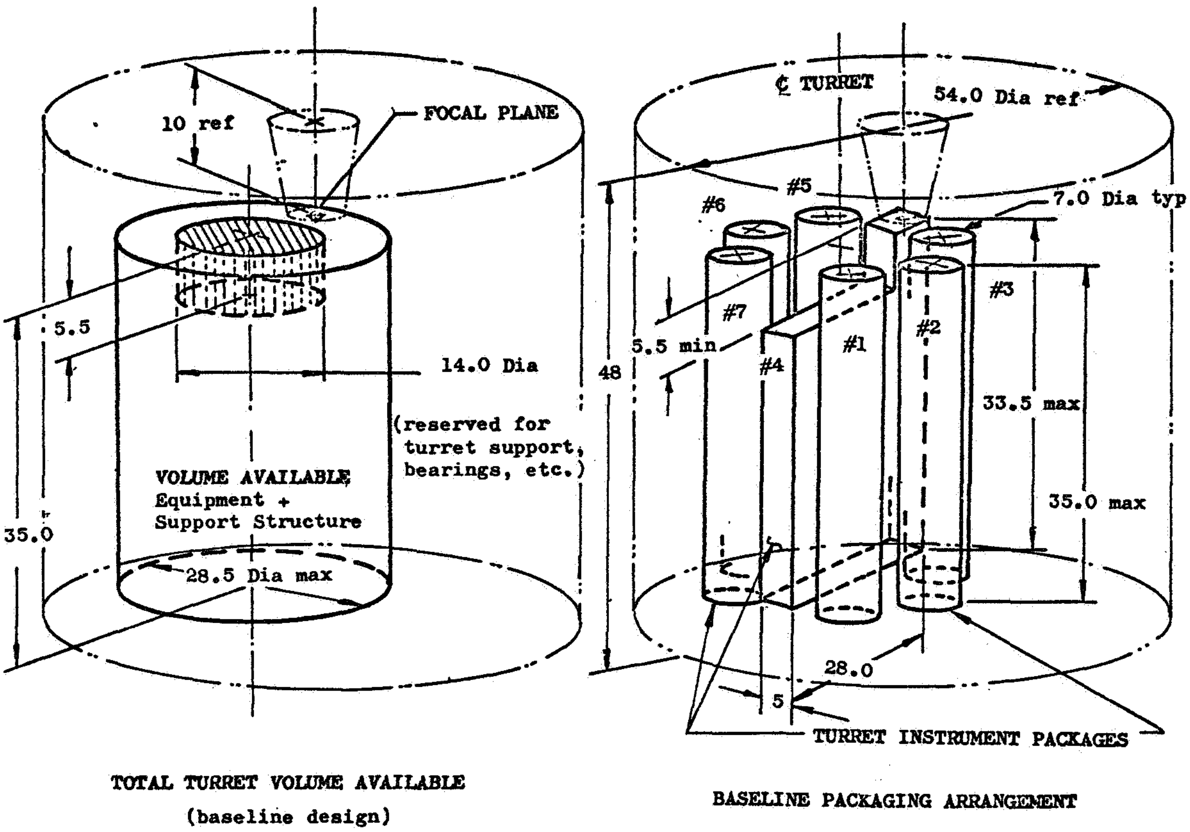


Figure I-5. Turret Packaging Volumes

Solar Panel/ACS Assembly

Four "A" frame support booms are hinged from the main cross members of the primary truss in the same plane as the SLA attachment points. The frames support four 35 sq. ft. solar panels and 4 attitude control modules consisting of three 0.5 lb. H_2O_2 thrustors and required propellant hardware. The assemblies are deployed prior to truss extension, and once deployed will provide gross vehicle stabilization. The booms deployment mechanisms are designed to be reversible to permit the astronauts to replace ACS modules and solar panels while firmly anchored to the primary truss structure. The ACS modules are designed with the "add-on" philosophy; i.e., the new packages can be added on the outboard end of the expended module attached only with a quick release mechanical fitting and an electrical connection. The solar cell panels are rigid cells on flexible substrate to permit them to be stowed, and transported to the worksite in compact rolls more easily than open panels.

Launch Support and Separation Systems

In addition to the four primary SLA reaction points the x-ray telescope design includes provisions to support the following assemblies during the boost phase.

(1) Forward truss
(2) ACS/Solar panels
(3) Reflective mirror assembly
(4) Image capsule equipment turret.

All of these systems are designed for remote release with EVA back-up except the turret release which is EVA primary.

b. Experiment Procedures

The x-ray telescope is launched into earth orbit using a Saturn I-B launch vehicle. The packaged arrangement is shown in Section III.

The initial deployment of the telescope follows the standard Apollo-LEM sequence. When the desired orbit is obtained the CSM separates from the adapter section (SLA), rotates 180^{o} and docks on the telescope docking adapter. Once the hard docked condition is achieved, the CSM crew commands the separation of the four telescope support points to the adapter. With the separation complete the CSM maneuvers clear of the spent stage, and makes any final necessary orbit corrections. Prior to extension of the telescope the CSM structure crew are only required to inspect the telescope package from the CM; i.e., no EVA is planned until after deployment. The four ACS/ solar panels are extended individually under the control of the CM crew, the mechanisms are required to be reversible for resupply, therefore, the crew has full control and would stop the cycle at the outset of any malfunctions. Once the ACS/solar panels are deployed, the lens truss is extended; this mode also energizes the forward lens sun shade.

Following deployment of the extendable structures EVA is planned to permit the astronaut inspection of the interior of the image capsule and removal of the turret launch restraints and test operation of the turret. The complete operational checkout is performed without EVA by ground control with the CSM still docked. On completion of successful systems checkout the CSM separates from the telescope and departs, leaving the telescope under command of the ground station.

Assuming the equipment continues to perform satisfactorily, the observational x-ray mission will be conducted under the control of the ground facility for two years, at which time a scheduled rendezvous for resupply is planned. Repair flights may be scheduled at 90-day intervals on the basis of current low earth orbit manned launch planning. Repair missions have not been synthesized. The two year resupply mission was assumed to include replacement of the four attitude control modules, solar panels, drag correction motor and three scientific packages.

Mission Profile

The principal investigators will dictate the mission profile; however, it is probable that the order of events might take this form:

(1) Command lock-on to within a $\pm$ 0.5 arc sec. of known coordinate spot in the sky -- accomplished by guidance system using on-board computer and star trackers.

(2) Visual telescope (slaved to the x-ray telescope) identification of target area. Electronic comparison of visual telescope image with desired coordinates holds the x-ray telescope to within a field of 1 arc min. by 1 arc min. centered on the desired coordinates.

(3) The imaging, spectrometry and polarimetry investigations of the target take place, while the automatic attitude control system holds the telescope pointed with the 1 arc min.

by 1 arc min. field centered on the desired coordinates. The desired image resolution of 2 arc sec. (or whatever is achievable by the optics) is obtained using an image motion compensator to correct for drift.

(4) This step would entail about 1-60 min. for imaging, from 10-150 min. per line for spectrometry, 60-1000 min. for the entire spectrum using a slitless spectrometer suggested by American Science and Engineering, Inc. and 5-60 min. for polarimetry, assuming a source on the order of 10^{-4} the intensity of SCO X-1, (SCP X-1 has an intensity of approximately 18 counts/cm^2-sec.) and an x-ray telescope on the order of 30-40 in. mirror internal diameter, with two to eight mirrors.

A traverse in pointing direction takes place and steps 1 - 3 are repeated for another x-ray source or interesting area of the sky. Most sources will be within $\pm$ 20° of the galactic equator. The Milky Way will provide a large number of viewing opportunities.

The data is stored in the spacecraft and transmitted to ground upon passing a ground station. If film is used for imaging, the exposed film will periodically be retrieved and resupplied by the astronaut. The data is analyzed on the ground.

Although the telescope is normally operated unmanned, it is designed to be compatible with manned support including resupply and refurbishment as well as maintenance and repair. The system concept provided is designed to maximum equipment accessibility to an EVA astronaut without specialized support equipment such as stabilized maneuvering aids, etc. Whereever possible astronaut restraints are incorporated in the basic structure; for example, the access doors on the image capsule.

The maximum interval between resupply was assumed to be two years based on extrapolations of existing hardware designs; the attitude control system was provided sufficient propellant for this period. Due to the highly refined nature of much of the scientific equipment it is doubtful that a two year unsupported mission could be achieved. The problems of protecting the slitless spectrometer gratings, filters, star tracker optics, and image tube surfaces from environmental erosions were not sufficiently analyzed to verify the time intervals between resupply; it is fairly clear that two years (without resupply) is an upper limit for this type of satellite. A detailed time line analysis of a resupply mission which replaced the four ACS units, four solar panels, drag velocity motor, and three of the seven scientific instrument packages. The resupply operation required 4 days, 18 hours and 12 min. of total time and 17 hrs. and 32 min. of EVA (external to the command module).

c. Measurements

Orbital measurements are required in each of the three flight objective areas: EVA, STRUCTURES, and SCIENTIFIC.

The assessment of man's capabilities will require the following biomedical measurements:

(1)	Oxygen Consumption	(5)	Suit Pressure
(2)	EKG	(6)	Suit Temperature
(3)	Body Temperature	(7)	Suit Oxygen Concentration
(4)	Respiration Rate	(8)	Suit Relative Humidity

The above measurements will be handled by the Apollo CM data system and do not represent a design requirement for the telescope.

In addition, data will be gathered on the actual EVA tasks vs time for subsequent comparison to the schedule and for development of new baseline capability data. The time-task information will be supplemented with photographs of the EVA astronaut taken by the stand-by astronaut.

The evaluation of the telescope structure and mechanical performance will be in three basic phases. The deployment and resupply (or maintenance and repair) phases are manned and will utilize the flight crew observations and inspections augmented with photography, in addition to the measurements taken.

In the deployment phase the following specific measurements will be made.

		Range of Values
(1)	Solar/ACS boom deployment motor power	100 to 300 watts
(2)	Truss extension " "	"
(3)	Lens focal length extension " "	"
(4)	Lens alignment " "	"
(5)	Turret rotation " "	"
(6)	Access door drive " "	"
(7)	Strain measurements on primary trusses	

* The motor sizes have not actually been specified but are expected to be approximately 1/4 horsepower.

The principal value of the above measurements will be verification of the design conditions.

During the unmanned operation phase only the lens alignment and turret rotation motors are operating, these motor power levels and the image capsule and primary mirror temperatures will be recorded for short periods daily to monitor the effect of time in orbit on the bearing friction.

The optical axis distortion angle generated by the on-board computer and the lens alignment system will be recorded to provide a time-distortion history which will give an indication of the degradation of the structure thermal coatings. The truss will be instrumented in approximately six locations with an array of temperature sensors and strain gages which measure the tube circumferential temperature and stress distortions.

During repair or resupply cycles the solar/ACS boom deployment and access door drive motor operating power can be remeasured to provide information on the dormant degradation of the motors and bearings.

In the scientific experiment area the precise measurements are not defined, however, the telescope is intended to perform imaging, spectrometry, and polarization measurements on previously identified x-ray sources or areas. No attempt has been made to predict the number of sources or areas of the sky which will be of interest by the flight time of this instrument but a safe guess is that the presently known two dozen or so x-ray sources will expand to several hundred. The general objectives will be to:

(1) study their structural nature (with 1 arc sec. resolution)
(2) locate these sources precisely (within arc sec.)
(3) measure their intensity of x-radiation vs wavelength
(4) determine their expanse or physical size (arc sec.)
(5) determine polarization
(6) determine time intensity variations, if any.

The capabilities of the telescope to accomplish the above are:

(1) Field of View or Beamwidth: 30 arc min.
(2) Angular Resolution: 2 arc sec. or better -- depends on jitter rate and number of exposures per sec.
(3) Pointing Capability: 5 arc min. off-set.
(4) Lock-on Accuracy: Within a 1 arc min. square.
(5) Jitter: Less than 1 arc sec./sec.
(6) Sensors or Detectors: 2 Image intensifies plus film or electronic imaging tube. 2 Spectrometers, 2 Polarimeters, and 2 Space Stations.
(7) Bandwidth: $\sim 2\text{Å}$ to 300Å.
(8) Spectral Resolution: $\lambda/\Delta\lambda = 100$ to 1000 or better.
(9) Collecting Area: 200 cm^2 minimum effective collecting area.
(10) Lock-on Time: Up to 90 min.
(11) Data: $5\text{-}10 \times 10^6$ bits per orbit for scientific usage.

d. Data Analysis and Interpretation

Data analysis and interpretation is the responsibility of the principal investigator, and will be determined at a later date.

e. Prime Obstacles or Uncertainties

The design of the x-ray telescope is based on current technology with certain qualifications. The structural assemblies, truss, image capsule, and lens assembly (except the mirrors) are conventional structure design not requiring any unusual fabrication techniques. The basic truss material is titanium, the image capsule and lens support structure are aluminum.

Although it would be advantageous to develop new structure materials with extremely improved thermal stability, these developments were considered unlikely and the alternatives of thermal control (heat pipe) and compensation were evaluated. The gimbaling lens compensation system using laser reflection techniques were selected as a workable solution to the structure thermal distortion problem. The alignment system is considered current technology.

The 30 in. nominal diameter multi-segment mirror assembly is larger than any similar optics presently developed. The 9 in. ATM mirror currently being developed is the largest grazing incidence lens attempted. However, the uncertainty of developing the larger 30 in. mirror, using current design, materials and fabrication techniques, is considered small by American Science and Engineering and Perkin-Elmer.

The design and development of the vehicle subsystems, electrical power, ACS, navigation, drift error, TLM, command and lens alignment is considered to be within current technology. Some uncertainty exists with regard to the scientific instrumentation as these systems are not sufficiently well defined.

The substitution of film for the proposed electronic imaging will involve some compromises and require increased astronaut participation. Electronic imaging of equal resolution is preferred because of the reduced volume, and resupply requirements and the almost immediate availability of the observational data to the astronomers.

f. Astronaut Participation

The astronauts role in the experiment is to serve in the areas of deployment, observation, checkout, malfunction repair and gross refurbishment. The function of man has been analyzed and the results reflected in the basic design of the experiment to provide the greatest possible assurance of mission success.

The basic satellite systems and component parts have been analyzed on a matrix of expected failure rates, astronaut capabilities, hazards and system costs. This analysis serves as the basis by which the operational modes of the systems are selected. Component part redundancy and replacement by EVA were weighed for system effectiveness. Using this approach, man's activities have been designed into the mission only when justified on the basis of increased probability of mission success.

The objectives of planned EVA on the x-ray telescope are four-fold and generally apply to all similar experiments.

(1) Develop confidence in man's ability to perform the tasks necessary to support a useful scientific mission.

(2) Achieve the highest deployment reliability to guarantee the effective initiation of the scientific mission.

(3) Extend the useful lifetime of the system beyond the normal component and subsystem lifetimes by repair, replacement, and resupply of critical items.

(4) Update the experiment capability with the latest equipment designs to deter obsolescence.

These combined objectives are achieved by specific activities; in the x-ray experiment, the manned contributions for the respective mission phases are listed in Section IV.

5. BASELINE OR CONTROL DATA

While the design of the majority of the x-ray telescope subsystems is based on the current state-of-the-art support studies in the following areas will reduce the normal telescope development program effort.

a. Long memory high resolution imaging electronic equipment.

b. Improved packaging of focusing crystal spectrometers or new designs

c. High efficiency X-Ray polarimeters

d. "Zero" expansion coefficient structure materials

e. Compatibility of telescope with space station to permit film or improve maintainability

f. Degradation of grazing incidence lenses in space environment

g. Mechanisms and motors for four year space use

h. Methods to reduce cost of precision mirror assemblies

Additional support activities that are recommended to augment the x-ray telescope are two prerequisite orbital experiments:

a. "Clothesline" Supply. This is an orbital experiment in which the feasibility of supplying a remotely located EVA astronaut with tools and materials and for return of these items to the CSM through the use of a "clothesline" type system is demonstrated; i.e., any direct transfer system.

b. Astronaut Locomotion Loads. This experiment is recommended to determine the actual loads on a typical large structure resulting from astronaut movement on the structure and impact with the structure while transferring from the CSM in the space environment.

It is anticipated that these experiments will be flown to resolve requirements of other structures similar to the x-ray telescope. The experiments are not necessary to resolve feasibility; primarily they will contribute to reducing design safety factors.

SECTION III - ENGINEERING INFORMATION

1. EQUIPMENT DESCRIPTION

a. Experiment Hardware

The experiment hardware comprises the large space structure of the x-ray telescope, the image capsule, grazing incidence lens assembly, lens structures and the telescope subsystems:

(1) Electrical power and distribution

(2) Command

(3) Navigation

(4) Data, TLM, and communications

(5) Attitude control

(6) Optical drift error

(7) Lens Alignment

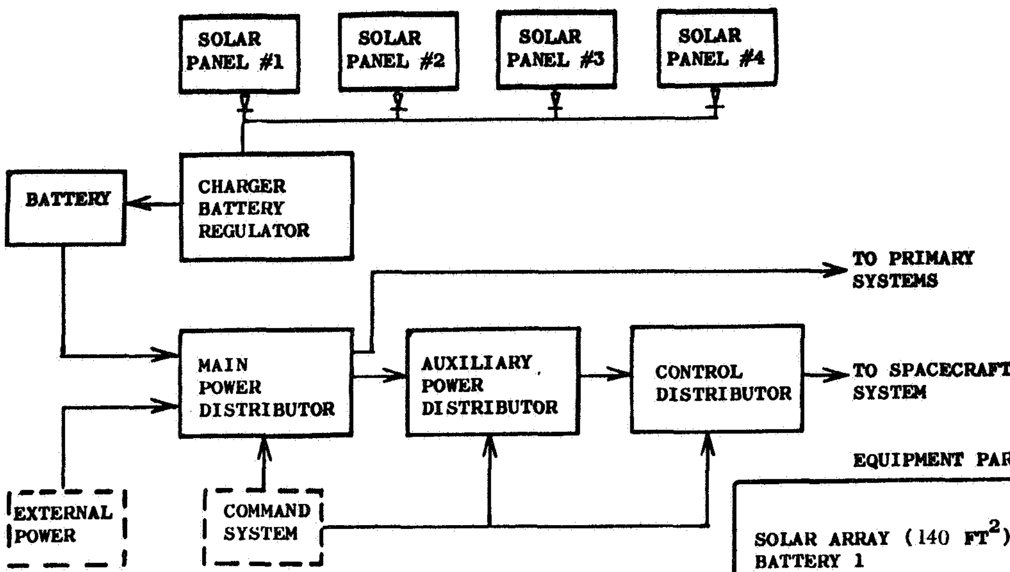
(8) Telescope instrumentation

The mission oriented telescope instrumentation and optical drift tracking subsystems were developed through consultation with American Science and Engineering, Inc.

The remaining lens alignment subsystem was developed after structural analysis indicated that the thermally induced distortions of the structure would at times exceed the maximum derived limits of 8 - 15 arc sec.

The electrical power subsystem consists of an omnidirectional, fixed solar panel array, batteries, conditioning and distribution equipment.

The average power to supply the various systems will be 260 watts with a peak power of 450 watts. Solar cell power is supplied for an average of 65 percent of each orbit. The solar cell systems consists of four 35 sq. ft. panels designed for projected 1972 performance of 13.5 watts/ft^2. These values are slightly more conservative than industry projections for this time period. The roll-up arrays (for simplified replacement) utilize rigid cells on a flexible substrates. The system block diagram is shown on Figure 1-6.



EQUIPMENT PARAMETERS

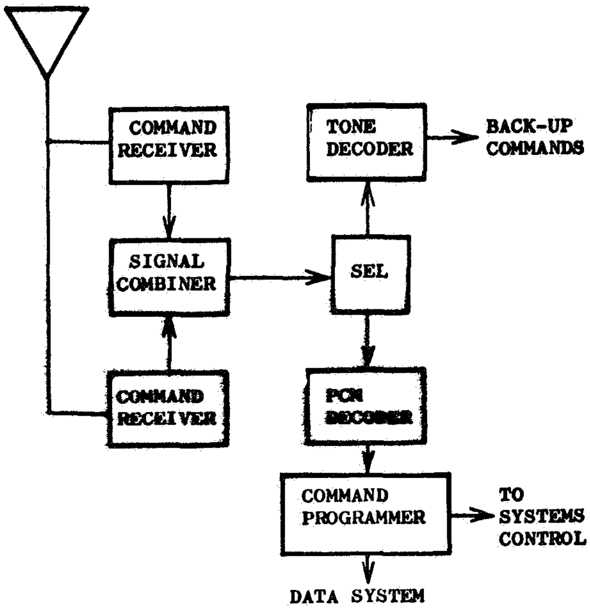
	IN^3	LBS
SOLAR ARRAY (140 FT^2)	13000	173
BATTERY 1	1500	36
BATTERY 2	1500	36
BATTERY 3	1500	36
REGULATOR	100	5
BATTERY CHARGER	80	3
MAIN PWR DISTR	146	5
AUX PWR DISTR	55	3
VOLTAGE BOOSTER	64	5
VOLTAGE LIMITER	8	1
LOW VOLTAGE SENSOR	8	1

Figure I-6. Electrical Power and Distribution System

Command System

The command system is designed to respond to approximately 100 remote command functions in the operation of the telescope and associated equipment. The commands will include deployment, vehicle pointing, control of sun shield, gratings, instrument turret, selection and modes of operation of data storage and multiplexing equipment.

The command system receives digital data from ground stations verifies it, determines the system and function to which it is addressed and routes it to the function. The system consists of two command receivers, a signal selector, a detector -decoder, an interface and buffer storage, relay drivers and a relay assembly. The command system maximum power consumption is 15 watts. See Figure I -7 for system block diagram.



EQUIPMENT PARAMETERS

	°C	IN^3	LBS	WATTS
COMM RCVRS (2)	+65°	85	4	4
SIGNAL CONDITION	-20°	15	1	1
TONE DECODER	"	40	1.5	2
PCM DECODER	"	38	2	3
COMMAND PROGRAM	"	70	5	5
HARNESSES, BRKTS			5	

Figure I-7. Command System

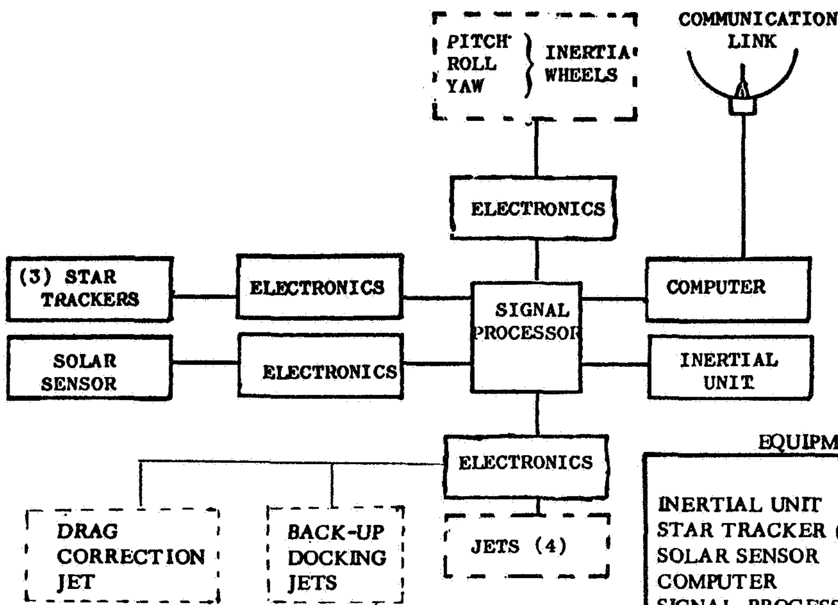
Navigation System

The x-ray telescope navigation system utilizes ground based sensor and computer facilities as much as possible to improve system reliability since the information to be transmitted is small and continuous ground contact is small. The ground based functions will include orbit determination orbit integration, source programming, star tracker pointing and control of drag velocity loss.

The vehicle operates with two star trackers and a solar sensor, a third back-up star tracker is installed and kept shielded from the space environment until needed. The solar sensor aids initial orientation and will maintain coarse vehicle stabilization for docking in the event that two star trackers fail.

The vehicle computer is similar to the IBM 4 π general purpose, stored program, binary operated airborne computer.

A simple inertial reference unit utilizing three displacement gyros is included for dark side stabilization if the reference stars become occulted. See Figure I-8 for block diagram.



EQUIPMENT PARAMETERS

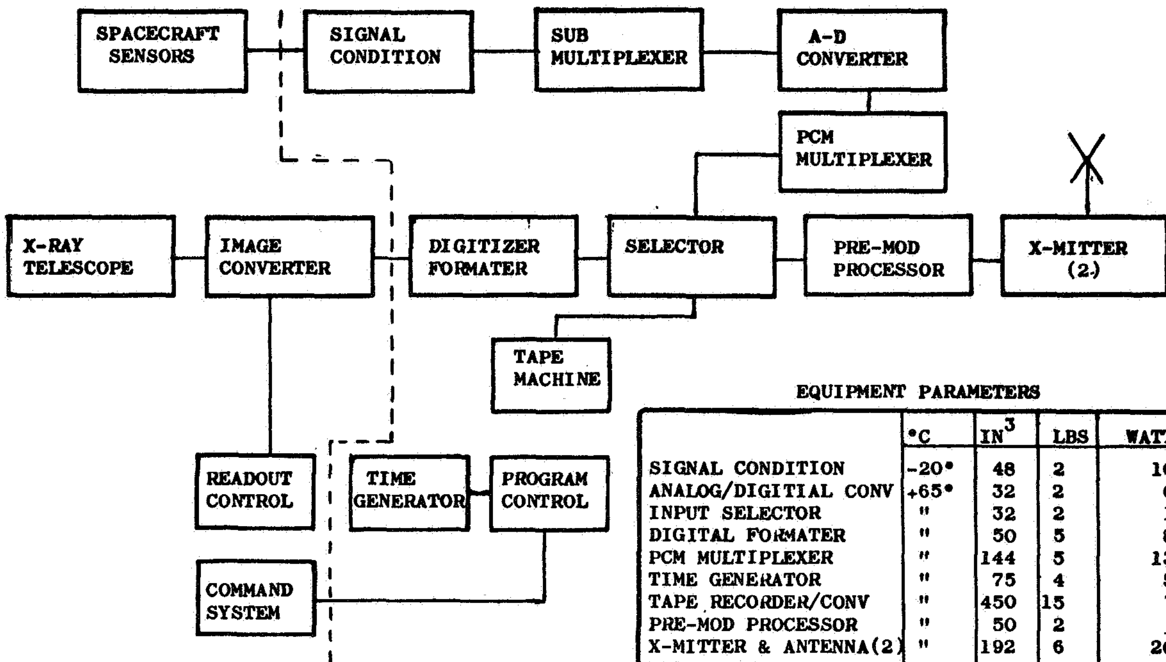
	IN^3	LBS	WATTS
INERTIAL UNIT		30	100
STAR TRACKER (3)	3000	30	45
SOLAR SENSOR	10	1	1
COMPUTER	700	35	75
SIGNAL PROCESSOR	700	18	10
PHASOLVER	210	8	2
SENSOR SIGNAL PROCESSOR(P&Y)	300	10	5
SENSOR SIGNAL PROCESSOR(ROLL)	180	6	3
MISC. ELECTRONICS	1340	53	24

Figure I -8. Navigation System

Data, TLM, and Communication System

The data handling system designed for the x-ray telescope is a conventional PCM telemetry system consisting basically of a signal conditioner, analog to digital converter, input selector, digital formatter, CM multiplexer, time generator, tape recorder/converter, pre-mod processor, transmitter and two antennas and associated multipliers, filters and selectors.

The tape recorder capacity required is approximately eight million bits, the maximum transmission rates to ground will be approximately 260 Kbps, a single machine can handle this load with a second as back-up. See Figure I -9 for system block diagram.



EQUIPMENT PARAMETERS

	°C	IN^3	LBS	WATTS
SIGNAL CONDITION	-20°	48	2	10
ANALOG/DIGITIAL CONV	+65°	32	2	6
INPUT SELECTOR	"	32	2	1
DIGITAL FORMATER	"	50	5	8
PCM MULTIPLEXER	"	144	5	13
TIME GENERATOR	"	75	4	5
TAPE RECORDER/CONV	"	450	15	7
PRE-MOD PROCESSOR	"	50	2	1.5
X-MITTER & ANTENNA(2)	"	192	6	20
MULTIPLIER	"	48	2.5	5
PRE-MODUL FILTERS	"	7	.5	1
PROGRAMMED SELECTOR	"	60	2	2
HARNESSES, BRKTS			10	

Figure I-9. Data, TLM and Communications

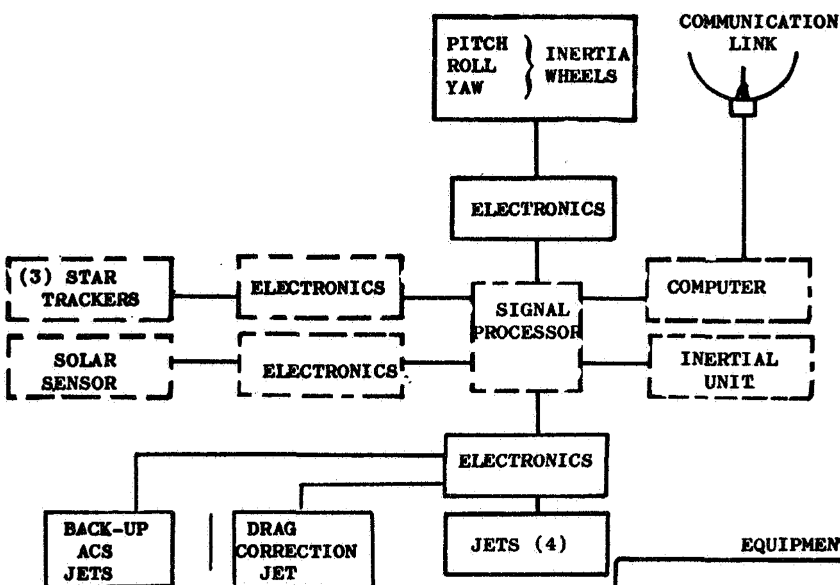
Attitude Control System

The telescope attitude control system consists of three independent attitude orientation systems and a drag correction system. The three orientation systems are the 12 jet hydrogen peroxide course stabilization system, the three axis inertial wheel precision pointing system and the emergency cold gas docking stabilization system.

The system is sized to maintain the required pointing accuracy of $\pm$ 0.5 arc min. and drift rates $\leq$1.0 arc sec/sec. for up to 90 min. (one orbit). The lock-on time can be extended to five hours with a total vehicle weight penalty of approximately 400 lb. Hydrogen peroxide was selected as the propellant on the basis of a two year mission without resupply.

The nominal 30 in. x-ray telescope configuration has 0.0403 and 0.00034 ft/lb. of gravity gradient and aerodynamic drag torques respectively. The inertia wheel system capacity is 10 ft/lb/sec. in roll and 84 ft/lb. in pitch and yaw.

The course orientation jet system utilizes 4 three-jet modules each with an individual propellant tank. Resupply of these modules is performed by EVA attaching replacement units to the expended unit. The precision inertia wheels are located in the image capsule assembly in close proximity to the star trackers to minimize reference coordinate errors resulting from structural distortions. The cold gas back-up docking system is provided to permit remote controlled astronaut stabilization of the telescope for docking. This emergency system is only required if both normal stabilization systems are malfunctioning or the control electronics fail. The reserve system has low propellant requirements permitting the use of the highly reliable nitrogen gas thrusters. The attitude control system weighs 575 lbs, occupies 5,400 cu. in. and requires 33 watts maximum power.



EQUIPMENT PARAMETERS

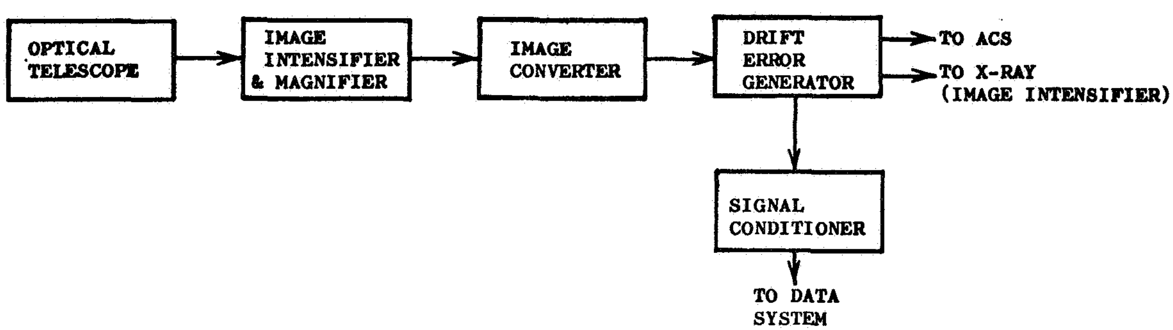
	IN	LBS	WATTS
ATTITUDE CONTROL JETS		328	
" " WHEELS	5300	185	33.5
DRAG CORRECTION JETS	32	32	
BACK-UP ACS JETS	30	30	
MISC ELECTRONICS *			

* Charged to Navigation System

Figure I-10. Attitude Control System

Optical Drift Error System

The drift error system permits the x-ray telescope to effectively lock-on optical sources in the general vicinity of the x-ray source being observed. The x-ray sources of interest to the proposed telescope mission do not have sufficient flux intensities to permit vehicle lock-on to the source being observed. It is necessary, therefore, to provide a guidance system which can lock-on to reference stars of adequate intensity. The vehicle star tracker would limit the long term image resolution to at least 10 arc sec.; this would be increased by the vehicle attitude control system. The optical drift error system consists of an optical telescope which is aligned with the primary x-ray mirror assembly. The optical telescope has a field of view of 2^{o} to assure that one or more stars of the tenth apparent magnitude will always be within the field. An image intensifier and vidicon record the moving reference star images, the star drift is translated into drift error signals which are transmitted to the x-ray image motion compensator. The image motion compensator will require a maximum travel of one arc min. in the Y and Z planes, the vehicle pointing accuracy. The drift error system permits extremely accurate imaging without requiring the vehicle to maintain precision attitude control. The drift system will not compensate for vehicle distortions or jitter between the lens assembly and the focal plane. The drift system block diagram and equipment parameters are shown in Figure I-11.



EQUIPMENT PARAMETERS

	IN^3	LBS	WATTS
OPTICAL TELESCOPE	13600	200	
IMAGE INTEN & MAG	250	20	10
IMAGE CONV (VIDICON)	300	20	10
DRIFT ERROR CONV	30	2	2
INTERFACE UNIT	32	2	2

Figure I-11. Optical Drift Correction System

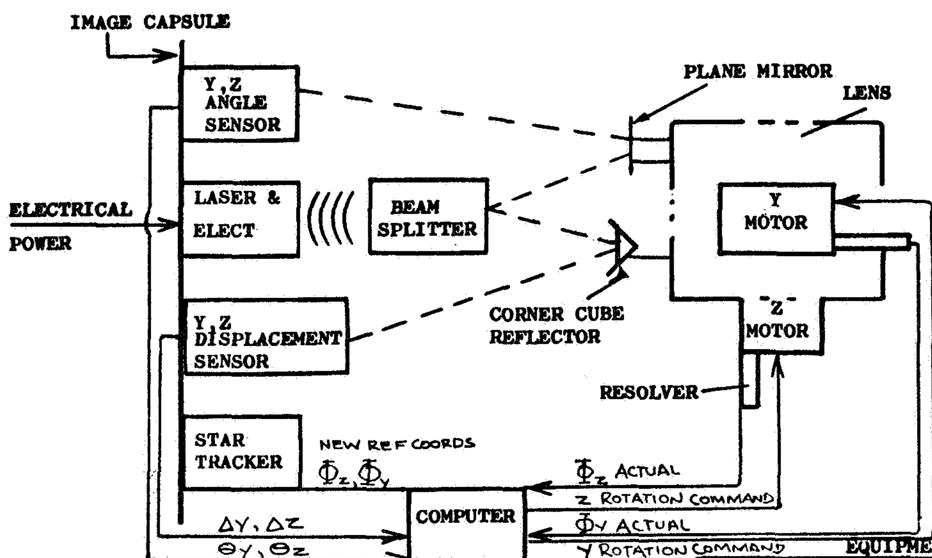
Lens Alignment

The lens alignment system compensates for the distortions of the primary truss structure.

The system consists of a gallium arsenide laser and sensor package mounted on the image capsule and a reflecting mirror assembly attached to the lens assembly.

A laser beam transmitted parallel to the optical axis is reflected by the plane and corner cube mirrors at the lens. The reflected beam impinges on the sensors and the beam location is registered. The plane mirror senses lens angle changes while the corner cube reflector senses lens displacement perpendicular to the optical axis.

As structural distortions occur the resulting changes to the optical axis are calculated and corrected for by rotating the two axis gimballed lenses. The optical axis changes are programmed into the navigation system to modify the star tracker reference axis. The alignment system block diagram and equipment parameters are shown in Figure I-12.



EQUIPMENT PARAMETERS

	IN^3	LBS	WATTS
LASER & ELEC	68	5	7
BEAM SPLITTER	8	1	
SENSOR	14	1	
SENSOR ELEC	96	5	2

Figure I-12. Lens Alignment System X-Ray

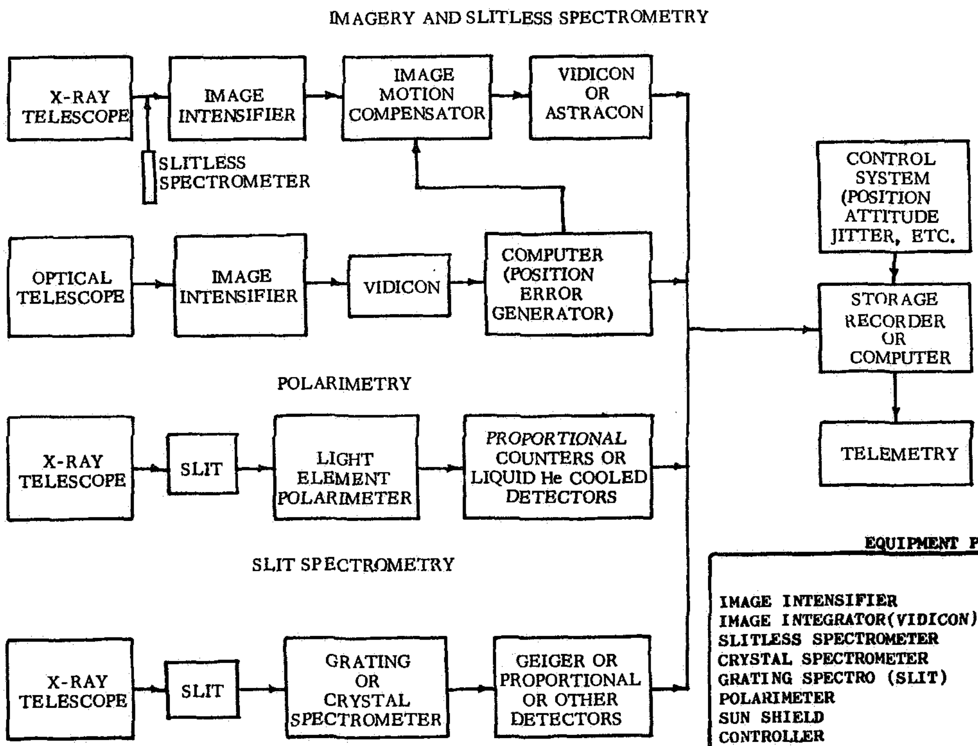
Telescope Instrumentation

The proposed telescope is capable of accomplishing a wide variety of soft x-ray observations. The following instruments were utilized during the preliminary design of the telescope. However, the telescope is not restricted to using these instruments alone.

There are three primary detection systems for use with the imaging x-ray telescope.

1. Imaging System - either film or electronic (with image motion compensation)
2. Spectrometer System - either a slitless spectrometer as discussed by A.S. & E. or a focusing crystal or concave diffraction grating type device.
3. Polarization System - a light element crystal scattering polarizer, or a Borrman effect crystal device.

Preliminary instrument parameters and an equipment block diagram is displayed in Figure I-13.



EQUIPMENT PARAMETERS

	IN^3	LBS	WATTS
IMAGE INTENSIFIER	2000	20	20
IMAGE INTEGRATOR(VIDICON)	2000	20	20
SLITLESS SPECTROMETER	-	30	-
CRYSTAL SPECTROMETER	20000	40	20
GRATING SPECTRO (SLIT)	29000	40	20
POLARIMETER	2000	50	20
SUN SHIELD	-	20	-
CONTROLLER	150	5	10
IMAGE MOTION COMPENSATOR	2000	20	20

Figure I-13. Telescope Instrumentation

SECTION III - ENGINEERING INFORMATION

Imagine Instruments

For effective imaging to two arc sec. resolution, all jitter and pointing stability total motion must be kept within a two arc sec. amplitude during that amount of time designated for each exposure. With a total jitter rate of two arc sec/sec. the image must be formed from information obtained once per sec. to achieve the desired resolution. Assuming a 1 m diameter telescope and the x-ray source intensity 10^{-4} Sco X-1, calculations show that at least 150 sec. are required to sensitize the best x-ray film Therefore, film requires a source 150 times stronger than Sco X-1 with a jitter rate or drift of two arc sec/sec. These calculations show the explicit need for image motion compensation with the pointing accuracy of the telescope of $\pm$ 0.5 arc min. Image motion compensation can be avoided only at the cost of decreasing the resolution requirements to that of the pointing requirement.

Image intensifiers allow some relaxing of this requirement, in that, under the same conditions, an identical image could be obtained in 60 sec. of time vs 150. Again using film at the rate of one frame per sec., the film usage rate becomes prohibitive.

Drs. Giacconi and Gursky suggested that an image motion compensator be introduced behind the image intensifier. This would electronically compensate for the movement of the telescope back and forth within the one arc min. pointing box, in order that the film or electronic imaging tube not be affected by any other motion than the allowed jitter. This device could be built deriving its signals from the optical telescope image. It is further suggested to completely depart from film, to using an electronic imaging tube as recording medium.

Thus, for imaging the recommendation is to use a system where the x-ray scanner would be either some type of vidicon tube, or possibly film plus a line scanner. The image intensifier - Image Motion compensator - Electronic Imaging Tube System is considered feasible before 1974. The size, weight and power requirements of the image intensifier are on the order of those of video system.

Spectrometers

Slitless Spectrometer

This is an A.S. & E. suggested device, consisting of a fine wire grid placed immediately behind the grazing incidence lens assembly. The grating itself blocks about 50 percent of the incident energy that has been collected by the lenses, and about 40 percent of that total power

Due to the resulting efficiency of 10 - 20 percent. Calculations shows that, for the telescope of 1 m diameter assumed above, the lock-on time for a 10Å entire spectrum with 0.025Å resolution would approach 10^4 sec. for a source the strength of about 10^{-4} that of Sco X-1. The sensor end would utilize either film or electronic imaging in conjunction with an image intensifier.

Focusing Crystal Spectrometer

This is a very high resolution device, and is used to study one spectral line at a time. Because of the requirement that only one line per crystal can be registered on the scanning equipment for each integration time, the available incoming energy is either split up on to several crystals, or studied one line at a time with one crystal; both alternatives imply long integration (lock-on) times. The entire instrument would be placed 1 - 2 ft. behind the focal point, where a collimating slit must be positioned. The energy is refracted from the crystal on to the single counters (photo muliplier tubes, e.g.) and different lines are obtained by rotating the crystal.

Slit Type Concave Grating Spectrometer

Compared to the Slitless Spectrometer this instrument has the advantages of better efficiency and higher dispersion. It would also be capable of providing spectra of extended sources, whereas the slitless spectrometer would only be able to handle point source spectra. However, the instrument would require a large distance (on the order of one meter) between the slit (located at the focal point) and the diffraction grating. It would also require a long bent film plate or image intensifier to register the spectrum.

Polarimeters

Polarimetry can be accomplished using a scattering crystal of a light element, the Borrmann Effect, or the Compton Effect.

In a scattering polarimeter, depending on a judicious scattering of the incoming photons in differing directions depending on their polarization, a light element crystal is used. This instrument has the advantage of being largely frequency independent, so that one crystal may suffice for the entire bandwidth under investigation.

One configuration that was suggested was the use of a solid or liquid hydrogen scattering crystal, cooled by liquid helium. The entire apparatus is placed behind a slit at the focal plane, and the crystal is surrounded by counters on all four lateral sides. These detectors could be liquid helium cooled germanium detectors, or proportional counters. The size of the crystal would be on the order of 30-40 cm long, wedge shaped, and perhaps 10 cm thick.

The Borrmann Effect depends on the selective transmission of energy along planar paths in selected crystals. The problem here is that each crystal is very selective as to wavelength, and two orthogonal crystals are required for each wavelength to ascertain the polarization components in two directions.

The Compton Effect utilizes a carbon block and would entail extremely long counting times.

Equipment List and Status

b. Required Equipment	c. State of Definition
1. Flight Article (FA)	Conceptual/Design
One complete flight article will be delivered.	
FA-1 Grazing Incidence Lens Capsule.	Conceptual/Design
FA-2 Lens Capsule Truss Assembly	Design
FA-3 Image Capsule	Conceptual/Design
FA-4 Image Capsule Truss Assembly	Design
FA-5 CSM Docking Drogue Assembly	Design
FA-6 Extension Carriage Assembly	Design
FA-7 Electronics Assembly	Design
2. Engineering Model (EM)	
Major assemblies are:	
EM-1 Lens Capsule Truss Structural Test Model	Breadboard
EM-2 Lens Capsule Weight and Balance Mockup	Design

b. Required Equipment (cont'd)	c. State of Definition (cont'd)
EM-3 Image Capsule Truss Structural Test	Breadboard
EM-4 Image Capsule Weight and Balance Mockup	Design
EM-5 Extension Carriage Structural Test Model	Design
EM-6 Extension Mechanism Weight and Balance Mockups	Breadboard
EM-7a, b, c, TT&C Weight and Balance Mockups	Design
EM-8a, b, c, d, ACS Weight and Balance Mockups	Design
EM-9a, b, c, d, Solar Panel Weight and Balance Mockups	Design
EM-10 CSM Docking Drogue Structural Simulator	Design
EM-11 LEM Adapter Structural Simulator	Breadboard
3. Engineering Subassembly Test Articles (SA)	
SA-1 Optical Telescope Assembly	Design
SA-2 Tracking Subsystem Electrical Test Model	Design
SA-3 Telemetry Subsystem Electrical Test Model	Design
SA-4 Command Subsystem Electrical Test Model	Design
SA-5 Solar Power Panel Electrical Test Model	Design
SA-6 Solar Power Panel Structural Test Model	Design
SA-7 Extension Drive and Drive Mechanism Electrical Test Model	Breadboard

b. Required Equipment (cont'd)	c. State of Definition (cont'd)
SA-8 Attitude Control Module - Air Bearing Test Model	Breadbroad
SA-9 Subassembly Test Harness (Quantity - As required)	Design
SA-10 CSM Control Panel Soft Mockup	Breadboard
SA-11 CSM Control Panel Electrical Test Model	Design
SA-12 LEM Adapter Electronic/ Electrical Interface Simulator	Design
4. Training Articles (TA)	
TA-1 Neutral Buoyancy Test Tank (30 ft. depth)	Construction Complete
TA-2 X-Ray Source Search Pattern Simulation Console	Breadboard
TA-3 Discrete X-Ray Source Observation Simulation Console	Breadboard
TA-4 Orbital Flight Simulator	Breadboard
5. Ground Support Equipment (GSE)	
GSE-1 Lens Capsule Handling Cart	Design
GSE-2 Image Capsule Handling Cart	Design
GSE-3 X-Ray Telescope Spacecraft Handling Cart	Design
GSE-4 Equipment Handling Carts (4)	Design
GSE-5 X-Ray Telescope Spacecraft Shipping Container	Design
GSE-6 Equipment Shipping Containers (4)	Design
GSE-7 Hoists and Slings (as required)	Design
GSE-8 Spacecraft Checkout Console	Design
GSE-9 ACS Fill and Drain Unit	Design
GSE-10 Launch Control Panel	Design

2. ENVELOPE

Geometrical sketches and dimensions are included in Figures I-14 through I-20 for the flight hardware assemblies.

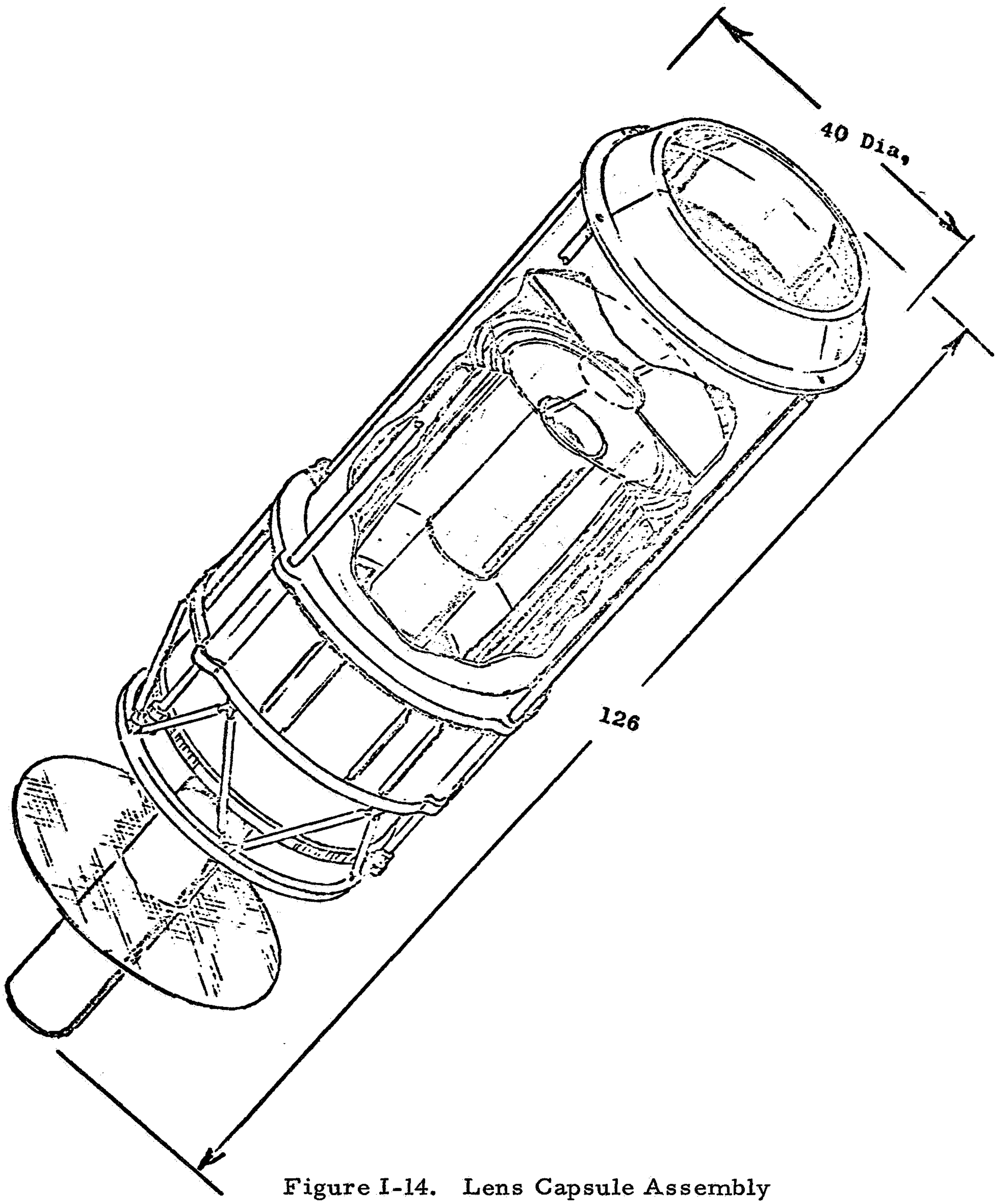


Figure I-14. Lens Capsule Assembly

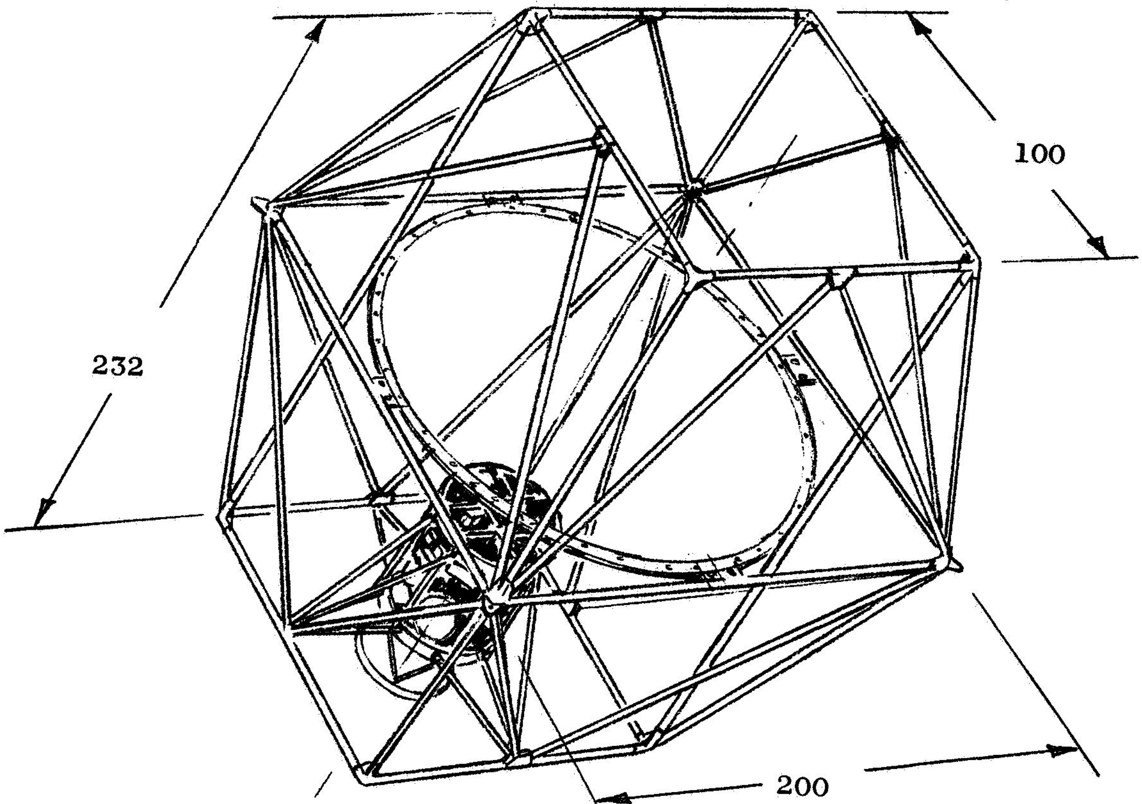


Figure I-15. Image Capsule Truss Assembly

Figure I-15 Image Capsule Truss Assembly

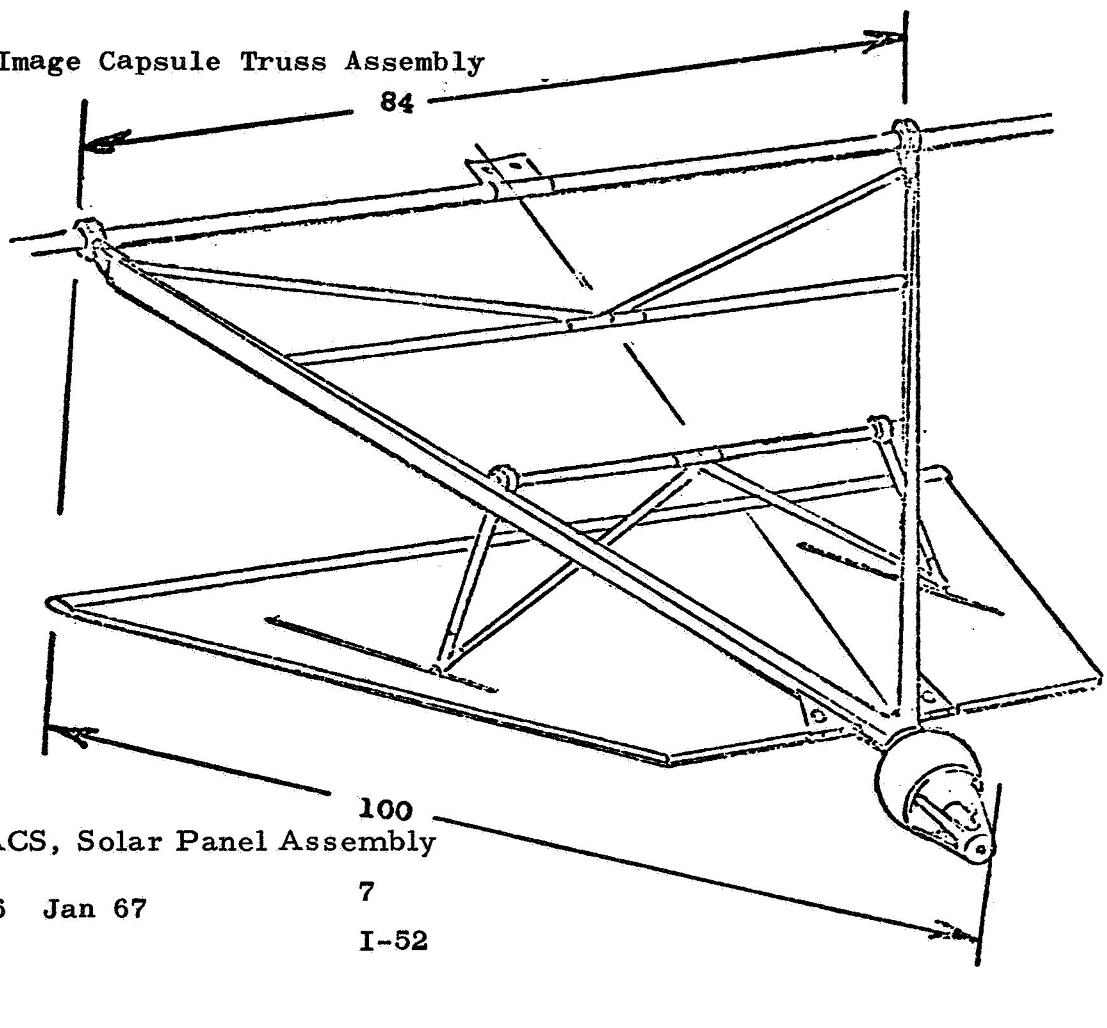


Figure I-16. ACS, Solar Panel Assembly

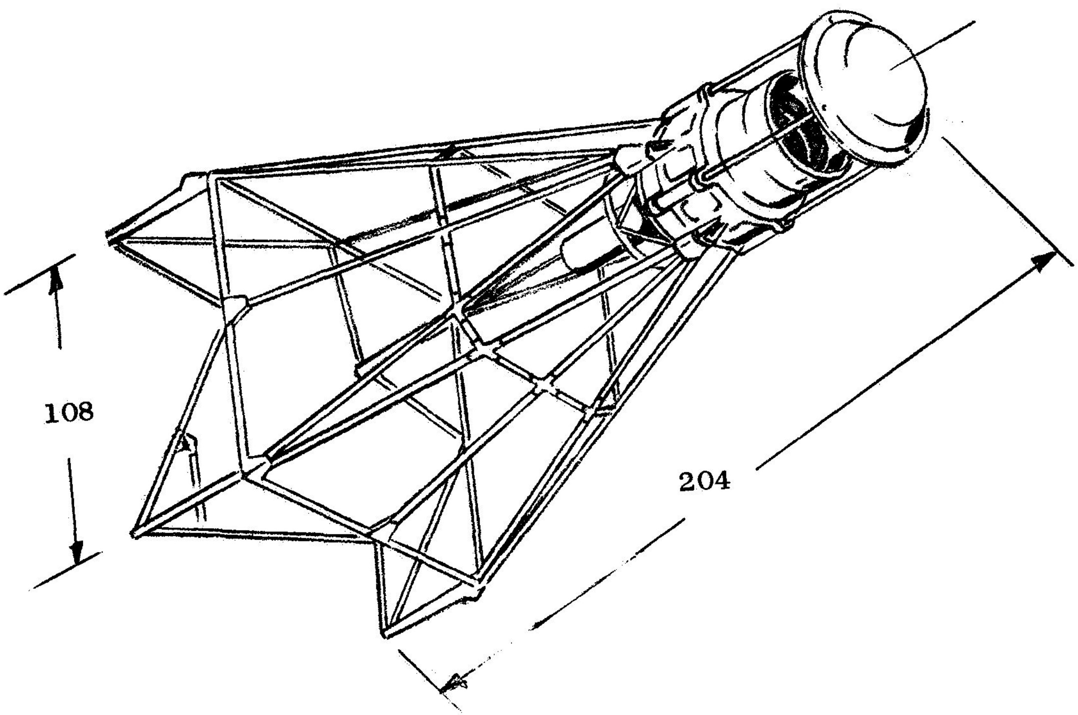


Figure I-17. Lens Capsule Truss Assembly

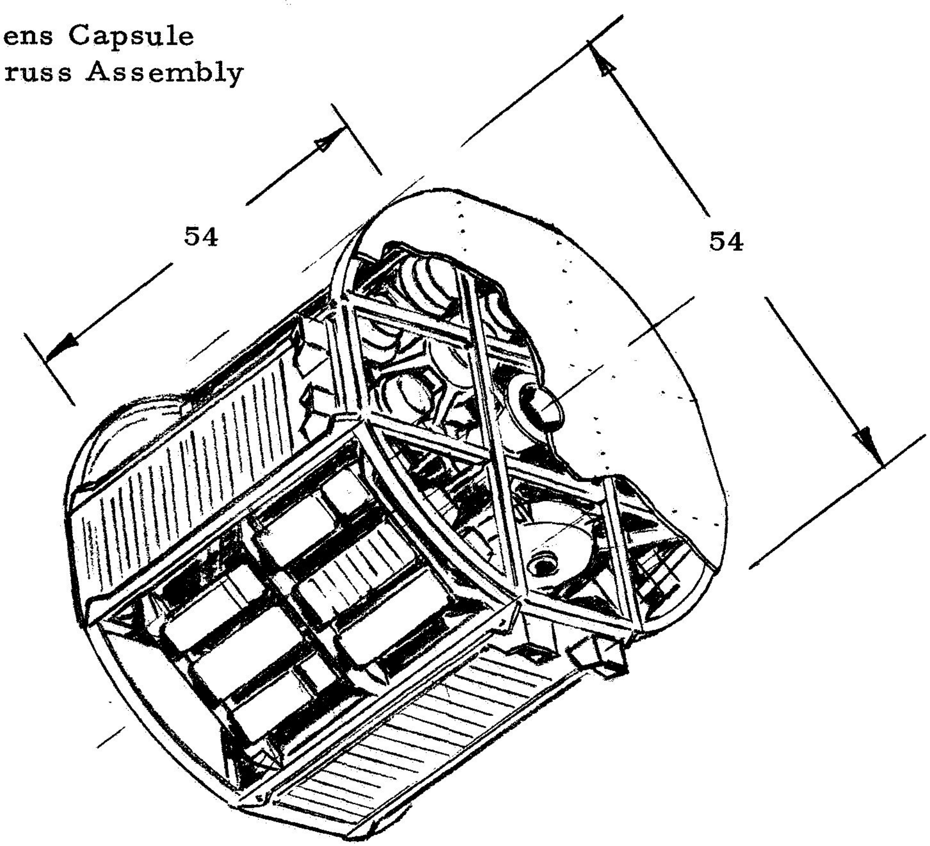


Figure I-18. Image Capsule Assembly

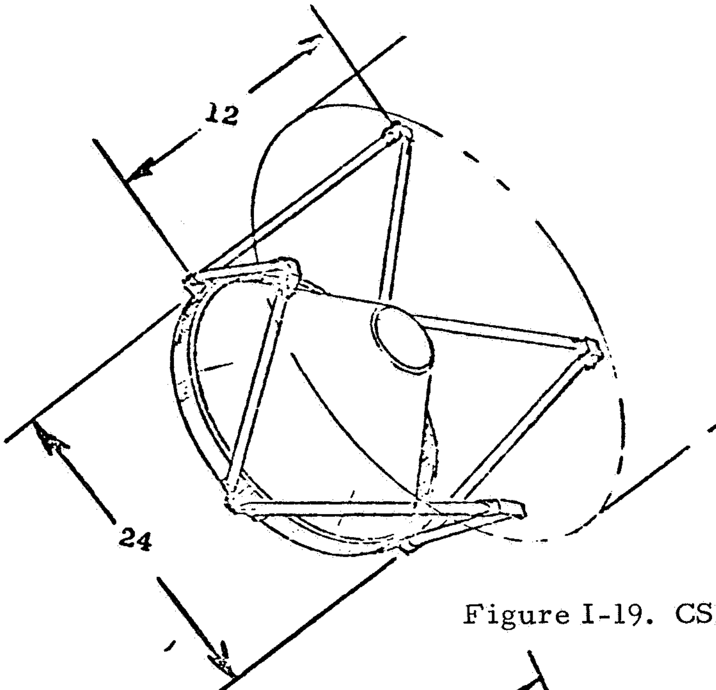


Figure I-19. CSM Docking Drogue

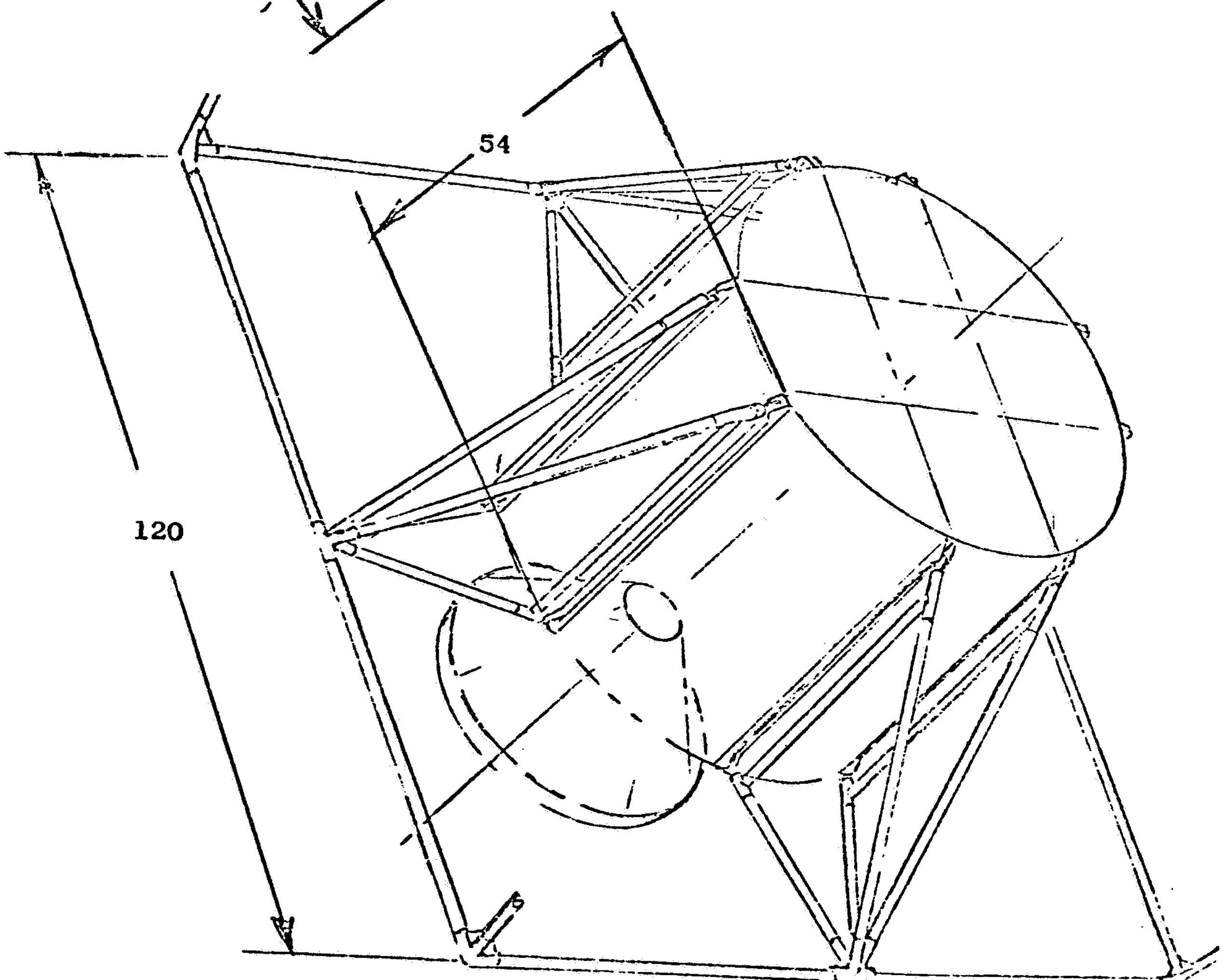


Figure I-20. Image Capsule Support Truss

3. WEIGHT AND SIZE

Equipment Item	Weight (lb)	Volume (cu ft) Stored Operation	Dimensions (in.)	Shape
1. Lens Capsule Assembly	493	92	40 x 40 x 126	Cylinder
2. Lens Capsule Truss Assy.	77	685	108 x 108 x 204	Pyramid
3. Image Capsule Assy.	1236	71	54 x 54 x 54	Cylinder
4. Image Capsule Assy.	192	2617	200 x 200 x 232	Back to Back Pyramid
5. ACS, Solar Panel Assy (4)	587	120	84 x 12 x 100	Trapezoid
6. CSM Docking Drogue	50	3	24 x 24 x 12	Conical
7. Image Capsule Support Truss	50	54	120 x 120 x 54	NA
TOTAL	2695	2620 *		

* Total Enclosed Volume of Packaged Configuration

4. POWER

The x-ray telescope power requirements are:

Average	260 watts
Maximum	467 watts

Component power is supplied by the x-ray telescope power system and does not represent a drain on the CSM system. Typical system power requirements are shown below:

System	Maximum Power
Navigation	226
Telescope Instrumentation	70
Data, TLM & Communications	59
Attitude Control	57
Optical Drift Correction	24
Command	12
Lens Alignment	9
	467

Note: All the above systems are contained within assembly number 3, the image capsule.

5. SPACECRAFT INTERFACE REQUIREMENTS

a. Desired Location

The x-ray telescope is designed to be launched on a manned Saturn I-B within the LEM adapter system, as shown in Figure I-21.

Figure I-21. Desired Experiment Location

b. Mounting Requirements

The x-ray telescope mounts within the LEM adapter system and attaches to the same four support points that the LEM would use. Interface requirements within this installation are the same as for LEM.

c. Subsystem Support Requirements

Checkout of the x-ray telescope subsystems, when mated to the launch vehicle, is accomplished both by landline and R-F data links, as follows:

Subsystem	Support	Method
Scientific Instruments	Status Monitoring	R-F Data Link
Attitude Control System	Status Monitoring	R-F Data Link
Telemetry System	Status Monitoring	R-F Data Link
Command/Control System	Status Monitoring	R-F Data Link
	Command Exercise	R-F Data Link & Landlines
Power System	Status Monitoring	R-F Data Link &
	Power Control	Landlines
	Battery Charge	Landlines
	On/Off Control	Landlines
	Launch Control Access	Landlines

d. Special Mechanical Control

Injection of the x-ray telescope into its final orbit requires the use of the CSM as follows:

(1) The CSM is docked to the telescope and extracts if from the LEM adapter system.
(2) The CSM provides the initial orientation and stabilization to the telescope.
(3) The CSM provides the life support requirements of the orbital crew during initial manned operations and during subsequent maintenance and repair activities.

(A LEM docking drogue is provided on the Image Capsule of the telescope for mating with the CSM.)

6. ENVIRONMENTAL CONSTRAINTS

a. Tolerance

Constraint \ Assembly	(1)	(2)	(3)	(4)	(5)	(6)	(7)
Thermal							
Stored	(8)	(8)	(8)	(8)	(8)	(8)	(8)
Operational	(9)	(9)	(9)	(9)	(9)	(9)	(9)
Atmospheric Pressure (Stored)	(10)	(10)	(10)	(10)	(10)	(10)	(10)
Relative Humidity (Stored)	50%	50%	50%	50%	50%	50%	50%
Air Movement Rate (Stored)	(11)	(11)	(11)	(11)	(11)	(11)	(11)
Atmospheric Composition (Stored)	(11)	(11)	(11)	(11)	(11)	(11)	(11)
Contaminants (Stored)	(11)	(11)	(11)	(11)	(11)	(11)	(11)
Acceleration (Storage)	(12)	(12)	(12)	(12)	(12)	(12)	(12)
Positive							
Negative							
Transverse							
Acceleration (Operational)	(13)	(13)	(13)	(13)	(13)	(13)	(13)
Positive							
Negative							
Transverse							
Vibration (Storage)	(12)	(12)	(12)	(12)	(12)	(12)	(12)
Random							
Sinusoidal							
Vibration (Operational)	(13)	(13)	(13)	(13)	(13)	(13)	(13)
Random							
Sinusoidal							
Acoustic Noise (Storage)	(13)	(13)	(13)	(13)	(13)	(13)	(13)
Light Tolerance	(9)	(9)	(9)	(9)	(9)	(9)	(9)
Intensity							
Wavelength							
Radiation Tolerance	(9)	(9)	(9)	(9)	(9)	(9)	(9)
RFI	(14)	(14)	(14)	(14)	(14)	(14)	(14)
EMI	(14)	(14)	(14)	(14)	(14)	(14)	(14)

NOTES: Reference Tolerance Table

(1) Lens capsule assembly.

(2) Lens capsule truss assembly.

(3) Image capsule assembly.

(4) Image capsule truss assembly.

(5) ACS, Solar panel assembly.

(6) CSM docking drogue.

(7) Image capsule support truss.

(8) Normal storage temperature range -65°F to +160°F, except for the lens assembly which is maintained at room temperature.

(9) The x-ray telescope is an independent vehicle designed to operate in the space environment, including thermal, light intensity, radiation and vacuum. After separation from the launch vehicle it imposes no constraints on either the launch vehicle or on the CSM.

(10) The assemblies are not sensitive to changes in atmospheric pressure.

(11) No unusual constraints.

(12) Acceleration during shipping and handling is typical for space hardware.

(13) Operational acceleration, vibration and noise conditions are based on Saturn I-B launch vehicle conditions.

(14) In accordance with MIL-STD-826.

b. Interference

The x-ray telescope is an independent vehicle and is not expected to cause interference with other vehicles in orbit. During launch, potential interference due to RFI and/or EMI is limited to the levels established in MIL-STD-826.

7. DATA MEASUREMENT REQUIREMENTS

With the exception of observational and biomedical data recorded by the astronauts in the CSM, there is no data interface with the CSM. The on-board telemetry and data processing systems are compatible with the requirements of the MSFN and DSIF.

SECTION IV - OPERATIONAL REQUIREMENTS

1. SPACECRAFT ORIENTATION REQUIREMENTS

There are two primary spacecraft involved: the CSM and the x-ray telescope. The interactions of these two vehicles, and the astronaut crew are critical to the successful conduct of the experiment.

a. Maneuvers

Deployment of the x-ray telescope by the CSM is shown in Figure I-22.

b. Type of Orbit

The orbit most suitable for the objectives of the x-ray telescope is a circular, low inclination earth orbit at 260 n. mi. or lower.

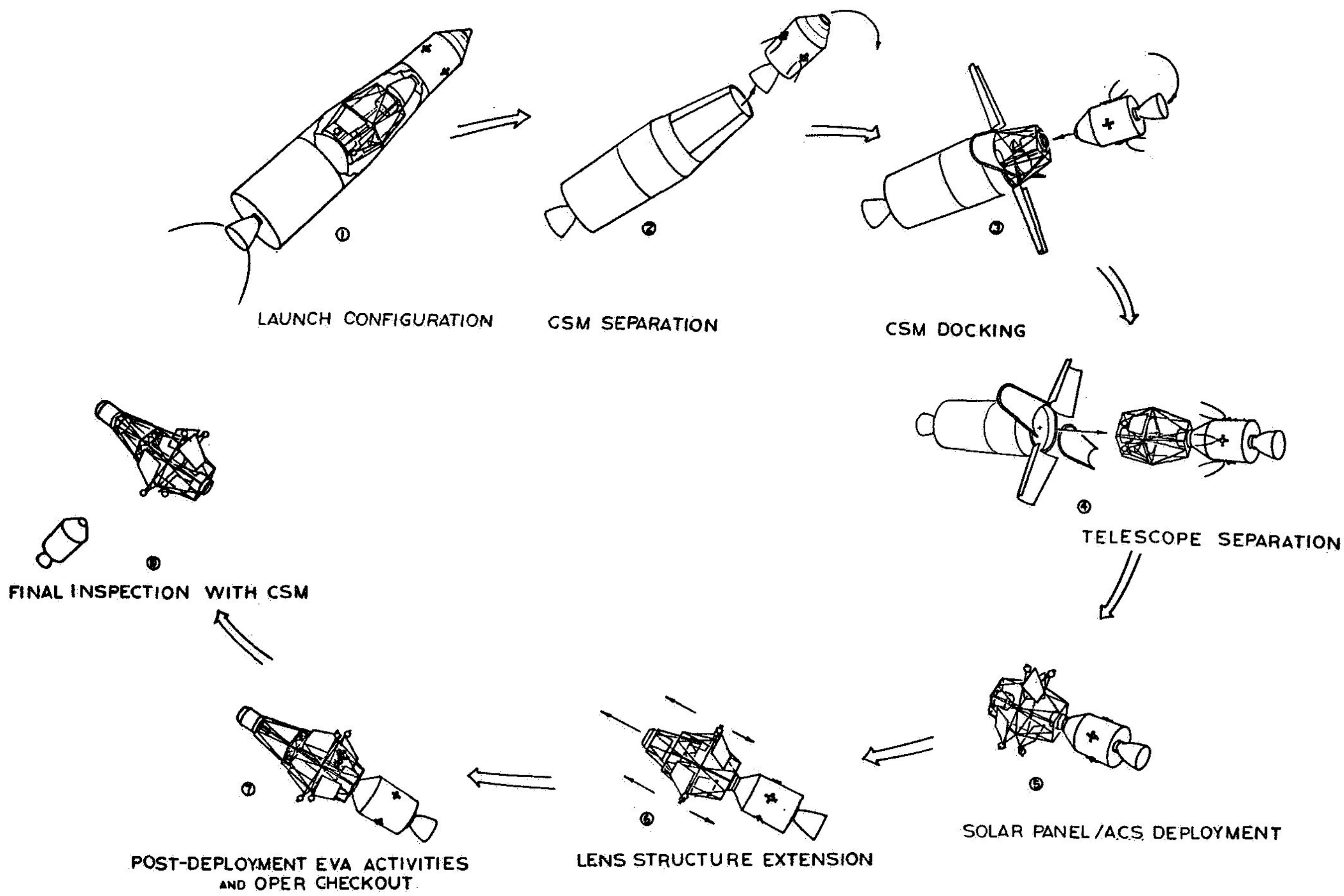


Figure I-22. X-Ray Telescope Orbital Deployment

c. Orbit Parameters

Altitude	260 n. mi.
Inclination	28.5^{o}
Period	94 min.
Sunlit Time	65 min. /orbit
Dark Time	29 min. /orbit
Orbital Lifetime	4 years

d. Lighting Constraints - None

e. Launch Time Contraints - None

f. Number of Measurements Required - to be determined

g. Time Per Measurement

X-ray telescope observation measurement periods are dependent on source intensity; typical ranges are:

(1) Lock-on to target	-	10 - 20 min.
(2) Imaging	-	1 - 60 min.
(3) Spectrometry		
Per spectral line	-	10 - 150 min. (as many lines as possible)
Entire spectrum	-	60 - 1000 min. (slitless spectrometer)
(4) Polarimetry	-	5 - 60 min.

Selected measurements are repeated for each x-ray target throughout the mission lifetime.

h. Orbital Location

Orbital location is not critical to the taking of data. Data is accumulated in a tape recorder and dumped when the spacecraft is in RF contact with appropriate ground station.

i. Spacecraft Pointing Accuracy

Telescope pointing requirements are within an accuracy of $\pm$ 0.5 arc min. deviation from any desired celestial coordinate for 90 min. periods. (360 min. lock-on capability will require a 400 lb. weight increase.) The telescope is not docked to the command module during this time.

j. Allowable Spacecraft Rate

The maximum allowable drift rate during measurements is 1 arc sec. /sec.; image motion compensation of the x-ray energy during measurements is used to achieve $\leqq$ 1 arc sec. /movement on the recording equipment.

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2. ASTRONAUT TRAINING

Refresher Course in X-Ray Astronomy Theory — 80 Hours

Equipment Usage — 80 Hours

Malfunction Detection, Analysis & Repair Techniques — 120 Hours

Safety Procedures — 40 Hours

Task Simulation

1. X-Ray Astronomy Tasks:
   a. Alignment
   b. Calibration
   c. Data Analysis

2. EVA Structural Tasks:
   a. Detection and repair of malfunction in deployment
   b. Electrical connections
   c. Installation and change of components
   d. Inspection
   e. Maintenance
   f. Tool Handling

240 Hours

Mission Simulation and Familiarization

Sequential steps in Mission Procedure for deployment and initial operation of the x-ray telescope.

120 Hours

Total time required: 680 hours.

At least 3 astronauts are trained at the same time.

Preferable skills: Electrical Engineering

Figure I-23. Summary of Astronaut Training Program

3. ASTRONAUT PARTICIPATION PLAN

The scheduled requirements of the astronaut during the experiment are as follows:

a. Initial Deployment

(1) Predeployment status check (IVA).

(2) Initiate and control deployment of solar cell frames and ACS booms and check lock indicators (IVA).

(3) Test solar panel and ACS operation (IVA).

(4) Initiate and control extension of lens capsule (IVA) (manned observation of the extension can assure safe deployment by stopping operation if binding occurs -- then requires EVA to correct malfunction).

(5) Separate lens capsule from launch support ring (IVA) (the optimum focal length adjustment may require the CSM crew to monitor the image screen with a repeater scope, while setting the lens focal length within the $\pm$ 5 in. adjustment).

(6) Subsequent to proper deployment - when extension indicates locked (8 places), release system cable tension electro-pyrotechnic (4 places) IVA.

(7) Check lens shade tubes locked (4 places) IVA.

(8) Egress CSM, locomote to image capsule, open all four access doors, remove launch supports on turret and inspect for wire bundle interferences, etc. Free wheel turret by hand full travel (270^{o}) to verify clearances, energize master turret switch (EVA).

(9) Cycle turret with local control, switch to auto control, and observe ground control operation (EVA).

b. Initial Operational Checkout

(1) Actuate spectrometer grating and observe, slew lens gimbal motors and observe correction made by lens alignment system - check visual alignment indicators - return to CSM (EVA).

(2) EVA

(a) Installation of equipment in image capsule. The turret is designed for seven stations. Three are occupied by primary packages A, B and C, capable of four separate observations. Two more stations contain duplications of packages A and C. Two stations are therefore available for addition on resupply. Add-on is preferred to replacement; however, all packages are removable by EVA astronaut and will fit into CM for earth return, if desired. For failure analysis, etc., sufficient volume and strength exists on the structure to stow expended equipment.

(b) The ACS and solar panels are replaceable by the astronaut. Each ACS unit contains jets, valves and tanks, has a single mechanical attachment collar and an electrical connector. The solar panels are flexible roll-out type contained in a cylindrical dispenser. The astronaut secures himself to the ACS deployment frame to replace the ACS unit or solar panel.

(c) Battery replacement will likely occur coincident with solar panel replacement. These batteries are located in a sector of the image capsule.

d. Refurbishment

(1) Refurbishment consists of replacement of certain items as per the normal resupply except additional equipment will be replaced to prolong useful life of the facility. Also, new and probably different types of mission equipment will be added to broaden the scope of the facility investigations. The total items to be replaced or the specific ones are not completely known. Those identified here can be considered typical. EVA will be required extensively to perform some of these tasks:

(a) Image capusle mission equipment.

(b) Image capsule operational systems (momentum wheels, turret motor/gear box, laser receiver system, electronic modules, batteries, star tracker, etc.).

(c) Lens capsule operational systems (focal length drive motors, gimbal motors, spectrometer grating and/or motors, lens shade patching, telescope electronics package laser beam generator package, filter discs, etc.).

(d) ACS units.

(e) Solar panel units.

e. Crew Time Line Analysis

The crew tasks for the deployment and operational checkout phase and resupply phase shown on the following time line analysis charts, Table I-1 and Table I-2.

The refurbishment phase tasks are dependent to a large extent on the type and reliability of the scientific equipment. This equipment is not sufficiently defined to permit development of a realistic refurbishment plan and therefore, no detailed time line analyses were prepared. It is estimated that the orbital refurbishment task will require at least double the time and effort required for the resupply missions.

Table I-1. **TIME LINE ANALYSIS**

X-RAY TELESCOPE INITIAL DEPLOYMENT TO OPERATIONAL STATUS

Final orbit altitude achieved (260 n.mi.). Crew is in CM in spacesuits. CSM has transpositioned, SLA panels have deployed and CSM is docked onto Image Capsule drogue. This is point of beginning.

EVENT NO.	EVENT AND TASKS	EVENT TIME (MIN)	ACCUM TIME (MIN)	EVA ACCUM TIME (MIN)
1	Time starts at point of beginning	0.0	0.0	
2	Crewman equalizes pressure in drogue. Removes forward hatch and stows. Latches the 12 manual latches. Connects electrical umbilical to X-Ray Telescope system	30.0	30.0	
3	Crew initiates separation of pyro bolts at 4 SLA hard points. CSM maneuvered to extract telescope from SLA.	5.0	35.0	
4	CSM maneuvered to correct orbit parameters and gain separation distance from S-IVB.	30.0	65.0	
5	Orbit circularized (assume occurs at half orbit period). This period is also used to conduct spacecraft housekeeping and biomedical checks. Shirtsleeve atmosphere.	45.0	105.0	
6	All recording systems checked relative to strain gauge data, temperature, ICSD, etc.	15.0	120.0	
7	Status for go is established through critique with ground control.	5.0	125.0	
8	Crew initiates signal to release #1 ACS/solar frame. Pyro bolt release. Elect indicator.	0.1	125.1	
9	Crew operates momentary switch to deploy. Other crew members observe deployment via windows and/or periscope. Frame rotates 103° and locks. Solar panel is automatically unlocked with extension of frame and is automatically signaled to extend within last quarter of frame travel. Solar panel extends and locks. Frame extension time 25 sec and panel extension time 10 sec with a 5 sec overlap. Total = 30 sec. Switches indicate lock-up of both.	0.5	125.6	
10	Crew observes deployment. Verifies deployment in conjunc- with lights. Decision to proceed.	0.5	126.1	

Table I-1. Time Line Analysis (Cont'd)

EVENT NO.	EVENT AND TASKS	EVENT TIME (MIN)	ACCUM TIME (MIN)	EVA ACCUM TIME (MIN)
11	#2 ACS/solar frame deployed, observed	1.0	127.1	
12	#3 ACS/solar frame deployed, observed	1.0	128.1	
13	#4 ACS/solar frame deployed, observed	1.0	129.1	
14	Crew critique. Re-check of all systems. Decision to proceed.	5.0	134.1	
15	Signal energizing X-Ray Telescope power generating system. Output of each solar panel monitored. Electrical characteristics at buses monitored.	15.0	149.1	
16.	Individual signals to fire each ACS jet. Verification that each operates, (CSM readout). Note: Attitude control is still maintained by CSM. Telescope ACS system is <u>not</u> fully activated.	15.0	164.1	
17	Critique by crew. Ground critique. Decision making. Go ahead.	5.0	169.1	
18	Signal to release lens lateral restraints. Pyro tape cutters sever tapes (or cables). Tapes retract into cases. Indicator lights verify completion.	0.4	169.5	
19	Crew initiates signal to release lens capsule truss carriage hold-down bots. Pyro release. Signal indication of release.	0.5	170.0	
20	Crew initiates signal to truss extension motor system. Run-around cable system extends truss. Lens shade is extended as part of cycle.	5.0	175.0	
21	Two-stage locks engage at end of stroke. Second stage pyro locks automatically initiated; extension system terminated. Electrical indication of lock..	0.1	175.1	
22	Crew initiates signal to release cable tension by retraction of pulley. Pyro or solenoid actuation. Strain gauges on tracks indicate release.	0.5	175.6	
23	Crew initiates signal to energize lens capsule separation bolts. Indicator lights show release.	0.4	176.0	
24	Crew initiates signal to advance lens capsule to mid-position. Capsule travels to position. Encoder relays position information.	1.0	177.0	
25	Crew initiates individual override signals to both axis of gimbals for trial run. Readout system also checked for data output.	5.0	182.0	
26	Crew critique. Rest period. Meal.	90.0	272.0	
27	Crew dons spacesuits, suit checkout, PLSS check, etc.	120.0	392.0	
28	Depressurize CM	4.0	396.0	

Table I-1. Time Line Analysis (Cont'd)

EVENT NO.	EVENT AND TASKS	EVENT TIME (MIN)	ACCUM TIME (MIN)	EVA ACCUM TIME (MIN)
29	One crew member opens hatch and exits CM (Hatch left open)	30.0	426.0	34.0
30	Crew member travels to image capsule (tether), Bay #1	5.0	431.0	39.0
31	EV astronaut engages feet in door stirrups and engages waist restraint onto door.	4.0	435.0	43.0
32	Astronaut plugs portable power tool into door receptacle and retracts door. (Approximately 3/4 open)	1.0	436.0	44.0
33	Rest period. Flood lights in capsule on.	2.0	438.0	46.0
34	EV astronaut actuates switch on interior panel energizing the laser alignment system. Beam impinges on target. A search mode is assumed to obtain target. Gimbal motors can be given override commands by astronaut. Astronaut must detach himself from door to make visual check on gross miss of target.	10.0	448.0	56.0
35	Rest period	2.0	450.0	58.0
37	Completion of laser alignment. Ground check.	5.0	470.0	63.0
38	EV astronaut checks for clearances around turret. All harnesses checked and any losse items secured.	2.0	472.0	65.0
39	EV astronaut disengages launch restraint pins at top of turret. Duplex thrust bearing glange cinched up. Drive motor clutch is checked for disengagement. (Actually disengaged prior to launch.)	10.0	482.0	75.0
41	Rest period	2.0	484.0	77.0
42	EV astronaut slowly hand rotates free-wheeling turret. Turret elec power is off. All scientific packages checked for securing to turret and wire harness receptacles engaged.	5.0	489.0	82.0
43	EV astronaut engages drive clutch. Switches on turret power and rotates turret through all seven stations. This is override control. Position indicators note turret position. Scientific instruments are "off". Limit switches prevent over-travel.	5.0	489.0	87.0
44	Turret rotation control switched to ground control and cycling repeated. EV astronaut stands by as observer. Turret left at Station #1 (image signature). Rest for EV astronaut.	7.0	496.0	94.0
45	EV astrnaut actuates power tool and closes door.	1.0	497.0	99.0
46	Astronaut disengages from door and returns to CM.	9.0	506.0	108.0
47	Hatch is closed. CM pressurized.	30.0	536.0	138.0

Table I-1. Time Line Analysis (Cont'd)

EVENT NO.	EVENT AND TASKS	EVENT TIME (MIN)	ACCUM TIME (MIN)	EVA ACCUM TIME (MIN)
48	Astronauts doff space suits. PLSS regenerates, etc.	60.0	596.0	
49	Repressurize CM	5.0	601.0	
50	Complete check of all systems, strain gauges, temperature sensors, etc. Ground to spacecraft critique.	60.0	661.0	
51	CSM ACS switched "off". X-Ray ACS system "on". Inertial platform "on".	1.0	662.0	
52	Attitude control system check.	15.0	677.0	
53	Sleep, personal time, hygiene, awake, eat.	480.0	1157.0	
54	All systems check especially CM housekeeping. Ground to spacecraft critique.	60.0	1217.0	
55	Attitude of spacecraft altered relative to sun and laser alignment system monitored to reveal compensation capability. Repeated minimum 3 times to gain assurance of performance.	45.0	1262.0	
56	All X-ray systems energized, CSM ACS and navigation "off". Telescope inertial platform "on". Scientific equipment "on".	5.0	1267.0	
57	Ground control sends up-date information to navigation system and commands pointing to particular sky coordinates. (Prime X-ray reference source.) Telescope slows and locks on.	5.0	1272.0	
58	Optical telescope locks on guide star. Verified at ground station.	1.0	1273.0	
59	Scientific instrument package at Station #1 energized by ground control. Held on target for approximately 24 min. Verifies instrument is receiving, IMC operating, control system compensation. (Image signature) Pick-up and communication time.	30.0	1303.0	
60	Station #3 commanded into receive position. Held 24 min and pick-up and communications time.	30.0	1333.0	
61	Auxiliary tasks, housekeeping, systems checks until target is again available.	30.0	1363.0	
62	Station #4 commanded into receive position. Held 24 min and communications and pick-up time.	30.0	1393.0	
63	Station #5 commanded into receive position. Held 24 min and communications and pick-up time.	30.0	1423.0	
64	Auxiliary tasks, housekeeping, systems check until target is again available. Eat.	30.0	1453.0	
65	Station #7 commanded into receive position. Held 24 min and communication and pick-up time.	30.0	1483.0	

Table I-1. Time Line Analysis (Cont'd)

EVENT NO.	EVENT AND TASKS	EVENT TIME (MIN)	ACCUM TIME (MIN)	EVA ACCUM TIME (MIN)
66	Check of all systems. Prepare for separation. Telescope stabilized by reference to inertial platform. All systems check-off.	30.0	1513.0	
67	Space suits donned. Suit systems checked.	120.0	1633.0	
68	Docking manual latches disengaged. Pressure hatch installed.	60.0	1693.0	
69	CSM undocks from x-ray telescope. CSM is flown around telescope. Movies, stills, close observation. Formation at approximately 200 feet distance.	90.0	1783.0	
70	Rest, eat, exercise.	60.0	1843.0	
71	Ground control re-energizes scientific equipment. Reference x-ray source picked up. Instruments cycled through operational check similar to before except all accomplished in 2 orbits. Turret final cycle is at Station #1. CSM flies formation during this check.	180.0	1923.0	
72	Telescope commanded to new source.	5.0	1928.0	
73	CSM flys formation for 8 orbits (12 hours). This completes normal preparation to operational status.	720.0	2648.0	

Total time to operational status: 1 day, 20 hrs, 8 min

Accumulated EVA time: 2 hrs, 18 min

Table I-2. Time Line Analysis - Resupply

Rendezvous with telescope has been achieved and CSM has docked onto the x-ray telescope image capsule drogue. This is point of beginning. Following EVA tasks include approximately 2 min. rest for each 10 min. of moderate to strenuous effort.

EVENT NO.	EVENT AND TASKS	EVENT TIME (MIN)	ACCUM TIME (MIN)	EVA ACCUM TIME (MIN)
1	Point of beginning. Crew in spacesuits.	0.0	0.0	
2	Crewman equalizes pressure in drogue. Removes forward hatch and stows. Latches the 12 manual latches. Connects electrical umbilical to x-ray telescope system. Telescope systems de-energized.	34.0	34.0	
3	CM depressurized. Hatch opened. Tether rigged, etc.	64.0	98.0	
4	Astronaut "B" exits CM. Maneuvers to SM. Erects conveyor line and transfers units to main truss center frame.	46.0	114.0	46.0
5	Retraction and installation of #1 ACS unit.	34.0	178.0	80.0
6	Retraction and installation of #2 ACS unit. Return to CM.	34.0	212.0	114.0
7	CM hatch closed. CM pressurized.	35.0	247.0	
8	Space suits doffed. PLSS regenerated. All systems check, biomed checks. Crew and ground critique.	60.0	307.0	
9	Eat, personal hygiene, housekeeping, biomed.	90.0	397.0	
10	Space suits donned. Suit and PLSS check-out.	120.0	517.0	
11	CM depressurized. EVA hatch opened. Tether rigged, etc.	64.0	581.0	
12	Astronaut "C" exits. Maneuvers to SM. ACS units #3 and 4 attached to conveyor and conveyed to truss center and secured.	42.0	623.0	156.0
13	Retraction and installation of #3 ACS unit.	34.0	657.0	190.0
14	Retraction and installation of #4 ACS unit. Astronaut "C" returns to SM.	39.0	696.0	229.0

Table I-2. Time Line Analysis - Resupply (Cont'd)

Event No.	EVENT AND TASKS	EVENT TIME (MIN)	ACCUM TIME (MIN)	EVA ACCUM TIME (MIN)
15	Astronaut "C" attaches #1 and #2 solar panel units to conveyor. Returns to CM.	32.0	728.0	261.0
16	Hatch closed. CM pressurized.	35.0	763.0	
17	Space suits doffed. PLSS regenerated. All systems check, biomed checks. Crew and ground critique.	60.0	823.0	
18	Eat, personal hygiene, housekeeping.	60.0	883.0	
19	Sleep.	480.0	1363.0	
20	Eat, personal hygiene, housekeeping, biomed checks.	60.0	1423.0	
21	Space suits donned. Suit and PLSS check-out.	120.0	1543.0	
22	CM depressurized. EVA hatch opened. Tether rigged, etc.	64.0	1607.0	
23	Astronaut "A" exits CM. Maneuvers to #1 solar panel area. Installs #1 solar panel.	50.0	1657.0	311.0
24	Extension of #1 ACS/solar panel frame.	12.0	1669.0	323.0
25	Installation of #2 solar panel.	50.0	1719.0	373.0
26	Extension of #2 ACS/solar panel frame. Returns to CM.	28.0	1747.0	401.0
27	CM hatch closed. CM pressurized.	35.0	1782.0	
28	Space suits doffed. PLSS regenerated. All systems check, biomed checks. Crew and ground critique.	60.0	1842.0	
29	Eat, personal hygiene, housekeeping biomed.	90.0	1932.0	
30	Space suits donned. Suit and PLSS check-out.	120.0	2052.0	
31	CM depressurized. EVA hatch opened. Tether rigged.	64.0	2116.0	
32	Astronaut "B" exits and maneuvers to SM, removes and attaches solar panels #3 and #4 and conveys to center of main truss.	38.0	2154.0	439.0
33	Installation of solar panel #3.	50.0	2204.0	489.0
34	Extension of #3 ACS/solar panel frame.	12.0	2216.0	501.0
35	Installation of solar panel #4.	50.0	2266.0	551.0
36	Extension of #4 ACS/solar panel frame. Astronaut "B" returns to CM.	18.0	2284.0	569.0

Table I-2. Time Line Analysis - Resupply (Cont'd)

Event No.	EVENT AND TASKS	EVENT TIME (MIN)	ACCUM TIME (MIN)	EVA ACCUM TIME (MIN)
37	CM side hatch closed. CM pressurized.	35.0	2319.0	
38	Space suits doffed. PLSS regenerated. All systems check, biomed checks. Crew and ground critique.	60.0	2379.0	
39	Eat, personal hygiene, housekeeping, biomed.	60.0	2439.0	
40	Operational check-out of power system.	30.0	2469.0	
41	Operational check-out of ACS system. Telescope ACS "on", CM ACS "off".	30.0	2499.0	
42	Telescope ACS "off". CM ACS "on".	1.0	2500.0	
43	Sleep.	480.0	2980.0	
44	Eat, personal hygiene, housekeeping, biomed.	60.0	3040.0	
45	Space suits donned. Suit and PLSS check-out.	120.0	3160.0	
46	CM depressurized. EVA hatch opened. Tether rigged.	64.0	3224.0	
47	Astronaut "C" exits and maneuvers to SM, removes propulsion unit, straps to self. Maneuvers to image capsule and secures unit to truss.	24.0	3248.0	593.0
48	Old propulsion unit removed, new unit installed, old unit secured to truss. Returns to CM.	50.0	3298.0	643.0
49	CM hatch closed. CM pressurized.	34.0	3332.0	
50	Suits doffed. PLSS regenerated. All systems checked. Biomed checks. Crew and ground critique.	60.0	3392.0	
51	Eat, personal hygiene, housekeeping, biomed.	90.0	3482.0	
52	Space suits donned. Suit and PLSS check-out.	120.0	3602.0	
53	CM depressurized. EVA hatch opened. Tether rigged.	64.0	3666.0	
54	Astronaut "A" exits and maneuvers to SM, removes and attaches Station #3 scientific package to conveyor. Maneuvers to main truss center frame and alters location for accessibility to image capsule. Bay I. Hauls cargo to capsule and secures.	44.0	3710.0	687.0
55	"A" secures self to Bay I elevator door, engages portable power tool and opens door, exposing turret area.	10.0	3720.0	697.0

Table I-2. Time Line Analysis - Resupply (Cont'd)

Event No.	EVENT AND TASKS	EVENT TIME (MIN)	ACCUM TIME (MIN)	EVA ACCUM TIME (MIN)
56	Turret Station #3 scientific package removed and temporarily secured to nearby truss.	14.0	3734.0	711.0
57	Turret Station #3 scientific package installed. Operates turret for clearance check. Turret left at #4 Station. Door re-positioned.	32.0	3766.0	743.0
58	Astronaut "A" disengages from door. Removes old package from temporary stowage on strut and returns with package to CM (or SM).	25.0	3791.0	768.0
59	CM side hatch closed. CM pressurized.	35.0	3826.0	
60	Space suits doffed. PLSS regerated.	60.0	3886.0	
61	Eat, personal hygiene, housekeeping, biomed.	60.0	3946.0	
62	Sleep.	480.0	4426.0	
63	Eat, personal hygiene, housekeeping, biomed.	60.0	4486.0	
64	Space suits donned. Suit and PLSS check-out.	120.0	4606.0	
65	CM depressurized. EVA hatch opened. Tether rigged.	64.0	4670.0	
66	Astronaut "B" exits and maneuvers to SM, removes and attaches Station #4 scientific package to conveyer. Maneuvers to image capsule Bay #I. Hauls cargo to capsule and secures.	44.0	4714.0	812.0
67	"B" secures self to Bay I elevator door, engages portable power tool and opens door, exposing turret area.	10.0	4724.0	822.0
68	Turret Station #4 scientific package removed and temporarily secured to nearby truss.	14.0	4738.0	836.0
69	Turret Station #4 scientific package installed. Operates turret for clearance check. Turret left at #5 Station. Door repositioned.	32.0	4770.0	868.0
70	Astronaut "B" disengages from door. Removes old package from temporary stowage and returns with package to CM (or SM).	25.0	4795.0	893.0
71	CM side hatch closed. CM pressurized.	35.0	4830.0	
72	Space suits doffed. PLSS regenerated.	160.0	4890.0	
73	Eat, personal hygiene, housekeeping, biomed.	90.0	4980.0	

Table I-2. Time Line Analysis - Resupply (Cont'd)

Event No.	EVENT AND TASKS	EVENT TIME (MIN)	ACCUM TIME (MIN)	EVA ACCUM TIME (MIN)
74	Space suits donned. Suit and PLSS check-out.	120.0	5100.0	
75	CM depressurized. EVA hatch opened. Tether rigged.	64.0	5164.0	
76	Astronaut "C" maneuvers to SM, removes and attaches Station #5 scientific package to conveyor. Maneuvers to image capsule Bay I. Hauls cargo to capsule and secures.	44.0	5208.0	937.0
77	"C" secures self to Bay I door, engages portable power tool and opens door, exposing turret area.	10.0	5218.0	947.0
78	Turret Station #5 scientific package removed and temporarily secured to nearby truss.	14.0	5232.0	961.0
79	Turret Station #5 scientific package installed. Operates turret for clearance check. Door repositioned.	32.0	5264.0	993.0
80	Astronaut "C" disengages from door. Removes old package from temporary stowage and returns package to CM (or SM).	25.0	5289.0	1018.0
81	Astronaut "C" maneuvers to terminal end of conveyor line, detaches and returns to SM. Retracts conveyor line anchor device and secures. Returns to CM.	34.0	5323.0	1052.0
82	CM side hatch closed. CM pressurized.	34.0	5357.0	
83	Suits doffed. PLSS regenerated	60.0	5417.0	
84	Eat, personal hygiene, housekeeping, biomed.	60.0	5477.0	
85	Sleep.	480.0	5957.0	
86	Eat, personal hygiene, housekeeping, biomed.	60.0	6017.0	
87	Mission instrument checkout. All x-ray telescope systems "on". CSM ACS "off".	5.0	6022.0	
88	Ground command to lock-on prime reference source.	5.0	6027.0	
89	Turret Station #1 commanded into receiving position and "on". Stored data.	15.0	6042.0	
90	Turret Station #2 commanded into receiving position and "on". Stored data.	15.0	6057.0	
91	Turret Station #3 commanded into receiving position and "on". Stored data.	15.0	6087.0	

Table I-2. Time Line Analysis - Resupply (Cont'd)

Event No.	EVENT AND TASKS	EVENT TIME (MIN)	ACCUM TIME (MIN)	EVA ACCUM TIME (MIN)
92	Target not available. Verification from ground station that turret Stations #1, 2 and 3 receiving.	30.0	6117.0	
93	Turret Station #4 commanded into receiving position and "on". Stored data.	30.0	6147.0	
94	Turret Station #5 commanded into receiving position and "on". Stored data.	30.0	6177.0	
95	Target not available. Verification from ground station that turret Stations #4 and #5 receiving.	30.0	6207.0	
96	Turret Station #6 commanded into receiving position and "on". Stored data.	15.0	6222.0	
97	Turret Station #7 commanded into receiving position and "on". Stored data.	15.0	6237.0	
98	Critique and verification from ground that Stations #6 and #7 are receiving.	45.0	6282.0	
99	Eat, personal hygiene, housekeeping, biomed.	90.0	6372.0	
100	Preparation for CSM separation from telescope.	30.0	6402.0	
101	Space suits donned. Suit systems check.	120.0	6522.0	
102	Docking locks and electrical umbilical disconnected. Pressure hatch installed.	60.0	6582.0	
103	CSM undocked. Crew maneuvers CSM in fly-around inspection. Photographic coverage.	90.0	6672.0	
104	Detached operational check. CSM crew on standby as telescope is commanded from ground inputs to cycle all instruments through operation and verify systems operational. Assumed accomplished in approximately two orbits.	18.0	6852.0	
105	Mission terminated.			

Total time to operational status:	**4 days**	**18 hours**	**12 minutes**
Accumulated EVA time outside of CM		**17 hours**	**32 minutes**

Notes: (1) See Appendix I for detailed work sheet of the resupply mission.

(2) See Figure I-2 for summary bar graph of the resupply mission.

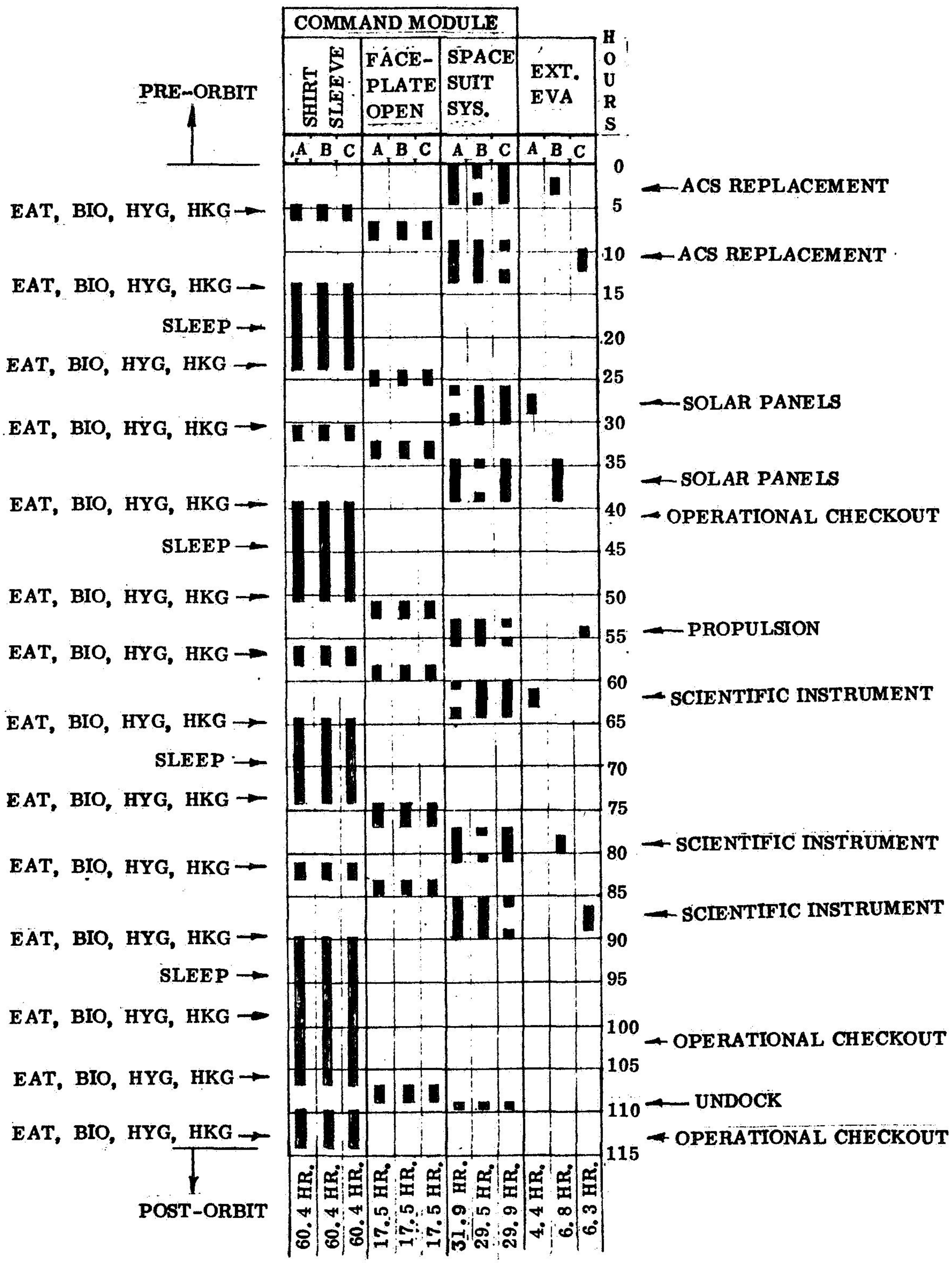

Figure I-24. Resupply Mission Summary

f. Astronaut Support Equipment Requirements

The following equipment will be required to accomplish the inspection, repair, and/or replacement of damaged components and refurbishment of the x-ray telescope:

(1) A portable life support system (PLSS)

(2) Fixed and portable worksite anchors including dutch shoes

(3) A communication system

(4) Illumination system on the experiment

(5) Fixed and portable illumination system

(6) Fixed and portable power supply system

(7) The necessary maintenance and repair tools and equipment and a tether to prevent tools and equipment from floating away

(8) Hand-held maneuvering unit and hand rails

(9) A 50 ft. tether

The (PLSS) will supply pressurized oxygen to the pressure garment assembly (PGA), clean and cool the expired oxygen, circulate cool liquid in the liquid-cooled garment (LCG) and function as a transceiver for the astronaut bioinstrumentation and communication. The operating time of this unit is 5 hr. at 1200 Btu/hr; 3.25 hr. at 1600 Btu/hr; and 2 hr. at 2000 Btu/hr. The oxygen and water systems may be recharged in either the Airlock module or Command Module.

The use of an umbilical in lieu of a PLSS was the choice for this experiment but the present design of Apollo CM does not allow its use due to pressurization limitations. Modification of the CM to allow for the use of an umbilical in this system would not only provide oxygen, H_2O, biomedical/communications and use as a tether but would afford the astronaut more dexterity, mobility and would not create any weight problems. The use of an umbilical for EVA is the ideal life support system to use versus the other life support systems now available.

The fixed worksite anchors will consist of a waist restraint which will provide astronaut torso stability in any given position while leaving both hands free for work. This restraint is an assembly consisting of a waist harness with two controllable flexible legs attached which are approximately 45 in. long. The application of this equipment will be primarily at the worksite to aid the astronaut in maintaining body position with a minimum of energy expenditure. It has the advantage when compared to flexible waist tehters, of maintaining the body in a specific position with respect to the worksite and of freeing both hands for any required work tasks. The variable waist restraint is used in conjunction with foot restraints which will be "built into" the design. Dutch shoes will be made available if required.

The communications system is part of the PLSS and this affords voice communication between the EV astronaut, the parent spacecraft and the backup man who is acting as a "life guard" in case of an emergency. A reliable communication system is vital for mission success and the safety of the astronauts.

Due to the low orbit (260 n. mi.) in which this experiment will be located, it will be exposed to 60 min. of daylight and 30 min. of night (dark) per orbit. If EVA were to be limited to daylight, the refurbishment task would take too many days to accomplish, therefore, adequate illumination on the x-ray telescope structure has been provided in the design. This allows for EVA at night which will expedite the completion of the required task. It is recommended that the EVA crewman's workload schedule take advantage of as much daylight as possible. This will avoid overtaxing the power supply (for structural illumination) and will reduce any safety constraints involved by EVA at night. Other lighting required should consist of helmet-mounted or some other portable lighting system. The helmet-mounted lights are preferred to minimize tool or limb interference with the illumination of the tasks.

A power supply system is required to provide power for the astronaut's equipment, illumination system, telemetry, voice communication, environmental control, etc. This power supply should be designed to preclude the need for the EV crewman to periodically return to his parent spacecraft for recharging his power supply during the normal EVA period.

Tools and equipment should be available for the repairs, replacement of damaged components and the refurbishment of the experiment. A Black and Decker tool kit would or could contain all the tools required for EVA on this experiment which would consist primarily of a small hand motor, pliers, screwdriver and a small plastic mallet.

The tool kit should have a temporary holding device which can be utilized while the EV crewman is working. Equipment that may require replacement are solar cell panels, altitude control system units, etc. Restraints will be required which will prevent equipment and tools from floating away.

The mobility of the EV astronaut will be greatly enhanced by the fact that the CM will be docked to the x-ray telescope (decreasing travel distance) when EVA is required. In addition, hand rails will be available on the structure of the telescope. The use of a hand-held maneuvering unit is recommended as an added mobility assist and would be a great asset in the event of an emergency. This unit cannot be used exclusively for mobility primarily due to its limited operational life. The use of a AMU was considered and rejected due to the mass of the unit and the excessive effort and time required to don and remove the unit. These factors also can be responsible for fatigue which would cut into the EVA time available. The required time for checkout and donning of this unit is 25 min. and its lifetime is approximately one hour. The use of an AMU would also adversely affect the astronaut's dexterity and mobility and would not allow him to work in small areas.

The EVA for this experiment will require a 50 ft. tether which will serve as a safety factor. It can prevent the astronaut from floating away but care must be taken when egressing or ingressing to avoid entanglement.

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4. PRE-LAUNCH SUPPORT

a. Shipping and Handling Procedures

Material handling and packaging is controlled by National Aerospace Standards (NAS). These standards establish the methods, materials and devices to be used throughout the procurement, receiving, manufacturing and shipping phases of the program.

Containers are marked in accordance with MIL-STD-129.

b. Installation and Checkout Procedures

Since the x-ray telescope is installed on the launch vehicle in the place of the LEM, and uses the same attachment points, the procedures used for installation will be modelled after the LEM procedures and will to the greatest extent practicable, use actual blocks from the LEM procedures.

Pre-launch checkout of the x-ray telescope is accomplished by means of its own checkout console. This console will have the capability to monitor all functions necessary for launch control. It will also have the capability to exercise all on-board functions through either a hardline connection or through an R-F command data link. Detailed analysis of the on-board systems will be accomplished by means of data transmitted over the spacecraft telemetry data link. The console will have the capability to receive and record this data. Procedures for the checkout of the spacecraft with the console at the launch site will be the same as those used for checkout and sell-off at the contractors facility.

As a further check of the compatibility of the telescope systems with the launch vehicle and with the MSFN and DSIF ground stations the spacecraft will be shipped to NASA/MSFC for fit checks and R-F data link checks prior to shipment to the launch site.

c. Facilities

In addition to the facilities contained at Complex 34 and/or Complex 37 at KSC, the x-ray telescope will require the use of a Missile Assembly Building (MAB) for the performance of engineering confidence tests by the contractor and general checkout prior to delivery to the launch pad for installation on the launch vehicle. The MAB will accommodate the spacecraft in either the horizontal or vertical positions. As such, a minimum clear height of 42 ft. is required with appropriate overhead cranes and support capability. The assembly area shall be environmentally controlled to the requirements of Federal Standard 209, Class 100,000 Clean Rooms.

d. Test Equipment

Checkout of the x-ray telescope is accomplished with the checkout console. Standard calibration and validation services are required for the quality control of this unit. In addition, standard electrical/electronic test equipment will be required.

e. Services

(1) Fill and drain of the H_2O_2 AttitudeControl System.

(2) Helium purge and pressurization of the Attitude Control System.

(3) Compressed nitrogen (4,000 psi) is required for fueling the emergency docking attitude control system.

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5. FLIGHT OPERATIONAL REQUIREMENTS

The x-ray telescope is an independent spacecraft operating in a low-altitude orbit. Initial deployment and checkout operations are manned. Subsequent astronomy mission operations are unmanned and automated for remote control by ground stations, as follows:

a. Command Control

b. Telemetry Data Acquisition

c. Data Reduction and Evaluation

The on-board systems are designed to be compatible with the MSFN stations.

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6. RECOVERY REQUIREMENTS

None

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7. DATA SUPPORT REQUIREMENTS

Not defined

SECTION V - RESOURCE REQUIREMENTS

1. FUNDING REQUIREMENTS

a. *Summarize total experiment cost by major category of expenditure as outlined below:*

ITEM	AMOUNT
DIRECT LABOR (Separate by Labor Category; Rate per hour or man-month; Personnel involved, what they will do, etc.)	$ 15,065
MANUFACTURING BURDEN (Overhead) RATE (%) (Flight experiments normally will be supported by contracts rather than grants.)	
MATERIALS (Total) (Bill of Material, including estimated cost of each major item.)	7,185
SUBCONTRACTS (List those over $25,000) (Specify the vendor if possible, and the basis for estimated cost. Include baseline study contracts.)	17,290
SPECIAL EQUIPMENT (Total) (List of lab equipment, purposed uses, and estimated costs.)	595
TRAVEL (Estimated number of individual trips, destinations, and costs.)	125
ANY OTHER ITEMS (Total) (Explain in detail similar to the above.) Phase B Completion Studies	200
TOTAL COSTS	$
General and Administrative Rate ( )	$
TOTAL ESTIMATED COST	$ 40,460

b. Funding Obligation Plan. Provide the preliminary funding requirements of the experiment by quarter as indicated on the attached sheet (Quarterly Funding Requirements). Funding should be broken into the general areas indicated on the following page and should identify the source of funding for each area.

SECTION V - RESOURCE REQUIREMENTS (Cont'd)

C. Quarterly Funding Requirements (Dollars in Thousands)

ITEMS	FUNDING SOURCE	FY 1				FY 2				FY 3				FY 4				TOTALS
		QUARTERS				QUARTERS				QUARTERS				QUARTERS				
		1	2	3	4	1	2	3	4	1	2	3	4	1	2	3	4	
Definition, Breadboard, Design, Development, Fabrication, Test (mock-ups, prototypes, and support equipment)		580	1080	1520	1840	2510	2410	2750	1985	2085	1935	2105	1710	1165	1125	1040	990	26, 830
Fabrication, Test and Delivery (Flight units and spares)									550	750	900	1450	1500	1550	1300	665	110	8, 775
Supporting Studies and Other Implementation Efforts		90	100	150	250	275	285	325	325	335	350	360	350	370	380	380	330	4, 655
Data Analysis and Publication*																		
YEARLY TOTALS		5, 610				11, 415				13, 830				9, 405				
*Not Estimated Separately														GRAND TOTAL				40, 260

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SECTION V - RESOURCE REQUIREMENTS (Cont'd)
2. PRELIMINARY DEVELOPMENT SCHEDULE

MAJOR MILESTONES	PLANNED DEVELOPMENT SCHEDULE																			
	FY 1				FY 2				FY 3				FY 4				FY 5			
	QUARTERS				QUARTERS				QUARTERS				QUARTERS				QUARTERS			
	1	2	3	4	1	2	3	4	1	2	3	4	1	2	3	4	1	2	3	4
EIP COMPLETE			▼																	
MSFEB Action				▼																
Hardware Contract				▼																
ICD Complete					▼															
Design Complete													▼							
DEP Complete										▼										
Prototype Delivered													▼							
Qualification Testing Complete														▼						
Flight Units Fabricated																	▼			
Delivery of Flight Hardware																		▼		

SECTION V - RESOURCE REQUIREMENTS (Cont'd)

3. MANPOWER

Provide a brief summary of manpower requirements both in-house and contract.

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4. FACILITIES

Provide a brief listing of facilities and major lab equipment requirements. Specifically identify new facility requirements. Whenever possible, indicate the schedule of usage for each item.

APPENDIX II
X-RAY TELESCOPE RE-SUPPLY OPERATION

The following tables document a preliminary orbital task analysis which was performed to examine the impact of manned participation on the x-ray telescope design and to assess the time, equipment and procedures necessary to integrate man effectively into the telescope system. Most of the analysis was limited to the 'normal operations" as the study schedule would not permit examination of the many failure modes which, can be postulated.

 PART A

Table II-1. X-ray Telescope Resupply Phase Normal Operation

GROSS FUNCTION	SYSTEM OR COMPONENT FUNCTION	EVENT TIME (MIN)	ELAPSED TIME (MIN)	DISPLAY INDICATION	CREW ACTION OR PARTICIPATION (AT CM STATION UNLESS OTHERWISS SHOWN)	EVA EQUIPMT.	POSSIBLE EVA HAZARDS	SAFETY OR EMER. PROCD.	REMARKS
Deactivate Telescope Systems	1 Scientific instrument "off". CSM is docked onto port of image capsule. AC is via X-Ray facility immediately following docking.	Point of beginning T = 0	0	Normal CM display. ICSP shows pertinent data of telescope status. Ground displays	Monitor displays. Crew in pressure suits. Deactivate CSM ACS.				The time interval for this resupply is at the scheduled depletion of the attitude control expendables and solar panel replacement. Intermediate resupply intervals for replacement of scientific instruments or other items are note defined at this time.
	2 CM/drogue pressure equalized	5.0	5.0	Pressure guage at CM hatch	Astro "A" crawls to docking hatch and manipulates valve				
	3 Hatch removed. Manual docking locks engaged. Elect umbilical connected, checked.	25.0	30.0		Astro "A" performs locking and connection task.				
	4 Hatch reinstall.	3.0	33.0		Astro "A" secures hatch & returns to CM station.				
	5 Telescope ACS "off"	.5	33.5		Monitor of displays				
	6 CSM ACS "on"	.5	34.0		Activate CSM ACS				6 Total spacecraft is under control of CM crew.
Operational Systems Replacement	1 ACS units replacement	0.0	34.0						One ACS unit is approximately 13" diameter x 24" long, weighs 75 lbs. Storage is in SM.
	1.1 CM depressurized.	4.0	38.0						
	1.2 CM hatch opened. Tether riggig, etc.	60.0	98.0		Astro "B" opens side hatch and exits. Astro "A" and "C" in pressure suits in CM.	*Equipment	Entanglement with tether and/or structure contact.		*Equipment consists of spacesuit, PLSS, tether, dutch shoes, handheld maneuvering unit and portable power tool.
	1.3 Conveyor line erection & transport	46.0	114.0	Remote TV	EVA "B" traverses to SM Sector I, secures self, opens door & erects conveyor line anchor device. Attaches carriage containing 2 ACS units to rigid portion of line. Traverses to main truss center frame & secures conveyor to truss receptacle. Hauls cargo to frame and secures.		Structural items within and/or around Sector I of SM Entangelment of tether in truss structure.		1.1 & on - All EV crew actions shown and event times include 2 min. rest for approximately every 10 minutes of effort.
	1.4 Retraction of #1 ACS/solar panel frame	4.0	148.0		EVA "B" plugs portable hand power tool into ACS frame gear box. Retract frame and solar panel.		Equipment in motion	Astro. to be at assigned, captured position before activation of retract system	
	1.5 ACS #1 unit installed	30.0	178.0		EVA "B" removes 1 unit from carriage. Traverses to work position & secures self. Trips isolation switch inerting old unit. Engages new unit onto old unit. Secures with integral ring clamp. Engages electrical connector. Returns to conveyor.		Proper maneuvering and securing of self while retaining the ACS unit.	Unit to be secured to Astro. while maneuvering. Unit tethered while being attached.	
	1.6 Retraction of #2 ACS/solar panel frame.	4.0	182.0		EVA "B" plugs portable hand power tool into #2 ACS frame gear box. Retracts frame and solar panel.		ditto	ditto	
	1.7 ACS #2 unit installed	30.0	212.0		EVA "B" removes unit from carriage. Traverses to work position & secures self. Trips isolation switch inerting old unit. Engages new unit onto old. Secures with integral ring clamp. Engages		ditto	ditto	

FOLDOUT FRAME 1

PART B

1.8 CM side hatch closed	30.0	242.0		"B" closes CM hatch				1.1 through ... Total EVA time = 3 hours 33 minutes EVA work time = 1 hour 54 minutes
1.9 CM pressurized	5.0	247.0	CM normal displays	Monitor displays				
1.10 Spacesuits doffed. PLSS regenerated	60.0	307.0	ditto	All systems check				
1.11 Eat	90.0	397.0						
1.12 Spacecuits donned. Suit and PLSS checkout.	120.0	517.0	ditto	Don spacecuits				
1.13 CM depressurized	4.0	521.0						
1.14 CM EVA side hatch opened. Tether rigging, etc.	60.0	581.0		EVA "C" opens side hatch & exits. "A" & "B" in pressure suits.	*Equip. See remark 1.2	Entanglement w/tether and/or structure contact.	Hand holds on CSM exterior. CM hatch left open.	
1.15 #3 & #4 ACS units installation	12.0	593.0		"C" traverses to SM sector I and secures self		Structural items within and/or around Sector I of SM		
1.16 Conveyance of #3 & #4 units	30.0	623.0		"C" retracts conveyor carriage & attaches ACS units #3 & #4. Traverses to main truss center frame & hauls cargo to frame & secures.		Entanglement of tether in truss structure.		
1.17 Retraction of #3 ACS/solar panel frame	4.0	627.0		"C" plugs portable hand power tool into ACS frame gear box. Retracts frame & solar panel.		Equipment in motion	Astro to be at assigned, captured position before activation of retract system	
1.18 ACS #3 installed	30.0	657.0		"C" removes one unit from carriage. Traverses to work position & secures self. Trips isolation switch inerting old unit. Engages new unit onto old. Secures with integral ring clamp. Engages electrical connectors. Traverses to #4 frame.		Proper maneuvering & securing of self while retaining the ACS unit	Unit to be secured to astro while maneuvering. Unit tethered while being attached.	
1.19 Retraction of #4 ACS/solar panel frame	4.0	661.0		"C" plugs portable tool into #4 frame gear box. Retracts frame and solar panel.		ditto	ditto	
1.20	35.0	696.0		"C" removes unit from carriage. Traverses to work position & secures self. Trips isolation switch inerting old unit. Engages new unit onto old. Secures with integral ring clamp. Engages elec. connector. Returns to SM.		ditto	ditto	
2 Solar panel replacement	0	696.0						
2.1 Conveyance of #1 and #2 units	32.0	728.0		"C" secures self to SM. Retracts conveyor carriage. Removes two cannisters from SM & attaches to carriage. Traverses to main truss center frmae & hauls cargo, via conveyor, to frame and secures. Returns to CM.		Structural items within and/or around Sector I of SM and truss structure		
2.2 CM side hatch closed	30.0	758.0		"C" closes CM hatch				
2.3 CM pressurized	5.0	763.0	CM normal displays	Monitor displays				Total EVA time = 4 hours 6 minutes EVA work time = 2 hours 27 minutes

FOLDOUT FRAME 2

Table II-1. X-ray Telescope Resupply Phase (cont'd.) Normal Operation

GROSS FUNCTION	SYSTEM OR COMPONENT FUNCTION	EVENT TIME (MIN)	ELAPSED TIME (MIN)	DISPLAY INDICATION	CREW ACTION OR PARTICIPATION (AT CM STATION UNLESS OTHERWISE SHOWN)	EVA EQUIPMT.	POSSIBLE EVA HAZARDS	SAFETY OR EMER. PROCD.	REMARKS
	2.4 Spacesuits doffed. PLSS regenerated	60.0	823.0	CM normal displays	All systems check				
	2.5 Eat	60.0	883.0						
	2.6 Sleep	480.0	1363.0						
	2.7 Eat	60.0	1423.0						
	2.8 Spacesuits donned. Suit & PLSS checkout	120.0	1543.0	CM normal displays	Don spacesuits. Mutual assistance & checkout.				
	2.9 CM depressurized	4.0	1547.0						
	2.10 CM EVA side hatch opened. Tether rigging, etc.	60.0	1607.0		"A" opens side hitch and exits. "B" & "C" in pressure suits in CM.				
	2.11 Solar panel #1 installed	50.0	1657.0	ICSP	"A" traverses to #1 frame. Operates isolation switch, disengages clamp fasteners along 2 sides & end of panel frame. Releases ratchet lock & panel retracts into cannister. Traverses to conveyor, extracts #1 cannister & returns to work position. Installs cannister adjacent to old cannister & secures. Routes leading edge lanyard thru pulley & extends. Ratchet lock permits extension only. Clamp fasteners engaged. Elec. harness connected. Isolation switch operated "on"		Tether entanglement and/or structure contact. Large size of cannister presents difficult handling task.	Handholds on CSM exterior. CM hatch left open. Foot & body restraints at work site.	Cannister size is 8-10 inches diameter by 8 ft long.
	2.12 Extension of ACS/solar panel frame #1	12.0	1669.0	Visual lock indication of panel locks by "A". CM display	"A" traverses to frame gear box, inserts portable power tool & extends frame. Checks for lock.		Equipment in motion	Astro to be at assigned captured position before activation of retract system	
	2.13 Solar panel #2 installed	50.0	1719.0	ICSP	"A" task same as for #1 panel		As previously noted	As previously noted	
	2.14 Extension of ACS/solar panel frame #2	28.0	1747.0	Visual lock indication of panel lock.	"A" task same as for #1 panel. "A" returns to SM and retracts conveyor carriage. Returns to CM.		ditto	ditto	
	2.15 CM side hatch closed	30.0	1777.0		"A" closes CM side hatch				
	2.16 CM pressurized	5.0	1782.0	CM normal displays	Monitor displays				Total EVA time = 3 hours 59 minutes EVA work time = 2 hours 20 minutes
	2.17 Spacesuits doffed. PLSS	60.0	1842.0	ditto	All systems check				
	2.18 Eat	90.0	1932.0						
	2.19 Spacesuits donned. Suit and PLSS checkout	120.0	2052.0		Don spacesuits. Mutual assistance and checkout				
	2.20 CM depressurized	4.0	2056.0						
	[illegible]	[illegible]	[illegible]		"B" opens side hatch & exits.				

2.22 Conveyance of solar panels #3 and #4	38.0	2154.0		"B" traverses to SM Sector I & secures. Removes two cannisters from SM & attaches to carriage. Traverses to main truss center frame and hauls cargo, via conveyor, to frame and secures.		As previously noted	As previously noted	
2.23 Solar panel #3 installed	50.0	2204.0		"B" task same as for #1 panel.		As previously noted	As previously noted	
2.24 Extension of ACS/solar panel frame #3	12.0	2216.0		"B" task same as for #1 panel		ditto	ditto	
2.25 Solar panel #4 installed	50.0	2266.0		"B" task same as for #1 panel		ditto	ditto	
2.26 Extension of ACS/solar panel frame #4	18.0	2284.0		"B" task same as for #1 panel. "B" returns to CM.		ditto	ditto	
2.27 CM side hatch closed	30.0	2314.0		"B" closes CM side hatch				
2.28 CM pressurized	5.0	2319.0						Total EVA time = 4 hours 27 minutes EVA work time = 2 hours 48 minutes
2.29 Spacesuits doffed. PLSS regenerated	60.0	2379.0		Doff spacesuits. All systems check.				
3 EAT	60.0	2439.0						Many items will be replaced in this resupply. To avoid unknowns associated with interactions between subsystems and thus a time consuming trouble-shooting procedure, the replacement philosophy is thus: where possible, each subsystem having replacement items will receive an operational check following replacement and before other systems are reworked. Proper scheduling should cause checkout to coincide with crew in CM for other purposes such as rest. Will result in many pressure cycles of CM.
3 Operational checkout of solar panels Power "on"	30.0	2469.0	CM & ground displays of pertinent elec characteristics	Actuates power switch. Monitors displays. Ground critique.	None	None		
5 Operational checkout of ACS system. CM ACS "off". Telescope ACS "on"	30.0	2499.0	CM & ground displays of pitch, yaw and roll.	Actuation of override control to energize each thrustor. Monitor displays				
6 Telescope ACS "off". CM ACS "on"	1.0	2500.0						
7 Sleep	480.0	2980.0						
8 Eat	60.0	3040.0						
9 Propulsion system replacement	0	3040.0						
9.1 Spacesuits donned. Suit and PLSS checkout.	120.0	3160.0						
9.2 CM depressurized	4.0	3164.0						
9.3 CM side hatch opened. Tether rigging, etc.	60.0	3224.0		"C" opens hatch. "A" & "B" in spacesuits in CM	*see remark L.2			
9.4 Transport of new unit	24.0	3248.0		"C" traverses to SM Sec.I & secures self. Removes propulsion unit. Straps to self. Traverses to image capsule truss support aft end. Straps new unit to truss structure.		As prev.noted	As prev.noted	

Table II-1. X-ray Telescope Resupply Phase (cont'd.) Normal Operation

GROSS FUNCTION	SYSTEM OR COMPONENT FUNCTION	EVENT TIME (MIN)	ELAPSED TIME (MIN)	DISPLAY INDICATION	CREW ACTION OR PARTICIPATION (AT CM STATION UNLESS OTHERWISE SHOWN)	EVA EQUIPMT.	POSSIBLE EVA HAZARDS	SAFETY OR EMER.PROCD.	REMARKS
	9.5 Propulsion unit removed	20.0	3268.0		"C" disconnects elec harness. Disengages retaining locks. Secures unit to truss frame receptacle.				
	9.6 Propulsion unit installed	30.0	3298.0		"C" removes new unit from temporary attach to truss. Secures with retaining locks. Connects elec harness Returns to CM.				
	9.7 CM hatch closed	30.0	3328.0		"C" closes hatch.				
	9.8 CM pressurized	4.0	3332.0						Total EVA time = 2 hours 52 minutes EVA work time = 1 hours 13 minutes
	9.9 Suits doffed. PLSS regenerated	60.0	3392.0						
	10 Eat	90.0	3482.0						
Scientific Instrument Replacement	See remarks column								The scientific instruments are mounted on a rotating turret. The turret contains 7 stations. Stations 3,4, and 5 are occupied by 3 primary scientific packages, capable of 4 separate observations. Sta. 1 contains a backup package for 5 and Sta. 7 likewise for 3. Station 2 and 6 are vacant at launch for addition of more advanced instruments via resupply. Add-on is preferred to replacement, however, all packages are removable by astronaut and will fit CM for earth return. Installation or removal consists of 2 latch devices and an electrical harness connector. Resupply interval for replacement or add-on has not been determined however, the nature of such equipment forces the assumption that all stations are full from previous resupply missions. The task on this missin will consider replacement of 3 scientific packages.
	1 Scientific instrument circuits "off"	0	3482.0						
	2 Spacesuits donned. Suit & PLSS checkout.	120.0	3602.0		Don spacesuits, mutual assistanee & checkout.				
	3 CM depressurized	4.0	3606.0						
	4 CM EVA side hatch opened. Tether, rigging, etc.	60.0	3666.0		"A" opens side hatch and exits. "B" & "C" in spacesuits in CM.	*see remark 1.2			
	5 Handling of first package	44.0	3710.0		"A" traverses to SM Sec. I & secures self. Retracts conveyor carriage. Extracts Sta. #3 scientific package & secures to carriage. Traverses to main truss center frame & alters location of conveyor line to place cargo readily accessible from Bay I of image capsule. Hauls package via conveyor line to Bay I & secures.	As previously noted	As previously noted		
	6 Access to instrument bay	10.0	3720.0		"A" positions & secures self to Bay I elevator door. Engages drive head of portable power tool into door receptacle & triggers. Door tra-		Powered door raises possibility that any contact of spacesuit with sur-		

FOLDOUT FRAME /

							Adverse suit damage.		
	7 Sta #3 package removed	14.0	3734.0		"A" operates turret manual switch to place Sta #3 in replacement position. Disconnect elec harness. Disengages latches & removes package. Temporarily secures to receptacle provided on nearby strut.		Possibility of loose instrument package.	Astro. must attach retainment tether prior to instrument removal.	
	8 Scientific package installed at Station #3	32.0	3766.0		"A" disconnects package from conveyor. Installs onto turret. Engages latches. Connects harness. Checks for clearances. Operates turret for clearance checks. Stops with Sta #4 in receive position. Repositions door.		Equipment in motion	Astro to be at assigned captured position before activation of turret sys.	
	9 Disposal of old package	25.0	3791.0		"A" disengages self from door. Disconnects old package from temporary stowage on strut & returns to CM with package. See remarks column.		Proper maneuvering & securing while retaining instrument package.	Unit to be secured to astronaut while maneuvering. Unit tethered while being attached.	Disposal may also consist of securing package within SM. CM stowage is for purposes of engineering study of speciment on earth return.
	10 CM side hatch closed.	30.0	3821.0		"A" clses CM hatch.				
	11 CM pressurized	5.0	3826.0						Total EVA time = 3 hours 44 minutes EVA work time = 2 hours 5 minutes
	12 Spacesuits doffed. PLSS regenerated	60.0	3886.0						
	13 Eat	60.0	3946.0						
	14 Sleep	480.0	4426.0						
	15 Eat	60.0	4486.0						
	16 thru 26 are duplicate events of 11.3.2 thru 12 except #4 station equipment replaced.	404.0	4890.0						26 Total EVA time = 3 hours 39 minutes EVA work time = 2 hours 5 minutes
	27 Eat	90.0	4980.0						
	28 thru 35 are duplicate events of 2 thru 9 except #5 station equipment is replaced.	309.0	5289.0		"C" performs last task per 9 except does not enter CM.		As previously noted	As previously noted	
	36 Conveyor line stowage	34.0	5323.0		"C" maneuvers to terminal end of conveyor line, detaches & maneuvers to SM Sec. I. Retracts conveyor line anchor device & secures. Maneuvers to CM.				
	37 CM side hatch closed	30.0	5353.0		"C" closes hatch.				
	38 CM pressurized	4.0	5357.0						Total EVA time = 4 hours 13 minutes EVA work time = 2 hours 34 minutes
	39 Suits doffed. PLSS regenerated	60.0	5417.0						
	40 Eat	60.0	5477.0						
	41 Sleep	480.0	5957.0						

FOLDOUT FRAME 2

Table II-1. X-ray Telescope Resupply Phase (contd.)

Normal Operation / Abnormal Operation

GROSS FUNCTION	SYSTEM OR COMPONENT FUNCTION	EVENT TIME (MIN)	ELAPSED TIME (MIN)	DISPLAY INDICATION	CREW ACTION OR PARTICIPATION (AT CM STATION UNLESS OTHERWISE SHOWN)	FAILURE MODE	FAILURE INDICATION	CREW ACTION OR PARTICIPATION	EVA EQUIPMENT	POSSIBLE EVA HAZARDS	REMARKS
	42 Eat	60.0	6017.0								
	43 Mission instrument checkout. All x-ray systems "on". CSM ACS and navigation "off".	5.0	6022.0	CM display Ground display							
	43.1 Command data to lock-on prime reference source	5.0	6027.0	Ground display		Electronics malfunction	Optical telescope telemetered data fails show evidence of receiving image field. No response to adjustments.	Space suits donned. CM depressurized. Astronaut maneuvers to SM Sector I, obtains new electronic package. Maneuvers to telescope. Secures package on structure. Removes and replaces electronic package. Thoroughly secures old package or returns to SM. Astro returns to CM, CM pressurized, suits doffed. Elapsed time = 240 minutes.	*Equipment. Securing devices would be part of replacing package for lash-down of replaced package.	Tether entanglement. Close work area. Contact with light weight, fragile thermal shroud.	43.1 Total vehicle slews onto target and locks-on. Ground originated command.
	43.2 Mission instrument at turret Sta 1 commanded into receiving position and "on".	15.0	6042.0	Ground display CM display	CM normal activities. Standby. Ground communication						43.2 Ground or CM control. Imagery. This instrument was not replaced and only cursory check was made to verify that other tasks did not cause instrument to be inoperative. Stored data.
	43.3 Turret Sta #2 commanded into receiving position and "on".	15.0	6057.0	Ground display CM display	CM normal activities. Standby. Ground communication						43.3 Same remarks as above.
	43.4 Turret Sta #3 commanded into receiving position and "on".	30.0	6087.0	Ground display CM display	CM normal activities. Standby. Ground communication	Grating motor malfunctions.		Space suits donned. CM presurised. Astro. maneuvers to SM Sector I, obtains new grating including drive motors and maneuvers to aft end of lens capsule. Temporarily secures grating. Retracts aft lens shade, removes old grating & installs new. Returns old shade to SM. Returns to CM, pressurises, suits doffed. Elapsed time = 240 minutes	*Equipment		43.4 Ground or CM control. Imagery and slitless spectrometry. Verify instrument is receiving, compensation systems operating, grating system operating. Stored data. Malfunction. The grating design is not sufficiently progressed for assurance that replaceability is entirely feasible. May be necessary to negate grating by EVA.

FOLDOUT FRAME /

Table II-1. X-ray Telescope Resupply Phase (contd.) Normal Operation

GROSS FUNCTION	SYSTEM OR COMPONENT FUNCTION	EVENT TIME (MIN)	ELAPSED TIME (MIN)	DISPLAY INDICATION	CREW ACTION OR PARTICIPATION (AT CM STATION UNLESS OTHERWISE SHOWN)	REMARKS
	46.5 Detached operational check.	180.0	6852.0		CM displays. Standby. Visual observation.	46.5 Ground control selects target. Telescope slews to X-Ray reference source. Mission instruments cycled through operational. Accomplished 2 orbits. CSM flies formation.
Mission terminated.	See Remarks column					Total elapsed time = 4 days, 18 hrs, 12 min.

					C = 60.4 hr Total time open faceplate: A = 17.5 hr B = 17.5 hr C = 17.5 hr Total time, full suit CM system: A = 31.9 hr B = 29.5 hr C = 29.9 hr Total EVA: A = 4.4 hr B = 6.8 hr C = 6.3 hr
43.5 Target not available	30.0	6117.0	Ground display	CM normal acitivities. Auxilliary tasks systems checks until target is again available.	43.5 Verification from ground stations that turret stations #1,2 & 3 receiving.
43.6 Turret Sta #4 commanded into receiving position and "on".	30.0	6147.0	Ground display	CM normal activities.	43.6 Ground control. Crystal spectrometer. Verify instrument receiving properly.
43.7 Turret Sta #5 commended into receiving position and "on".	30.0	6177.0	Ground display	CM normal activities.	43.7 Ground control. Polarimetry. Verify instrument receiving properly.
43.8 Target not available.	30.0	6207.0	Ground display	CM normal activities. Auxilliary tasks.	43.8 Same as 43.3
43.9 Turret Sta #6 commanded into receivin position and "on".	15.0	6222.0	Ground display	CM normal activities.	43.9 Same as 43.3
43.10 Turret Sta#7 commanded into receiving position and "on".	15.0	6237.0	Ground display	CM normal activities.	43.10 Same as 43.3
44 Critique	45.0	6282.0			
45 Eat	90.0	6372.0			
46 Undocked check out.	0	6372.0			
46.1 Preparation for separation	30.0	6402.0		All systems check	
46.2 Space suits donned. Suit systems check.	120.0	6522.0			
46.3 Docking locks and electrical unbilical disconnected. Pressure hatch installed.	60.0	6582.0		Astro. crawls to docking tunnel & performs tasks of unlocking, disconnecting, installing pressure hatch and venting drogue.	
46.4 Undock & fly around. CSM ACS "on".	90.0	6672.0		Crew maneuvers CSM around telescope. Visual observations, movies, stills.	46.4 CSM remains at approx. 200 feet from telescope and maintains formation.

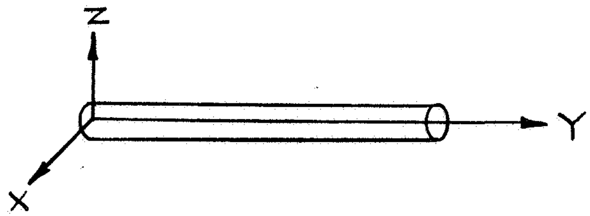
FOLDOUT FRAME 2

APPENDIX III

THERMAL GRADIENT STRESS ANALYSIS, X-RAY TELESCOPE

THERMAL MOMENT

A completely unrestrained beam as shown below is subjected to a temperature gradient, T. T is constant with respect to x and y but varies with z.

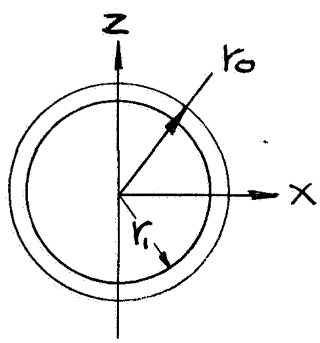


In this case the thermal moment which developes is given

$$M_{T_x} = E\alpha \int_A T(z)z \quad dA \tag{1}$$

where integration is over the cross-sectional area.

In particular for a tube and a linear gradient



$$T(z) = T_L + \frac{T_u - T_L}{2\, r_o} (r_o + z)$$

$$T(z) = \frac{T_u + T_L}{2} + \frac{T_u - T_L}{2\, r_o} z$$

Upon substitution into (1),

$$\frac{M_{T_x}}{E\alpha} = \frac{T_u + T_L}{2} \int_A z \, dA + \frac{T_u - T_L}{2\, r_o} \int z^2 \, dA \tag{2}$$

By definition, the first integral is the first area moment and, by symmetry, is zero. The second integral is

$$\frac{\pi}{4} \left(r_o^4 - r_i^4 \right)$$

Then,

$$\frac{M_{T_x}}{E\alpha} = \frac{T_u - T_L}{8} \left(r_o^4 - r_i^4 \right) \qquad (3)$$

Consider a tube with $r_o = 1.25$, $r_i = 1.23$, and let $(T_u - T_L) = 1^o$

$$M_{T_x} = E \alpha \; (.015)$$

In the case of beryllium with $E = 46 \times 10^6$, $\alpha = 6.4 \times 10^{-6}$,

$$M_{T_x} = 4.4 \text{ lb. in./}^oF$$

For aluminum, $E = 10 \times 10^6$, $\alpha = 13 \times 10^{-6}$

$$M_{T_x} = 1.95 \text{ lb in/}^oF$$

The discrete element method of analysis is formulated

$$[K] \{r\} = \{R\} \qquad (1)$$

Where $[K]$ is the stiffness matrix of a structure properly constained against rigid body displacements,

$\{r\}$ is the nodal displacement vector,

$\{R\}$ is the vector of external forces.

(Forces and displacements are used in the generalized sense to include moments and rotations)

Suppose a structure meets the following requirements:

1. No element is ever loaded beyond the material yield point
2. The principle of super position is always valid
3. The elastic properties G and E are constant and independent of load
4. The shear modulus, G, is related to Young's modulus, E, by

$$G = \frac{E}{2\,(1+\gamma)}$$

5. All elements in the structure are of the same material

Then if $\{R\}$ is composed solely of mechanical loads, i.e. not thermal loads, equation 1 can be written as

$$E \; [K] \; \{r\} = \{r\} \tag{2}$$

and the unknown vector $\{r\}$ is found as

$$\{r\} = \frac{1}{E} \; [K]^{-1} \; \{R\} \tag{3}$$

where $[K]^{-1}$ is the inverse matrix of $[K]$

From equation (3) the displacements for some material, i, are

$$\{r_i\} = \frac{1}{E_i} \; [K]^{-1} \; R \tag{4}$$

and for a different material, j,

$$\{r_j\} = \frac{1}{E_j} \; [K]^{-1} \; \{R\} \tag{4a}$$

From (4) and (4a)

$$\{r\} = \frac{E_i}{E_j} \; \{r_i\} \tag{5}$$

When $\{r\}$ is known, the vector stress resultants, or internal load, in a structural element, a, is given as

$$\{p^a\} = [k^a][c^a][a^a]\{r\} \tag{6}$$

where $[k^a]$ is the element stiffness matrix for element a in a local coordinate system

$[c^a]$ is a transformation matrix

$[a^a]$ is an assembly matrix

Under the restrictions above, (6) becomes

$$\{P^a\} = \frac{1}{E}[K^a][C^a][a^a]\{r\} \tag{7}$$

and from equation 3,

$$\{P^a\} = E[K^a][C^a][a^a]\frac{1}{E}[K]^{-1}\{R\} \tag{8}$$

Equation 8 shows that element internal loads are independent of material, in the case considered here.

If account of deformations caused by thermal loads is to be considered, the procedure is to compute for each element a a thermal load vector in local coordinate system; this is

$$\{P_t^a\} = E\alpha\{f(T)\} \tag{9}$$

Where α is the linear coefficient of thermal expansion and the column matrix on the right is in terms of the temperature of the element. This element thermal load vector is written in terms of the entire structure as

$$\left\{ P_t^a \right\} = - \left[{}^a \right]^T \left[C^a \right]^T \left\{ p_t^a \right\} \tag{10}$$

where the suffix T denotes transposition. The thermal load vector for the structure is

$$\left\{ R_t \right\} = \sum_{a=1}^{N} \left\{ P_t^a \right\} \tag{11}$$

where there are N elements in the structure

For the cases considered here, may be written

$$\left\{ R_t \right\} = (E \alpha) \left\{ R_t \right\} \tag{12}$$

and upon substitution of (12) into (4) and (5),

$$\left\{ r_i \right\} = \alpha_i \left[K \right]^{-1} \left\{ R_t \right\} \tag{13}$$

$$\left\{ r_j \right\} = \alpha_j \left[K \right]^{-1} \left\{ R_t \right\} \tag{14}$$

which leads to

$$\left\{ r_j \right\} = \frac{\alpha_j}{\alpha_i} \left\{ r_i \right\} \tag{15}$$

(The assumption is that $\left\{ R_t \right\}_i = \left\{ R_t \right\}_j$

As a result of equation (13), an internal load vector will be found for element a. This is $\left\{P_e^a\right\}$, and from equation 8,

$$\left\{P_e^a\right\}_i = \left[K^a\right]\left[C^a\right]\left[a^a\right]\left\{r_i\right\} \tag{16}$$

From 15,

$$\left\{P_e^a\right\} = \frac{\alpha_j}{\alpha_i} \cdot \left[K^a\right]\left[C^a\right]\left[a^a\right]\left\{r_i\right\} = \frac{\alpha_j}{\alpha_i} \left\{P_e^a\right\}_i \tag{17}$$

The internal loads in element a, $\left\{P^a\right\}$ is

$$\left\{P^a\right\}_j = \left\{P_e^a\right\}_i + \left\{P_t^a\right\}_i \tag{18}$$

$$\left\{P^a\right\}_j = \frac{\alpha_j}{\alpha_i} \left\{P_e^a\right\}_i + \frac{(E\alpha)_i}{(E\alpha)_j} \left\{P_t^a\right\}_i \tag{19}$$

To determine $\left\{P_i^a\right\}_j$ when $\left\{P^a\right\}_i$ is known, compute

$$\left\{P_t^a\right\}_i = E_i \alpha_i \left\{f(T)\right\} \tag{20}$$

and

$$\left\{P_t^a\right\}_j = E_j \alpha_j \left\{f(T)\right\} \tag{21}$$

Then

$$\left\{P_e^a\right\}_i = \left\{P^a\right\}_i - \left\{P_t^a\right\}_i \tag{22}$$

Find

$$\left\{P_e^a\right\}_j = \frac{\alpha_j}{\alpha_i} \left\{P_e^a\right\}_i \tag{23}$$

Addition of (21) and(23) yields $\left\{P^a\right\}_j$