ATM
STUDY PROGRAM

FINAL REPORT

PREPARED FOR
THE NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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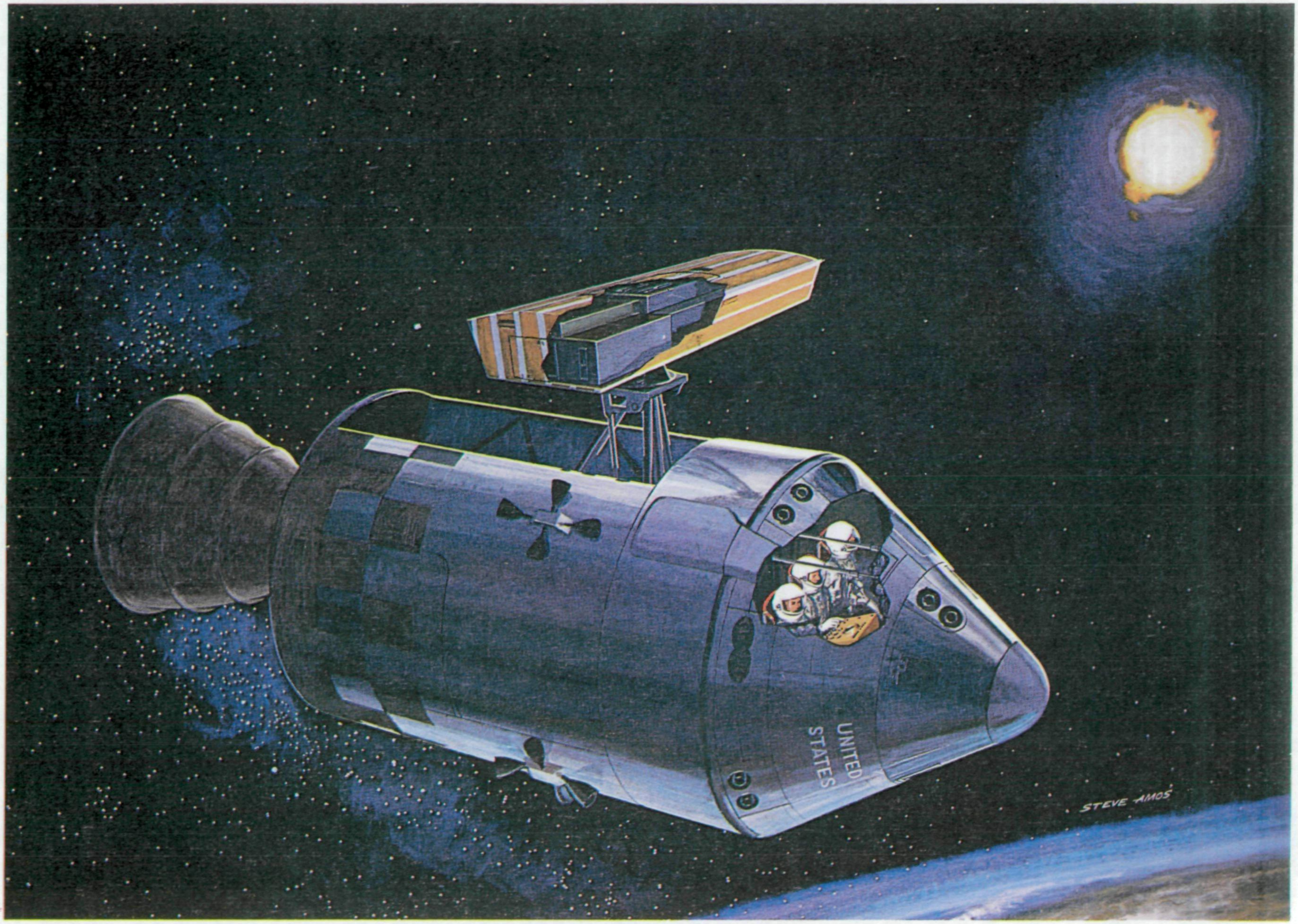
BALL BROTHERS RESEARCH CORPORATION
BOULDER, COLORADO

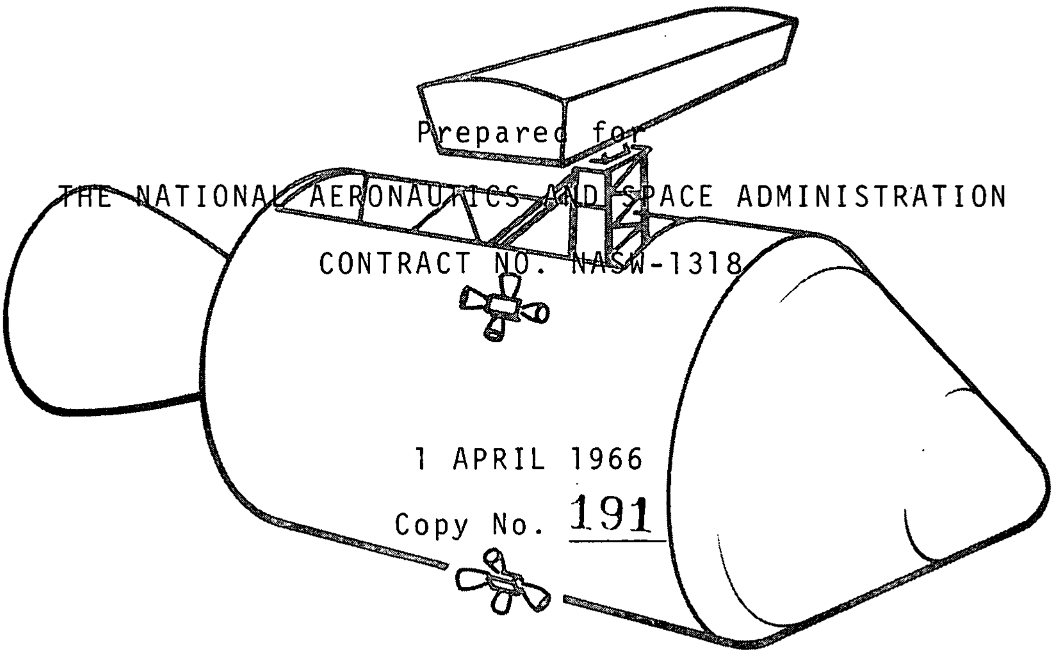


ATM/APOLLO

BALL BROTHERS RESEARCH CORPORATION

APOLLO TELESCOPE MOUNT
STUDY PROGRAM

FINAL REPORT



Prepared for

THE NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

CONTRACT NO. NASW-1318

1 APRIL 1966

Copy No. 191

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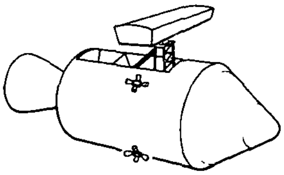

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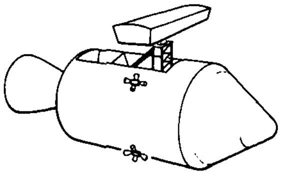

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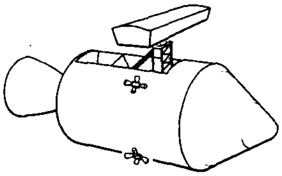

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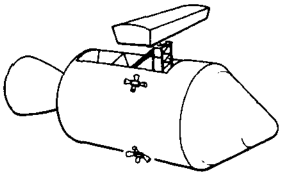


TABLES

SECTION 1

INTRODUCTION

Section 1
INTRODUCTION

This report presents the results of the 7½ month technical requirements study for a conceptual Apollo Telescope Mount (ATM). The primary guidelines under which this study was conducted are as follows:

(1) Minimum interface with the Apollo spacecraft.
(2) Near earth orbit mission.
(3) Maximum utilization of existing technology and hardware.
(4) Maximum experiment versatility for future Apollo science missions.
(5) Solar oriented system concept with adaptability to a variety of other celestial targets.
(6) Location in Sector I of Apollo Block II Service Module.
(7) Optimum utilization of the Apollo astronaut crew.

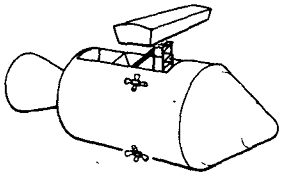
The primary objective of the study was to develop a conceptual ATM system compatible with the requirements of the scientific instrumentation it is to carry and the interface constraints imposed by the Apollo spacecraft and mission. For the purposes of this study, four typical solar experiments currently under development were considered. The principal investigators for these experiments provided the information from which the ATM performance criteria were established. The Apollo spacecraft interface data were derived from numerous NASA and NAA documents and from technical conferences with NASA and North American Aviation.

Due to the developmental nature of the Apollo spacecraft, much of the data thus derived have fluctuated during the course of this study and may now be obsolete in certain areas. Some minor changes are known as of this report writing, but were discovered too late to influence the results and recommendations of this study.

For convenience, the report is organized so that an overall picture of the ATM system concept can be obtained by reading Section 2 which describes the overall system, the subsystems, and the rationale for their selection, and the system operation.

Design considerations applicable to the total system are discussed in Section 3, where the experiment requirements are tabulated and the Apollo interface constraints discussed. Additional discussion is devoted to the experiment pointing and distortion problems, photographic considerations, contamination control, astronaut crew utilization, and reliability considerations.

The individual subsystem concepts are described in detail in Section 4 through 10 along with the trade-off considerations which led to the concept. Some of the subsystem approaches may change significantly as the experiment requirements become more firmly established and the constraints imposed on the ATM/Apollo mission become better defined.



The anticipated major ground support equipment requirements for ATM are described in Section 11.

As the ATM concept was developed, continual attention was devoted to the adaptability of the ATM concept to extended applications and where possible these considerations are reflected in the concept. Section 12 presents some of the possible extentions of the ATM concept.

The many abbreviations associated with the Apollo program and others used in this report are defined in the Glossary of Section 13.

Appendixes are included at the end of the report which present some of the more detailed information as well as a preliminary ATM performance and interface specification.

The study conclusions are:

(1) Significant space science can be accomplished via the ATM system.

(2) The astronaut crew can be very effectively utilized in the acquisition of scientific data.

(3) The Apollo spacecraft presents no constraints precluding the successful accomplishment of an ATM mission and conversely an ATM mission can be accomplished without major changes in the Apollo spacecraft and mission.

(4) The ATM concept is readily adaptable to a variety of other vehicles and targets, both celestial and terrestial.

(5) ATM can be designed using primarily state-of-the-art engineering with a minimum of new development.

Additional effort recommended as a result of this study includes:

(1) Study analysis of adaptation of ATM for operation from LEM.

(2) Further investigation into the problems of contamination of the experiments in orbit by residual particles and gas in the vicinity of the Apollo spacecraft.

(3) Study analysis of ATM system adaptation required for orientation and support of stellar and geophysical experiments.

(4) Preliminary design of the ATM system for a solar mission during the 1968 - 1970 peak solar activity period.

SECTION 2

SYSTEM CONCEPT

Section 2
SYSTEM CONCEPT

The ATM system concept has been developed to accommodate a complement of solar experiments, which will acquire data on specific solar phenomena. The system and experiments will operate with the Apollo spacecraft and will utilize the astronaut crew as operators and scientific observers.

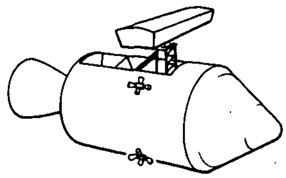
2.1 SYSTEM DESCRIPTION

The ATM system is a three-axis oriented solar research platform that is mounted in Sector I of the Apollo Service Module. It will accommodate several solar experiments capable of resolving solar features of about five arc seconds in size. The ATM system will point such experiments to any position on the solar disk and will hold the selected alignment to within ±5 arc seconds in pitch and yaw, and limit roll about the line-of-sight to one arc sec/sec during data acquisition.

The ATM consists of three major sections: (1) the Sector I mounting structure (cradle); (2) the extension mechanism; and (3) the oriented section. The ATM system is shown pictorially in Fig. 2-1 and in block diagram form in Fig. 2-2. The cradle supports the ATM components and provides attachment to the Apollo Service Module (SM). Major portions of the data, command, and power subsystems as well as the extension mechanism are mounted to this structure.

The extension mechanism and oriented section are stowed in Sector I during launch. The extension mechanism is a folded tripod, parallel linkage that is pneumatically driven through a nominal 90 degree arc. The oriented section is attached to the top of this structure and extends out of the SM parallel to its original position, as shown in Fig. 2-3. The extension mechanism locks in place when extended and positions the long axis of the oriented section parallel to the longitudinal axis of the Apollo spacecraft. When extended, the center of mass of the ATM system is approximately in the cross section plane of the CSM center of mass.

Attached to the top of the extension mechanism is a three-axis gimbal housing which supports the oriented section. An experiment support structure (spar) is attached to this "inside-out" gimbal housing which supports the experiments, portions of the ATM electronics, the solar sensors, the solar monitor telescope, and the thermal shield. (See Fig. 2-1.) Integral experiments are attached to this spar on three sides and occupy the majority of the oriented section volume. Surrounding the oriented section is a thermal shield and heaters that maintain the desired thermal environment within the section. Aperture cover mechanisms are on the sun-facing end of the thermal shield to protect the experiment optics and for thermal control. Hatches are provided in the outer face of the thermal shield for access to the experiment film magazines.



A command subsystem consisting of a control unit located in the lower equipment bay of the CM, and command decoding and distribution electronics located in the cradle and in the oriented section, enables the astronaut to control the ATM and experiments. The control unit contains all the controls and indicators necessary for this control with the exception of the solar monitor subsystem video display which is located adjacent to the control unit in the CM. The control unit and video display are depicted in the CM. (See Fig. 2-4.)

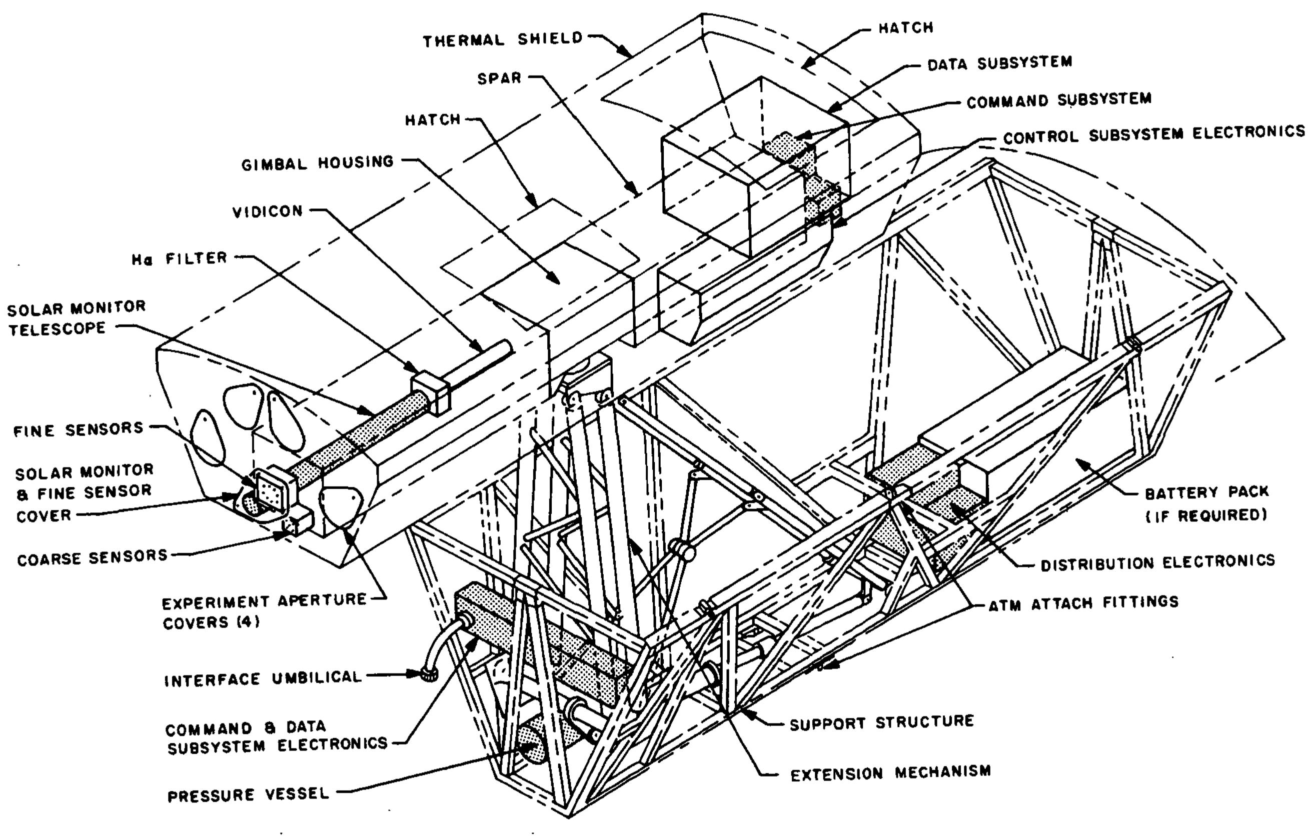
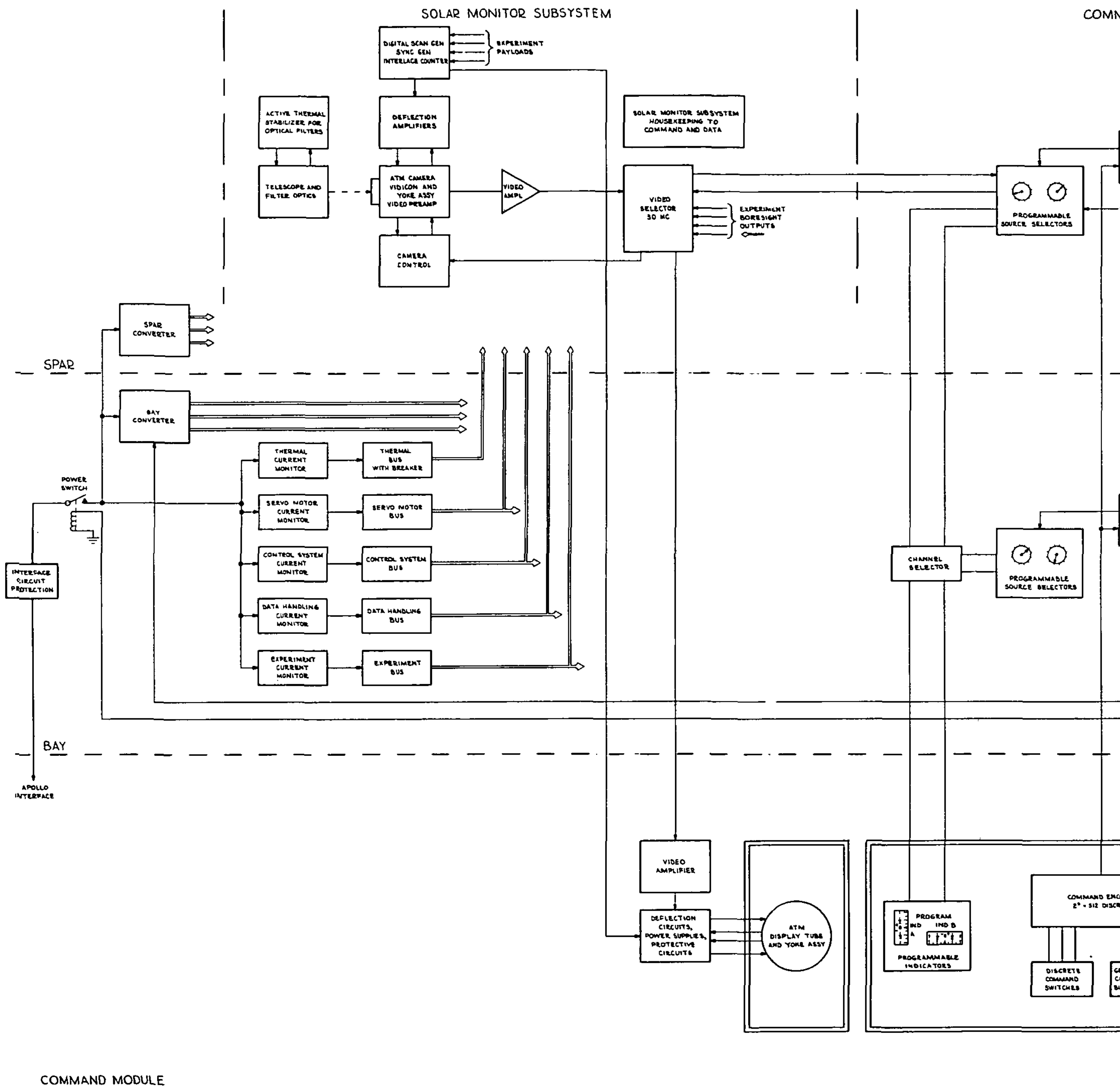
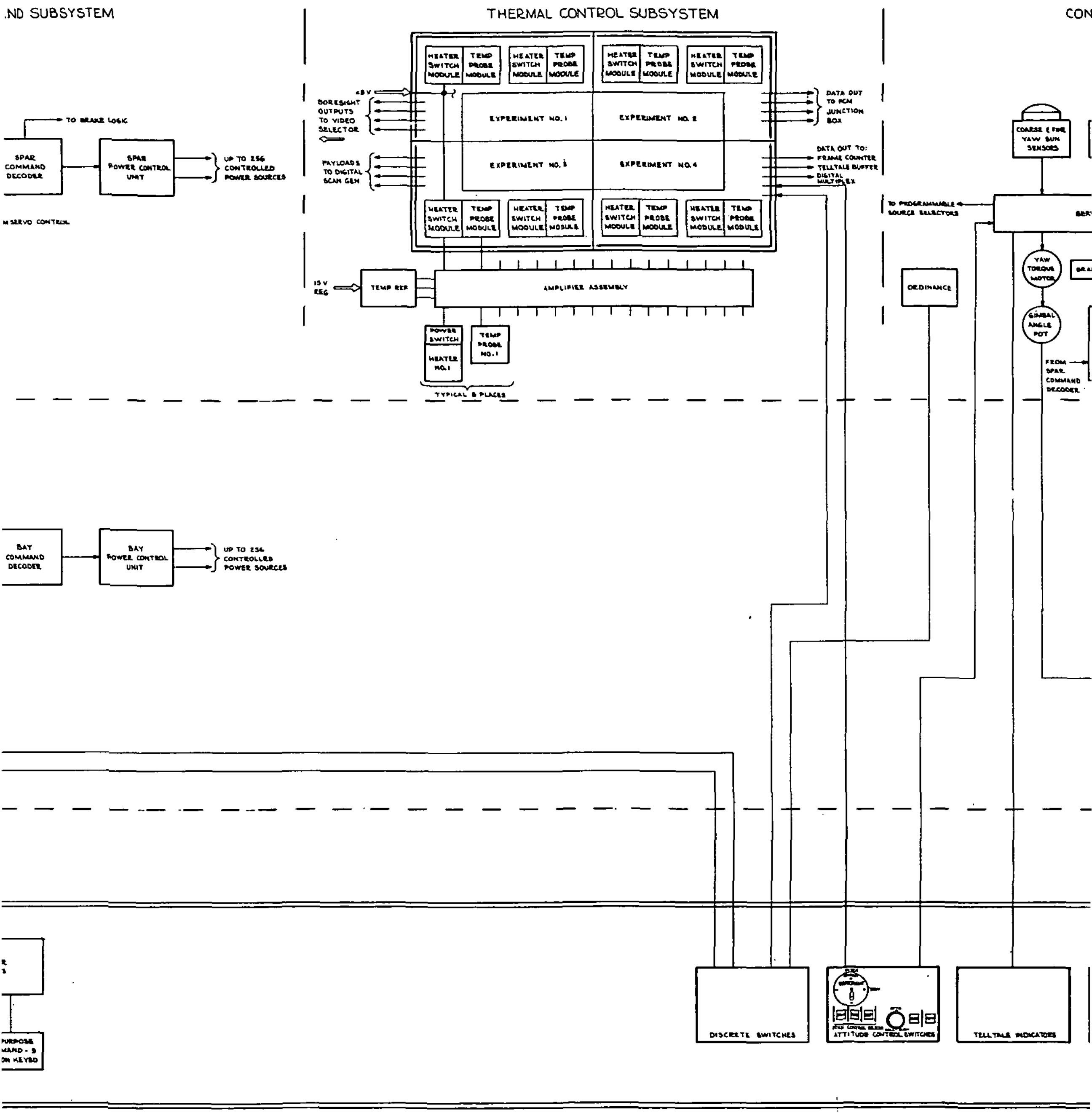


Fig. 2-1 ATM System Diagram


SOLAR MONITOR SUBSYSTEM
DIGITAL SCAN GEN SYNC GEN INTERLACE COUNTER
EXPERIMENT PAYLOADS
ACTIVE THERMAL STABILIZER FOR OPTICAL FILTERS
DEFLECTION AMPLIFIERS
SOLAR MONITOR SUBSYSTEM HOUSEKEEPING TO COMMAND AND DATA
TELESCOPE AND FILTER OPTICS
ATM CAMERA VIDICON AND YOKE ASSY VIDEO PREAMP
VIDEO AMPL
VIDEO SELECTOR 30 MC
EXPERIMENT BORESIGHT OUTPUTS
CAMERA CONTROL
PROGRAMMABLE SOURCE SELECTORS
SPAR CONVERTER
SPAR
BAY CONVERTER
THERMAL CURRENT MONITOR
THERMAL BUS WITH BREAKER
POWER SWITCH
SERVO MOTOR CURRENT MONITOR
SERVO MOTOR BUS
INTERFACE CIRCUIT PROTECTION
CONTROL SYSTEM CURRENT MONITOR
CONTROL SYSTEM BUS
CHANNEL SELECTOR
PROGRAMMABLE SOURCE SELECTORS
DATA HANDLING CURRENT MONITOR
DATA HANDLING BUS
EXPERIMENT CURRENT MONITOR
EXPERIMENT BUS
BAY
APOLLO INTERFACE
VIDEO AMPLIFIER
DEFLECTION CIRCUITS, POWER SUPPLIES, PROTECTIVE CIRCUITS
ATM DISPLAY TUBE AND YOKE ASSY
PROGRAM IND A IND B
PROGRAMMABLE INDICATORS
DISCRETE COMMAND SWITCHES
COMMAND MODULE



.ND SUBSYSTEM
THERMAL CONTROL SUBSYSTEM
CON
HEATER SWITCH MODULE
TEMP PROBE MODULE
EXPERIMENT NO. 1
EXPERIMENT NO. 2
EXPERIMENT NO. 3
EXPERIMENT NO. 4
BORESIGHT OUTPUTS TO VIDEO SELECTOR
PAYLOADS TO DIGITAL SCAN GEN
DATA OUT TO PCM JUNCTION BOX
DATA OUT TO: FRAME COUNTER TELLTALE BUFFER DIGITAL MULTIPLEX
TO BRAKE LOGIC
SPAR COMMAND DECODER
SPAR POWER CONTROL UNIT
UP TO 256 CONTROLLED POWER SOURCES
TEMP REF
AMPLIFIER ASSEMBLY
POWER SWITCH
HEATER NO. 1
TEMP PROBE NO. 1
TYPICAL 8 PLACES
ORDINANCE
TO PROGRAMMABLE SOURCE SELECTORS
COARSE & FINE YAW SUN SENSORS
YAW TORQUE MOTOR
GIMBAL ANGLE POT
FROM SPAR COMMAND DECODER
BAY COMMAND DECODER
BAY POWER CONTROL UNIT
UP TO 256 CONTROLLED POWER SOURCES
DISCRETE SWITCHES
ATTITUDE CONTROL SWITCHES
TELLTALE INDICATORS

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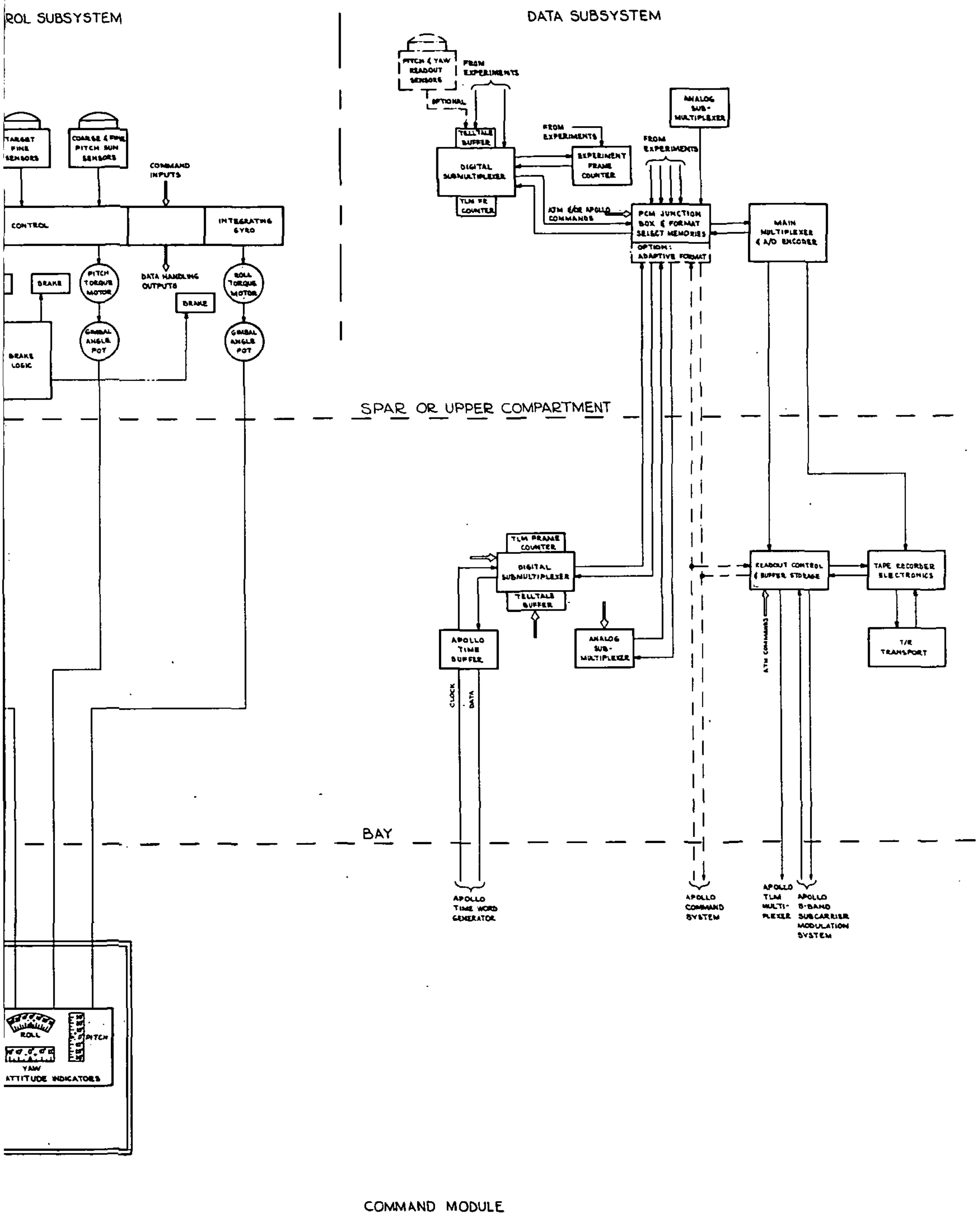

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Fig. 2-2 ATM System Block Diagram

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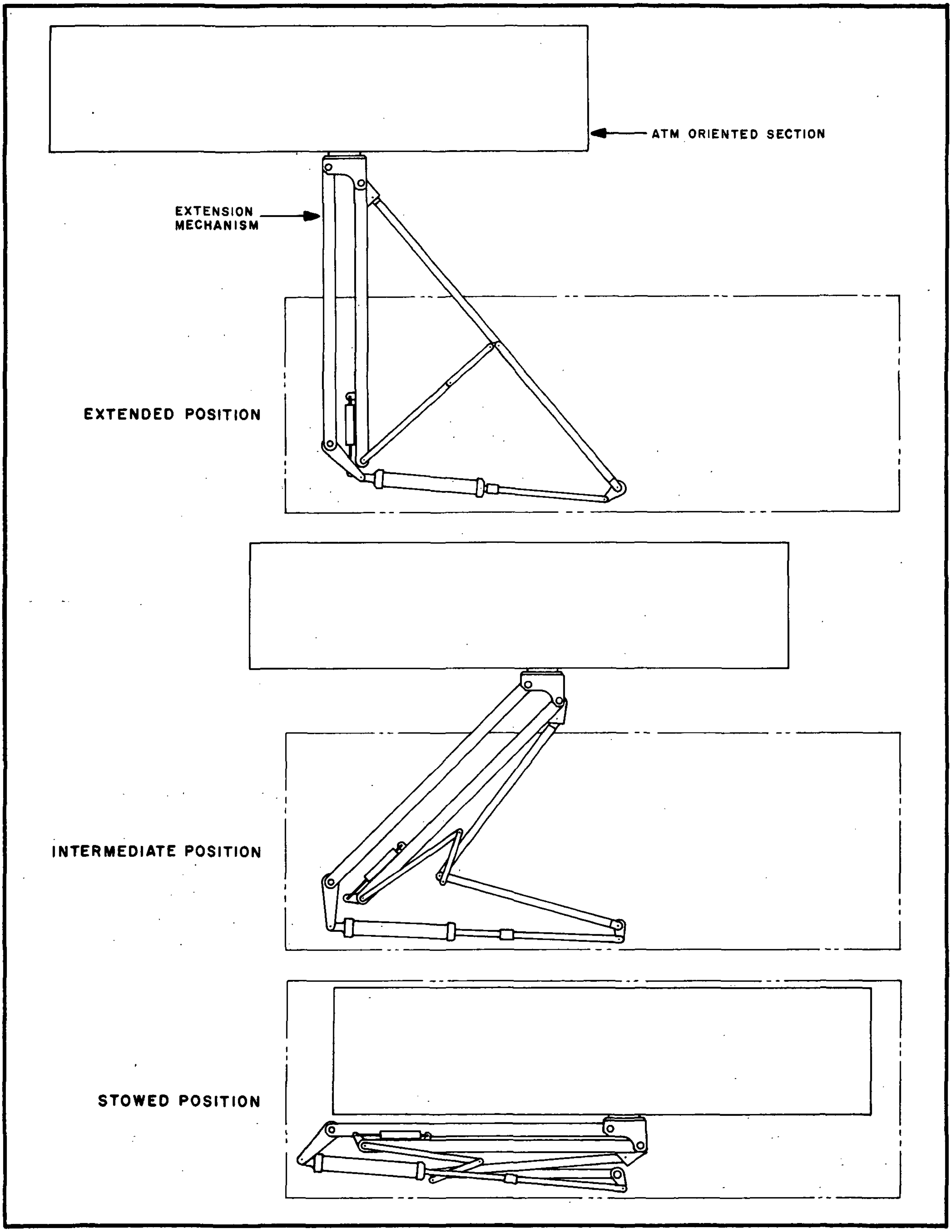
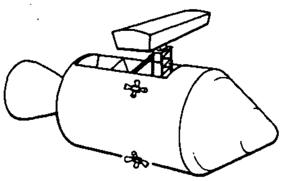


Fig. 2-3 Parallel Extension of ATM Oriented Section



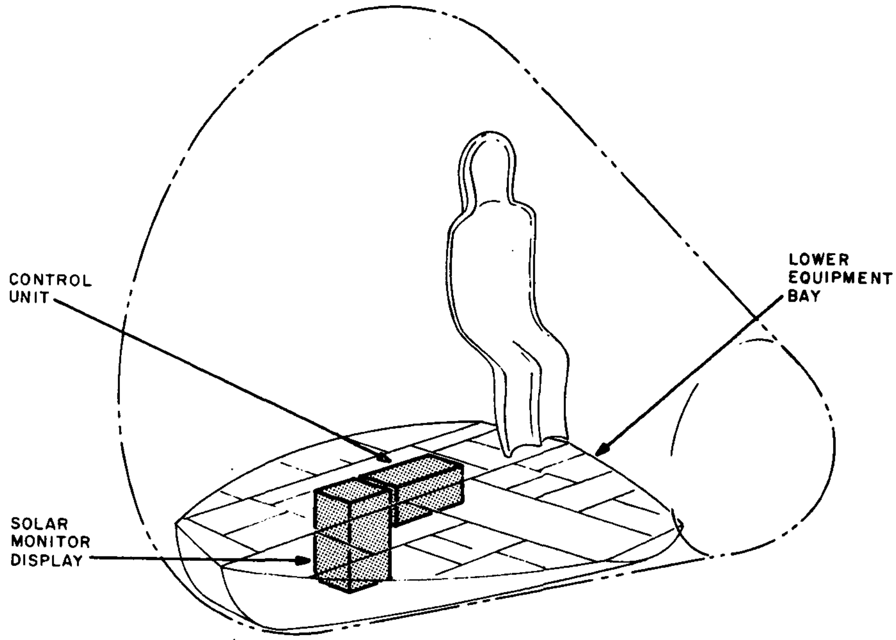


Fig. 2-4 ATM Control Unit-Mounted

The pointing control subsystem automatically points the oriented section at the sun in three axes once the solar target has been located by the astronaut. Under astronaut direction it can be slewed to any position with respect to the CSM within the freedom of the three gimbals to optimize CSM orientation during pointing. The gimbal freedoms are ±170 degrees in yaw, ±30 degrees in pitch, and ±25 degrees in roll, as depicted in Fig. 2-5. In addition, while pointing, the astronaut can offset the oriented section to any position on the solar disk and ATM will hold this alignment. The astronaut maneuvers the oriented section using a "fly to" control stick and the roll control knob of the control unit, which operates both the slewing and offset pointing functions. (See Fig. 2-4.)

Power for the ATM system is obtained from the Apollo power system, augmented by surge batteries within the ATM. Power and electronic signals are distributed throughout the ATM system by wiring harnesses, and by flexible cables across the gimbals. Digital data are recorded on an ATM buffer tape recorder and are played back into the Apollo telemetry for transmission to the ground.

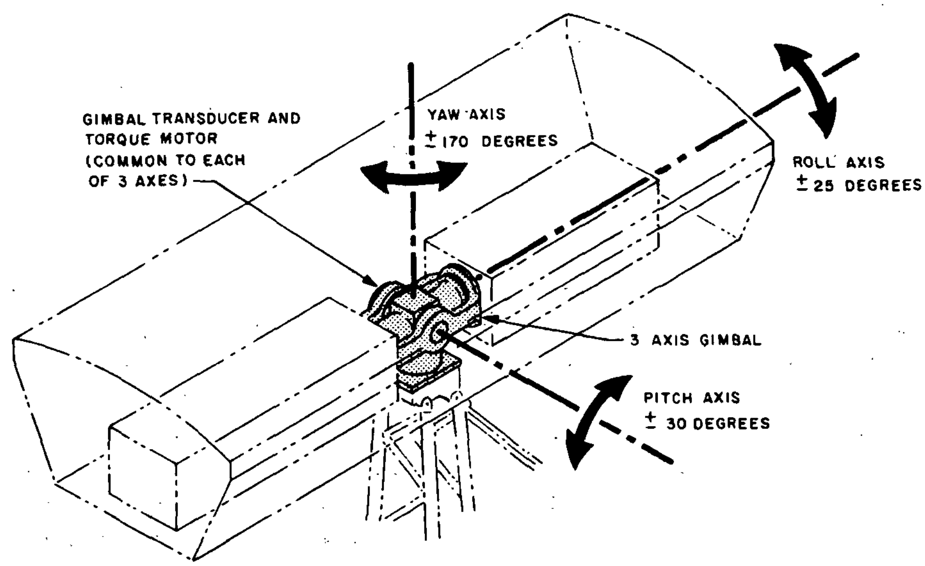


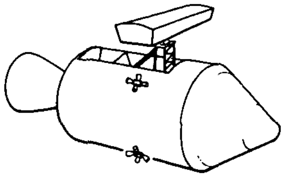
Fig. 2-5 Gimbal Freedom

A solar monitor subsystem is provided within the ATM to image the solar disk in Hα light, which is characteristic of solar activity. A vidicon camera mounted on the spar provides a video image which is displayed to the astronaut observer in the cockpit. (See Fig. 2-4.) This display enables the astronaut observer to monitor solar activity and can be used to display images from the experiments for precise boresight alignment to solar features.

2.2 SUBSYSTEMS DESCRIPTION

For ease of identification, description and control of interfaces, the ATM system is divided into eight subsystems as follows:

- Structural
- Thermal control
- Pointing control
- Solar monitor



- Command
- Data
- Power
- Electrical distribution

To get a better picture of the ATM system concept each of these subsystems and its role in the overall system is discussed independently in this section.

2.2.1 Structural Subsystem

The SM is 154 inches in diameter and 155 inches long (Fig. 2-6). The cross section is divided into six pie-shaped segments (Sectors I through VI) and a central tunnel. The segments vary from 50 degrees to 70 degrees; two segments are 50 degrees, two are 60 degrees, and two are 70 degrees. Sectors II and V contain the SPS oxidizer tanks, Sectors III and VI contain the SPS fuel tanks, and Sector IV houses the fuel cells and associated tanks. ATM will be mounted in Sector I, which is a 50 degree segment that has been allocated for the housing of experimental payloads. The dimensions within Sector I, which control the maximum allowable experiment system volume, are 151 inches length, 50.5 inches height, 17 inches width at the inboard edge of the sector and 62.5 inches width at the outboard edge of the sector. This gives a total volume for ATM accommodation within Sector I of 308,600 cubic inches.

The ATM system provides a maximum volume consistent with Sector I physical dimensions for stowing solar experiments during launch, and then extending these solar experiments, after the Apollo has achieved its desired orbit. The experiments are attached to the support structure (spar) via isolation mounts to minimize the effects of spar temperature changes and distortion on the experiments. One possible cross section arrangement of the representative experiments considered during the study is shown in Fig. 2-7. The structural subsystem provides sufficient strength to react the launch loads imposed upon it by the Apollo and provides support of the oriented section during launch to relieve stresses on the gimbal bearings. The ATM system components are mounted within a structural mounting cradle that provides the structural attachment interface to the SM. The attachment is accomplished by eight attach points. The four primary attach points are located at Sta. 290.66 as already specified for experiment mounting in Sector I, and the remaining four are located in pairs on the forward and aft outboard ends of the cradle.

The extension system is located at the lower forward end of the cradle (Sta. X = 343.75). The system is a parallel-bar linkage stabilized by a strut when extended. The system is extended using a pneumatic cylinder for the driving force and is capable of operation in a one-g field for ground testing. The three-axis gimbal is mounted to a platform on top of the extension structure. Yaw is the inside gimbal, pitch the intermediate gimbal, and roll the outside gimbal. The roll gimbal shaft is structurally mounted to the spar. The allowable travel in these gimbals is ±170 degrees in yaw, ±30 degrees in pitch, and ±25 degrees in roll. No combination of gimbal angles can cause mechanical interference between the oriented section of the CSM.

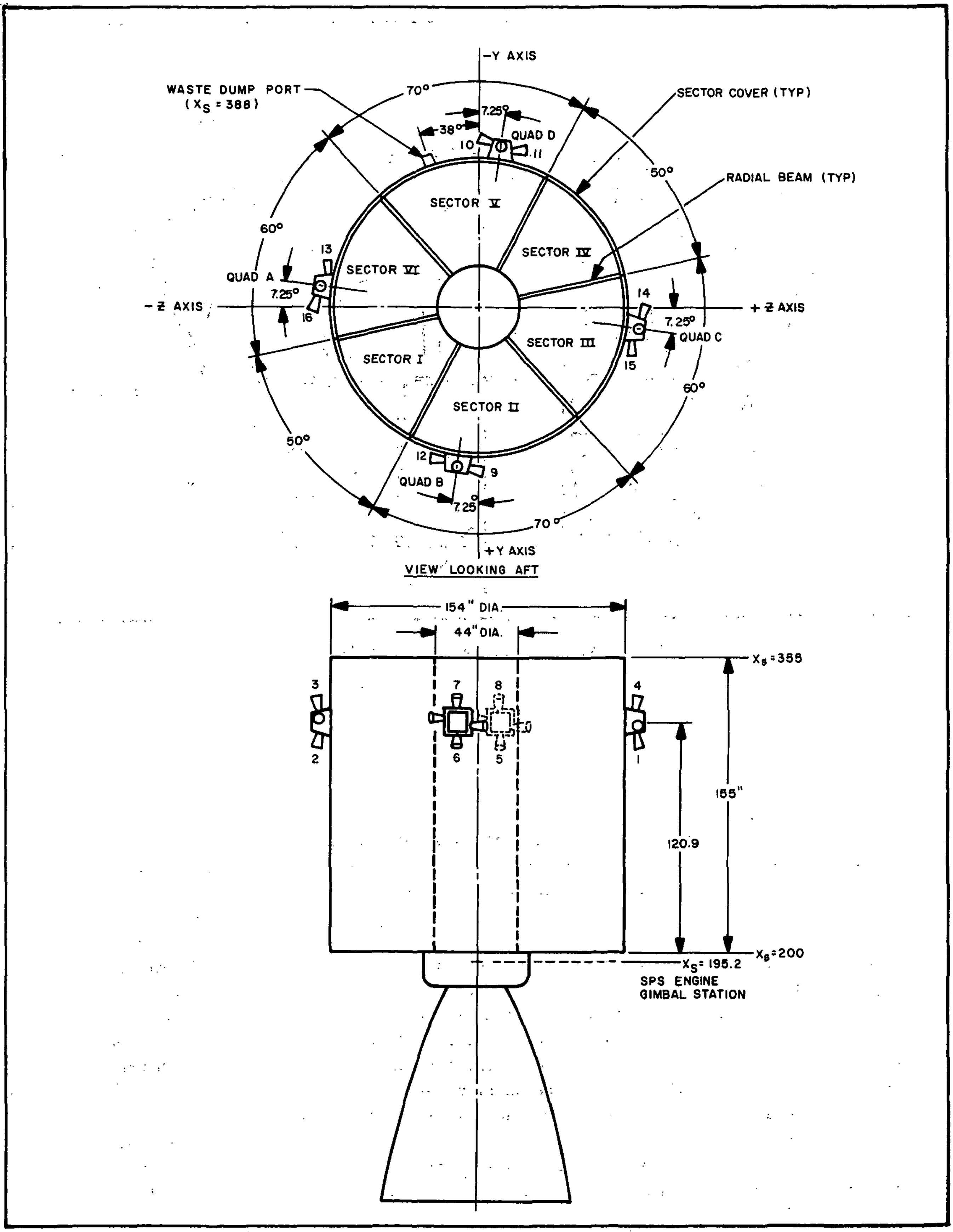
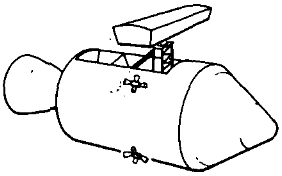


Fig. 2-6 Apollo Service Module Diagram



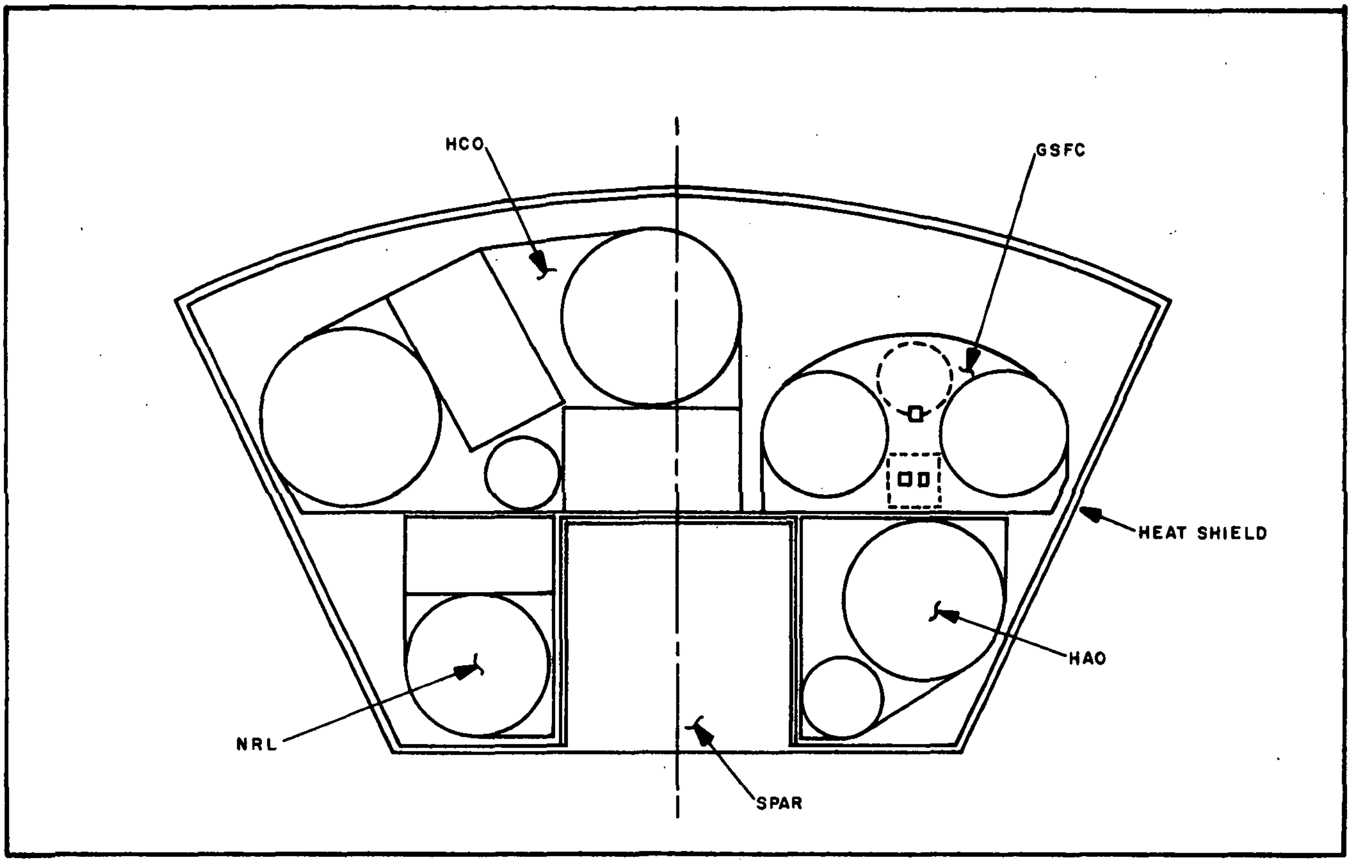


Fig. 2-7 Possible Arrangement of Study Experiments

A heat shield encompasses the oriented section structure and the experiments, and is fitted with aperture covers operable from the CM for protection of the experiment and ATM optics. The maximum length for experiments within the heat shield is 134 inches.

The oriented section will be jettisoned prior to reentry at the end of the mission. This capability is provided to assure a reliable reentry maneuver and to conserve reentry fuel. This consideration stems from the effects of the ATM mass on the CSM c.g. which will shift it out of the SPS thrust line envelope if it is tightened as currently contemplated. With the contemplated limited gimbal travel, ATM would shift the c.g. outside the thrust line envelope even in the stowed condition. These considerations, along with increased system complexity associated with a restow operation and the lack of a strong requirement for it, led to the present recommendation that a restow capability not be provided. Provision has been made, however, in the system concept for the addition of restow capability at a later date should it be required.

The composite c.g. of the CSM and ATM is within the thrust line envelope of the SPS engine for both the stowed and extended position with the present thrust line envelope definition.

The Sector I cover, which is an integral part of the SM structure, will deploy and will have reinforced access ports for prelaunch film loading. Due to the structural role of this cover it is recommended that it be designed and implemented by the SM contractor with ATM interface requirements established by the ATM contractor. The estimated total weight of the ATM system, including 750 lbs for experiment and 600 lbs of batteries, is approximately 2200 lbs. Note that the batteries may not be required if Apollo power can be used for the ATM system.

2.2.2 Thermal Control Subsystem

2.2.2.1 General

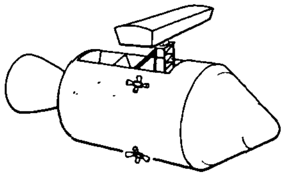
The design concept presented is intended as a feasible approach to a detailed design to meet the requirements for a 14-day mission. The concept takes into account the thermal effects of the CSM, earth albedo, solar radiation, and reradiation from the ATM as a function of the varying relative orientation of the ATM and the CSM. In addition, the varying solar and earth albedo reflections from the CSM have been considered.

In the present design concept thermal control of the oriented section is accomplished by a combined passive/active system, while temperatures in the bay are passively controlled. The major features of the conceptual system are presented in the following paragraphs.

2.2.2.2 Thermal Shield

The oriented section of the ATM is completely enclosed in a thermal shield. The shield consists of two shields in series, the external skin and an inner liner. The outer and inner surfaces of the external skin will be prepared so that the passive thermal properties will provide a mean orbital external skin temperature approximately 30°F below the desired internal equilibrium temperature. The required thermal control coating for the external skin will be determined when the specific mission profile is established. This coating can be applied or changed as late as a few days prior to the mission. The inner liner radiation shield is attached to the inside of the external skin by means of standoffs. This liner has an emissivity of 0.1. The thermal shield is shown in the frontispiece of this volume and is depicted in cross section in Fig. 2-8.

Apertures in the thermal shield are provided for each experiment, the fine control sensors, and the solar monitor subsystem optics. The aperture covers can be commanded open or closed by the Apollo crew during specific data acquisition periods.



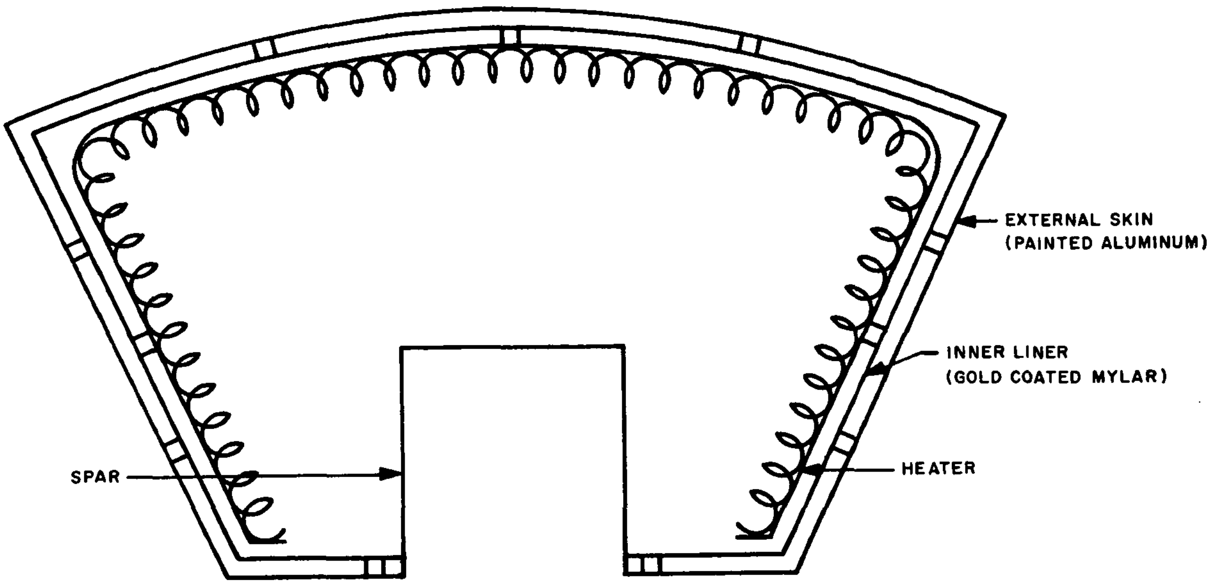


Fig. 2-8 Experiment Thermal Shield Enclosure

2.2.2.3 Active Heating System

Electric heating elements are attached to the inside of the thermal shield liner to radiate into the oriented section. The heaters will be thermostatically controlled to maintain the internal environment within ±5°F of the design temperature. The active heating system consists of eight heating zones. Each zone temperature is controlled by an independent temperature sensor mounted in the environment. The sensor and control system activates the heaters when the temperature falls 5°F below the preset temperature and de-energizes the heaters when the temperatures rise 5°F above the preset temperature. The active heating system will be operated throughout the 14-day mission.

2.2.2.4 Critical Component Thermal Control

Several subsystem and structural components must be maintained at a relatively constant temperature or within some narrow range of absolute temperature. These include the spar, the solar monitor subsystem interference filter and vidicon, the control sun sensors, critical electronics and gyroscopes. Thermoelectric devices and auxiliary filters are used to control the temperature of the interference filter.

Similar means are used to control the sun sensor temperature. Critical electronics, gyroscopes, etc., are to be surface treated to passively maintain the proper temperature.

The spar is a critical component from a thermal control standpoint due to the necessity to limit the dynamic thermal distortion. Although static distortion can be compensated by the astronaut by boresight alignment, dynamic bending of the spar must be limited to considerably less than five arc seconds during a one-minute data acquisition period to meet experiment requirements of the overall system.

Significant heat-producing components inside the spar such as the solar monitor subsystem vidicon are thermally isolated from the spar, and a conductive heat transfer path to the outer heat shield is provided. The surface of the heat shield in the vidicon area is coated with a high emissivity, low-absorptivity coating so that it readily radiates to space the heat transferred from the vidicon. To minimize heat gains (or losses) to the spar, it is shielded from heat sources or sinks within the oriented section. Aluminized mylar is used to shield the spar from surrounding equipment and from equipment (such as the vidicon) enclosed within the spar. All equipment attached to the spar will be mounted on high-thermal-resistance standoffs to limit heat flow to the spar. Manganese standoffs having thermal resistance of 275°F/watt are feasible. (See Appendix B for further analytical considerations.)

2.2.2.5 Bay Components

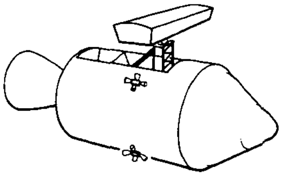
The extension mechanism and subsystem components mounted to the support cradle will be surface treated to maintain passively the proper temperature. The thermal interface between Sector I and the rest of the SM is planned to be maintained by the SM contractor using multilayer insulation.

2.2.3 Pointing Control Subsystem

2.2.3.1 System Considerations

Pointing Stability. The basic function of the pointing control subsystem is to provide precision three-axis attitude control of the oriented section. In Section 3.1 and 3.3, it is shown that while there is no requirement for precise absolute pointing, there is a requirement for pointing stability of a few seconds of arc per minute. That is, once the ATM is pointed at a solar target, it must not drift away at a greater rate than this.

An additional requirement is that of roll rate stability. This rotation would be whatever the component of the CSM rate happens to be in the ATM roll axis. For reasons of image smearing on the photographic film, and of roll rate coupling into pitch and yaw during offset, the ATM inertial roll rate must not exceed four arc seconds per second. (See Sections 3 and 6.)



The subject of manual CSM control was discussed with members of the MSC Astronaut Office* and it was predicted that the limit cycle (or residual) rates could be manually reduced below the 0.05 deg/sec (180 arc sec per sec) capability of the G & NC control, but probably not below 0.01 deg/sec (36 arc sec per sec). Therefore, even the most optimistic residual CSM rates will not satisfy the requirements and a roll axis rate control is necessary.

Operating Modes. The requirements for the various operational modes of the pointing control subsystem stem from a variety of considerations. The basic operating mode is automatic fine control which provides the high stability required by the experiments. Fully automatic operation is essential in this mode to cope with transient disturbances such as those arising from man motion (Section 6).

The other extreme is provided by a manual slewing, or coarse positioning mode in which the astronaut may command the ATM to slew in any direction regardless of the position of the sun. A coarse sun sensor readout is provided to allow the astronaut to use this mode to position the ATM accurately enough to allow the automatic fine control mode to take over.

In some instances, notably when the CSM attitude must be changed, it is desirable to keep the ATM locked on the sun during the maneuver. This not only saves power, but also avoids a possibly severe thermal transient. While the automatic fine control mode might be designed to cope with the resulting high gimbal rates, it has been found necessary to shield the fine sun sensors, which control this mode during this time, to avoid contamination from RCS engine exhaust products. (See Section 3.5.) As the coarse sun sensors are not as sensitive to contamination, an automatic coarse pointing mode has been provided to maintain the sun direction. This also allows a fully automatic acquisition operation once the ATM is within range of the coarse sun sensors, thereby reducing the accuracy to which the crew must position the CSM prior to sun acquisition.

In order to point experiments similar to the HCO and NRL study experiments at specific regions on or near the sun, it is necessary to add an offset capability to the automatic fine control mode. This is accomplished by adding variable biases to the fine sun sensor error signals. Precise absolute positioning is not required since the necessary information for adjustment is available to the astronaut from a boresight indicator in the experiment being pointed.

Restow. As discussed in Section 2.2.1, there is no present requirement to be able to restow the oriented section in the bay and this capability is therefore not provided. In the event that this assumption must be changed, it would be relatively easy to provide circuitry to feed the gimbal angle pickoff signals as errors to the position servos while in the automatic coarse pointing mode. All gimbal angles would then be driven to zero automatically after which the restow maneuver could be accomplished.

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*R. Chaffee, Dr. J. Kerwin, and Dr. F. C. Michel.

2.2.3.2 Subsystem Description

A brief description of the pointing control subsystem is presented here which is consistent with the system considerations just presented.

A DC torque motor is the gimbal torque source in each axis. Sun sensors, both coarse and fine, are used as error sensors in the automatic control mode of the pitch and yaw axes. Sensing for the roll axis automatic mode is obtained from a rate integrating gyro. Necessary amplification, stability networks, and mode switching interconnect the sensors and commands to the torque motors.

The pointing control subsystem is operated by the astronaut from the control unit in the CM via the command subsystem. A rate-limited slewing capability of six degrees per second is provided in each axis so that the operator can manually position the spar to any desired attitude with respect to the CSM within the allowable gimbal travel. Transducers on each gimbal generate signals that drive the gimbal angle indicators on the control unit which show the attitude of the oriented section relative to the CSM.

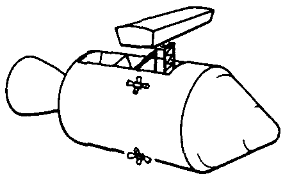
The automatic pitch-yaw coarse pointing mode is controlled by the coarse sun sensors, which are unobstructed by any aperture cover. The pitch-yaw coarse pointing mode can be used for coarse acquisition and for control of the oriented section during periods when the spar aperture cover is closed (covers the fine sun sensors), such as during the patrol mode.

The pitch-yaw fine pointing mode is controlled by the fine sun sensors that are located behind an aperture cover. A vernier positioning capability is provided that allows the operator to point an experiment line of sight to any selected spot within a 40-arc minute square centered on the sun. The pitch and yaw offset signals, along with necessary subsystem housekeeping data, are sent to the data subsystem for recording. The recorded offset signals will assist the experimenter's post flight data reduction in determining the area of the solar disc under observation at the time of data acquisition.

2.2.3.3 Solar Monitor Subsystem

The need for a solar monitor is a direct result of the basic ATM premise which establishes the astronaut as observer-in-charge of the ATM system. The astronaut must be able to examine the sun's disk in monochromatic light in order to find significant chromospheric features. He must then be in a position to align the optical axis of any one of the experiments to a pertinent feature on the disk with the precision required by that particular experiment.

Seated in the CM the observer is physically isolated from the ATM and its experiment complement. Therefore, a remote sensing system is needed to enable him to monitor the sun's image with the same facility as if he were actually able to sight along the axis of the experiment telescope.



The solar monitor subsystem presents a video image of the solar disk to the observer in the CM. The solar monitor subsystem also provides the capability for video sensor inputs from the experiments for presentation on the CM display. The astronaut is able to look at the display while he manipulates the "fly to" control stick to offset the axis of the oriented section to intersect off-axis features, out to the edge of the disk. The solar monitor as a whole is used primarily in the patrol mode of observation, with the spar pointing at the center of the sun and the astronaut viewing the solar activity over the entire disk and out beyond the limb. Should an event of particular interest be observed, the astronaut will command the oriented section to offset from the sun's center in such a way as to bring the important activity into the center of the field. At this time, he will switch the video input to his display from the ATM solar monitor to the specialized experiment, narrow-field sensor which has been accurately boresighted with the optical axis of the particular experiment to be activated.

The spectral response, field of view, and spatial resolution of each experiment's boresight sensor will be a function of the requirements of that particular experiment to insure that the experiment axis can be aligned with the salient feature on the disk with the required precision. A reticle pattern in each camera tube appears in the display to give the astronaut a geometric reference which helps him point the experiments at the specific detail of interest.

Some experiments may present their boresight readout in analog form from null detectors in each axis. These signals can be displayed to the observer via the program indicators located on the control unit. (See Section 2.2.4.)

The solar monitor subsystem concept is presented in detail in Section 7.

2.2.4 Command Subsystem

The command subsystem is the interface between the astronaut and the ATM. It is possible for the astronaut to erect the ATM, control the attitude of the ATM, control experiment modes of operation, and monitor the status of the ATM and experiments. This is accomplished by controls and indicators on a control unit (Fig. 2-9) located in the CM.

There are two types of commands: namely, hardwire commands, and encoded commands. Hardwire commands are used where crew safety is involved, or where undue complexity would result if encoded commands were used. The hardwire commands are ATM power control, ordnance control and ATM attitude control. ATM pitch and yaw are controlled with the "fly to" control stick on the control unit. Roll is controlled by the roll control knob. Supplemented with selector buttons, the "fly to" control stick produces 25 discrete commands which result in different combinations of pitch and yaw direction and rate. In addition to providing pitch and yaw signals to ATM, the "fly to" control stick can produce commands for pitch and yaw motion of elements in the experiments, e.g., the internal occulting disk in the HAO study experiment.

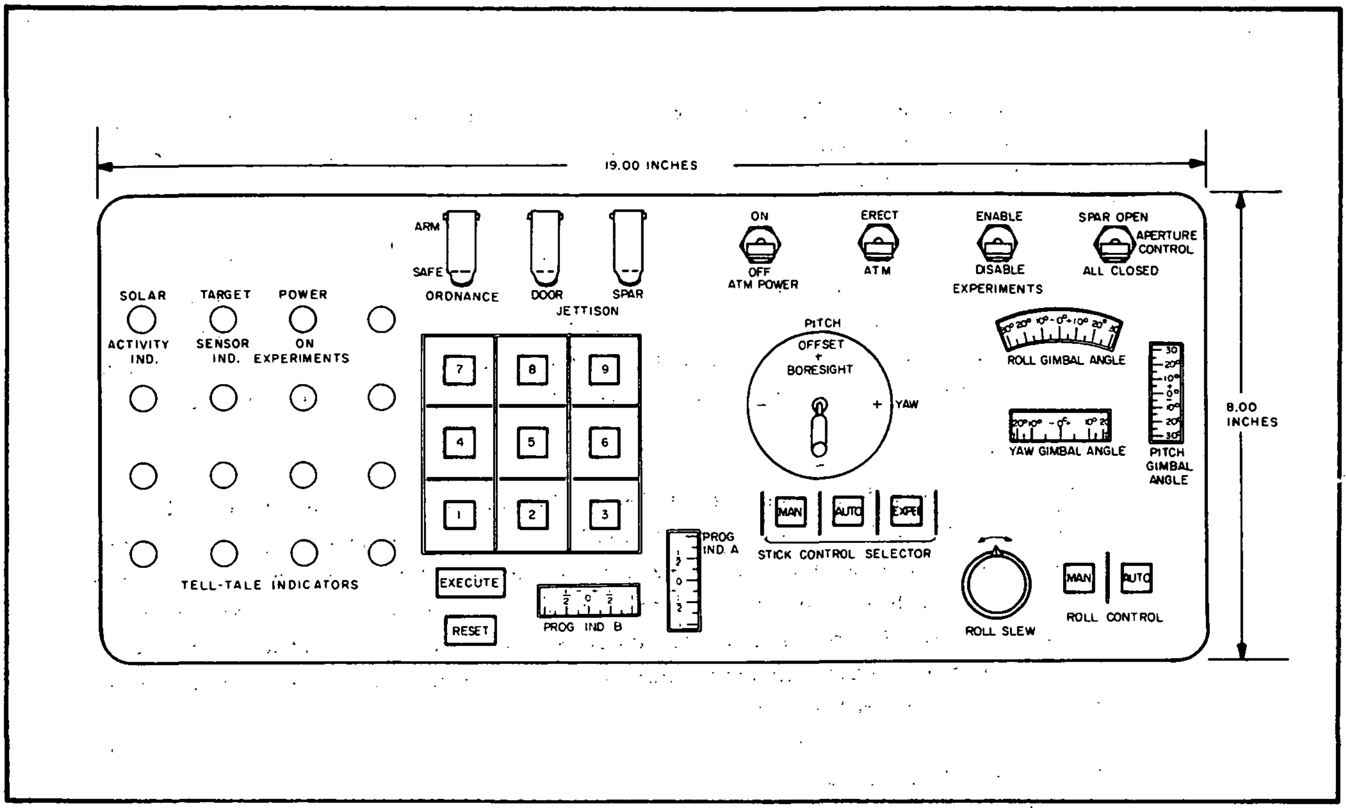
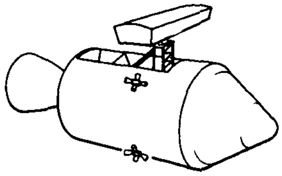


Fig. 2-9 ATM Control Panel Layout

There are two types of encoded commands: commands operated by discrete switches, and commands operated by a nine-button keyboard. (See Fig. 2-9.) The encoding concept is used to limit the number of conductors between the CM and SM. Commands which either are frequently used or require quick reaction are operated by a single switch. This switch automatically formats a nine-bit word into the command encoder. Less frequently used commands are formatted by punching the appropriate buttons on the nine-button keyboard.

The total number of encoded commands is 512. One half of these are available for connection of bay and oriented section analog sources to indicators on the control unit. The remaining 256 encoded commands are available for operating experiments and other ATM subsystems.

There are three types of indicators on the control unit. These are: (1) attitude indicators for continuous monitoring of the ATM pitch, yaw, and roll gimbal angles; (2) program indicators for monitoring two analog sources at a time; and (3) flag or light indicators for monitoring status sensors in the oriented section. The program source/indicator concept is used again to limit the number of conductors between the CM and SM. The 16 digital indicators are flags (or lights, if important enough) which will indicate such conditions as "on target" and "experiment power on".



Each indicator is connected to its respective sensor with a conductor. If the need should arise to reduce the number of conductors further, it is possible to multiplex these digital indicators.

There is a command decoder, power control unit and program source selector in the bay and in the oriented section. The power control unit is used to amplify low energy pulses from the command decoder so that command relays can be energized. The power control unit also provides relay logic. These three units are placed in both the bay and in the oriented section to reduce the number of conductors across the gimbals.

The command system as described herein uses 58 conductors between the CM and SM, and 37 conductors across the gimbals. These conductors include power, power ground, and signal ground, which can be used for the remaining ATM subsystems as well as the command subsystem. A further reduction in the number of conductors is possible if the need should arise. Currently, 10 conductors are used for attitude control, which could be accomplished with encoded commands. However, this would significantly complicate the "fly to" control stick.

2.2.5 Data Subsystem

2.2.5.1 Systems Considerations

The data subsystem is responsible for collecting all digital and analog data from the ATM and its experiments, and processing it for storage or transmission to the earth via the Apollo telemetry. Since most of these data are generated by photoelectric instruments (HCO typical), two basic channel capacities have been designed into the subsystem. These are 120 bits/second and 4800 bits/second, where the latter rate is needed only when the HCO study experiment is operating.

The most important system decision affecting the design of this subsystem is the question of the possible use of the Apollo telemetry. The alternative is total data storage and return of tape magazines. For these reasons, it is the recommendation of this study that a buffer tape recorder be used to gather all data, and that it be read out over the Apollo telemetry no less than every two orbits. This tape would not be recovered.

2.2.5.2 Design Concept

In the design concept developed during the study, all data except high rate primary data from an experiment such as the HCO study experiment are collected in a set of digital and analog submultiplexers. These in turn are serially read out and combined in a PCM junction box, as are the high rate experiment data. This box offers the choice of two fixed data formats, selected on command, which correspond to the two data rates mentioned above. The formatted data is then sent to the main multiplexer where it is combined in a single serial data channel. The main multiplexer also supplies clock pulses to synchronize the entire subsystem and experiment users.

A two-speed tape recorder is provided to store the output of the main multiplexer, and has a capacity to store two near-earth orbits of data with the higher rate used for experiment data acquisition during an activity mode. A readout control and buffer storage are included in the subsystem to provide compatibility between the tape recorder and the Apollo telemetry multiplexer. The recorder can be read out in seven minutes with this equipment.

A complete description of the subsystem concept as well as current data requirements are given in Section 9.

2.2.6 Power and Electrical Distribution Subsystems

2.2.6.1 Power Subsystem

This section presents the system conditions as they affect the design of the power subsystem. For detailed equipment descriptions and power summaries, see Section 10.

The overall ATM system power requirement for a 14-day mission is 30 kw-hr at 28 VDC. About 10 kw-hr of energy is available for experiment use from the primary Apollo fuel cell system.* The voltages and regulation are quite suitable for ATM requirements.

While the energy balance available from the Apollo does not match the ATM requirement, the great advantages of weight saving, cost, and simplicity resulting from the use of Apollo power make this approach very attractive. Thus, the main study recommendation for an ATM power source is that the Apollo energy be increased to meet the ATM requirements. This can be accomplished by either increasing the Apollo fuel cell reactants, or shortening the mission the few hours necessary to make available the additional 20 kw-hr required. In the event that this latter approach is not acceptable, it would be possible to allot space within the ATM bay for the storage of additional fuel cell reactants.

If neither of the above approaches is acceptable, then it is the recommendation of the study that a silver-zinc battery be included as a part of the power subsystem. It is also recommended that further consideration be given to an autonomous fuel cell as a possible replacement for the above battery.

While the primary recommendation is to use Apollo power, the description of the ATM power subsystem in this report includes a battery pack located in the bay in order to show that it can be accommodated if necessary. Additional subsystem equipment includes a backup battery for ATM ordnance activation, regulators, power controls, protection circuitry, DC convertors and convertors.

*NPC 500-9, "Apollo Experiment Guide", 6-15-65, NASA Office of Manned Space Flight, Section 7.2.1.

2.2.6.2 Electrical Distribution Subsystem

The primary elements of the ATM electrical distribution subsystem are the wire harnesses in the bay and in the oriented section and the interface with the CM/SM umbilical. The main consideration in the electrical distribution subsystem is the transmission of electrical signals and power across the gimbals with minimum static and dynamic torques applied to the pointing control subsystem. The recommended concept utilizes flexprint leads for signals and power, and light weight coax for video transmissions.

2.3 OPERATIONS

The ATM system and experiments are to be operated in orbit by the astronaut crew, with support from scientists on the ground. Many of the ATM operations are automatic and require no crew effort other than initiation and an occasional check. Other ATM operations are crew controlled and require direct crew participation. The operations associated with a particular mission are discussed in this section in four categories: (1) prelaunch operations, (2) ATM orbital operation, (3) ATM experiment orbital operation, and (4) post-mission operations.

2.3.1 Prelaunch Operations

The major prelaunch effort to support the orbital mission is the loading of film magazines. One complete set of magazines will be loaded in the experiment cameras two days prior to lift off or at the last available access time. This loading will be accomplished through access ports in the SM Sector I cover. The second set of film magazines will be stowed in the designated CM compartments.

Extensive astronaut training on the ATM system operations will be conducted prior to the mission. A final astronaut briefing by the cognizant experimenters will be required within the last few days prior to liftoff. This briefing will enable the scientists to describe current solar activity, set forth specific regions of interest, and present last minute instructions for activity mode operation. The activity condition of the sun will be determined from ground based solar observatories.

2.3.2 ATM System Orbital Operations

One crew member will be the primary ATM operator, but the other two crew members will also operate the ATM system to some extent during the 14-day mission. The observer will operate ATM by means of the control unit (Fig. 2-9) mounted in the CM. The major ATM operational functions are presented in the following sections.

2.3.2.1 Extension of the Oriented Section

The ATM oriented section is to remain stowed during the first 24 hours of orbit. This period provides time for the station keeping activities of the crew and protects the

ATM from the initial Apollo outgassing contamination. After this initial stowed period, the observer will jettison the SM Sector I cover, and activate the extension mechanism.

The extension mechanism automatically erects the oriented section and locks the structure in place. An additional outgassing period of several hours is recommended following extension to allow further outgassing of trapped particles in the ATM system. The active thermal control subsystem will be activated at this time.

2.3.2.2 Solar Acquisition

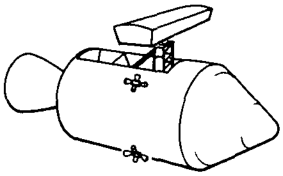
The astronaut observer activates the ATM pointing control subsystem in the manual mode. This enables him, with the "fly to" control stick, to slew the ATM oriented section to the desired position with respect to the CSM in all three axes within the gimbal freedom. Angles are indicated by the gimbal angle indicators on the control unit. This desired position will most likely be at the center of the pitch and roll gimbals, and will be established for yaw by the required thermal balance of the CSM or the position that requires the minimum fuel consumption for maneuver of the CSM to bring the sun into the ATM acquisition range.

Once the ATM gimbal angles are set, the observer releases the "fly to" control stick which automatically activates the gimbal brakes. The observer maneuvers the CSM with the RCS thrusters until the sun is within the field of view of the ATM coarse target sensor. This sensor has a square field of view of 40 degrees on a side, and when illuminated activates a sun acquisition indicator on the control panel. Coarse angular relationships between ATM and the solar vector will be indicated in pitch and yaw on the control panel program indicators. Using these indications the observer can continue to maneuver the CSM until he brings ATM as close to the nominal target as he desires. At any time after the sun is within the coarse target indicator field of view (40 deg square) the observer may activate the coarse acquisition system which will release the gimbal brakes and will automatically point the oriented section to the center of the sun to within ±2 degrees. The amount of displacement of the gimbals from their original locked position after coarse acquisition will depend upon how closely the optical axes were aligned to the solar vector prior to coarse acquisition. With ATM pointing in the coarse mode, the CSM will be further maneuvered to again center the gimbals if required and reduce the residual rates in all three axes to a minimum value to maximize the observing period.

2.3.2.3 Fine Pointing

After minimizing the CSM rates while pointing in the coarse mode, the observer activates fine pointing which automatically points the ATM to the center of the sun within approximately ±1 arc minute.

The fine pointing command automatically opens an aperture cover over the fine pointing sensors allowing them to take over. After a few seconds of fine pointing the coarse



sensors are switched out of the control loop. If the observer closes this aperture cover, the pointing control automatically returns to coarse pointing. When the aperture cover is reopened the fine pointing is restored to about the same alignment on the sun.

2.3.2.4 Offset Alignment

When the ATM is pointing in the fine mode, the observer can maneuver the ATM through small offset angles to any point on the solar disk. This maneuvering is performed with the "fly to" control stick and covers a range of ±20 arc minutes from the center of the solar disk. The observer can offset from the center to any point on the disk that he can identify on the solar monitor display at a rate of 40 arc seconds per second. It will take about 24 seconds for the observer to offset from the center of the solar disk to the limb.

Any offset angle put in the system will remain in effect until manually removed by the observer. The alignment will hold to any offset position to within ±5 arc seconds for at least one minute, and will hold to within ±1 arc minute for periods up to 20 minutes or more. Absolute alignment of an experiment optical axis to a specific target area on the sun is obtained by individual boresight signals from that experiment, as discussed in a later paragraph. The offset alignment pointing is depicted in Fig. 2-10.

2.3.2.5 ATM Jettison

At the completion of the mission, the astronaut crew will jettison the ATM oriented section prior to reentry. This is performed by separate covered "arm" and "execute" switches on the ATM control unit panel. This action causes the oriented section to separate from the top of the extension mechanism.

2.3.3 ATM Experiment Orbital Operation

Similar to the operation of the ATM system, the astronaut crew will perform all ATM experiment operations via the control unit. The various functional operations to be performed are discussed in the following paragraphs.

2.3.3.1 Target Selection

The scientific observing to be performed will have been established during the premission training by the cognizant experimenter. It includes prescheduled standard mode observing of specific solar features, and nonscheduled activity mode observing of active solar phenomena. The astronaut observer determines the experiment mode of operation, and the location of the solar features from the ATM solar monitor subsystem video display in the CM.

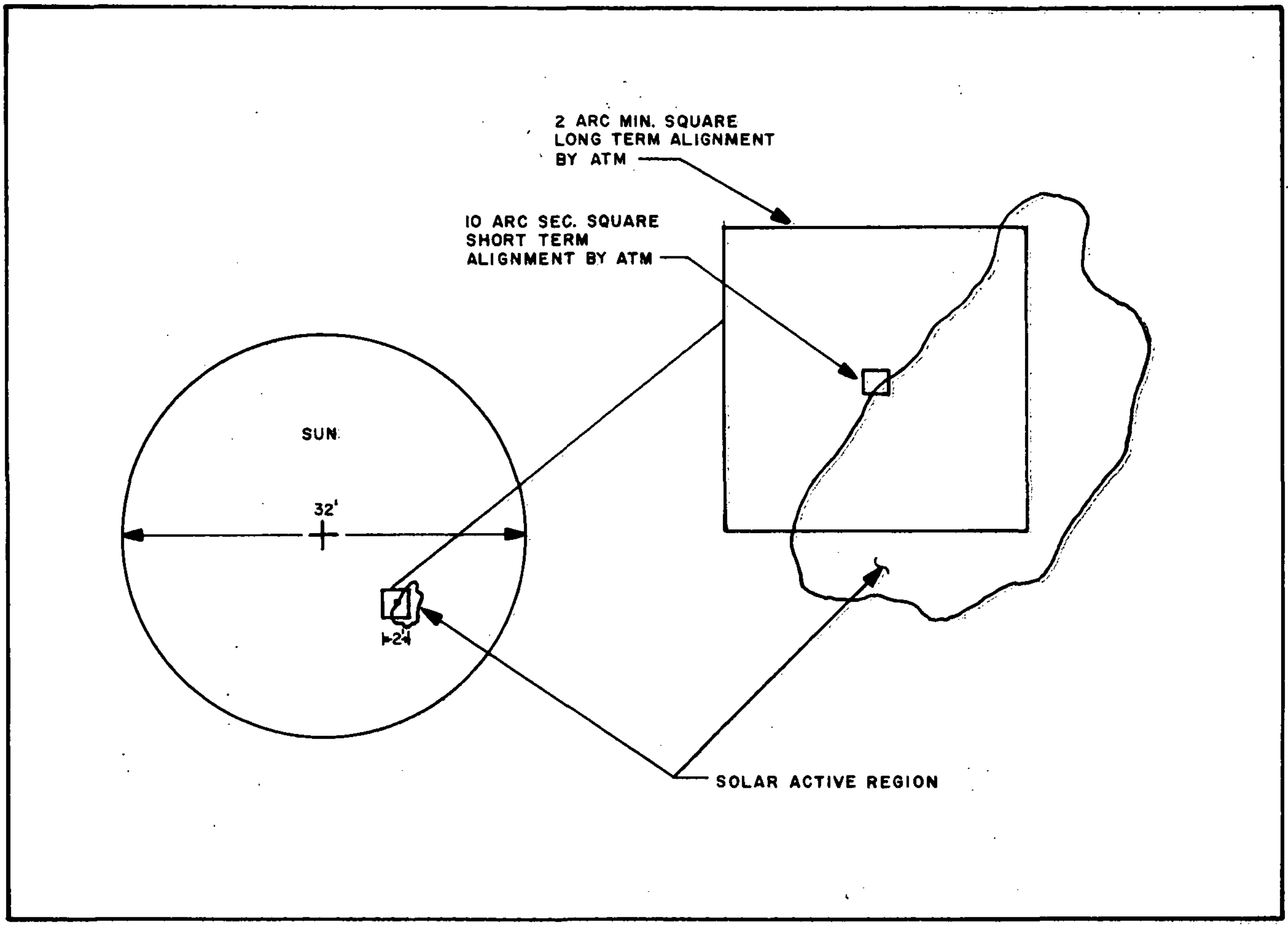


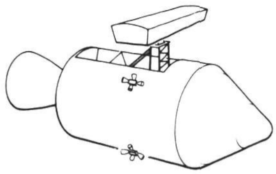
Fig. 2-10 ATM Offset Pointing

This display is a video image of the entire solar disk, presented in the light of the solar Hα (6563Å) emission. This emission is distinctively characteristic of major solar features, and will also be used on the ground by cognizant experimenters during the mission. The Hα emission will also be used during the astronaut training program, so that they are familiar with the features to be observed in orbit. A typical ground observation photograph in Hα light is shown in Fig. 2-11.

This solar display will be used by the astronaut observer to select the region for data acquisition. Between scheduled standard mode observations, the observer will use the monitor in a patrol mode to locate potential active phenomena.

2.3.3.2 Instrument Boresighting

The astronaut observer will boresight each experiment to the desired alignment on the solar disk, based on the information displayed from the solar monitor subsystem.



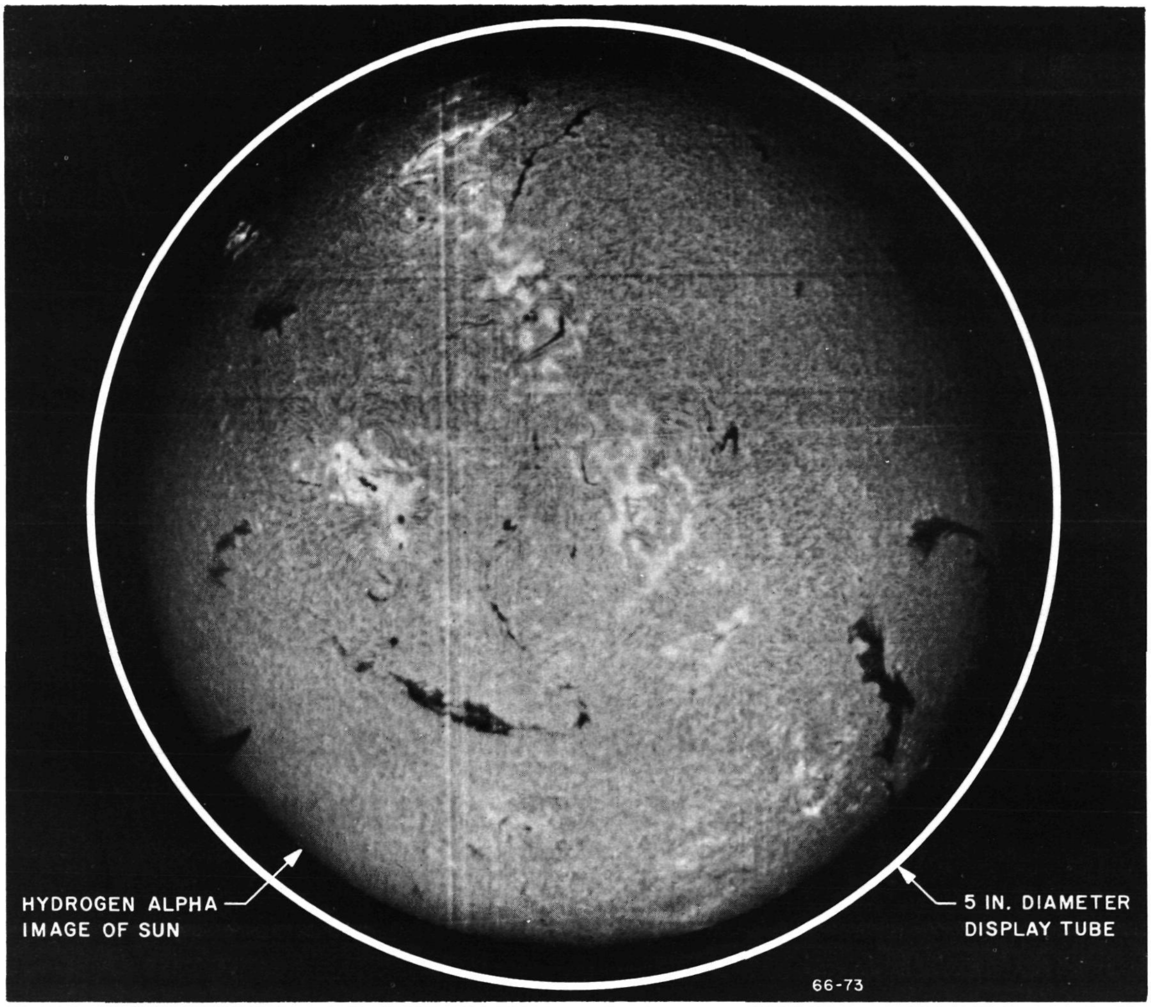


Fig. 2-11 Solar Hydrogen Alpha Display in Solar Monitor

Experiments requiring higher resolution and accuracy than available from the solar monitor display will provide separate boresight signals. Typical of these will be video signals from smaller field of view vidicon telescopes located within the experiment. These signals will be keyed into the ATM solar monitor display which can be selected by the astronaut. A field of view of ±5 arc minutes will provide sufficient resolution for the experiments. Such a view derived from a photograph similar to that of Fig. 2-11 would appear as shown in Fig. 2-12.

The observer will offset the ATM to a coarse alignment to the desired disk region, using the ATM display, and switch to the experiment display for the fine alignment.

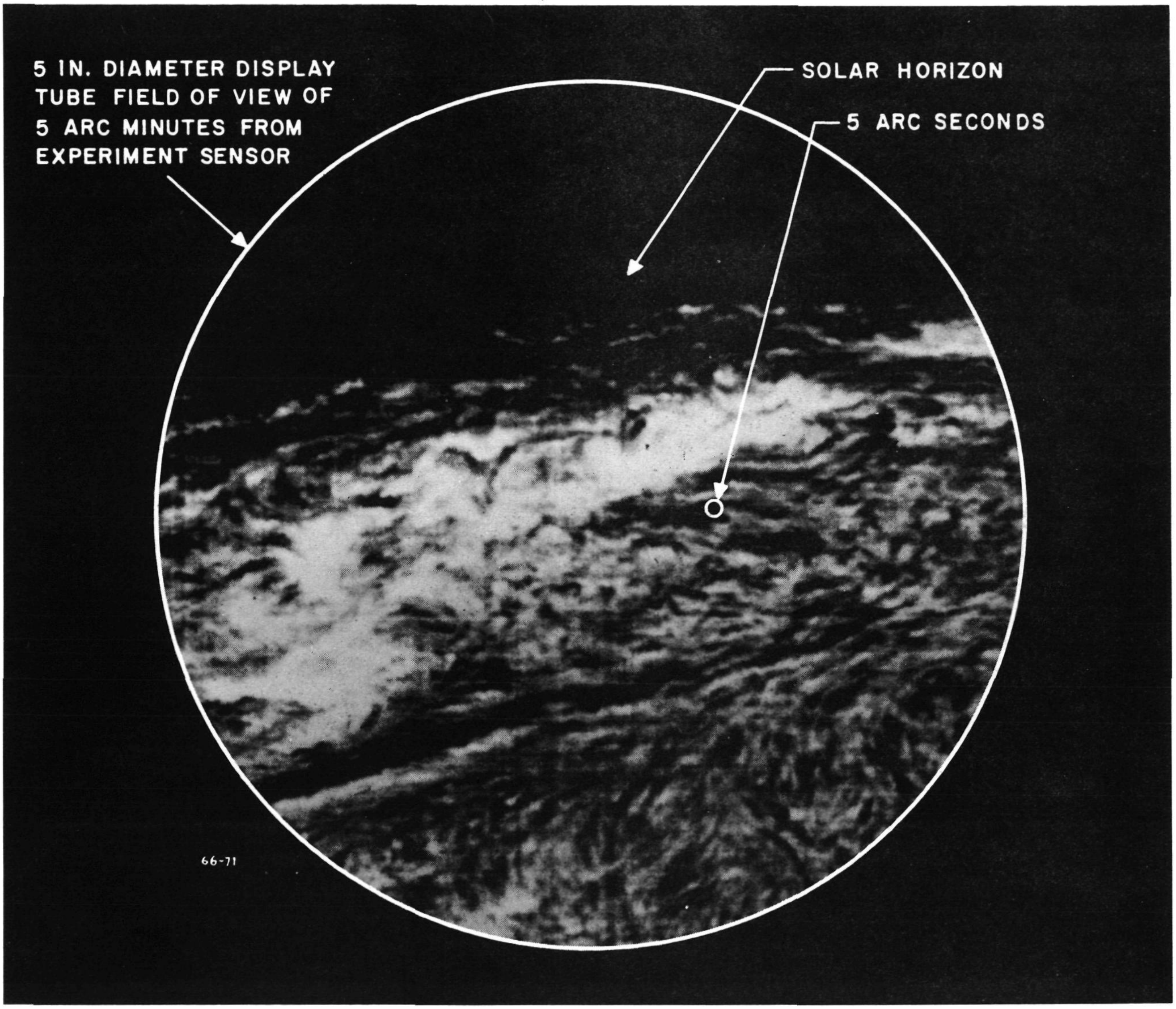
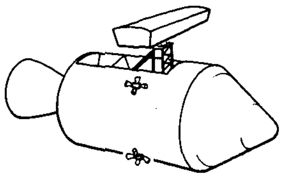


Fig. 2-12 Five Arc Minute Solar Hydrogen Alpha Display in Solar Monitor

During data acquisition, the observer can monitor the relative pointing on this experiment display and make trimming adjustments if the instrument optical axis drifts off the desired point. Similarly, if the data acquisition period is long, the observer may have to adjust the ATM gimbal angles via CSM maneuver to avoid hitting the gimbal stops. If this becomes imminent, the observer will temporarily stop the data acquisition, perform the CSM maneuver, and restart the data acquisition, retrimming the boresight alignment as necessary.



2.3.3.3 Data Acquisition

When the instrument has been boresight aligned to the desired solar disk region, the observer initiates the data acquisition. The data acquisition function will vary between experiments, and may be automatic once started or may require that the observer stop the acquisition.

The duration of data acquisition time remaining after CSM quieting will depend upon several factors as depicted in Table 2-1. Table 2-1 shows the estimated times for the various operations associated with the operation of ATM for two cases: (1) "First sequence of spacecraft day", which includes all the initial target acquisition activities, and (2) "CSM drift correction", which shows the operations and times required to recenter the ATM gimbals and get back to data acquisition after the CSM has drifted into a gimbal stop. The estimates presented assume use of the same experiment looking at the same target. These times will increase slightly if a different experiment or different target is used due to the new offset and boresight times, but will still be less than the first sequence times. It will be noted from Table 2-1 that the time available for data acquisition is greatly increased if the CSM residual rates can be reduced to 0.02 deg/sec prior to fine pointing. Recent conversations with NASA personnel have indicated that this rate may be readily achievable. All times indicated in Table 2-1 were established on the basis of the time required for the CSM to drift into the gimbal with the smallest range (±25 deg in roll) for the two cases: 0.05 deg/sec and 0.02 deg/sec drift rates. Time counting is started at the completion of the last CSM quieting maneuver immediately prior to opening the fine sun sensor aperture. The additional two minutes of time available on sequences after the first sequence of the spacecraft day are the result of less time being required to boresight and no time required to offset.

It is anticipated that after a few acquisition sequences the astronauts will become more proficient, and operating times can be reduced. It should also be emphasized that these times are preliminary estimates only, and will most certainly change as the astronaut training program progresses and more definition of factors such as RCS exhaust gas diffusion rates are established.

Based on the timing analysis of Table 2-1, periods of approximately 18 minutes of uninterrupted data acquisition appear possible depending upon the quieting of the CSM and the astronaut operation times. It is likely that a few small boresight trim adjustments will have to be made during the 18-minute acquisition period. In most cases, this can be done without interrupting the data sequence by trimming during an exposure, or between one minute exposures as the adjustments will be small and will take only a few seconds to accomplish. This will, of course, be dependent upon the exposure characteristics of the experiment being operated.

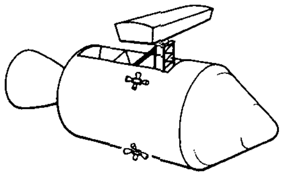
Note from the times required for the initial offset and boresight maneuvers that several minutes may be required from the sighting of an active region until the data acquisition can be initiated. It is expected that with increasing experience during the mission, the observer could perform these maneuvers more rapidly and initiate data acquisition earlier in the activity period.

Table 2-1
ATM SEQUENCE TIMING

	Est. Time Required To Perform Function (min)	Remaining Time Available Until Gimbal Pickup at 0.05 deg/sec CSM Drift (min)	Remaining Time Available Until Gimbal Pickup at 0.02 deg/sec CSM Drift (min)
(1) First Sequence of Spacecraft Day:			
Function			
a. Set and lock ATM gimbals	1		
b. CSM maneuver to point ATM at sun	5		
c. ATM automatic coarse sun acquisition	<0.1		
d. CSM maneuver to recenter gimbals and quiet CSM	3	8.3	20.8
e. RCS exhaust dispersion wait period	2	6.3	18.8
f. ATM automatic fine sun acquisition	<0.1	6.2	18.7
g. Offset (max. travel to sun limb)	0.5	5.7	18.2
h. Boresight	2.0	3.7	16.2
Total first sequence acquisition time	13.6		
Remaining data acquisition time in first sequence		3.7	16.2
(2) CSM Drift Correction:			
a. CSM maneuver to recenter gimbals and quiet CSM	3.0	8.3	20.8
b. Exhaust dispersion wait period	2.0	6.3	18.8
c. ATM automatic fine sun acquisition	<0.1	6.2	18.7
d. Boresight	0.5	5.7	18.2
Total drift correction time (experiment sequence interrupted)	5.6		
Remaining data acquisition time		5.7	18.2

The observer will maintain surveillance over the CSM waste dump so that he can close all ATM aperture covers prior to and during the dump. This may occur during a data acquisition and cause the loss of some data. Similarly, the observer will ensure that the RCS thrusters are not operated unless all ATM aperture covers are closed. The observer should wait at least two minutes following either a waste dump or an RCS thruster firing before reopening the aperture covers.

The observer will record the video images during data acquisition to obtain data on the ATM absolute roll angle about the line of sight. Sketches in the log book of the



solar activity will be accurate enough to define the roll angle, but it may be necessary to photograph the video image to eliminate this duty from the ATM observer. Such a camera could be jettisoned during the final EVA if storage is not available.

2.3.3.4 Camera Film Recovery and Stowage

The expected observing profile will expend the initial film magazines after approximately seven days in orbit. At this time in the mission, the film magazines will be recovered by one of the astronaut crew in an EVA operation. The ATM will be positioned in the gimbals at the desired angle for the EVA operation, and the gimbal brakes will lock the ATM relative to the CSM.

The EVA astronaut will take the new film magazines with him when he exits the CM. The film magazines are stored in the CM rock boxes or other suitable compartments during the first half of the mission. These magazines will be on lanyards attached to his suit or to the CM during EVA to avoid accidental loss.

The EVA astronaut will open the hatches in the thermal shield to gain access to the film magazines in the instruments. He will remove the expended film magazines and attach them to the lanyards. The new film magazines will be inserted into place and will be ready to operate with no further manipulation. The astronaut will close the hatches before returning to the CM.

At the end of a mission, a second EVA operation will be performed to recover the second set of film magazines. The procedure will be the same. During this EVA, the astronaut may choose to remain outside the CM at a safe distance to verify that the ATM oriented section jettisons properly.

The recovered film magazines will be stowed in containers which will insert into CM compartments. The empty food compartments may be used as well as the designated rock boxes.

2.3.3.5 Ground Scientist-Astronaut Communication

During the mission, the ground scientists and the astronaut observer will exchange information on data acquisition and solar activity. Such information will be relayed through the respective capsule communicators. Typical of such communications will be the description of an activity region that the scientist has observed through a ground telescope.

A polar coordinate reticle could be used on the video display in the CM and the scientist could define the activity location from a similar reticle and video image on the ground. The astronaut observer could align his reticle along the solar equator defined by the distribution of the solar activity regions as shown in Fig. 2-13. Similar alignment of the ground reticle to the same angle θ and callout of polar coordinates should eliminate any ambiguities.

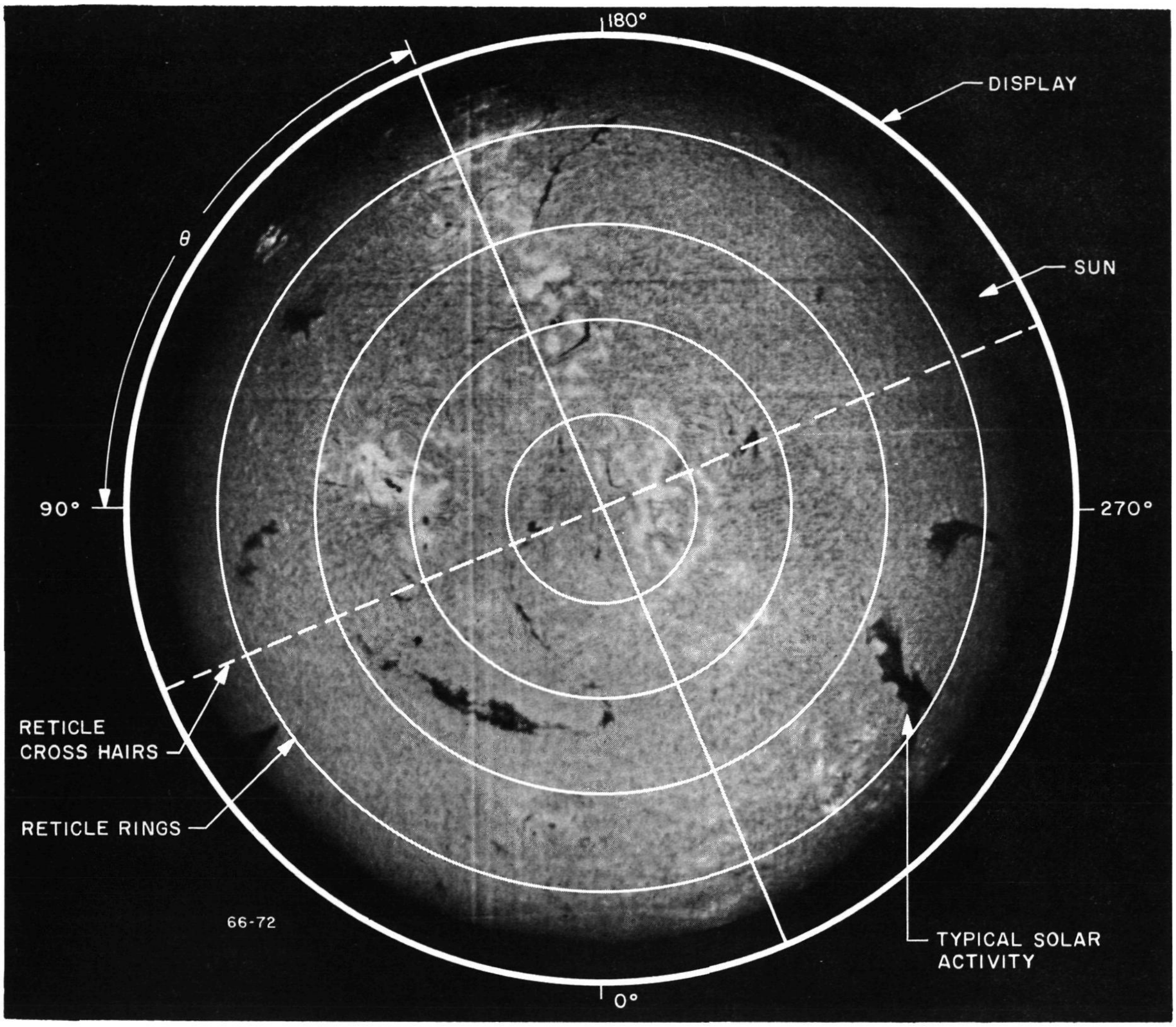
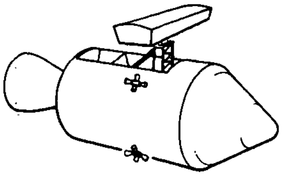


Fig. 2-13 Solar Monitor Display Reticle Adjustment to Sun's Equator

Other communications might alter the observing plans to change the data acquisition periods or to utilize the film if the expenditure rate is lower than expected.

2.3.4 Post-Mission Operations

The experiment film magazines must be removed from the CM for processing. The film is not expected to be damaged by the salt humidity during CM recovery since it is stowed in containers. However, excessive temperature would have an adverse effect,



and the film containers should be removed from the CM as soon as possible for special handling.

Post flight debriefing between the astronaut crew and the cognizant scientists will be conducted to ensure the maximum exchange of information applicable to the data analysis.

SECTION 3

GENERAL DESIGN CONSIDERATIONS

Section 3
GENERAL DESIGN CONSIDERATIONS

The systems design of ATM must effectively fill the void between Apollo constraints and criteria, and the experiment complement requirements in a manner to efficiently and effectively perform solar research missions. The proper interpretation of these criteria form the basic framework for the ATM design concept. The following sections present both the criteria and the interpretation of the criteria that have been used in evolving the ATM system concept.

3.1 EXPERIMENT REQUIREMENTS

The system performance capabilities of ATM have been established as a result of the requirements of four typical solar experiments investigated during the study. These four experiments were supplied by NASA as being representative of the types of experiments that would utilize the ATM system. These representative experiments and their respective cognizant scientists, who provided the experiment definition are:

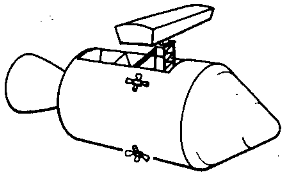
- Ultraviolet Spectrometer,
  Harvard College Observatory (HCO) - Dr. L. Goldberg
- Ultraviolet Spectrograph,
  Naval Research Laboratory (NRL) - Dr. R. Tousey
- Extreme UV/X-ray Spectroheliographs,
  Goddard Space Flight Center (GSFC) - Mr. W. White
- White Light Coronagraph,
  High Altitude Observatory (HAO) - Dr. G. Newkirk

Each of these experimenters has submitted requirements for experiments to be conducted on an ATM system (Ref. TN 65-302 "Summary of ATM Experimenters' Meeting"). These requirements have been reviewed during the study and have been used as the basis for establishing the general scope of experiment accommodation on a solar ATM system. The ATM experiment requirements in each major interface area are presented in this section and are based on the interpretation of requirements of these four experiments.

3.1.1 Experiment Objectives

The objective of the solar ATM experiments will be to investigate specific solar features in detail with a complement of instruments measuring in the white light, ultraviolet, extreme ultraviolet, and X-ray regions of the spectrum. A typical complement of experiment instruments could include a coronagraph, an ultraviolet spectrometer/spectrograph, and X-ray and ultraviolet spectroheliographs. Major emphasis of such experiments would be to investigate activity regions on the solar disk or in the corona to determine the characteristics of specific phenomena.

The experiment instruments will be designed to resolve solar features of approximately five arc seconds in size, and in the case of a spectrometer or spectrograph will



observe solar energy from a nominal five arc second square field of view. The latter type instruments will require pointing to a specific activity area on the solar disk to investigate only that area in the narrow field of view. These activity regions will have varied characteristics and it is expected that the astronaut observer will select the target regions to be investigated and align the experiment optics to that region.

The experiments will be designed to enable the astronaut observer to control the acquisition of data from the selected target regions. Based on displays of the selected target activity, the astronaut can initiate data acquisition to extend for a period of a few minutes up to about 20 minutes. These data acquisition periods will vary depending on the characteristics of the solar activity. Solar flares typically have an active period of as long as 20 minutes, but major flares could extend for over one hour.

It is expected that the astronaut observer will be able to control the experiment data acquisition to include many observations of general solar features, as well as a significant number of observations of short-lived activities such as solar flares and prominences.

3.1.2 Instrument Configuration

The experiments considered during this study represent a variety of instrument configurations, which vary in size and optical arrangement. These optical arrangements define the expected instrument sizes and are depicted for typical experiments (Figs. 3-1 and 3-2) which are discussed in the following paragraphs.

3.1.2.1 Cross Section

The cross section requirements of ATM experiments will vary according to the type of optical arrangement within the instrument. Clear viewing access of the sun is defined by the size of the primary optics, and the total cross section required is defined by the off-axis optics and auxiliary components of the instrument. Typical requirements are indicated in Table 3-1 for the four experiments analyzed. It is expected that a complement of experiments for ATM would include instruments that vary in cross section from a minimum of approximately 100 square inches up to several hundred square inches.

The maximum telescope diameter indicated in Table 3-1 is 10 inches. However, it is expected that experiments will be proposed with telescopes from 15 to 20 inches in diameter. Compound instruments with more than one telescope system are expected.

Specific experiments may have cross section profiles that vary along the instrument length, possibly allowing a cross-sectional trade between experiments as a function of location along the length. However, in general each experiment will require a given cross section for its entire length.

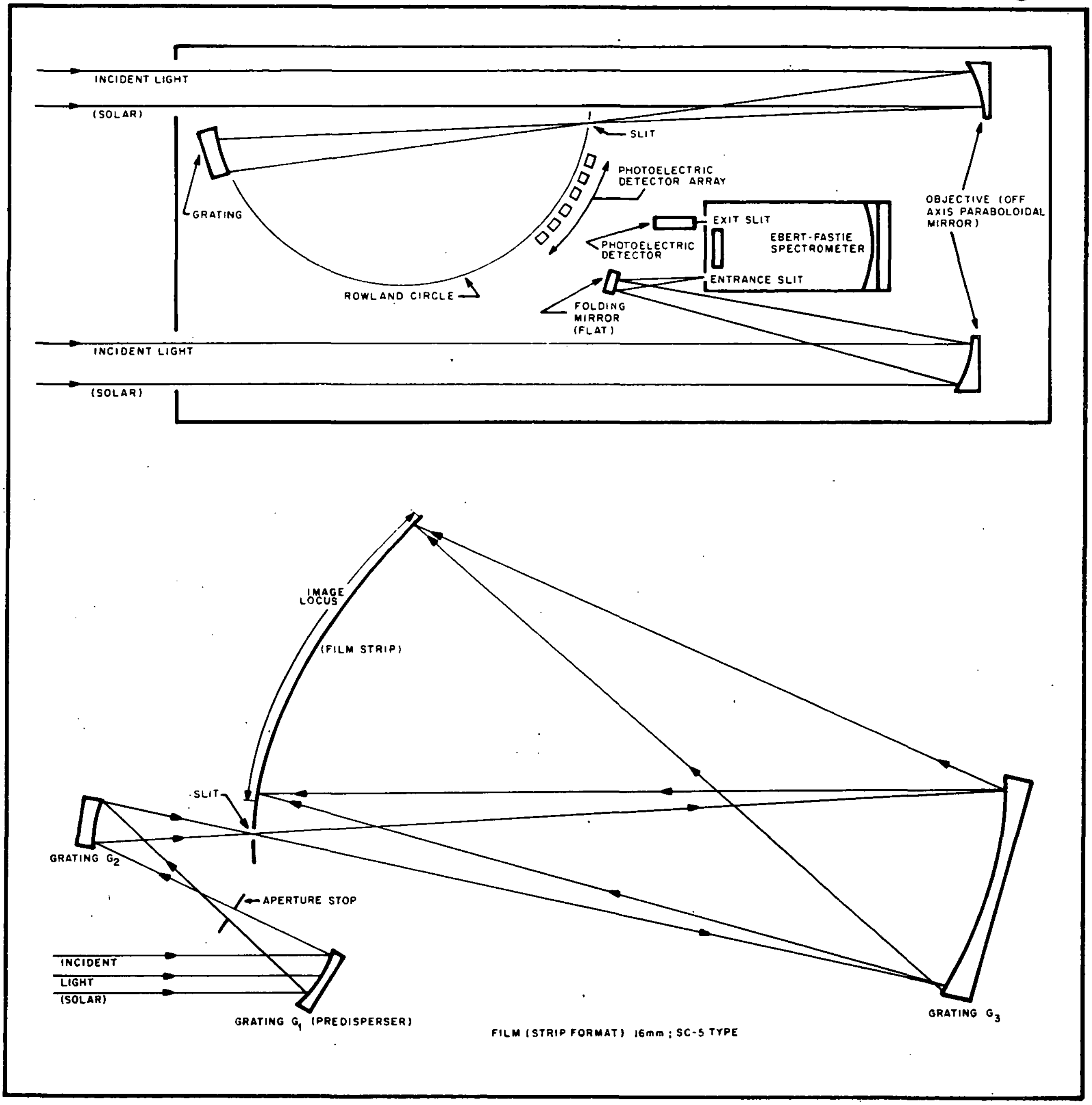


Fig. 3-1 Typical Experiment Optics (Ultraviolet Experiments)

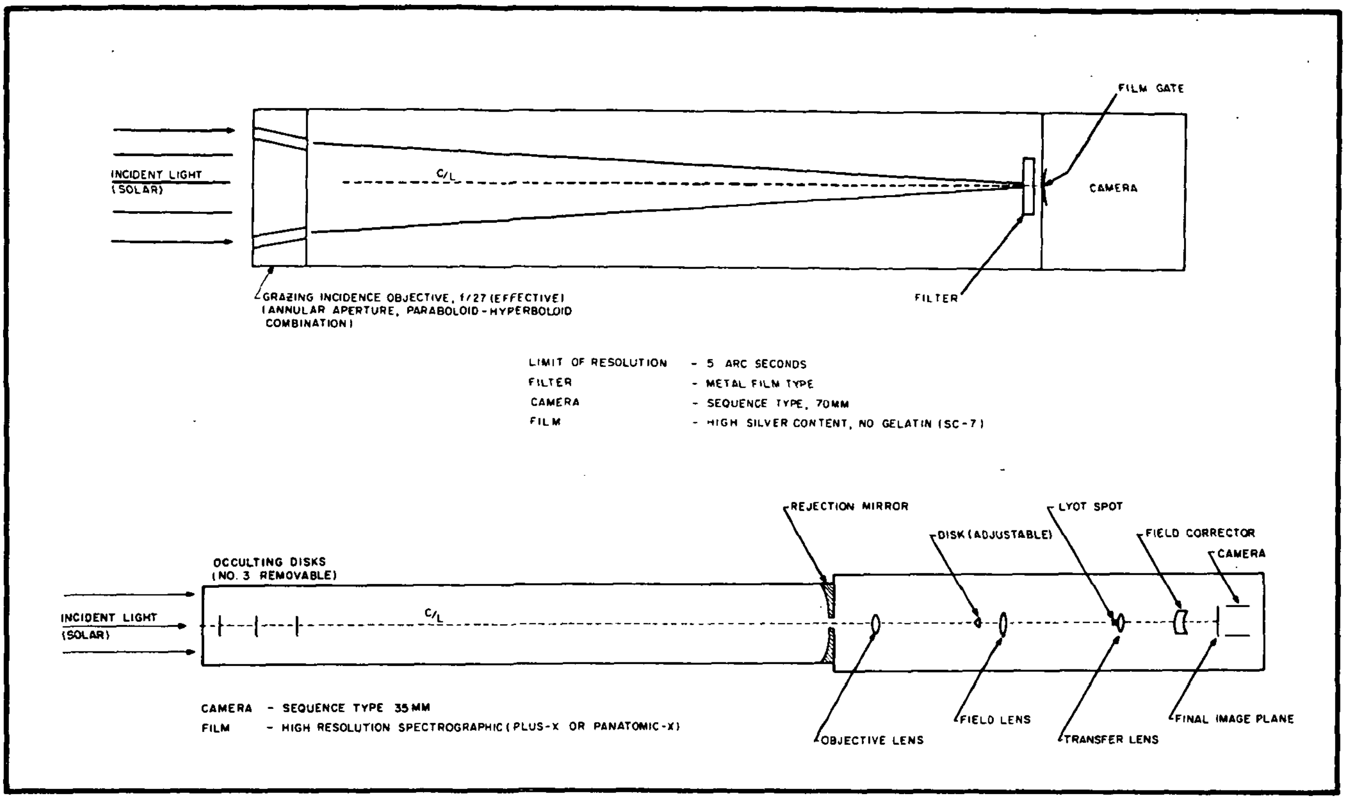


Fig. 3-2 Typical Experiment Optics (X-Ray Telescope and White Light Coronagraph)

3.1.2.2 Length

The overall length requirements of ATM experiments will vary according to their optical characteristics. The lengths of typical instruments are indicated in Table 3-1 for the study experiments. It is expected that the minimum instrument length will be about eight feet and the maximum about 11.5 feet. The lengths indicated in Table 3-1 include only the major structure of the instrument and do not include light baffling structures that may be required on the front of the instrument to reduce reflections from neighboring equipment.

3.1.2.3 Weight

The weight of an ATM experiment will vary according to its optical design and the associated cameras and electronics. Weights for the typical study experiments are indicated in Table 3-1. These estimated weights are based on experiment structures that are self-contained, and that provide the necessary optical stability. It is expected that the maximum single experiment weight to be accommodated will be approximately 300 pounds.

Table 3-1
EXPERIMENT CONFIGURATION REQUIREMENTS

Study Experiment	Maximum Telescope Diameter (in.)	Number of Primary Telescopes	Number of Auxiliary Telescopes	Total Cross Section (sq. in.)	Length (ft.)	Weight (lbs.)
HCO	10	2	1 [(a)]	336	11.5	250
NRL	4	1	1 [(a)]	112	8	150
GSFC	7	2	3	180	9	150
HAO	8	1	1 [(a)]	156	10	150

(a) Boresight telescopes.

3.1.3 Experiment Pointing

The experiment objectives, indicated in Section 3.1, define the accuracy of the experiment pointing required. These requirements are discussed in more detail in the following paragraphs and are summarized in Table 3-2.

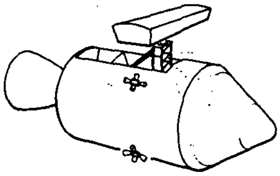
3.1.3.1 Solar Disk Pointing Reference

The typical experiments analyzed in the study included two types of solar disk orientation requirements. The coronagraph (HAO) and spectroheliograph (GSFC) require pointing to the center of the solar disk, and the spectrometer/spectrographs and spectroheliographs require referencing to activity regions anywhere on the solar disk. Pointing to the center of the solar disk must be maintained to within ±20 arc seconds, and activity region or offset pointing must be maintained to within ±5 arc seconds during any one data acquisition.

It is expected that experiments requiring center pointing in the range from ±1 arc minute to ±5 arc seconds will provide an error signal to the required accuracy that can be displayed to the astronaut observer. This display will enable the astronaut to adjust ATM pointing to effect the required experiment alignment. Similarly, experiments requiring offset alignment in the range from ±1 arc minute to ±5 arc seconds of a given solar disk feature will provide display data to the astronaut observer. This display will enable the astronaut observer to select the desired solar feature, offset the ATM pointing to boresight on that feature, and monitor the experiment alignment throughout the data acquisition period.

3.1.3.2 Relative Pointing to Reference

The requirement for relative pointing of the experiment to the selected reference is based on the instrument resolution, field of view and the duration of data acquisition. These data are given for the study experiments in Table 3-2, and



indicate that the limiting factors expected for solar experiments are:

- Instrument resolution: 5 arc seconds
- Minimum field of view: 5 arc seconds (square)
- Maximum photographic data period: 1 minute

These limiting factors impose a relative pointing requirement of less than five arc seconds drift from the starting alignment during a one-minute exposure period. They also impose the requirement for a dynamic pointing error (pointing noise) of less than five arc seconds integrated deviation from the starting alignment during a one-minute exposure period.

It is expected that ATM experiments will require a sequence of data acquisition or exposure periods. Typical data periods are indicated in Table 3-2 for the study experiments, which range from a minimum of one minute to a nominal 20 minutes for solar flare or activity observations. Since the astronaut observer will be monitoring the experiment alignment to the desired reference point throughout a data sequence, the relative pointing may accumulate a drift from the starting point. It is expected that the astronaut observer will maintain the proper alignment to counteract any accumulative drift, by manipulating offset pointing controls. This action can be performed between exposure periods and it should not require more than a few seconds to correct the alignment. In the case of an offset raster or a wavelength scan (similar to the HCO requirements - Table 3-2) that lasts for several minutes, the astronaut observer will maintain proper alignment throughout the period as required.

Table 3-2
EXPERIMENT CONFIGURATION REQUIREMENTS

Study Experiment	Pointing Reference	Spatial Resolution (arc sec)	Field of View	Offset Range (arc min)	Boresight Display	Data Sample Duration (sec)	Data Sequence Duration	
							Quiet (min)	Active (min)
HCO	Selected Active Regions	5	5 arc sec square slit;	±20	Yes	0.040	5(raster)	25(raster)
			1.9 arc sec wide slit	±20	Yes	0.040	8.5 (scan)	25.5 (scan)
NRL	Selected Active Regions	5	5 arc sec wide slit	±20	Yes	60	10	20
GSFC	Solar Disk Center	5	40 arc min		No	10	1	20
HAO	Solar Disk Center	15	225 arc min		Yes	2	1	20

3.1.4 Thermal

ATM experiments will require a thermal environment that is stable to ±5°F within the range 30°F to 80°F. The experiment optics and support structure will be designed to accommodate the dynamic increase in heat flux when the apertures are opened during data acquisition.

3.1.5 Observing Schedule

The ATM mission profile will be based on the observing requirements of each experiment. Observing requirements for the four study experiments are given in Table 3-3. Three modes of experiment observing are defined by these requirements:

(1)	Patrol mode:	Experiments in standby condition with coarse sun orientation and astronaut monitoring solar activity. No data acquisition performed.
(2)	Standard (quiet) mode:	Scheduled Observations of solar regions for data acquisition periods of 1 to 10 minutes.
(3)	Activity mode:	Nonscheduled observations of active solar regions for data acquisition periods of 20 minutes or greater.

A typical observing schedule for one calendar day for the four study experiments is shown in Fig. 3-3. The activity modes are superimposed on the standard mode observing and the patrol mode is depicted to complete the major gaps in data acquisition. Common observing between experiments with overlapping fields of view or approximately parallel optical axes (such that they are aligned to the same active region) is indicated during the 10th and 15th orbits.

The total observing time for each study experiment is shown in Table 3-3, resulting in the following typical breakdown by mode:

(1)	Patrol mode:	50 hours
(2)	Standard mode:	30 hours
(3)	Activity mode:	20 hours
	Total:	100 hours

The accumulative data acquisition for the four study experiments is shown in Fig. 3-4. An EVA recovery of the film magazines should be performed at about the seventh day, coinciding with completion of 50 percent of the expected data acquisition. Late in the 14-day mission the astronaut observer and the cognizant ground scientist may agree to revise the relative amount of standard activity mode observing to optimize the use of the remaining film.

3.1.6 Photographic Data

Some of the ATM experiments will utilize photographic cameras for primary data acquisition, as shown in Figs. 3-1 and 3-2. These data cameras will be mechanisms that place rolled film or strip film at the experiment image plane. The film

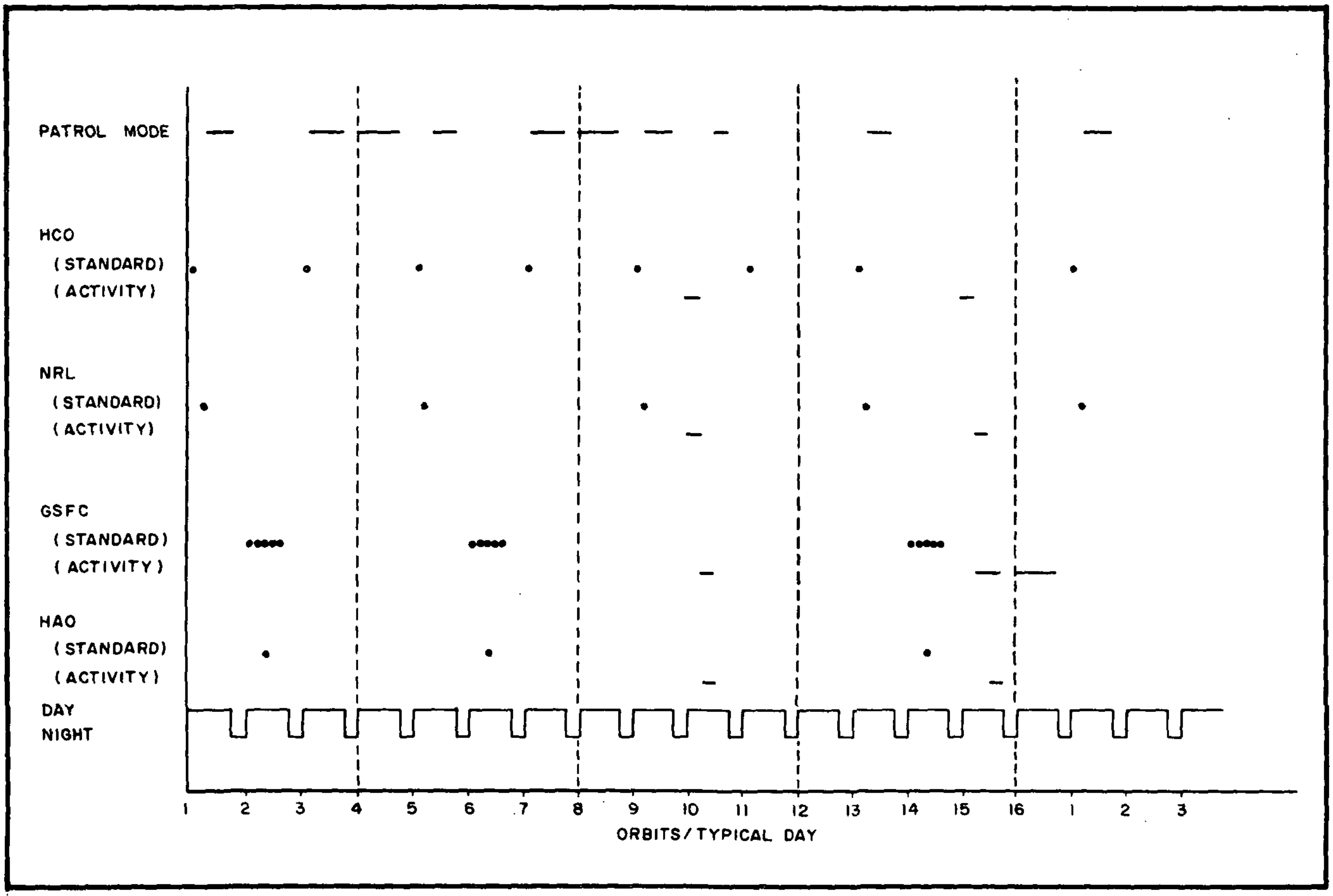


Fig. 3-3 Experiment Observing Experiment Observing Schedule for One Calendar Day

magazines will be separate from the experiment cameras to enable astronaut recovery during EVA. The location of cameras in the experiments will vary from the front to the rear of the instrument as depicted in Fig. 3-1 and 3-2.

The film magazines will be self-contained and will not require any threading to load in the camera. Film will be of several different types ranging from Plus X for white light instruments to specialized Schumann emulsions for the ultraviolet and X-ray instruments. Film sizes will vary from 16 mm to 70 mm as indicated for

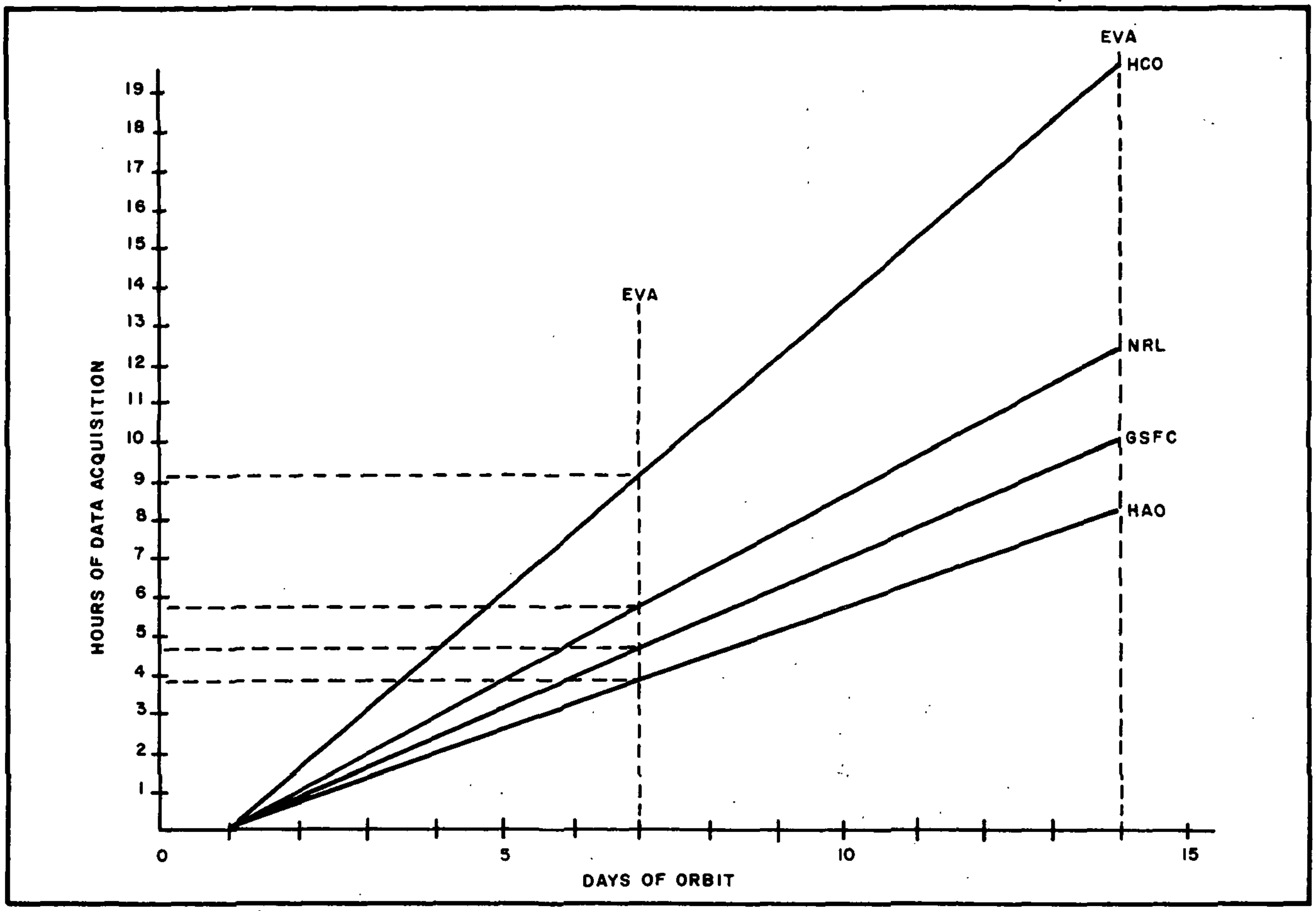


Fig. 3-4 Accumulative Data Acquisition Time For Study Experiments

the study experiments in Table 3-4. The amount of film to be used is also shown in this table, and it can be seen to range up to several hundred feet per magazine.

The film footage requirements are based on the expected observing times for the study experiments (Table 3-3) and on the use of film bases of 0.005 inch thickness. The recoverable film magazine sizes are indicated in Table 3-4 for rolled film. If specialized strip film cassettes are used for the Schumann films, these sizes may increase.

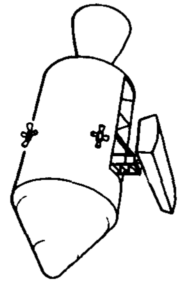


Table 3-3
EXPERIMENT OBSERVING REQUIREMENTS

Study Experiments	Standard Mode: Duration of Data Sampling Sequences (min)	Standard Mode: Number of Data Sampling Sequences	Standard Mode: Number of Orbits Between Sequences	Standard Mode: Cumulative Observing Time (hr)	Activity Mode: Duration of Data Sampling Sequences (min)	Activity Mode: Number of Data Sampling Sequences	Activity Mode: Number of Orbits Between Data Sequences	Activity Mode: Cumulative Observing Time (hr)	Patrol Mode: Cumulative Observing Time (hr)	Total Time (hr)
HCO	8.5	100	2	14.1	25.5	13	15	5.5	--	19.6
NRL	10	50	4	8.3	20	13	15	4.3	--	12.6
GSFC	5[a]	50	4	4.1	20[a]	13	15	4.3	50[b]	58.4[b]
					100	1	200	1.7	--	1.7
HAO	5[a]	50	4	4.1	20	13	15	4.3	--	8.4
Totals				30.6				20.1	50	100.7

(a) Assumes 5 data samples (approximately 1 minute each) for each sequence distributed over one orbit at 12 minute intervals.

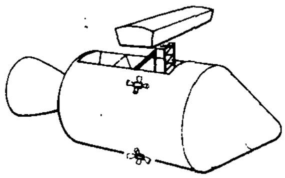
(b) GSFC experiment in standby operation as activity detector.

Table 3-4

EXPERIMENT CAMERA CHARACTERISTICS

Study Experiment	Film Type	Film Size	Standard Mode					Activity Mode					Summation		
			Camera Rate exp/min	Exposures per foot	Total Observing Time (hr)	Cumulative Film Footage (ft)	Number of Exposures	Camera Rate exp/min	Exposures per foot	Total Observing Time (hr)	Cumulative Film Footage (ft)	Number of Exposures	Total Film Footage (ft)	Number of Magazines	Probable Magazine Diameter (in)
HCO	None	--	--	--	--	--	--	--	--	--	--	--	--	--	--
NRL	Schumann SC-5	16mm	1	1	8.3	500	500	1	1	4.3	260	260	760	2	12
GSFC	Schumann SC-7	70mm	--	4	4.1[a]	62[a]	250[a]	6	4	6.0	540	2160	602	2	12
HAO	Plus X or Panatomic X	35mm	4	13	4.1	77	1000	12	13	4.3	240	3120	317	2	8

(a) Five data exposures (approximately one minute each) for each orbit of operation (50 orbits of operation).



3.1.7 Digital and Housekeeping Data

Some ATM experiments will produce digital data in the primary measurement mode or as secondary measurements. In some cases, secondary digital data generated within an experiment will be recorded on the camera film. Experiments that generate primary digital data will require data handling and data recovery. The digital data requirement for an ATM experiment complement will be similar to that given for two of the study experiments indicated in Table 3-5.

Table 3-5
EXPERIMENT DIGITAL DATA REQUIREMENTS

Study Experiment	Channel No.	Standard Mode				Activity Mode				Total Number of Bits
		Samples (per sec)	Bits per Sample	Observing Time (hr)	Number of Bits	Samples (per sec)	Bits per Sample	Observing Time (hr)	Number of Bits	
HCO	1	25	16	8.3	$1.2x10^7$	25	16	4.3	$6.2x10^6$	$1.8x10^7$
	2	25	16	8.3	$1.2x10^7$	25	16	4.3	$6.2x10^6$	$1.8x10^7$
	3	25	16	8.3	$1.2x10^7$	25	16	4.3	$6.2x10^6$	$1.8x10^7$
	4	25	16	8.3	$1.2x10^7$	25	16	4.3	$6.2x10^6$	$1.8x10^7$
	5	25	16	8.3	$1.2x10^7$	25	16	4.3	$6.2x10^6$	$1.8x10^7$
	6	25	16	8.3	$1.2x10^7$	25	16	4.3	$6.2x10^6$	$1.8x10^7$
	7	25	16	8.3	$1.2x10^7$	25	16	4.3	$6.2x10^6$	$1.8x10^7$
	8	25	16	8.3	$1.2x10^7$	25	16	4.3	$6.2x10^6$	$1.8x10^7$
	9	25	16	14.1	$2.0x10^7$	25	16	5.5	$7.9x10^6$	$2.8x10^7$
	10	25	16	14.1	$2.0x10^7$	25	16	5.5	$7.9x10^6$	$2.8x10^7$
				Total	$13.6x10^7$				$65.4x10^6$	$20.0x10^7$
GSFC	1	0.017	16	4.1	$4.0x10^3$	0.1	16	6	$3.5x10^4$	$3.9x10^4$
	2	0.017	16	4.1	$4.0x10^3$	0.1	16	6	$3.5x10^4$	$3.9x10^4$
	3	0.017	16	4.1	$4.0x10^3$	0.1	16	6	$3.5x10^4$	$3.9x10^4$
	4	0.017	16	4.1	$4.0x10^3$	0.1	16	6	$3.5x10^4$	$3.9x10^4$
				Total	$16x10^3$				$14.0x10^4$	$15.6x10^4$

The digital data output from experiments will be over discrete observing periods and will not be a continuous requirement. Between observing periods no digital data will be produced by the experiments, except housekeeping data.

The summary requirements for minimum experiment digital data handling are:

(1) 15 data channels
(2) Minimum sampling rate per channel: 0.01 samples per second
(3) Maximum sampling rate per channel: 25 samples per second
(4) At least two channels with different sampling rates
(5) 16 bits per data channel word
(6) 3 times 10^7 total bits per data channel
(7) Sampling rate change from standard to activity mode

In addition, five sampled, analog, housekeeping channels are required for each experiment.

Experiments with cameras will provide data indicating the amount of film used at any given time for astronaut surveillance.

3.1.8 Supporting Measurements

The ATM experiments will require supporting measurements from the ATM system. The position of the experiment optical axis on the solar disk must be determined during data acquisition as follows:

(1) Absolute average X-Y position to within ±1 arc minute

(2) Average relative position from starting alignment to within ±5 arc seconds

(3) Absolute roll angle about the line of sight to within ±5 degrees

All experiment data (digital and photographic) requires absolute time correlation to within ±1 second of a general time reference. Time signals can be received by the experiments and recorded on film, or the experiments can provide a signal designating film frame number, which can then be related to time in the data handling subsystem.

3.1.9 Commands and Displays

The ATM experiments will require commands to be executed by the astronaut observer to operate functions within the instrument, as well as read out the analog housekeeping channels to the astronaut. A list of typical commands for one experiment is shown in Table 3-6.

The total number of commands required for a complement of four experiments will be at least 60. In addition to the on demand display of the five housekeeping channels, at least one full time indicator per experiment will be required to indicate experiment status to the astronaut.

3.1.10 Power

The ATM experiments will require DC power for operation. The power requirement of the four study experiments is given in Table 3-7. These data are based on experiment operation only during data acquisition periods. Assuming no more than two ATM experiments are operative in parallel, the maximum load would be 45 watts at any one time. The total amount of energy that should be made available for ATM experiments is 2.0 kilowatt-hours.

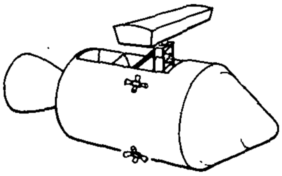


Table 3-6

EXPERIMENT COMMAND AND DISPLAY REQUIREMENTS

Number	Command	Purpose	
		Functional Operation	Display
1	Data Camera Start	X	
2	Data Camera Stop	X	
3	Activity Mode/Standard Mode Select	X	
4	Optics raster 1 slew on	X	
5	Optics raster 1 slew off	X	
6	Automatic Data Mode In	X	
7	Automatic Data Mode Out	X	
8	Spare	X	
9	Spare	X	
10	Spare	X	
11	Data Channel No. 1		X
12	Data Channel No. 2		X
13	Data Channel No. 3		X
14	Data Channel No. 4		X
15	Data Channel No. 5		X

Table 3-7

EXPERIMENT POWER REQUIREMENTS

Study Experiment	Average Power (watts)	Total Operative Time (hr)	Total Energy Required (kw-hr)
HCO	10	19.6	0.196
HCO (Boresight)	15	19.6	0.294
NRL	5	12.6	0.063
NRL (Boresight)	15	12.6	0.294
GSFC	10	60.1[a]	0.601[a]
HAO	10	8.4	0.084
Totals	65	--	1.532

(a) Includes 50 hours of patrol mode standby operation.

3.1.11 Contamination Control

The ATM experiments will utilize precision optics that must be maintained in a clean environment. The optics must not become contaminated during ground checkout, launch, or orbital operation. During orbital operation the oriented section of the ATM must be nominally dust proof, with the exception of the apertures when open. Care must be taken to avoid the firing of RCS thrusters or the waste dump during any period that the aperture covers are open.

3.2 EXPERIMENT POINTING AND DISTORTION

An examination of the pointing requirements of the experiments (Table 3-2) shows that structural and pointing stabilities measured in microns and arc seconds are required to exploit the high-resolution capability of these instruments. This section presents the results of the study in this area and serves as an introduction to the analysis and design of the structural, thermal control, and pointing control subsystems.

3.2.1 Definitions

There are a series of effects which, if not adequately controlled, can cause serious degradation of the experiment data. Among these are a group which taken together result in stringent requirements upon the above mentioned ATM subsystems.

The term "instrument axis" shall be used to describe the direction of the center of the field of view of an instrument. As some of the experiments have several optically independent instruments, there will be one instrument axis for each independent instrument.

The term "pointing error" shall mean the angle between the true and the desired instrument axis.

For each instrument, there is some alignment of the optical components which results in optimum performance of the instrument. Any mechanical disturbance to this alignment will cause some degradation from expected performance. The set of such disturbances, whether in position or in orientation, will be termed the "distortion" of the instrument.

3.2.2 Average Pointing Errors

ATM experiments similar to three of the study experiments (HAO, NRL, GSFC) will utilize framing cameras. During any given frame or exposure, the instrument axis will describe some tour in space relative to the desired instrument axis. By integrating over the exposure time, an average direction of the instrument axis may be found and its angular distance from the desired instrument axis may be called the average pointing error. Other experiments (similar to the HCO study experiment) will utilize photoelectric techniques with discrete data sampling. For the purpose of this analysis, these may be considered equivalent to framing instruments, and the averaging definition given above is applicable to all instruments.

From the definition given here, it may be seen that the average pointing error does not contribute to the smearing of film or photoelectric data. Instead, it is a measure of the uncertainty in the experimenter's knowledge of the solar location of the actually observed field.

The field of the HCO 1250Å to 2250Å study spectrometer is 1.9 arc sec by 30 arc sec. A serious error in the knowledge of the field could result if the average pointing error during data acquisition exceeded the field dimension. Thus, the average pointing error should be held to about 1.5 arc seconds.

Similarly, the NRL study instrument (spectrograph) has a field of view of five arc seconds in one dimension, and in some operating modes may be several arc minutes in the other.

The other HCO study instrument has a five arc sec square field of view; but by tilting the main mirror, it scans through a five arc min square. It thus has a lesser average pointing error requirement, since this raster will encompass a relatively large area of the solar disk.

The GSFC study instruments image the entire solar disk with a field of view greater than the solar disk. No average pointing error difficulties are foreseen for this type of instrument.

The average pointing error for the HAO study coronagraph must be held to less than 20 arc seconds from the center of the solar disk. With an exposure time of two seconds, the initial fine adjustment should last through many exposures.

Each experiment will be equipped with an individual boresight system with which the astronaut may make initial instrument orientation to specific solar features or the center of the disk as required. Systems considered in this study include auxiliary telescopes with image display to the astronaut and internal null indicators for disk centering. These boresight systems will require visual presentation to the operator compatible with the desired experiment pointing accuracies. In some cases, provisions are to be made for checking the alignment of the boresight system to the instrument axis during the mission.

The sources of average pointing error may be grouped under three general headings and are depicted in Fig. 3-5:

(1) Experimental boresight error. This is the error between the instrument axis and its associated boresight system.

(2) Structural-thermal drift. Structural deformation contributes to average pointing error, causing the relative alignment, between the instrument axis and the pointing control subsystem fine sun sensor mechanical axis, to vary before the end of an exposure and following the last boresight correction. Creep and thermal bending in the experiment structure, and effects of spar creep and thermal bending on the experiment contribute to this error.

(3) Pointing control error. This is drift defined just as above except that the drift sources are all within the pointing control subsystem. These include thermal and electrical drift within the fine sun sensors, instability of the offset bias, and noise and drift in the servo electronics. Also included here is the pitch and yaw drift due to residual roll rate.

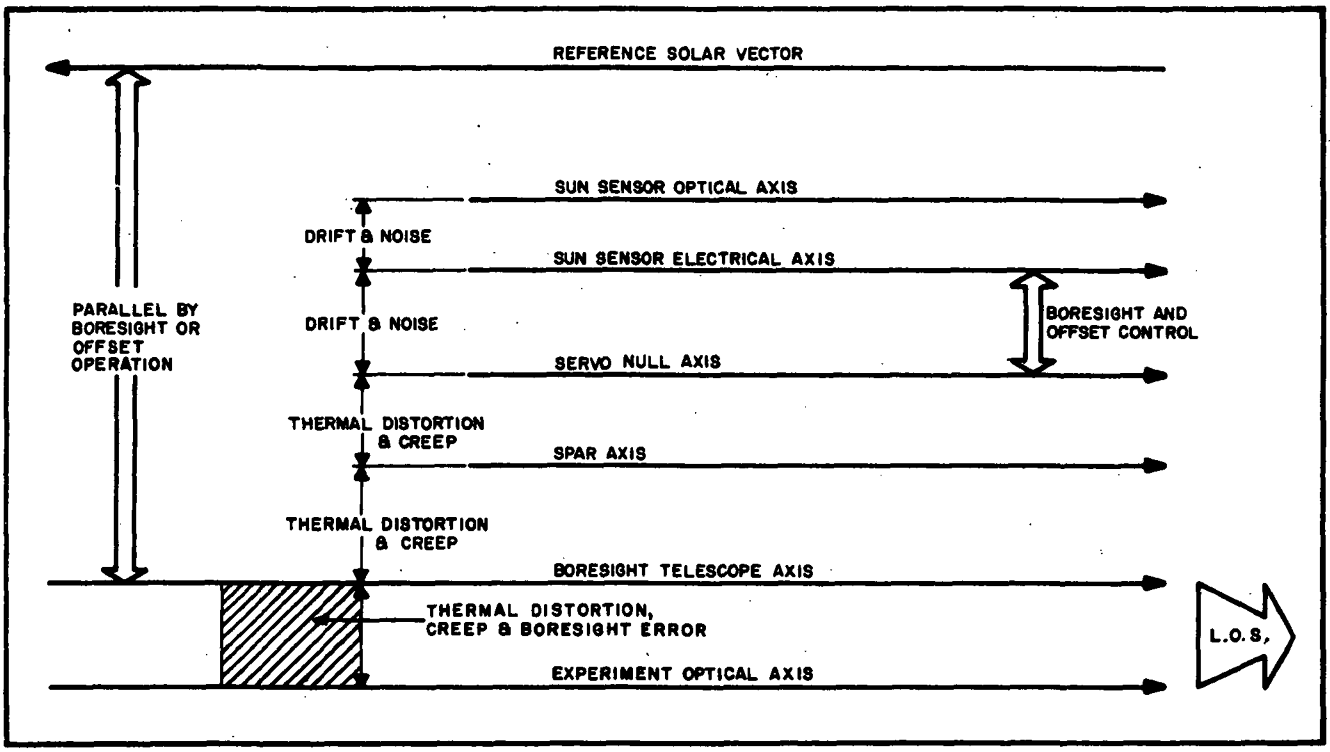
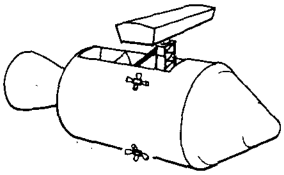


Fig. 3-5 ATM Pointing References



Some illustrative values of average pointing errors in each category above are given here to indicate the scope of the problem. Assume that the experiment boresight error is held to one arc second. Also let the structural-thermal drift be composed of one arc second per minute of thermal drift in the spar and in the experiment structure, and one arc second per minute of creep in each structure. Let the overall pointing control error be 2.5 arc seconds per minute except for roll rate effects, which at two arc sec per second roll rate and a 10 arc minute offset from the center of the solar disk, result in an additional error rate of 0.35 arc sec per minute. If the time between the last boresight adjustment and the middle of the exposure is 3/4 minute, then the root-sum-square average pointing error is 2.6 arc seconds. It may be inferred from this that for the above errors and rates only, for exposures of one minute or greater, will the average pointing error become excessive. This calculation is only illustrative, and additional systems analysis will be required.

3.2.3 Dynamic Pointing Errors

The dynamic pointing error is defined by again considering the time average direction of the instrument axis during an exposure. The instantaneous dynamic pointing error is the angle between the instantaneous instrument axis and the average instrument axis defined above. Since this instantaneous error is a function of time during any one exposure, another integral average is needed to describe its effect on film resolution.

Probably the best integral measure for the purpose is the RMS integral:

$$\overline{E} = \left[\frac{1}{T} \int_0^T E^2 \, dt\right]^{1/2} \qquad (3.1)$$

where:

E is the instantaneous dynamic pointing error
T is the exposure time

A more complex measure of the effect of dynamic errors on resolution could be formulated, but this will serve to illustrate the magnitude of the effect. If the instantaneous error is a linear function of time and has a drift rate $\mathring{E}$, then $\overline{E} = 0.288\mathring{E}T$.

The GSFC study experiment full disk imaging X-ray telescopes have five arc second resolutions and 10 second exposure times. Thus, their maximum tolerable linear drift rate is 1.7 arc seconds per second. The GSFC study auxiliary instruments are photoelectric, observing the entire solar image and thus have no dynamic error problems.

The HAO study coronagraph has a resolution of 15 arc seconds and an exposure time of two seconds. Its maximum tolerable linear drift is 26 arc seconds per second.

Of the three photographic systems, the worst dynamic error problem is presented by the NRL study instrument, with a five arc second resolution and a 60 second exposure time. The maximum linear drift rate across the slit in the spectrograph is 17 arc seconds per minute (or 0.28 arc seconds per second).

For photoelectric systems, the data integration time will be short (0.04 seconds for the HCO study experiment); thus, the dynamic pointing error is not a significant problem. The sources of dynamic error are essentially the same as those contributing to the average pointing error except that experiment boresight errors have no dynamic component, thus the errors may be grouped as either structural-thermal drift or pointing control error.

Some illustrative values of dynamic errors in these categories are given here to indicate the scope of the problem. As a first case, consider an instrument axis offset to the limb. Let the structural-thermal drifts be as in the last section, i.e., one arc second per minute each in thermal drift and creep, in both the spar and the experiment structure. Also, let the overall pointing control error be 2.5 arc seconds per minute, not including roll rate effects. Taking the roll rate to be two arc seconds per second and the offset to be the full 20 arc minutes results in an additional error rate of 0.7 arc seconds per minute. The root-sum-square of these errors is 3.3 arc seconds per minute, which is well within the 17 arc seconds per minute maximum allowed calculated above for the NRL study spectrograph.

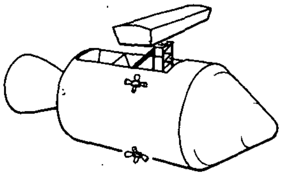
From the other example, consider an instrument axis on the center of the solar disk and a field of view out to 6.5 solar radii. Let the structural-thermal drifts and pointing control errors (not including roll rate effects) be as above. This time the roll rate enters as a direct rotation of the field rather than as cross coupling in the solar sensors because of offset. Taking the extreme field radius of 113 arc minutes and 2 arc seconds per second roll rate, the error rate from this cause is 3.93 arc seconds per minute. This gives a root-sum-square error rate of less than 0.1 arc sec per second, which is again well below the maximum allowed of 26 arc sec per second calculated above for the HAO study coronagraph.

In the case of experiments, similar to the HCO study spectrometers, requiring hold times up to 8.3 minutes, it is expected that the astronaut will make manual pointing corrections to offset dynamic system changes during the data acquisition period, insuring long-term pointing errors within acceptable limits.

3.2.4 Distortion

Control and/or maintenance of internal alignment of experiments cannot be generally solved. Each particular problem will require its own particular resolution. It is expected that these solutions will include consideration of structural rigidity, thermal control and balance, automatic adjustments, and astronaut controlled adjustments.

An example of the latter is the HAO study coronagraph, where in their present layout, a set of occulting disks is mounted well forward on the spar and the imaging optics



are mounted several feet to the rear. Preliminary analysis indicates that a misalignment between the axes of these two independent optical elements of up to 20 arc seconds can be tolerated.

To correct for this misalignment, a two-axis adjustment of a final internal occulting disk (D4) is provided. This can be adjusted by the astronaut using a two axis "fly to" control stick on the control unit to achieve a null indication from the internal image alignment indicator. Because of this provision, it may be possible to hard mount the HAO study experiment components directly to the ATM spar.

3.3 CONSIDERATIONS OF PHOTOGRAPHIC DATA RECOVERY

ATM experiment requirements include photographic detection of prime scientific data. Since there has been limited experience in space photography connected with scientific experiments, this study has included analysis of the design requirements and critical areas. The critical areas concerning photographic techniques that have been investigated during the study include thermal and vacuum restrictions, film handling, and the effects of radiation environment. These areas are discussed in the following paragraphs.

3.3.1 Vacuum Restrictions

It is expected that the various cameras will have mechanisms to transport film to the image plane of the telescopes. The operation of moving parts in the vacuum at 200 nautical miles requires specific attention in their design. Moving parts have been extensively used on spacecraft and space experiments which have been specially designed or treated for vacuum operation. Treatment with BBRC VAC KOTE vacuum lubrication process has proven to be effective against cold welding. These designs have proven successful during space operation, and camera mechanisms can be designed and treated to operate in vacuum for 14 days.

3.3.2 Thermal Restrictions

The camera mechanisms must be maintained within a temperature range of 30°F to 100°F for proper operation. Previous experience has shown that space mechanisms operating within this range have functioned properly. It is expected that the ATM thermal environment within the experiments will be in the range of 30°F to 80°F and no temperature problems should be encountered.

Thermal control of the film must be maintained so that it does not become brittle and lose any of its photographic qualities. The allowed thermal range for the film emulsions expected to be used on ATM experiments is from 30°F to +75°F, which is compatible with the temperature range expected in the experiments.

It should be noted that the highly sensitive Schumann emulsions have been evaluated by NRL (Dr. R. Tousey) at vacuum temperatures as high as 110°F. Results indicate no degradation in photographic properties.

Exposed film stored in the CM during reentry may be exposed to higher temperatures than specified above. This will depend on the storage location and may require that the film be stored in thermally insulated containers.

3.3.3 Film Handling

The film types to be used in some of the ultraviolet and X-ray experiments will be the specialized Schumann emulsions. This type emulsion is extremely sensitive and it must not come in contact with any other surface. This requires that specialized magazines or cassettes be used to handle the film.

Of the types of Schumann emulsions that may be used, SWR, SC-5, and SC-7, the latter two are the most sensitive and have only been manufactured in short strips. Therefore, the magazines for the SC-5 and SC-7 may have to be designed to handle individual strips. Such magazines are currently being developed by Dr. Tousey, NRL, and by Dr. Hiseman, Northwestern University.

The SWR Schumann emulsion film can be rolled in a special magazine that would separate the successive rolls from each other. Contact with Eastman Kodak, the manufacturer of this film, has indicated that a developer type magazine would probably maintain the required separation between the film rolls. This type of magazine supports the film at the edges in a continuous spiral. It may also be possible to roll the SC-5 and SC-7 films in such a continuous spiral.

The film handling of the more standard emulsions can be designed in the conventional manner. Magazines of 500 to 1000 feet can be used, with a film base thickness of 0.0005 inches. A 500 and a 1000 foot roll of this film will form a reel of about 8 inches and 11 inches in diameter, respectively.

The initial loading of film magazines in the cameras will be performed on the launch tower. The second set of film magazines will be loaded under EVA conditions in orbit, as will the recovery of both sets of magazines. This EVA operation will be performed by the astronaut wearing pressurized gloves and probably with poor visibility. Therefore, the film magazines must be self-contained, so that when inserted into the cameras all film threading is automatic. Retention mechanisms and handles on the film magazines must be designed to permit handling by the astronaut with the pressurized gloves on during EVA conditions.

3.3.4 Radiation Effects of the Apollo Fuel Gauging System

The Apollo RCS fuel gauging system consists of a cobalt 60 gamma radiation source that may affect the ATM experiments. This source is distributed on the outer sides of the eight RCS fuel and oxidizer tanks in the SM (Fig. 3-6) and has an aggregate strength of 25 millicuries. It produces two gamma rays: one at 1.18 Mev and the second at 1.33 Mev. These energetic gamma rays will penetrate to the ATM experiments, and probably cause adverse effects, particularly with photographic film.

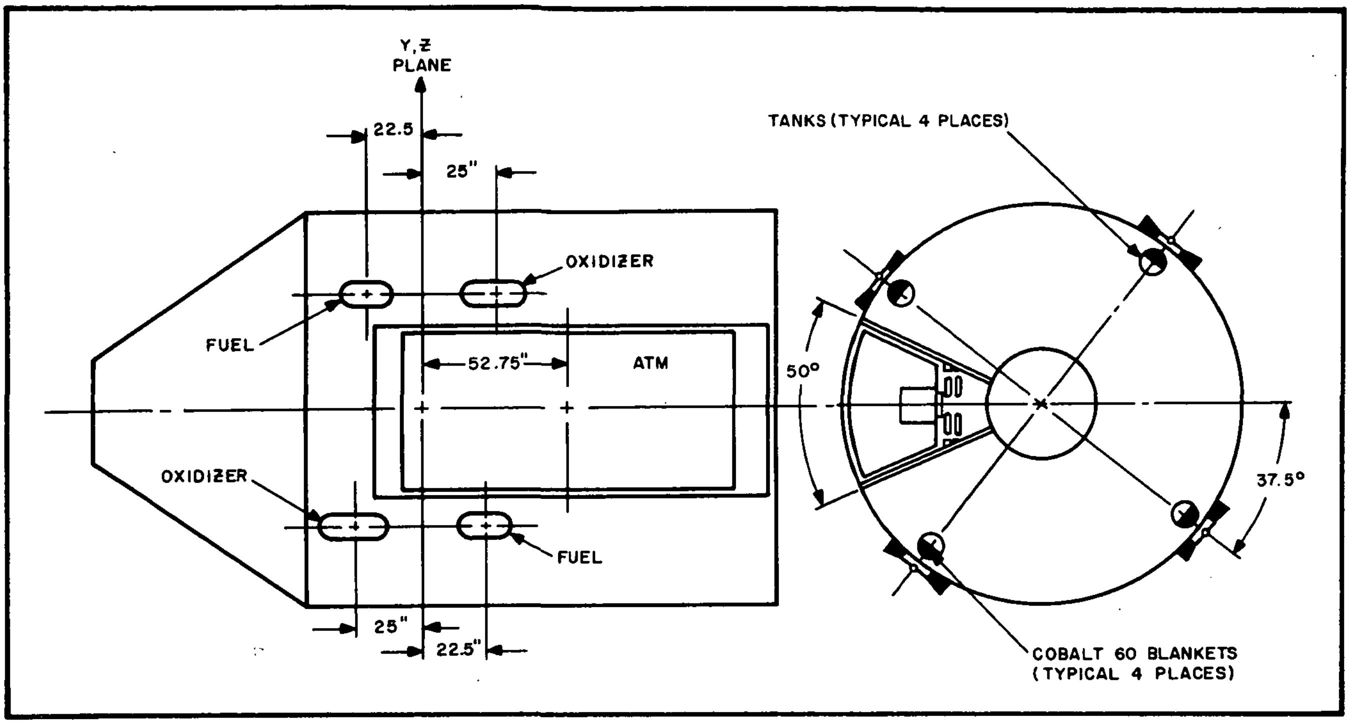


Fig. 3-6 Apollo Fuel and Oxidizer Tank Relationship to ATM

The effects of this source have been analyzed to determine the amount of film fogging that might be produced in the ATM experiments. This analysis has included laboratory tests of exposure of three types of film emulsion to cobalt 60 radiation. The exposed films were quantitatively analyzed by means of a densitometer to determine the relative fogging. The results of this analysis indicate that the cobalt 60 source will fog ATM experiment film, but that lead shielding about the film can probably reduce the extent of fogging to acceptable levels.

These tests and analyses are described in detail in Appendix C, and the results are summarized in the following paragraphs.

The cobalt 60 dose data* were projected to several ATM locations depicted in Fig. 3-7. The maximum dosage expected is determined by assuming that the film is loaded in the experiment two days prior to lift-off, and that the ATM is extended one day after lift-off. It is further assumed that seven days of operation in the extended position

*Dose data furnished by NAA.

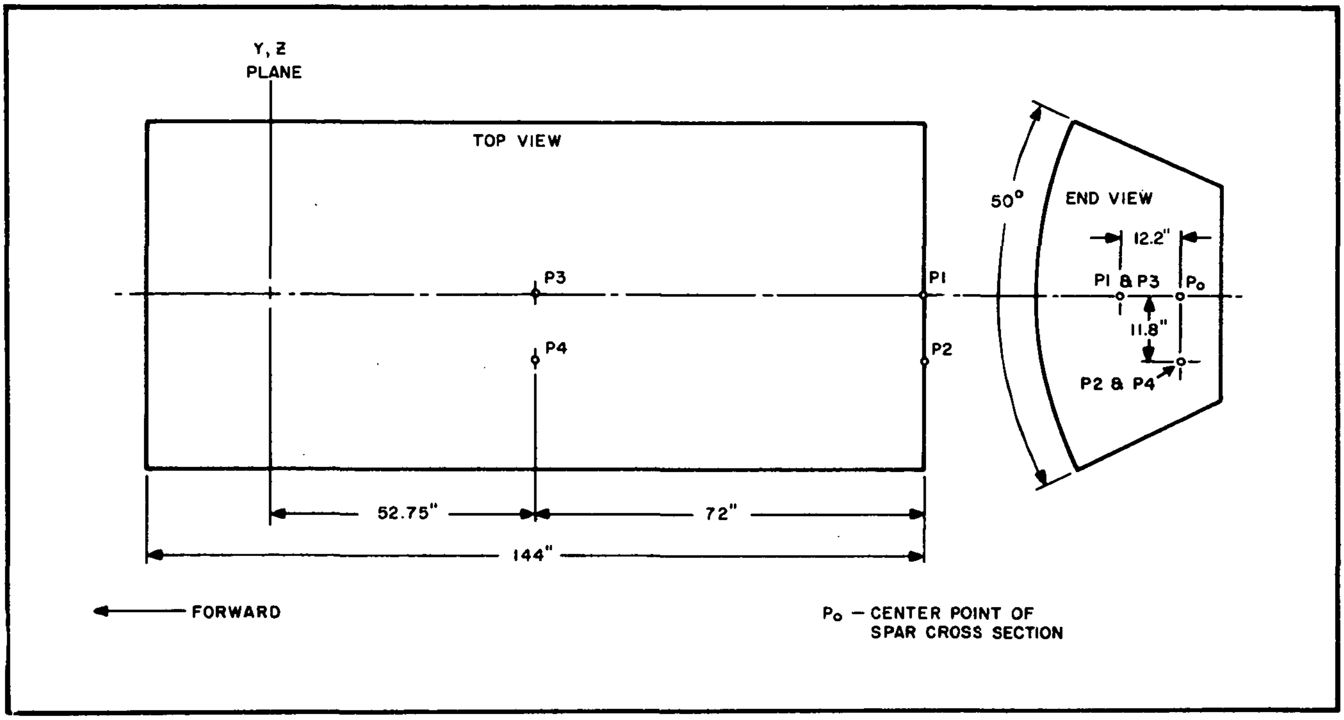


Fig. 3-7 ATM Camera Locations for Radiation Analysis (Stowed)

will take place prior to EVA film removal to the CM. The integrated dose was calculated for each position with the ATM stowed for three days and with it extended for seven days. The integrated dose for the balance of the mission (six days) would be slightly less than that for seven days.

The three film emulsions tested covered a wide range of photographic speed (ASA) and should be typical of film that will be used in ATM experiments. These film types are listed below:

- Plus-X (ASA 160)
- Tri-X (ASA 320)
- Royal-X Pan (ASA 1250)

The films were exposed to a cobalt 60 source for different lengths of time at a constant distance to derive the relationship of fogging as a function of time.

The exposed films were analyzed in a densitometer referenced to unexposed film. The resulting fog density data, interpolated to the expected ATM exposures, are presented in Table 3-8. Data are shown for the four ATM camera positions analyzed (Fig. 3-7). Three dosage conditions are indicated in this table for the various possible stowed and extended time periods. The first portion of the table indicates the film fogging for a 72-hour period stowed (two days on the launch tower and one day orbital outgassing period before extension). The second portion of the table indicates the film fogging for a seven day period extended (one half of the 14-day mission). The third portion of the table indicates the total effect of the first two exposures. It is expected that this represents the maximum effect on the film since after seven days of orbital operation, the film magazines will be removed from the experiments and stowed in the CM where the radiation levels are significantly lower. The fourth portion of the table indicates the film fogging for six days extended for the second set of magazines.

It is expected that none of the Apollo or ATM structure will attenuate the gamma ray dose; however, lead could be used to absorb the gamma rays. One centimeter of lead would reduce the total dosage by one half, which would reduce the fogging indicated in Table 3-8 to normally acceptable levels: 0.10 to 0.15.

Table 3-8
POSSIBLE ATM FILM FOGGING AT SELECTED LOCATIONS

ATM Location	Mission Period											
	Stowed 3 Days			Extended 7 Days			Total After 7 Days Orbital Operator			Extended 5 Days		
	Plus-X	Tri-X	Royal-X	Plus-X	Tri-X	Royal-X	Plus-X	Tri-X	Royal-X	Plus-X	Tri-X	Royal-X
P-1	0.02	0.02	0.07	0.08	0.09	0.17	0.11	0.14	0.22	0.06	0.07	0.15
P-2	0.02	0.02	0.07	0.09	0.11	0.19	0.12	0.15	0.24	0.07	0.09	0.17
P-3	0.08	0.10	0.18	0.09	0.10	0.18	0.17	0.23	0.32	0.07	0.09	0.17
P-4	0.10	0.12	0.20	0.10	0.13	0.21	0.19	0.27	0.35	0.09	0.11	0.19

3.4 APOLLO INTERFACE CONSTRAINTS

The ATM conceptual design is based on the Apollo system interface requirements and constraints, including the specific mission characteristics. The mission characteristics have been defined by NASA in the scope of the study. The interface requirements and constraints have been defined by NASA and NAA documentation, as amplified by coordination with the experiment working groups at NAA and NASA/MSC. The various requirements are presented in this section as they have affected the ATM conceptual design.

3.4.1 Mission Characteristics

The ATM system is conceived to be used in conjunction with the Apollo Block 2 configuration. ATM missions are to be primarily scientific missions and are to be conducted in low-altitude earth orbits. The orbital inclination is to be nominally 30 degrees from the equator, at an altitude of 200 nautical miles. The mission duration is to be 14 days. The launches will be from the Kennedy Space Center (KSC) at an initial azimuth of 72 degrees.

The mission profile is to be developed for each mission based on the specific experiment requirements for Command Service Module (CSM) orientation and operation, and for crew support. The amount of reaction control system (RCS) maneuvering and guidance and navigation (G&N) operation possible during the 14-day missions is limited and will be defined by the specific mission profile. For Apollo/ATM missions, there will be three crew members. Each may be scheduled for experiment operations for as much as 12 hours a day, or for a total time of 168 hours. All crew experiment activity must be time phased with other crew tasks, and experiment operations requiring more than one man at a time should be minimized. Crew support of experiment requirements for extra-vehicular activity (EVA) is expected, but will be limited in the number of separate EVA's possible.

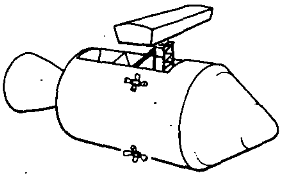
In all conditions of post lift-off, abort, or reentry, the Service Module (SM) is separated from the Command Module (CM) and is not recovered. Any experiment system that affects crew safety must have man-rated reliability.

3.4.2 Structure, Mass, and Storage

3.4.2.1 Structure

Sector I of the Apollo Block 2 Service Module (SM is a bay that encompasses a 50-degree opening as shown in Fig. 3-8. The volume of this sector is defined by: a forward and aft bulkhead 151 inches apart; two radial beams, No. 1 and No. 6, which run longitudinally and define the sides of the sector; an outer shell of 77-inch radius which attaches to the edges of the two outboard radial beams; and an inner shell having a 22-inch radius which is attached to the inboard edges of the two radial beams. The outer shell or sector cover acts as the major structural load path for axial loads. This shell provides continuity for the body bending shear generated by the aerodynamic loads acting on the Command Module (CM) and launch escape system (LES). The shell also reacts its own air and inertial loads experienced during launch and distributes these loads plus the body bending and axial loads along the forward end of the Lunar Excursion Module (LEM) adapter.

The structural interface between the ATM and Sector I of the SM must be made at the inboard and outboard rib caps of the radial beams. The attachments must assure that point loads created by the presence of the ATM are reacted by the radial beams and distributed into the sector cover prior to reaching the SM/LEM interface at Sta X_s = 355.00. Hard points for mounting are provided on the inboard and outboard



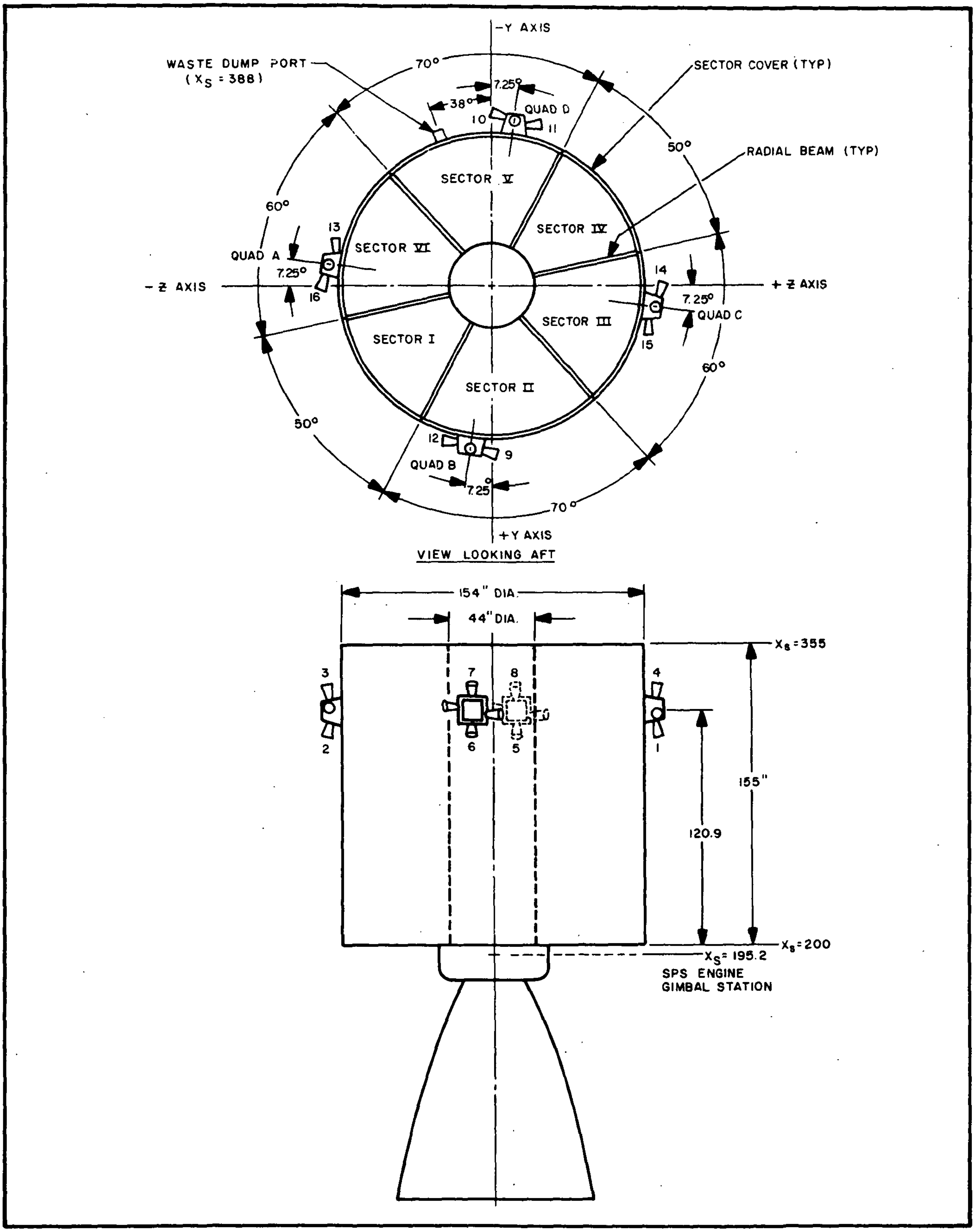


Fig. 3-8 Apollo Service Module Diagram

areas of radial beams No. 1 and No. 6 between Sta X_s = 279.25 and X_s = 290.66. The Sector I cover may be deployed or jettisoned prior to extending the ATM for observation. The design for deploying the sector cover must guarantee that the cover will still provide the structural integrity required for reaction to and distribution of Apollo launch loads. A sector cover that will deploy is planned for the Sector I bay on Apollo missions.

3.4.3 Mass Constraints

The allowable mass and distribution of mass in Sector I are directly limited by the extent of travel of the SPS engine gimbal located at the aft end of the SM. They are also a function of the amount of propellant weight carried by the Apollo spacecraft. In order to accomplish CSM orientation and reentry maneuvering, the center of gravity of the CSM/ATM must fall within the specific SPS engine gimbal angle limits.

The gimbal angle excursions considered for this study are ±8.5 degrees for the Y-axis and ±6.0 degrees for the Z-axis. A null position of 4.0 degrees in the positive Y-axis and zero degrees in the Z-axis are also considered. The resulting total gimbal angles are:

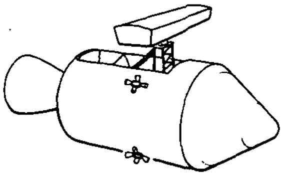
- +Y: 12½ degrees
- -Y: 4½ degrees
- -Z: 6 degrees
- +Z: 6 degrees

The gimbal thrust line envelope in relation to the Sector I is shown in Fig. 3-9. This addition of mass in any part of Sector I that overlaps the gimbal thrust line envelope will aid in maintaining the total c.g. within the desired location. The addition of mass outside that overlapping volume will cause the total c.g. to move away from the desired location. The addition of CSM propellant has the effect of shifting the spacecraft c.g. laterally from the allowable gimbal angle limit of the +Y coordinate.

For low-altitude abort conditions, a safety factor of three degrees is being considered for the gimbal angles. Therefore, when the ATM is in the stowed position, the gimbal thrust line envelope is reduced from that shown in Fig. 3-9 to +Y = 9½ degrees, -Z = 3 degrees.

3.4.4 Stowage

Scientific data can be stored in several locations in the CM during reentry. The



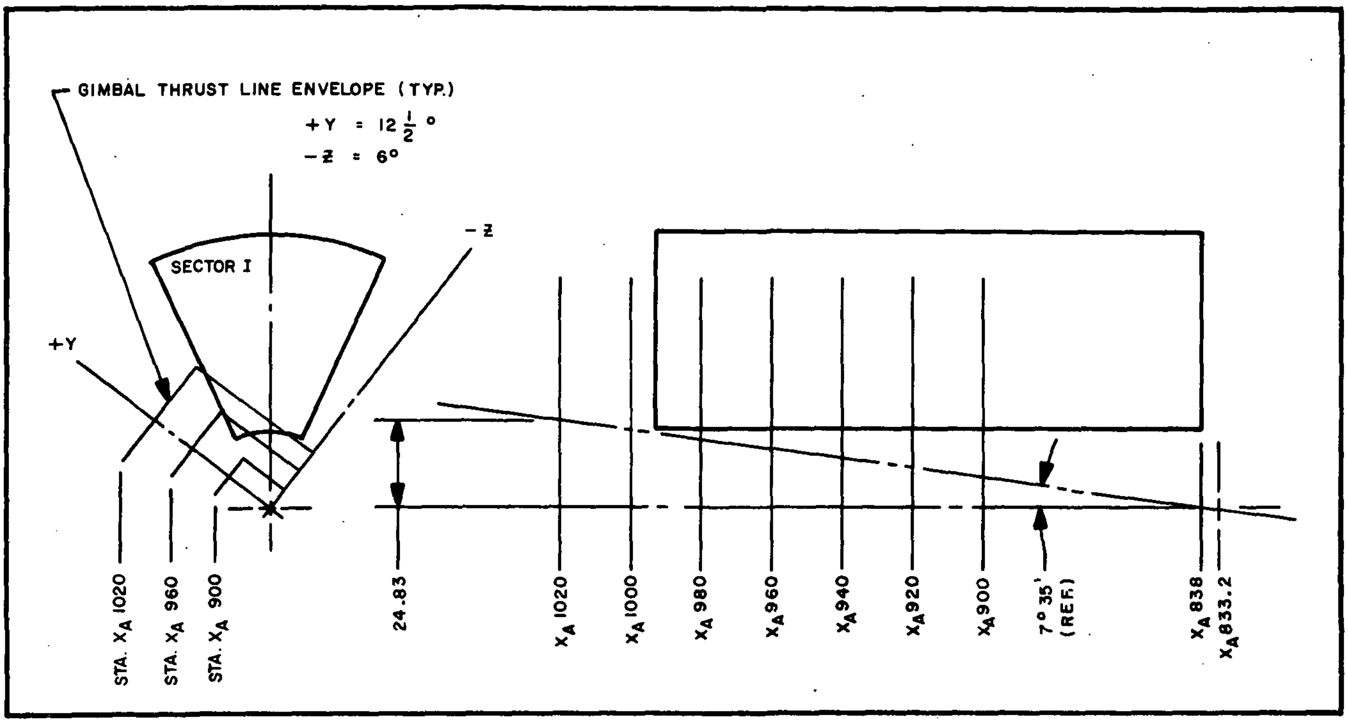


Fig. 3-9 Sector I Gimbal Thrust Line

possible storage areas at the end of a 14-day mission are as follows (Fig. 3-10):

Area	Weight	Volume
Food Compartment (lower equipment bay)	30 lb	1.0 cu ft
Food Compartment (L.H. equipment bay)	50 lb	1.7 cu ft
Food Compartment (R.H. equipment bay)	30 lb	0.9 cu ft
LIOH Canisters (transfer to SM)		4.5 cu ft
Total	110 lb	8.1 cu ft

3.4.5 Environmental

3.4.5.1 Thermal Restraints

The Service Module will have insulation on the Sector I surfaces of the radial beams and inner shell to maintain thermal control. No requirements exist for the programming of CSM aspect relative to the sun for thermal control in the near-earth orbits considered for ATM.

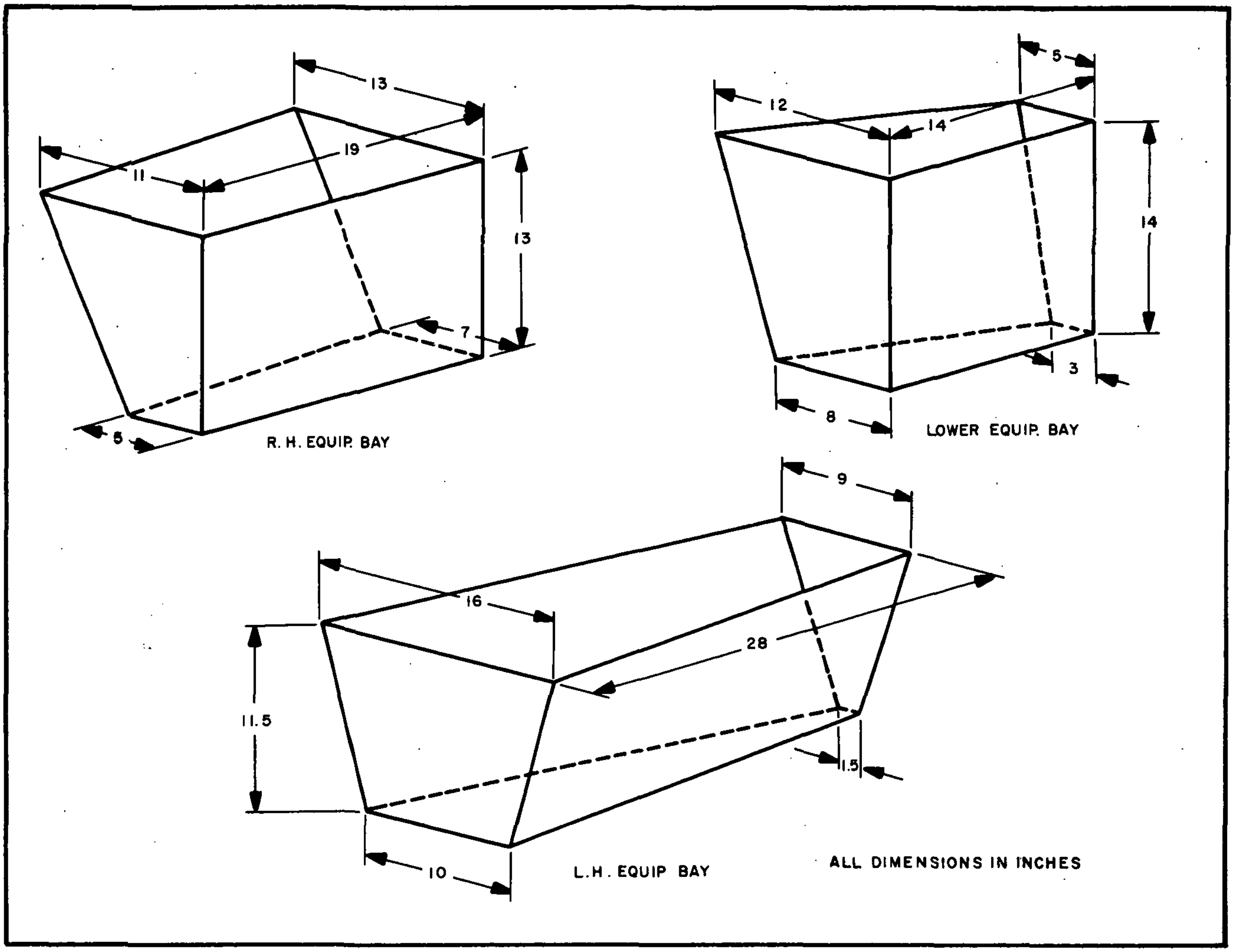
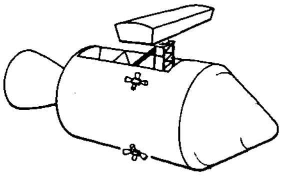


Fig. 3-10 Possible Storage Areas

3.4.5.2 Acceleration, Acoustics, Shock and Vibration Restraints

Maximum acceleration, acoustic, vibration and shock loads imposed on the ATM structure by the Apollo spacecraft are those loads which occur during launch and maximum dynamic pressure. Imposed on the ATM structural designs are the environmental test levels specified in NASA General Working Paper No. 10032, "Environment Specifications for Apollo Scientific Equipment."

The maximum acceleration launch load factors experienced by the Service Module occur during Saturn V first stage boost. These loads are 4.90 g longitudinal and 0.16 g



lateral. During maximum dynamic pressure, a 0.30 g lateral load factor is experienced by the Apollo spacecraft.

All Apollo systems are to be tested to the appropriate environmental requirements and procedures as specified in NASA Working Paper No. 10032 and the following tables and procedures are applicable to the environments under consideration for the ATM:

- Acceleration - Table 3; 6 g along all axes for 140 sec
- Acoustic Noise - Table 3; Procedure XIX
- Shock - Table 1; Procedure VIII
- Vibration - Table 1; Procedure IX
  Table 3; Procedure XI and XIII

3.4.6 Apollo Attitude Control

The Apollo CSM is controlled in attitude by two independent systems: (1) the guidance and navigation computer (G&NC); and (2) the spacecraft control system (SCS). Both systems use the reaction control system (RCS) for control of attitude. The RCS has four quads of four engines each which thrust the CSM in three axes. These thrusters are shown in Fig. 3-8.

The G&NC and the SCS systems both activate the RCS thrusters in an automatic limit cycle mode. The minimum limit cycle rate for the G&NC is 0.05 degrees per second, and for the SCS is 0.2 degrees per second. The dead band limits for the G&NC limit cycling are ±0.5 degrees; for the SCS limit cycling from ±0.2 to ±8.0 degrees.

Manual control through the use of differential RCS thrusting permits net minimum impulses less than those which can be obtained from other existing control modes. Astronaut experience in performing this technique indicates that residual rates of 0.02 degrees per second can be obtained.

Specific RCS quads can be turned off by the astronaut crew with switches, and specific thruster pairs can be turned off by the astronaut crew with circuit breakers.

3.4.7 Electrical Power

Apollo Block 2 missions normally allocate a maximum of 10 kw-hr of equivalent DC power for scientific experiments on a nonessential basis. Any further allocation of power may decrease the total mission life depending on the detailed power profile. The Apollo spacecraft can supply both DC power at a nominal 27.5 volts and AC power at 115 volts, 3-phase, 400 cps.

3.4.8 Command Module/Service Module Interconnections

The Apollo spacecraft allocates 58 conductors and two coax cables between the CM and SM to scientific experiments. Of the 58 conductors, eight are 16-gauge for power, and fifty are 20-gauge for signals.

3.4.9 Data Transmission

The CSM provides for transmission of real-time and recorded scientific data to the manned space flight network (MSFN) via the CSM communications subsystem. Three channels can be transmitted by frequency modulation of three subcarrier oscillators with the composite modulating an S-band transmitter. Further capability is provided for data transmission through the CSM/PCM telemetry equipment which operates at 51.2K bits/second. A total of 15 analog channels at one sample/second and 10 digital channels (8-bits/word) at 10 samples per second are normally allocated for scientific experiments. Time correlation is provided by simultaneous transmission of data containing time-from-launch information. The three subcarrier channels allocated can be recorded by CSM data storage equipment and be played back during normal data transmissions.

The data transmission capability may not be fully available for ATM depending on the primary mission requirements. Further investigation and negotiation should be conducted to determine the exact capabilities available for use by the ATM system.

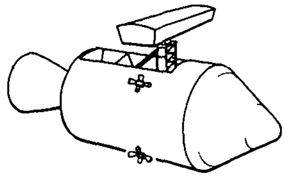
3.4.10 Human Factors

The Apollo CSM is designed to be flown by three men, and a major telescope system such as ATM must be designed to be compatible with the CSM and normal crew operations. Since the crew lives and functions in the CM cockpit, all ATM operations must be performed within the confines of this cockpit. The crew cannot actually see the ATM, mounted to their rear in the SM; so they must operate it by using controls and indicators. The crew must be able to pilot the spacecraft and operate the ATM at the same time. All features of the ATM must consider the practical problems of this dual task.

3.4.10.2 Crew Operations

All crew operations take place within the CM cockpit, except for extra-vehicular activity (EVA). These operations are conducted primarily from the couches, but the crew are able to leave the couches and work in the vicinity of the forward equipment bay. CSM attitude controls are located near the couches so that any one of the crew can operate them. ATM controls should be located to permit coordinated operation of both the CSM and the ATM.

The crew will spend the majority of time in the CM cockpit in spacesuits, and should not be required to make extensive physical maneuvers to operate the ATM. All ATM switches should be operable with a gloved hand and critical switches should be covered, requiring two steps to actuate. Indicators should be flag type rather than lights, since lights can confuse the crew. Indicators should be avoided unless required to operate the ATM or unless the crew can correct some malfunction.



3.4.10.3 Observing Schedule

The crew will be following a regular work/rest cycle that will permit the scheduling of scientific investigations. In general, individual experiment duties should be limited to one man. The estimated time available for all experiments on one mission is from 5 to 12 hours per day per man. The combined experiment requirements for crew participation will be scheduled into a mission profile. Deviations to this profile may be accommodated during a mission, but will require approval of the mission controller and the commander of the spacecraft.

3.4.10.4 Extra-Vehicular Activity (EVA)

Members of the crew can leave the CM through a hatch to perform specific duties directly on the ATM. The entire CM cockpit must be depressurized during this egress. The crew member performing the EVA will wear an additional EVA suit and gloves, and any manipulation required must be of an elementary nature and possible with bulky gloves. Planning for EVA must also consider limitations imposed by the space suit, the tether, total darkness and zero-G environment.

Hand and foot holds may be required if the crew member is to install or remove equipment from the ATM. Any portions of the ATM that are to be returned to the CM cockpit should be designed so as to be secured to the crew member by tether or to some other tether on the CSM.

3.4.11 Ground Operations and Testing

Current planning of ground operations and testing for the Apollo application program require experiment systems to be at the Kennedy Space Center (KSC) 60 days prior to launch for installation and integration tests with the Apollo Command and Service Modules. Installation into the Apollo is to be performed in the manned space operations building (MSOB). Vertical or horizontal installation is possible; however, if the ATM is to be removed or replaced after the Apollo CSM has left the MSOB, this operation must be done with the CSM in the vertical position. The contractor is to be responsible for any special hoisting jigs or attachment fixtures. KSC will provide and operate all necessary hoists or cranes.

Electromagnetic interference (EMI) system compatibility testing will be accomplished in the MSOB with the CSM in a vertical position. End-to-end testing of all critical flight hardware is to be conducted. If a critical subsystem fails after the EMI system compatibility testing and a backup unit is scheduled for substitution, the CSM will be recycled to the MSOB for additional EMI testing with the backup unit.

3.4.12 Launch Tower Access

Access to the ATM will be from two work platforms on the missile service structure (MSS) which encircles the Service Module. Film loading into the experiments and

and final checks can be made no later than T -12 hours in the count down. At T -8 hours, the MSS is removed from the launch vehicle eliminating further access to the Service Module. Present configurations of the MSS work platforms provide extensions which can be slid back making a clearance of approximately 18 inches between the platform and the spacecraft skin, or an entire platform segment can be lifted or folded back providing a clearance of five feet. Current plans are to load hypergolic propellants into the Service Module 10 days before launch. Since the Sector I cover is a load-bearing member of the Service Module, it can not be removed after propellants have been loaded. Access to the ATM system and experiments after propellants have been loaded will have to be accomplished through access doors in the cover.

Umbilical access to the ATM system and experiments for ground operations and testing must be through the Sector I cover, or possibly through the CM hatch directly to the ATM control unit.

3.5 CONTAMINATION

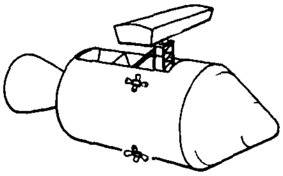
3.5.1 General

Control of contamination and contamination potential is a significant problem of the ATM system. Items that can be adversely affected by contamination include experiment optical elements, control sensors, thermal control and lubricated components. Known sources of contamination other than outgassing include Apollo RCS exhaust products and waste dump systems. Some measure of contamination control of these sources is accomplished by the ATM oriented section thermal shield and aperture covers. However, some question remains as to the dispersal time constants of this contamination.

Also unanswered at this time is the question of the contamination potential of the remainder of materials in the Apollo. Since the Apollo was not initially designed to perform optical missions, certain material selections were undoubtedly made that are not necessarily compatible with a research observatory function. A typical example of such a material selection is the ablation shield of the CM, which may trap an extensive amount of gas that will outgas during the mission.

Many of these problems presumably can be diminished with a vacuum bake-out of the CSM prior to launch. However, this approach should not negate a materials and materials handling review of the CSM to evaluate the materials compatibility. Materials selection and handling for ATM must include optical contamination potential as a criterion.

The following sections discuss in more detail some aspects of contamination control.



3.5.2 Ground Handling

Controlled environment facilities will be required during all phases of assembly, integration, test and transport of the ATM system and experiments. These facilities will consist of clean rooms with humidity control, special containers and fixtures, and dry gas purging.

The primary design constraints with ground handling are associated with the oriented section. The thermal shroud must allow access to experiments for prelaunch removal of any internal protective covers, and provide sufficient protection for critical elements when the ATM is out of a controlled environment. The latter case may be limited to prelaunch operations.

3.5.3 RCS Exhaust Products

The Apollo RCS thrusters will be used extensively during the orbital operations. Although the ATM system aperture covers are to be closed by the astronaut crew during any thruster operation, some of the gas may diffuse inside the ATM experiments, or the crew may inadvertently fail to close the aperture covers. The exhaust consists of combustion products from the reaction of Aerozine -50 (i.e., mostly Hydrazine and unsymmetrical dimethylhydrazine) fuel and nitrogen tetroxide oxidizer. Some of the constituents of the combustion are as follows:

Constituent[a]	Percent	Boiling Temperature
H_2O	37.0	100°C
N_2	33.0	-196°C
CO	7.0	-192°C
[OH]	4.5	---
CO_2	5.0	-78°C
NO	1.0	---
O_2	2.0	-183°C
[O]	1.0	---
H_2	0.7	-252°C
[H]	2.4	---
TOTAL	93.6%	

(a) Reference - NASA document, Apollo Experiments Pallet Request for Proposal No. 13G721-17-22P, "Questions and Answers".

The remaining 6.4 percent of the combustion products are unknown at this time but probably consist largely of unburnt propellants.

Most of these ingredients pose some kind of contamination problem, but the highly reactive free radicals [OH], [O] and [H] may be particularly troublesome. Even with the apertures closed, these products may damage the thermal coatings and the ATM gimbal equipment.

Although no analysis of the dynamics of the gas jets emitted by the RCS thrusters is known, it is reasonable to assume that all 16 engines will cause some gas to impinge on the ATM equipment. In most of these the level may be too low to be damaging, but in two cases damage is probable.

Actuation of RCS thrusters No. 9 and 16 allows direct impingement of exhaust gasses on ATM. (See Fig. 3-6.) Because of this, these two thrusters should not be actuated after the Sector I cover has been deployed.

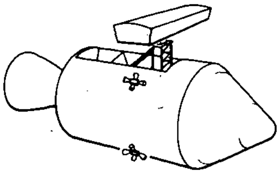
Thrusters 9 and 16 are easily deactivated by opening two circuit breakers (Fig. 3-11) that supply solenoid power from Main Bus A. Manual operation of these breakers can be accomplished in the Service Module.

Deactivation of thrusters 9 and 16 also turns off thrusters 12 and 13. The consequence of this is that all roll control torques must be provided from RCS Quads C and D. Since each quad has its own fuel and oxidizer tankage and there is no interconnection between the tanks of the various quads, there will be a propellant deplation of Quad C and D relative to Quads A and B. With the present mechanization of the circuit breakers, it is not possible to deactivate appropriate pitch or yaw thrusters of Quads C and D in order to balance propellant utilization. If balance is a critical item, then the roll thrusters of Quad C and D should be deactivated during the period prior to ATM deployment to conserve propellant for use when ATM is extended.

3.5.4 Liquid Wastes

In the Apollo system, liquid human wastes and fuel cell products (almost entirely water) are combined in an automatic dumping chamber. The astronaut crew cannot control the dumping of these wastes. However, it is expected to be reasonably periodic (in the order of every two hours). Also, indicators on the storage chambers provide information to the crew so that they can anticipate when a dump will occur. All ATM system aperture covers must be closed by the crew prior to such a dump. The common dumping port is located at Station Xc = 26, 38 degrees off the -Y axis towards the -Z axis. This location is approximately 90 degrees from the ATM system (Fig. 3-6).

The fuel cell and crew liquid wastes consist primarily of water and salts. When ejected into space, the liquid will freeze into crystals (mostly ice). The majority of these crystals will diffuse away from the vicinity of the ATM, but some may impinge on the ATM thermal shield. These crystals should melt and evaporate rapidly due to the relatively hot surface of the thermal shield, leaving some of the salts on the outer surface. It is difficult at this time to estimate the seriousness of this problem, and further study is required.



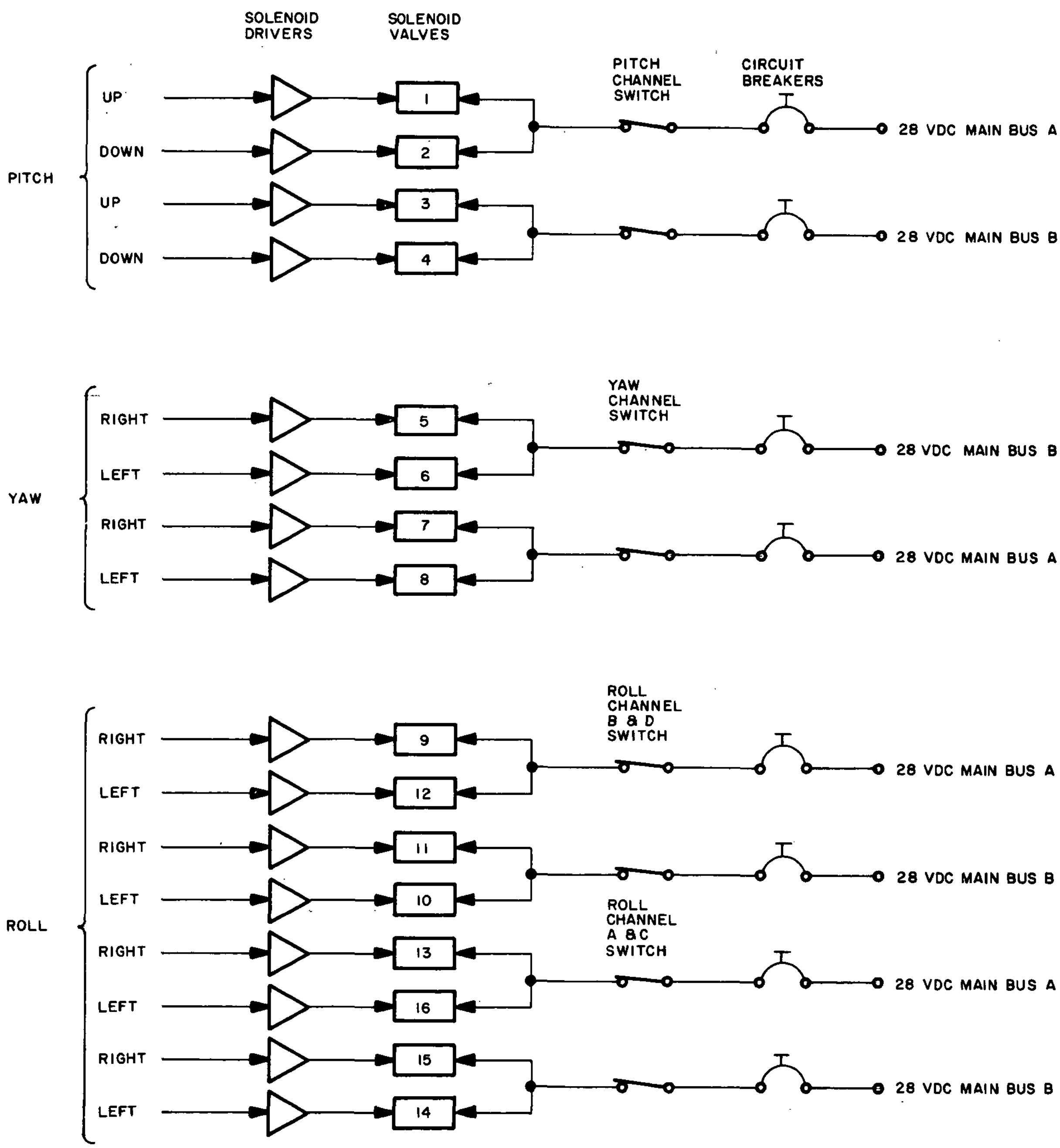


Fig. 3-11 RCS Engine Control Switch and Circuit Breaker Combinations

3.5.5 Outgassing

The Apollo and ATM outgassing and sublimation products will probably have a negligible effect on the thermal shield, but may prove troublesome to the more sensitive optics. Since the outgassing and sublimation is greatest early in the orbital operation, it will be necessary to leave the ATM oriented section stowed for about one day.

The Sector I cover should probably be kept in place during this initial one-day period so that it will protect the ATM from the majority of the Apollo sublimation products, particularly those from the ablative heat shields. The Sector I bay will not be hermetically sealed and pressure within the ATM should approach that of the environment in this 24-hour period. Following extension of the ATM oriented section a further outgassing period should be scheduled to permit the diffusion away of remaining gasses trapped in the various parts of the ATM.

3.6 ASTRONAUT UTILIZATION

The ATM system is an astronomical space research platform that is operated by the Apollo astronaut crew. Crew participation in an ATM mission is a major factor in the acquisition of significant scientific data. Crew operation of the ATM system and experiments must be optimized to be compatible with the Apollo mission and the normal work load assignment of the crew. Crew utilization has been analyzed as an ATM design consideration and major items are discussed in this section.

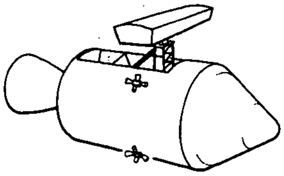
3.6.1 Crew Availability

The ATM/Apollo crew will consist of a normal complement of three men. For this study, it has been assumed that one of the crew members would be the primary ATM observer, and that the other two crew members would perform ATM operations when he is "off duty". It is expected that the primary observer could operate the ATM about 12 hours a day from the second through the 14th day of the mission.

The 156 total hours available by the primary observer is sufficient for the expected 100 hours of ATM operation. However, the ATM operation continues around the clock, and other members of the crew will be utilized during the sleeping and eating periods of the primary observer. The other two crew members will also be used to assist the primary ATM observer during major ATM operations, such as the EVA recovery of photographic film.

3.6.2 Manual Versus Automatic Operations

The major design consideration of this study with respect to crew utilization is the trade-off analysis of manual versus automatic operations. This analysis is based on the desirability to utilize the crew to perform the majority of the ATM operations that otherwise would require extensive automatic design.



Of the many ATM operations that could be either manual or automatic, the ones that have been designated to be performed by the astronaut crew are as follows:

(1) Selection of solar activity regions for data acquisition
(2) Coarse solar acquisition within ATM field of view
(3) Offset pointing to selected solar activity regions
(4) Boresighting of experiment telescopes to desired alignment
(5) Management of experiment data acquisition
(6) Management of ATM system and experiment variables
(7) Recovery of photographic film

This selection utilizes the astronaut crew to perform those operations in which judgment is required to select scientific targets and to align telescopes to these targets. In addition, it provides for astronaut control over all ATM and experiment operations affecting the acquisition of scientific data, similar to observer duties in a ground based observatory. Astronaut judgment exercised on the spot will enhance the probability of obtaining transient data, and data of unexpected phenomena. The astronaut operations can be augmented by radio communication with the ground scientists, who can redirect the observing program based on ground data or verbal descriptions from the astronaut crew.

3.6.3 Status Monitoring

The ATM operations designated in Section 3.6.2 for astronaut utilization require that the astronaut observer have sufficient display information to determine the functional status of each subsystem and experiment. Functions with variable settings should be displayed in real time, so that the astronaut observer can exercise judgment during the ATM operation. Additional status monitoring of less critical functions should be displayed on a demand basis.

The display of status monitors should be in the form: (1) of lights if the astronaut observer is expected to perform some further action; (2) of flags if it is a critical status report; (3) of general purpose shared meters if needed only occasionally; and (4) of special purpose individual meters if required for the proper control of ATM.

3.6.4 Astronaut Control of ATM

The astronaut control of ATM functions should be made from the CM, using switches and buttons that execute the proper response in the ATM system or experiments. Discrete switches or buttons should be used for all major or critical functions that require fast astronaut response or that are repetitively used. Coded button formats can be used for all other functions that are used only occasionally. Continuous control operations, such as slewing or offsetting, should be performed using a "fly to" control stick, similar to that used in the Apollo attitude control system.

3.6.5 Astronaut Training

The astronaut utilization discussed above requires that the astronaut crew be thoroughly trained in all facets of the ATM and experiment operations. Design consideration must be given to the ability to train the crew prior to the mission, using ATM hardware and following similar techniques to those to be used during the orbital mission.

Typical operations that should be performed during astronaut ground training are as follows:

(1) Functional operation of all ATM system modes
(2) Operation of ATM slewing and offset pointing
(3) Experiment boresight maneuvers
(4) Experiment target selection using displays of actual or recorded solar conditions similar to those expected during orbital operation
(5) Functional operation of all ATM experiment modes
(6) Simulated EVA recovery of film magazines

3.7 RELIABILITY CONSIDERATIONS

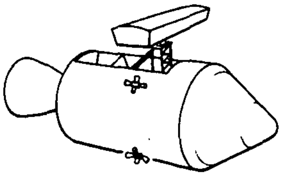
Several specific reliability factors pertaining to an Apollo/ATM mission can be identified at this time. System reliability is divided into two types: (1) those portions of the system that affect crew safety; and (2) those portions of the system that affect mission success. This study has not included an extensive reliability analysis of the ATM system and the ATM/CSM combination. However, mission reliability considerations have been included in the design tradeoffs used in arriving at the present concept. Several critical areas, which have influenced the ATM design concept, have been identified with crew safety reliability.

3.7.1 Crew Safety Reliability

Any of the ATM items or functions that interface with the CSM were considered to have an effect on crew safety. The primary criteria imposed during the study consisted of either minimizing or eliminating interfaces. When an interface had to be established, the approach was restricted to concepts whose reliability was already established, or whose reliability could be unquestionably established during the program.

Significant interfaces that were established consist of the following:

(1) Attachment of ATM support cradle to the SM. Proper reliability can be established through analysis and test of the ATM structure.
(2) Jettison of Sector I cover. A jettisonable cover is required for other Apollo missions. It appears that ATM can interface adequately with a standard cover design.



(3) CSM center of gravity shift with ATM stowed and deployed. The primary requirement for c.g. control results from SPS engine gimbal limits. If the ATM/CSM c.g. remains within requirements for both deployed and stowed conditions, the requirement for restow or jettison can be eliminated. If this is not possible, then the reliability of restow operation will affect crew safety.

(4) Effects of ATM on Apollo attitude control system. The requirement to deactivate the roll thrusters of RCS Quads A and B may have some effect on crew safety. This will require more analysis.

(5) Presence of ATM command subsystem equipment in the CM. The ATM concept has established requirements for command subsystem equipment to be mounted in the CM lower equipment bay. Since hardware design techniques for this type of equipment are quite well established, the effect on crew safety reliability should be slight.

(6) EVA film recovery operations. Reliability and human engineering requirements must be established to allow proper and safe interfacing of the astronaut with ATM during EVA. One example of this in the concept is the inclusion of gimbal brakes on ATM to keep it immobile with respect to the SM.

SECTION 4

STRUCTURAL SUBSYSTEM

Section 4
STRUCTURAL SUBSYSTEM

4.1 SUBSYSTEM REQUIREMENTS

The ATM structural subsystem must provide for the following:

(1) Structural interface attachment to the SM
(2) Structural interface attachment and volume accommodation for all other ATM subsystems
(3) Extension mechanism for extending the oriented section
(4) Structural support and stability for the oriented section in the stowed and extended position
(5) A three-axis gimbal and housing for maneuvering the oriented section
(6) Structural support and attachment of the experiment payload
(7) Thermal shield structure
(8) Aperture covers and actuating mechanisms
(9) Interface control of the ATM mass distribution
(10) Interface control of the SM Sector I cover
(11) Interface control of the CM storage requirements for ATM data

The structural subsystem design must be compatible with the physical and environmental interface constraints of the Apollo spacecraft as stated in Section 2.3. Four structural attachment locations were established by NAA at sta. $X_s = 290.66$: two attachment points located on the inboard rib caps of Section I, and two points located on the outboard rib caps. Attachment fittings must be designed to maintain a minimum distance between the point of applied load and the rib cap centroid. This design requirement minimizes the induction of undesirable bending moments into the radial beams. Reactions of the ATM system imposed on the SM must not exceed the structural capability of the SM. The total mass of the ATM system in the stowed position must be located so that the composite c.g. of the CSM and ATM shall not exceed the gimbal position thrust line envelope of the SPS engine. The structural subsystem must accommodate a complement of experiments that will be up to approximately 11 feet long and will require a large portion of the available volume.

The longitudinal and lateral accelerations experienced by the SM during launch of the Saturn V and Saturn IB boost vehicles are shown in Table 4-1. The 4.9 g maximum longitudinal load factor occurs with a 0.16 g lateral load factor at the end of Saturn V first stage boost. A 0.3 g lateral load factor was used for the maximum critical condition since it also occurs during first stage boost.

The condition stated above is the limit design for static loads. The maximum load to which the structure is subjected during operation is the limit load. The design yield load is equal to the limit load, and the design ultimate load is equal

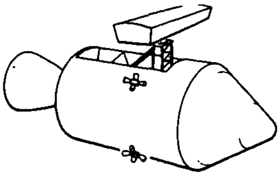


Table 4-1
APOLLO LAUNCH LOAD FACTORS

Vehicle	Flight Condition	Apollo X_a Station	Lateral g Load Factor	Axial g Load Factor
Saturn V	lift-off	993	0.46 (b)	2.06 (b)
	lift-off	838	0.25 (b)	1.74 (b)
	max q	993	0.30	1.90
	max q	838	0.30	1.90
	S-IC end boost	993	0.16	4.90
	S-IC end boost	838	0.12	4.90
	S-IC thrust decay and separation	993	--	0.83 tension
	S-IC thrust decay and separation	993	--	2.08 (b) compression
	S-IC thrust decay and separation	838	--	1.36 (b) tension
	S-IC thrust decay and separtion	838	--	0.05 (b) compression
	S-II h.o.e. (a) @ burnout	993	0.80	2.15
	S-II h.o.e. @ burnout	838	0.63	2.15
	S-IVB h.o.e. @ burnout	993	0.24	1.65
Saturn V	S-IVB h.o.e. @ burnout	838	0.02	1.65
Saturn IB	S-IVB h.o.e. @ burnout	993	0.26	3.40
Saturn IB	S-IVB h.o.e. @ burnout	838	0.05	3.40

(a) h.o.e. - Hard Over Engine.
(b) Includes effects of overall body mode excitation.

to 1.5 times the limit load. The structural members are designed so no failure will occur at or below the design ultimate load. The ultimate design load factors are:

7.35 g axial, 0.45 g lateral

In the design of the extension drive system, the maximum lateral deflection of the extension mechanism and cradle were based on a one g lateral load factor applied at the gimbal intersection with the system extended.

The design of the spar is based on sufficient structural rigidity to provide a natural resonant frequency of 10 cps or greater, and a maximum change of slope between the ends of 60 arc seconds.

Vibration, acceleration, shock, and acoustic noise test levels, that the ATM system must endure, were taken from NASA working paper No. 100.32 "Environment Specifications for Apollo Scientific Equipment." The vibration and acceleration levels specified for Apollo scientific experiments are:

Acceleration:	6 g all axes (140 sec)
Vibration:	all axes
g (0 to peak)	5-27.5 cps, 2.08 g
	27.5 to 52 cps 4.15 g
	52 to 100 cps 2.9 g
	(For random see Fig. 4-1.)

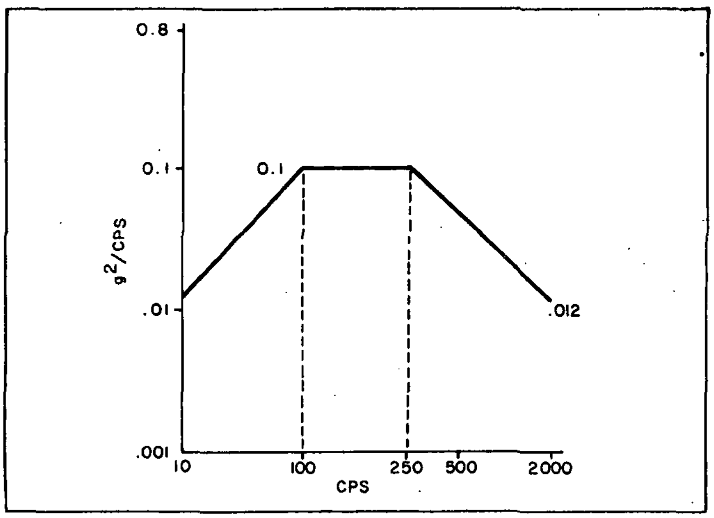


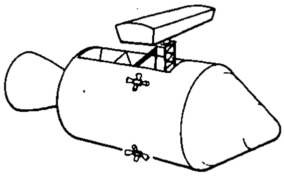
Fig. 4-1 Randon Vibration Levels For Apollo Experiments

The design weight used for the ATM attach points was 2,000 lbs. This includes a payload experiment weight of 750 lbs mounted to the spar. For a complete weight analysis of the ATM system and discussion of weight limitation see Section 4.2.9.

4.2 GENERAL DESCRIPTION

The structural subsystem will be installed in Sector I of the SM. The dimensions within Sector I which control the maximum allowable ATM volume are 151 inches length, 50.5 inches height, 17 inches width at the inboard edge of the sector, and 62.5 inches at the outboard edge of the sector. The total volume within Sector I is 308,600 cubic inches.

Some ATM subsystem equipment is contained within a structural mounting cradle that conforms to the 50 degree segment shape of Sector I. A nominal minimum clearance of 0.75 inch is provided between the cradle and the Sector I structure. The structural attachment to the SM is provided on the cradle. The extension mechanism and structure are located at the forward inboard end of the cradle. The extension mechanism is a parallel-bar linkage supplemented with a folding support strut



for additional stability. The driving force to actuate the extension mechanism is provided by a pneumatic cylinder. A three-axis gimbal is mounted to the aft end of the parallel-bar linkage. The spar attaches to the outside or roll gimbal. The spar is 124 inches long and provides for the interface of the experiment payload and the ATM equipment that requires being mounted on the spar. The experiment components, spar, and ATM equipment mounted to the spar are enclosed in a heat shield. Aperture covers for individual experiments and ATM sensors are provided on the front surface of the heat shield. An access door or doors will be provided in the heat shield for retrieval of experiment data. Additional ATM equipment will be mounted to the forward end of the cradle. Volume at the inboard aft end of the cradle is reserved for batteries or other electrical power subsystem equipment. (See Fig. 4-2 for the conceptual layout of the ATM structural subsystem.)

4.2.1 Structural Cradle and Apollo Structural Interface

The structural support cradle shown in Fig. 4-2 is a five-sided, open-truss structure of trapezoidal shape conforming to the 50 degree sector of Sector I in the SM. The two sides and bottom are truss members with the forward and aft ends also closed by truss members to provide structural rigidity. The outboard side is open to accommodate the extension of the oriented section. Four mounting points for the extension mechanism are located at the lower forward end. Power supply volume is provided at the lower aft end. The oriented section is accommodated in the upper or outboard volume and extends approximately the length of the support cradle. Structural support points for the oriented section in the stowed position are provided at the forward and aft ends. These load-carrying support points provide structural support of the oriented section during launch and thus protect the gimbal bearings from the launch loads. Longitudinal and lateral loads are distributed through the truss members as tension or compression loads and are reacted at the Apollo-ATM attach points as point or axial loads.

The open-truss type of structure was selected for the cradle rather than a skin-to-stringer type for the following reasons: (1) flexibility of member locations and equipment installation, (2) accessibility to equipment through the sides and ends, and (3) minimum weight.

The cradle is constructed of aluminum extruded tubing and the truss members are constrained at the joints by riveting the members in each joint to a common gusset plate. Extruded square tubing is used for the side and end trusses to provide flat mounting surfaces for the ATM subsystem components.

The structural attachment interface to the SM is accomplished through a four point major attachment pattern located at sta. 290.66. These points are supplemented by four additional attachment points, two at each end of the cradle on the outboard edges.

The major attachment points at sta. 290.66 are located on the inboard and outboard

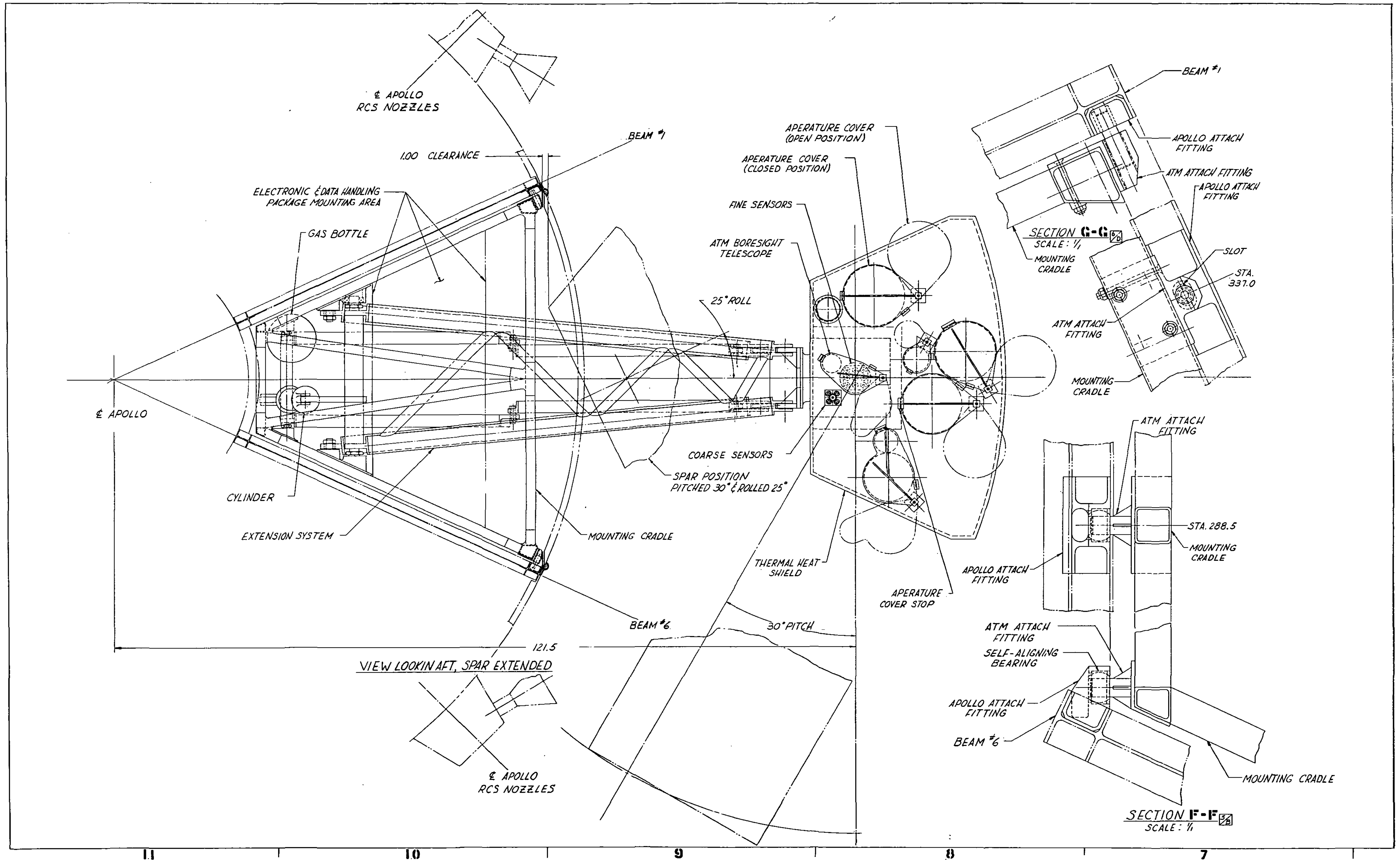


Fig. 4-2 ATM Structural and Mechanical Concept Layout (Sheet 1 of 4)

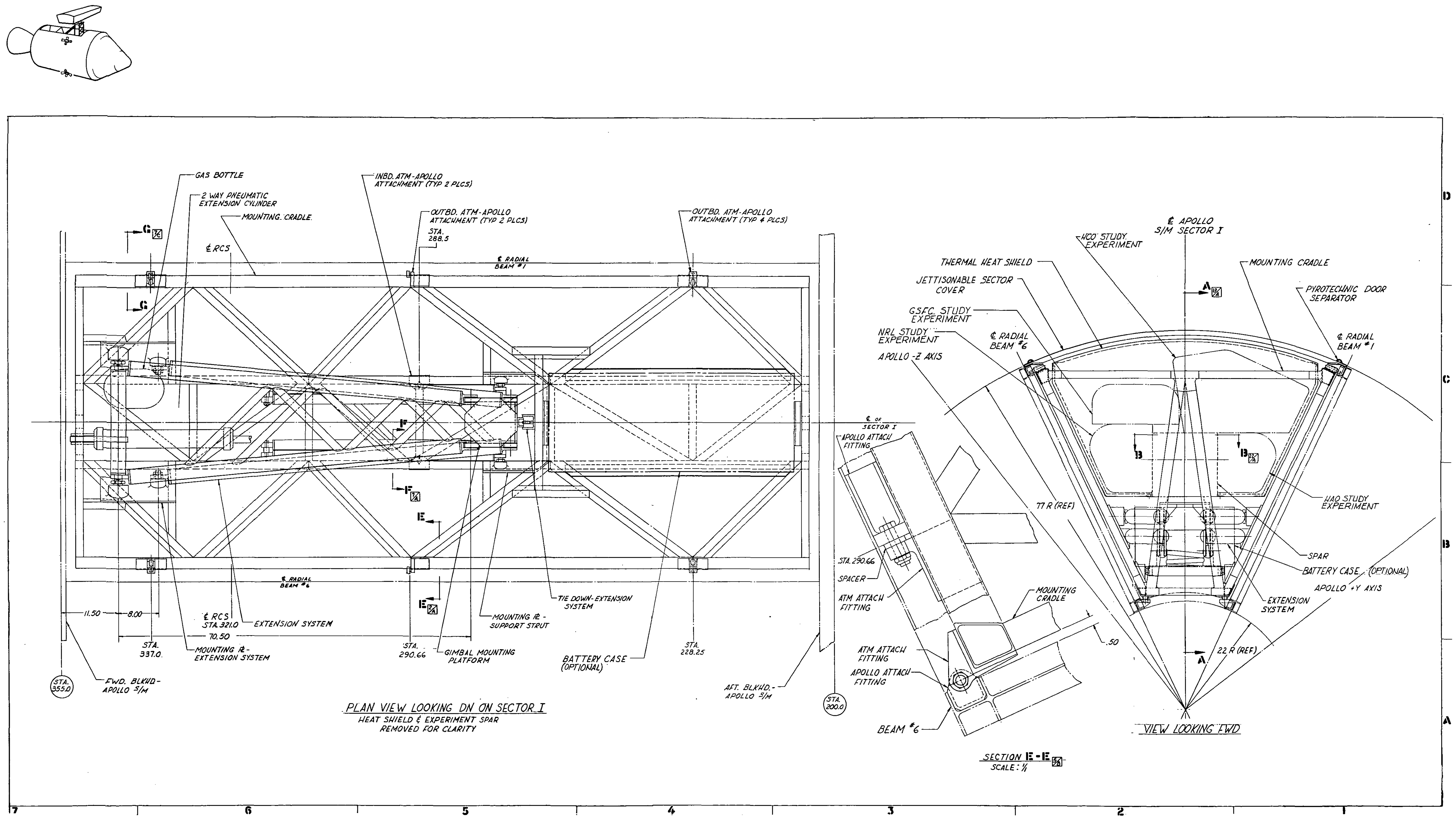


Fig. 4-2 ATM Structural and Mechanical Concept Layout (Sheet 2 of 4)

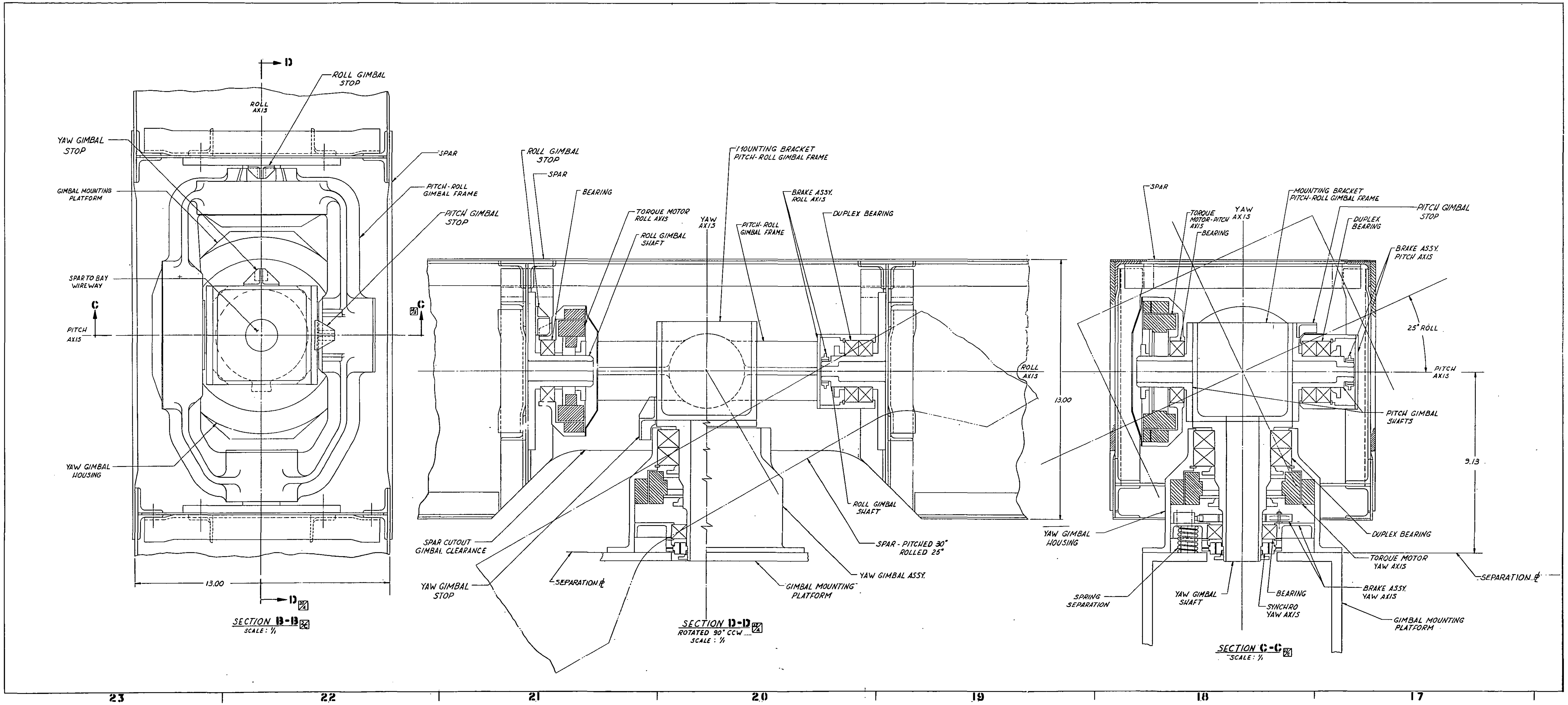


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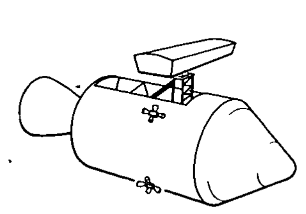
Fig. 4-2 ATM Structural and Mechanical Concept Layout (Sheet 3 of 4)

4-7-3



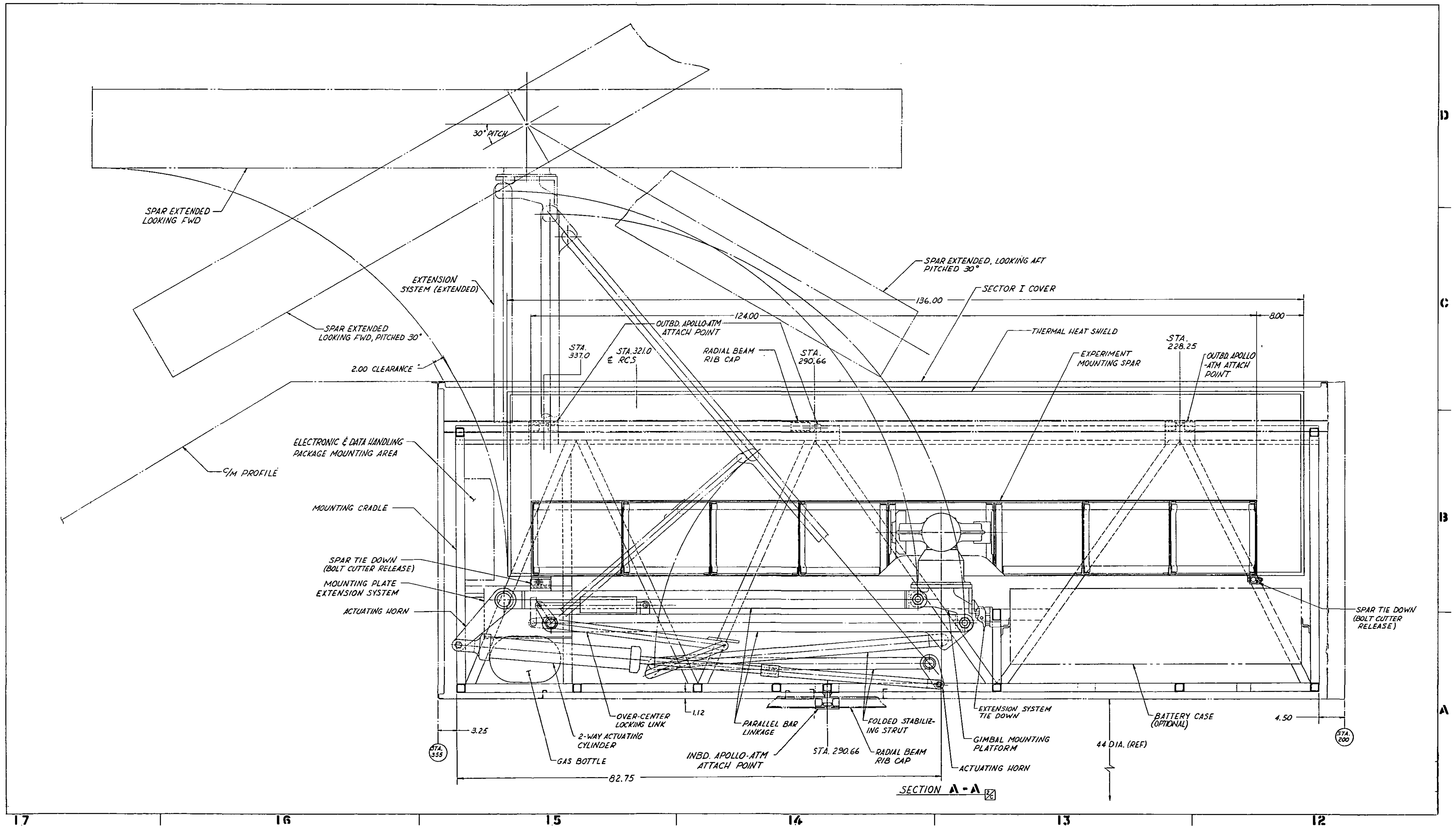


Fig. 4-2 ATM Structural and Mechanical Concept Layout (Sheet 4 of 4)

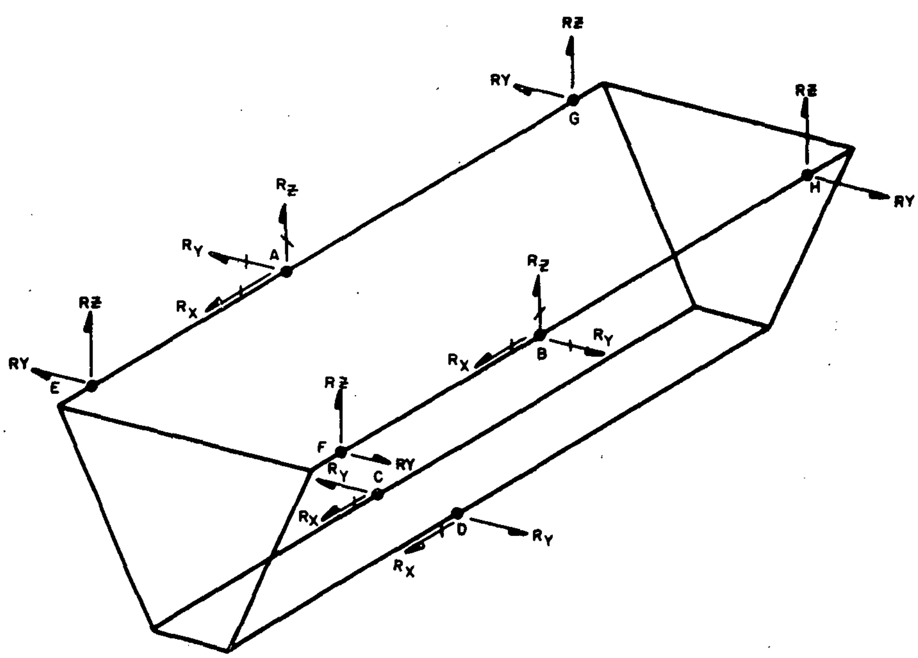
4-8-1

4-8-2

rib caps of the radial beams confining Sector I. (See Fig. 4-2 for detail of attach points.) The two outboard attach points are capable of reacting loads in all three orthogonal axes; the two inboard attachment points react loads only in a longitudinal and side direction. The attach points on the forward and aft ends of the structural cradle are required for additional stability of the cradle. They react loads in the two lateral directions, but not the longitudinal direction. No single attach point reacts bending moments by itself. Bending moment is reacted as a couple between two attach points depending on the direction of the applied moment. ATM mounting points are located between the flanges of the SM radial beam caps, with the axis of the mounting point close to the flange centroid to reduce the effect of eccentric loading. Inertia loads from ATM are transferred directly into the SM radial beams.

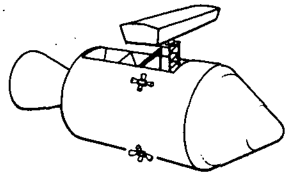
Attach points at sta. 290.66, which react the launch thrust loads, take advantage of existing hard points within Sector I that accommodate the Lunar Mapping and Survey System. The two forward and two aft outboard attach points provide structural stability, and minimize deflections caused by applied lateral side loads by reacting this loading as a couple. The physical outboard attachment at sta. 290.66 is made with vertical bolts through horizontal lugs, and the inboard attachment is made with horizontal pins into self-aligning bearings. Forward and aft outboard physical attachments at sta. 228.25 and 337.00 are made with a lateral pin placed into a fitting which is a close-tolerance vertical slot.

Reaction loads occurring at the attach points for the critical design condition as stated in Section 4.1 are shown in Fig. 4-3. These reaction loads are within



POINT	LOAD	CASE I (POUNDS)
A	R_X	1900
	R_Y	-152
	R_Z	-195
B	R_X	1350
	R_Y	-562
	R_Z	-1610
C	R_X	4340
	R_Y	-173
D	R_X	4380
	R_Y	273
E	R_Y	-212
	R_Z	-51
F	R_Y	0
	R_Z	171
G	R_Y	213
	R_Z	900
H	R_Y	-836
	R_Z	960

Fig. 4-3 Reaction Loads on ATM/SM Attach Points



the structural capability of the SM since the structure within Sector I is capable of supporting a package with twice the mass of the ATM system.

4.2.2 Extension Mechanism

As stated in Section 4.1, the extension mechanism is required to extend the oriented section, and also to provide structural stability for the oriented section while in the extended position. It was assumed that the extension mechanism must be capable of extending the oriented section, weighing 1100 lbs., in a 1 g environment for ground tests. The distance of 43.5 inches for extension of the oriented section beyond the SM is established by the full freedom of the gimbal without interference to the CSM and the stowed configuration in Sector I.

Three methods for extending the oriented section were considered during the study: (1) a scissors jack, (2) telescoping tubes, and (3) a parallel-bar linkage. Analysis of each of these methods shows that the parallel-bar linkage gives the minimum height when stowed, has the best structural stability, and results in the best position relative to the CSM when erected. The disadvantages are that the maximum experiment length is about six inches shorter, and that more force is required to activate it in a one g environment.

The parallel-bar linkage concept adopted during this study is shown in Fig. 4-2. Manufacturing tolerances for the other two systems considered would add a large amount of play or deflection within the systems, which would reduce their structural stability. Locating the extension system toward the forward end of the SM is beneficial to the combined Apollo-ATM c.g. location and minimizes contamination from the RCS plumes.

The two structural links that form the parallel link are 70.5 inches long, 22 inches wide at the base, and 10 inches wide at the top. Each link consists of two 3-inch diameter tubes that form the sides with tubular truss members between the side members. The configurations of the two links are identical except for the base attachment. The upper link, in the stowed position, has a torque tube across its base for reacting the torque induced by the extension mechanism. Base plate supports tied to the forward end of the cradle provide the pivot point mounts for the parallel links at the base of the extension system. All pivot points are provided with either ball bearing or self-aligning bearings to minimize friction and prevent binding at these points. The narrow ends of the parallel links, the ends which actually move radially 90 degrees for extension, are tied to a platform that mounts the yaw axis of the pointing control gimbal, and also acts as a restraining link that keeps the side links parallel through the 90 degree travel of the system. The extension mechanism keeps the spar in a stable horizontal position throughout its 90 degree travel from the stowed position to the extended positon.

Structural stability of the parallel-bar linkage in the fore and aft direction of the SM is provided by a support strut that folds when the spar is stowed. The support strut attaches to the cradle at the inboard edge and to the aft leg of the parallel-bar linkage close to the outboard edge (when extended). The support strut is a tubular truss member, which has self-aligning bearings at all attach points and joints that rotate. (See Fig. 4-2.) An overcenter linkage is connected from the support strut's folding joint to the aft leg of the parallel-bar linkage. By driving the linkage slightly overcenter against stops with a spring force when the spar is extended, the extension mechanism becomes a stable structural support. Restow of the oriented section is not considered a requirement for the extension mechanism except during ground testing, but can be accommodated into the system. The only additions required for restow would be a force-generating system to override the spring force in the overcenter locking link and the extension force generating system would be reversible. For loads in the structural members of the extension mechanism and the natural resonant frequencies, see BBRC TN66-47 "Preliminary ATM Structural Analysis".

Drive systems considered for extending the extension mechanism were electrical and pneumatic. Electrical systems were considered inferior to the pneumatic system due to reliability and force generated vs. volume required. The extension mechanism is driven to its extended position by a pneumatic system which is partially shown in Fig. 4-2. This system has a cylinder that provides for extending the oriented section. The system uses nitrogen gas stored at 3000 psi in a pressure vessel, which drives the cylinder through a pressure regulator, a two-way check valve, and flexible pneumatic tubing.

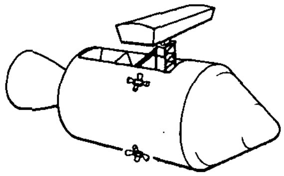
One end of the double-acting cylinder is attached to an activating horn that is cantilevered from the base pivot point on the forward leg of the parallel-bar linkage. The opposite end is attached to an activating horn that is cantilevered from the base pivot point of the support strut. Tied between these two points, the center of the cylinder translates in an inboard plane as the extension mechanism is actuated.

The pneumatic pressure regulator may be adjustable to permit increased pressure for ground testing in the one-g field. A bleed-off orifice is used to control the acceleration and velocity of the oriented section during the extension maneuver.

The forces required for extending the oriented section in a one-g field are presented in Fig. 4-4.

4.2.3 Gimbals

The three ATM oriented section axes are defined as the yaw, pitch and roll axes. The gimbal angle travel of these axes is: ±170 degrees in yaw; ±30 degrees in roll, from the null position. The yaw axis null position is established when the long dimension of the ATM oriented section is parallel to and coplanar with the longitudinal axis of the CSM; that is, looking forward along the launch thrust axis. The pitch null position is parallel to the CSM longitudinal axis; the roll null position is tangent to the surface of the SM projected to the center of the gimbal.



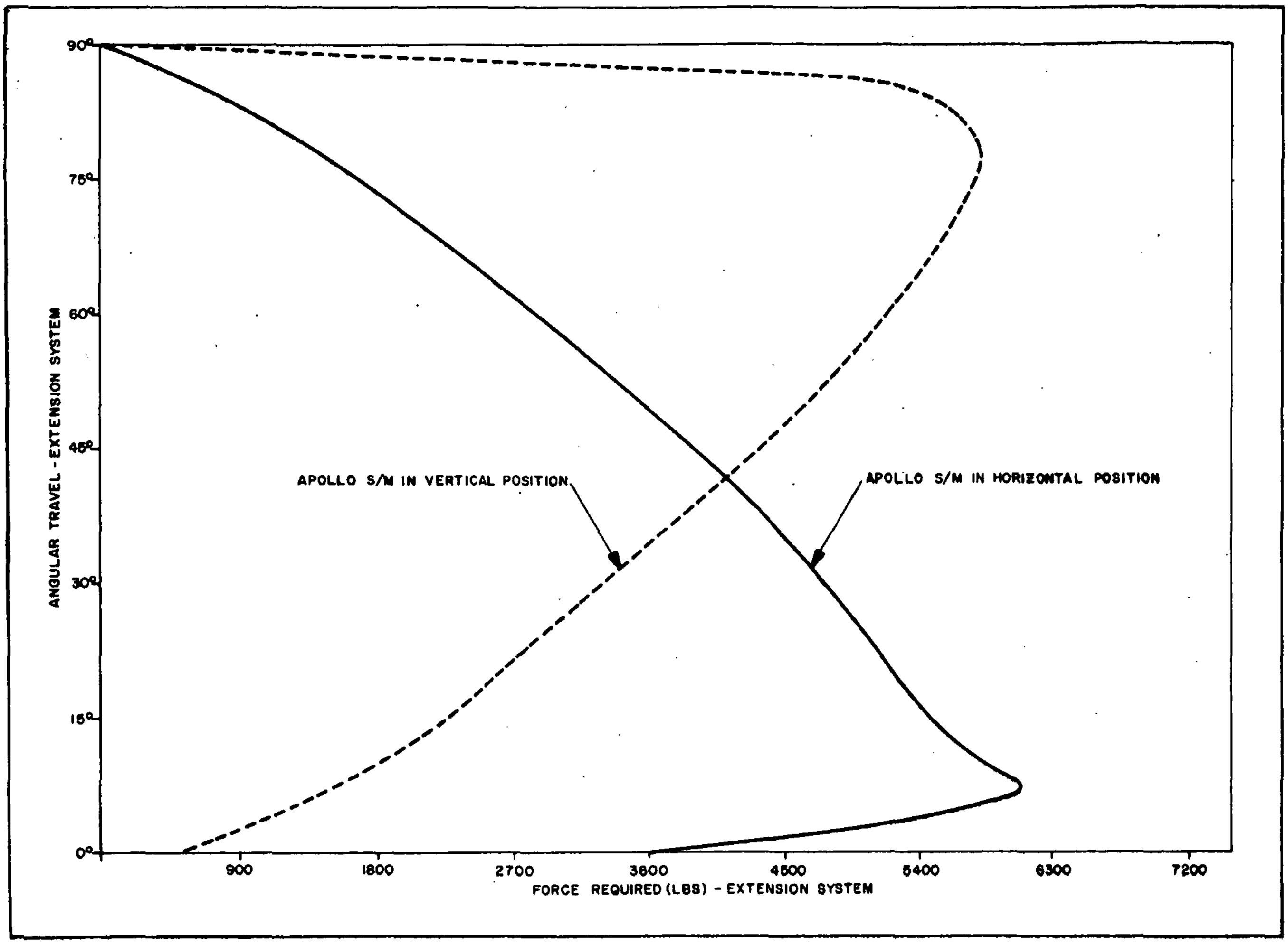


Fig. 4-4 Oriented Section Extension Force Requirements

Yaw gimbal travel is optimized for the view angles of the experiments since orientation at 180 degrees aft, and pitch toward the SM might damage the experiments due to scattered light reflected from the aft portions of the SM. Pitch gimbal travel is the maximum possible angle that maintains clearance between the oriented section and the CSM structure for negative angles, and that permits an unrestricted angle that maintains clearance between the oriented section and the extension mechanism structure.

The tri-axial gimbal mechanism attaches to the top of the extension mechanism structure and provides a structural attachment between the oriented section and the extension mechanism, as shown in Fig. 4-2. The three gimbal axes are orthogonal and intersect at a common point. The oriented section will be balanced about this point so that it rotates about its center of mass. The concept shown in Fig. 4-2 indicates that a large amount of ballast would be required to balance the oriented section about the roll axis. This concept is slanted toward providing the maximum experiment cross section. Moving the roll axis outboard, and extending the length of the yaw housing and gimbal shaft would provide for better experiment balance around the roll axis, but this design change would decrease either the available experiment cross-sectional area or the roll gimbal angle.

The yaw-axis shaft is supported by preloaded duplex bearings at one end, mounted in a cylindrical housing attached to the extension mechanism. It is supported by a ball bearing at the other end of the shaft. The angular-contact, duplex-pair bearings react thrust loads along the shaft as well as the radial loads. The Conrad deep-groove ball bearing reacts only the radial loads. The bearings are stainless steel, 52100, similar to but larger than those used on the OSO spacecraft and will be lubricated using the VacKote process, a space-qualified BBRC lubrication. The two bearings are fixed in relationship to the extension platform; therefore, the yaw shaft rotates relative to the CSM. The pitch-roll gimbal frame is attached rigidly to the flange of the yaw shaft by means of mounting brackets. Two stub shafts are attached rigidly to these mounting brackets, and constitute the pitch shaft. The pitch-roll gimbal frame is supported by these stub shafts through a similar ball bearing arrangement to that of the yaw axis. Two additional stub shafts forming the roll shaft, located 90 degrees radially from the pitch axis, are supported through a similar ball bearing arrangement to that of the pitch axis. The stub shafts are attached rigidly to the forward and aft bulkhead of the spar.

Stops are provided in each gimbal axis to prevent excessive gimbal travel that might cause damage to the ATM system or the CSM. These stops are shown in Fig. 4-2 and are shear lugs padded with Buna N rubber. They are capable of stopping the oriented section traveling at a velocity of six degrees per second.

Gimbal brakes are provided to hold the oriented section at a fixed orientation with respect to the CSM. These brakes will hold the oriented section in a one-g field with an unbalance of c.g. offset of 0.25 inches from the gimbal axis intersection. The gimbal brakes utilize a friction brake concept as shown in Fig. 4-2. The braking force is provided by a spring-loaded clamp which fits around the gimbal shafts. A double-acting solenoid provides the force required to apply and release the brakes.

The gimbal drive motors and associated electronics are presented in Section 6 as part of the pointing control subsystem.

4.2.4 Oriented Section Spar

The experiments, the thermal shield, the solar monitor telescope-vidicon, the pointing control solar sensors, and portions of the ATM electronics will mount to the spar as indicated in Fig. 2-1. The spar attaches to the gimbal housing thus becoming the primary support for all oriented section equipment and the experiments.

The use of the spar to support the experiments and all other portions of the oriented section has been analyzed to determine the effects of distortion. These analyses have included the trade-off of thermal and structural distortion over the 10-foot-long spar, the system requirement for experiment pointing stability, and the availability of the astronaut to adjust experiment optical alignment in orbit.

With the implementation of thermal control techniques, the dynamic thermal distortion of the spar can be limited to less than five arc seconds during a one minute data acquisition period. This dynamic distortion and static distortion of the spar will be precluded from distorting the experiment by means of experiment attachment mounts, which do not react bending moments. These mounts will also prevent differential expansion between the experiment and the spar from distorting the experiment base plate. This establishes the requirement that experiments requiring fixed relationships between elements to a high degree of precision must build an integral instrument on a rigid base plate. High thermal impedance will be designed into the mounts to minimize the flow of heat from the spar to the experiment base plate.

An alternative approach would be to provide a spar with sufficient stiffness to act as an optical bench to which the experiments could be rigidly attached in one piece or by major elements. This approach was rejected as an initial concept because of the resultant mass of the spar, control of thermal distortion problems, and the difficulty of preventing spar sag-distortion of experiment alignments in a one-g field.

With the recommended concept, the experimenter would align his integral experiment in his laboratory prior to mounting to ATM and/or provide the capability of in-flight adjustment of critical elements by the astronaut via the command system.

Further analysis after the experiment requirements are more fully known is needed to finally establish the feasibility of the approach recommended above.

The spar, as shown in Fig. 4-2, is a skin-stringer structure, with a 13 inch square cross section and a length of 124 inches, to which the experiments, the thermal shield, ATM equipment, and the solar sensors attach. See Figs. 2-7 and 4-2 for mounting arrangement of experiments and equipment on the spar. Structural requirements of the spar are: (1) natural resonant frequency of 10 cps or greater (imposed by pointing control subsystem), and (2) maximum angular slope of 60 arc seconds between ends when loaded with 1000 lbs. Designing for the second requirement will provide a natural resonant frequency somewhat greater than 10 cps.

The spar concept consists of extruded aluminum angles and aluminum face sheets. Internal bulkheads are located approximately 16 inches apart to provide local stability to the structure. The support structure is attached to the gimbal housing by bolting the face plates of the roll shafts to reinforced bulkheads within the spar. The face sheets of the spar are cut away in the vicinity of the gimbal housing to permit the specified gimbal travel. This also provides for a minimum distance between the extension mechanism mounting platform, and the center line of the gimbal axis, thus reducing the bending moment between these two points. It also maximizes the cross-sectional area for the experiment instruments. The structural continuity across the cutout portion of the face plates is maintained by a splice plate.

4.2.5 Jettison of Oriented Section

A jettison feature for separating the oriented section from the extension structure, after completion of the ATM mission, is required to conserve fuel during the reentry maneuver. Figure 4-2 indicates the plane of separation. Separation will be accomplished using explosive bolt cutters and compression springs to push the oriented section away from the spacecraft. The wire bundle from the oriented section will be separated with an explosive connector or a squib-actuated guillotine.

4.2.6 Thermal Shield Structure

The thermal shield encompasses the experiments, the spar, and the ATM equipment mounted on the spar. Structurally, the thermal shield must be capable of withstanding the launch loads.

The thermal shield attaches to the spar at the bottom edges on both sides, and at the forward and aft ends of the spar. Construction is of thin aluminum honeycomb, reinforced with longitudinal and radial stringers. An interior liner of mylar is mounted to the thermal shield structure with insulated standoffs. Both surfaces of the mylar are prepared with a gold diffused deposit.

Access doors are provided in the top surface of the thermal shield for access to the experiment cameras. This facilitates the loading of film on the launch tower and the recovery of film during EVA operations. The access doors are dust tight to minimize the possibility of experiment contamination.

4.2.7 Aperture Cover Mechanisms

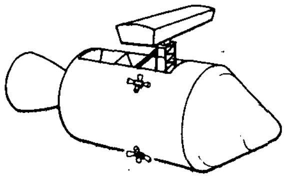
Aperture covers will be used to close each of the apertures in the front of the thermal shield. Aperture covers for each experiment, the fine control sensors, and the solar monitor subsystem optics will be required. A typical arrangement is shown in Fig. 4-2.

The aperture covers are actuated open by a torque motor and travel until reaching a stop. The aperture cover mechanisms are spring loaded for return to the closed position. Retaining channels support the covers through the launch environment, and provide stops to control the travel from the open to the closed position.

The arrangement of the aperture covers is such that they can all be opened at the same time or may be opened individually. The aperture cover mechanisms must be designed so as not to affect the oriented section balance in either the open or closed condition. The covers must be dust tight to minimize contamination of the experiments.

4.2.8 Sector I Cover

Prior to the extension of the oriented section, the Sector I cover must be deployed. This cover is an integral load-carrying member of the SM and should be designed and fabricated by the Apollo spacecraft contractor.



Analysis of methods for deploying the Sector I cover to enable the oriented section to extend included hinged doors and the jettisonable cover. Mechanical hinged doors were investigated and no method analyzed could provide a continuous shear panel for the structural continuity required by the Apollo spacecraft. Mechanical actuators would also decrease the available experiment volume.

Jettisoning is the most feasible approach for deploying the cover and providing for the structural integrity of the door. Jettisoning can be accomplished with jet-cord or primacord held in place with a retaining strip. A U-notch in the aluminum extruded attach flange of the sector coincides with the primacord and insures that the cover is severed at this point. Internal shielding is provided for protection of the ATM system against any possible shrapnel from the primacord. Compression springs located at the four outboard corners of Sector I provide for pushing the door away from the Apollo spacecraft. An existing Apollo sector cover can be used; the only modification required is the addition of the U-notch in the attach flange.

Interface control will insure that the jettison mechanism does not adversely affect the ATM system or the normal function of the cover.

A relatively small hatch will be required in this cover for access to the experiment cameras for film loading on the launch tower.

4.2.9 ATM System Weight

The ATM system weight and c.g. location are constrained by the SM structure and the gimbal angles of the SPS engine, located at the aft end of the SM. The structure within Sector I of the SM is capable of supporting 5000 lbs. The composite c.g. of the CSM and ATM cannot exceed the gimbaled position thrust line envelope of the SPS engine. The SPS engine gimbal station is X = 295.2. The associated gimbal angles are 12.5 degrees in the positive Y-direction, 4.5 degrees in the negative Y-direction, and six degrees in the positive and negative Z-direction from the center line of the CSM. For low altitude abort conditions, a margin of approximately three degrees of gimbal capability must be maintained along the Y-and Z-axes in both the positive and negative direction.

The individual equipment weights, and coordinate locations of these weights for the stowed and extended positions, are presented in Table 4-2. The c.g. for the combined masses of the ATM system is as follows:

Stowed	Extended
X_s = 263.7	X_s = 301.6
Y = 27.54	Y = 50.88
Z = -35.72	Z = -65.82

The composite c.g. of the CSM and ATM is well within the gimbaled position thrust line envelope of the SPS engine for both the stowed and extended positions of the ATM.

Table 4-2
ATM WEIGHT DISTRIBUTION

Subsystem (a)	WEIGHT (lbs)	Coordinates (Inches) Stowed Sta.Xs	Stowed Y	Stowed Z	Extended Sta.Xs	Extended Y	Extended Z
<u>Structure</u>							
1. Cradle	300	277.00	30.5	-39.5	277.00	30.5	-39.5
2. Extension Mechanism	157	305.00	21.75	-28.25	335.00	41.5	-53.75
3. Oriented Structure(b)	95	269.00	31.00	-40.13	399.75	74.4	-96.1
4. Gimbals(b)	65	269.00	31.00	-40.13	339.75	74.4	-96.1
5. Thermal Shield (b)	51	269.00	31.00	-40.13	339.75	74.4	-96.1
6. Electrical Power (c)	600	232.38	21.00	-27.38	232.38	21.00	-27.38
<u>Data</u>							
1. Oriented Section (b)	65	269.00	31.00	-40.13	339.75	74.4	-96.1
2. Cradle (b)	20	347.75	31.50	-40.75	347.75	31.50	-40.75
<u>Command</u>							
1. Oriented Section	8	269.00	31.00	-40.13	339.75	74.4	-96.1
2. Cradle	9	347.75	31.50	-40.75	347.75	31.50	-40.75
<u>Solar Monitor (b)</u>	30	269.00	31.00	-40.13	339.75	74.4	-96.1
<u>Pointing Control (b)</u>	45	269.00	31.00	-40.13	339.75	74.4	-96.1
<u>Experiments</u>	750	269.00	31.00	-40.13	339.75	74.4	-96.1
Total	2195						

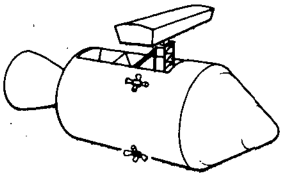
(a) Weights for the thermal control subsystem (10 lbs.) and the electrical distribution subsystem (100 lbs.) are included in the weights shown.

(b) Oriented section assumed to be balanced about center lines of gimbals.

(c) Optional subsystem considered for worst case calculations.

Calculations for determining the composite c.g. location were based on the empty weight and corresponding c.g. coordinates of the CSM. Propellant weight added to the fuel and oxidizer tanks within the SM shift the location of the CSM center of gravity. No information for the fuel weight or the center-of-gravity location of the fuel was available for this study. Further investigation and information will be required to determine the effect of the ATM system on the CSM c.g. with the fuel load on board.

At the present time, NAA and NASA are jointly considering different gimbal angles for the SPS engine. These gimbal angles are 5½ degrees in the plus Y-direction, 3½ degrees in the minus Y-direction, 6½ degrees in the plus Z-direction and 2½ degrees in the minus Z-direction. These values contain 2½ degree control safety factor.



Should these new gimbal angles come into being, the CSM and extended ATM composite c.g. would exceed the gimbaled position thrust line envelope of the SPS engine. This would require jettisoning the ATM oriented section prior to firing the SPS engine.

4.2.10 Interface Control of CM Storage

The ATM control unit and solar monitor display unit will be stowed in the CM during ascent and throughout the orbital mission. The ATM experiment film magazines will be stored in the CM during portions of the orbital mission and for re-entry.

Various storage locations for these assemblies will be used. The control and display units will be mounted in the lower equipment bay as indicated in Fig. 2-3. (See NASA document "MSC Internal Note 65-ET-16" NAA Dwg. No. S209-6.) This experiment control space allocation will accommodate the control unit, which has an 8 by 19 inch panel face and is 6 inches deep, and the display unit, which has an 8 by 8 inch panel face and is about 15 inches deep.

The experiment cameras will be preloaded with film magazines that will be used during the first half of the mission. The second set of film magazines will be stored during ascent and the first half of the mission, either in the CM "rock box", or in available volume in the ATM oriented section, or in Sector I.

The exposed film magazines and possibly data tapes will be stored in the CM "rock box", or in one of the following areas that should be available at the end of the mission:

Food Compartment (Lower Equipment Bay)	1.0 cu. ft.
Food Compartment (L. H. Equipment Bay)	1.7 cu. ft.
Food Compartment (R. H. Equipment Bay)	0.9 cu. ft.
Li OH Canister (Transferred to SM)	4.5 cu. ft.
	8.1 cu. ft.

At present, six exposed film magazines are planned for: two of these are approximately 8 inches in diameter and 2 inches thick, and four are about 12 inches in diameter and 2 inches thick. These will be installed in two thermally insulated containers, each of which is about 5 by 9 by 17 inches. These containers will be stowed in the left and right hand food compartments, as shown in Fig. 4-5.

SECTION 5

THERMAL CONTROL SUBSYSTEM

Section 5
THERMAL CONTROL SUBSYSTEM

The design concept presented in Section 2.2.2 for the thermal control subsystem is based upon analysis of the requirements and constraints of the other ATM subsystems, the CSM, and the experiments. This analysis is summarized in the following paragraphs and is discussed in detail in Appendix B. The numerical values presented for thermal design are not intended to be design values; they are intended only to show the feasibility of the approach.

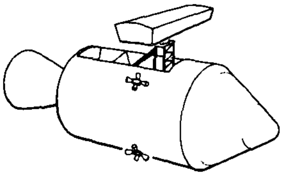
5.1 SPAR ALIGNMENT

The spar of the ATM serves as the base upon which the individual experiments are mounted. Thermal distortion of this spar will result in pointing misalignment of the attached instruments during a data acquisition period. Steady-state pointing errors can be nullified by the Apollo crew.

If the spar is subjected to a thermal environment for which a temperature gradient exists across the tube diameter, the differential thermal expansion of the tube sides will result in tube bending. An aluminum spar that is 12 feet long and 1 foot in diameter requires a temperature difference across the tube diameter of only 0.032°F for a one arc second bend. (See Appendix B.) Since one arc second per minute appears to be a tolerable error (Section 3.2), the value of 0.032°F per minute has been taken as a design goal for the rate of change of the temperature difference across the spar.

5.2 TEMPERATURE DIFFERENTIAL

The temperature difference across the spar is caused by the difference in magnitude of the heat transfer to the spar from the experiment packages, which vary in their heat dissipation, and from different areas of the heat shield as the individually controlled heaters are energized and de-energized. There is a steady-state or mean orbital heat input to the mounting spar and, also, there is a time-varying or oscillating heat input. At any particular time or orbital position, the total bend due to thermal stress will consist of the constant amount plus the particular value of the time-varying component. By adjusting the offset bias in the pointing control subsystem, the astronaut can nullify the misalignment effect of the constant bend. The thermal design effort for the system is, therefore, directed toward the dynamic component of misalignment. The analysis defining the allowable dynamic heat input rate to the spar is described in Section 2 of Appendix B. In addition, in Section 4 of Appendix B, the insulation required between the spar and the experiments and other components to limit heat flow to the spar is estimated. In general, two to three sheets of aluminized mylar are sufficient. Also, in Section 3 of Appendix B, order-of-magnitude estimates are calculated for the spar insulation required to restrict the steady-state bend of the spar to 100 arc seconds. These calculations indicate that about 30 layers of aluminized mylar are required.



5.3 THERMAL CONTROL

The time-averaged temperature level of the oriented section can be controlled by passive methods to within approximately ±10°F for a prescribed mission orbit, a prescribed CSM attitude pattern during the mission, and a prescribed orientation of the ATM with respect to the CSM during the mission. The first two of these items may be known before a mission, but the last item will be known only for specific ATM operation in orbit; that is, when the ATM is solar pointing. Therefore, even the ±10°F figure cannot be insured passively, and active thermal control must be imposed upon the passive thermal control methods in order to reduce the variation to ±5°F.

The active thermal control concept is that of employing electrical heaters combined with temperature sensors and a heater power control network. The heater power will be controlled by a simple on-off system. By designing the exterior surface finish to result in a skin temperature some 30°F lower than the desired interior temperature, only heat input to the interior, by means of the heaters, is necessary for temperature control. Should interior temperatures rise above the desired control level, the power to the heaters is shut off by the switching network, and the interior cools by heat flow to the external heat shield skin. The active thermal control design computations are presented in Section 4 of Appendix B.

Less power to the heaters will be required if the exterior skin temperature is only 10°F lower than that of the interior. However, ATM orientation, unknowns, combined with the variation of earth albedo with cloud cover, require a margin of safety in the design. Should skin temperature exceed the interior temperature, the heater power would, of course, cease since the interior temperature would rise above the control design temperature. However, cooling the instrument compartment interior is not desired and would be necessary should the heat flow be reversed from outward to inward.

Since the skin temperature will vary as a function of position on the skin, it is necessary to control separately the interior temperatures at different locations within the oriented section. In this conceptual design stage, eight heaters with eight sensors are specified for the four quadrants, fore and aft in the system. Additional detailed design studies will indicate whether more or less control is necessary, and establish exact locations.

Active thermal control is necessary for the optimum performance of the optical systems of the experiments. Changes in the overall temperature level of the instruments can result in degradation of optical focusing and alignment due to differential thermal expansion of the instrument cases, optical mounts and optical components.

It is pointed out in Appendix B that the above design is not flexible enough to meet all the contingencies possible for the ATM system. It is recommended that a louvre system be added to this concept to vary the thermal impedance between the interior and the exterior environment. The precise requirements and design of this system must be determined in detailed design studies.

The additional thermal control necessary to limit the thermally induced drift of pointing to one arc second per minute of time will be provided by appropriate shielding of the spar from transient heat inputs. These transient heat inputs and the required shielding have been calculated and are presented in Appendix B. Only two layers of aluminized Mylar shielding are required between the spar and the experiment packages. Approximately seven layers are required between the spar and the exterior environment for the transient response; however, in order to limit the steady-state bending of the spar to some arbitrary value (100 arc seconds in this case), 30 layers of shielding are required. The use of this latter amount of shielding is recommended since both the transient and the arbitrarily assumed steady-state requirement are satisfied. The heat exchange between the spar and the experiment packages by conduction through the mounting structure must be restricted by the use of mounts having thermal resistances of the order of 275°F/watt.

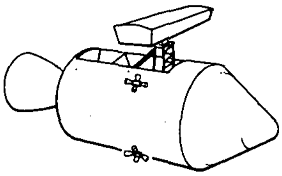
5.4 POWER REQUIREMENTS

The power requirements for active thermal control will depend upon the Apollo-ATM mission profile, and the external surface finishes of the heat shield. The external surface finish of the heat shield must be designed for a particular mission profile. The computed heater power estimate given in Appendix B for one possible average case is approximately 55 watts average, and an additional five watts are required for the thermal electric devices and thermal control circuitry.

5.5 ABSOLUTE TEMPERATURE CONTROL

Certain components of the ATM such as photographic film, electronics, the solar monitor interference filter, and the vidicon require specific temperature ranges for optimum performance. Most of these components can be maintained at the proper temperature level by the active thermal control system if mounted in the instruments or by the application of passive thermal control coatings--whether they are installed in the instruments or in the bay area.

The most sensitive components within the system are the light interference filter and the vidicon used in the solar monitor subsystem. The temperature necessary for their proper operation must be maintained within a few degrees of their design temperature. Both of these components are located within the spar. Since the spar is also a critical temperature component, these components must be isolated from the spar and a heat transfer path to the space environment provided. The interference filter temperature can be controlled using temperature sensors and thermoelectric devices to heat or cool as required. The heat input to the filter from solar radiation is restricted to a very narrow band of wavelengths in the region of the 6563Å line of hydrogen. The interference filter allows a 1 Å band of wave-



lengths centered upon this hydrogen line to pass. Additional filters are placed in series optically in front of the interference filter; each of these filters permit a successively smaller band of wavelengths to enter the next filter. The heat input to the 1Å filter from solar radiation and other sources is of the order of 0.1 watt. Assuming an efficiency of 10 percent for the thermoelectric device, approximately one watt of power will be required. The heat will be transferred by radiation or conduction to the heat shield below the spar and out to the space environment. The heat shield surface in this area is coated with a high-emissivity, low-absorptivity coating. An estimate of the requirements for the thermoelectric system for the vidicon interference filter is made in Appendix B. By reversing the electric current through the thermoelectric modules, they can be used as heaters when the filter temperature falls below that required for optimum performance.

The vidicon must also be thermally isolated from the spar, and a conductive heat transfer path to a radiating plate must be provided. The vidicon dissipates approximately five watts of power. For this reason, thermoelectric devices were not considered for this application--the power requirements of about 50 watts would be excessive. The vidicon is encased in a copper sheath (Appendix B) with sufficient contact pressure to insure good thermal contact. The sheath is mounted to a radiating plate exposed to the exterior. The surface of the plate is coated with a high-emissivity, low-absorptivity surface so that it will radiate to space the heat transferred from the vidicon. The vidicon analysis is presented in Section 3 of Appendix B.

5.6 MECHANIZATION

The basic concept of the thermal control system requires sensing the temperatures of the experiment complement. By using these sensors to control heaters, the experiments' environment will be kept at the desired temperature.

Such a system (Fig. 5-1) will require an accurate temperature reference for control of the system to ±5°F of some fixed nominal temperature. The temperature sensors and the heaters will require extensive design considerations as to the areas and materials to be thermally controlled.

The heaters most likely will have to be ribbon elements over a wide area with high output capabilities. The power switches will be mounted in the heater area to utilize their heating properties as well. The entire system will be housed in the oriented section.

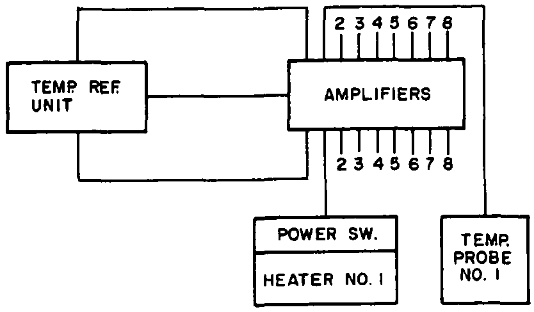


Fig. 5-1 Thermal Control Block Diagram

SECTION 6

POINTING CONTROL SUBSYSTEM

Section 6
POINTING CONTROL SUBSYSTEM

In this section the pointing control subsystem requirements, conceptual design and possible implementation are considered from an operational standpoint; that is, the modes of operation form the basis for logical divisions in the text. While this is a somewhat artificial breakdown because there is much hardware commonality, it provides a better insight as to how the study was performed and the design evolved.

There is a minor exception to this division in that all sun sensors are described in Section 6.4. However, since the sun sensors are essentially a separate problem, the breakdown by modes of operation within Section 6.4 is similar to that used for the entire subsystem.

6.1 REQUIREMENTS

6.1.1 General

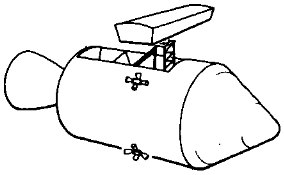
The pointing control subsystem (PCS) must be designed to satisfy the maneuvering and solar orientation requirements of the ATM system. Additionally, the design must be consistent with the limitation and constraints imposed by the other subsystems and should avoid concepts which complicate the subsystem interface or impose severe restrictions on the other subsystems.

Certain ground rules are evident from the system concept (Section 2). Thus, the concept design of the PCS is based on the assumption that three axis control is required, that dc gimbal torquers will provide the control of each axis, that it is sufficient to control rate only in roll (i.e., roll position control is not required), and that maximum use will be made of the control inputs obtainable from the astronaut. Furthermore, at the completion of the mission the ATM oriented section will be jettisoned and the PCS will not be required to provide accurate gimbal position control for restow positioning. However, the possibility that restow may be a growth requirement is considered in the design.

The performance requirements, from which the concept design is derived, are set forth below. These requirements include the operating modes of the PCS, interfaces with other subsystems, and environmental constraints.

6.1.2 Fine Pointing

The fine pointing operation constitutes the primary function of the pointing control subsystem. During this phase of the operation the astronaut selects a particular feature or event on the solar disc (using the solar monitor subsystem display and additional information from the experiment, if necessary), and guides the ATM until the experiment optical axis is correctly centered on the target. When the proper position has



been achieved, it must be accurately maintained during the data acquisition period of the applicable experiment. Furthermore, the response of the subsystem to commands by the astronaut observer must be sufficiently rapid to allow data to be obtained from relatively short-lived events.

The specifications stated and discussed below define the fine pointing performance:

- Accuracy: ±3 arc sec for any one minute period
- Offset range: 40 x 40 arc minute square containing the solar disc
- Resolution increment: commanded motions of ±1 arc sec
- Time to achieve target: one min (max), 0.5 min average

Accuracy as specified above refers to relative pointing error which is defined as the peak angular excursion of a fine sun sensor reference plane measured from its initial pointing direction for a one minute, experiment data acquisition period. The components of the relative error are drift and jitter: the former is a secular change while the latter is a noise-like variation with zero average value. Thus, the relative pointing error is the peak value of the instantaneous sum of the drift and jitter amplitudes in the one minute interval.

While the absolute pointing accuracy is not of major importance for this mission, a one arc minute absolute accuracy is to be achieved. Absolute pointing error is defined (for zero offset) as the angular difference between a line normal to the fine sun sensor reference plane, and a line from the ATM gimbal axis intersection to the geometrical center of the solar disc.

The resolution increment is defined as the smallest offset change that can be commanded. The other requirements are self-explanatory.

6.1.3 Acquisition

The acquisition requirements are concerned with the maneuvers which bring the ATM into an operational position. Both manual and automatic maneuvers are to be utilized.

Manual Control. The subsystem shall incorporate a slewing capability on all three axes, which allows the ATM to be positioned to any arbitrary attitude within the range of gimbal travel. During the manual slewing operation the maximum velocity of the ATM shall be limited to six deg/sec (Section 6.3.1) in each axis.

Automatic Control. The PCS shall have the capability to perform automatic acquisition whenever the angle between the solar vector and the ATM pointed axis (as defined by the normal to the fine sun sensor reference plane) is less than 25 degrees. The PCS shall be capable of automatically aligning the ATM pointed axis to within two degrees of the solar vector. The time required to complete the acquisition shall not exceed 15 seconds.

6.1.4 Roll Control

The roll axis control must be capable of maintaining the roll drift rate to a value of less than two arc sec/sec. The roll axis must incorporate the same slewing capabilities as the pitch and yaw axes.

6.1.5 Operational Features

There are several requirements that are not contained in the above performance specifications that should be incorporated into the design. These are:

(1) The PCS shall supply signals which permit determination of the pitch, yaw, and roll gimbal angles to within ±1 degree.

(2) An "in-field" signal shall be supplied whenever the sun is in a position that permits automatic acquisition.

(3) In the fine pointing mode, the PCS shall supply signals which permit determination of the angles between the solar vector and the pitch and yaw null axes.

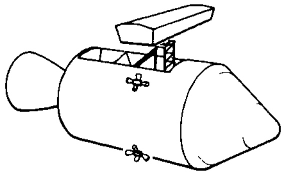
(4) When a gimbal limit is encountered, the PCS shall automatically prevent further drive torque in the direction of the gimbal stop.

(5) During periods when no control is required the ATM shall be immobilized.

6.1.6 Interface Requirements and Constraints

The pointing control subsystem interfaces with all of the other subsystems and, therefore, must be designed to accept the applicable interface requirements. Similarly, the PCS imposes restrictions on the other subsystems and these requirements must be stated and considered as input requirements to the particular subsystem involved. These are enumerated as follows:

• PCS energy allotment:	2.0 kw-hr	
• Maximum cable torque per gimbal:	0.083 lb-ft	
• Maximum bearing torque per gimbal:	0.052 lb-ft	
• Weight of oriented section:	1000 lb	
• ATM gimbal inertias:	270 slug-ft^2	Yaw
	270 slug-ft^2	Pitch
	40 slug-ft^2	Roll
• Gimbal travel:	±170 degrees	Yaw
	±30 degrees	Pitch
	±25 degrees	Roll
• Min. resonant freq. of first bending mode of structure:	10 cps	Yaw
	10 cps	Pitch
	15 cps	Roll
• Max balance (offset between c.g. and gimbal intersection):	0.010 in.	
• Gimbal stop energy capability:	2.0 ft-lb	



6.2 FINE POINTING MODE

With the fine pointing requirements given in Section 6.1.2, the servo design to satisfy those requirements is described below. It has been established that fine sun sensors supply the error signals and that dc torque motors provide the gimbal control torque. A gimbal rate signal is derived from the transducer. Gimbal friction, structural resonance, and man-motion effects are the major considerations in the design, particularly with regard to the stabilizing networks.

6.2.1 Design Considerations

6.2.1.1 Error Sources and Error Budget

Since the general configuration of the control loop is established (i.e., a sun sensor, signal amplifiers, power amplifier, d.c. gimbal torquer), a reasonable assessment of the error sources and the amount of error that can be allocated to each can be formulated. The error budget presented is the end product of several iterations wherein the allowable errors were distributed on an "equal hardship" basis.

It is assumed that errors arising from the various sources are random and uncorrelated so that the expected error will be the root sum square of the individual errors. An exception to this occurs in the case of "extreme" man motion where the motion-induced torque disturbance must oppose the gimbal drag torque in order to produce the maximum error. This subject is covered in more detail in Section 6.2.3. The eight error sources listed below are the major contributors to the pointing error:

Error Source (Pitch and Yaw)	Error Budget (Arc Sec/min)
(1) Fine sensor drift	2.0
(2) Variation in gimbal drag torques	1.0
(3) Fine servo electronics drift	0.5
(4) Offset command signal drift	0.5
(5) All sources of pointing jitter	0.5
(6) Roll-induced pitch/yaw error	1.0
(7) Error due to roll-induced gain change (cross-coupling)	0.1
(8) Error due to "moderate" man-motion	0.1

Expected error (RSS) 2.6 arc sec/min

Errors (1) and (6) are discussed in more detail in Sections 6.4. The error allocated to item (2) is the principal factor in determining the requisite torque sensitivity of the loop. Items (3), (4), and (5) are basically requirements on the electronics necessary to implement the design concept. Item (7) arises due to the fact that roll motion causes the sensor input axes to be angularly displaced from the gimbal axes. Thus, the effective torque sensitivity is decreased. (See Section 6.2.4 for a more detailed discussion.) Finally, in Item (8) "moderate" man motion is defined as a motion which does not reverse the sign of the CSM-ATM relative velocity in any axis.

6.2.1.2 Gimbal Drag Torques

The predominate yaw and pitch gimbal torques are gimbal friction and gimbal drag caused by the flexible cables which are displaced with gimbal travel. Assuming a CSM unidirectional angular rotation of 0.01 to 0.05 deg/sec with respect to either the yaw or pitch gimbal, frictional torque will be applied to the ATM through the bearings, torque motor brushes and motor magnetic drag.*

The flexible cables, which supply electrical connections across each gimbal, are expected to exert a friction-like drag at about the midtravel point of each gimbal. A drag increase proportional to displacement from this position, opposing the position change, would then be expected. The maximum drag will occur in yaw for extreme gimbal angles of 170 degrees; the estimated torque level, considering a 70-conductor cable, is ±16 oz-in.

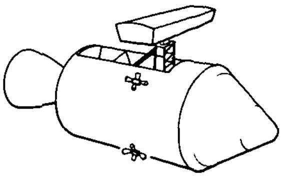
Magnitudes of maximum static gimbal friction/drag torques are tabulated as follows:

Source	oz-in.	lb-ft
Bearings	±10	±0.052
Motor Brushes	±13.5	±0.070
Magnetic Drag	± 2.5	±0.013
Cable Drag, max.	±16	±0.083
Totals	±42	±0.218

The actual torque motion characteristic of the cable is undoubtedly more complicated than a simple spring-like restraint. For example, the cable might represent a compliant force over a range of travel, and then (due to ductile flow, position readjustment, or change in position between insulator and conductor in the case of a coax lead) exhibit a sudden reduction in torque. Thus, the variation in torque over a small range of travel cannot be considered a small portion of the total cable drag. The torque variation considered (in the next section) is somewhat greater than would be experienced if the cable drag suddenly changed from maximum to zero.

It should be noted that the torque motor sizing is not established by the gimbal drag torque, but rather by the acceleration requirements arising from the need to provide rapid acquisition.

*This small torque drag is considered to result from a magnetic hysteresis phenomenon peculiar to a PM field torque motor. With rotation, armature magnetization lags behind the magnetizing forces of the PM field poles, and a magnetic torque is coupled between armature and field.



6.2.1.3 Torque Sensitivity

The necessary torque sensitivity can be established from the requirement that torque variations must not result in pointing errors exceeding one arc sec.

The torque variation due to the flexible cables has already been discussed. In addition to this there will be minor variations in the motor brush friction and to a lesser extent in the bearings. The torque variation due to all causes is assumed to be 0.1 lb-ft; thus, the minimum torque sensitivity must be:

$$K_T = \frac{\Delta T}{\epsilon_T}$$

$$K_T = \frac{0.1}{1.0} = 0.1 \text{ lb-ft/arc sec} \qquad (6.1)$$

$$= 2.06 \times 10^4 \text{ lb-ft/radian}$$

This establishes a lower bound on the control loop gain. The restrictions and limitations which determine the upper bound are discussed below.

6.2.1.4 Bandwidth Consideration

The principal concerns in designing the control loop are to maintain a high static gain, a stable system, and a minimum bandwidth. The bandwidth is of considerable importance since it is necessary to avoid interaction between the servo loop and the structure; more specifically, the servo-loop cut-off frequency should be well below the first bending mode of the structure.

Since a detailed model of the structure was not warranted at this stage of the design, the structure was assumed to be adequately represented by a first bending mode at 10 cps with a bending gain approximately 10 percent of the rigid gain. On the basis of this assumption, it is necessary to restrict the control loop bandwidth to a maximum of 1.5 - 2 cps in order to insure that structural characteristics do not affect the performance of the servo.

6.2.1.5 Compensation Network Selection

A simplified block diagram of either the pitch or yaw control loop is given in Fig. 6-1. In this diagram only the network and the load are considered as frequency sensitive elements and the only inputs are position commands and torque disturbances. As will be seen, time constants associated with the electronics and motor drive are significantly above the frequency range of interest. Furthermore, the motor damping term is negligible for the 0.01 to 0.05 deg/sec rates considered.

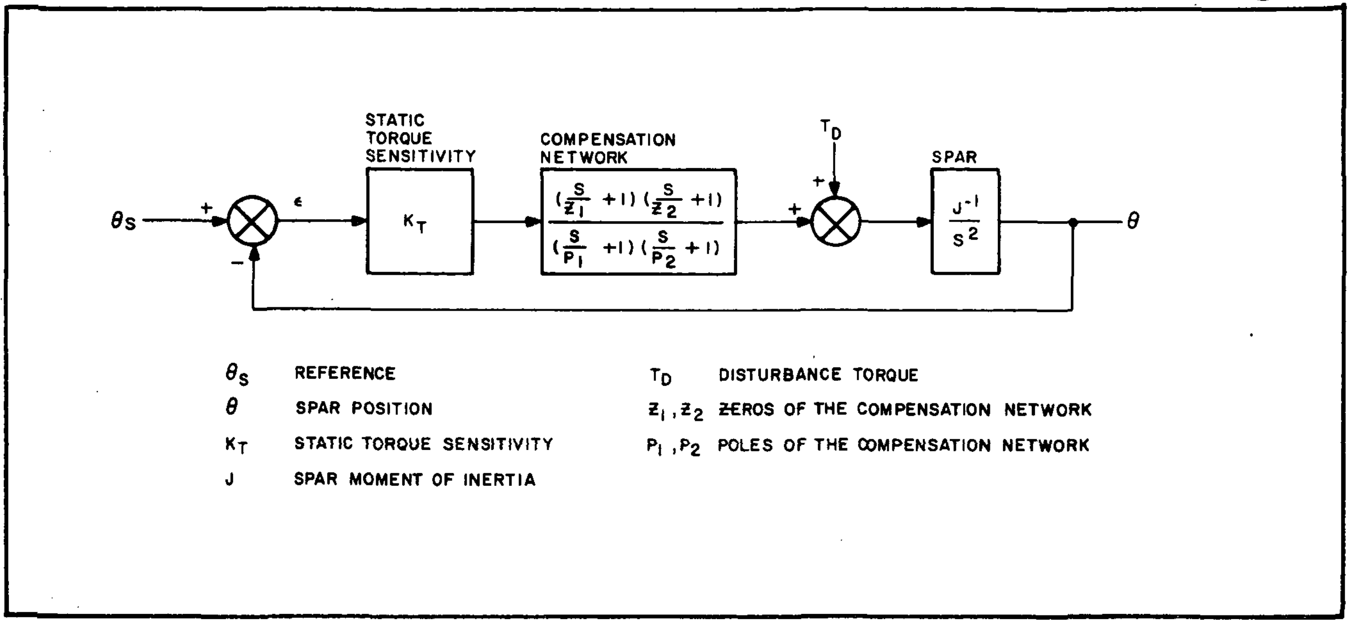


Fig. 6-1 Pointing Control System Block Diagram

The network assumed contains two zeros and two poles. For a simple lead-lag configuration one zero and one pole would have finite values, and the remaining zero and pole would move to infinity. For integral compensation (lag-lead-lead-lag) all four values appear.

The lead-lag compensation is desirable because of simplicity and the minimization of transient overshoot due to torque disturbance (see below). On the other hand, the torque sensitivity for a given bandwidth is lower than that achievable with integral compensation.

If, in Fig. 6-1, the input point is assumed to be the torque summing junction, then the compensation network appears in the feedback path of the loop and the error transfer function is:

$$\frac{\epsilon(s)}{T_o(s)} = \frac{J^{-1}\,(s/p_1+1)\,(s/p_2+1)}{s^2(s/p_1+1)\,(s/p_2+1) + K_T J^{-1}\,(s/z_1+1)\,(s/z_2+1)} \qquad (6.2)$$

Note that the network poles become zeros in the closed-loop, torque-to-error transfer function; hence, any poles in the network transfer function which are placed near the origin contribute increased derivative weighting, and large transient overshoots result.

The root locus plots of Figs. 6-2 and 6-3 compare similar compensation networks with and without integral compensation. Note that System A achieves a torque sensitivity of 0.143 lb-ft/arc sec with closed loop poles at position 2 on the diagram. For system B (no integral compensation), position 2 (same torque sensitivity) has greatly decreased gain and phase margin and correspondingly higher bandwidth. However, it is necessary to examine the effect of a step torque disturbance.

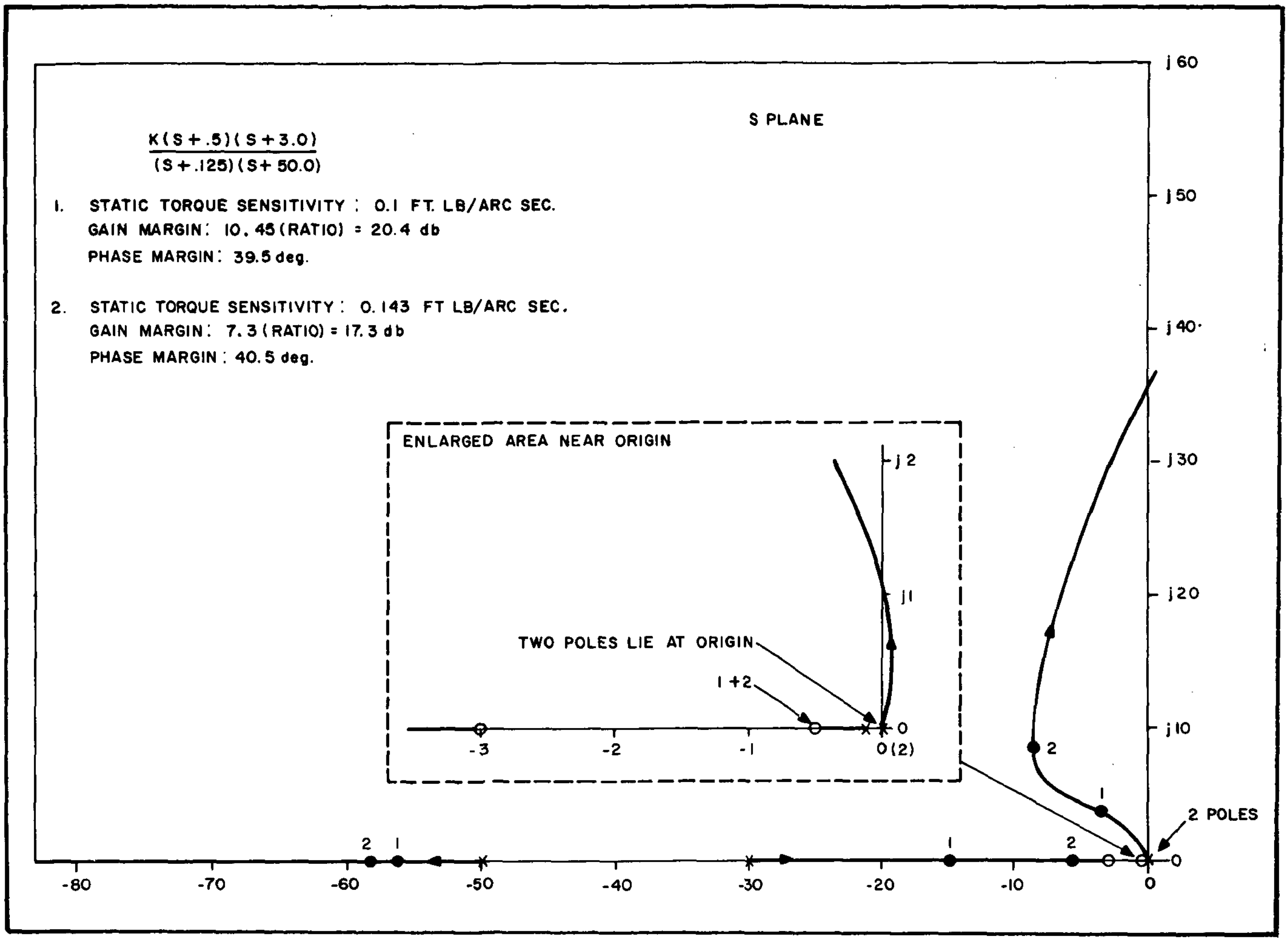


Fig. 6-2 Compensation Network Root Locus Plot of System A with Integral Compensation

For each system, the error response following a step torque disturbance was obtained using an analog computer. Figure 6-4 shows the two responses superimposed. Curve A shows an overshoot of 338 percent caused by the network pole at 0.125 radian/sec. Curve B shows no overshoot, but the "wiggles" indicate that the system does not have a high stability margin.

The best compromise is to maintain sufficient (but not excessive) torque sensitivity delete the integral compensation, and readjust the pole-zero to obtain better stability. Figure 6-5 is a root locus plot of the system with a gain of 0.1 lb-ft/arc sec and a network having a zero at six rad/sec and a pole at 60 rad/sec. Figure 6-6 gives the Bode plot for this system. Adequate stability is obtained as can be seen from the gain margin of 16.5 db and phase margin of nearly 40 degrees.

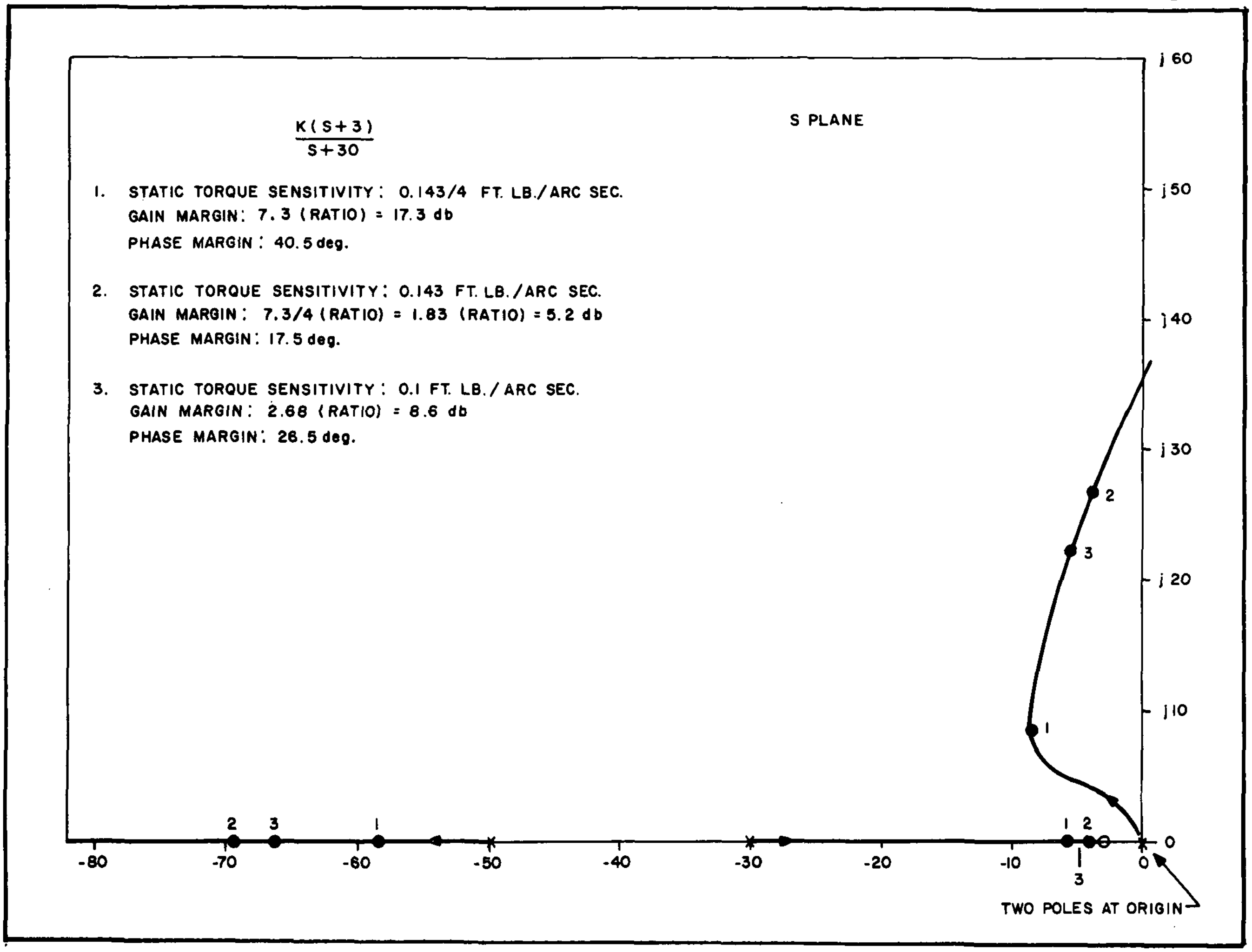
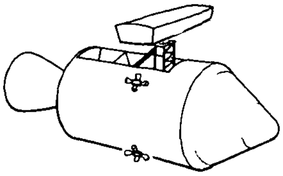


Fig. 6-3 Compensation Network Root Locus Plot of System B Without Integral Compensation

6.2.1.6 Selection of Offset Rates

A system requirement is that the oriented section must be capable of being positioned by the astronaut anywhere within a 40 x 40 arc minute square, approximately centered on the sun, from any other place in this square in not more than one minute and within one arc second of the commanded position.

This offset is accomplished by the introduction of a position signal into the fine pointing servo loop that increases linearily with time (i.e., a ramp). The resolution requirement of one arc second sets the scale factor of the ramp at not more than two arc seconds per second since it does not seem likely that an astronaut wearing space gloves could be expected to send a command of shorter duration than ½ second.



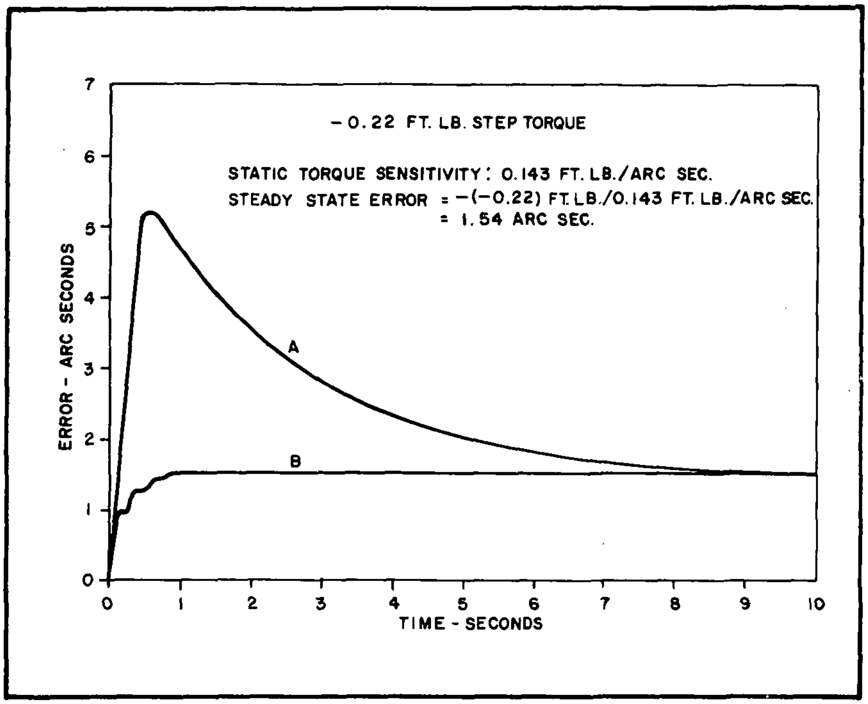


Fig. 6-4 Plot of Servo Response to Torque Disturbance Input

Simultaneous offset control in both axes is available; hence, it is permissible to discuss the operation in one axis. At the rate of two arc seconds per second, it takes 20 minutes to turn through 40 arc minutes. Obviously, the transport time requirement cannot be satisfied unless a two-speed concept is employed. From a human engineering standpoint, the ratio of fast to slow speed should not exceed 25. At the maximum speed ratio, the fast speed is 50 arc seconds per second, and the time needed to traverse 40 arc minutes is 48 seconds. Even though this figure is within the requirements, it will take training for an astronaut to become sufficiently proficient to realize this performance. The ability to correctly anticipate overshoot when shifting from fast to slow will be a big factor in making an offset quickly.

The advantage of using a position command is that the overshoot, except that resulting from pure delays such as astronaut reaction time and command activation, will settle out to the commanded position. For convenience, the slow rate is called the "boresight" and the fast rate the "offset".

An alternate approach is to provide the astronaut with a capability of proportional control where the commanded rate is continuously variable from zero to 50 arc seconds per second. This would probably be preferable for ease of operation, and the pointing control subsystem is designed to accept variable rate signals.

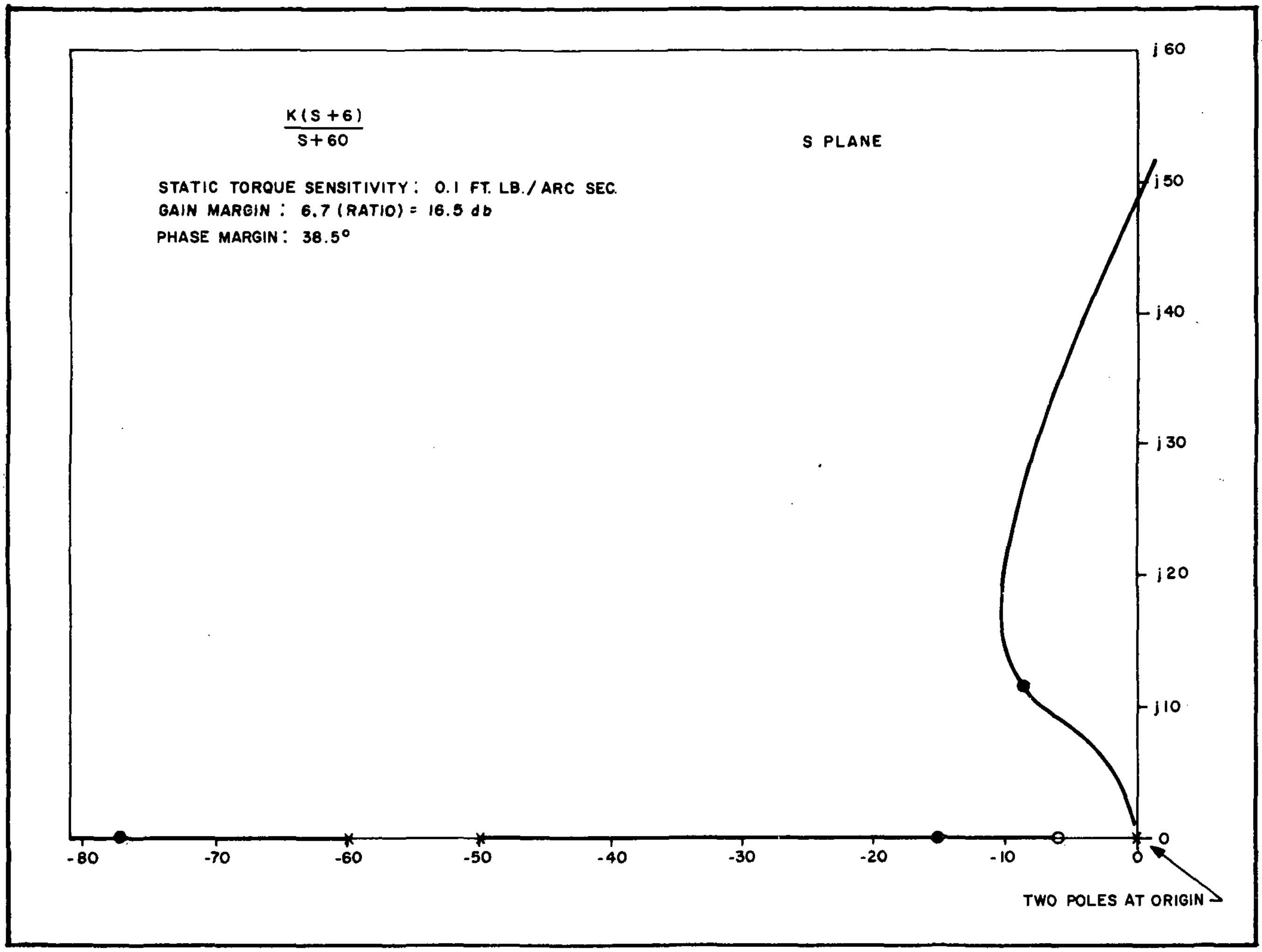


Fig. 6-5 Root Locus Plot of System C

6.2 Implementation Concepts

In the following paragraphs the various elements comprising the control loop are enumerated, and means for achieving the required performance are discussed. The subject of implementing the sun sensors must be considered in considerable detail, and to preserve continuity this discussion is presented in Section 6.4.

Figure 6-7 is a block diagram of the fine pitch or yaw control channel. The elements are described in the following paragraphs.

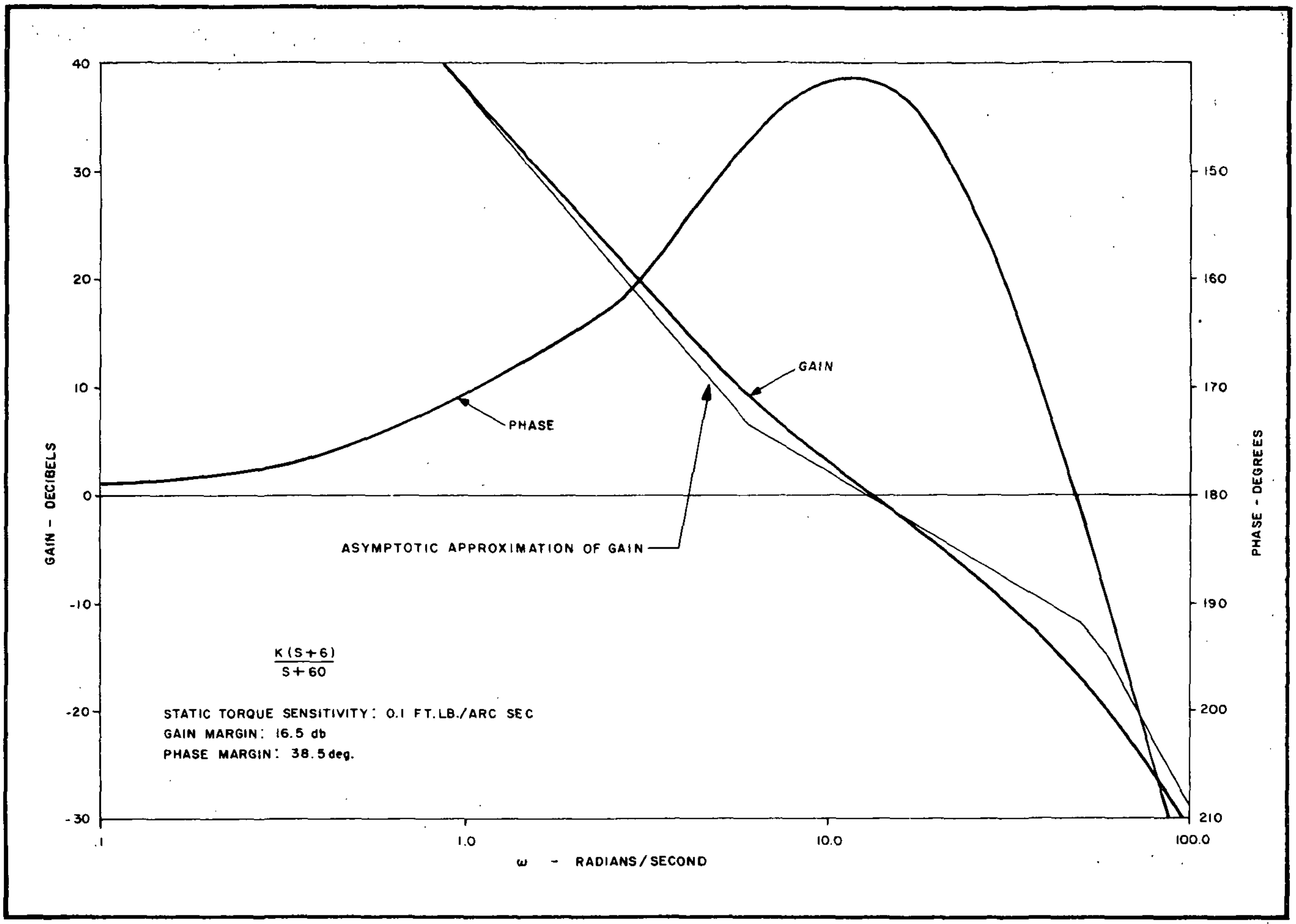


Fig. 6-6 Bode Plot of System C

6.2.2.1 Voltage Amplifier/Compensation Network

The most critical requirement in the ATM fine servo electronics is that total electronic drift be held to an equivalent pointing error of 0.5 arc sec or less (see error budget, Section 6.2.1.1) during a one minute experiment sample period. Two basically different design concepts have been considered to obtain the amplification and dynamics required:

<u>Cascaded Design.</u> In this design a fine sensor preamplifier, the compensation network and a chopper AC amplifier-demodulator combination are cascaded. Stability is achieved by designing most of the gain in the chopper-amplifier section. Bias input is summed as voltage at the preamplifier output.

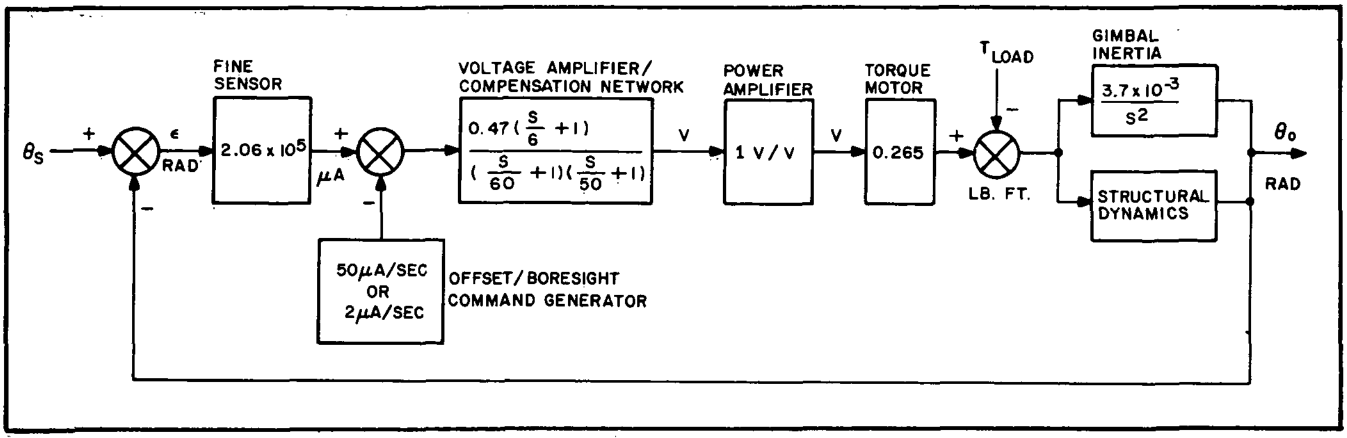


Fig. 6-7 Block Diagram of the Fine Servo Yaw/Pitch Control

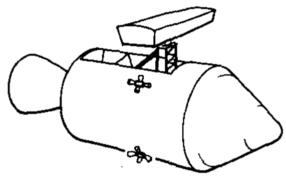
Feedback Design. This design employs a closed loop with a dynamic lag-lead feedback transfer around a DC voltage amplifier. A current summing point at the input receives both the fine sensor signal current and the biasing current for offset pointing control. Closed-loop gain is relatively low, and drift stability is excellent. As output saturation is approached, feedback gain change automatically permits the closed-loop lead transfer to remain effective for dynamic input signals. The number of components in this feedback design is about one-half the number in the cascade design. Thus, from the standpoint of simplicity and operational performance, the feedback design concept appears better for ATM electronics.

6.2.2.2 Offset Command Generator

As previously discussed, the offset command generator must furnish the bias signal which changes the pointing line-of-sight. Furthermore, the commands are obtained from rate and direction signals (that is, the astronaut commands the ATM to pitch up at a fast or slow rate). Implementation of the command generator is intimately connected with the sensor design and, as will be seen in Section 6.4, the recommended sensor design requires that the offset signal be a controlled current.

Since the initiating command from the astronaut is rate and since it is a critical requirement that the offset position be accurately maintained when the astronaut returns his control stick to the zero position, the command generator must be an integrator with essentially zero drift and jitter.

Two design concepts that were considered and rejected for generating bias currents with short-term (one minute) stability of one part in 2400: (1) to employ operational amplifier integrators, and (2) to use 14-stage digital-to-analog converters. Method (1) was rejected since the requirement is generally beyond the state of the art for analog computer type integrators even under carefully controlled environmental conditions. Method (2) suffers from the requirement that each stage of the D-to-A converter must have the same resolution as the least significant digit in order to obtain a monotonic output.



A conventional electromechanical concept has been selected: biasing currents are generated by a motorized potentiometer similar to those used in analog computers. The astronaut commands the motor to run at high and low speeds in either direction. Excitation voltage on the potentiometer is regulated to better than 0.01 percent over a one minute time period. The extremely small current requirement from the potentiometer minimizes bias signal change due to loading of the potentiometer. Resolution of the potentiometer is substantially greater than one part in 10^4, which gives a command position resolution of about 0.25 arc sec. The entire assembly can be housed in a hermetically sealed container to avoid lubrication and outgassing problems. This scheme also makes proportional rate control feasible.

6.2.2.3 Power Amplifier

The critical requirement in power amplifier design is minimization of the electrical power consumed in the electronic circuits. Power consumption in a linear design has been considered for quiescent pointing, and the total amplifier dissipation in yaw and pitch would be greater than 20 watts.

The design approach selected is to incorporate pulse width modulator-power switch amplifiers for all ATM gimbal motor drive designs. Here, the DC voltage input signal is converted into a constant frequency, square wave form, with pulse durations being modulated so that the pulse length is proportional to the input voltage. The square wave signal turns pairs of power switching transistors on and off alternately; an LC filter then smooths the current and supplies the motor armature circuit with a DC drive voltage. With power switching elements saturated most of the time, electronics power dissipation can be held to a few watts for quiescent pointing.

6.2.3 Man Motion Error Analysis

The disturbance produced by man motion depends to a great extent on whether the impulse resulting from the motion is sufficient to reverse the sign of the relative angular velocity between the ATM and the CSM in any axis. Obviously, the smaller the initial relative angular velocity, the greater the probability that a given motion will reverse a gimbal rate. It will be assumed herein that 0.02 deg/sec is the minimum (practical) drift rate for the CSM. Man motions which create CSM transient velocities in excess of 0.02 deg/sec will have a 50 percent probability of reversing a gimbal rate and will, therefore, be classified as "extreme". Motions with less impulse will be called "moderate".

The significance of this division is apparent when the effects of man motion on ATM are separated into disturbances resulting from acceleration produced torques (due to the noncoincidence of the ATM c.g. and the gimbal intersection) and those arising from reversing the sign of the gimbal friction torque.

First, consider the error produced by reversing the gimbal rate. With a torque sensitivity of 0.1 lb-ft/arc sec, the 0.44 lb-ft change in gimbal friction will cause the pointing angle to change by 4.4 arc seconds. It should be noted, however, that this is not a random error; that is, the sign will be opposite that of the torque-induced error that existed prior to the disturbance. Another way of stating this is to consider that only the nonvarying component of torque (i.e., 0.12 lb-ft) is reversed so that the additional error resulting from gimbal rate reversal is 2.4 arc seconds.

In order to determine the extent of man motion that is required to reverse the 0.02 deg/sec minimum drift rate, it is necessary to make several assumptions regarding the position of the crew relative to the CSM c.g., and to postulate a minimum value for the CSM inertia to determine the worst-case condition. These are:

(1) The CSM inertia is 4.0×10^4 slug-ft^2
(2) The maximum distance, L, between the moving mass (i.e., the moving astronaut) and the CSM c.g. is 6.4 ft
(3) The motion has a negligible effect on the CSM c.g.

The change in angular momentum represented by a $\Delta\omega$ of 0.02 deg/sec is

$$\Delta H = J_{csm}\,\Delta\omega = (4.0\times 10^4)\;(0.02)\left(\frac{\pi}{180}\right) = 14 \text{ lb-ft-sec} \qquad (6.3)$$

With a moment arm of 6.4 ft the corresponding impulse is

$$\text{Impulse} = \frac{14}{6.4} = 2.2 \text{ lb-sec}$$

This can be interpreted in terms of varying activities in the following manner. Suppose that the astronaut swings his arm through an arc of 90 degrees.

Let M_a = mass of arm = 0.3 slug (10 lb weight)

I_a = inertia of arm = 0.6 slug ft^2

Δx = change in c.g. of arm = one ft

θ = angle through which arm is moved = $(\frac{\pi}{2})$

Then

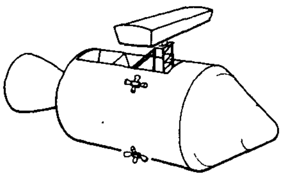
$$\Delta H = I_a\,\Delta\omega_a + M_a\,\Delta V_a\,L \qquad (6.4)$$

$$= I_a\frac{\Delta\theta}{\Delta t} + Ma\frac{\Delta x}{\Delta t}\cdot L$$

And

$$14\,\Delta t = 0.6 \times \frac{\pi}{2} + 0.3 \times 1 \times 6.4$$

$$\Delta t = 0.2 \text{ sec}$$



Since it is unlikely that the astronaut can extend his arm at this rate, it can be concluded that arm motions fall into the "moderate" category.

Consider next the effect of body motions and suppose the astronaut arises from the couch. The motion of the c.g. of the body will be assumed to be one ft and the angular velocity of the body essentially zero.

Then, with

$$M_\beta = 6 \text{ slug},$$

$$14\Delta t = 6 \times 1 \times 6.4$$

$$\Delta t = 2.7 \text{ sec}$$

If the astronaut stands up in less than 2.7 seconds, the motion is classified as "extreme". By restricting motion during the data sample period to arm and head motions, the error due to man motion can be minimized. It should also be noted that for drift rates of 0.05 deg/sec even a rapid, full-body motion can be accommodated.

The effect of disturbances produced by acceleration-induced torques remains to be considered. For this analysis, a motion considerably more severe than any discussed above will be assumed. The motion and assumed conditions are described below:

- A 200 lb mass moves one ft in 0.5 sec.
- The motion is described by a constant acceleration (for 0.25 sec) followed by a constant deceleration.
- The line of the forces applied is 6.4 ft from the CSM c.g.
- The pitch and yaw gimbal axes lie in the plane containing the axis of CSM rotation, which results from the reaction torque impulses applied to the CSM.
- The particular motion and ATM gimbal axis geometry will arbitrarily be considered to affect the ATM pitch orientation only.

Figure 6-8 describes the motion in further detail. Values shown are derived as follows:

Since x at t = 0.25 sec is 0.5 ft, then for $0 \leq t \leq 0.25$ sec,

$$a = \frac{2x}{t^2} = \frac{2 \times 0.5}{(0.25)^2} = +16 \text{ ft/sec}^2$$

and

$$F = ma = \frac{200}{32.2} \times 16 = +99.5 \text{ lbs}$$

and

$$v_{max} = at = +16 \times 0.25 = +4 \text{ ft/sec}$$

also,

$$x(t) = 1/2\, at^2 = 8t^2 \text{ ft}$$

As shown in Fig. 6-9, the ± 99.5 lb reaction force at point A is replaced by the same force applied in a parallel direction at the CSM c.g., and a ± 635 lb-ft couple applied about an axis BC, normal to the force direction and to the 6.4 ft moment arm.

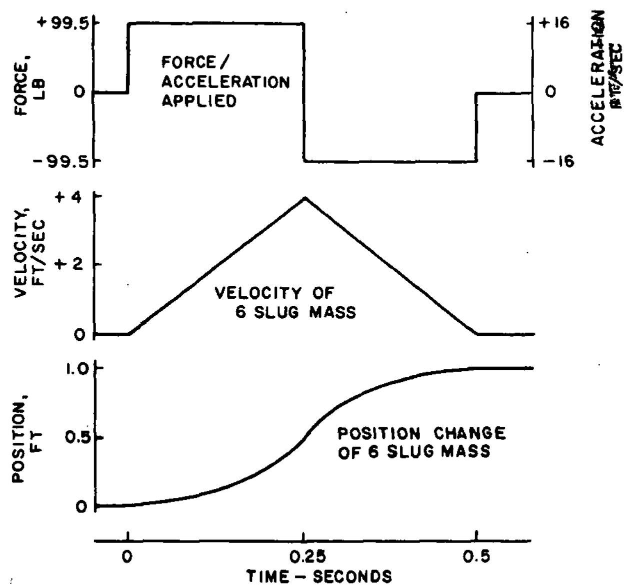


Fig. 6-8 Man-Motion Dynamics

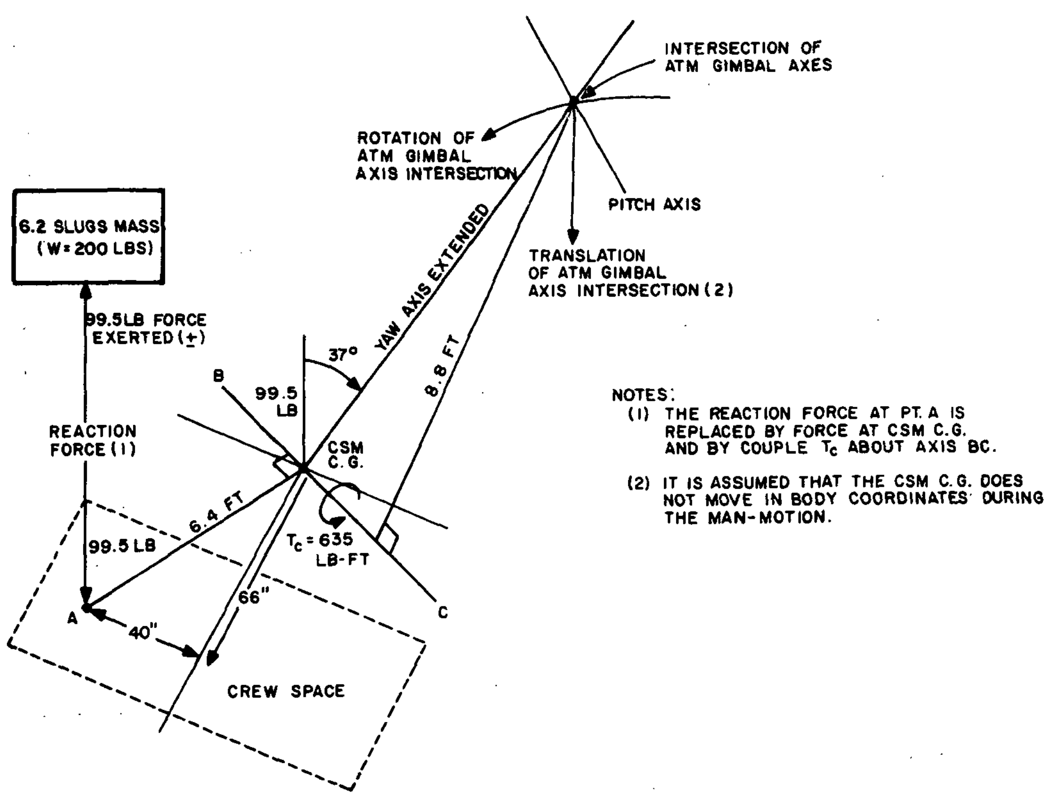
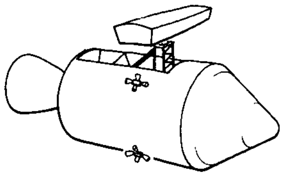


Fig. 6-9 Geometry of "Worst Case" Man-Motion



Accelerations of the ATM gimbal axis intersection can now be calculated. In Fig. 6-10, an arbitrary pitch gimbal center of mass offset, relative to the axis, is shown to be 0.010 inch. Both translational and tangential accelerations of this point cause appreciable reaction forces and torques to be exerted on the pitch gimbal. (Note that the gimbal considered here could also be the yaw or the roll gimbal.)

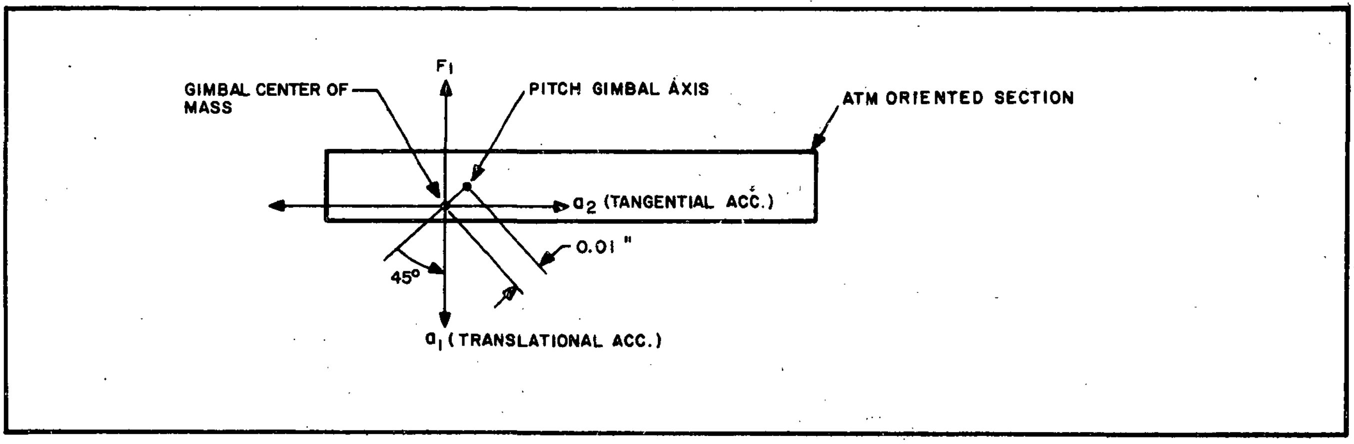


Fig. 6-10 Reaction Forces on ATM Pitch Gimbal

The translational acceleration a_1, for F = ± 99.5 lb and for an estimated minimum CSM weight of $2.0\mathrm{x}10^4$ lb worst-case condition, is given by

$$a_1 = \frac{F}{m} = \pm \frac{99.5 \times 32.2}{2.0 \times 10^4} = \pm 0.16 \text{ ft/sec}^2$$

The tangential acceleration a_2, for T_c = ± 635 lb-ft, for an estimated CSM inertia of $4.0\mathrm{x}10^4$ slug-ft^2, and for a radius from the axis of CSM rotation of 8.8 ft (Fig. 6-8), is given by:

$$a_2 = \frac{T_c r}{J} = \pm \frac{635 \times 8.8}{4.0 \times 10^4} = \pm 0.14 \text{ ft/sec}^2$$

The radial acceleration a_3 of the gimbal center of mass is maximum when ω is maximum at 0.25 sec. At t = 0.25 sec, we have:

$$\omega_{max} = \alpha t = \frac{635 \times 0.25}{4.0 \times 10^4} = 3.96 \times 10^{-3} \text{ rad/sec}$$

and therefore:

$$a_{3(max)} = (\omega_{max})^2 r = (3.96 \times 10^{-3})^2 (8.8) = 1.38 \times 10^{-4} \text{ ft/sec}^2$$

This small maximum radial acceleration is approximately $10^{-3} a_2$ and will be ignored.

Assuming that pitch gimbal motion is negligible as a result of the reaction forces applied, and that the CSM is a rigid body, then for a 31 slug gimbal mass (W = 1000 lb), total reaction torque applied to the pitch gimbal is given by:

$$T_p = \pm \frac{0.707 \times 0.01 \times 31}{12} (a_1 + a_2) \text{ lb-ft}$$

$$T_p = \pm 5.48 \times 10^{-3} \text{ lb-ft}$$

Thus, the error due to this torque is less than 0.06 arc seconds.

6.2.4 Fine Pointing Error Analysis

6.2.4.1 Quantitative Discussion of Errors

For the expected magnitudes of each disturbance input that is considered appreciable (those error sources listed in Section 6.2.1.1) the resulting relative error magnitudes will be computed where possible, or assigned numerical values based on referenced analysis, judgment, and experience.

Fine Sun Sensor Output Signal Drift. Analysis of fine sun sensor performance is presented in Section 6.4, and the conceptual approaches in designing a fine sensor with thermal drift equivalent relative error of ±2 arc sec is presented.

Error Resulting From Man Motion. The man motion-induced errors are discussed in Section 6.2.3. For "moderate" man motion the error is 0.05 arc seconds(max), and when "extreme" man motion occurs the added error is 2.4 arc seconds.

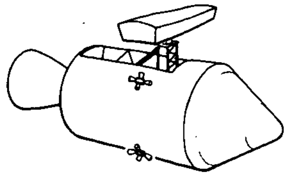
Error Due To Roll-Induced, Pitch-Yaw Cross Coupling. It is shown in Section 6.4.4.2 that the error resulting from the sun sensor cross coupling induced by roll motion is less than one arc sec/min.

Fine Servo Electronics Drift. The scale factor at the power amplifier output is 0.47 V/arc sec (considering the design represented on the fine servo block diagram, Fig. 6-7). For constant inputs to the servo, drift at this output point in a one minute time period, for a 70 ± 5^{o}F electronics environment, can be held to 0.1 V or less. This assertion is based on experience. Equivalent electronics drift relative error is thus expected to be 0.21 arc sec/min or less.

Offset Command Signal Drift. For a ± 1200 arc sec (± 20 arc min) offset range, the offset command generator drift requirement will be specified at ± 0.5 arc sec equivalent angle, for a one minute period, and in a 70 ± 5^{o}F environment. This short-term stability of 0.04 percent is reasonable, considering that the excitation voltage to the potentiometer of this circuit will be regulated to 0.01 percent, and that signal current supplied to the bias input summing point will be only 1.2 ma maximum.

All-Sources of Pointing Jitter. The total of the dynamic error components at fundamental frequencies higher than the two cps servo bandpass will be referred to as jitter. The probable sources of this error are as follows;

(1) Random and periodic small reaction torques on the ATM gimbals (assuming small static unbalance in the gimbals)caused by ATM structural vibrations, CSM vibrations, and random, "moderate" man motions.



(2) Fine servo dynamic error at a structural resonance frequency, with stable servo design. (Transverse ATM structural resonance could cause this effect in yaw, with pitch gimbal angle at an extreme.)

(3) Ripple on motor drive torque

(4) Power amplifier dead zone

(5) Discontinuous gimbal friction and drag characteristic for extremely low velocity operation.

Detailed computation of jitter error for all sources will not be made here. Consider, though, the effects of dynamic torque on the pointed inertia of 270 slug-ft^2, which is the problem implicit in items (1), (3), and (5) above.

At 2.0 cps the sinusoidal torque amplitude required to cause a 0.5 arc sec peak excursion of the 270 slug-ft^2 yaw or pitch inertia is given by

$$T = V_g \omega^2 \theta$$

$$T = 270 \times (2.0 \times 6.28)^2 \times (2.42 \times 10^{-6}) \text{ lb-ft} \qquad (6.5)$$

$$T = 0.103 \text{ lb-ft}$$

Sustained occurrence of 0.1 lb-ft torque load variation at 2.0 cps is highly improbable. Disturbance at higher frequencies should cause less pointing change because of the inverse-square response of the inertia. It appears reasonable at this time to assign a ± 0.5 arc sec value to the total of all jitter error.

<u>Fine Servo Gain Change With Roll Gimbal Angle Change.</u> As roll gimbal angle changes, the yaw and pitch fine sensor gains will decrease with travel away from the zero degrees position, the reduction reaching a factor of about 0.91 (≈cos 25 degrees) at ± 25 degrees. (This cosine factor is not precisely correct.) For a nominal torque load of 0.2 lb-ft, and assuming no other changes, relative error is given by

$$\varepsilon_r = \frac{T}{0.91\, K_{ts}} - \frac{T}{K_{ts}} = \frac{T\,(1-0.91)}{0.91\, K_{ts}} \qquad (6.6)$$

$$\varepsilon_r = \frac{0.20 \times 0.09}{0.91 \times 2.6 \times 10^4} = \pm 7.6 \times 10^{-7} \text{ rad} = 0.16 \text{ arc sec}$$

In a one minute time period, the roll angle change at a CSM inertial drift rate of 0.05 deg/sec is only three degrees, or about 1/8 of the change considered above. In this short time, therefore, a conservative estimate of relative pointing error of 0.05 arc sec will be made.

<u>Change in Gimbal Drag Torque With Yaw Or Pitch Gimbal Angle Change.</u> Maximum gimbal drag change for any range of gimbal travel in yaw or pitch is estimated at 0.1 lb-ft, or about ½ of the maximum static drag level of 0.22 lb-ft.

Relative error for this torque change T is given by

$$\varepsilon_r = \frac{T}{T_{cs}} \qquad (6.7)$$

$$\varepsilon_r = \frac{0.1}{2.6 \times 10^4} = 38 \times 10^{-7} \text{ rad} = 0.78 \text{ arc sec}$$

6.2.4.2 Error Total

Error Source	Peak Relative Error Expected, Arc Sec in one min time
• Fine sun sensor output drift	±2
• Torque disturbance caused by man motion	--
• Error due to roll-induced, pitch-yaw cross coupling	±1
• Fine servo electronics drift	±0.21
• Boresight command signal drift	±0.5
• All sources of pointing jitter	±0.5
• Fine servo gain change with roll gimbal angle change	±0.05
• Change in gimbal drag torque with yaw or pitch gimbal angle change	±0.78
Root-sum-square total:	±2.24
Peak total:	±5.04

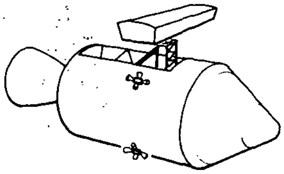
6.3 ACQUISITION

The pitch-yaw acquisition servos can be used for coarse acquisition and for maintaining gross pointing (±2.0 degrees) when the fine sun sensors are disabled by closing the spar aperture cover. To the extent possible, the same equipment is used for the coarse servos as in the fine, and this is primarily the power amplifiers and torque motors. Separate sun sensors, called coarse sun sensors, are required as well as different stability networks.

6.3.1 Design Considerations

6.3.1.1 Controlled Slewing Rates

When in the manual control mode, the astronaut commands controlled-rate slewing maneuvers to either change gimbal angles to a desired orientation, or to actually acquire the coarse solar pointing range (50 deg x 50 deg) required by the automatic control mode. The concept employed to limit the slewing rates is to obtain rate information from linear gimbal angle transducers, and to apply this rate to the drive electronics to form rate-limiting slewing loops. At this time, the rate limit will be about 6 deg/sec in each slewing loop(including roll). This value is readily changeable, and will be set by operational considerations. To slew ±170 degrees in yaw at a 6 deg/sec rate, the approximate elapsed time of 30 seconds appears reasonable. The slewing rate limit is sized by other factors also, such as gimbal stop capability and gimbal angle overshoot



that occurs when the astronaut applies brake torque. At 6 deg/sec the gimbal stop would have to absorb 1.35 ft-lb of energy in pitch or yaw. Other means to limit slewing rates that were considered in the study are to use gyro or tachometer rate sensing devices for each gimbal. Gyro controls were rejected primarily because of their power requirements; the tachometer concept was discarded because of the additional mounting problems it entailed.

6.3.1.2 Torque Motor Sizing

Yaw and pitch motor torque requirements are primarily derived from the acceleration periods allowed for the gimbals to reach their controlled slewing rates of 6 deg/sec. This acceleration time will be set at five seconds. Maximum motor torque required is given by:

$$T = J_g \times \frac{\omega}{t} + T_f$$

$$T = 270 \times \frac{6}{57.3 \times 5} + 0.22 \text{ lb-ft} \qquad (6.8)$$

$$T = 5.8 \text{ lb-ft}$$

(This torque value, computed on the basis of required gimbal acceleration, is 25 times the static friction torque loading of 0.22 lb-ft; a ratio of 10 would be adequate.)

6.3.1.3 Transducer Requirements

Gimbal Angle Transducers. DC signal voltages proportional to instantaneous gimbal angle through the full gimbal travel ranges in each ATM axis will be generated. The design concept here is to use high-resolution, linear potentiometers. These gimbal angle transducers will be suitable as position sensing elements in automatic ATM gimbal restow position servo designs, should restow control capability become a requirement in the future.

Coarse Sun Sensors. For yaw and pitch, DC signal voltages that indicate solar orientation of the ATM spar over ±25 degree ranges will be generated. These voltages will afford the astronaut an analog indication of coarse solar pointing error during manual acquisition maneuvering. The design concept is to employ silicon solar cells in differential pairs in a special wide-view indicator sensor, and then to transform the output signal current to voltage in simple signal conditioning electronics amplifiers. The astronaut can select these yaw and pitch signals on the programmable indicating meters on the control unit when desired.

6.3.1.4 Automatic Control

Torque Sensitivity. A gimbal frictional torque in either yaw or pitch of up to 0.22 lb-ft will be driven by a static coarse error in steady state coarse pointing. The pointing requirement of the coarse automatic control is to reduce this static error to within the fine sun sensor ±5 deg field of view. Ideally, the null axes of both coarse and fine control sensors should coincide so that settling transients in the fine pointing control would be negligible as the aperture cover is opened. Because of stray light reflections perturbing the coarse sun sensors, however, this alignment coincidence is unlikely. The criterion in selecting coarse servo torque sensitivity will be to allow a "moderate" static coarse error of ±0.20 deg to sustain the static ±0.22 lb-ft friction. Torque sensitivity is given by:

$$K_{ts} = \frac{T_f}{E_c} \tag{6.9}$$

$$K_{ts} = \frac{0.22}{0.20} = 1.1 \text{ lb-ft/deg}$$

Selection of Compensation Dynamics. The compensation network design has two requirements: (1) It must stabilize the servo loop, which is paramount, and (2) it must allow acquisition in a reasonable time from the maximum possible error.

A phase plan plot of automatic coarse acquisition in yaw or pitch, for initial errors of -20 deg/sec and zero deg/sec is shown in Fig. 6-11. The switching line has been selected to minimize acquisition time, and is shown for a rate-to-position gain ratio of 2.04. The two segments, A and B, are defined by the following equations:

- General Form:

$$\varepsilon = \frac{J_g}{2(\pm T_m \pm T_f) \times 57.3} (\dot{\varepsilon}^2 - \dot{\varepsilon}_0^2) + \varepsilon_0 \text{ deg} \tag{6.10}$$

- Segment A (For + T_m, - T_f):

$$\varepsilon_a = 0.437 (\dot{\varepsilon}^2 - 0) - 20 \text{ deg} \tag{6.11}$$

- Segment B (For - T_m, - T_f) :

$$\varepsilon_b = 0.407 \left[\dot{\varepsilon}^2 - (4.9)^2\right] - 10 \text{ deg} \tag{6.12}$$

In Equation (6.10) $\pm T_m$ is the maximum static motor torque of ±5.6 lb-ft, $\pm T_f$ is the static friction level of ±0.2 lb-ft, and J_g is the yaw/pitch gimbal inertia of 270 slug-ft^2.

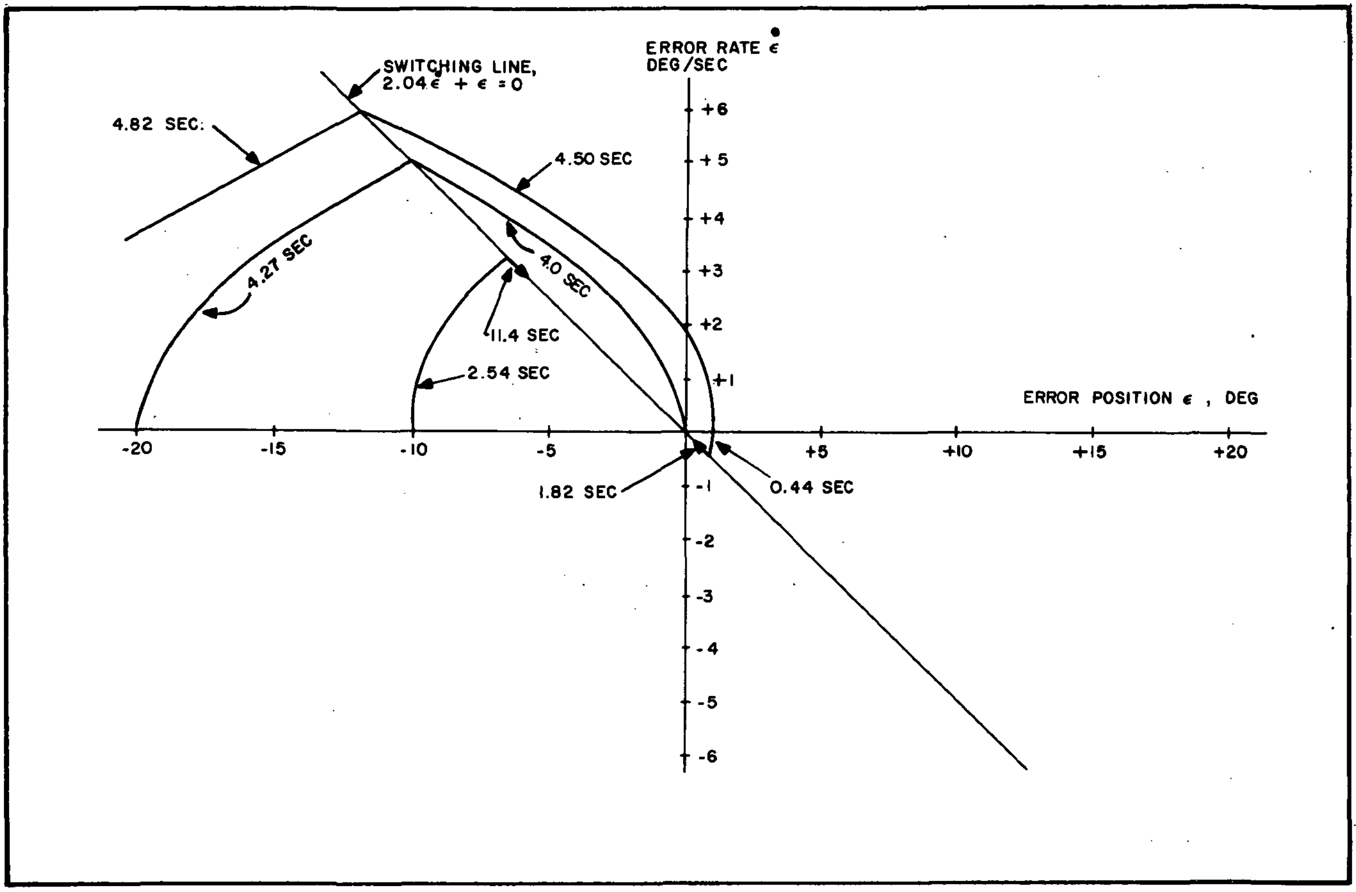


Fig. 6-11 Phase Plane Plot of Yaw/Pitch Automatic Coarse Acquisition

The time interval for segment A is given by:

$$t_A = \left| \frac{J_g}{(+T_m + T_f) \times 57.3} (\dot{\varepsilon}_2 - \dot{\varepsilon}_1) \right| \qquad (6.13)$$

$$t_A = \left| 0.87\ (4.9-0) \right|$$

$$t_A = 4.27 \text{ sec}$$

and for segment B by:

$$t_B = \left| \frac{J_g}{(+T_m + T_f) \times 57.3} (0-\dot{\varepsilon}_2) \right| \qquad (6.14)$$

$$t_B = \left| 0.81\ (o-4.9) \right|$$

$$t_B = 4.0 \text{ sec}$$

When the servo enters the linear region the above equations do not apply. The time to go from θ_1 to θ_2 in linear operation is approximately

$$\int_0^t dt = -\frac{1}{z}\int_{\theta_1}^{\theta_2} \mathrm{ctn}\theta d\theta \qquad (6.15)$$

Where "z" is the zero in the transfer function,

or

$$t = \frac{1}{z} \ln\frac{\sin\theta_1}{\sin\theta_2} \; ; \; \theta_2 > 0 \qquad (6.16)$$

Not surprisingly, the longest coarse acquisition is not the largest initial error. As shown in Fig. 6-11, the time to stabilize at a 0.2 deg error (the angle where the control torque counterbalances the friction) is indicated, and summarized below.

Initial Coarse Error (degrees)	Time to Stabilize at 0.2 deg (seconds)
25	10.58
20	8.27
10	13.94

Since the gimbals are locked prior to coarse acquisition, the assumption of zero initial rate is justified.

The switching line on the acquisition plot is given by

$$2.04\,\dot{\varepsilon} - \varepsilon = 0$$

This relationship can be effectively achieved in a simple coarse compensation lead network if the lead time constant is two seconds and the network α is at least 10. An α of 20 can be readily mechanized and is shown in the block diagram that follows.

6.3.1.5 Block Diagram

Figure 6-12 is a simplified block diagram of the yaw/pitch acquisition control. The numerical values shown are those derived in concept design, undertaken to define possible mechanization problems. The block diagram is simplified in that all transfer functions are not represented numerically, and that several high frequency dynamic responses are not shown. The purpose here, however, is to present conceptual design, and the omitted design detail does not affect this effort.

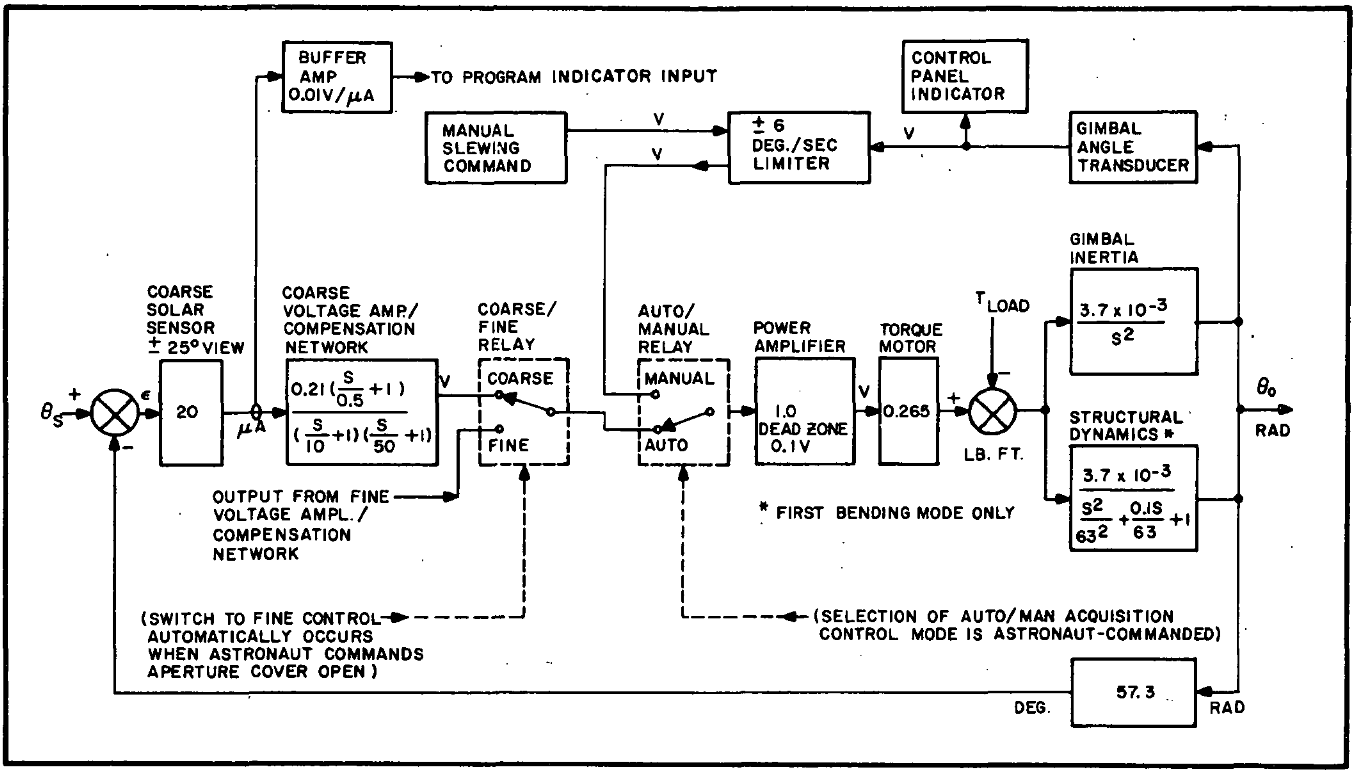


Fig. 6-12 Yaw/Pitch Acquisition Control Block Diagram

As many of the fine pointing control elements as possible are employed in the yaw/pitch acquisition control. These are the power amplifiers, the torque motors, and the gimbal structural configurations. The block diagram shows that if the astronaut commands the manual acquisition control mode, the coarse (or fine) position control loop is opened, and only the capability to slew the gimbals at a ±6 deg/sec limited rate is available. When in automatic mode operation, the pointing control becomes operable for automatic coarse acquisition provided the aperture cover is closed (the coarse/fine relay is in the COARSE position), and provided that the following coarse pointing requirement is met: the solar direction must lie within the ±25 degree viewing ranges of both the yaw and pitch coarse sun sensors at the time automatic control is commanded. This is indicated by the coarse sun indication on the control unit. The orientation must be performed manually by the astronaut by commanding, if necessary, both the CSM RCS control and the ATM manual acquisition control, with the aid of the gimbal angle and coarse solar orientation indicators.

6.3.1.6 Electronics

Coarse Voltage Amplifier/Compensation Network. The concept design approach in mechanizing this circuitry is exactly the same as that already described for the fine servo voltage amplifier/compensation network. The similarity between these coarse and fine circuits will permit them to have identical designs except for a few resistors and capacitors, which establish the gain and dynamics required.

Slewing Rate Limiter. The gimbal angle rates of change will be generated by differentiating the linear output signals from the gimbal angle position transducers. Passive circuit elements will perform this function, and filtering of the rate signal noise introduced by small discontinuities in the transducer signals will be employed to avoid electronics amplifier saturation and high power consumption. The large gimbal inertias will slew smoothly for the drive torque noise levels allowed. Controlled slewing rate achieved will lie within ±10 percent of the nominal rate requirement (at present ±6 deg/sec), since precise rate control is not necessary.

6.4 SUN SENSORS

The requirements outlined in Section 6.1 define the performance features and requirements for the pointing control subsystem. In this section these requirements on the PCS are interpreted in terms of the restrictions and demand they place on the sun sensors. After having established the requirements on the error sensor, several conceptual approaches to the solution of the sensor design problem can be formulated. These conceptual approaches can then be evaluated and compared to select the approach which best satisfies the requirements. A detailed examination of the recommended approach is made in Section 6.4.4.

6.4.1 Design Requirements

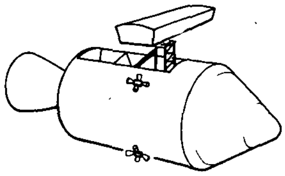
The design requirements for the sun sensors are categorized into four groups as follows:

(1) Those requirements which are common to both coarse and fine modes (reliability, power, etc.).
(2) Those requirements which apply only to the coarse control mode.
(3) Those requirements which apply to the fine control mode.
(4) Those requirements which apply to the in-field sensor.

These requirements are discussed with notation of the relevant PCS requirement and the implications on the general sensor design in the following sections.

6.4.1.1 General Design Requirements

In addition to the usual minimum size, power and weight requirements generally associated with spaceborne hardware, the ATM requires highly reliable designs and mechanizations. The reliability requirement is considered to be one of the prime criteria in the evaluation of sensor concepts. As a result of this requirement, concepts incorporating moving parts and/or fragile components must suffer a severe penalty during their evaluation. On the other hand, those concepts possessing design simplicity and utilizing space-proven techniques will be emphasized.



6.4.1.2 Coarse Sun Sensor

The design requirements on the coarse sun sensor are summarized below:

(1) The sensor must not be disabled when the aperture covers of the instrument package are closed. This requires that the sensor be mounted outside the oriented section and, therefore, be immune to degradation from contamination.

(2) The sensor must have a field of view greater than 25 degrees but less than 35 degrees half-angle centered about the solar vector. The sensor null accuracy must be better than 1.5 degrees. In satisfying this requirement it will be assumed that the earth is not within the acquisition field; therefore, albedo will be ignored except as a requirement to limit the field of view of the coarse sensor to the acquisition field.

(3) The sensor output must be a monotonically increasing function of increasing error angle for error angles in the range from -25 degrees to +25 degrees about each of two orthogonal axes normal to the sensor line of sight (pitch and yaw).

(4) The peak-to-peak noise on the output within the frequency range dc to 100 Hz must be smaller than the output corresponding to a 0.5 degree change in the error angle.

(5) The angular gain or scale factor of the error signal in the vicinity of the on-target condition should be as high as conveniently obtainable.

6.4.1.3 Fine Sun Sensor

The design requirements on the fine sun sensor are summarized below:

(1) The sensor must have a field of view sufficient to assure that the sun is within this field when control is transferred from the coarse to fine mode. Since the on-target or null accuracy of the pointing in the coarse mode is better than two degrees, a conical field of view with a half-angle of four degrees is sufficient to meet this requirement.

(2) The sensor long-term, on-target accuracy must be better than 40 arc seconds.

(3) The sensor must have a linear range sufficient to accommodate offsets of at least 20 arc minutes. Ten percent linearity over a range from -20 arc minutes to +20 arc minutes in both pitch and yaw axes is considered sufficient to satisfy the offset requirement.

(4) The cross-coupling between the sensors for pitch and yaw control axes must be small enough to enable the astronaut to easily position the instruments. An equivalent nonorthogonality between the sensor control axes of less than one degree is adequate.

(5) The scale factor of the sensor error signal over the linear range should be as high as conveniently attainable within the size and weight requirements of the sensor.

(6) The short-term (for one minute of time) stability of the sensor error signal over its linear range must be better than two arc seconds. This is considered the most critical performance requirement on the sensor, since reduced performance will result in smear and field offset and, therefore, reduced resolution and knowledge of the field in the experiment data.

(7) The peak-to-peak noise on the output within the frequency range dc to 100 Hz must correspond to an equivalent angular noise of less than 0.5 seconds.

6.4.1.4 Design Requirements for the In-Field Sensor

The design requirements for the in-field sensor are summarized below:

(1) The sensor must not be disabled when the aperture covers of the instrument package are closed. This requires that the sensor be mounted outside the oriented section and, therefore, be immune to degradation from contamination.

(2) The sensor must have a conical field of view of half-angle greater than 25 degrees but less than 35 degrees. The sensor must produce a conveniently high output when the sun is within the field of view and negligible output when the sun is outside the field of view.

(3) The peak-to-peak noise on the output in the frequency range dc to 100 Hz must be smaller than five percent of the output produced when the sun is within the field of view.

6.4.2 Types of Sensor Designs Considered

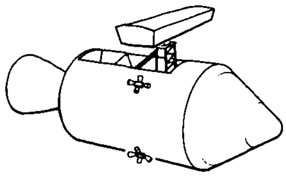
In this section are presented the various types of sensor designs considered in this study. The considerations are divided into three categories: coarse sun sensor, fine sun sensor, and in-field sensor. The results of the evaluation of the various types discussed in this section are given in Section 6.4.3.

6.4.2.1 Coarse Sun Sensor Design Concepts

Three basic coarse sun sensor design concepts were considered:

- Illumination-balancing concept
- Scanning spot concept
- Multiple detector concept

Illumination-Balancing Concept. In this concept the field of view of the sensor is divided into four quadrants and the illumination reaching the sensor from each of the quadrants is compared. Consider the illustration of the sensor's field of view shown in Fig. 6-13. If the sun is seen against a relatively dark background, the error in the pitch orientation, θ_p, is a function of the difference between the sum of the illumination from quadrants A and B and the sum from C and D. Similarly, the yaw error is a function of the difference between the sum from B and C and the sum from A and D. This scheme can easily be implemented by combining four sensors in an array so that each sensor can only detect illumination from one of the four quadrants. However, since outputs which are not cross coupled between the pitch and yaw control axes are highly desirable, it is easier to use four sensors in such manner that each sees half the field. If the sensors are arranged in pairs so that one sensor looks at half the field (say, quadrants A and B) and the other sensor looks at the other half (C and D), then the algebraic difference in their outputs is a function of the



error angle (θ_p in the case considered). Two such sensor pairs mounted in an array with their field separations at right angles provide both pitch and yaw error signals which are not cross coupled. This concept is illustrated in Fig. 6-14.

Scanning Spot Concept. This concept utilizes a sensor with a small instantaneous field of view, ω, which is deflectable over a larger field, Ω. Consider Fig. 6-15 in which a linear-type scan is illustrated (spiral and rosette scans could also be used). If the output of the sensor is used as a trigger to sample the deflection when the sun is within the small instantaneous field of view, then the deflection readouts in x and y can be used as the pitch and yaw error signals.

This scheme can be mechanized in several ways, using both mechanical and electrical techniques. For example, a sensor with a small instantaneous field of view can be moved as a unit to generate the scan, or the beam of light incident on the sensor can be deflected by a moving mirror or rotating optical wedges. The scanning can be achieved electrically by imaging the scan field on the face of an image disector tube and deflecting the sensitive spot on the tube face with magnetic fields. This scheme is illustrated schematically in Fig. 6-16.

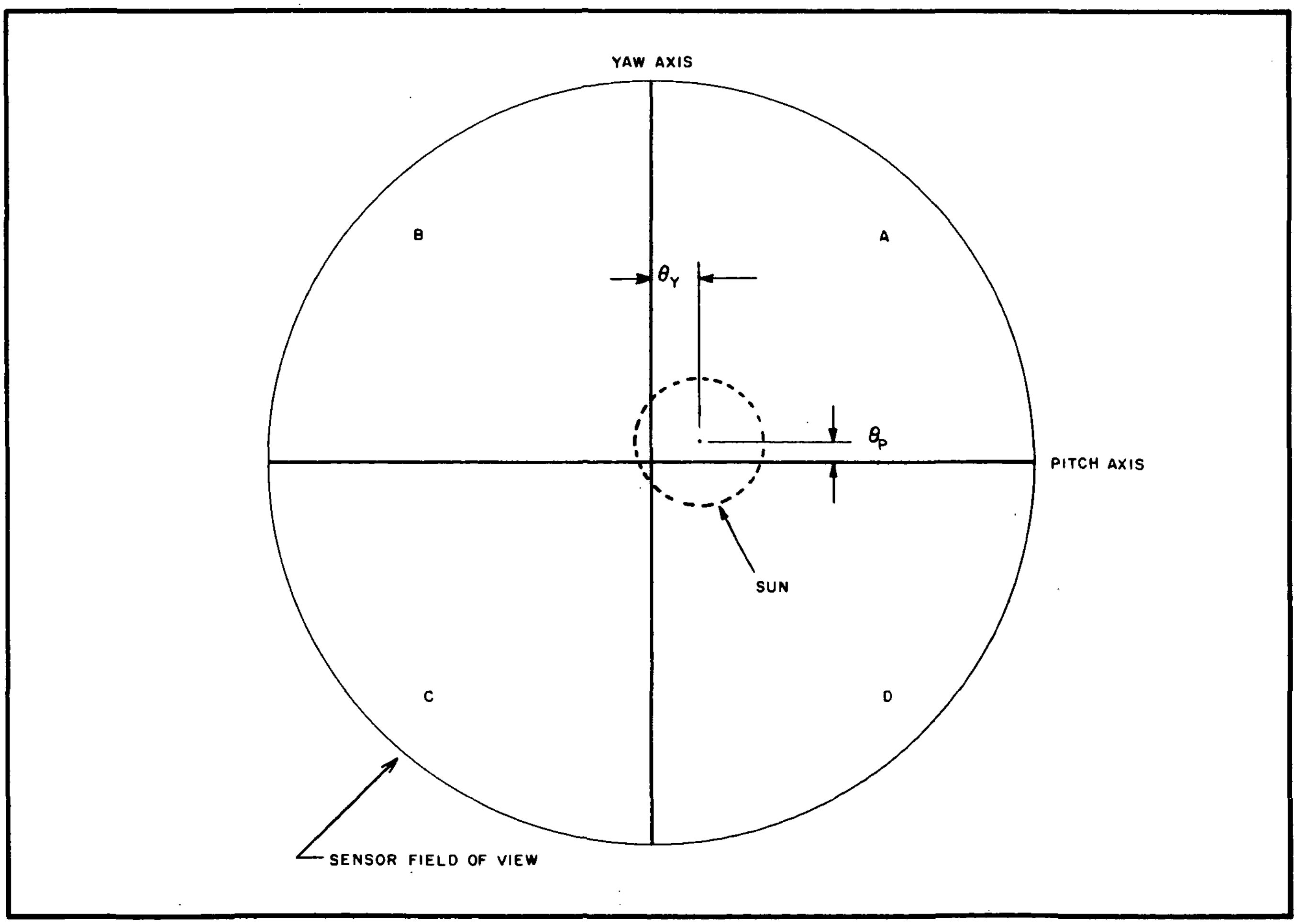


Fig. 6-13 Coarse Sensor Illumination Balancing Technique

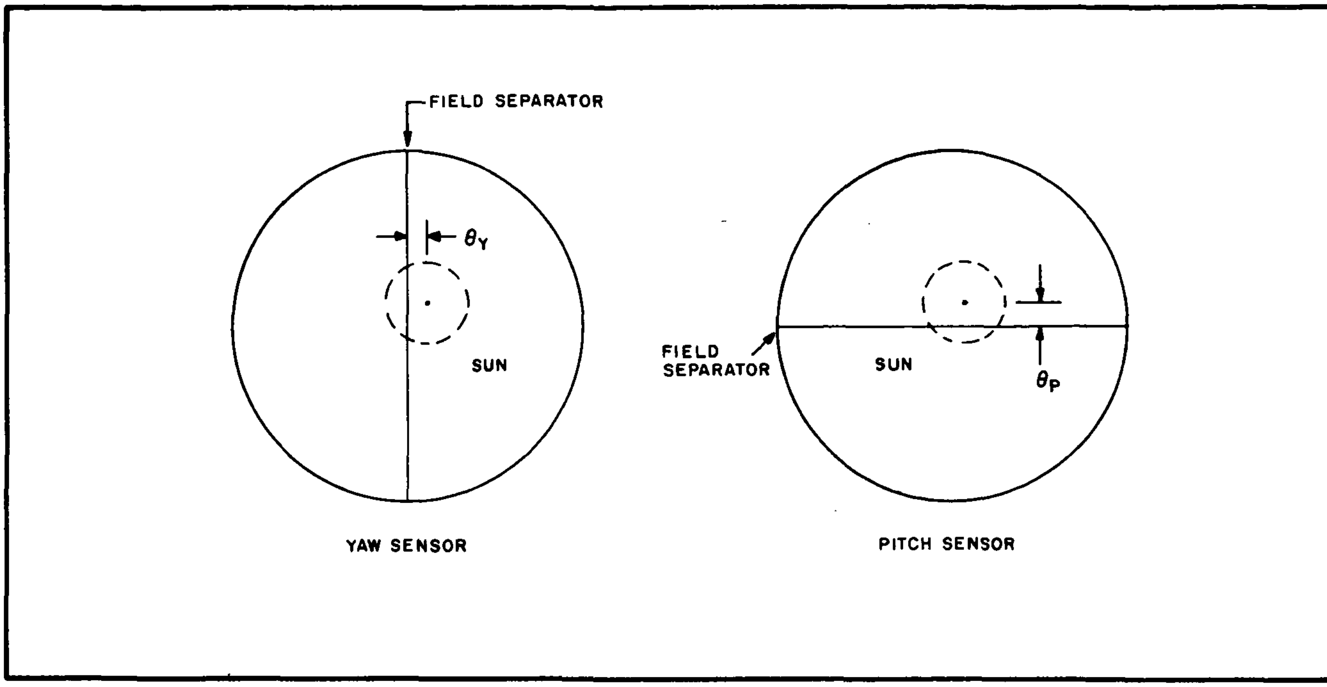


Fig. 6-14 Coarse Sensor Illumination Balancing Concept

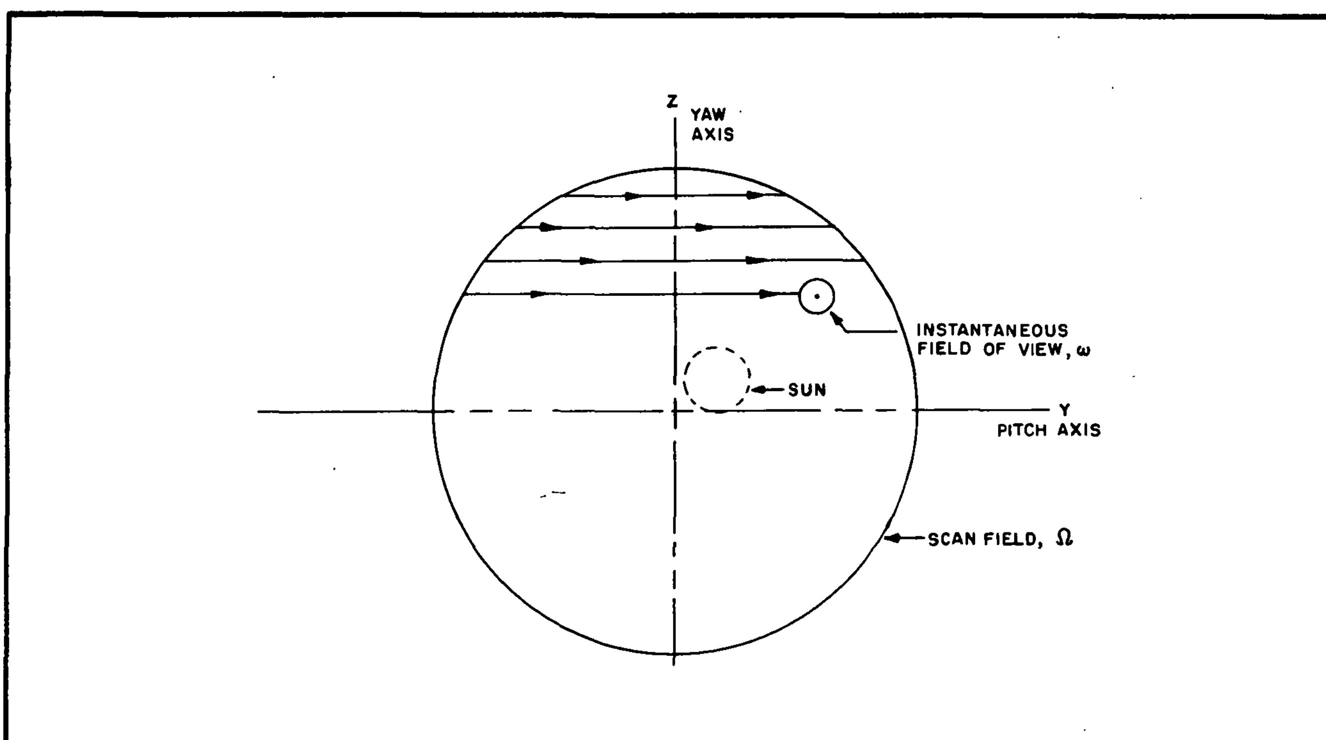
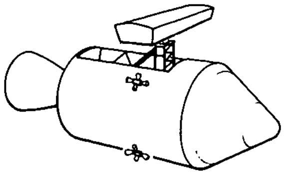


Fig. 6-15 Coarse Sensor Scanning Spot Technique



Multiple Detector Concept. This concept utilizes a lens to image the field on a detector array. The solar position is then determined by interrogating the detector array to find which element is illuminated. The position of the illuminated detector in the array is resolved into pitch and yaw components to generate the error signals. This concept is illustrated schematically in Fig. 6-17.

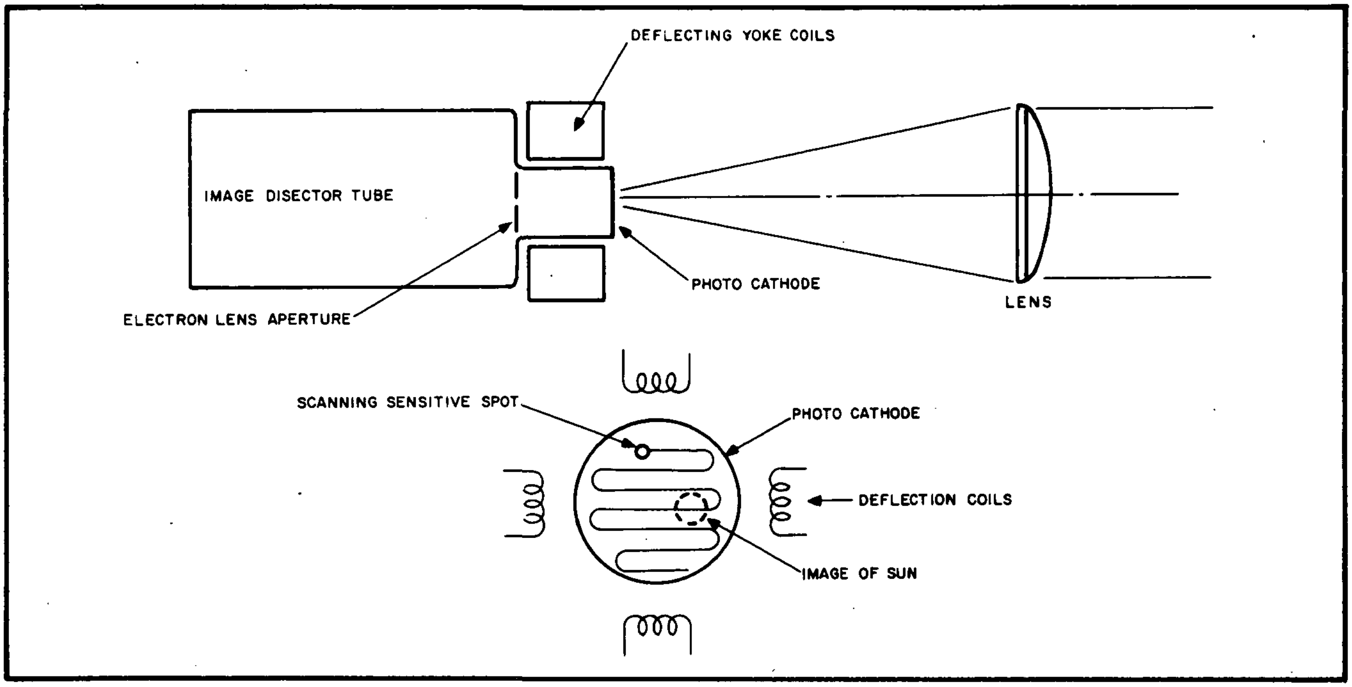


Fig. 6-16 Coarse Sensor Scanning Image Detector Concept

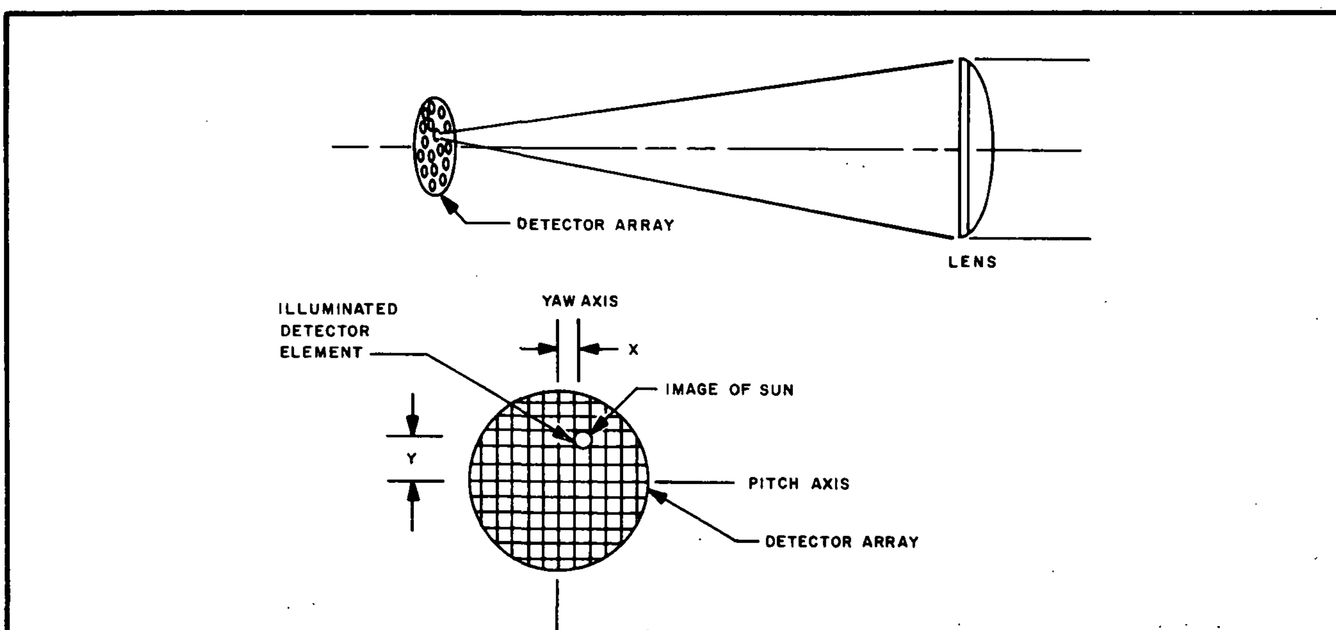


Fig. 6-17 Coarse Sensor Multiple Detector Technique

6.4.2.2 Fine Sun Sensor Design Concepts

The design problems posed by the fine sun sensor are in many ways directly related to the coarse sun sensor design problems and, therefore, concepts which satisfactorily solve the coarse design problems are in general applicable to the fine sun sensor design. However, there are two basic problems which are unique to the fine sensor, which are:

(1) The accuracy and hence the resolution increment for the fine sensor is smaller than the angular diameter of the sun.

(2) The sensor must not only operate as a nulling sensor, but must also provide offset point capabilities.

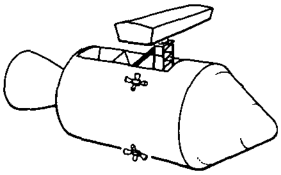
In addition to these problem areas, there is a basic change in the mechanization of the pointing control subsystem. This change occurs because of the availability of the astronaut as a control element. The prime requirement of the PCS becomes one of maintaining the attitude of the system in some orientation selected by the astronaut. This control concept allows the use of rate controlling sensors as position controlling sensors since the control function can be interpreted as one of holding the rate of motion of the system to some small value. The initial position is selected by the astronaut using the solar monitor subsystem.

The sensor mechanization concepts are categorized into the following groups for this discussion:

- Illumination-balancing concepts
- Scanning spot concepts
- Rate-sensing concepts

Illumination-Balancing Concepts. The illumination-balancing concept is based on the same principle as that described in Section 6.4.2.1, except that the total field is much smaller and the concept of field separator toe-in is utilized. Consider the illustration shown in Fig. 6-18. The illumination reaching the sensor from side A is transmitted to detector A, while the illumination from side B falls on detector B. The toe-in of the field separators is indicated by the cross-hatched area. The toe-in angle, τ, is adjusted to optimize the linearity of the error signal. Figure 6-19 illustrates a typical mechanization of this concept with the resulting sensor characteristic.

From the error signal characteristic it is clear that offset pointing can be implemented by simply summing an electrical bias with the sensor output. This bias shifts the apparent zero point presented to the PCS, resulting in a stable null condition at the offset point corresponding to the particular bias introduced. The offset can also be implemented by articulating the sensor directly or by deviating the incident beam with a movable mirror or a rotatable optical wedge. Figures 6-20 and 6-21 illustrate possible mechanizations of the articulated detector and optical wedge concepts. In the articulated detector concept the detector array is mounted



on a traveling slide. In order to change the null point, the slide is commanded to the new position, thus moving the detector array. The servo will then renull the system at the new offset point. In the rotating optical wedge concept, the null point is changed by rotating a wedge directly in front of the sensor lens. Rotation of this wedge causes the beam to be deflected through an angle, α, given by:

$$\alpha \approx \Phi \sin \beta \tag{6.17}$$

where α is the projection of the beam deviation on the sensitive axis of the detector

Φ is the deviation angle of the wedge

β is the angle between the wedge axis and the sensor insensitive axis

Since the sensor is insensitive to image motion along the occulting blade, the effective offset angle is equal to α and the PCS will null the system at this offset angle.

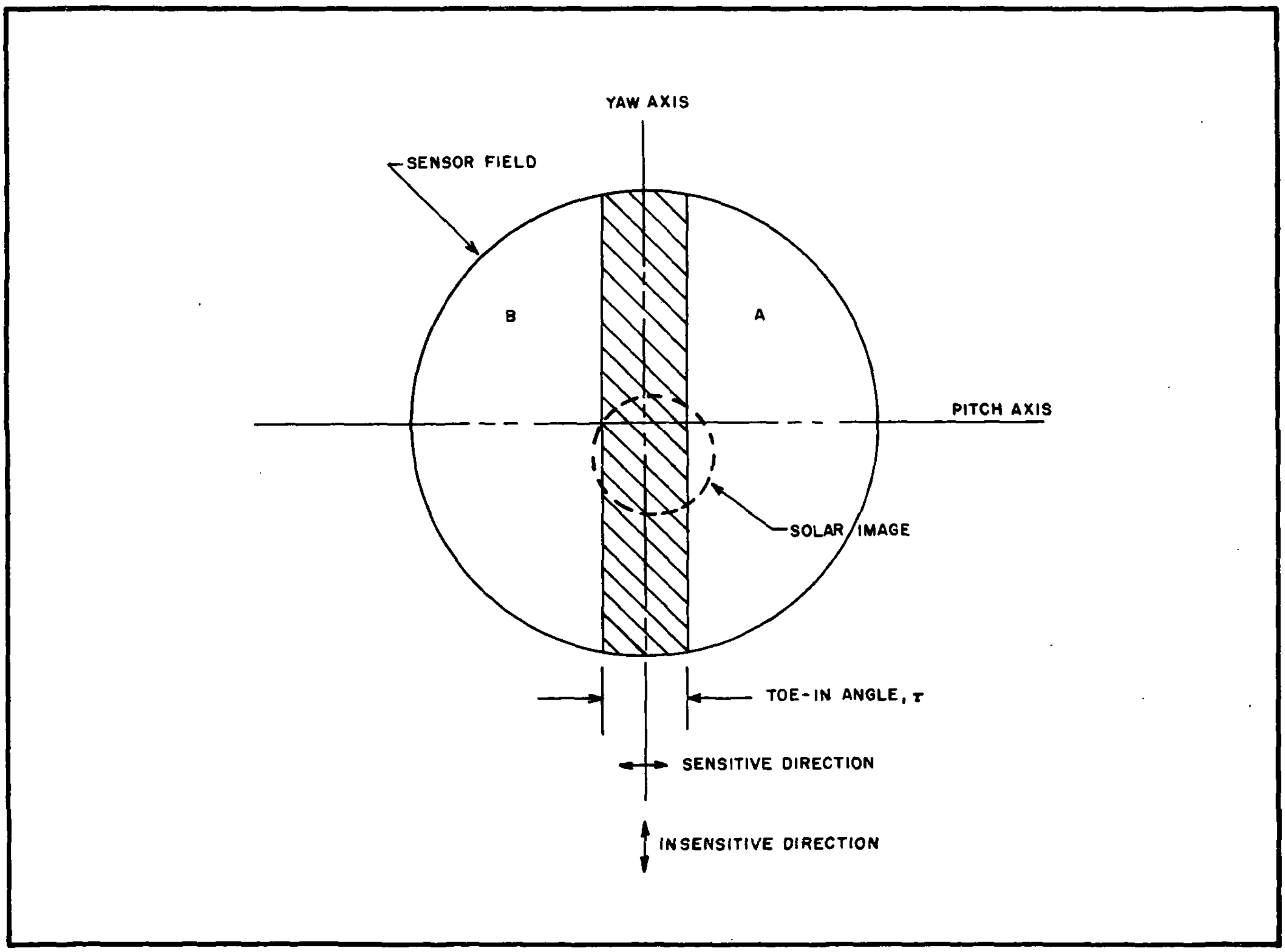


Fig. 6-18 Fine Sensor Illumination Balancing Technique

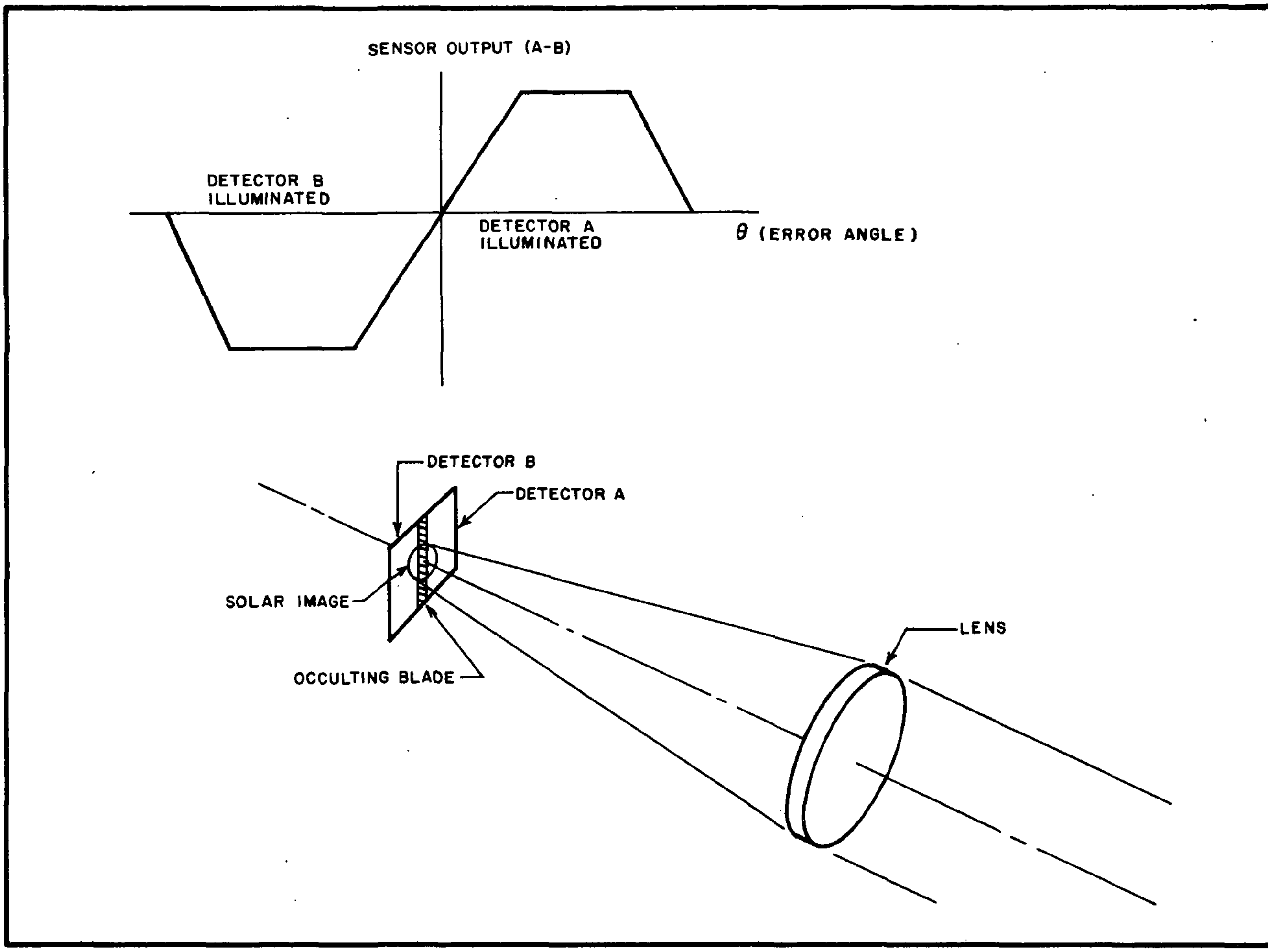


Fig. 6-19 Fine Sensor Illumination Balancing Mechanization Concept

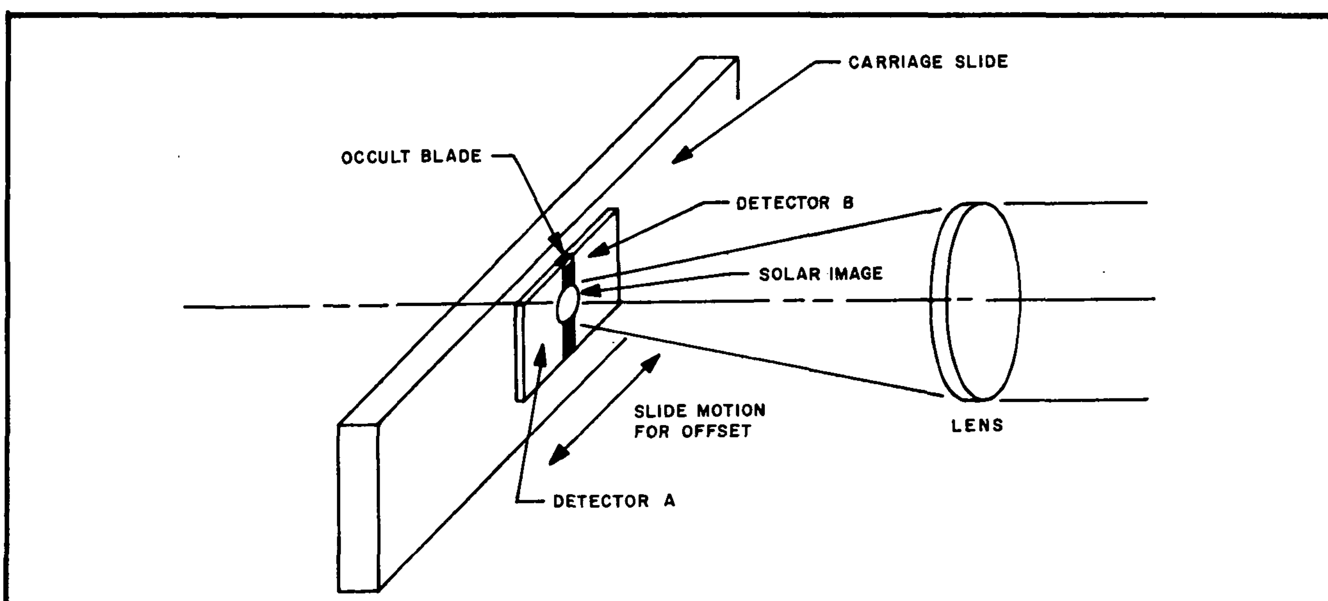


Fig. 6-20 Fine Sensor Detector Array Concept

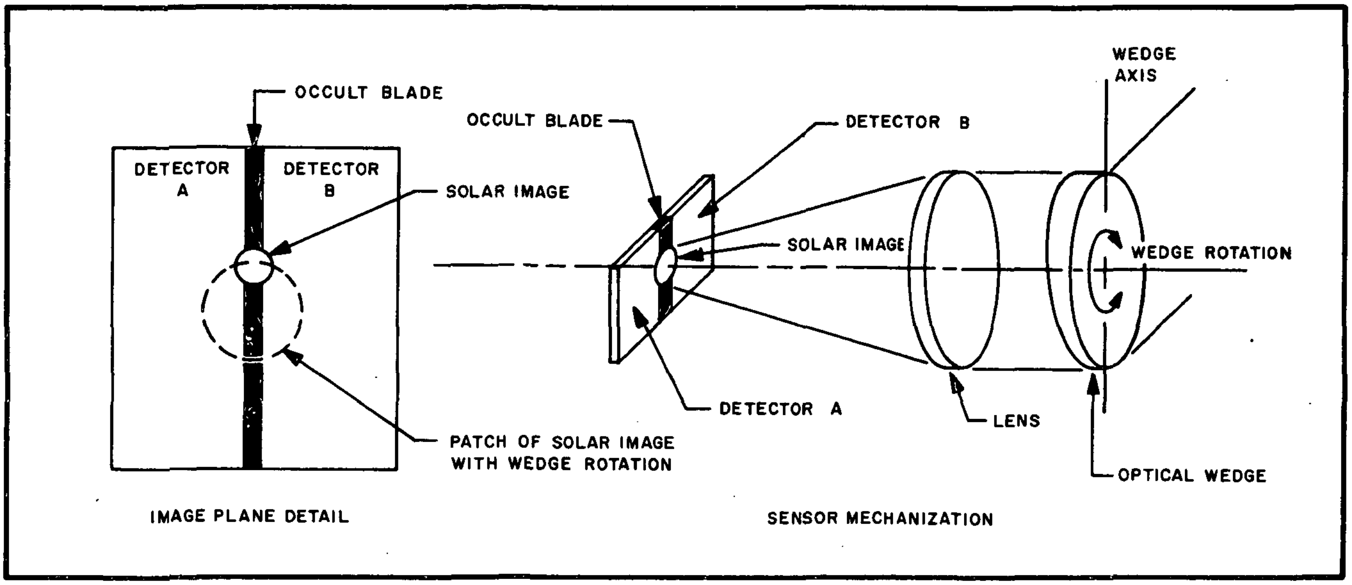


Fig. 6-21 Fine Sensor Optical Wedge Concept

Scanning Spot Concepts. Several scanning spot concepts were considered. In this section, only those concepts which generate position information are discussed. Those concepts which generate rate signals are discussed in the next section.

The scanning spot concepts considered here are much the same as those considered in the coarse sun sensor discussion with the exception that the mechanization is complicated by the requirement of achieving resolution smaller than the apparent angular diameter of the solar disc. This resolution requirement necessitates that we obtain more information rather than rely on a simple sampling of the deflection when the sensor is triggered by the spot sweeping across the sun. In fact, the resolution requirement associated with a short-term accuracy of two arc seconds dictates that resolution be made of just when the spot crosses the sun to within better than one part in a thousand.

Consider the general (in the sense that the scheme does not depend upon the particular method of generating the "sun-presence signal" or the deflection signals) method of sampling the deflection signals illustrated in Fig. 6-22. The deflection signal is sampled by the gate only during the time when the sun is within the instantaneous field of view of the sensor. This sample is fed to an averaging circuit which integrates the signal during the pulse duration. The clipped sun-presence signal is also integrated during the pulse time. The ratio of the integral of the deflection signal to the integral of the clipped sun-presence signal is stored in a holding circuit until the next pulse arrives. The output of the holding circuit is presented to the PCS as the error signal.

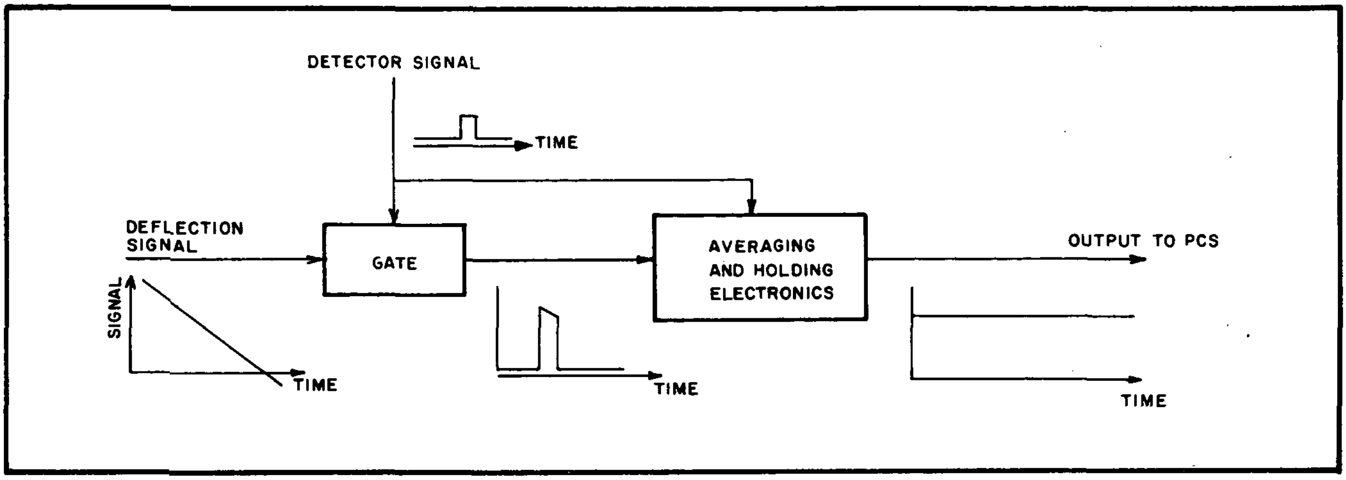


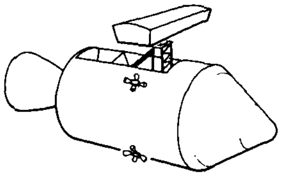
Fig. 6-22 Fine Sensor Scanning Spot Error Signal Generation

With the exception of the added concept discussed above, the approaches discussed for the coarse sun sensor can be scaled to the smaller field of view and applied to the fine sun sensor. Another concept which was considered consisted of utilizing the scanning spot technique with a very small instantaneous field to track solar disturbances directly. The mechanization of the sensor requires the use of a narrow-band filter centered on a strong emission line such as Hα. The sensor would track either the disturbance of interest or one nearby. Selection of the particular disturbance to be tracked could be implemented by allowing the sensor to search a small field near the desired offset point and lock on to any suitable target within this field. The solar monitor display, displays derived from the experiments, or a separate sensor camera could be utilized as the sensing element. The method of generating error signals would be similar to the method described earlier in this section.

Rate-Sensing Concepts. The scanning spot technique can also be used to generate rate signals about pitch, yaw, and roll. These signals can then be presented to the PCS as rate signals or integrated in a short-time-constant integrator to provide position signals. Consider the concept illustrated in Fig. 6-23. Signals can be generated from the step in intensity at the limb of the sun (excluding the roll sensor), or if a narrow-band filter is used, from detail on the solar disc. These signals can be synchronously demodulated against the scan frequency to produce a signal proportional to the rate of motion of the field. Rates in the direction of the scan cause the frequency content in the output to be shifted to a frequency lower than the basic scan frequency, and rates against the scan shift the frequency to a higher level. Linear-type scans can be used to detect pitch and yaw rates while a pin wheel or spiral scan would be used to detect roll rates.

6.4.2.3 In-Field Sensor Conceptual Design

The in-field sensor requirements are amenable to several approaches including the scanning spot technique. However, only one concept was considered in the study



since this concept possessed all the features required in a very simple and reliable design. The concept considered is illustrated in Fig. 6-24. The aperture limits the field of view of the detector to a cone of approximately 25 degree half-angle. When the sun is within this field, the output of the detector follows a cosine curve centered about the on-target condition. As the sun leaves the field, the output drops to a negligible value.

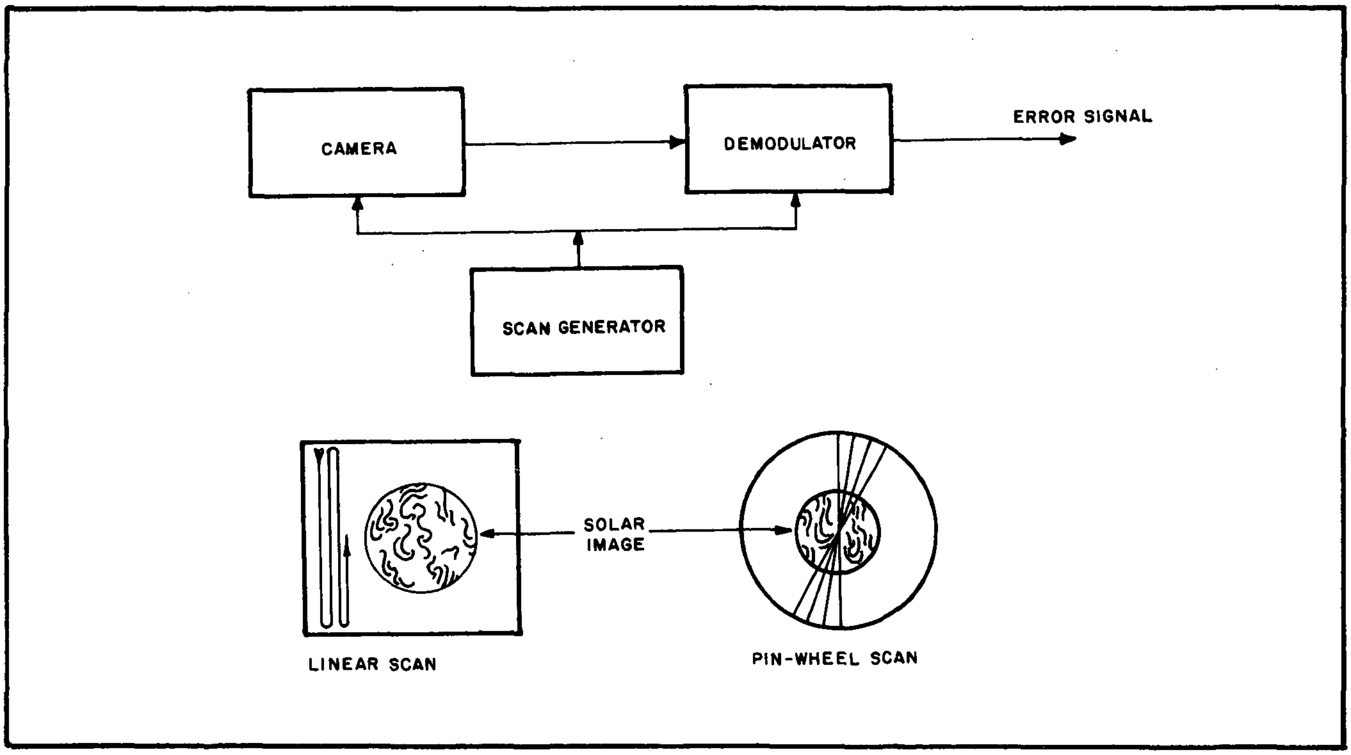


Fig. 6-23 Fine Sensor Rate Signal Concept

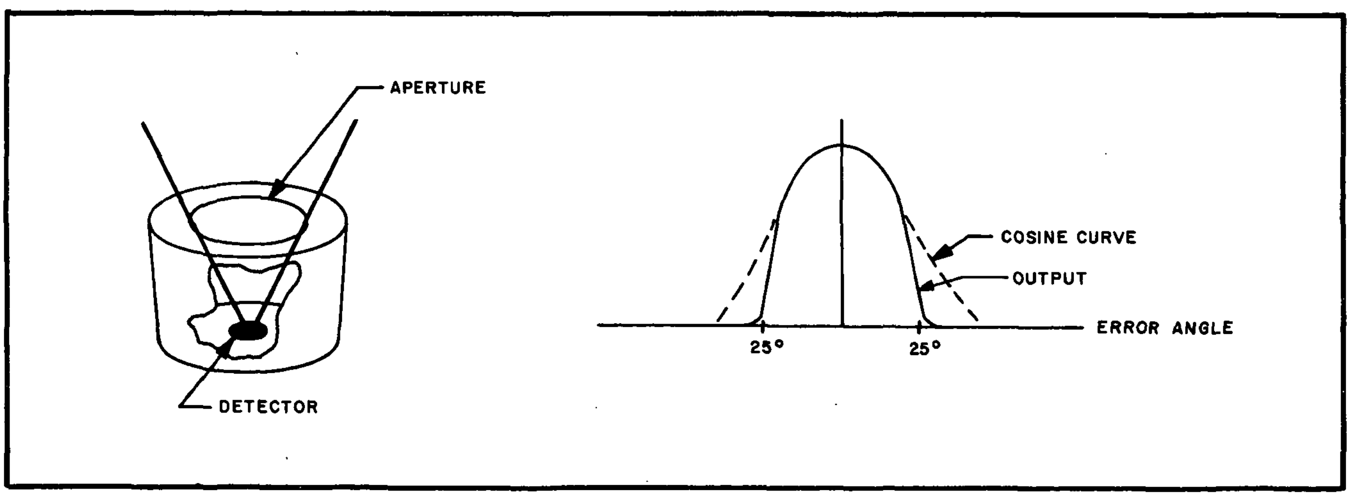


Fig. 6-24 In-Field Sensor Design Concept

6.4.3 Evaluation of Sensor Design Concepts

In this section the design concepts presented in Section 6.4.2 are evaluated and compared. On the basis of this comparison the concepts which best satisfy the design requirements for the coarse and fine sun sensors are selected. Detailed consideration of the selected concepts are given in Section 6.4.4.

6.4.3.1 Coarse Sun Sensors

Illumination-Balancing Concept. This concept is an extremely simple and reliable approach which has been utilized on many rocket and satellite systems. Its chief attributes are summarized below:

(1) No moving parts or fragile components are required in the concept implementation.

(2) Silicon detectors can be utilized. These detectors are stable and reliable. Their suitability for use in space environments has been demonstrated both in the capacity of solar energy convertors and optical detectors.

(3) The associated electronics necessary to produce usable signals is simple, consisting of a low-gain dc amplifier.

(4) Sensors based on this concept have been used in several space programs and suitable sensors are available as standard, space-proven hardware.

The major disadvantage inherent in this concept is its susceptibility to background-induced inaccuracies. However, since the required accuracy for the ATM coarse mode is not stringent, the design can easily satisfy it.

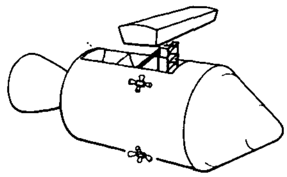
This concept has been selected as the best suited for the ATM because of its design simplicity and proven performance.

Scanning Spot Concept. This concept eliminates the problem of background-induced inaccuracy for most background situations which occur during the mission. However, it introduces an undesirable reliability problem. If mechanical scanning is utilized, moving parts are required. If electrical scanning is incorporated, a fragile component (image dissector) is required. Furthermore, the signal conditioning electronics are more complicated than for the illumination-balancing concept.

Multiple Detector Concept. This concept can also be implemented to negate the background-induced inaccuracy, but the large number of detector elements and complicated electronics associated with this approach are undesirable.

6.4.3.2 Fine Sun Sensors

Illumination-Balancing Concept. The concept considered has been utilized on several space programs and is, therefore, a proven one. It has not, however, been incorporated in a proven design having accuracy requirements as stringent as those imposed



by the ATM. (For that matter, neither has any other sensor design.) The design concept has the same attributes delineated in the coarse sun sensor discussion. However, the stringent short-term accuracy requirement necessitates sophisticated design of both the sensor and the associated electronics. Thermally induced drifts are of particular concern. In fact, in order to satisfy the short-term accuracy requirement, thermal control capable of holding the sensor temperature environment constant to within 5°C over any one minute period will be required. The mechanical stability of the sensor can be held to better than 0.5 arc seconds (in a controlled environment) in a good design reflecting care in choice of materials. The signal-to-noise ratio will be high enough in any practical design to render noise on the output negligible.

While the short-term accuracy requirement necessitates extreme care in implementing the concept, this approach can be mechanized to yield a sensor which will satisfy the ATM requirements. The articulated and mechanical deflection concepts are considered undesirable due to their reduced reliability and mechanical complexity.

This concept is considered to be best due to its high reliability, design simplicity, and proven performance (at least for lower accuracy systems).

<u>Scanning Spot Concepts.</u> These concepts tend to eliminate the problem associated with thermal drift of the detectors inherent in the illumination-balancing concept. Since only one detector is incorporated in these concepts and, since the output of the detector is used only as a gating signal, thermally induced gain changes have little effect on the sensor accuracy. However, the problem is not completely eliminated, but rather transferred from the detector to the deflection system. Drifts and instabilities of both the actual deflection mechanism and the deflection readout contribute directly to sensor inaccuracy. In the case of mechanical scanning, requirements near the state of the art (readout to better than two arc seconds) are placed on the deflection readout system. In the case of the electrical scanning system, thermally induced variations in the deflection coil impedance and mechanical position become problem areas. Furthermore, the mechanical structure deflections with temperature and vibration become serious problems in both cases. In the mechanical scanning technique, play in bearings and/or shaft deflections present difficult design problems. In the electrical scanning technique the problem of mounting the glass image dissector in a metal housing with mechanical stability tolerances of only a few microinches poses a definite challenge.

The electronics necessary to derive suitable signals are complicated and require sophisticated designs to achieve the resolution and stability requirements of ATM. While these approaches offer promise, the complexity of the design implementations and the reduced reliability resulting from the incorporation of moving parts and/or fragile components dictates that their evaluation be lower than for the illumination-balancing concept.

<u>Rate-Sensing Concepts.</u> The rate-sensing concepts must be given a lower evaluation

rating than the scanning systems. These concepts are not particularly suited to the control problem during operation of a coronagraph since this experiment requires position centering on the solar disc and error rates are of secondary importance.

6.4.4 Detailed Examination of the Recommended Design Concepts

In this section, a more detailed description of the recommended design concepts and their problem areas is given. The emphasis is on examining the design concepts and potential problem areas rather than detail design problems associated with the implementation of the concepts.

6.4.4.1 Coarse Sun Sensor

The recommended coarse sensor concept is illustrated in Figs. 6-25 and 6-26. Figure 6-25 illustrates the sensor output characteristic while Fig. 6-26 is a detail illustration of an individual sensor element. The composite output characteristic for a coarse sensor pair is shown in Fig. 6-27. This output characteristic is relatively insensitive to errors about the other control axis; each point is simply multiplied by the cosine of the cross-axis error.

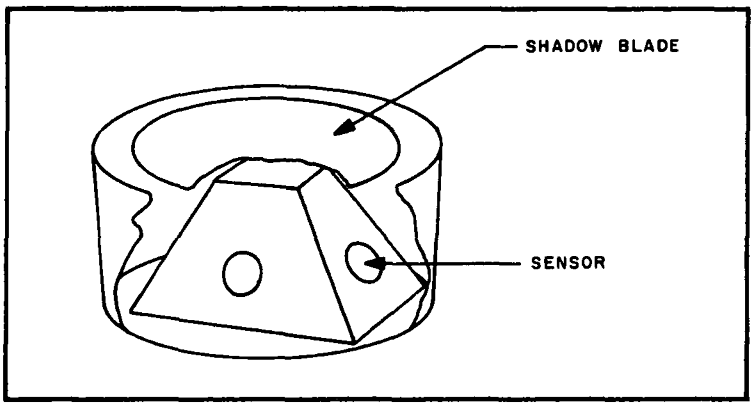


Fig. 6-25 Recommended Coarse Sensor Design Concept

The sensors are operated as current sources and the scale factor is approximately 20 to 30 microamps per degree.

The blades restrict the field of view to the acquisition field in order to minimize the effects of stray light, albedo, reflections off the space craft, etc. However, the location of the sensor must be chosen carefully to minimize errors due to reflections from the spacecraft.

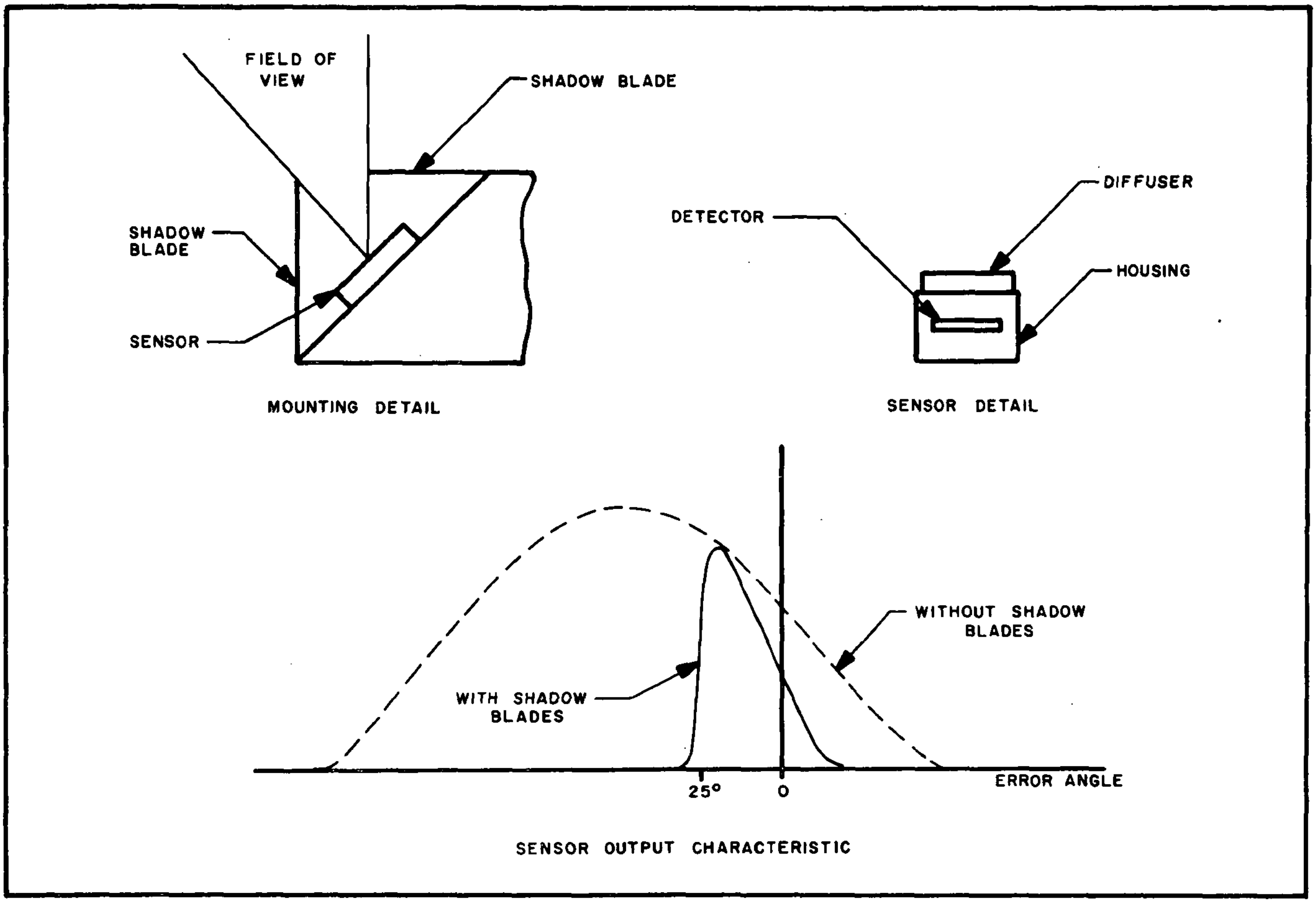


Fig. 6-26 Individual Coarse Sensor Element Detail and Output Characteristics

6.4.4.2 Fine Sun Sensor

The fine sun sensor consists of a pair of single-axis sensors (one sensor for pitch and one for yaw axis control) as illustrated in Fig. 6-19. These sensors are current sources in the electrical configurations illustrated in Fig. 6-28. For designs within reasonable size limitations the scale factor in the linear region of the output signal is in the neighborhood of one microamp per arc second. The potential problem areas and error sources are summarized below:

Thermal Drifts. The largest, and hardest to control, source of error in the fine sun sensor approach is the thermally induced error due to gain changes in the detectors. Since the sensor must not only operate at null, but also at offset angles as large as 20 arc minutes, and since the offset is generated by a fixed bias source, both differential and absolute drifts contribute to the sensor inaccuracy. The source of this thermally induced gain change lies in the shift in the long wavelength cutoff of the detector response with temperature. This shift is illustrated in Figs. 6-29 and 6-30 for an unfiltered detector. The thermal gain change contributes an error of approximately 0.5 arc sec/$^{\circ}$C change in temperature for an unfiltered detector. If a narrow-band

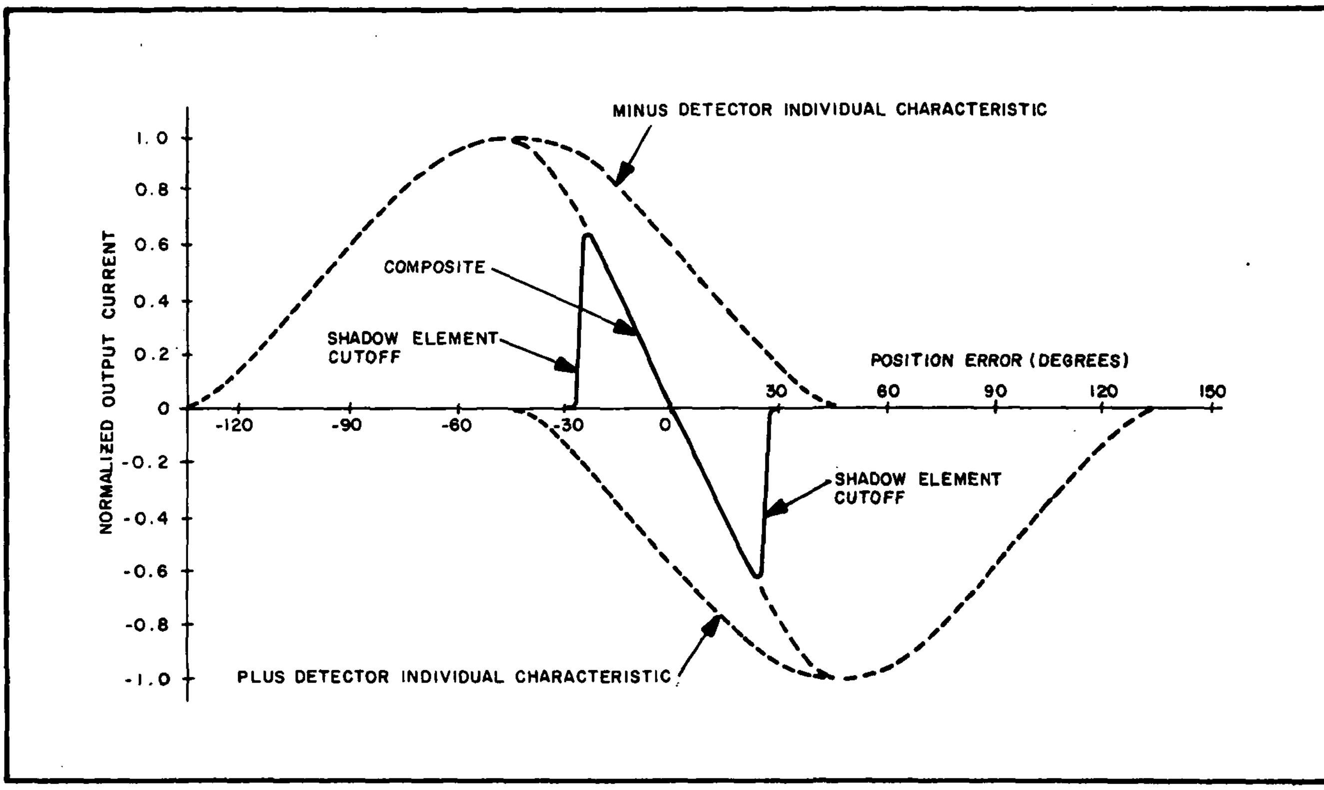


Fig. 6-27 Composite Output Characteristics for 4 Coarse Sensor Pair

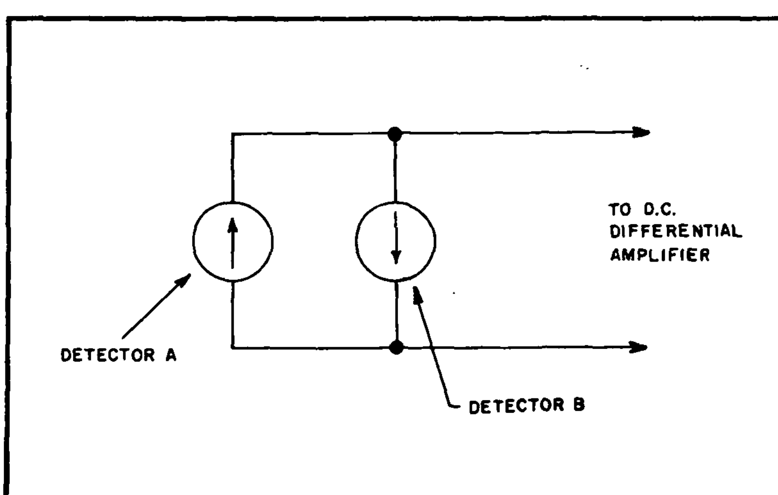


Fig. 6-28 Fine Sensor Electrical Configuration

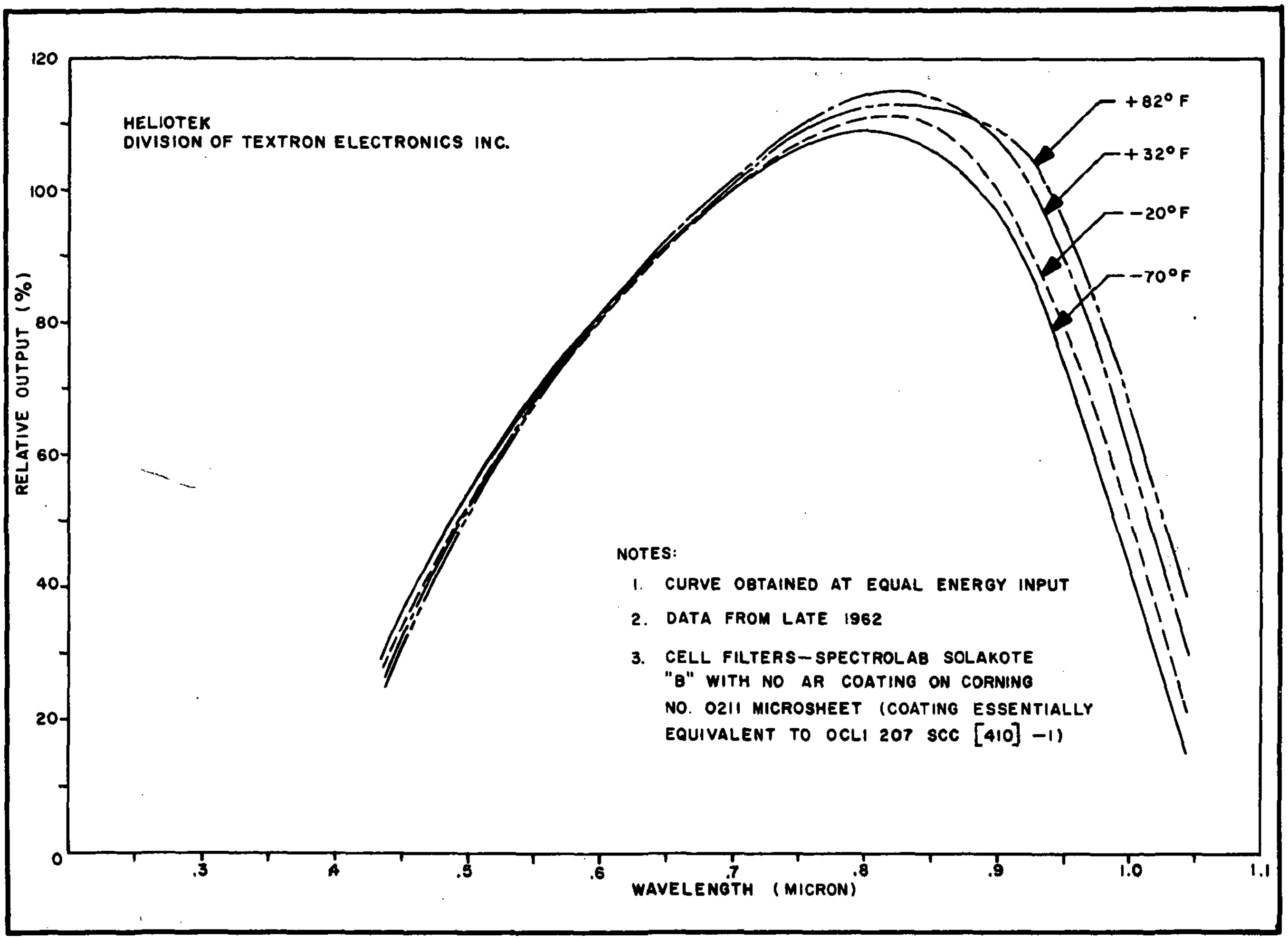


Fig. 6-29 Spectral Response vs Temperature for Filtered P/N Solar Lens

spectral filter is incorporated to limit the detector response as shown in Fig. 6-31, the error can be reduced by about an order of magnitude. This thermal gain change is emphasized by the change in heat input from the sun as the solar image moves across the detector array. With careful design, the thermally induced errors can be held to less than 1.5 arc seconds.

Mechanical Shifts. Since the thermal environment of the sensor is controlled to within ±5°C, thermally induced mechanical deflections and shifts will be less than 0.5 arc seconds in a careful design.

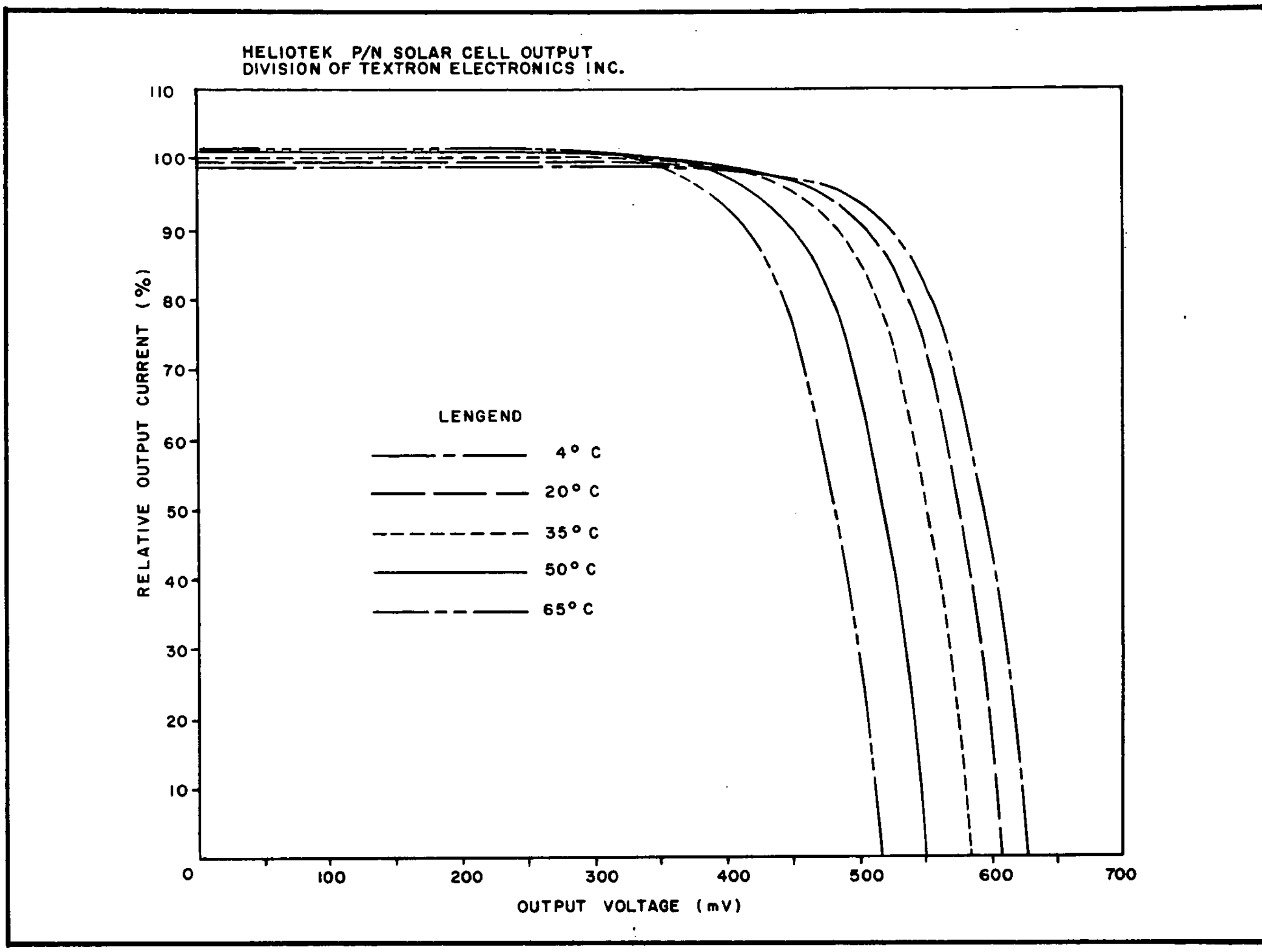


Fig. 6-30 Filtered P/N Cell Output

Errors Induced By Amplifier Input Offset. These errors can be held to less than 0.1 arc seconds by holding the offsets to less than five millivolts.

Roll-Induced Errors. These errors are not included directly in the sensor error budget since they are not controllable by the design of the sensor. The discussion below shows how roll errors are coupled with pitch and yaw, and demonstrates why the roll rate limitation is primarily set by the pitch/yaw error requirements.

Figure 6-32 illustrates schematically the situation that occurs when there exists a maximum offset (20 arc minutes) in yaw and no offset in pitch. In Fig. 6-32 (b) a roll error (grossly exaggerated) is shown. Note that the energy balance in both pitch and yaw has been altered. The PCS causes the sensors to return to the initial position (in terms of energy balance) and the situation is then as shown in Fig. 6-32 (c).

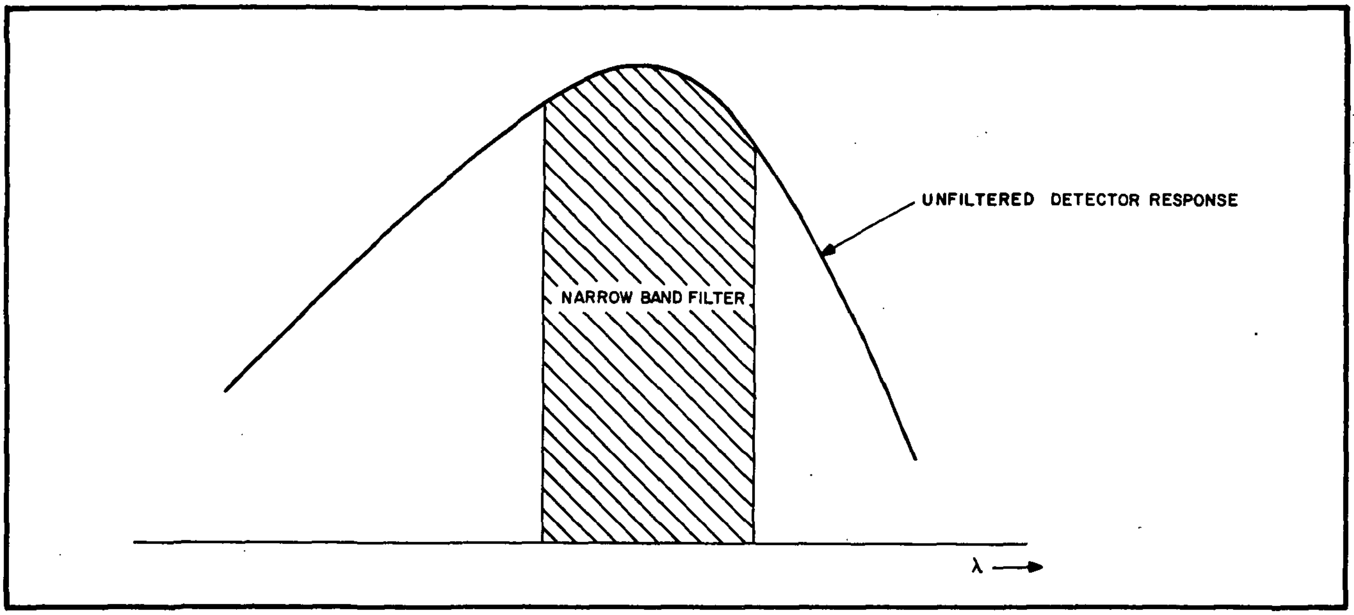


Fig. 6-31 Narrow Band Spectral Filter Response

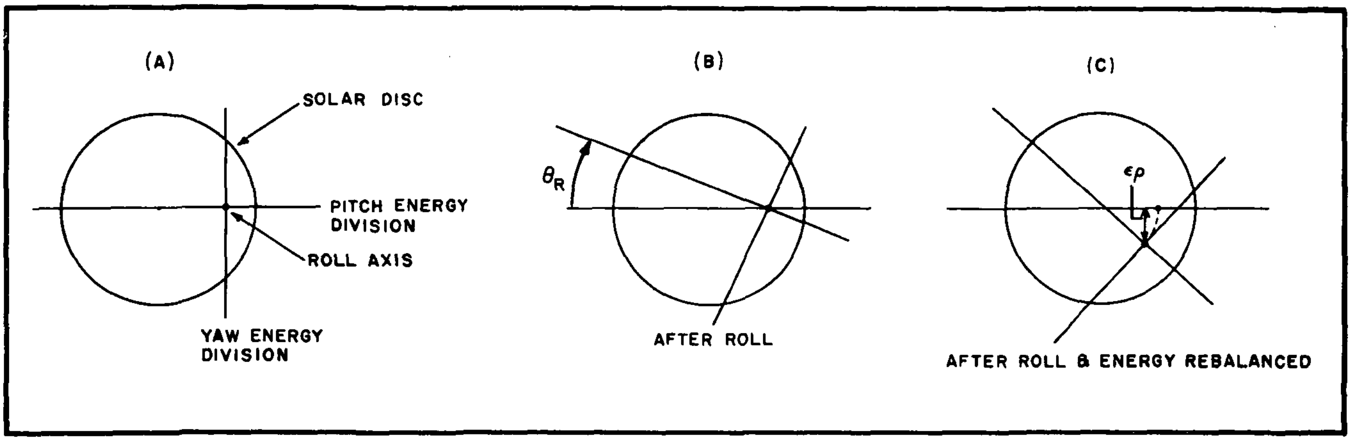


Fig. 6-32 Fine Sensor Roll and Energy Rebalance

Thus, the error in pitch due to roll is seen to be:

$$\varepsilon_p = 20.60 \sin \theta_R \text{ (arc secs)} \qquad (6.18)$$

Since θ_R is small, $\sin \theta_R \simeq \theta_R$ and

$$\varepsilon_p = 1200\ \theta_R \quad (\theta_R \text{ in radians}) \qquad (6\text{-}19)$$

If ε_p is to be maintained at one arc sec for one minute, then

$$\frac{\Delta \theta_R}{\Delta t} = \frac{1}{(1200)(60)} \text{ rad/sec} = 2.85 \text{ arc sec/sec}$$

Error Summary. Fine sun sensor errors are summarized as follows:

Thermal error	1.5 arc sec
Mechanical stability	0.5
Amplifier offset	0.1
Total peak	2.1 arc sec
Total RSS.	1.58 arc sec

6.5 ROLL CONTROL

6.5.1 General

The roll axis servo uses a floated, rate-integrating gyro for position reference and a dc torque motor for gimbal drive. Necessary electronics, stabilizing networks, and mode switching are provided. In the automatic mode, the servo maintains the roll attitude to the inertial attitude of the gyro when it is uncaged. Roll slew (six deg per sec) is accomplished by torquing the caged gyro. A roll gimbal angle transducer provides the control unit with a signal that drives an indicator displaying the roll gimbal angle.

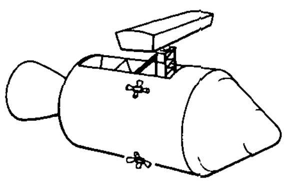
6.5.2 Performance Requirements

The maximum allowable roll rate is two arc sec per sec. This requirement is imposed by pitch-yaw cross coupling in the fine sun sensors when the spar is offset. (See Section 6.4) A slewing capability of six deg per sec is a PCS requirement.

6.5.3 Roll Sensor Selection

The types of roll sensors considered are magnetometers, rate gyros, and rate integrating gyros. The magnetometer is very sensitive to local magnetic field from both the CSM and ATM, and was rejected for this reason. It also has other drawbacks such as variable gain depending on the relationship of the ATM roll axis and the earth's magnetic field vector, and the variation in field strength with altitude. A rate gyro has to have a rate threshold substantially below the two arc sec per sec ($\approx$ 0.0006 deg per sec) requirement, which is beyond the state of the art for rate gyro.

A position control has a net rate set by the drift rate of the integrating gyro and by jitter rates made negligible by proper servo design. A medium-grade, floated, rate-integrating gyro has a drift rate of about one degree per hour (one arc sec per sec) which satisfies the requirements. Thus, the rate integrating gyro was selected for the ATM roll control sensor.



6.5.4 Principal Disturbance Inputs

The principal disturbance input tending to cause roll displacement is the friction/drag torque coupled to the roll gimbal through bearings, motor brushes, and flexible cables by the CSM drift rate component with respect to the ATM roll axis. Without roll control, drift at 0.05 deg/sec would cause a roll angle change of 3.0 deg in a one minute time period, and 1.2 deg at 0.02 deg/sec rate. Both of these considerably exceed the permissible displacement.

Other disturbances that can cause measurable roll angle change in one minute of time are servo sensor drift, electronics drift and cable drag change with gimbal travel, but the roll rates caused by these inputs are small and the resulting displacements not significant.

6.5.5 Roll Servo Design Concept

Design Approach. The preliminary roll control design concept employs a rate integrating gyro as a sensing element in a one-axis, stable-platform, position-controlled servo. Total drift requirements can be achieved with a random gyro drift rate of about 1.0 deg/hr (one arc sec/sec). The torque sensitivity requirements are based on a displacement allowance of ±0.015 degrees, occurring with CSM-roll gimbal relative drift rate reversals.

Torque Sensitivity Computation. For a static friction/drag torque on the roll gimbal of ±0.22 lb-ft, the torque sensitivity K_{ts} required to maintain gimbal displacement to ±0.015, for a CSM-roll gimbal relative rotation reversal, is given by:

$$K_{ts} = \frac{0.44 \text{ lb-ft}}{0.015} = 29.4 \text{ lb-ft/deg}$$

$$K_{ts} = 1680 \text{ lb-ft/rad}$$

Mechanization of Roll Servo. The following elements are cascaded to make up the roll servo loop:

- Rate-integrating gyro
- AC amplifier - demodulator
- RC compensation network
- Voltage-power amplifier
- Torque motor
- Inertia/structure

The product of the frequency invariant gain factors associated with the transfer function of each of the above elements is the loop gain, K_a, and is given by:

$$K_a = K_{ts} \times \frac{1}{\text{Roll Inertia}} = 1680 \times \frac{1}{40} \text{ rad/sec}^2$$

$$K_a = 42 \text{ rad/sec}^2$$

Sizing the Roll Torque Motor. There are two functional requirements on maximum roll motor torque: (1) that the saturation level be about 10 times the 0.22 lb-ft frictional torque, and (2) that it accelerate the 40 slug-ft^2 roll inertia to a gyro-controlled slewing rate of six deg/sec in a few seconds of time. A two lb-ft accelerating torque will cause the roll gimbal to reach six deg/sec in a time given by:

$$t = \frac{J}{T} \omega = \frac{40}{2} \times \frac{6}{57.3} = 2.2 \text{ sec}$$

An available torque motor considered for roll control develops 2.7 lb-ft torque with 22.9 volts applied. Maximum power amplifier drive will apply about 90 percent of this level to the motor, and the developed torque will be 2.4 lb-ft.

Selection of Rate-Integrating Gyro. The type of gyro considered for the roll servo in this preliminary design is a miniature, floated, rate-integrating gyro with a random drift rate of 1.0 arc sec/sec in a zero gravity environment. Its characteristic time constant of 0.005 sec is negligible in the loop dynamics. Gyro gain is 0.75 V/deg = 43 V/radian. Torquer scale factor is 10^4 deg/hr/ma. This level of performance can be met by several gyro manufacturers.

Discussion of Transfer Functions. (See Fig. 6-33.)

(1) Integrating gyro: $\frac{V_o}{\theta_r}(s) = \frac{43}{s}$ (V_{AC}/rad)

The gyro operates as an integrator of inertial roll gimbal rate, θ_r, and provides an AC output voltage proportional to the roll gimbal "error" angle, referenced to the gyro spin vector at the moment the gyro gimbal is uncaged. (This occurs the instant that roll control is switched from manual slewing mode to automatic control.)

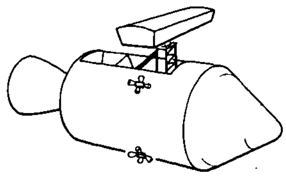
The gyro characteristic time constant of 0.005 sec can be ignored in the loop analysis because it means that the gyro will integrate the roll rate accurately for sinusoidal variations up to at least 100 rad/sec (16 cps), a frequency well above the seven rad/sec servo bandpass.

(2) AC amplifier - demodulator: $\frac{1000}{\frac{s}{50} + 1}$ V_{DC}/V_{AC}

This circuit amplifies the gyro pick-off AC voltage signal and demodulates it in a full-wave phase detector, referenced to the gyro excitation. The lag filter smooths the output ripple and noise, and has a corner frequency of 8.0 cps (50 rad/sec).

(3) Compensation network: $\frac{0.085 \left(\frac{s}{5} + 1\right)}{\frac{s}{50} + 1}$ (V/V)

This response is given by a conventional lead-lag network with an α of 10. The static gain factor is less than the theoretical value of 0.10 because of circuit loading effects. The reduction to 0.085 V/V assumes "light" circuit loading.



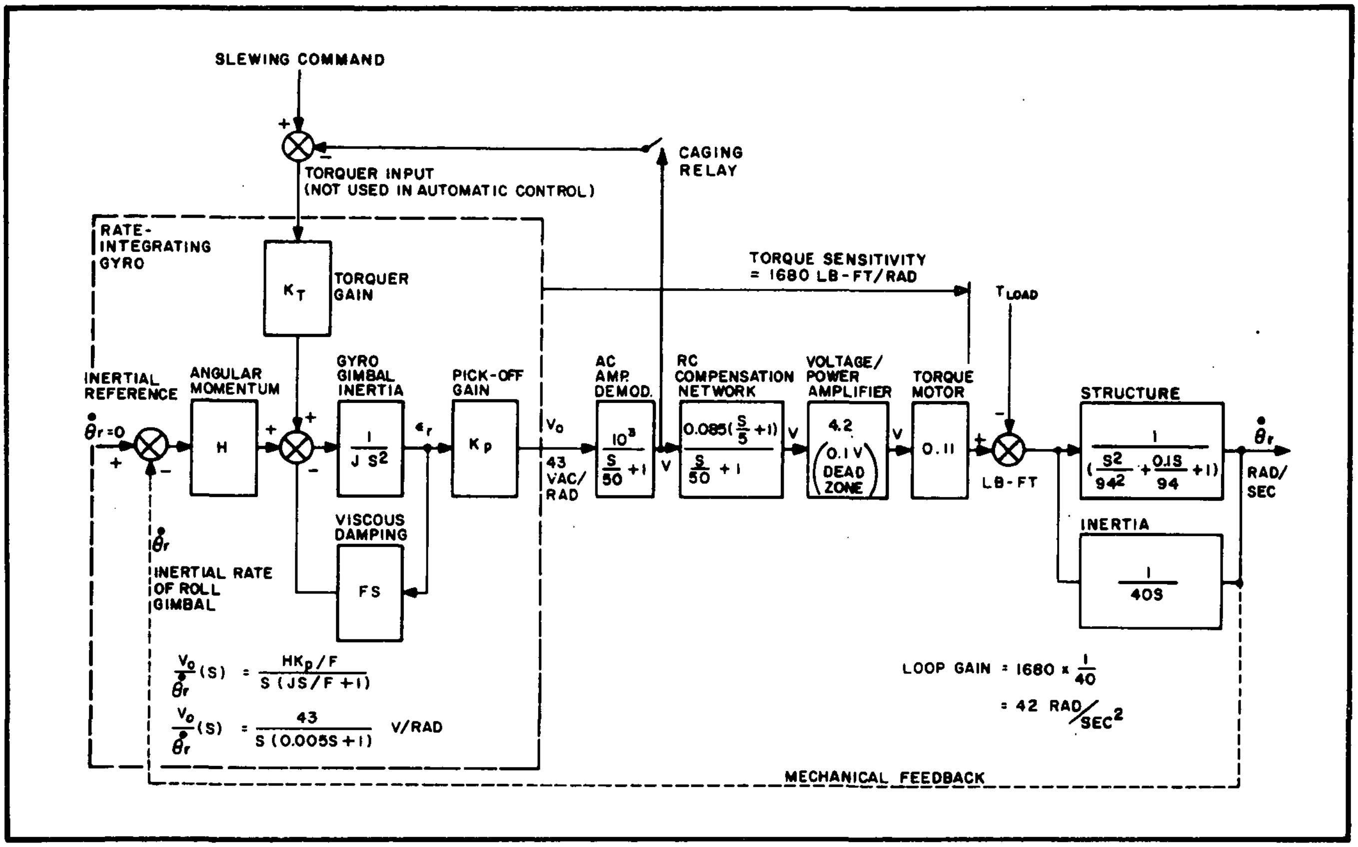


Fig. 6-33 Roll Control Servo Block Diagram

(4) Voltage - power amplifier: 4.2 V/V; dead zone, 0.1 V

The network output must drive into an impedance-matching, voltage amplifier, which then drives the power amplifier. Identical to the yaw-pitch amplifier design, the roll power amplifier will be a pulse width modulator-power switch circuit with modulation frequency of about 5000 cps. Assuming the gain factor 4.2 V/V is in the driver voltage amplifier, and that the power amplifier gain is unity, a 0.1 V power amplifier input dead zone will be equivalent to a roll angle of about 2.7 arc sec. This dead zone should cause no problems, however, because the low CSM inertial drift rate will couple frictional drag torque to the roll gimbal, and the servo will drive to maintain its inertial position. Power amplifier output impedance is 0.5 ohms; motor armature resistance is 6.1 ohms. For ±22 volts maximum open-circuit voltage, the motor armature maximum current and voltage will be ±3.3 amps and ±20.4 V, respectively. Quiescent motor current and power to offset 0.2 lb-ft friction will be 0.28 amp and 0.47 watts, respectively. A smoothing filter at the power amplifier output, with an effective time constant of $\leq$ 0.005 sec, is assumed to have negligible effect on servo dynamics.

Torque Motor - Inertia/Structure Transfer Function. In the block diagram, a simplified transfer of the torque motor and load is shown. The motor back EMF input is omitted because its maximum amplitude for rates of 0.05 deg/sec will be less than one mv. The roll structural resonance is assumed to be at 15 Hz and similar in nature to the pitch/yaw resonance. Open-loop dynamic gain is down 34 db at this frequency, which is considered adequate. (See Bode plot, Fig. 6-34.)

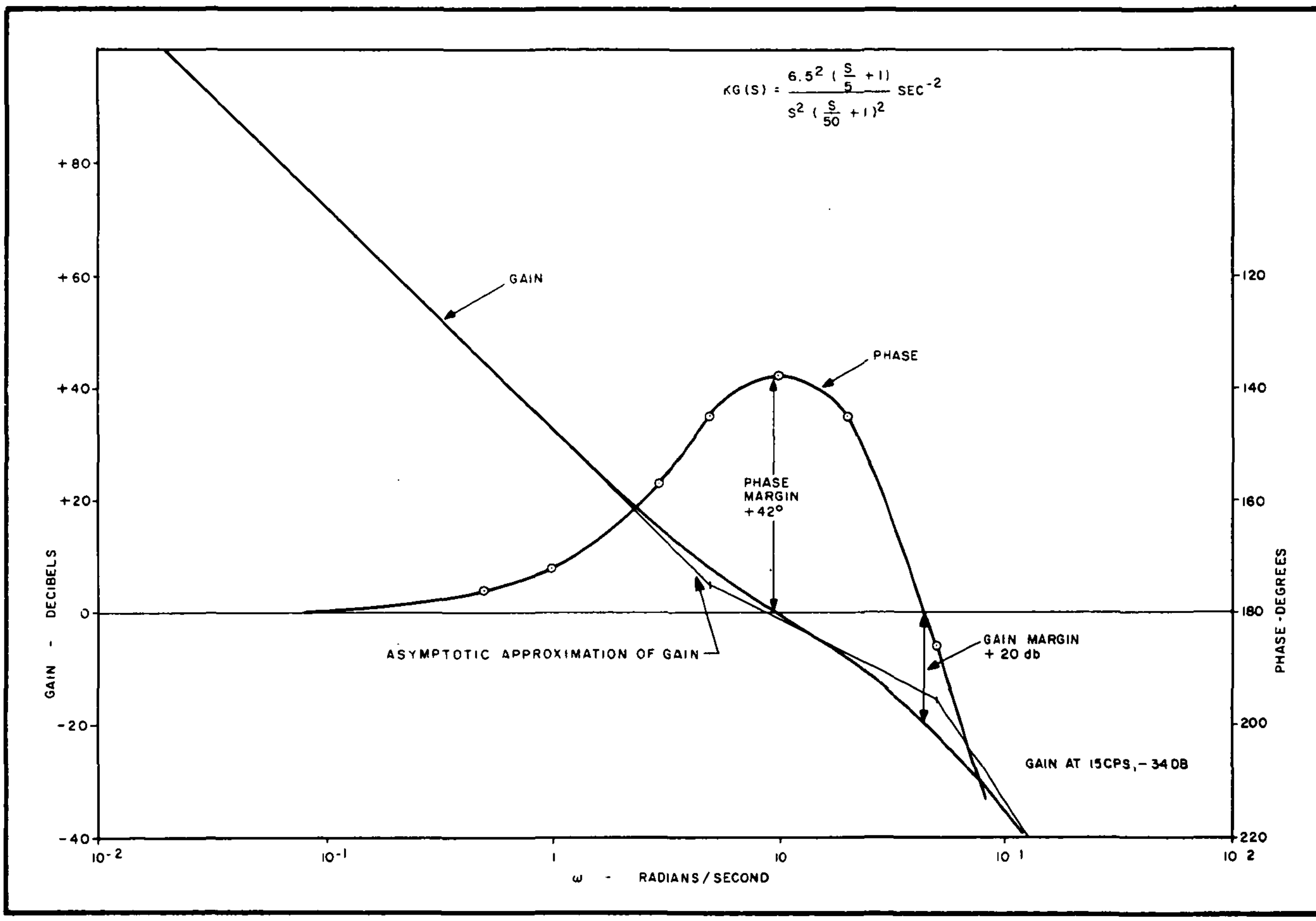


Fig. 6-34 Roll Control Bode Plot

Stability Analysis. The cascaded transfer functions shown in Fig. 6-33 are plotted in the Bode plot, Fig. 6-34. Phase and gain margins of +42 degrees and +20 db, respectively, indicate good loop stability. If the gyro characteristic lag at a frequency of $\frac{1}{5.0 \times 10^{-3}}$ = 200 rad/sec were plotted, the phase and gain margins would be about +40 degrees and +16 db, respectively. This points out that any roll servo dynamic lags effective above 30 Hz (188 rad/sec) can be ignored.

Roll Servo Performance Discussion. As implied in the discussion on roll servo disturbance inputs, the CSM drift rate is the principal factor that makes a roll control servo necessary. The selected design is a position-controlled servo employing an integrating gyro to provide an inertial roll position reference, and having a static torque sensitivity of 1680 lb-ft/rad.

6.6 ADDITIONAL FUNCTIONS

In this section certain features and readout functions associated with the pointing control subsystem are discussed. The principle ground rule observed is that the required feature (indicator or function) be incorporated in a manner which does not compromise any of the operational requirements of the subsystem.

6.6.1 Solar Angle Indication

Since there are coarse sun sensors used for the coarse pointing mode, and their field of view is unobstructed by the heat shield, there is sun angle information available in the concept that could be useful to the astronaut during coarse acquisition. Therefore, it seems worthwhile to make this information available by giving the operator the capability of displaying it by command on the program indicators of the control unit. However, the circuitry providing this indication must be isolated so that any malfunction of circuitry or indicator cannot appreciably degrade the coarse control function. (See Fig. 6-10.)

A coarse target detector drives a telltale indicator on the control unit so that the operator can tell when the sun is in the field of view of the coarse sun sensors, even though their outputs are not driving the programmed indicators.

6.6.2 Gimbal Angle Indication

It is required to provide to the control unit the pitch, roll, and yaw gimbal angle information. The accuracy of these readouts is determined by the operational requirements. The basic function, of course, is to display the relative attitude of the ATM to the CSM. However, a restow capability is a growth requirement, and for this operation the ATM must be positioned within 0.5 deg of zero degrees in each axis. The transducer must be aligned to zero degrees within 0.25 deg and its resolution must be at least 0.1 deg. These latter specifications are imposed on the gimbal angle transducers, but for the present concept the indicators on the control unit do not need to have this resolution.

Both digital and analog readouts were considered. A digital readout for the yaw axis total range (340 deg) requires a 13-bit encoder to obtain the necessary resolution. The pitch axis requires 10-bit information, and the roll axis needs nine bits. Parallel readout is not practical because of the large number of wires required. Serial readout does not have this drawback, but this type of readout is unnecessarily complex. However, the deciding consideration against any digital concept is that the rate signal for the slewing rate limit is derived from differentiating the output of the gimbal angle transducer. (See Section 6.3.) For this, a very smooth linear position signal is necessary, and since it is generated for the servos there is no reason not to use it to drive the gimbal angle transducers. The resolution required by the servos is much better than that needed by the astronaut to visually detect the gimbal rate when estimating the time remaining before a stop is reached.

An analog device such as a synchro or resolver does not have an acceptable characteristic for obtaining a rate signal for the servo, and furthermore requires a relatively large number of wires. Thus, the best device for this application is a center-tapped pancake type potentiometer since it satisfies all the aforementioned requirements with a reasonable number of wires. It does need a regulated excitation and the sliding contact must be lubricated for the vacuum environment.

6.6.3 Housekeeping Data

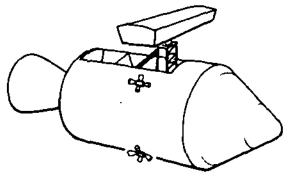
The significant PCS data points are made available to the data subsystem for recording in order that the post flight subsystem performance evaluation can be made. These same data will be necessary for ground checkout. Section 9 includes a discussion of the PCS data points and the sampling rates.

6.6.4 Immobilization of Oriented Section

Three methods of immobilizing the oriented section were considered: closed-loop position control, pins and detents, and friction brakes. Closed-loop position control is a position servo with the loop closed through the gimbal angle transducer. Any disturbance torque tending to move the gimbal is resisted by the torque motor according to the stiffness of the servo. With an angle command input, the oriented section can be held at any desired attitude relative to the CSM. The major drawback to this scheme is that the PCS must remain energized. Therefore, the concept was rejected.

The idea of using solenoid-actuated pins pushed into detents was studied. Only one, or at best a few, gimbal hold positions would be available on each gimbal, and it was decided that more flexibility was required to satisfy the ATM thermal control requirements.

The friction brakes satisfy all of the requirements and have been selected. For power conservation reasons they must be a bistable mechanism so that power is only required to change from "set" to "release", or vice versa.



6.7 DISCUSSION OF INTERFACE ASSUMPTIONS

A number of interface assumptions have been made in order that the PCS conceptual design could proceed. However, if any of these requirements cannot be met, the following paragraphs discuss the consequences to the PCS performance and some possible solutions.

6.7.1 Balance of Oriented Section

It has been shown in Section 6.2.3 that the oriented section must be balanced so that the center of mass is very close to the intersection of the gimbal axes if the magnitudes of the pointing transients due to "moderate" man motion are to be held to tolerable levels. The center of mass-gimbal intersection offset is assumed to be 0.01 in., and with a 1000-pound oriented section in a 1-g field, the unbalance torque is 0.83 lb-ft. This is 3.8 times the assumed gimbal friction and there is no reason why the oriented section cannot be statically balanced to within its gimbal friction. Anisoelastic effects and center of mass shift within the experiments due to film transport and aperture cover movement will tend to degrade the static balance. Good design can minimize these disturbances, and those that remain will result in man motion pointing transients.

6.7.2 Cable Torques

One alternative approach in the electrical distribution subsystem concept (Section 10) is the use of wire wrap mechanisms for routing signals and power across the gimbals. The torque produced by this friction is assumed to be 0.087 lb-ft. It is also assumed that the wires slip and grab as the gimbals rotate such that the total torque variation due to the wires in any angular increment can vary from zero to 0.087 lb-ft. If this assumption proves to be optimistic, it is possible that the torque sensitivity of the PCS can be increased to compensate for the increased torque. This would cause greater quiescent power drain as well as reduce the gain margins. If necessary, slip rings can be used and the present PCS would perform satisfactorily.

6.7.3 Structure

It has been assumed that the spar structure has a first bending mode frequency of 10 Hz or greater. If a more detailed design of the structure shows that this frequency is less than 10 Hz, a form of integral control such as that discussed in Section 6.2.1 will probably be necessary. Unfortunately, this will aggravate the magnitude of the man motion-induced pointing transients. Other schemes, such as attaching sensors to appropriate places on the spar, should be investigated.

6.7.4 Offset Commands

It has been assumed in the design that only discrete commands are available to control the offset and boresight operation. As discussed in Section 6.2.1, a proportional control of the offset rate would be desirable if the command subsystem can mechanize it.

6.7.5 Restow

The present ATM concept does not include a capability to restow the oriented section. The PCS gimbal angle transducers have been selected so that they have sufficient resolution to permit closed-loop position control if the restow capability is added later. If the position servo loops are added, the oriented section immobilization trade-off should be reopened because it might be that the "pin and detent" approach would be more attractive than the friction brakes.

SECTION 7

SOLAR MONITOR SUBSYSTEM

Section 7
SOLAR MONITORING SUBSYSTEM

7.1 FUNCTIONAL REQUIREMENTS

The need for a solar monitor is a direct result of the basic ATM premise which establishes the astronaut as observer-in-charge. Seated in the CM, the observer is physically isolated from the ATM and its experiment complement. Therefore, a remote sensing system is needed to enable him to monitor the sun's image in the same manner as if he were actually able to look along the axis of the experiments.

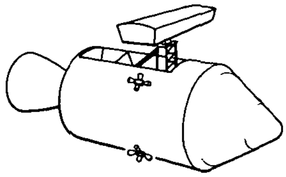
The image of the solar disc is to be displayed in the CM so that the astronaut can look steadily at the display while he manipulates the "fly to" control stick to offset the oriented section. The solar monitor is used primarily in the patrol mode of observation, with the ATM pointing at the center of the sun and the astronaut viewing the solar activity over the entire disc and out beyond the limb. Should an event of particular interest be observed on the monitor, the astronaut will command the pointing control to slew off center to bring the important activity into the center of the field. Once the coarse offset is aligned, the astronaut can switch the video input to his display from the main solar monitor to a specialized, narrow-field sensor within an experiment, which has been accurately boresighted with the optical axis of that experiment. The spectral response, field of view, and spatial resolution of each experiment's boresight sensor is specified by the individual experimenter to ensure that the experiment axis can be aligned with the salient feature on the disc with the required precision. A reticle pattern on the flux-sensitive image plane of each boresight sensor appears superimposed on the displayed image. This provides the observer with a positive geometric reference for use in pointing the experiment at the desired point.

7.2 DESIGN CONSIDERATIONS

The functional requirements outlined above for the solar monitor subsystem provide the basis for the following subsystem definition and design approach.

A sharp image of the sun is formed by a telescope mounted on the spar and aligned with the pointing control sun sensors. The solar image at the focal plane falls on the photosensitive surface of the camera tube and is converted to an electronic signal which is transmitted to the remotely situated display. There, it is reconstituted into a visual image for observation by the astronaut. (See Fig. 7-1.) The image transfer must occur in real time, or nearly so, in order to avoid introducing too much lag in this man-optical link of the pointing control subsystem. Provision can be made for photographing the displayed image should this be judged desirable.

A dominant factor involved in defining a solar monitor is the spectral response characteristic desired. This is specified by the experimenters concerned and is determined by the type of activity and wavelength region under study. Since the ATM



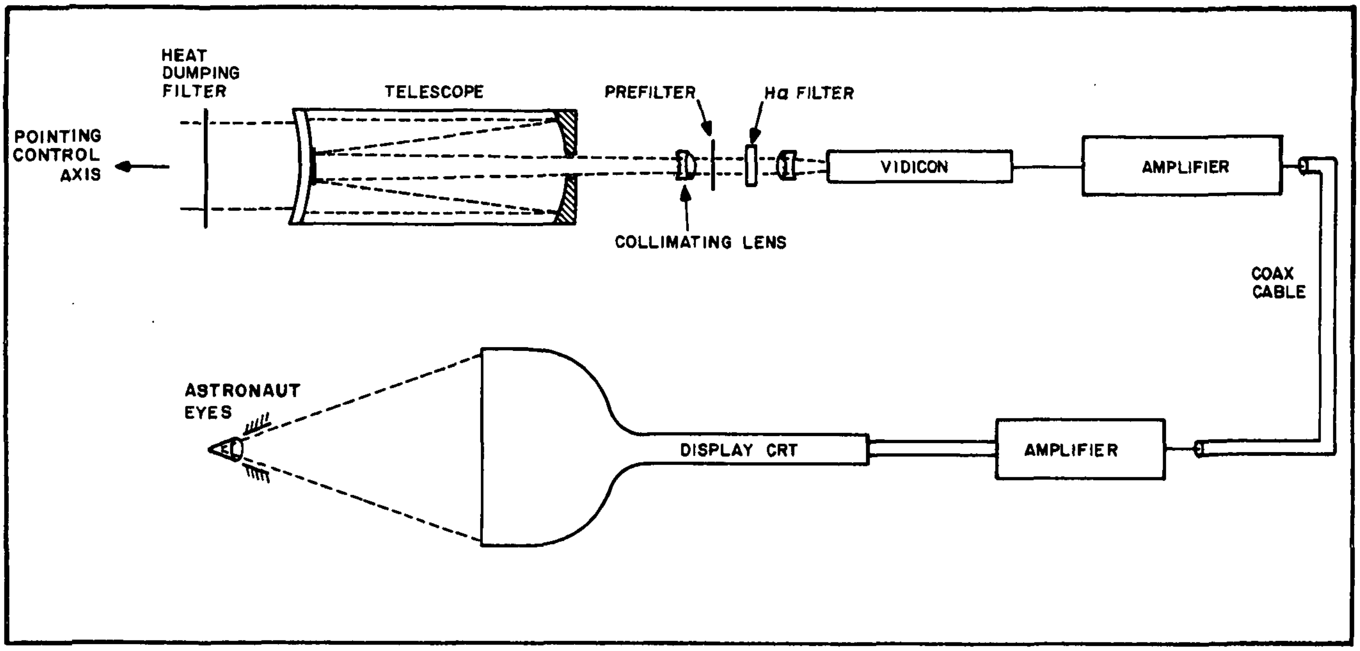


Fig. 7-1 Solar Monitor Subsystem

carries several independent experiments, it is not possible to select a unique spectral characteristic for the monitor which is optimum for all requirements.

For the purpose of this study,the solar monitor subsystem will view the sun in monochromatic light. The study experimenters have indicated that chromospheric activity will be most usefully monitored at the wavelength of the Balmer alpha line of hydrogen, $\lambda = 6563\text{Å}$; however, many of the ATM experiments will be designed to measure nonvisible radiation such as solar X-rays and various emissions in the ultraviolet portion of the spectrum, and may require other monitor wavelengths. Chromospheric activity such as flares, plages, prominences and sunspots, however, are readily observed by their Hα radiation, and these phenomena are probably the best visual manifestation of the centers of solar activity that are of interest in nonvisible portions of the spectrum.

The first sign of a new center of activity is the appearance of a magnetic field, but the first visible indications soon follow in the form of plages and faculae. Since plages occur in the chromosphere, their presence is clearly evident as bright areas in Hα light. The plages continue to brighten and increase in area, and the magnetic field also becomes more intense. After a few days, flares and prominences may commence and in a few more days all activity reaches a maximum. Then the activity dies down, with the plage being the last visible manifestation to disappear.* The individual experimenter will define the activity regions to be studied in terms

*Solar Flares, H.J. Smith and E.P. Smith, 1963, page 12.

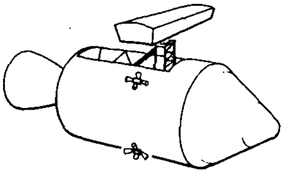
of the major visual telltales seen in a particular monochromatic light. Most flares cannot be observed in other than monochromatic light, and must be viewed through a system of rather high spectral purity.

The chromospheric fine structure in the neighborhood of an active region is very complex. Monochromatic Hα filtergrams reveal a larger amount of detail not visible in white light. (See Figs. 2-11 and 2-12.) It is essential that the scientists be able to train the astronauts in the techniques of solar observation to an extent which will enable them to recognize the various types and forms of features, and to be aware of their relative importance. This provides a powerful motivation for the selection of Hα, since this wavelength is the one most commonly used at ground-based observatories and flare-patrol stations across the country.

The major elements of the solar monitor are sketched in Fig. 7-1. Heat dumping filters ahead of the instrument aperture reflect a large portion of the thermal energy before it reaches the remainder of the optical system. The imaging telescope has a three or four-inch aperture and about a 50-inch focal length. It is diffraction limited at about two arc seconds. The temperature-controlled monochromatic filter is most likely centered at λ = 3934Å for Hα, but the design of the system is sufficiently flexible that a filter centered at some other wavelength such as λ = 3934Å for the calcium K line can be used if desired. An additional heat and light dumping, narrow-band prefilter is placed immediately ahead of the monochromatic filter. The latter has a bandwidth of one angstrom or less in order to achieve the necessary spectral purity (5000 to 10,000) for the detection of solar flares. This is discussed further in Section 7.3.1.3.

The monochromatic image of the sun is focused on the face plate of the photosensor. A scanning type of photosensor is required to transmit the detail over the disc. Inasmuch as the entire disc is to be viewed in the patrol mode, the solar monitor must have a field of view of at least 32 arc minutes. In order to include the sky near the limb, a field of about 40 arc minutes diameter is recommended. In this way, activity which may extend beyond the limb (prominences and flares) will be seen, and centering of the image in the field of the display is not unduly critical.

There is ample radiant flux from the chromosphere in a one angstrom band at 6563Å to enable high resolution to be obtained with a standard, space-qualified vidicon camera tube. The use of image orthicons or image intensifiers is therefore not required. The sensor is to operate in real time with a two-to-one interlace and at a rate in the neighborhood of 10 to 15 frames per second in order to conserve bandwidth without introducing enough "flicker" to degrade the ocular response of the astronaut. A resolution, at 15 percent modulation transfer, of 800 TV lines or 400 line pairs is required. This figure applies in any direction and at any point in the field of view from the center of the sun out to the limb. Four hundred line-pairs across the 40 arc minute field gives one line pair for each six arc second element and represents a spatial frequency of 30 line-pairs/mm on the vidicon faceplate. A 30 arc second detail then represents six line-pairs/mm and nearly



100 percent contrast is transferred at this and all lower spatial frequencies. Preservation of this resolution through the transmission line and on to the cathode ray tube display (CRT) is a matter of careful design.

A high resolution CRT is used for the display, with a phosphor having spectral and time-decay characteristics which are optimum for visual monitoring of the image. Magnetic deflection will be required and the display will be linear with very little distortion of a nature that would cause eye fatigue, or introduce uncertainty in interpretation of complex solar features. As a result of this design, the eye of the observer is the limiting factor in detecting small features of low contrast with the background.

The only manual adjustments to the CRT required are those affecting intensity, contrast, and spot size. This enables the astronaut crew to adjust the composite vidicon-kinescope "gamma" characteristic for optimum visual interpretation under the background lighting conditions which prevail at the time. Such adjustments may be required to examine various features of the solar disc, and particularly when switching to the narrow-field, boresight sensors on individual experiments.

The CRT face is a possible hazard since it is exposed on the display panel. A cover will be provided to enclose the CRT face when not in use. Also, the accelerating potential on the CRT will be kept below the level where X-ray generation can become significant. The cover is primarily to be used during the ascent stages when there is a chance of an abort and landing, as well as during the reentry phase if the ATM display unit is not jettisoned.

7.3 DETAILED SUBSYSTEM DESCRIPTION

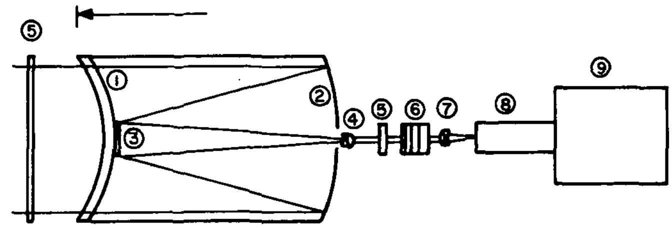
7.3.1 Optical System

The function of the optical system is to project a well focused, monochromatic image of the sun on the photosensitive target of the scanning camera tube. A telescope is required to collect the incident flux and the focal length must be chosen so that a 40 arc minute field appears in the image plane as approximately a one-half inch circle. The effective aperture must be sufficiently large that the camera tube can produce a high resolution signal with the monochromatic radiant flux available. The aperture must also be sufficient to place the diffraction limit of the telescope well below the aperture response cutoff of the vidicon tube. The resolution of the telescope must not be marginal if satisfactory performance is to be maintained in the presence of environmental vibration extremes, thermal gradients and radiation fields. At the same time, size and weight must be kept low.

7.3.1.1 Telescope

The telescope design is of the Bouwers catadioptric type (Fig. 7-2) operating in the Cassegrain configuration. This design presents a compact instrument of light weight

OPTICAL ELEMENTS



1. MENISCUS LENS (SHELL OR CONCENTRIC LENS)
2. PRIMARY MIRROR (SPHERICAL OR RETOUCHED SPHERICAL)
3. SECONDARY MIRROR (INSIDE SURFACE OF ①)
4. COLLIMATING LENS (NEGATIVE LENS)
5. HEAT DUMPING FILTER (METAL-DIELECTRIC, NARROW BAND PASS, HIGH REFLECTIVE $\lambda > 8000 Å$)
6. OPERATING FILTER (VERY NARROW BANDPASS, BIREFRINGENT, OR THIN FILM)
7. FOCUSING LENS (POSITIVE LENS)
8. VIDICON TUBE
9. ELECTRONICS

- BETWEEN ④ AND ⑦ LIGHT IS PARALLEL
- LENGTH ① TO ⑧ 36 INCHES (MAXIMUM)
- DIAMETER 3.5 - 4.25 INCHES

Fig. 7-2 Optical System Telescope Design

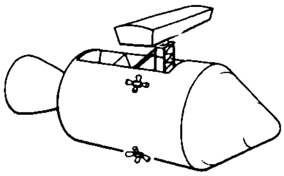
and wide field. The effective focal length (EFL) is typically three times or more greater than the length of the main telescope tube. The Cassegrain configuration allows telescope self compensation for thermal effects. The optical elements will be fabricated for nonbrowning materials. The focal length will be 45 to 60 inches and the clear aperture 3.0 to 3.5 inches.

Focal Length. Required focal length is calculated from the formula:

$$d = \theta F \tag{7.1}$$

where d = image diameter = 0.5 inches
and θ = angular diameter of object = 32 arc min
= 0.0093 radians

Therefore, $F = d/\theta$ = 54 inches = focal length of telescope



Diffraction Limit. The diffraction limit of a telescope, based upon the Rayleigh criteria and the Airy disk, is given by:

$$w = \frac{1.22\lambda}{D} \tag{7.2}$$

where w = angle subtended by image in the focal plane (arc sec)
λ = wavelength of light - (6563Å at Hα)
D = diameter of objective of telescope (inches) (entrance aperture)

To present a satisfactory diffraction limit would require a telescope of about three inches diameter. (Bouwers-Maksutov type telescopes are commercially manufactured in apertures of 3.5, 4 and 5 inches.)

Formulation of the resolution, based upon the maximum intensity of the diffraction disk due to object number two being located upon the first minimum of the diffraction disk due to object number one, has been found to be too severe. It has been found empirically that, for visible light, the detectable separation between two objects of equal intensity is given by the formula: w (in arc seconds) = 4.5/D (in inches), quantities defined above. (This is the well-known Dawes Limit.)

The entire question of resolution of detail is one of considerable complexity; the resolution of any optical system is related to aperture, focal length, obstructions, vignetting, image contrast, detector sensitivity and response, quality of optical components, and wavelength of light. Certainly resolution of detail beyond the Rayleigh limit is known and of practical importance if the necessary trade-off is acceptable. A rule of thumb restricts the optical circle of confusion to one-third the size of the vidicon scanning beam.

Field of View. To obtain the desired field of view and image size requires a long focal length system. Requirements of instrument package size cause rejection of long focal length refractors. Cassegrain or corrected Cassegrain systems would give the most satisfactory results.

Thermal. One of the properties of the Cassegrain configuration is the comparative ease of thermal compensation: proper choice of optical and mechanical components would allow complete self compensation over the operating range of temperatures.

7.3.1.2 Narrow-band Filter

One of the principal elements in the ATM solar monitor subsystem is an extremely narrow-band optical filter which provides an image of the sun at a wavelength chosen to evoke a maximum of information relevant to active regions on the disc. As discussed in Section 7.2 above, some of the most important features, such as prominences and flares, emit visible radiation only in narrow spectral lines. Wavelength selective filtering serves to isolate an emission line of interest so that the pertinent features are not washed out by the brilliance of the photosphere in the continuum. The only alternative is continuous, high-speed scanning of the entire disc with a spectrohelioscope slit.

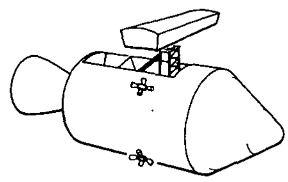
A second, equally important reason for the use of selective filtering is the necessity for preventing the absorption of heat from the incident solar radiation. The solar monitor and spar in general, and the narrow-band filter and vidicon faceplate in particular, put severe constraints on the permissible heat flow into the system. The thermal considerations are discussed more fully in Section 5.

One heat rejection filter, having a bandpass of the order of 50Å centered on Hα, is placed ahead of the telescope as shown in Fig. 7-2. This reflects a large part of the heat before it reaches the telescope proper. A second prefilter of about 10Å bandpass is placed immediately ahead of the principal filter to further alleviate any possible heating or internal reflection problems.

To relieve filter sensitivity to angles of incidence, auxiliary lenses are incorporated in the system. One lens collimates incident light on the filters and the second focuses the solar image on the vidicon target. (See Fig. 7-2.) These lenses are fabricated from nonbrowning materials.

Until quite recently, the only type of applicable full field filter was a birefringent monochrometer of the type invented by Lyot, and variations of this. Solar observatories commonly employ birefringent filters having a bandpass 0.5Å wide for flare patrol work, giving a spectral purity of better than 10,000. It has been brought to our attention* that commercial thin-film interference filters are now available which approach the spectral selectivity obtainable with birefringent filters. The latter are much larger and heavier and have a passband sensitivity to temperature 10 or 20 times greater than that of the interference filter. This temperature dependence requires that a birefringent monochrometer be held at a specific absolute temperature within about two tenths of a degree in order to keep it centered on the spectral line of interest. The interference filter, on the other hand, has a temperature slope of 0.05Å/°C. This enables the flare detection requirements of the solar monitor to be met with a ±5°C variation from the center temperature. This kind of temperature control can be achieved on the ATM spar, with careful thermal analysis and control. On the other hand, the few tenths of a degree limit demanded by the birefringent filter presents an extremely demanding situation in this type of application. Therefore, it is recommended that a thin-film type of narrow-band filter be used, and that appropriate thermal control be designed into the system. These filters in the 0.5 to 1.0Å bandwidth range are particularly sensitive to angle of incidence, and this is the reason for the auxiliary collimating lens system mentioned above and shown in Fig. 7-2.

*Conversation with J. Firor and R.H. Lee of HAO.



7.3.1.3 Design Considerations at Hα

The Hα emission is a convenient indicator because of the sharpness of its Fraunhofer absorption line. A rough sketch of the profile is given in Fig. 7-3. The chromospheric intensity at the bottom of the line is given in Allen* as 16 percent of the continuum (photosphere). Areas brighter than the background chromosphere, which are seen floating in the lower atmosphere, are called plages. These plage areas usually accompany sunspots. A solar flare typically manifests itself as a sudden, very rapid increase in intensity in a plage region. (See Fig. 7-4.) Both the area and intensity increase rapidly for a few minutes followed by a slow decay, which may take from a few minutes to several hours.** As shown in the figure, the plages are about twice the intensity of the undisturbed chromosphere.

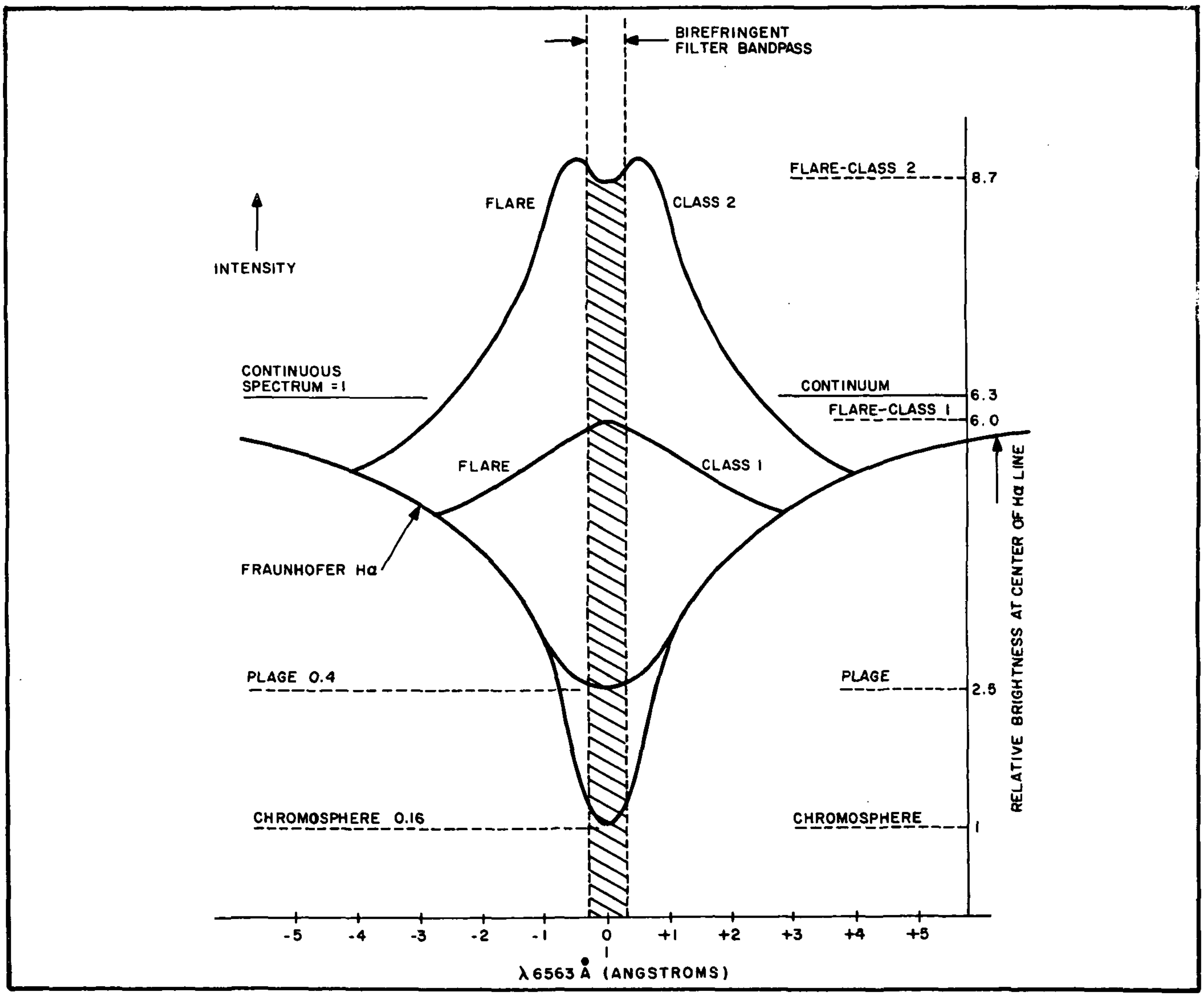


Fig. 7-3 Spectral Profile of Flare Hydrogen Alpha Emission

*Astrophysical Quantities, C. W. Allen, 1963, pages 168, 186.

**Solar Flares, H. J. Smith & E. P. Smith, 1963, pages 65-70, 99-103.

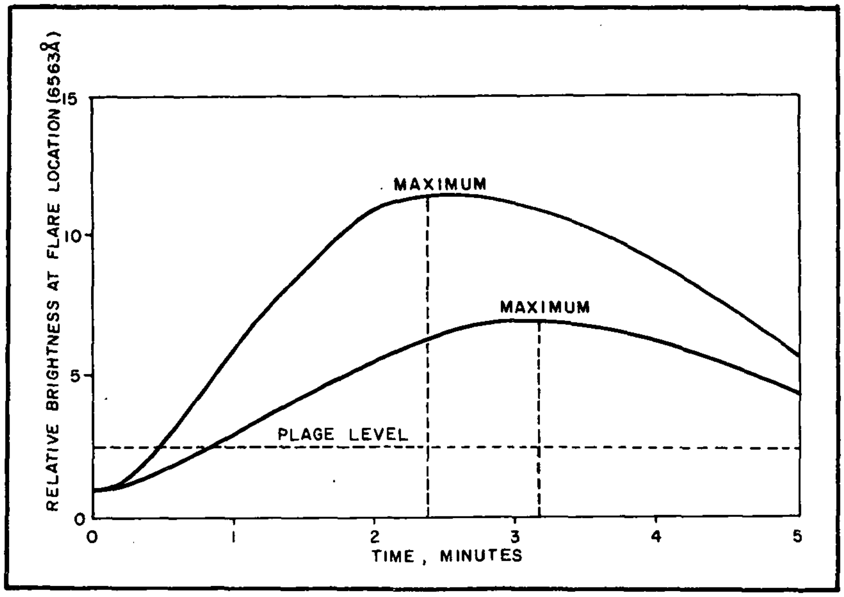


Fig. 7-4 Two Typical Time Profiles For Class 2 Hα Flare Development

A class one flare, the most frequent type, may be more than double the intensity of the plage where it originates, but, as shown, this is well below the brightness of the photosphere itself. Therefore, even a filter with a bandpass only a few angstroms wide would not offer enough contrast for the flare to be detected by eye. Unless very large and bright, it would have low contrast with respect to the continuum. The maximum possible contrast in the image is needed for visual detection of typical flares because of their small area relative to the entire disk. The visual angle subtended by a flare to the eye, viewing the solar disk, can be small enough to increase the difficulty of seeing a class one flare, especially during the initial rise. This is all important if observations or measurements are to be started before the peak intensity is reached. The mean area of many flares in the class one category is approximately 100×10^{-6} of the solar hemisphere.* This implies a width of roughly 10×10^{-3}, or 1/100, of the solar diameter. (This would be about 0.04 inch on the monitor screen or a visual angle of eight arc minutes at the most comfortable viewing distance.) An area of this size may escape notice until significant contrast is developed.

The only way to obtain sufficient detectability is to observe the Hα line with a spectroheliosope or through a birefringent monochromator. The latter allows the entire disk to be viewed simultaneously without resorting to scanning techniques. Solar observatories commonly use birefringent filters such as described by Evans** and Ohman*** having a passband about 0.5Å wide for flare patrol work. In this way

*Astrophysical Quantities, C. W. Allen, 1963, pages 168, 186.
**Astron J., Vol. 64, No. 8, page 330, 1959.
***Observatory, Vol. 73, page 203, 1953

the full contrast ratio, as indicated at the right side of Fig. 7-3 relative to the undisturbed chromosphere, is made available to the observer. It will be important that the combined "gamma" characteristic of vidicon and monitor kinescope does not unduly compress the hard-to-get contrast in the flare image.

The area of a flare is used as a measure of its importance as shown in the following table:*

Table 7-1
SOLAR FLARES

Flare Class	Mean Area in 10^{-6} of Sun's Hemisphere	Mean Duration	No. of Flares Reported 7/57-12/58
<1	<100		835
1	100-250	20 min.	5389
2	250-600	30 min.	497
3	600-1200	60 min.	41
3+	>1200	3 hr.	

Typical time profiles during flare development are shown in Fig. 7-4.** This illustrates how a large amount of the important initial rise to the peak can occur within a minute or two. The visual observational threshold should be as low as possible on the steep-rise portion if the astronaut is to perceive the onset of an event and orient the experiments to the critical feature before the radiation peak is reached. If the experiment involved has a field of view of only five arc seconds, this means that the astronaut must put a 0.01-in. circle (the field of view) at the center of the display tube inside an area about 0.04-in. across (the flare) anywhere on the disk. (See Fig. 2-12.) This must be done rapidly if the peak radiative flux from a fast rise flare is to be measured.

In view of the demand on the eye to locate a five arc second element in the display of the 40 arc minute field of the solar monitor with enough certainty to make reasonably full use of the experiment capability, it becomes necessary to provide magnification of the image of the center portion of the display tube picture. The ability to magnify the center portion by a factor of 5 or 10 by increasing the focal length would be invaluable, assuming adequate spatial resolution of the ATM telescope and vidicon combination. However, at present there is no way of guaranteeing that the optical axes of the several experiments can be kept within five arc seconds of the axis of the solar monitor. Therefore, a separate boresight optical system within each experiment with a narrower field of view is required, commensurate with the angular resolution requirements of the particular experiment. The display CRT is capable of being switched to any experiment boresight sensor as desired.

*Applied Optics Vol., No. 2, page 88, March, 1962.

**Solar Flares, Smith & Smith, page 102.

7.3.2 Video Sensor

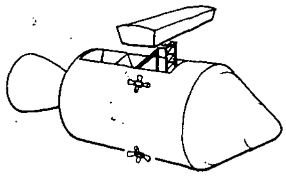
The video sensor is the next link in the chain of electro-optical elements which functions to effectively place the eye of the astronaut at the image plane of the ATM experiment telescopes. The maximum capability for quick and accurate acquisition of a small active feature on the solar disk will be determined mainly by the astronaut and the quality of the remotely observed solar image.

In view of the role of the solar monitor subsystem in a closed-loop pointing control system, where fast response is desired, the visual link should introduce no lag other than that due to the perception and reaction time of the astronaut. A minute spent balancing out overshoots in order to settle down on a specific point would be costly in terms of capturing the fast initial rise of Hα radiation during a flare. Hence the link must operate in real time, and the many slow-scan techniques, whether or not image conversion or storage techniques are used, must be ruled out.

A time of 1/30 second per frame is ordinarily required to avoid a flickering image. This rate satisfies the integrating time of the eye, but a 1/15 second frame time is nearly as good except at high intensities with standard phosphors. (The advantage of going to a 1/15 second frame time is a reduction of the system bandwidth requirement by a factor of two.) Frame times shorter than 0.1 second exceed the integration time of the human eye sufficiently to cause noticeable flicker, eye strain and degraded ability to perceive small-area, low-contrast features. Low-persistence phosphors on the monitor can reduce the flicker, but in so doing inject a lag into the pointing control loop.

An additional point in favor of a relatively fast scan is the absence of any requirement to use a vidicon having a special "slow-scan" photo conductor, thus eliminating the possibility of erasure problems which might otherwise interfere with the closed-loop response.

One of the most important camera-tube parameters for the ATM is the two-dimensional resolving power throughout the field. The final performance of the entire system depends on how well the astronaut can observe and track fine structure, regardless of its orientation or of its position on the solar disk. Along with the small size of many significant features, such as flares, is the fact that these entities do not necessarily exhibit a great intensity contrast relative to their surroundings. It is important that contrast degradation be held to an absolute minimum at spatial frequencies up to and including those represented by the smallest resolution element of interest. It also appears that, to the extent possible, the high frequency response should be inherent in the aperture response of a high quality camera tube rather than rely on aperture correction and peaking in the associated circuitry. The reason for this is the large amount of "noise" present in the object of study, itself: namely, the constantly varying and complex detail of the sun's chromosphere. Much of the randomly shifting mottles and fibrilles, for example, may represent spatial frequencies equivalent to the five arc second angular resolution element. It is conceivable that undue accentuation of this structure might produce a confusing image rather than a sharp, clear image.



Resolution figures given in terms of the "number of TV lines" in a given direction usually refer to the "limiting" resolution, a condition wherein objects having 100 percent contrast are imaged as having 3 to 10 percent contrast--just detectable by eye. Thus, to detect objects of low inherent contrast, the modulation transfer must be well above the limiting value for spatial frequencies considered significant. Imaging the 40-arc minute field onto a 0.525-inch circle on the retina of the vidicon established a relationship wherein a 30-arc second element of resolution is equivalent to a spatial frequency of 6 line-pairs/mm. Virtually 100 percent contrast transfer can be achieved at this frequency by specifying that limiting resolution occurs at 30 line-pairs/mm. A frequency of 30 line-pairs/mm represents fine detail equivalent to a six arc second element. This is not an excessive requirement, however, in view of the fact that while the solar monitor is required to resolve accurately details like 30 arc seconds, it is also desired to retain detectability of details down to 5 arc seconds. Accurate resolution of such fine detail can only be obtained by switching to one of the individual experiment boresight telescopes, which have a total field of view of only about 5 arc minutes in place of the 40 arc minute wide field of the monitor. Providing the diffraction limit of the individual boresight telescope is within bounds, a 5 arc second detail now represents a spatial frequency of 6 line-pairs/mm on the vidicon retina, permitting 100 percent contrast transfer of the fine structure as required.

Additional impetus for obtaining high resolution from the telescope and vidicon is presented by the manner in which all the links in the closed-circuit video chain act like a series of cascaded low-pass filters to the spatial frequencies needed to transfer full object detail. Degradation of the final image can be minimized only by requiring that the modulation transfer function of each component be significantly greater than that specified for the entire system.

7.3.3 Video Transmission

The video image generated by the monitor vidicon sensor in the oriented section will be transmitted by coax to the ATM control unit in the CM. All video signals from this sensor and the experiment sensors will be selected in a multipole RF switch located in the oriented section, since only one sensor image will be viewed at a time. This switch will maintain precise impedance termination for each signal line and will maintain cross-talk isolation between adjacent lines of at least 40 db below the signal. The astronaut will select the video channel by command from the ATM control unit.

This single video signal will be transmitted by coax from the oriented section, across the gimbals to the interface coax connector in Sector I of the SM. Existing coax leads will be used in the SM and CM to transmit the signal to the ATM display unit. The need for high resolving power in real time presents a severe bandwidth requirement. The transmission of high spatial frequencies requires high electrical bandwidth if the picture must be scanned rapidly. For example, if the image is scanned with 1150 lines in order to have a vertical resolution of 800 TV lines, and the aperture response gives equal horizontal resolution, then the electrical bandwidth requirement for a 1/30 second frame time is about 20 MHz. This does not include the extra demands for blanking, which could increase the figure close to 30 MHz.

High video bandwidth such as this may be achieved if appropriate care is taken,* but more power is required, S/N requirements increase, and more quality is required by any transmission cable. The latter is particularly important in view of the effort to use miniature, flexible coax around the ATM gimbals. Choice of a 1/15 second frame time will halve the bandwidth requirement. Data compression schemes may conceivably yield further bandwidth reduction without sacrificing the quality of the system. Numerous bandwidth reduction schemes have been used,** and it is possible that one of them could be used in this system.

Two undesirable situations would be introduced by attempting the use of the flexprint leads for the video signal. One of these is the absence of shielding, and the other is the likelihood of high frequency reflections due to imperfect matching at the flex lead interface. Poor terminations along the video transmission line would result in signal overshoots and spurious images. This kind of phenomenon could seriously interfere with the ability of the astronaut to observe and track fine structures on the solar disk. The lack of shielding could allow electromagnetic interference to contaminate the video signal and would permit the video signal to radiate energy which could interfere with other ATM subsystems and experiments. Thus, coaxial cable is to be used all the way, including around the gimbals.

7.3.4 Video Display

7.3.4.1 Illumination

The human eye is principally sensitive to light of wavelengths between 0.380 to 0.760 microns. (For various individuals and various sources limits extend from 0.310 to 1.050 microns.) People can see well enough to walk with illuminance of 10^{-9} lumens per square centimeter. Illuminance becomes painfully bright if it rises above 50 lumens per square centimeter. (Maximum illuminance due to direct and scattered sunlight at the surface of the earth is about 15 lumens per square centimeter.)

The phenomenon of glare discomfort is very complex, and very many factors must be considered. Even ordinary levels of illuminance become painful if an observer is surrounded by an unbroken field of diffusely reflecting material of high albedo (such as fresh snow). The resulting physiological state is termed snow blindness.

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*"A 60-Megacycle Video Chain for High-Definition Television Systems", O. H. Schade, Sr., RCA Rev. Vol. 26, pages 178-199, June 1965.

**"The Improved Gray Scale and the Coarse-Fine PCM Systems, Two New Digital TV Bandwidth Reduction Techniques." W. T. Bisignani, G. P. Richards and J. W. Whalen IEEE Int. Conv. Rec., Vol. 13 Part 4, pages 55-73, March 1965.

"An Experimental Study of the Possible Bandwidth Compression of Visual Image Signals", C. Cherry et al, Prog. IEEE VSI, page 1507, Nov. 1963.

"Digital Television: Shrinking Bulky Bandwidths", Knight et al, Electronics Vol. 37, No. 31, page 77, December 14, 1964.

The above discussion refers to overall illuminance, and not to illumination entering the eye. Much lower tolerances exist for an eye directly viewing a bright source. The upper limit here is about 0.05 lumens per square centimeter. The lower limit of faint sources against a dark background (faint star against sky) is about 10^{-13} lumens per square centimeter. This corresponds to a star of magnitude 8. (Under best conditions it is difficult to see stars fainter than magnitude 6.5.)

7.3.4.2 Resolution

The definition of the limit of resolution of the eye is usually given as being the minimum angle subtended by two neighboring points, where the eye can separate the two points as being in fact, two points. Such limits are a function of a considerable number of variables, even in physical and physiological factors and neglecting psychological effects.

Four ocular phenomena are involved in limiting visual resolution: (1) diffraction by the pupil of the eye; (2) geometrical and chromatic aberrations of the eye; (3) discontinuity in the structure of the retina (detector array of the eye); and (4) the mechanisms known as day vision (photopic, or high light intensities), and night vision (scoptopic or weak intensity). While it is a simplification to discuss resolution of the eye as a function of these four factors, it would seem that these factors are the major ones and lead to reasonable results in predicting visual resolution.

In practice, the resolution of the eye is established by testing normal eyes under such conditions as to remove psychological factors from consideration. Results of such tests by many investigators and subjects indicate the best possible resolution of the eye as one minute of arc. For most subjects, under most conditions, the best obtainable resolution was slightly better than two minutes of arc.

It is generally acceptable for system considerations to assume a resolution for the eye of two minutes of arc.

7.3.4.3 Cathode Ray Tube

A console display for a single observer is normally viewed at a distance of about 18 inches. The two arc minute resolution limit of the eye then corresponds to an element size of about 0.01 inch on the display. Taking this limiting condition as referring to a just-detectable (not accurately resolvable) element corresponding to a 5 arc second solar detail, it then follows that an active display screen width is equal to 400* (0.01 inch) or 4 inches. To achieve this requires a so-called "five inch" cathode ray tube (CRT). A solar detail of 30 arc second size then occupies about 0.06 inch on the display, presenting a visual angle of 12 arc minutes to the eye of the astronaut. A feature of this size should be resolvable quite well by the observer, even if the object contrast is not great. Figure 7-5 illustrates an example of how the low-pass filtering effect of the eye to spatial frequencies can be one

*400 (five arc second) diameters is approximately one solar diameter.

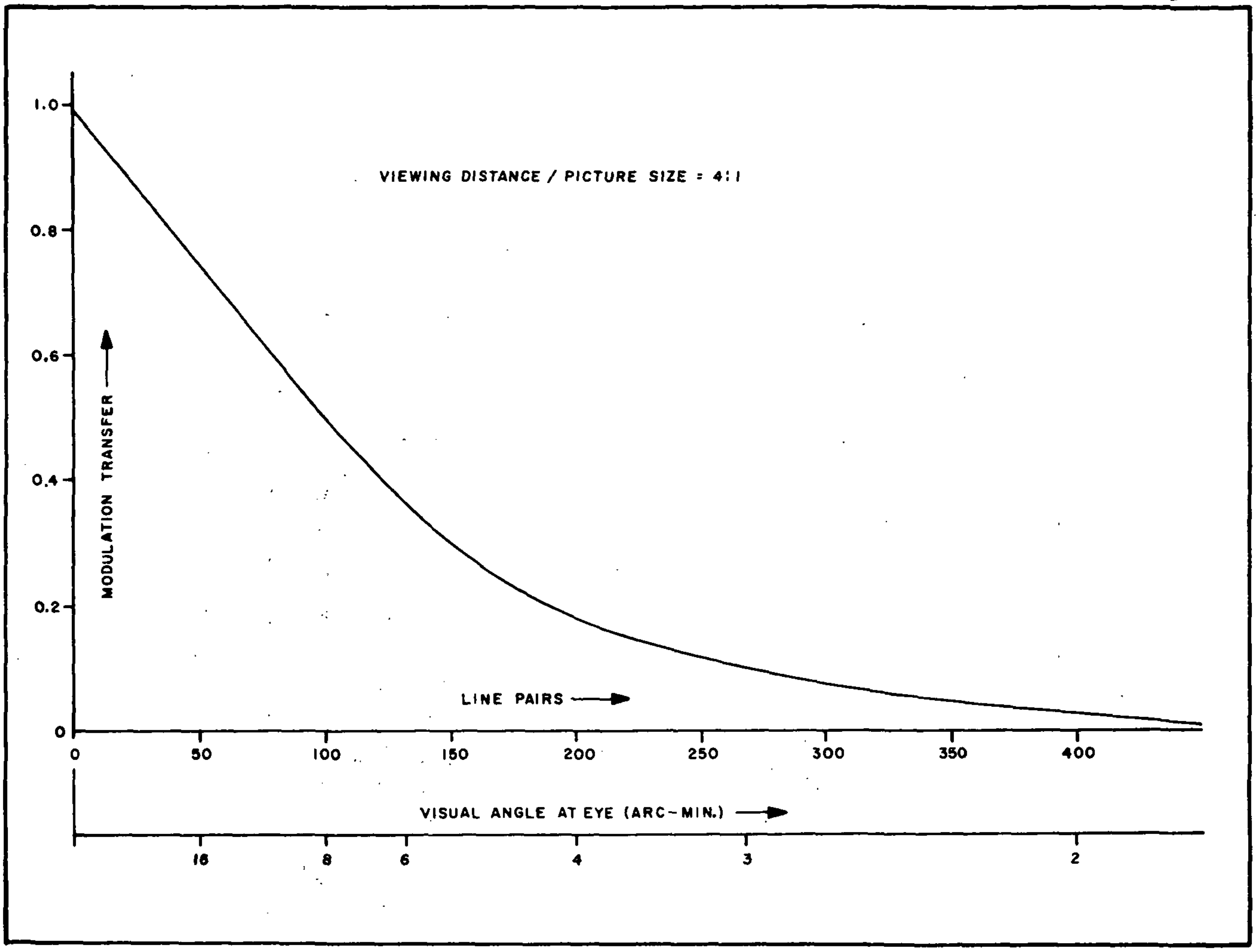


Fig. 7-5 Aperture Response of the Eye

of the more serious factors leading to loss of detail in the overall transmission. For the expected five arc minute field of view of experiment boresight sensors, a five arc second diamter solar detail will occupy 0.1 inch on a "five inch" CRT.

In order to match the resolving power and quality of the rest of the system, the CRT used for display must have a high bandwidth, a small spot size, and good deflection linearity. A number of high resolution, ruggedized tubes are available in this diameter. These tubes, in general resolution, have spot sizes in the neighborhood of 0.001 in. to 0.005 in. and are magnetically deflected to a maximum angle of 42 degrees. (This requires 10-20 kv accelerating potential.) The electron-optical geometry under these conditions results in a minimum tube length in the 13 to 14-inch range.

For a deflection angle of 70 degrees or more, the length could be reduced to something around eight or nine inches. (The accelerating potential is 8 to 10 kv.) The result would be an increase in spot size by a factor of 10. This would limit the maximum number of lines which can be displayed to 400 or less compared to 2000 to 4000 for the longer tubes. The deflection linearity and the flatness of the field would also be degraded as the deflection angle is increased. These figures indicate that use of any tube shorter than 13 inches will not yield the required image. One possible configuration of the display panel is shown in Fig. 7-6. Provision for mounting a camera is included to allow photographic recording of the CRT picture.

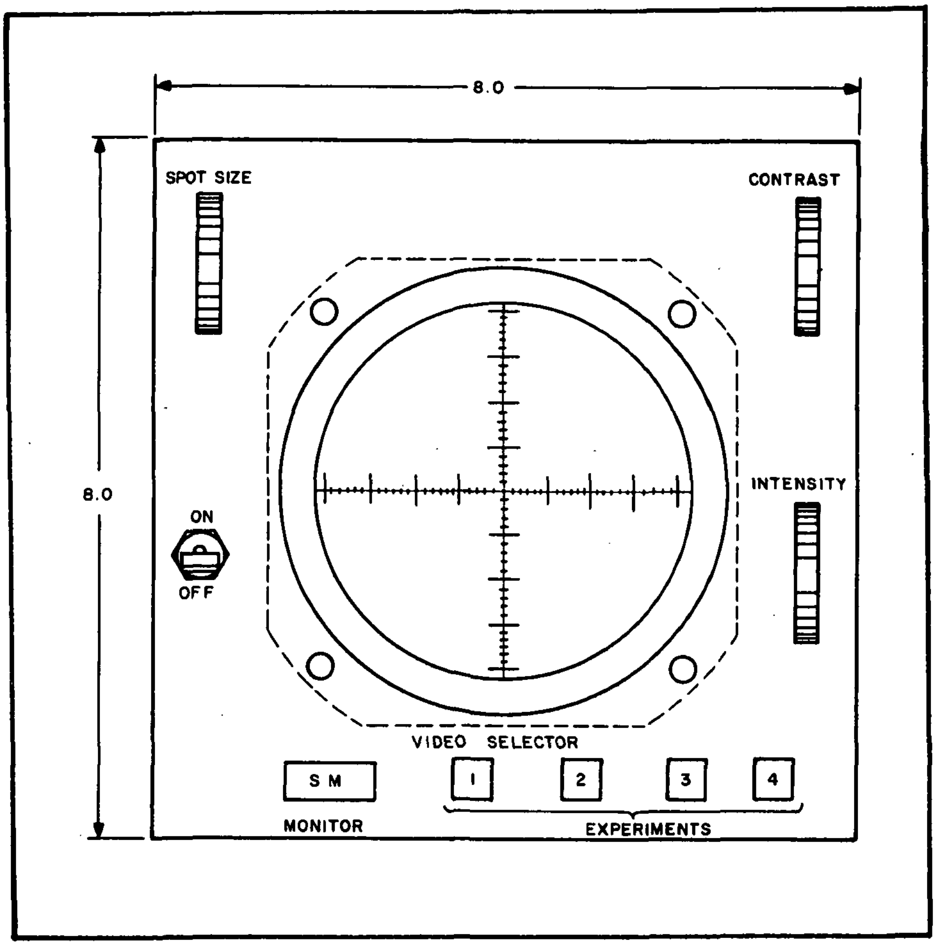


Fig. 7-6 Solar Monitor Display Panel

SECTION 8

COMMAND SUBSYSTEM

Section 8
COMMAND SUBSYSTEM

8.1 REQUIREMENTS

The fundamental requirement of the command subsystem is to provide a means by which an astronaut in the CM can operate the ATM. This involves erection of the ATM from within the SM, pointing the ATM at a desired target, operating various experiment modes, monitoring the status of ATM subsystems, and jettisoning the oriented section.

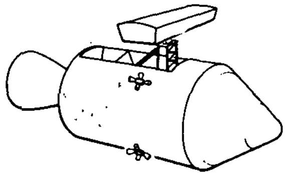
The necessity for operating and monitoring the status of the ATM established the requirement for a control unit which can be conveniently located in the CM. The control unit must have switches to enable the astronauts to perform the following operations:

(1) Jettison the door
(2) Erect the ATM
(3) Control power to subsystems and experiments
(4) Control the position of the ATM and experiment aperture covers
(5) Control the attitude of the ATM in pitch, yaw and roll
(6) Control the position of experiment occulting discs and other operating modes
(7) Control the operating modes of the other subsystems
(8) Jettison the oriented section

The control unit must also have indicators to monitor the result of operating the switches mentioned above and to monitor the status of the subsystems. Indication of yaw, pitch and roll position must be continuously monitored. Other indications may be programmed.

8.2 DESCRIPTION

A block diagram of the command subsystem, which will satisfy the requirements stated in Section 8.1, is shown in Fig. 8-1. The front panel layout of the control unit is shown in Fig. 8-2. The command subsystem provides discrete and keyboard commands, continuous indicators for essential data, and programmed indicators for status data. Hardwire commands are used where crew safety is involved, or where undue complexity will result if encoded commands are used. There are 512 commands which are encoded and transmitted to the bay and oriented section. The encoded command concept and the programmed indicator concept is used to reduce the number of conductors between the CM and SM, and across the gimbals.



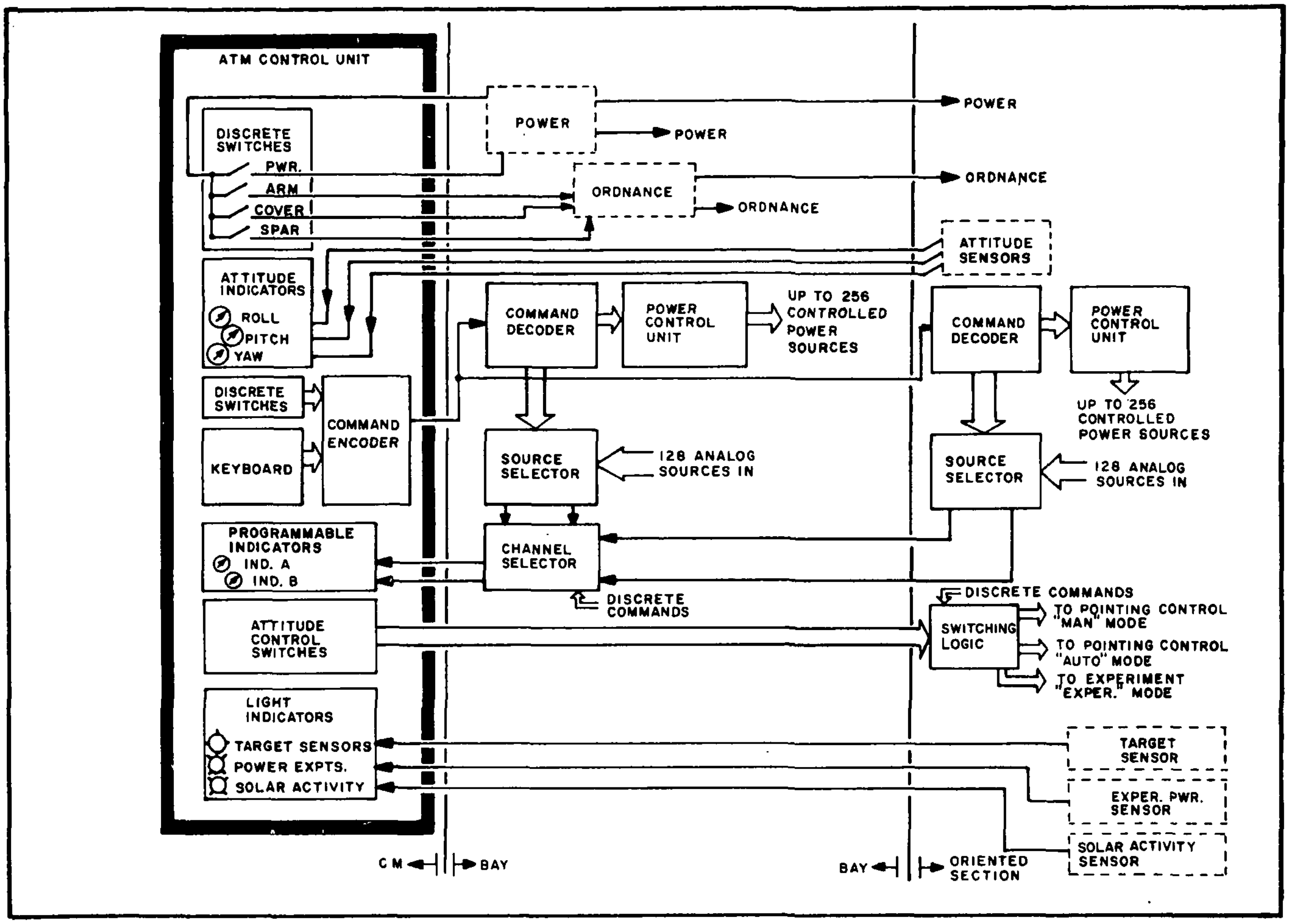


Fig. 8-1 ATM Command Subsystem Block Diagram

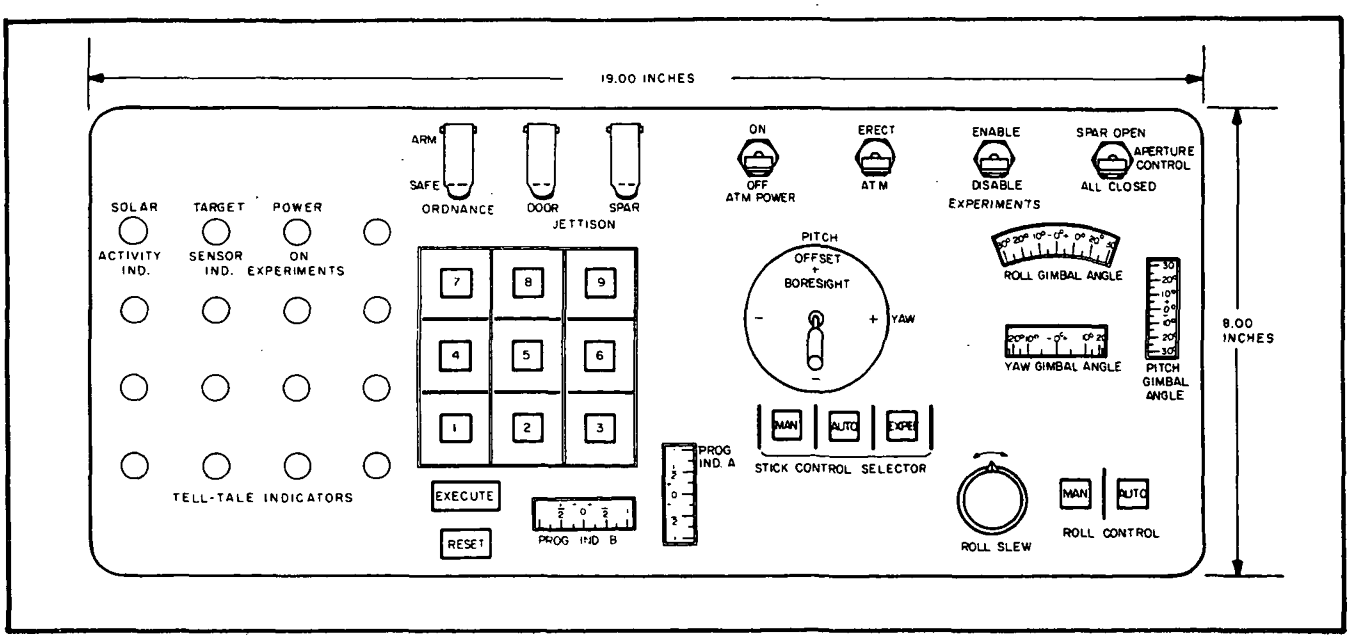


Fig. 8-2 ATM Control Panel Layout

8.2.1 Control Panel

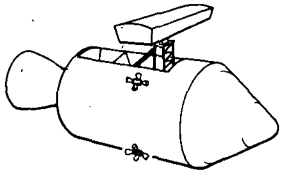
The control unit panel layout of Fig. 8-2 shows several types of switches and indicators. Each type was chosen after considering the comments of several members of the Astronaut Office at MSC, the frequency of use and crew safety. Ordnance switches are covered so that the operator must raise the cover before operation of the switch is possible. Ordinary toggle switches are used for some functions where the position of the switch indicates the state of the function or functions. A stick is shown for controlling the oriented section in pitch and yaw, and for positioning an occulting disc in the HAO study experiment. When the pointing control subsystem is in the manual mode, the stick is used to slew the oriented section to the solar disc. When the pointing control subsystem is in the automatic mode, the stick is used to offset-point the oriented section to a desired target on the solar disc. For both the manual and automatic modes, the stick will control the oriented section in 25 combinations of pitch, yaw and rate. Roll position of the oriented section is controlled with a single switch. This switch is used to slew or offset-point the oriented section in the manual mode and automatic mode, respectively. The nine-button keyboard, which is used to send encoded commands, uses push button switches. After the operator depresses the appropriate buttons for a specific command, the "execute" bar is depressed to send the command. The keyboard switches remain depressed until the "reset" button is depressed.

Meters are used to indicate continuously the oriented section position relative to the CSM position. Two meters are used for up to 256 programmed indications to monitor the status of a selected data channel. The data channel is selected or connected to one of the indicators by sending the appropriate command with the keyboard. The remaining indicators are shown as lights or flags and are used for monitoring events such as solar activity or experiment power. Sixteen lights are shown. More study is required to determine the optimum number of indicators.

8.2.2 Commands

Two types of commands are used: hardwire commands and encoded commands. Items such as ordnance control, power control and attitude control are hardwire commands. Attitude control requires 10 conductors between the CM and the oriented section. Attitude control can be accomplished with the encoded commands; however, additional complexity will result.

Encoded commands are transmitted by using either the keyboard or discrete switches. For example, the manual and automatic pointing control system modes are transmitted with discrete switches. This is done for ease of operation on frequently used commands. When a discrete switch is used to transmit an encoded command, a nine-bit word is automatically formatted into the command encoder. Less frequently used commands are formatted into the command encoder by depressing the appropriate buttons on the nine-button keyboard. There are a total of 512 encoded commands transmitted on one conductor from the control unit to the bay command decoder and to the oriented-section



command decoder. Two hundred and fifty-six of these commands are used to control the ATM system, and the remaining 256 are used for programming analog sources to indicator A and B on the control panel. The distribution of commands in the bay and oriented section depends on the number of commands required in each location; however, the oriented section will require more commands than the bay since 40 commands are allocated for experiment use. Each command decoder drives a power control unit. The power control unit will provide pulses to drive relays, or will provide relay logic depending on the subsystem and experiment requirements.

8.2.3 Indicators

Each command decoder drives a programmed source selector. Each source selector connects two analog sources to the channel selector. Discrete encoded commands sent to the channel selector connect two of these four sources to indicators A and B on the control panel.

The 16 control panel indicator lights or flags are hardwired to sensors in the oriented section. These indicators could be time multiplexed if the number of conductors between the CM and the oriented section became excessive; however, in the interest of simplicity, hardwires are shown at this time.

8.3 CONTROL UNIT LOCATION

The location of the control unit in the CM is in the experiment control panel in the lower equipment bay, as shown in Fig. 2-7. The control unit weighs approximately 10 pounds, is 6 inches deep, and the front panel is 8 x 19 inches.

SECTION 9

DATA SUBSYSTEM

Section 9
DATA SUBSYSTEM

9.1 GENERAL REQUIREMENTS

The data subsystem is required to provide the following features:

(1) Recovery of prime measurement data in digital form from the ATM experiments.

(2) Retrieval of secondary data from, and in support of, the ATM experiments to allow optimum use (correlation, sequence, operating mode, time, etc.) by the cognizant scientists of prime data when recovered.

(3) Recovery of the necessary minimum data from the experiments to allow required post-flight analysis and correction of any major malfunction or contingency which could affect mission success.

(4) Retrieval of analog and digital data from supporting subsystems as needed for mission operations and post-flight analysis.

The minimum design requirement must accommodate the experiment prime data requirements discussed in Section 3.2. The major user is the HCO study experiment, whose requirements are listed in Table 3-5. The required ten 16-bit input channels gated at 25 samples per second indicate a maximum data rate in excess of 4000 bits per second will be needed for the overall subsystem.

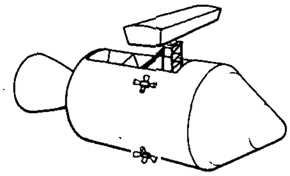
Other data requirements more dependent on final subsystem design and operational planning are not well defined at this time. Nevertheless, the data subsystem, just as other subsystem conceptual requirements, must be determined in order that the amount of useful data of scientific value will not be unnecessarily constrictive.

9.1.1 Interrelation Between Data and Command Subsystems

The command subsystem discussed in Section 8 provides all basic services, including data selection and presentation, needed to allow the astronaut to properly operate the ATM system and obtain optimum results from the experiments. The data subsystem, on the other hand, serves the mission in a somewhat different way.

The information obtained by the data subsystem lends itself more to ground-based processing and interpretation techniques. Access to this data by representatives of the experimenters, located at a mission control center during the flight, can lead to added confidence in optimum mission success.

The scientist/astronaut data-link conferences will take on more meaning when the scientist has useful detailed data on ATM environment and experiment performance to complement the astronaut's "on-the-spot" observations. Because of this difference in basic use and in functional requirements, it has been concluded in this study that the basic implementation of the two subsystems (data and command) should be handled separately.



9.1.2 Basic Goals

Among the not always compatible, but important data subsystem design goals are:

- Reliability - A carefully balanced system design requires that the data subsystem, regardless of functional requirements, be implemented in such a way that it does not become the "weak link" in the reliability chain.
- Multi-Experiment Configuration Adequacy - In balance with other goals, the data subsystem should be capable of accommodating a somewhat different complement of experiments than that referenced in Section 3.1. This is in line with cost effectiveness goals, since a complete redesign of the subsystem for different flights would be too costly.
- Flexibility - The ability to accommodate different operating modes (discussed in Section 3.1) during a given flight in an effective manner is desirable.
- Growth Potential - The data subsystem designs conceived as adequate today may not be adequate for 1970. Therefore, the ability to expand and improve capabilities in a modular manner is desirable.

9.1.3 Constraints

9.1.3.1 Apollo Interface

It appears likely that the highly capable and flexible Apollo unified S-band communication/telemetry/command systems could accommodate the minimal ATM-to-ground data interchange requirements without measurably affecting other services, and with a simple design/implementation interface.

Therefore, an initial ATM data subsystem design concept will be based on the availability of this service. However, if this service cannot easily be accommodated, an alternate ATM data subsystem design concept is considered.

The primary Apollo S-band links are:

(1) Apollo 2272.5 mc down-link - FM carrier with 1.024 mc subcarrier often modulated by Apollo telemetry tape recorder for playback to ground. (ATM could play back the stored data seven minutes out of every 180 minutes of flight.)

(2) Apollo 2287.5 mc down-link - PM carrier with 1.024 mc subcarrier normally modulated by 51,200 bit per second Apollo PCM telemetry system. (ATM could use any one, or combination, of available Apollo telemetry channels to output "patrol" or "housekeeping" real-time data to ground at rates not in excess of 400 bits per second.)

(3) Apollo 2106.4 mc up-link - PM up-link carrier with 70 kc subcarrier modulated by command subsystem at information rates of 120 bits per second. [ATM option might use this to transfer maximum of under 1000 bits at infrequent intervals (once/day) to modify adaptive telemetry format.]

Only a very small fraction of the Apollo data transfer capabilities will be utilized by ATM.

The most important service the Apollo communication link might perform is referred to in paragraph (1) above. The capability to read out stored ATM data to the ground distribution, processing and storage would benefit ATM in the following ways:

(1) Reduce necessity for providing long-term, high-capacity bulk storage of data in ATM.

(2) Reduce need for additional EVA maneuvers (added to film retrieval) to recover data storage device.

(3) Provide for some experiment data to be partially processed/ analyzed by scientific community during early flight orbits in order that operation plans and experiment settings might be varied to optimize later data acquisitions.

(4) Enable mission analysts to assist the astronaut to diagnose possible contingencies and recommend "fixes" which would least detract from the attainment of major mission objectives.

9.1.3.2 Astronaut Safety

Proper utilization of ATM data by ground based NASA operations personnel can enhance astronaut safety by:

(1) Reducing the need for EVA operations of a diagnostic nature or for unduly burdening the astronaut with detailed investigations using his necessarily limited on-board data processing and display capabilities.

(2) Reducing the need for an EVA recovery of data (not film) storage devices and subsequent restorage and recovery operations in the Command Module.

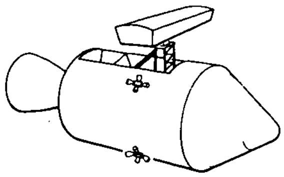
9.1.3.3 Power

A major design constraint is the need to minimize power requirements. This constraint was emphasized during early study phases when the need to provide an ATM incorporated battery pack for all power requirements was postulated. The present design concept allowing limited use of the more weight-efficient CSM fuel cell may allow more efficiency in implementing the data subsystem.

It is currently believed that the data subsystem average power can be kept in the range of 5 to 10 watts without going to expensive designs or new development in data assemblies. The higher power limit (10 watts) might be approached if the Apollo communications interface (Section 9.1.3.1) is not implemented.

9.1.3.4 Wires Across Gimbals

The difficulty in maintaining the accurate pointing control, while compensating for the forces imparted by wires running across gimbal assemblies, is discussed in Sections 6 and 10. Therefore, it may prove to be good overall design practice to incorporate data submultiplexers in both the ATM bay area and the oriented section in order to minimize the number of wires crossing the gimbals.



9.1.3.5 Weight

It is currently believed that the data subsystem can be kept to about 35 pounds on the oriented section and 20 pounds in the bay. This could increase to about 65 pounds in the oriented section if a total mission data storage requirement is imposed through denial of the ATM/Apollo data interface, or if the desirability for parallel storage and EVA recovery is determined.

9.1.3.6 Volume

Preliminary results of this study indicate that an adequate data subsystem could be incorporated in about 1000 cubic inches in the oriented section, and 600 cubic inches in the bay. The need for additional data storage and EVA recovery of electronic data, if Apollo interface is not possible, might increase the required volume to about 2000 cubic inches in the oriented section.

9.1.4 Conceptual Data Requirements

Table 9-1 indicates some of the basic data input channel requirements at present envisioned. It is highly possible that further definition of requirements during the detailed design stage may prove the need for more channel capacity than indicated, in order to allow for:

(1) Urgent needs that may not be identified until the overall ATM system design is better detailed.

(2) Higher response rates than here postulated (due to limited knowledge).

Table 3-5 indicates the total digital bits required for the HCO and GSFC study experiments. The 200 plus megabits indicated sets the minimum for total data storage requirements if data must be stored for the entire mission; that is, if the Apollo communications link is not used to transfer data.

More important, particularly if normal frequency or time-division multiplexing techniques are used, is the maximum data rate requirements. These rates are shown as a different requirement for two different data modes of operation in Table 9-2. The modes are in turn related to ATM operations modes as follows:

Data Mode	Operations Mode	Operations Definition
Slow	Housekeeping	All ATM experiments Off, but housekeeping data is required to maintain records on temperature, stress, command status, and other possible contingency and post-flight analysis monitors.
Slow	Patrol	Some ATM subsystems and experiments activated but no prime data being acquired.
Fast	Standard	Experiments collecting prime data at low rate
Fast	Active	Experiments collecting prime data at maximum rate.

The data rates postulated for "tell-tale" and analog housekeeping channels are probably adequate for a majority of signals. However, additional studies made after other subsystems become better defined will undoubtedly lead to higher rate requirements on some channels.

Some spare channel capability in the original conceptual design is desirable in order that these problem input channels may be accommodated by "supercommutation" (multiple multiplexer channels serving a single input channel).

Table 9-1
CONCEPTUAL DATA REQUIREMENTS
FIRST ATM MISSION

Data Source	Bay		Oriented Section		
	Digital tell-tales	Analog channels	8-bit Word channels	Digital tell-tales	Analog channels
Study Experiments					
1 HCO (10-16bit)	--	--	20	--	5
2 GSFC (4-16 bit)	--	--	8	--	5
3 NRL (Primarily Photo)	--	--	--	--	5
4 HAO (Primarily Photo)	--	--	--	--	5
ATM Internal					
5 Yaw Control	--	--	--	1	2
6 Pitch Control	--	--	--	1	4
7 Roll Control	--	--	--	1	4
8 Other Control	--	--	--	1	1
9 Temperature Probes	1	2	--	1	40
10 Solar Monitor S/S	1	2	--	1	10
11 Power S/S	1	3	--	--	--
12 Operate SPAR/Ordinance	1	1	--	1	--
13 Command S/S	3	1	--	5	1
14 Data S/S	2	1	--	2	1
15 Apollo Interfaces	3	5	--	--	--
TOTALS	12	15	28	14	83

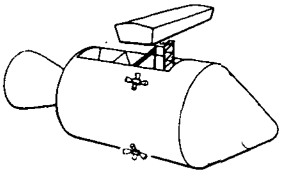


Table 9-2
CONCEPTUAL DATA RATE REQUIREMENTS

INPUT CHANNELS	SAMPLING RATES FOR DIFFERENT OPERATIONAL MODES	
	Housekeeping and Standard	Activity
(10) HCO 16-bit inputs	25 samples/second	25 samples/second
(4) GSFC 16-bit inputs	0.017 samples/second	0.1 samples/second
Other Digital (Tell-Tale)*	0.05 samples/second	0.25 samples/second
Other Analog (Housekeeping)*	0.05 samples/second	0.25 samples/second

*Based on a conservative sampling ratio of 5 as needed to avoid "aliasing error" when prefiltering is not investigated - See Stiltz, "Aerospace Telemetry" - Prentice Hall - 1961.

9.2 ALTERNATE DESIGN CONCEPTS CONSIDERED

Among the alternate design concepts considered during this study were the following:

(1) Use of Apollo PCM multiplexer channels exclusively - It does not appear that the Apollo spacecraft's PCM multiplexing equipment can adequately serve the ATM requirements without severely limiting its use for basic Apollo functions and making interface coordination very difficult. However, there appears to be several ways that spare Apollo channel capacity can be used to accommodate a real-time ATM data rate below 190 bits/second, and one way for accommodating a maximum total ATM data rate below 9600 bits per second for direct transfer to an Apollo ground station.

(2) Use of Apollo S-band link for stored data playback - It would appear a simple matter to time share the same subcarrier modulation bus with the Apollo tape recorder, since it is doubtful that either the Apollo tape recorder or the ATM data storage system will require a major percentage of the available data readout passes over appropriate Apollo ground stations. It appears that a relatively uncomplicated ATM data storage device can be read out no oftener than once every two orbits in less than seven minutes pass time allowed over a given Apollo ground station at rates comparable to the basic 51.2 kilobit/second Apollo playback rate. This concept can be combined with Item 1 or used independently. The study concluded that the two capabilities should be combined.

(3) Use of Apollo commands to coordinate stored ATM data playback - In accordance with the views expressed in Section 9.1.1 (Interrelation Between Data and Command Subsystems), it was a conclusion of this study that the Apollo command system can best control the stored data playback from ATM at a time and station location (preferably over the U.S. for fast transfer to control center) that best meets the combined Apollo/ATM mission profile.

The astronauts can, of course, initiate this action through their combined Apollo/ATM control units. However, their unique "on-the-spot" capabilities might be better used in other areas where less verbal integration with the ground station will be needed. Another use of the Apollo command link by ATM will be in the area of updating the optional adaptive telemetry format memories. The benefits of this optional capability are discussed in Section 9.4.

(4) Variable data rates as indicated in Table 9-2 - The difference in basic data rates between the two (slow/fast) data rate requirements leads naturally to consideration of a variable clock rate. Based on data needs thus far identified, this study concludes that two basic PCM clock rates (time coordinated with experiment modes) of about 120 bps and 4800 bps will adequately service the ATM requirements.

(5) Apollo time words - A simple telemetry frame counter was considered for providing necessary time correlation with celestial, Apollo mission and ATM function events. However, it is more desirable to read the available Apollo spacecraft time word into the ATM data storage device at frequent intervals. The study determined that primary effort should be directed to use the time word generated within the CSM.

(6) Experiment frame counters - The need was considered for providing experiment photo-frame counters. Many experiment designers preferred to superimpose this data directly on the film frame.

The functional superiority of end-item data on the same frame as the photo containing prime data cannot be seriously challenged. However, two minor disadvantages are noted:

a. Data will either have to be preexposed onto the film ahead of time or a somewhat complex photoelectric/optic device will have to be included in experiment design.

b. Data on the film will provide no housekeeping type data back-up for analysis in the event the film is not recovered.

It was determined that a simple binary counter working off the same command lines initiating exposures can be incorporated.

(7) Experiments provide own data systems - The experiments can incorporate their own data multiplexing and storage subsystem. The astronaut might then recover the data storage device at the same time he recovers the film package.

The study determined that it will be far more efficient for a central data subsystem to provide services to the experiments at the same time that needed ATM diagnostic and engineering data are being collected.

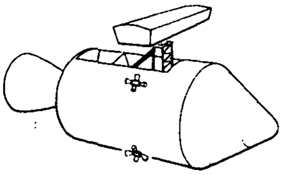
(8) Multiplexing - Alternate data collecting techniques were considered including:

a. Direct data recording on multi-channel tape recorders.

b. Frequency division multiplexing onto one or several main data-record channels.

c. Conventional time division recording into a single main data-record channel.

The latter technique, together with conventional PCM encoding to an experiment-compatible eight bit level, was selected as the optimum approach.



(9) Data storage - With the extensive plans for implementing Apollo ground stations to cover a large percentage of earth orbit operations, the electronic data storage may not be required. Direct real-time telemetry through an already operating Apollo or a supplementary ATM communications link was the alternative.

ATM data storage for at least two orbits of the Apollo was determined to be a favorable design approach.

Various data storage techniques were considered, such as:

- Film
- Magnetic cores
- Magneto-strictive delay lines
- Electromagnetic tape recorders
- Intermittent start/stop operations
- Start/stop with intermediate data storage ("buffer") techniques

A multi-speed (two initially) tape recorder with adequate storage capability for two orbits, and with ability to play back in seven minutes, was selected for the concept design.

9.3 DATA SUBSYSTEM DESIGN CONCEPT

Tables 9-3, 9-4 and 9-5 show schedules for a two-rate PCM multiplexing that would:

- Meet the minimum data requirements listed in Section 9.1
- Honor the study trade-off analysis discussed in Section 9.2
- Provide some flexibility and growth capability

Note that the GSFC study experiment data lines are not multiplexed directly by the main multiplexer (Table 9-3). An efficient PCM telemetry design requires that the relatively slow sampling rates (0.017 to 0.1 sample/second) be serviced by a sub-multiplexer rather than by the main multiplexer, whose speed has to be "sized" to the relatively higher (25 samples/second) requirement of the HCO study experiment. Table 9-4 indicates the minimum sampling rate requirements for the GSFC study experiment (0.017 samples per second during the standard mode and 0.1 samples per second during activity mode) have been more than met.

Figure 9-1 shows a block diagram of the data subsystem concept. While conceptually adequate, the exact channel capacity and arrangement may be changed to accommodate precise future requirements. Emphasis should be placed on investigating the suitability of already developed assemblies that can be adopted to this application.

Several assemblies, which are somewhat unique to the ATM conceptual design, are the Apollo time word buffer and telemetry R/O control and buffer storage, and involve interface design between Apollo and ATM. Their design does not appear difficult; however, more interface definition is needed before detailed design can begin.

Table 9-3

CONCEPT ANALYSIS - MAIN MULTIPLEX UTILIZATION

Main Multiplexer Inputs	(120 bps) Housekeeping and Standard Mode			(4800 bps) Activity Mode		
	No. of Input Channels	No. of Multiplex Channels (8 bits ea)	Input Channel Sampling Rate (Samples/sec)	No. of Input Channels	No. of Multiplex Channels (8 bits ea)	Input Channel Sampling Rate (Samples/sec)
HCO Study Experiment Prime Data	10(16 bit)	20	25	10(16 bit)	20	25
Bay An. Sub-Mux	1(8 bit)	4	1.25	1(8 bit)	1	12.5
Bay Dig. Sub-Mux	1(8 bit)	4	1.25	1(8 bit)	1	12.5
SPAR An. #1 Sub-Mux	1(8 bit)	4	2.50	1(8 bit)	1	12.5
SPAR An. #2 Sub-Mux	1(8 bit)	4	2.50	1(8 bit)	1	12.5
SPAR Digital Sub-Mux	1(8 bit)	4	1.25	1(8 bit)	1	12.5
Frame Sync	1(8 bit)	2	0.62	1(16 bit)	2	12.5
Spares	18(8 bit)	26	0.31	1(8 bit)	21	12.5
Totals	--	48		--	48	

Table 9-4

CONCEPT ANALYSIS - ORIENTED SECTION SUBMULTIPLEXER UTILIZATION

Oriented Section Sub-Multiplexer Inputs	Housekeeping and Standard Mode			Activity Mode		
	No. of Input Channels	No. of Sub-Mux Channels	Input Channel Sampling Rate (Samples/sec)	No. of Input Channels	No. of Sub-Mux Channels	Input Channel Sampling Rate (Samples/sec)
GSFC Secondary Data	4(16 bit)	8	0.052	4(16 bit)	8	0.52
Tell-Tale 8 bit Word Inputs						
Control S/S	4(8 bit)	4	0.052	4(8 bit)	4	0.52
Thermal S/S	1(8 bit)	1	0.052	1(8 bit)	1	0.52
Solar Monitor	1(8 bit)	1	0.052	1(8 bit)	1	0.52
Ordinance	1(8 bit)	1	0.052	1(8 bit)	1	0.52
Command S/S	5(8 bit)	5	0.052	5(8 bit)	5	0.52
Data S/S	2(8 bit)	2	0.052	2(8 bit)	2	0.52
Sub-Frame Sync	1(8 bit)	1	0.052	1(8 bit)	1	0.52
Spares	1(8 bit)	1	0.052	1(8 bit)	1	0.52
Totals	--	24		--	24	

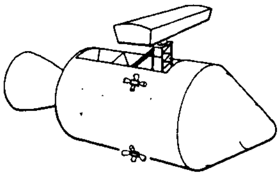


Table 9-5
CONCEPT ANALYSIS FOR SUBMULTIPLEXERS

Inputs	Bay Digital Sub-Mux			Bay Analog Sub-Mux			Oriented Section Digital Sub-Mux			Oriented Section Analog Sub-Mux		
	No. Inputs	Sampling Rate Samples/ Sec		No. Inputs	Sampling Rate Samples/ Sec		No. Inputs	Sampling Rate Samples/ Sec		No. Inputs	Sampling Rate Samples/ Sec	
		Std	Act		Std	Act		Std	Act		Std	Act
HCO Study Experiment Housekeeping Data							3	0.052	0.26	2	0.052	0.26
GSFC Study Experiment Housekeeping Data							3	0.052	0.26	2	0.052	0.26
NRL Study Experiment Housekeeping Data							3	0.052	0.26	2	0.052	0.26
HAO Study Experiment Housekeeping Data							3	0.052	0.26	2	0.052	0.26
Control S/S							5	0.052	0.26	6	0.52	0.26
Thermal Control S/S	1	0.052	0.52	2	0.052	0.52	20	0.052	0.26	20	0.052	0.26
Solar Monitor S/S	1	0.052	0.52	2	0.052	0.52	5	0.052	0.26	5	0.052	0.26
Power S/S	1	0.052	0.52	3	0.052	0.52						
Ordinance	1	0.052	0.52	1	0.052	0.52						
Command S/S	3	0.052	0.52	1	0.052	0.52				1	0.052	0.26
Data S/S	2	0.052	0.52	1	0.052	0.52				1	0.052	0.26
Apollo Interfaces	3	0.052	0.52	5	0.052	0.52						
Sub-Frame Syncs	1	0.052	0.52	1	0.052	0.52	1	0.052	0.26	1	0.052	0.26
Spares	11	0.052	0.52	8	0.052	0.52	5	0.052	0.26	6	0.052	0.26
Totals	24			24			48			48		

The PCM junction box constitutes the nerve center for the data subsystem and has fixed multiplex-format circuits using "and" gate addressing techniques quite conventional for spacecraft telemetry applications. One of the inputs to the "and" gates selects one of the two available formats in coordination with varying mission and experiment requirements (same command that selects overall data-rate).

Some additional circuits are required for this dual-format capability. However, for reliability considerations, the parts count should not be directly added on the negative side of the ledger since it is a parallel instead of an in-series function.

The design requirements cannot be finalized until experiment and other subsystem data requirements and interface agreements have been established.

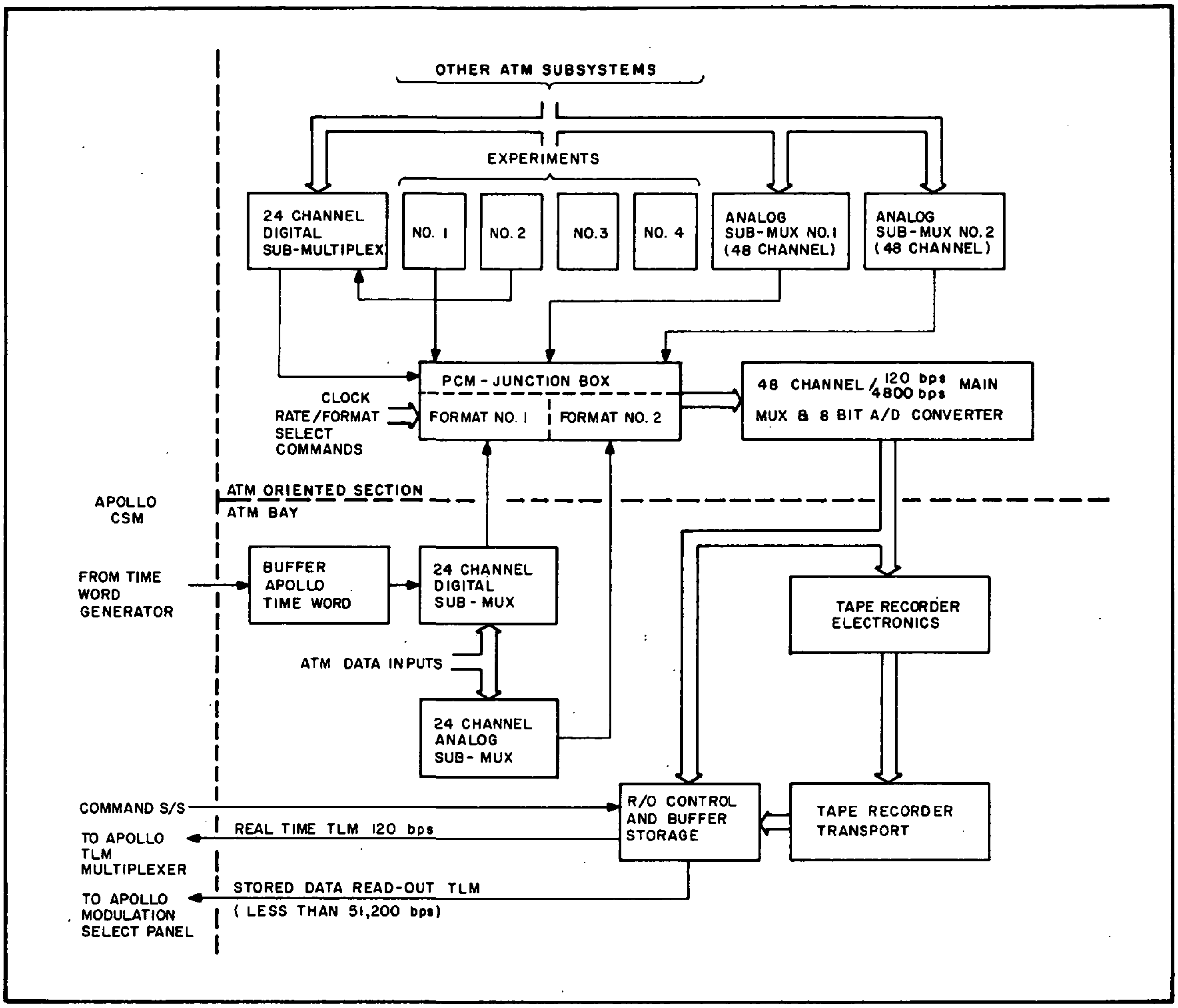
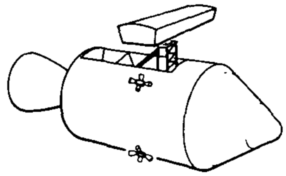


Fig. 9-1 ATM Data Subsystem Concept Block Diagram

9.4 OPTIONS

Data requirements thus far established, leading to the design concept given in Section 9.3, are subject to some change as experiment configurations are firmed and operations plans are better defined. Some further study during definition phase may lead to the justification of alternate design options not shown in Section 9.3. Some of these options considered during the study are discussed in the following paragraphs.



9.4.1 Adaptive Format

One option considered was an ATM operations concept in which multiple modes of data requirements are envisioned to best serve the scientific mission in accordance with such dynamic considerations as:

(1) Sun activity mode:

- Quiet sun
- Sun flare: first build-up, peak, decline

(2) Experiment:

- All photographic
- Part photographic - part electronic
- All electronic
- Fixed spectrum scan
- Variable spectrum scan
- Fixed counter overflow length
- Variable counter overflow length
- Flare priority

A particular experiment might be given top priority (maximum number of time-multiplexed channels and fastest telemetry clock rate) to fulfill its high-priority scientific mission during the fast build-up of a specified flare condition. As the sun's flare radiation charge rate declines, the system might go back to a more equal time-sharing basis with other experiments. On another occasion, another experiment might be given this priority consideration.

Figure 9-2 illustrates a concept for implementing this function that would take the place of the 48-channel multiplexer shown in Fig. 9-1. The concept of adaptive format is not unique. (See 1962-65 National Telemetry Conference papers.) However, the familiar systems being promoted for general spacecraft use are somewhat more complex and more expensive than required for a scientific mission such as ATM.

Another approach investigated during the study is shown in Fig. 9-3. In this case, one or more adaptive format memories may be selected alternately with the fixed format memory. In this case the additional adaptive format circuits are relatively simple, since the primary timing, encoding and frame-sync generation functions are performed by the conventional PCM multiplexer.

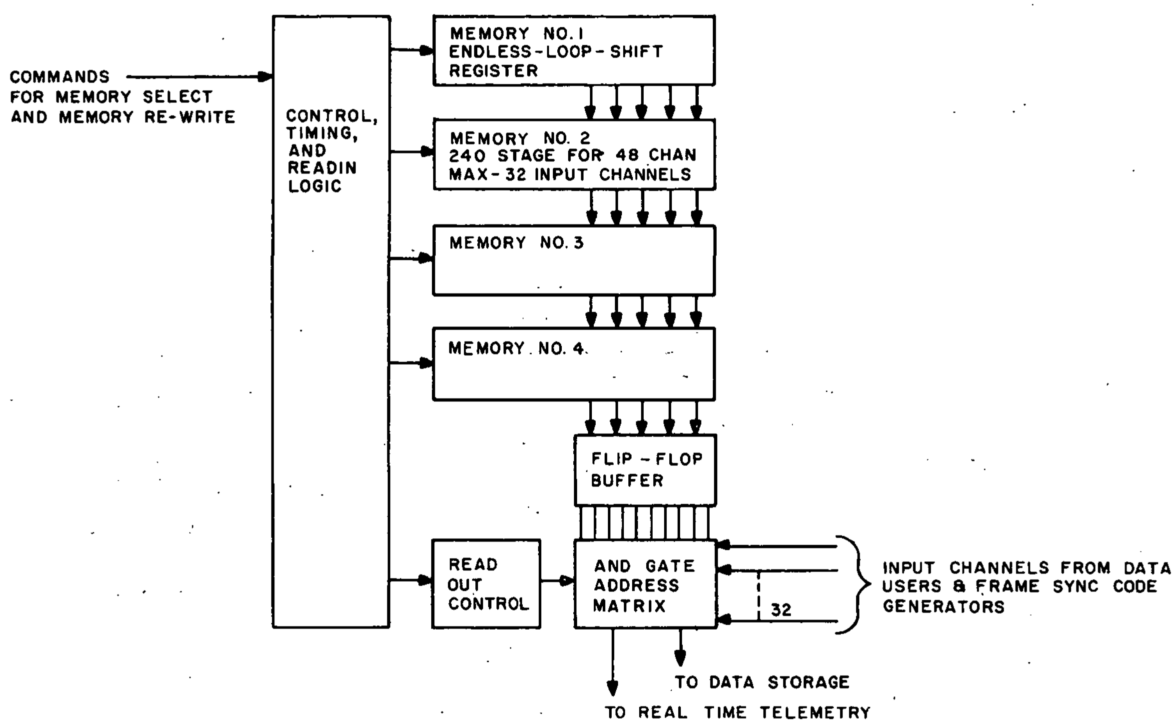
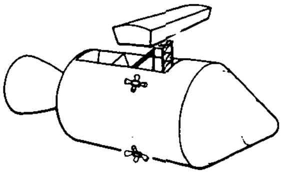


Fig. 9-2 Adaptive Format Multiplexer

With relation to the effect of adding the adaptive format supplement (Fig. 9-3), it should be understood that the single adaptive/fixed select switch must be considered a liability, and that the multiple component functions of multiplexing in some ways paralleling the services of the conventional multiplexer must be considered an asset (add to rather than detract from reliability).

9.4.2 EVA Data Retrieval

Another concept studied was to have the astronaut retrieve a tape recorder transport at the same time he is retrieving ATM film packs. One possible tape recorder transport considered was a multitrack machine with a weight of about 10 pounds and a volume of 400 cubic inches. Another approach used an 800 cubic-inch, 24 pound machine. Adequate data storage with the above mentioned machines did not appear to be a problem; that is, mission requirements can easily be met.



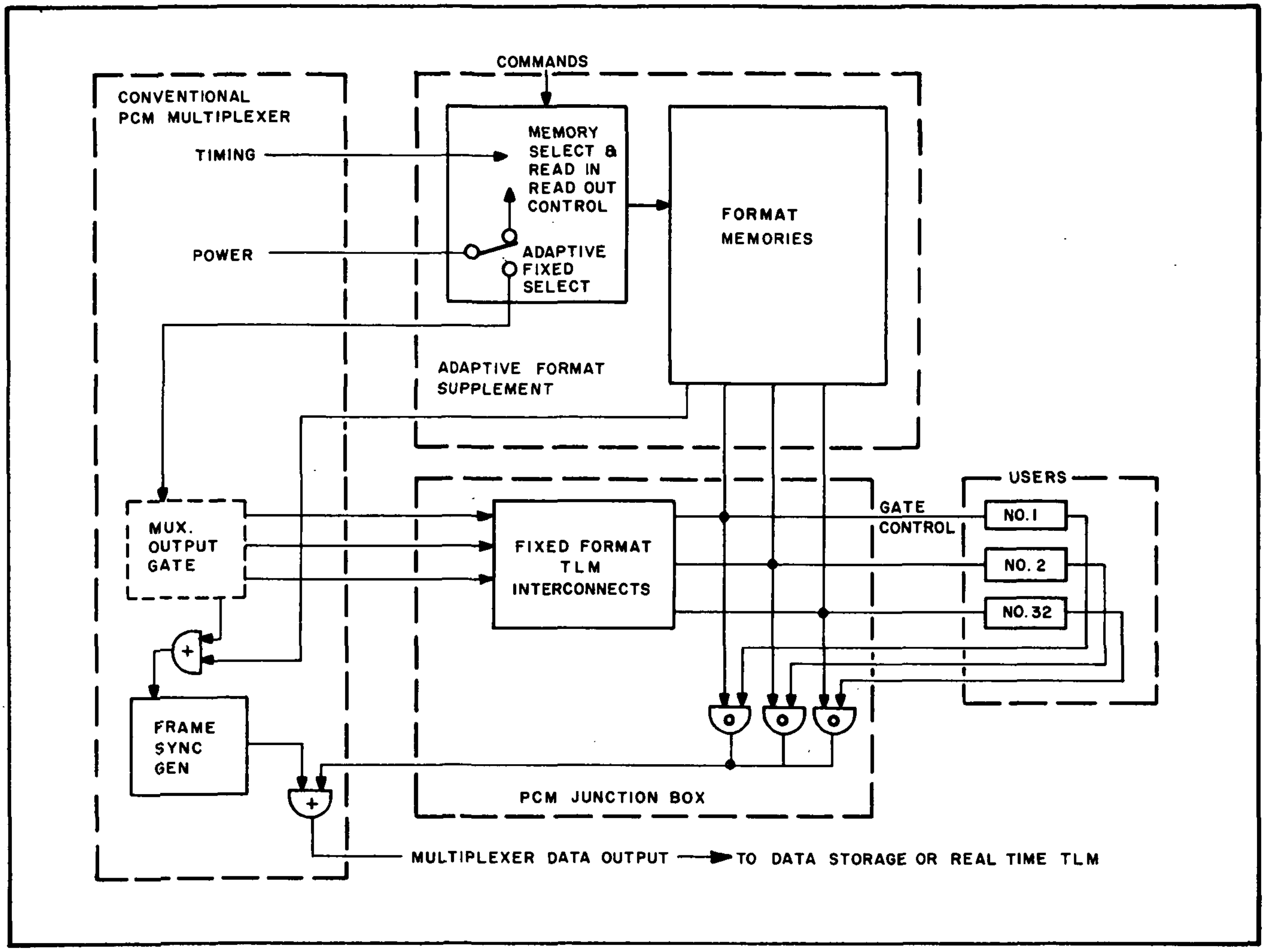


Fig. 9-3 Alternate Adaptive/Fixed Format Multiplexer

There may develop a necessity to use this approach if other Apollo telemetry commitments do not allow the minimal ATM interface requirements to be met. A functional need to parallel the two approaches could develop. There will be no major functional design problem in either case. However, this approach in place of the temporary store/playback does not appear to be functionally superior at this time.

SECTION 10

POWER AND ELECTRICAL DISTRIBUTION SUBSYSTEMS

Section 10
POWER AND ELECTRICAL DISTRIBUTION SUBSYSTEMS

10.1 GENERAL REQUIREMENTS

The ATM requires an energy source and an electrical distribution that fulfills the following functions:

- Provides 30 kilowatt-hours (KWH) at 27.5 $\pm$ 2.5 volts. (See Table 10-1.)
- Supports current surges up to 60 amperes while maintaining greater than 25 volts.
- Imparts a minimum mechanical loading on the directable spar assembly as a consequence of circuit connections having to pass over (or through) the gimbal assemblies.
- Incorporates grounding, shielding, electro-magnetic interference (EMI), and material selection techniques that are compatible with combined ATM/Apollo requirements.
- Complies with the Apollo restraints stated in Section 2.

Table 10-1
ATM POWER BUDGET

Subsystem	Watts	Watts Oriented Section Only	On-Time, Hours	Surge Current, Amps	Kilowatt Hours
Thermal	60	60	312	10	18.72
Pointing	18	18	100	40	1.80
Solar Monitor (Includes filter thermal control)	52	27	100	1.8	5.20
Command	4	2	100	0.5	0.40
Data	5	3	312	0.6	1.56
Structure					
Study Experiments					
HCO	25	25	19.6	0.9	0.49
NRL	20	20	12.6	0.7	0.25
GSFC	10	10	60.1	0.6	0.60
HAO	10	10	8.4	0.5	0.08
TOTAL	204	175			29.10
Max. Watts Steady (HCO, NRL, ON; GSFC and HAO OFF)					

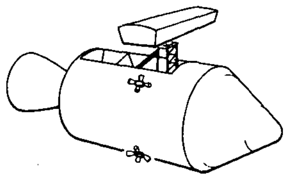
Note: Two kilowatt hours should be reserved for final experiment accommodation.
Note: Maximum surge current is 55.6.

10.2 DESIGN CONCEPT ANALYSIS

10.2.1 Energy Sources

The alternate energy source design concepts explored were:

- 100 percent use by ATM of CSM generated (fuel cell) power, with ATM supplying additional expendables (hydrogen and oxygen) for the Apollo fuel cell if acquired.



- 100 percent generation (or storage) of ATM power requirements within ATM design envelope (alternate batteries, solar arrays, and fuel cell sources considered).
- Partial use of CSM power supplemented by ATM power source (possibility of CSM using ATM power source as emergency back-up considered).

Figure 10-1 shows the relative effect of the ATM power load on the effective mission duration. Note that the mission duration would be reduced by approximately one-half day or 3.5 percent as a result of the ATM load.

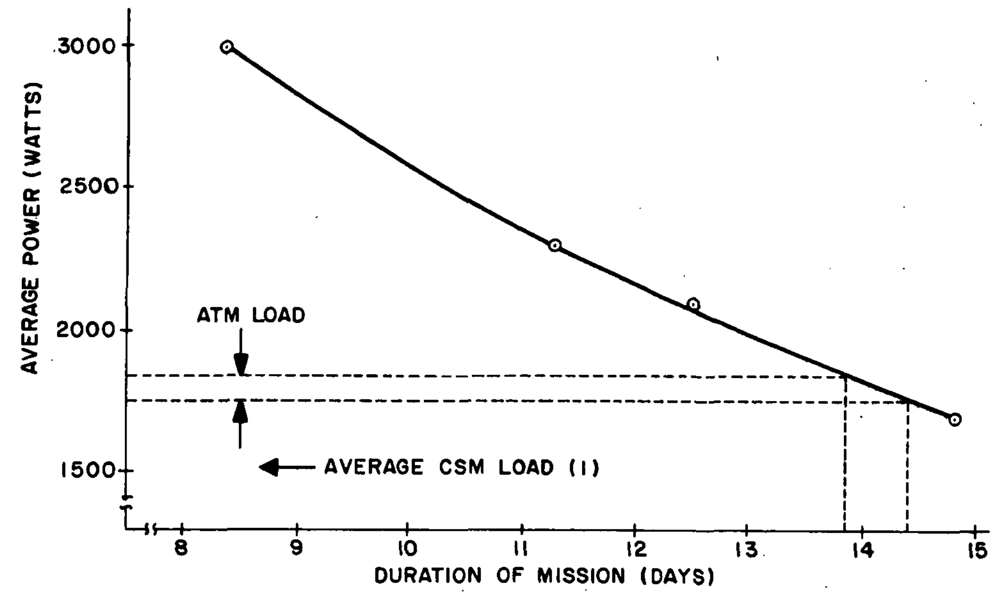


Fig. 10-1 ATM System Effect on Mission Time Using CSM Power

The added complexity and weight penalty incurred by incorporating a fully sustaining, independent ATM power source would be the result of retaining the 3.5 percent mission life capability.

As a primary conceptual consideration, the study concluded that the CSM could probably provide the ATM unregulated power requirements, and surge current capability. While complete data to justify this approach were not available during this study, it was concluded that this was the most overall efficient approach since:

- The CSM fuel cell would be expected to be far more efficient (due to size and long-term development support) than an especially sized ATM energy source. That is, added overall weight capability could be more efficiently used in supplying additional expendables for the CSM fuel cell than by adding additional fuel cells or alternate energy sources.
- The larger CSM fuel cell could more easily sustain the expected ATM surge currents.
- The CSM/ATM power interface could be effected in a simple reliable

way so as not to jeopardize the required reliability for either Apollo or ATM.

10.2.2 Electrical Distribution Across Gimbals

The primary design problem associated with this subsystem, that has been considered in this study, is associated with carrying power and signals across the gimbals. The use of either flexible cables or slip rings has been considered. Both of the methods appear feasible, but further details of the exact requirements of conductor size and number will have to be determined before either method is chosen.

10.3 DESIGN CONCEPTS

10.3.1 Source

Figure 10-2 shows a schematic of the power subsystem based on 100 percent use of the CSM energy source.

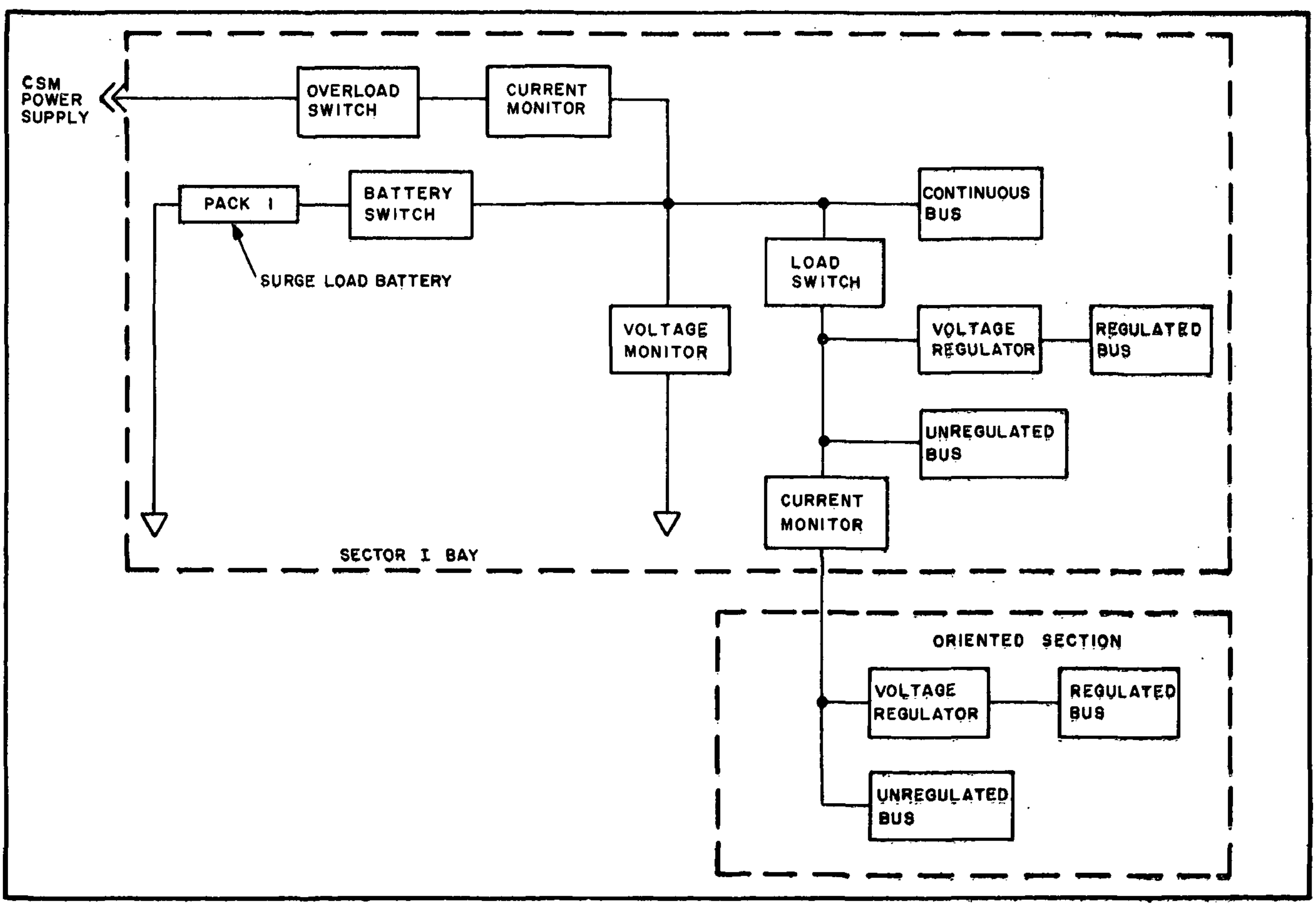
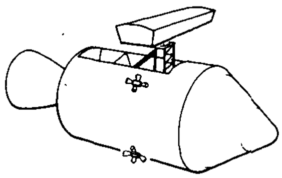


Fig. 10-2 Power Subsystem Block Diagram



The CSM can provide approximately 500 watts to ATM, which is sufficient to provide the ATM peak steady state loads. A secondary battery, to provide for the high transient loads occurring during acquisition, may be necessary. The necessity of such a battery will depend upon the allowed transient voltage that can be withstood by the ATM subsystems, and voltage which the CSM power system can provide under the surge loads.

If the addition of the ATM system loads on the CSM decreases the total life of the mission to an unsatifactory point, a portion of Sector I will be consigned for additional fuel cell tankage.

10.3.2 Circuit Protection

An overload switch is set to trip if the maximum surge load, that the CSM power supply can provide, is exceeded. A load switch is provided to drop, upon command, all the system loads except those on the continuous bus. In the event of an emergency, the surge load battery can be used to help power the CSM.

10.3.3 Power Conditioning

Direct-current regulated voltage (± 1 percent), as needed by the various subsystems, will be provided in the bay and in the oriented section. Assemblies requiring more precise regulation or higher than 20 volts will provide the additional regulation as part of the assembly.

10.3.4 Monitors

The following items will be monitored:

- Supply current
- Thermal heating current
- Main bus voltage
- Oriented section current
- Temperature of surge current battery

In the event that a monitor of the current to a particular subsystem is found to be essential to proper command decisions, the individual subsystem will also be provided with a current monitor.

10.3.5 Inter-Gimbal Wiring

Further study is needed to determine the single most optimum design approach for spanning the gimbal assemblies. Therefore, several alternate approaches will be analyzed until a more definite design decision can be made.

Flexible cables offer the advantage that no sliding contacts are required and, therefore, are less susceptible to failure and generation of noise. The disadvantage of flexible cables is the torque required to bend the cable and the increase in torque required as the cable is moved from its quiescent position. It is estimated that only eight ounce inches change in torque over a 60-degree travel can be tolerated without introducing excessive pointing errors or increasing power requirements. The increase in torque can be compensated somewhat by methods of coiling, which tend to make a constant-torque device.

In general, all methods will have some null position, but the design can be such that the change in torque is small. Whether the cable can be made to fulfill the allowable torque requirements will have to be determined as a result of further design analysis and a determination of the conductor requirements. A further consideration must be the temperature of the cable. Passive control may be required to maintain flexibility.

Slip rings can be used to solve the problem of wiring over the gimbals. They have been used for similar space applications. Noise generated in slip rings can cause problems with low level signals, but there is anticipated to be no low level signals except for the solar monitor signals. Tests would have to be run to determine the feasibility of carrying the high frequency signal of the solar monitor through a slip ring. Slip rings have the advantage of presentingessentially a constant torque to the system, but have disadvantages in complicating the mechanical structure for placement of the slip rings.

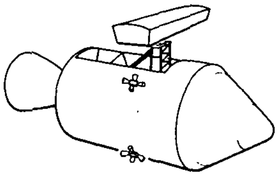
Depending on the final conductor requirements, it may turn out that a combination of flexible cable and slip rings is most advantageous; but it is recommended that first consideration be given to designing a flexible cable to wire over the gimbals.

Further study is also required on other problems associated with electrical distribution, such as grounding methods, shielding methods, EMI requirements, and material selection.

The requirements for these items have to be compatible with CSM requirements, and be consistent with acceptable spacecraft design practices. The number of junction boxes needs to be determined as required by the various subsystems involved. It is anticipated now that a minimum of three junction boxes will be required, one each at the bottom and top of the extension mechanism, and one in the oriented section. The electrical distribution will use junction boxes to eliminate extraneous switching problems in the subsystem design. The final design of junction boxes must be organized to allow easy check out and accessibility of components for changes. This effort will require consideration of the testing connections required by AGE.

10.4 ALTERNATE POWER SOURCE DESIGN CONCEPTS

The design concept presented in Section 10.3 is based on the assumption that power can be supplied from the Apollo CSM. The various alternate methods for supplying the ATM system with power have been analyzed and are discussed in the following sections.



10.4.1 Silver-Zinc Battery

If power cannot be obtained from the CSM fuel cells, the approach that has been developed to satisfy the requirements of Section 10.1, consists of a self-contained power supply in the form of a silver-zinc battery. The battery is comprised of sufficient packs connected in parallel to provide the required ampere-hour capacity, and sufficient cells are connected in series within each pack to provide the necessary voltage. The battery has a 35 kilowatt-hour and a 1250 ampere-hour capacity. The weight of the battery and its supporting structure is less than 600 pounds.

The small amount of oxygen and hydrogen liberated during discharge is vented to the atmosphere through a filter, which traps any potassium hydroxide. The heat generated by the battery during discharge, if not dissipated, raises the temperature of the battery a maximum of 10°F. Therefore, the battery will be thermally isolated from its surroundings by means of coatings with low solar absorptivity and low emissivity. At the time of launch, the battery will be at a temperature from 70°F to 80°F.

Silver-zinc cells are available as a flight qualified component. The characteristics of silver-zinc cells vary depending upon the cell construction, which is altered to obtain optimum performance for a particular set of characteristics. In order to optimize the watt-hour per pound ratio, the separator thickness and amount of electrolyte is kept to a minimum. An extremely thin plate separator can reduce the wet storage life of the cell to a few hours, and reduction in the amount of electrolyte causes a high impedance at the end of discharge.

To obtain a long cycle life and maintain charge for a long period after activation, the plate separator thickness is increased. This increases the size and the impedance of the cell.

The characteristics of a cell also depend upon the manner in which the cell is used. The ampere-hour capacity increases with low discharge rates and decreases with repetitive cycling. The wetstand life decreases with high storage temperature. The impedance increases with low temperature.

For purposes of the ATM power source, the characteristics of a silver-zinc battery pack are given in Table 10-2.

Cells meeting the above requirements are available as off-the-shelf items. However, in order to obtain cells that have been already qualified for space flight usage, it may be necessary to relax some of the requirements.

10.4.2 Fuel Cells

Analysis of fuel cells to supply the ATM requirements has indicated that a fuel cell could be compatible. The following fuel cell features must be considered:

- Tankage for reactants
- Heat exchanger for preheating the reactants and cooling the water
- Radiation of heat from fuel cells
- Collection and cooling of the waste water for periodic dumping

A fuel cell capable of supplying 120 watts and 30 kilowatt-hours would weigh about 250 pounds and would require about eight cubic feet.

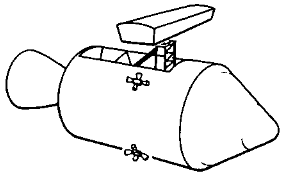
Adverse properties of a fuel cell system are the handling of cryogenic fuel and the water dump, which would add to the ATM contamination control problems.

Table 10-2
CHARACTERISTICS OF SILVER ZINC BATTERIES

Operating temperature during discharge	60 to 100°C
Open circuit voltage	1.86 volts
Plateau voltage at c/200 rate	1.50 to 1.57 volts
Watt-hour capacity per cubic inch at c/200 rate	6
Watt-hour capacity per pound at c/200	80 watt-hour per pound minimum
c/20 surge for 40 seconds at end of discharge with voltage remaining above	1.42 volts
Cycle life for flight cells	Three 25 percent discharges One 60 percent discharge One 90 percent discharge
Cycle life for test cells	Six 60 percent discharges
Dry storage life	2 years
Wet stand life (cell may self-discharge 25 percent during stand period, but must then be capable of being recharged and meeting watt-hour and surge capacity requirements)	3 months at 70° to 80°F, plus 6 months at 30°to 40°F
Gassing during discharge	3 litres of O_2 and H_2 maximum per 100 amp-hr capacity
Heat dissipation during discharge	0.04 BTU/amp-hr maximum

10.4.3 Battery and Furnace

Since the major factor in the ATM power budget is the heater load of the thermal control subsystem, a furnace has been considered to provide the heat requirement. In this furnace, combustion of fuels would heat a liquid, which would circulate through heaters in the oriented section. The combustion products of such a furnace could cause an additional contamination problem, and developing pneumatic lines to cross the gimbals would be prohibitive (the weight and size of the furnace and associated fuel tanks would require that they be located on the cradle). The momentum of the circulating fluid could affect the pointing control subsystem design. For such a system the balance of the power requirement would be supplied by batteries.



10.4.4 Nuclear-Thermionic Power Supply

Nuclear-thermionic power supplies were considered, but the possibility of radiation leakage affecting the SM fuel gauge system restricts the ATM from using them. The low efficiency of the thermoelectric energy conversion would require a radiator large enough to dissipate about 6000 watts, which would be prohibitive for the ATM concept.

10.4.5 Solar Array

A solar array to supply a maximum load of 118 watts would require an extensive area. The array would have to be folded in Sector I prior to extension of the oriented section, and then articulated perpendicular to the solar vector. Design of such an array would be extensive and would increase the problem of balancing the oriented section. The CSM attitudes would be restricted if battery charging was required when the ATM was not solar oriented.

SECTION 11

GROUND SUPPORT EQUIPMENT

Section 11
GROUND SUPPORT EQUIPMENT

In order to evaluate design parameters and verify proper performance of the ATM, extensive ground support equipment(GSE)will be required. Contained in this section are a list and brief description of the major items anticipated for aerospace ground equipment (AGE) and test equipment (TE). Aerospace ground equipment is defined as that ground equipment necessary to support system integration and checkout of the ATM with the Apollo spacecraft. Test equipment is defined as that equipment necessary to support all other ATM ground testing.

11.1 AGE LIST

11.1.1 Checkout Console and Accessories

- Checkout Console
- Pointing Subsystem Source Hood

11.1.2 Major ATM Handling Equipment

- ATM Handling Fixture
- ATM Transportation Fixture

11.2 TEST EQUIPMENT LIST

11.2.1 Major Electronic Test Equipment

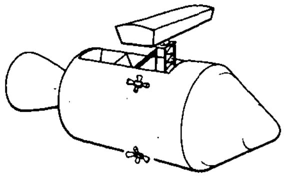
- Checkout Console Support Equipment
- Auxiliary Remote Data Handling Equipment
- Experiment Simulator Set
- Spar Control and Monitor Unit
- ATM Simulator
- Solar Simulator

11.2.2 Major Mechanical Test Equipment

- Apollo Motion Simulator
- Thermal Vacuum Chamber Fixtures
- ATM Shipping Container

11.3 AGE DESCRIPTION

Described in this section is the major equipment required for integration and test of the ATM with the Apollo CSM.



11.3.1 Checkout Console and Accessories

The checkout console and its various pieces of support equipment will be interconnected in several ways to provide control of ATM and monitor capability during the ground testing. Stimuli are supplied to the ATM through its own control unit by the pointing system and vidicon stimulators (source hood), the solar simulator, gyro torquers, experiment simulators, etc., all under the control of the console. Responses are monitored directly via displays and meters on the ATM control unit and the console, and by the auxiliary data handling equipment rack or the computer. During troubleshooting tests, stimuli and responses may also be controlled and monitored by special test equipment contained in the "checkout console support equipment rack."

The operator will have full manual control over the ATM at all times, with the auxiliary data handling equipment rack or computer acting only as data reduction centers; however, if desired, the console can be designed for automatic checkout by providing computer control capability. Automatic control capability would allow NASA acceptance checkout equipment (ACE) to conduct a full automated checkout of the ATM as part of the integrated Apollo test program.

The deliverable portion of the checkout console and its auxiliary equipment consists of only two items; the remaining test equipment is unnecessary for ATM system tests at NASA sites. This is partly based on the assumption that experiment stimulation can be effected artificially by built-in self-test circuits that the console can activate. The two major AGE items are listed and described below:

- Checkout Console
- Pointing Subsystem Source Hood

Checkout Console (See Fig. 11-1.) The console is of standard construction, weighing about 350 pounds, and provided with casters for portability. The control unit of the ATM under test is mounted at the left end of the console, exceptions being EMI, some vacuum tests, and Apollo integration tests. The remaining console face contains power supplies, miscellaneous circuitry, and other items. The console interfaces with its accessories through a test cable.

The console provides 28V DC power to the ATM during all test phases except where battery or Apollo power is used. In addition, the console provides signals to the ATM normally generated in the Apollo CM.

The operator will have manual command of all ATM subsystems through the ATM control unit and the console controls that interface with the various stimulators and simulators. In addition, the operator will be able to manually program the ATM data handling commutators to any desired signal for console display.

ATM responses to commands will be monitored with data displays on the ATM control unit, or with the data reduction accessory equipment, and by test points provided

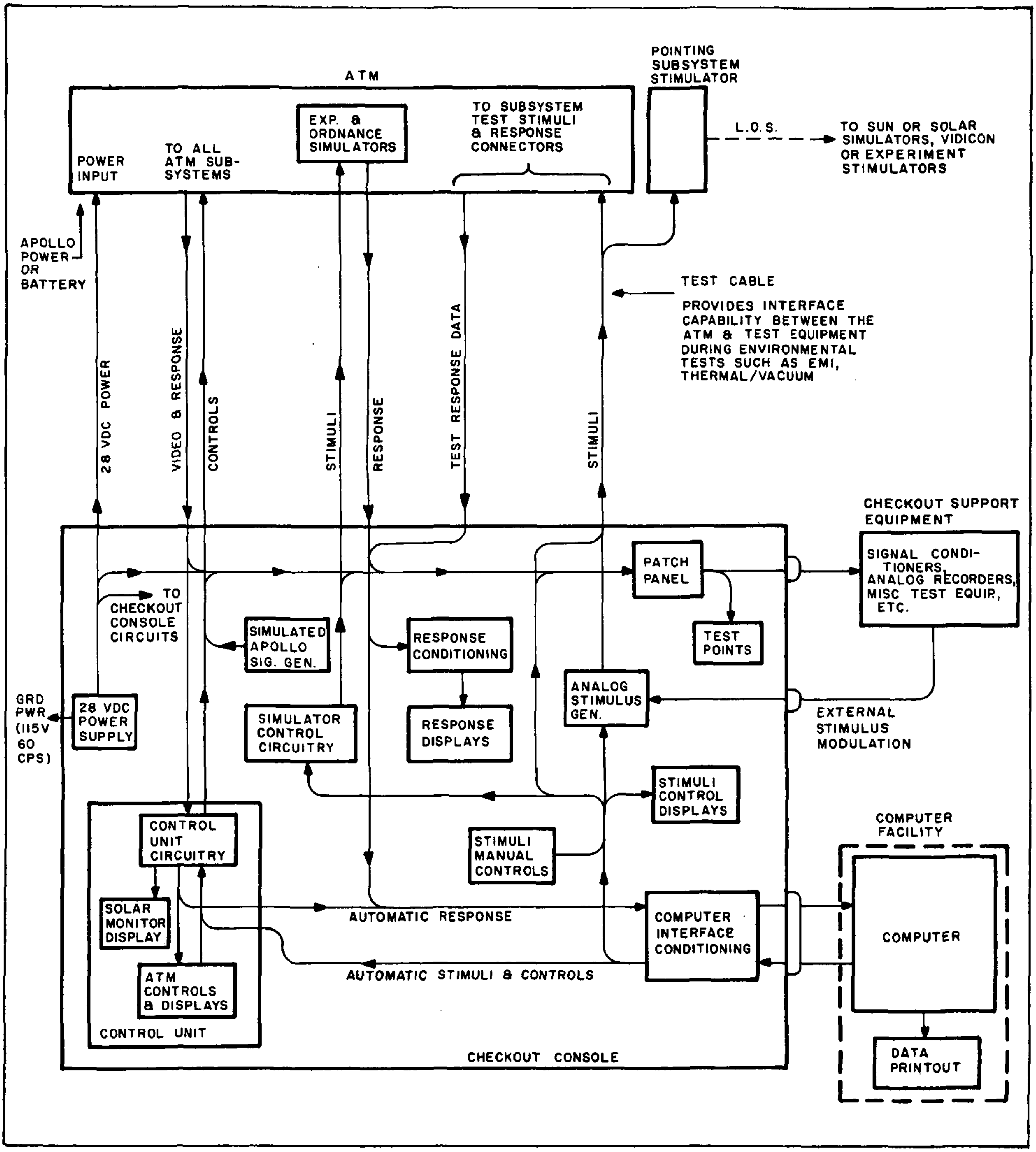
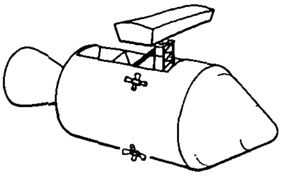


Fig. 11-1 Checkout Console Block Diagram



for troubleshooting with the checkout console supporting equipment.

If automated checkout capability is desired, the console will be provided with interface circuitry that will allow a computer to command all necessary console and control unit stimuli and controls. The various displays on the control unit and console will continue to operate during automated tests, providing a running indication of the test progress. ATM responses will be monitored by the computer on a "go-no-go" basis, a fault resulting in an automatic test hold. All data will be recorded by the computer printout system.

Fault isolation capability down to the "blackbox" level can be provided by the console and its support equipment if the required test leads will not compromise ATM performance.

Pointing Subsystem Source Hood. This device is used to stimulate the pointing control and solar monitor subsystem when it is undesirable to use either the sun or the solar stimulator. The unit attaches to the front of the ATM oriented section. The weight of the unit will be a minimum, and a suitable counterbalance installed at the opposite end of the oriented section to maintain system balance during testing. The source hood will be operated remotely from the checkout console and will contain simple illuminating devices capable of stimulating the fine, coarse, and target sensors and the solar monitor telescope-vidicon. The servo system will be tested in an open-loop mode. The design will permit the use of this unit in well illuminated areas without external light interference.

11.3.2 ATM Handling Equipment

The major handling equipment consists of the following two pieces of associated equipment:

- ATM Handling Fixture
- ATM Transportation Fixture

The handling fixture is used in conjunction with the facility hoisting equipment to handle the ATM in any of the various test set-ups and to install it in the CSM. This fixture will be capable of lifting the ATM from a horizontal position and installing it in a horizontal, vertical or sideways mode to provide installation capability in vertical test fixtures or an erected Apollo CSM. Suitable shock absorbing devices will be provided to prevent any possible damage to the ATM.

It will probably be adequate to design the fixture to work with a stowed ATM configuration only; however, further design investigations may indicate that handling capability for an extended ATM must also be provided.

The transportation fixture is a specialized dolly used to transport the ATM to its various test set-ups within a given area, i.e., where an ATM shipping container is not required. The transporter must be physically compatible with most of the test facilities, exceptions being the vacuum chamber, shake table, etc. Since this unit will also be used as an ATM support fixture in tests where extreme rigidity (such as a seismic mass) is not required, the transporter will be designed to be stable for any possible ATM position. In addition, the transporter must be provided with shock absorbing devices to prevent damage to the ATM or experiments during transportation. The transporter will be provided with screwdown foot plates to level and stabilize the device during tests. A 30 degree inclined ramp will allow the transporter to be used during some solar pointing tests.

11.4 TEST EQUIPMENT DESCRIPTION

Described in this section are the major pieces of equipment required for subsystem tests and ATM systems tests prior to integration with Apollo.

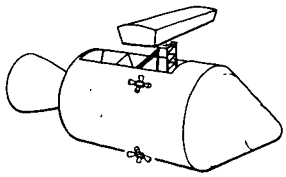
11.4.1 Major Electronic Test Equipment

Checkout Console Supporting Equipment. This unit consists of one or two racks of miscellaneous support equipment such as oscilloscopes, digital voltmeters, function generators, counters, etc. The equipment will be used primarily in conjunction with the checkout console for troubleshooting and design evaluation of the ATM and will be portable. All voltage requirements will be furnished within the rack, using facility power.

Auxiliary Remote Data Handling Equipment. This portable unit consists of one rack of equipment that will be used to reduce ATM digital data. The device will normally work in conjunction with the checkout console but will also have the capability of interfacing directly with the ATM data subsystem. Remote selection and readout of individual data channels will be possible through ATM commutator controls on the equipment rack, and display lights and meters. The data will also be recorded via tape deck or digital printer. The rack will be capable of performing all its functions using facility power only.

Experiment Simulators. Experiment simulators will be used for ATM tests when the experiments are not available. These simulators will consist of individual units having the same size, mass, c.g. and installation fittings as the experiments. These devices will contain sufficient electronics, working in conjunction with the checkout console, to provide an electrical interface to the ATM which simulates all experiment internal subsystems. The simulators can be furnished with dummy film magazines to allow astronaut training in film loading and unloading techniques.

Oriented Section Control and Monitor Unit. This piece of portable equipment provides the capability of operating the oriented section subsystems only, i.e., the cradle portion of the ATM is not required. This allows parallel testing of the oriented section and cradle assemblies. The unit consists of a simple control and display panel, necessary circuitry, and a power supply.



ATM Simulator. The ATM simulator will be capable of operating the experiments separate from the ATM in all modes and under all conditions during functional tests. The device will be portable and will have manual controls and simple readouts as well as a means of reducing and recording data either by digital printer or tape deck. To provide flexibility, a rack mounted oscilloscope and patch panel will also be provided. It will be capable of interfacing through the checkout console to the computer to enable processing of experiment data during checkout tests.

Solar Simulator. This unit delivers a beam of light of that spectral range to which the ATM pointing control subsystem is sensitive. The cross-sectional area of the beam must be large enough to adequately cover the fine sun sensors with the simulator located at some convenient distance from the ATM. A large beam area allows greater linear displacement of the ATM during ATM tracking tests utilizing the Apollo motion simulator (section 11.4.2). The light intensity is approximately one-third of the solar equivalent.

11.4.2 Major Mechanical Test Equipment

Apollo Motion Simulator. This device is used to simulate the normal rotational motions of the Apollo CSM in orbit. The ATM will be mounted on this unit and the pitch and yaw tracking capability and roll stability monitored with a tracking auto-collimator set-up. The fine eyes will monitor the solar simulator during these tests with the ATM roll axis oriented in the earth's east/west plane to eliminate roll gyro precessing due to the earth's spin. If this orientation is not possible, a small gyro torque signal, equivalent to normal earth rates, may be fed to the roll gyro to cancel precessing action as an alternate approach.

The actual CSM motions existing while the ATM is in the fine pointing mode are relatively slow, not exceeding a steady 0.05 degrees per second with maximum perturbations of several arc seconds due to astronaut motion. The fixture, then, need only provide a constant velocity type of motion about the three axes of ATM, variable over a range of zero to 0.1 degrees per second. The mechanical imperfections of the motion table itself will probably provide acceleration transients at least equalling those of the CSM. In addition, the fixture design should provide simulated CSM motion throughout the entire gimbal ranges of the ATM, i.e., ±30 degree pitch, ±25 degree roll and ±170 degree in yaw, in order to verify proper operation for any usable position of the ATM gimbals.

The table will be designed to carry only the ATM oriented section rather than the entire unit complete with the cradle and extension mechanism. This will allow the motion simulator gimbals to be as close as possible to the ATM gimbals, resulting in the following advantages: (1) the design of the table will be simpler due to the reduced mass it must move; (2) the linear displacement of the oriented section in the horizontal and vertical mode, due to table motion, would be relatively small (this will provide a much larger motion range before the fine control sensors are driven out of the solar simulator beam); finally, (3) the testing will be more realistic as unnatural flexing in the ATM extension mechanism due to the earth's gravitational field, will be eliminated. The oriented section will be left electrically connected to the remainder of the ATM during these tests to verify integrated system operation.

If for some reason it becomes necessary to eliminate all linear displacement of the oriented section during these motion tests, the fixture will be designed with its three gimbal axes in exact alignment with the corresponding ATM axes. This, however, would make the fixture quite large, and interference problems might develop.

At present, it would appear best to use a gimbaled structure equipped with ball bearings rather than an air bearing approach to allow operation of the motion simulator in a vacuum chamber. The large area of the ATM thermal shield may effect the pointing accuracy during tests in the normal atmosphere due to convective air currents. The problem would be further aggravated by bleed air from an air bearing device. The motion simulator drive system and bearings will be specially designed for vaccuum operation.

Thermal Vacuum Chamber Fixtures. These fixtures are used to support the ATM system in the vacuum test facility. It appears to be impractical to use the various transporters for this operation, due to the risk of these devices introducing contamination. It is assumed at this time that the chamber, which will be used, will be accessible from the top only, consequently the thermal vacuum fixtures and their loads must be lowered in place. The position of the ATM system during thermal vacuum tests is unknown at present, but will be dictated largely by the chamber design.

ATM Shipping Container. This is a special container for shipping or storage of the ATM system, which will be a sealed, metal container, provided with pressure, temperature and shock recorders during transportation. The container can be pressurized as required with dry nitrogen. The recorder data will be evaluated at the receiving point to determine the trip environment. The device will be equipped with wheels and lifting pads for transportation and handling.

SECTION 12

EXTENDED APPLICATIONS

Section 12
EXTENDED APPLICATIONS

The conceptual design presented in this study has been developed keeping in mind versatility to extended missions. A general definition of ATM might be that it is a connecting link between scientific instruments and a space vehicle, designed to provide stabilization of the instruments while allowing relatively gross motion of the vehicle. Both parts of this definition are important when coupled with the idea of manned operation and long orbital life of the ATM/vehicle system. To "provide stabilization of instruments" means that ATM is a research platform such that, by subsequent rendezvous, various experiments can be removed from ATM and others of different capability substituted, or the experiments can be reactivated. To "allow relatively gross motions of the vehicle" greatly reduces the requirements for attitude control of the vehicle itself and in particular allows the vehicle motions resulting from man motion within the vehicle.

Thus, the ATM is an important link in the overall concept of a manned orbiting observatory. It differs from the concept of orienting an entire laboratory to high precision at a particular target and is in fact much more versatile. For example, the vehicle laboratory can be crudely oriented to the sun for thermal control or, if desired, spin-stabilized for provision of artificial gravity while, through the use of ATM, maintaining the capability to observe targets with high precision.

12.1 GROWTH CAPABILITIES

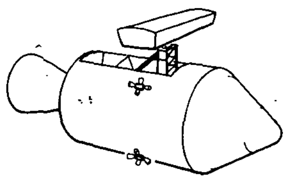
The factors that define capability of ATM can be listed as follows:

(1) Size of experiments that can be accommodated
(2) Weight of experiments that can be accommodated
(3) Accuracy of stabilization
(4) Targets that can be accommodated
(5) Useful lifetime
(6) Operational features
(7) Data handling capability
(8) Power supplied

Items (7) and (8) are considered to be supporting subsystems and are neither constrained by, nor do they constrain, the ATM concept nor configuration. Either of these subsystems can be physically accommodated in the vehicle and be increased in size and capability as desired. Items (1) through (6) are dependent on the ATM system concept. The growth capabilities of each of these factors is discussed in the following sections.

12.1.1 Size of Experiments

The inside-out gimbal configuration of ATM allows virtually unlimited experiment size. Size in general, however, will be associated with weight, and it is likely that constraints from cg location and structural compliance (resonant coupling of



experiments structure with the pointing control subsystem are limiting factors. A guess as to limitations of size is that telescopes 10 feet in diameter with tube lengths of 30 feet can be accommodated as far as in-orbit ATM operation is concerned.

Considering Sector I of the Apollo Service Module as the constraining envelope for early missions it is feasible to accommodate two 20-inch telescopes, 11.5 feet long, as shown in Fig. 12-1. Accommodation of a 30-to-40 inch telescope is also feasible assuming that a bulbous cover over Sector I can be used.

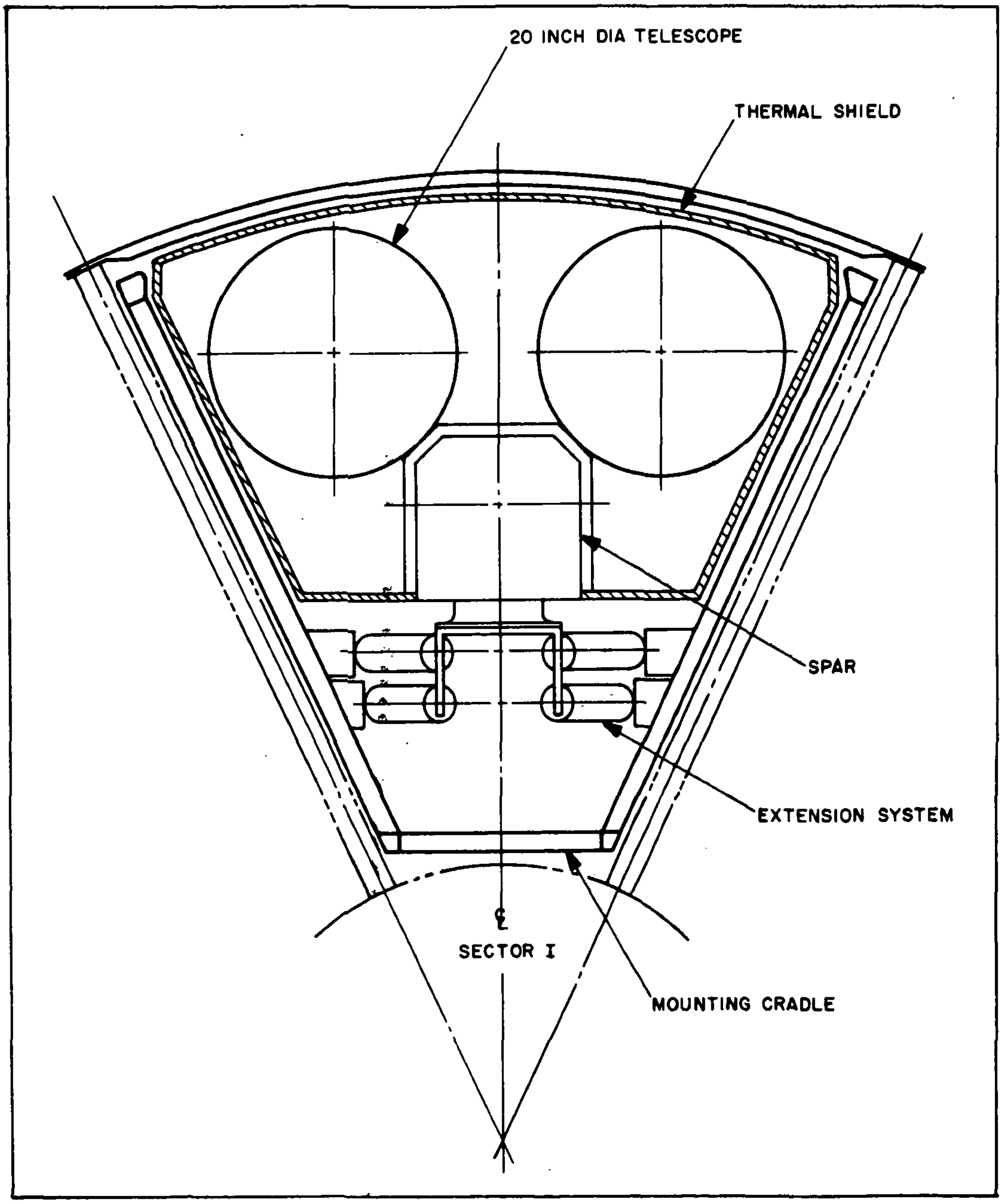


Fig. 12-1 ATM System Utilizing 20 Inch Diameter Telescope

12.1.2 Weight of Experiments

For early missions it is vital to conduct ground test operations of the integrated ATM/experiments package prior to launch. Therefore, a weight limitation is imposed by virtue of the load-carrying capacity of the gimbal bearings. The criterion is to support the ATM/experiments in a 1-g field without exceeding the rating of the bearings. In sizing the bearings, it must also be kept in mind that minimum size is desirable in order to minimize friction levels. Based on these considerations, the present ATM conceptual configuration will allow a design that can support about 4,000 pounds of experiments. This compares with the anticipated 750 pounds for initial flights.

For later flights on a space station where the operation of changing experiments is feasible, the 1-g test criterion is not applicable, and even greater weights can be accommodated; the ultimate limitation is caused by constraints on cg position and structural compliance, as mentioned in Section 12.1.1 above.

12.1.3 Accuracy

The conceptual design of ATM includes the short-term stabilization accuracy of about five arc seconds. It is likely that refinement of pointing control subsystem sensors and electronics can reduce this to about one to two arc seconds. To achieve stabilization better than this, the best approach is to build into the scientific instrument an image-motion-compensation control system which would then control an element of the telescope. Stabilization to a fraction of an arc second could then be achieved. The concept of deriving a fine control error signal from the experiment, which is used to drive the primary control system, is not workable with ATM and in general is bad because of problems with structural resonances.

12.1.4 Targets

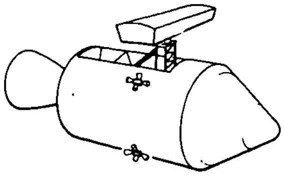
ATM is adaptable to any target that can be sensed and tracked within the gimbal freedom. The pointing control subsystem is very simple and is readily adaptable to the use of star trackers, horizon scanners, or whatever is appropriate.

12.1.5 Lifetime

Assuming adequate support from a power subsystem, the inherent life of ATM is probably limited by basic electronic system reliability and/or lubrication lifetime of the gimbal system. Because of simplicity of design, and based on extensive test and operational experience with BBRC Vac-Kote vacuum lubricants, this can be expected to be at least three years assuming "hands-off" operation. With the aid of manned maintenance, 10 years or longer is not unreasonable.

12.1.6 Operational Features

These can be so varied that a specific requirement would have to be analyzed for a meaningful statement. Continous 360 degree travel in yaw (azimuth) can be readily



accomplished by placing all electrical connections across the yaw gimbal on slip rings, thus allowing ATM to fit a variety of other missions requiring this freedom. A feature of possible interest involving basic Apollo systems is a video link to earth.

The ATM solar monitor subsystem can be designed to make its video signal compatible with the Apollo video link to the surface. This would permit transmission of the boresight images to the surface for monitoring by the ground scientists when in radio communication with the surface. Real-time monitoring of the ATM telescope images might enhance the scientific investigations by allowing the ground scientists to advise the astronaut observers about specific solar features to be investigated.

12.2 CONCLUSIONS

The ATM concept presented in this report will result in hardware that is versatile and useful for many different missions with little or no redesign of the basic gimbaled platform. The design is suitable for use in the future on vehicles other than the Apollo SM, and will be an important development for use with a manned orbiting observatory.

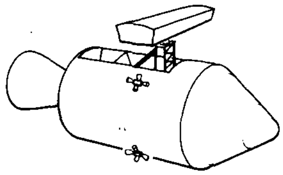
SECTION 13

GLOSSARY

Section 13
GLOSSARY

13.1 ABBREVIATIONS

ACE	Acceptance Checkout Equipment
AGE	Aerospace Ground Equipment
ASA	American Standards Association
ATM	Apollo Telescope Mount
ATU	Astronaut Training Unit
BBRC	Ball Brothers Research Corporation
CRT	Cathode Ray Tube
cg	Center of Gravity
CSM	Command Service Module
CPS	Cycle Per Second
db	Decibels
EFL	Effective Focal Length
EMI	Electromagnetic Interference
EVA	Extra-Vehicular Activity
GSFC	Goddard Space Flight Center
GSE	Ground Support Equipment
G&N	Guidance and Navigation
G&NC	Guidance and Navigation Computer
h.o.e.	Hard Over Engine
HCO	Harvard College Observatory
HAO	High Altitude Observatory
HF	High Frequency
KSC	Kennedy Space Center
L.H.	Left Hand
LOS	Line of Sight
LES	Lunar Excursion Module
LMS	Lunar Mapping and Survey
MSC	Manned Spacecraft Center
MSOB	Manned Space Operation Building
MSS	Missile Service Structure
MOI	Moment of Inertia
MUX	Multiplex
NASA	National Aeronautics and Space Administration
NRL	Naval Research Laboratory
NAA	North American Aviation, Inc.
OSO	Orbiting Solar Observatory
PCS	Pointing Control Subsystem
PSI	Pounds Per Square Inch
PCM	Pulse Code Modulator
PPM	Pulse Position Modulator



RF	Radio Frequency
RCS	Reaction Control System
R.H.	Right Hand
RMS	Root Mean Square
RSS	Root Sum Square
SM	Service Module
SPS	Service Propulsion System
SCS	Spacecraft Control System
SWR	Standing Wave Ratio
TE	Test Equipment
UV	Ultraviolet
STA	Vehicle Station Number
VHF	Very High Frequency

13.2 NOMENCLATURE

ALBEDO	Sunlight reflected by the earth.
ANGSTROM (Å)	A unit of length, used chiefly in expressing short wavelengths. Ten billion Angstroms equal one meter.
APOLLO BLOCK II	This term refers to a specific configuration of the Apollo spacecraft.
BIREFRINGENT FILTER	A filter comprised of polarizing plates and quartz crystals for isolating narrow spectral transmission bands.
CALCIUM K (CaK)	State of ionization of the calcium atom that is enhanced by flare activity. (λ = 3934Å)
CATADIOPTRIC	An optical system utilizing a lens-mirror combination.
CHROMOSPHERE	A thin layer of relatively transparent gases between the photosphere and the corona of the sun.
CORONA	The faintly luminous outer envelope of the solar atmosphere.
DETENT	Stop or checking device associated with the control sticker.
EXPERIMENT	The word experiment as used herein pertains to the scientific instruments as a system. It includes the instruments plus the operation and testing of the instruments, etc.
FLEXPRINT	Flexible conductors between two systems.
FIBRILLES	A type of hydrogen chromospheric structure associated and observed with solar flares.
GAMMA	The slope of the transfer characteristic of a television system when plotted on log-log paper.
HYDROGEN ALPHA (Hα)	First line in the Balmer series of neutral hydrogen (λ = 6563Å).
INSTRUMENT	The word instrument as used herein is the physical hardware used for scientific investigation.
KINESCOPE	Cathode ray tube in which electrical signals are transmitted into a visible picture on a luminescent screen.
LiOH CANISTER	Lithium hydroxide canister contained in the Apollo spacecraft.
MONOCHROMATIC	Pertaining to or consisting of a single color, radiation, or a single wavelength.

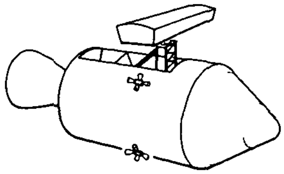
MOTTLES	Spots or blotches appearing on the sun as viewed in calcium light. Mottles is used to describe visible structures instead of granulation.
PHOSPHOR LAYER	Layer of luminescent material, applied to the inner face of a cathode-ray tube, which fluoresces during bombardment by electrons and phosphoresces after bombardment.
PHOTOSPHERE	The intensely bright portion of the sun visible to the unaided eye.
PLAGES	Clouds of calcium or hydrogen vapor that show up as bright patches on the visible surface of the sun.
S BAND TRANSMITTER	2,000 to 3,000 megacycle transmitter.
SCHUMANN EMULSION	Ultraviolet sensitive emulsion.
SENSORS	Portions of a control system which receive deviations from a reference and converts these deviations into signals.
SLEW	The horizontal and/or vertical movement of a device when picking up or tracking a target.
THERMIONIC	Pertaining to emission of electrons by heat.
TORR	Millimeters of mercury.
ULTRAVIOLET	(UV) Electromagnetic radiation in wavelength interval between 100 and 4000 Angstroms.
VACKOTE	BBRC lubrication process for vacuum environment.
VIDICON	A camera tube in which a charged density pattern is formed by photoconduction and stored on that surface of the photoconductor which is scanned by an electron beam, usually of low velocity electrons.
WHITE LIGHT	Radiation producing the same color sensation as average noon sunlight.
X-RAY	Electromagnetic radiation in wavelength interval between 0.1 and 100 Angstroms.

13.3 SYMBOLS

Å	Angstroms
C	Centigrade or Celsius
℄	Center Line
F	Fahrenheit
g	Gravity
Hα	Hydrogen Alpha
MAX. q	Maximum Dynamic Pressure
MHz	Mega hertz (megacycles per second)
μ	Micro
S/N	Signal to Noise Ratio
Hz	Hertz (cps)
UV	Ultraviolet

13.4 DEFINITIONS

ATM	The entire assembly of subsystems without the experiments (analogous to NASA designation: spacecraft).



ATM SYSTEM	The entire assembly of subsystems with experiments (analogous to NASA designation: observatory).
CRADLE	The ATM structure that supports the ATM extension mechanism, attaches to the Apollo Service Module Sector I.
EXTENSION MECHANISM	The truss network and extension drive mechanism that supports the ATM oriented section and extends this section out of the Apollo SM Sector I.
GIMBAL HOUSING	The housing which contains the gimbals, servo motors, etc., and to which the ATM spar is attached.
SPAR	The support structure that attaches to the gimbal housing, and to which the experiments and ATM electronics and thermal shield are attached.
ORIENTED SECTION	The portion of ATM that is movable with respect to the extension mechanism; includes the gimbal housing, spar, experiments, ATM electronics mounted in this section and the thermal shield.
"FLY-TO" CONTROL STICK	The control stick on the ATM control unit that is used by the astronaut crew to maneuver the ATM gimbals in: (1) slew mode; (2) offset pointing; and (3) X-Y experiment functions.
POINTING ERROR	The angle between the true and the desired instrument axis.
AVERAGE POINTING ERROR	The angular distance of the instrument axis from the desired instrument axis found by integrating the average direction of the instrument axis over the exposure time.
DYNAMIC POINTING ERROR	The instantaneous dynamic pointing error is the angle between the instantaneous instrument axis and the average instrument axis.
INSTRUMENT AXIS	This term shall be used to describe the direction of the center of the field of view of an instrument.

APPENDIX A

PRELIMINARY PERFORMANCE AND INTERFACE SPECIFICATION FOR APOLLO TELESCOPE MOUNT

Appendix A

Preliminary
PERFORMANCE AND INTERFACE
SPECIFICATION FOR
APOLLO TELESCOPE MOUNT

BBRC

BALL BROTHERS RESEARCH CORPORATION

BOULDER, COLORADO

DOC. NO. A20719

PRELIMINARY

PERFORMANCE AND INTERFACE

SPECIFICATION FOR

APOLLO TELESCOPE MOUNT

1. SCOPE

1.1 Scope - This specification defines the physical and functional performance requirements of the Apollo Telescope Mount (ATM) and the collateral requirements between the ATM, the scientific experiment equipment, and the Command and Service Modules (CSM). The equipment comprised of the ATM and the scientific experiments is referred to as the ATM system. The equipment comprised of the ATM subsystems without the scientific experiments is referred to as the ATM.

1.2 Objectives - This specification defines the spacecraft constraints imposed on the ATM system and the constraints imposed on the scientific experiments by the ATM. It further stipulates the performance requirements of the ATM to meet the scientific experiment objectives and the compatibility requirements for selection of materials, construction, design, safety, reliability, and environment which must be satisfied for installation and operation on the Apollo spacecraft.

2. APPLICABLE DOCUMENTS

The following documents, of exact issue shown, form a part of this specification to the extent specified herein.

SPECIFICATION

Military

MIL-P-oo7936B (Wep) 17 December 1961	Military Specification Parts and Equipment, Procedures for Packaging of
MIL-E-52720 2 January 1960	General Specification for Environmental Testing of Aeronautical and Associated Equipment
MIL-E-5400G 15 May 1964	Electronic Equipment, Aircraft General Specification for AGS
MIL-C-9435B 17 April 1962	Chamber, Explosion-Proof Testing
MIL-R-27542A 21 May 1963	Reliability Program for Systems, Subsystems, and Equipment

National Aeronautics and Space Administration (NASA)

MSFC-10M01071	Environmental Protection When Using Electrical Equipment Within Areas of Saturn Complexes, Where Hazardous Areas Exist, Procedure for
IESP 4 December 1964	Interference Control Requirements for Spacecraft Equipment

North American Aviation, Inc.

Space and Information Systems Division (NASA/S & ID)

SID 64-1388 5 January 1965	Scientific Instrumentation Performance And Interface Specification Block II
MC 999-0058 16 July 1964	Approved Materials for use in the Apollo Spacecraft

STANDARDS

Military

MIL-STD-130B Change 1 7 February 1964	Identification Marking of U. S. Military Property

MIL-STD-143A 14 May 1963	Specification and Standards, Order of Precedence for the Seclection of
MIL-STD-202E 2 December 1963	Selected Standards for R. F. and Acoustical Parts
MIL-STD-810 14 June 1962	Environmental Test Methods for Aerospace and Ground Equipment
MS-33586A 16 December 1958	Metals, Definition of Dissimilar

OTHER PUBLICATIONS

National Aeronautics and Space Administration (NASA)

NPC 250-1A 30 July 1963	Reliability Provisions for Space Systems Contractors
NPC 200-1A June 1964	Quality Assurance Provisions for Space Systems Contractors
NPC 200-2 20 April 1962	Quality Program Provisions for Space Systems Contractors
NPC 200-3 April 1962	Inspection System Provisions for Supplies of Space Materials, Parts, Components, and Services

3.0 REQUIREMENTS

3.1 Performance - The ATM is a three axis stabilized platform fixed to the Apollo Service Module (SM) on which solar telescopes are mounted. The ATM is remotely operated by an Apollo crew member from the Apollo Command Module (CM).

3.1.1 Operational Requirements

3.1.1.1 Mission - ATM missions may be conducted as part of on-line Apollo missions, or as a part of later, more scientifically oriented missions in the Apollo applications program. In flight scientific experiments will be conducted during these missions to the maximum extent possible without compromising the principal mission objectives. ATM missions are to be conducted in low-inclination, near-earth orbits. The orbital inclination shall be nominally 30 degrees from the equator, at an altitude of 200 nautical miles. The mission duration shall be 14 days.

3.1.1.2 Scientific Equipment - The scientific experiments shall be selected by NASA and the scientific equipment shall be in general NASA/government furnished equipment (GFE). They will provide for scientific and technological investigation of various phenomena observable on or near the sun.

3.1.1.3 Functional Characteristics - The ATM shall be capable of orienting the optical axis of scientific experiments to specified portions of the solar disk after acceptable initial conditions of CSM attitude and drift rates have been established. The ATM shall provide structure, stabilization, thermal control, power, operational control, and data handling as required for the scientific experiments within the Apollo constraints.

3.1.1.4 Pointing Performance Requirements - Solar pointing performance shall be as follows:

(1) Pitch and yaw axes
- Relative pointing error: 1 min (time) ±5 arc sec
  20 min (time) ±1 arc min
- Coarse pointing error: ±2 deg

(2) Roll drift rates
- Average: <1 arc sec/sec

Peak: <4 arc sec/sec

3.1.1.5 Gimbal Limits - The range and limits of gimbal travel shall be as follows:

- Pitch: ±30 degrees
- Yaw: ±170 degrees
- Roll: ± 25 degrees

The gimbals shall be held immobile when the limit of gimbal travel is reached, when there is no slewing command while in manual mode, and when power is removed.

3.1.1.6 Pointing Acquisition - The ATM shall be capable of automatically orienting the ATM optical axis within the specified accuracy requirements when the initial error is within the gimbal limits and is no more than the following:

- Coarse mode: ±25 degrees
- Fine mode: ± 4 degrees

3.1.1.7 Offset Pointing - The ATM shall have the capability to offset the ATM optical axis from the center of the solar disk within the range of ± 20 arc minutes, and maintain the accuracy requirements of paragraph 3.1.1.4.

3.1.1.8 Slew Control - Controls shall be provided for the crew operator to slew the oriented section in all three axes. Control shall be provided for setting the initial gimbal angles and for adjusting the offset pointing axis. Slewing rates shall be provided for all axes as follows:

- Initial setting (slew): - 6 degrees/second
- Offset high rate (offset): - 40 arc seconds/second

- Offset low rate (boresight): - 2 arc seconds/second

3.1.1.9 Thermal Control - The ATM shall have a thermal shield which encloses the scientific experiment complement, and shall provide passive and/or active control as required to maintain an internal thermal environment within ±5°F of nominal 60°F.

3.1.1.10 Aperture Covers - Aperture covers shall be provided as required for each experiment, the fine control sensors, and the ATM solar monitor optics to prevent contamination during launch, operation of the Apollo reaction control system (RCS), and operation of the crew and fuel cell waste dump. Aperture covers shall be remotely operated to both an open and closed condition by a crew member in the CM.

3.1.1.11 Access Doors - Access doors shall be provided in the thermal shield to facilitate loading and recovery of film magazines in the experiments.

3.1.1.12 Solar Monitor - A telescope, photoelectric sensor, and display shall be provided, which allows a crew member to observe the solar image. The solar monitor shall have:

(1) Its optical axis aligned to the pointing control axis within ±1 arc minute.

(2) An image resolution of five arc seconds.

(3) Filters to allow observation of the solar image at 6563 Å (Hα) or 3634 Å (CaK).

3.1.1.13 Experiment Mounting - An experiment mounting structure shall be provided which will:

(1) Provide insulation against heat flow across the mounting surface.

(2) Not introduce structural distortions into the

experiment structure.

(3) Allow alignment of the instrument axes relative to the pointing axis.

3.1.1.14 Data Requirements

3.1.1.14.1 Scientific Experiments - The ATM shall provide the following minimum capability for handling scientific experiment data:

(1) Fifteen digital channels (16-bit word)
- Minimum sampling rate: 0.01 samples per second
- Maximum sampling rate: 25 samples per second
- At least two channels with different sampling rates

(2) Twenty analog channels
- Sampling rate: at least once per 20-minute observing period
- Input voltage: zero to five volts peak

The specific data requirements for the scientific experiments shall be determined at a later date.

3.1.1.14.2 ATM Data Requirements - The ATM shall provide the following minimum capability for handling status and performance monitoring data of the ATM:

(1) Twenty-six digital channels (8-bit word)
Sampling rate: 0.05 samples per sec max
0.01 samples per sec min

(2) Eighty analog channels
Sampling rates: 0.05 samples per sec max
0.01 samples per sec min

3.1.1.14.3 Data Storage - The ATM shall provide means of data storage for accumulated data in at least two orbits if

Apollo communications link can be negotiated for use. In case the Apollo communication link cannot be used, the ATM shall provide data storage for the full 14-day mission. For the latter case, a means of retrieval of stored data shall be provided.

3.1.1.15 Electrical Power Requirements - Maximum use shall be made of available CSM power, and the ATM system shall be designed to be compatible with the CSM. The following electrical power requirements shall be provided for the ATM system for a 14-day mission:

- Total Energy requirement: 30 kilowatt hours
- Maximum surge current: 55 amperes
- Average power (during solar observations): 170 watts
- Average power (night or nonobserving): 60 watts

3.1.2 Operability

3.1.2.1 Reliability - The ATM shall be designed to the same level of reliability as any other component of nonmission-essential CSM equipment.

3.1.2.2 Maintainability - The maintainability of the ATM shall be the responsibility of NASA and shall be subject to agreement on the countdown requirement. All associated maintenance and GSE equipment shall be furnished by the ATM contractor and/or NASA.

3.1.2.3 Useful Life - The ATM shall be designed for a 14-day mission life.

3.1.2.4 Natural Environment - The ATM and associated ground support equipment (GSE) shall be designed to meet the requirements as specified in paragraph 3.1.2.4.of NAA specification SID 64-1388.

3.1.2.5 Transportability - Wherever possible, equipment shall be designed to be transported by common carrier with a minimum of protection. Special packaging and transportation methods shall be required to prevent system penalties.

3.1.2.6 Human Performance - The flight crew shall perform monitoring and control of the ATM system during those periods when it does not compromise normal monitoring and control of the CSM. Extra vehicular activity (EVA) shall be required to recover and reload film magazines approximately seven days after launch, and to recover film magazines at the end of the fourteen-day mssion prior to reentry.

3.1.2.7 Safety

3.1.2.7.1 Hazard Proofing - The design of the ATM system shall not contribute to the hazard of fire, explosion, and toxicity to the crew, launch area personnel and facilities. The hazards to be avoided include accumulation of leakage of combustible gases, the hazard of spark ignition sources including static electricity discharge.

3.1.2.7.2 Equipment - Where practicable, the various components shall be hermetically sealed or of explosion-proof construction.

3.1.2.7.3 Fail Safe - ATM System equipment or component failure shall not propogate sequentially; that is, design shall be "fail safe".

3.1.2.8 Induced Environment - The ATM System and associated GSE shall be designed to be capable of operating after exposure to the applicable induced environments as specified in paragraph 3.1.2.8 of NAA Specification SID 64-1338.

3.2 Interface Requirements

3.2.1 Power Requirements - The ATM shall be designed to be compatible with the Apollo electrical power system and use Apollo power as available. Where special equipment requires special voltage levels, batteries, inverters, and converters, they shall be supplied as part of the ATM system.

3.2.1.1 Power Allocation - All electrical power supplied by the CSM is defined as a nonessential load and may not be operated during an emergency. A power profile showing operational demands shall be outlined for each mission and shall be made available for allocation of power.

3.2.1.2 Power Characteristics - CSM power characteristics are as follows:

3.2.1.2.1 DC Power -

(1) Steady-state voltage limits: 27.5 ± 2.5 volts

(2) Transient voltage limits: 21 volts to 32 volts with recovery to steady-state within one second

(3) Ripple voltage: 250 millivolts peak to peak maximum

3.2.1.2.2 AC Power -

(1) Phases: three phases - displacement 120 degrees - tolerance plus or minus two degrees

(2) Nominal voltage limits:

- Steady-state; 115 ± 2 volts rms (average of three phases)

- Transient: 115 plus 35, minus 65 volts recovering to 115 plus or minus 10 volts within 15 milliseconds and steady-state within 50 milliseconds
- Unbalance: two volts (worst phase from average)
- Modulation: 0.5 percent

(3) Nominal frequency tolerance:

- Normal: negligible (synchronized to master timing)
- Emergency: 400 ± 7 cps (loss of master timing)

(4) Wave shape: sine wave

- Maximum total distortion: five percent
- Highest harmonic: four percent
- Crest factor: 1.414 plus or minus 10 percent

3.2.2 Cables and Harness - Electrical power and data recording/telemetry outlets shall be provided by the Apollo CSM as standard provisions.

3.2.2.1 CM/SM Umbilical Provisions - The following provisions within the CM/SM umbilical shall be allocated for the ATM system:

- Fifty-eight umbilical pins; eight 16-gauge for power, fifty 20-gauge for signal.
- Two coaxial cable connections, type Cx

3.2.3 Volume Requirements

3.2.3.1 ATM System Volume - The ATM system shall occupy Sector I of the SM. The total volume allotment is shown in Fig. 1.

3.2.3.2 Scientific Experiment Volume - The minimum volume allotment for scientific experiments in the ATM system is shown in unshaded portion of Fig. 2.

3.2.3.3 ATM Volume Requirements (CM) - Space shall be allocated in the CM for location of a control panel, display panel, and storage of film and tapes. Approximately two cubic feet shall be allotted for control and display panels, and three cubic feet for data storage. Location of space allotment shall be subject to negotiation.

3.2.4 Weight - The total weight of the ATM system contained in Sector I of SM shall not exceed 5000 pounds.

3.2.5 Center of Gravity - The ATM system center of gravity location shall not adversely affect the operation of the service propulsion system (SPS). The resultant center of gravity of the CSM and ATM must fall within the following SPS gimbal angle restraints anytime the SPS is operated:

- Y axis: + 12.5 degrees, -4.5 degrees
- Z axis: ± 6.0 degrees

3.2.6 Attachment - The primary attach points of the ATM to the Sector I bay shall be the same as that which accommodates the Lunar Mapping and Survey system. The CSM shall provide two additional attach points at both the forward and aft ends of the Sector I bay capable of reacting loads in the two lateral directions. Figure 3 shows typical attach point requirements and their principal axis of load reaction.

3.2.7 Data Requirements - The CSM shall provide for transmission of real-time and recorded data via the CSM communications subsystem to an extent which shall be subject to negotiation. It is desirable that the CSM PCM telemetry equipment operating in the high data rate mode, 51.2K bits/sec, be fully available to the ATM system on a time-shared basis of seven minutes every two orbits as a maximum utilization rate.

3.2.8 Time Correlation - The CSM shall provide data containing time-from-launch information, which can be recorded with the ATM data.

3.3 Design and Construction

3.3.1 General Design Features - NASA and the ATM contractor shall be responsible for the detail design, performance criteria and packaging of the ATM system. The CSM contractor shall be responsible for providing the stowage, volume, power, data utilization, and integration as outlined herein.

3.3.1.1 Installation Data - NASA and the ATM contractor shall provide the CSM contractor with the necessary installation information, including design drawings and unique equipment orientation, to ensure compatibility and to facilitate integration of the design with the CSM configuration. The CSM contractor shall provide matched drill plate tooling for each volume to indicate mounting provisions in each stowage area.

3.3.1.2 Integration - The ATM system shall be integrated with the CSM by the CSM contractor, in accordance with specific mission requirements. Conflict with the design configuration of the CSM, electrical power requirements, data readout modes and rates, and electromagnetic interference shall be subject to negotiation.

3.3.1.3 Mockups - Mockups or prototype orientation shall be provided and will define the physical dimensions, weight, and center of gravity of the ATM system.

3.3.1.4 Electrical Design - Electrical design criteria shall be as follows:

3.3.1.4.1 Grounding - All electrical grounding shall be "floating" and brought out to a separate pin on the interface electrical connector(s).

3.3.1.4.2 Shielding - All shielded wires shall have the shields brought out to individual pins on the interface electrical connector(s). Each shield shall be brought out to a separate pin, except for the umbilical connectors from CM to SM, which shall be terminated on a common pin.

3.3.1.5 Radiation Control - Scientific equipment using radioactive materials in detectors and sensors shall be properly shielded for side effects so as not to interfere with, influence the operation of, deteriorate equipment in adjoining and neighboring compartments, or contribute to radiation build-up tolerance of the crew.

3.3.2 Selection of Specifications and Standards - Materials, parts, and processes shall be selected in the following order of preference, provided coverage is suitable:

(1) Federal specifications approved for use by NASA
(2) Military specifications and standards (MIL, JAN or MS)
(3) Other governmental specification
(4) Specifications released by nationally recognized associations, committees, and technical societies

3.3.3 Materials, Parts, and Processes - Materials, parts, and processes shall be selected with the following considerations:

(1) Materials, parts, and processes shall be suitable

for the purpose intended. Safety, performance, reliability, and maintainability of the item are of primary importance.

(2) Except in those instances where their use is essential, critical materials shall not be used.

(3) Where possible, materials and parts shall be of kind and quality widely available in supply channels.

(4) When practicable, a choice between equally suitable materials and parts shall not be specified.

(5) Whenever possible, single source items shall be avoided.

(6) When practicable, circuits shall be designed with a minimum of adjustable components.

3.3.3.1 Flammable Materials - Nonmetallic materials shall not support combustion.

3.3.3.2 Toxic Materials - Materials that could release corrosive fumes or gases will not be used. No materials will be used that can liberate gases, fumes, or particles hazardous to human life or that may adversely affect comfort or efficiency of personnel. External finish of the scientific equipment shall not release toxic products when subjected to the CM cabin atmosphere as outlined per NAA/S&ID Specification MC 999-0058.

3.3.3.3 Unstable Materials - Metals shall be nonmagnetic and corrosion-resistant alloys, except where the use of magnetic materials is necessary to comply with the electrical design characteristics of the ATM system and does not compromise the operation and performance of other spacecraft equipment. No cadmium or zinc metals shall be used in the fabrication of the ATM system.

3.3.3.4 Dissimilar Metals - Dissimilar metals in direct contact with each other, which tend toward active electrolytic or galvanic corrosions, shall not be used in the fabrication of the ATM system. Dissimilar metals are defined in Specification Standard MS 33586.

3.3.4 Standard Materials, Parts, and Processes

3.3.4.1 Standard Parts - Standard parts Air Force-Navy (AN) or military standards (MS) or joint Air Force-Navy (JAN) shall be used wherever they are suitable for the service, and shall be identified by their numbers. Commercial parts having suitable properties may be used, if they conform to all the requirements and (1) no suitable standard parts are in effect on the date of manufacture of the equipment, or (2) the commercial part used is replaceable by a standard part without alteration.

3.3.4.2 Standard Materials and Processes - Materials and processes used in the manufacture of the ATM system shall conform to specifications applicable in Standard MIL-STD-143.

3.3.5 Moisture and Fungus Resistance

3.3.5.1 Fungus - Nonnutrient materials as specified in Specification MIL-E- 5400 shall be used where practical in the manufacture of the ATM system.

3.3.5.2 Moisture - Equipment located in the CM using electrical connectors shall be of the hermetically sealed type.

3.3.6 Corrosion - Materials used in the manufacture of the ATM system shall be corrosion resistant.

3.3.7 Interchangeability and Replaceability - All equipment having the same supplier's part number shall be physically and functionally interchangeable.

3.3.8 Workmanship - Equipment shall be fabricated in a workmanship manner complying with sound industry practice. Particular attention shall be given to neatness and thoroughness of working parts, assemblies, plating, welding, and freedom of parts from burrs and sharp edges that might damage associated equipment, or cause injury to operating personnel.

3.3.9 Electromagnetic Compatibility - The ATM system shall be designed and tested to the requirements of MIL-I-26600. NASA approved interpretations of this document shall be permitted.

3.3.10 Identification and Marking

3.3.10.1 Marking Requirements - Each item delivered as an individual part or individual assembly, must be identified by a part number and marked in accordance with MIL-STD-130. In addition, all functional, replaceable, repairable assembly items shall be serialized to differentiate between identical assembly units. Lower order assemblies such as functional, replaceable, repairable items of an assembly are also required to have a part number and a serial number. In each case, the part number will be the same on all identical parts or assemblies. The part numbers and serial numbers must be clearly and durably applied to the part or to the nameplate of the part. Where the part is small or fragile, a tag firmly affixed to the part may be used to bear the identification. Drawings, specifications, and procedures for the part must have the part number on the title block or title page.

3.3.11 Storage - The ATM shall have a shelf life of not less than three years under conditions outlined in paragraph 3.1.2.4.1.1 of NAA/S & ID Specification SID 64-1388.

4. QUALITY ASSURANCE PROVISIONS

4.1 General Quality Assurance Program - A quality assurance program shall be established in accordance with NASA publications NPC 200-2 and NPC 200-3. Inspections and tests to determine conformance of the system to contract and specification requirements shall be conducted prior to submission of the article to NASA for acceptance.

4.2 Reliability Program - A reliability program shall be established in accordance with NASA publication NPC 250-1.

4.3 Testing - NASA and the ATM contractor shall provide proposed preinstallation acceptance tests procedures which will outline the functional verification and CSM integration tests to the CSM contractor for his preparation and completion of the final preinstallation acceptance test (PIA). The proposed PIA procedures should be based on the verification tests to which the equipment was subjected at the time of acceptance. The CSM contractor will perform a PIA on the equipment after delivery to his plant and prior to installation into the CSM. Equipment also will be checked during the CSM systems checkout in the installed state. If special test equipment is required of the CSM contractor for the PIA, NASA and the ATM contractor will insure that the proper equipment is made available.

4.3.1 Test Program - The CSM contractor, in joint effort with NASA and the ATM contractor, shall establish criteria for scientific equipment and related GSE in accordance with MIL-STD-810, MIL-L-9435, MIL-STD-202, MIL-R-27542, NPC-2501, NPC-200-2, NPC-200-1, and MSFC-10M01071. The specific test requirement shall be defined at a later date.

4.3.2 Qualification Testing - NASA and the ATM contractor shall be responsible for performing all the required qualification testing on the ATM system to ensure that the equipment is compatible with the spacecraft's mission requirements.

5. PREPARATION FOR DELIVERY

5.1 Preservation, Packing, and Marking Requirements - The preservation, packaging, packing, and marking requirements shall be in accordance with MIL-P-007936B (Wep).

6. NOTES

Not applicable

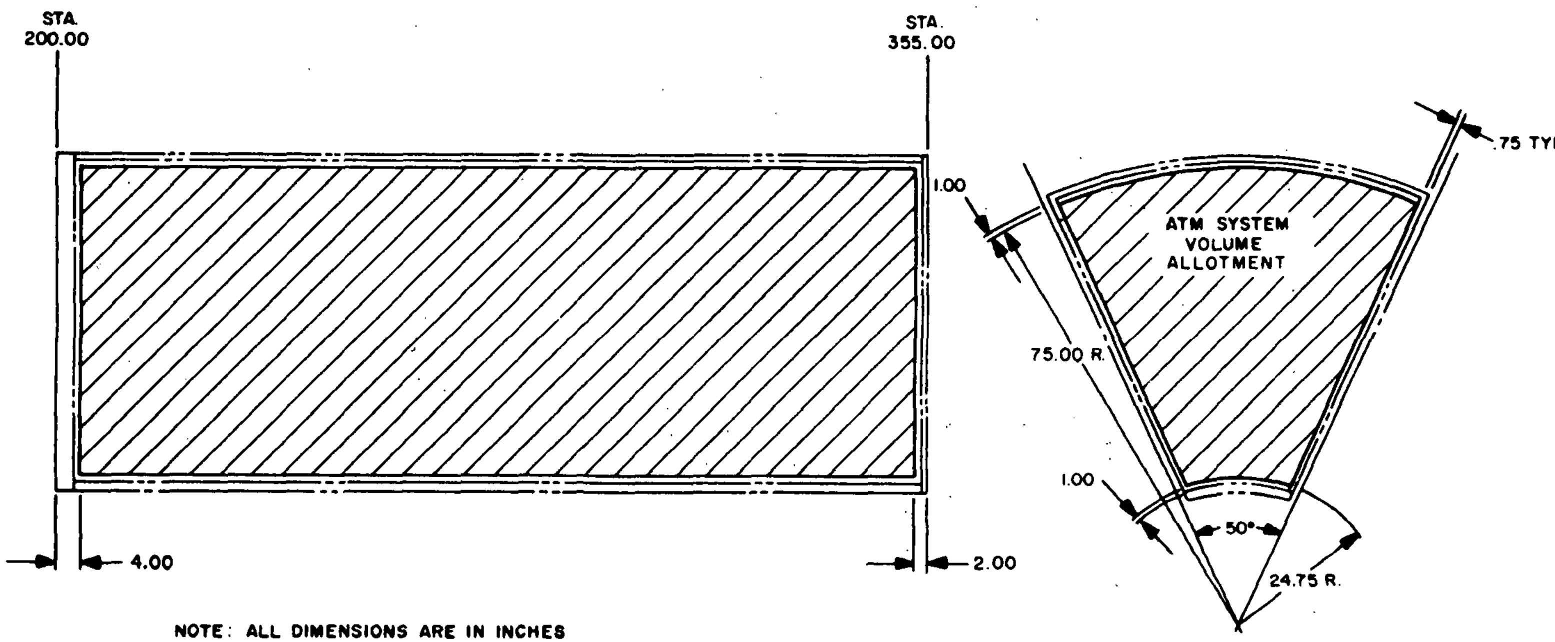
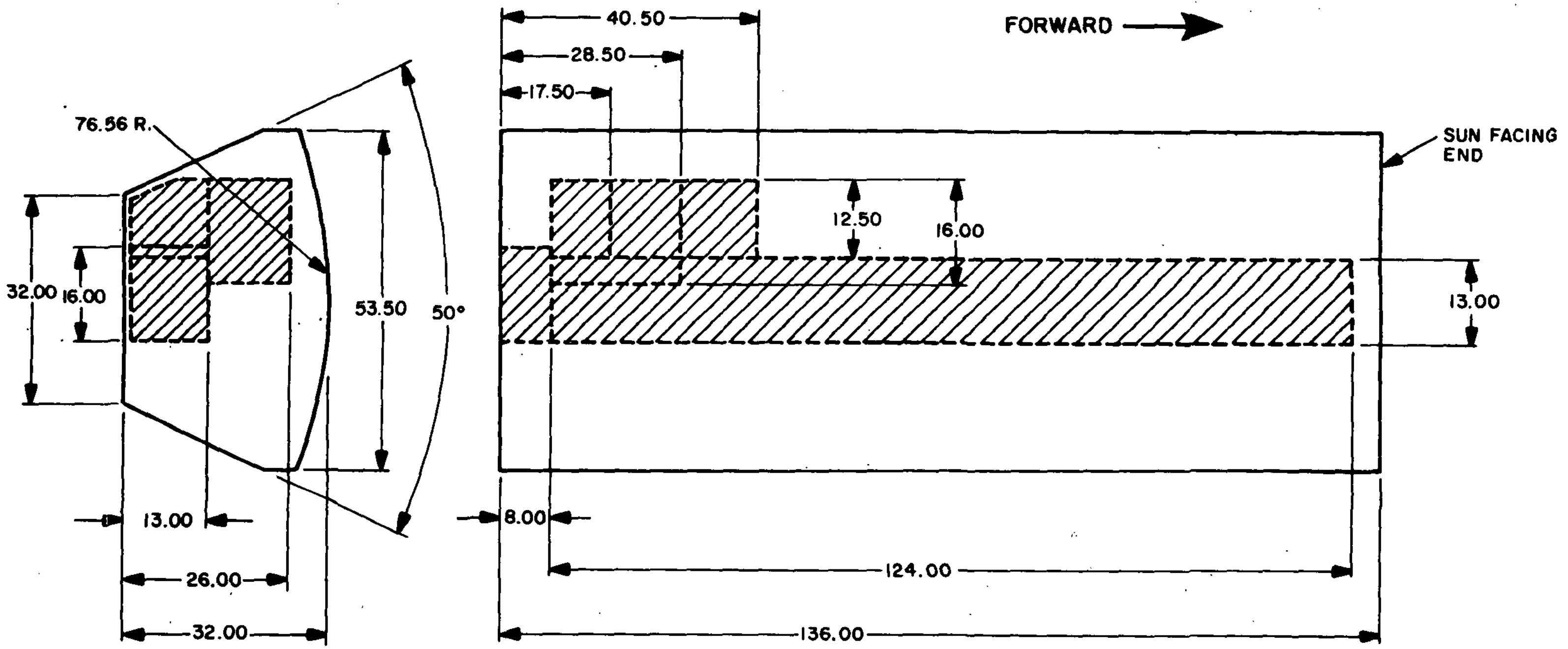


Fig. 1 ATM Volume Allotment - SM Sector I



NOTE: ALL DIMENSIONS ARE IN INCHES

SHADED AREA: ATM SUBSYSTEMS
CLEAR AREA: EXPERIMENT ALLOWED VOLUME

Fig. 2 Minimum Experiment Volume Allotment

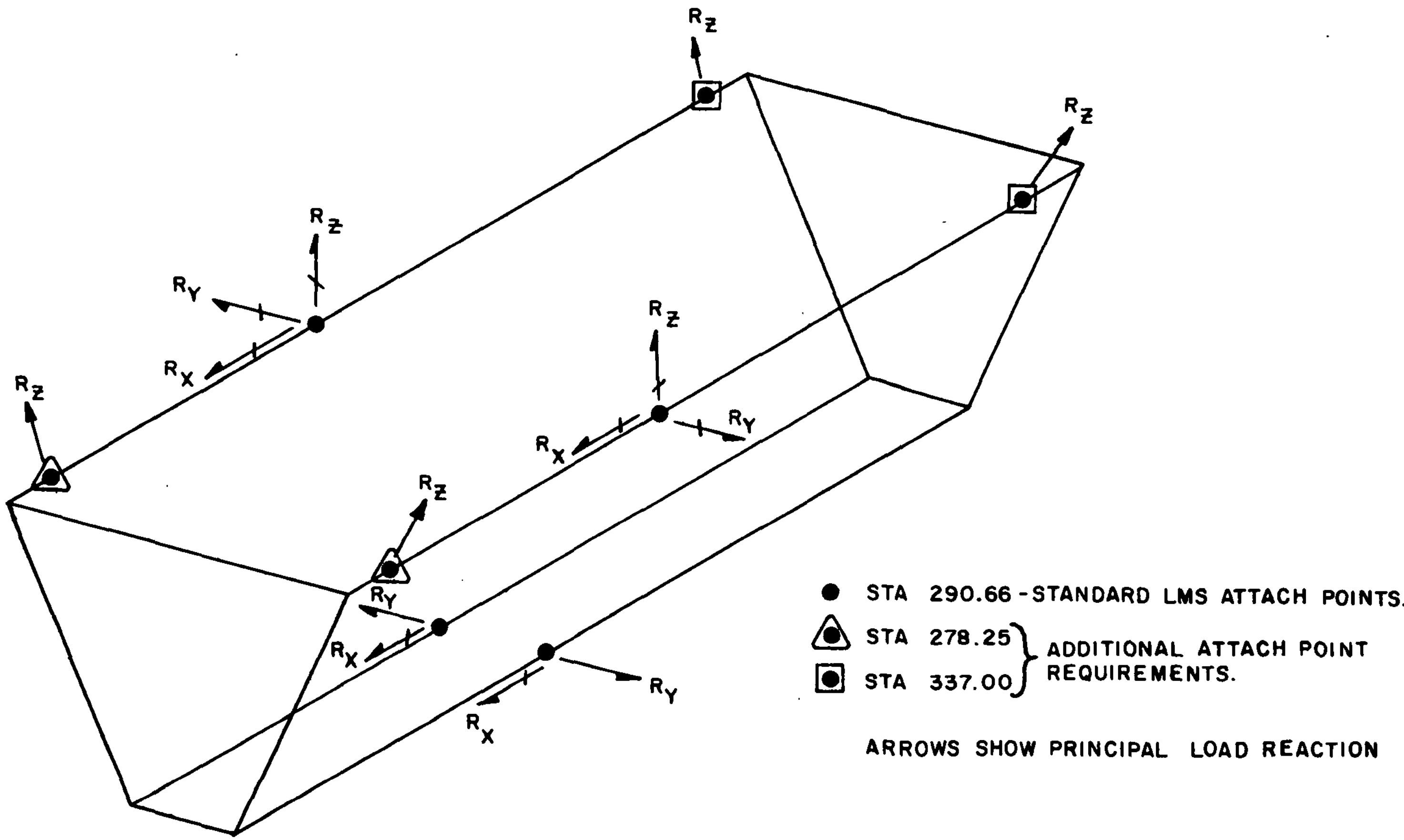


Fig. 3 Sector I Attach Points and Axis of Load Reaction

APPENDIX B

THERMAL ANALYSIS

Appendix B
THERMAL ANALYSIS

Section 1
ABSTRACT

B.1 INTRODUCTION

Throughout this appendix section it is assumed that the mounting spar can be represented by a tube 12 feet long and 1 foot in diameter. The final study spar configuration is an inverted box U. (See Fig. 4-1.) A detailed thermal analysis of the U-shaped spar configuration was not performed during this study. However, the analytical results obtained using the tubular model provide order-of-magnitude values for temperature differences, heat flow rates, and insulation requirements, and, therefore, are sufficient to show feasibility of the design approach.

Appendix B
Section 2
ALLOWABLE TEMPERATURE DIFFERENCE ACROSS THE MOUNTING SPAR

B.2 DIFFERENTIAL EXPANSION

The temperature difference across the spar causes the spar to bend into an arc because of differential expansion. This difference, ΔT ($^{\circ}F$), can be determined from the following equation (Fig. B-1):

$$\Delta T = \frac{D\theta}{L\alpha} \tag{B.1}$$

where:

D = spar diameter (ft)
θ = bend angle (rad)
L = spar length (ft)
α = thermal coefficient of expansion ($13 \times 10^{-6}/^{\circ}F$ for aluminum)

If the spar is assumed to be an aluminum tube 12 ft long and one foot in diameter, and the allowable spar thermal misalignment is one arc second (5×10^{-6} rad), Eq.(B.1) yields the following result:

$$\Delta T = \frac{(1)(5 \times 10^{-6})}{(12)(13 \times 10^{-6})} = 0.032^{\circ}F$$

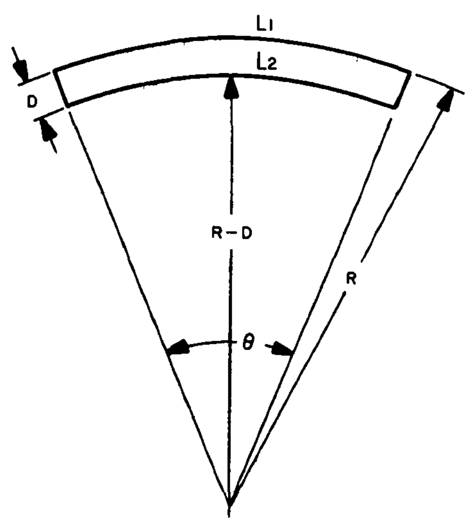


Fig. B-1 Allowable Temperature Difference Across the Alignment Spar

Appendix B
Section 3
THERMAL DESIGN REQUIREMENTS TO MAINTAIN EXPERIMENT POINTING ACCURACY

B.3 INTRODUCTION

Thermal distortion of the mounting spar will result in misalignment of the attached instruments during a data acquisition period. The dynamic instrument pointing accuracy required for the ATM system is of the order of five arc seconds for a one minute pointing period, and 20 arc seconds for a 20 minute pointing period. The dynamic thermal misalignments allowed during these pointing periods are one arc second and five arc seconds, respectively. Steady state bending produces pointing errors that can be nullified by the astronaut. However, the design requirement to restrict a steady-state bending to 100 arc seconds is of interest for overall system considerations.

In Section 2, Appendix B, above, it is shown that the maximum allowable temperature difference across the spar diameter is 0.032°F for a pointing accuracy of one arc second. Since this is a linear function, the allowable temperature difference for a misalignment bend of 100 arc seconds is 3.2°F.

An estimate of the allowable heat input rate to the spar can be made by dividing the spar longitudinally into two sections or elements. A lateral cross section of this model is sketched in Fig. B-2.

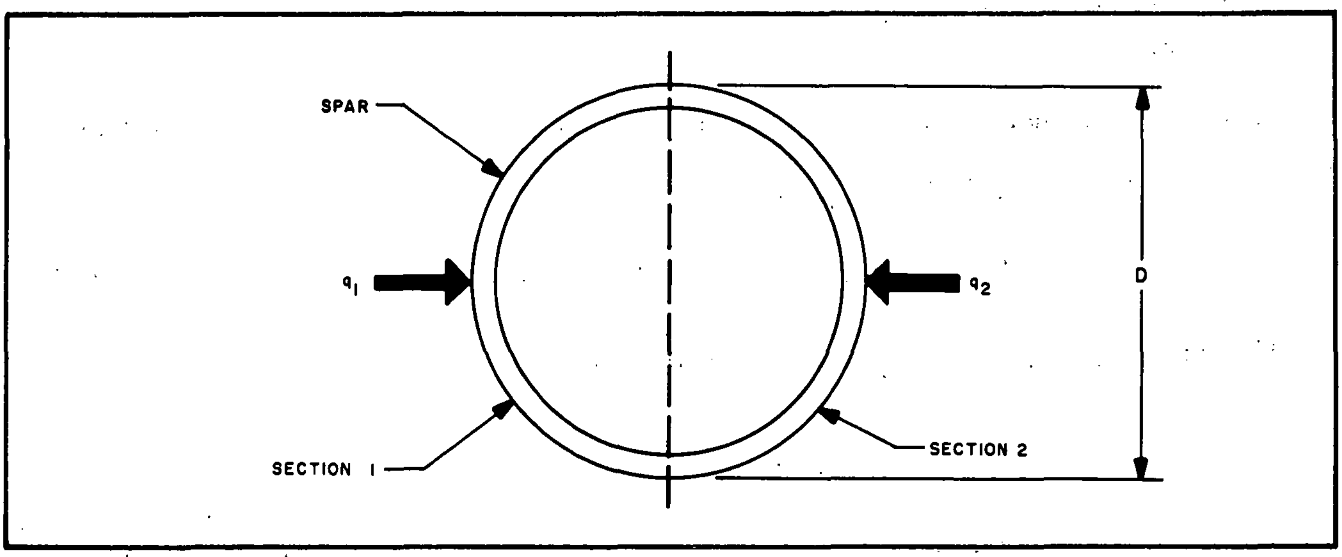


Fig. B-2 Allowable Heat Input Rate to the Spar

A heat balance for both elements results in the following equations:

$$m_1 c \frac{dT_1}{d\theta} = q_1 + \sigma b_{1,2} \left(T_2^4 - T_1^4\right) + C_{1,2} \left(T_2 - T_1\right) \tag{B.2}$$

$$m_2 c \frac{dT_2}{d\theta} = q_2 + \sigma b_{1,2} \left(T_1^4 - T_2^4\right) + C_{1,2} \left(T_1 - T_2\right) \tag{B.3}$$

where:

θ = time (hrs)

m_1, m_2 = masses of section 1 and 2 of the tube, respectively (lbs)

c = specific heat of the material (Btu/lb°F)

$b_{1,2}$ = radiation exchange factor between section 1 and 2

$C_{1,2}$ = thermal conductance between section 1 and 2 (Btu/hr°F)

q_1,q_2 = heat input rates from the environment to sections 1 and 2, respectively (Btu/hr)

σ = Stefan-Boltzmann constant $\left[(0.1714 \times 10^{-8}) \text{ Btu/hr ft}^2 {}^\circ R^4\right]$

Subtracting Eq. (B.3) from Eq. (B.2), setting $m_1 = m_2 = m$, and letting $\Delta T = T_1 - T_2$ yields the result:

$$mc \frac{d(\Delta T)}{d\theta} = (q_1 - q_2) + 2\sigma b_{1,2} \left(T_2^4 - T_1^4\right) - 2c_{1,2} \Delta T \tag{B.4}$$

If $T_1 \approx T_2$, then $\sigma \left(T_2^4 - T_1^4\right) \approx -4\sigma T^3 \Delta T$ and for $T \approx 530^\circ R$, $4\sigma T^3 \Delta T \approx \Delta T$ (since $4\sigma T^3 \approx 1$ at $530^\circ R$).

Therefore, for small ΔT and $T \approx 530^\circ R$, Eq. (B.4) can be simplified to the expression:

$$mc \frac{d(\Delta T)}{d\theta} = (q_1 - q_2) - 2b_{1,2} \Delta T - 2c_{1,2} \Delta T$$

or

$$mc \frac{d(\Delta T)}{d\theta} = (q_1 - q_2) - 2\Delta T \left(b_{1,2} + c_{1,2}\right) \tag{B.5}$$

By letting $\Delta q = q_1 - q_2$, Eq. (B.5) can be rewritten in the form:

$$\frac{d\theta}{mc} = \frac{d(\Delta T)}{\Delta q - 2(b_{1,2} + C_{1,2}) \Delta T} \tag{B.6}$$

For the initial condition ($\theta = 0$, $\Delta T = 0$), integration of Eq.(B.6) yields the result

$$\theta = \frac{mc}{2(b_{1,2} + c_{1,2})} \ln \left[\frac{\Delta q}{\Delta q - 2(b_{1,2} + c_{1,2}) \Delta T}\right] \tag{B.7}$$

Note that in Eq. (B.7) $\theta \rightarrow \infty$ as $\Delta q \rightarrow 2\,(b_{1,2} + c_{1,2})\,\Delta T$; this is the steady-state heat flow.

The value of the heat transfer coefficients $b_{1,2}$ and $c_{1,2}$ in the preceding equations should be as large as possible in order to permit a large value of Δq. However, a large value of $b_{1,2}$ requires that no structures such as electrical wiring or insulation obstruct the radiation between the interior sides of the tube elements. This is not a practical requirement and the value of $b_{1,2}$ should be set equal to zero in order to obtain a conservative thermal design.

The value of $c_{1,2}$ in the Eq. (B.7) is a linear function of the wall thickness of the tube; and, hence, of overall tube mass. It is necessary to repeat the solution for wall thickness versus practical values of Δq that can be expected for the temperatures to be encountered and the state of the art of experiment mounting techniques for the reduction of conductive heat flow. For a wall thickness of 0.16 inches for the aluminum tube having a diameter of one foot and a length of 12 feet, the value of $c_{1,2}$ is determined as follows:

$$c_{1,2} = \frac{kA_c}{L} \tag{B.8}$$

where k = thermal conductivity = 100 Btu/hr ft^2 $^\circ$F/ft

A_c = cross-sectional area = (2) (0.16") (12 ft)/12 = 0.32 ft^2

L = conductive path length = 1/2 πD = 1.57 ft

and

$$c_{1,2} = \frac{(100)\ (0.32)}{1.57} = 20.4 \text{ Btu/hr } ^\circ\text{F}$$

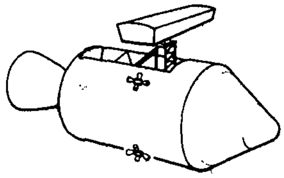
B.3.1 Steady-State Heat Input

It is assumed that the astronaut can nullify pointing errors caused by steady-state heat inputs to the spar. However, the order-of-magnitude requirements for shielding, necessary to restrict steady-state spar bending to about 100 arc seconds, are indicated by the following computations.

The steady-state heat input rate that will result in a bending angle of 100 arc seconds is computed using Eq. (B.7):

$$\Delta q = 2C_{1,2}\Delta T \tag{B.9}$$

As previously shown, the value of ΔT for a 100 arc second bend is 3.2°F; this value together with the above computed value, 20.4 Btu/hr $^\circ$F, for $C_{1,2}$ result in Δq = 13.06 Btu/hr.



This Δq represents the net steady-state heat flow, $q_1 - q_2$, where $q_1 = -q_2$, so that

$$q_1 = 13.06/2 = 6.53 \text{ Btu/hr}$$

This would be the evenly distributed heat flow entering one entire side of the spar and leaving the opposite side. The principal source of heat flow is the temperature difference existing between the experiments and the spar. The heat transfer is by conduction and radiation.

Conductive heat transfer occurs through the mounts attaching the experiments to the spar and its rate is given by the standard equation,

$$q = \Delta T/_R \qquad (B.10)$$

where q = rate of heat flow

ΔT = temperature difference between the experiments and the spar

R = the thermal resistance of the mounts

Assume that the experiments have a temperature that is of the order of 10°F different from that of the spar and, as a worst case, that the differences on opposite sides are of opposite signs. Then for a mount resistance of 275°F/watt the use of the above equation results in q = 0.036 watts.

If the experiment attachment points were at one foot intervals along the spar the above conductive heat flow for each attachment would result in a total of (12) (0.0364) = 0.436 watts, or 1.5 Btu/hr.

This value is much less than the allowable value of 6.53 Btu/hr previously computed. However, radiative heat transfer between the experiments and the spar must also be accounted for. In order to limit radiative heat transfer to 6.53-1.5 = 5.03 Btu/hr, aluminized mylar insulation must be installed between the spar and the experiments. The number of layers required can be estimated as follows.

The radiation heat transfer between the experiments and the spar is expressed by the equation,

$$q = F_A F_\epsilon A \sigma \left[(T_{EX})^4 - (T_{SPAR})^4 \right] \qquad (B.11)$$

where: F_A = configuration factor = 1

A = 18.8 ft^2 (1/2 spar area)

σ = 0.1714 x 10^{-8} Btu/hr-ft^2 - °R^4

T_{EX} = experiment temperature = 70°F or 530°R

T_{SPAR} = alignment spar temperature = 60°F or 520°R

F_ε = emissivity factor

The equation for F_ε is the standard equation for multilayered shielding.

$$F_\varepsilon = \frac{1}{N\left(\frac{1}{\varepsilon_M} + \frac{1}{\varepsilon_{AL}} - 1\right) + \left(\frac{1}{\varepsilon_{EX}} + \frac{1}{\varepsilon_{SPAR}} - 1\right)} \qquad (B.12)$$

where,

N = number of aluminized mylar shields required
ε_M = mylar emissivity = 0.8
ε_{AL} = aluminum emissivity = 0.05 (vacuum deposited)
ε_{EX} = experiment emissivity = 0.9 (assumed black)
ε_{SPAR} = alignment spar emissivity = 0.2 (aluminum)

For these values,

$$F_\varepsilon = \frac{1}{20.25\ N + 5.11}$$

Setting q in Eq. (B.11) equal to 5.03 Btu/hr as the allowable radiative heat transfer rate and solving for N results in

N = 1.6 layers.

Thus, two layers of aluminized mylar are required between the experiments and the spar to restrict the steady-state bending of the spar to about 100 arc seconds.

Since the mounting spar contains heat dissipating equipment that must radiate through the bottom of the spar to space (Section 4, Appendix B), the bottom side of the spar will also radiate to deep space unless adequately insulated. The number of layers of aluminized mylar required for this insulation can be estimated in the following manner.

Assume that (1) the lower section of the spar is exposed to the exterior environment, deep space at 0°R; (2) the upper surface of the spar is exposed to 70°F (experiments, electronic equipment, etc.); and (3) the spar will be maintained at 60°F.

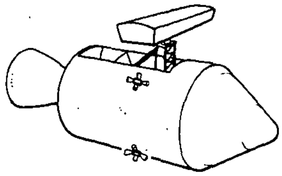
The radioactive heat transfer equations for the upper and lower surfaces are for the interior and exterior environments, respectively,

$$q_1 = F_{A1}\ F_{\varepsilon 1}\ A_1 \sigma \left[T_1^4 - T_2^4\right] \qquad (B.13)$$

and

$$q_2 = F_{A_2}\ F_{\varepsilon 2}\ A_2 \sigma \left[T_2^4 - T_3^4\right] \qquad (B.14)$$

Where the geometric view factors, F_{A1} and F_{A2}, between the spar and the respective environments are unity, and $A_1 = A_2$ = areas of the two sides of the spar



T_1 = interior environment temperature, ^oR
T_2 = mean temperature of the spar, ^oR
T_3 = exterior environment temperature, ^oR
$F_{\varepsilon 1}$ = radiation exchange factor between the spar and the interior
$F_{\varepsilon 2}$ = radiation exchange factor between the spar and the exterior

The radiation exchange factors through the multilayers of insulation are given by,

$$F_{\varepsilon 1} = \frac{1}{N_1 \left(\frac{1}{\varepsilon_M} + \frac{1}{\varepsilon_{AL}} - 1\right) + \left(\frac{1}{\varepsilon_C} + \frac{1}{\varepsilon_{SPAR}} - 1\right)} \quad (B.15)$$

$$F_{\varepsilon 2} = \frac{1}{N_2 \left(\frac{1}{\varepsilon_M} + \frac{1}{\varepsilon_{AL}} - 1\right) + \frac{1}{\varepsilon_{SPAR}}} \quad (B.16)$$

Where N_1, N_2 = number of layers of aluminized mylar on sides 1 and 2, respectively,

ε_M = emissivity of mylar
ε_{AL} = emissivity of aluminum
ε_C = emissivity of interior
ε_{SPAR} = emissivity of spar

For the values,

ε_M = 0.8
ε_{AL} = 0.05
ε_C = 0.9
ε_{SPAR} = 0.2

the above equation results in

$$F_{\varepsilon 1} = \frac{1}{20.25N_1 + 5.11}$$

$$F_{\varepsilon 2} = \frac{1}{20.25N_2 + 5}$$

For steady-state, the continuity of flow requires that $q_1 = q_2$. This requirement for the above equations, together with the numerical values

T_1 = 530^oR
T_2 = 520^oR
T_3 = 0^oR
$A_1 = A_2 = 18.8\ ft^2$

result in the following relationship between N_1 and N_2:

$N_2 = 12.2\ N_1 + 4.7$

The number of shields required to restrict the bending to about 100 arc seconds may be computed by using either of the above equations for heat flow for a value of q = 6.53 Btu/hr, and by using the above relationship between N_1 and N_2. This results in a value of N_1 = 1.2 or two layers with $N_2 \simeq 30$ layers. This shielding requirement is not impractical; 30 layers require less than an inch of thickness to achieve the optimum insulating effectiveness.

B.3.2 Transient Heat Transfer

Since the astronaut can null out the pointing error caused by steady-state temperature differences across the spar, then it is the transient temperature difference between the environment and the spar that will produce the pointing error.

The magnitude of the transient heat input to the two sides of the spar necessary to result in a one arc second per minute pointing drift can be computed using Eq. (B.7). That equation is

$$\theta = \frac{mc}{2(b_{1,2} + C_{1,2})} \ln \left[\frac{\Delta q}{\Delta q - 2(b_{1,2} + C_{1,2})\,\Delta T}\right] \quad (B.7)$$

Solving for Δq results in the expression:

$$\Delta q = \left\{\frac{2\,(b_{1,2} + c_{1,2})\,\Delta T}{e^{\left[\frac{2(b_{1,2} + c_{1,2})\,\theta}{mc}\right]} - 1}\right\}\left\{e^{\left[\frac{2\,(b_{1,2} + c_{1,2})\,\theta}{mc}\right]}\right\} \quad (B.17)$$

Substitute the following values into Eq. (B.17):

θ = 1/60 hr

m = the mass of one side of the aluminum spar

= $\frac{\Pi DL}{2}\,\tau\delta$

where δ = material density = 170 lb/ft^3 (aluminum)

D = spar diameter = 1 ft

τ = wall thickness = 0.16 in = 0.0133 ft

L = spar length = 12 ft

m = 42.7 lbs

$b_{1,2}$ = 0

$C_{1,2}$ = 20.4 Btu/hr - $^\circ$F

ΔT = 0.032 $^\circ$F

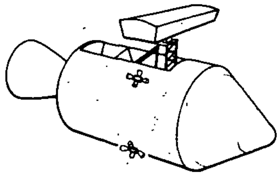
C = 0.2 Btu/lb - $^\circ$F

Δq = 16.97 Btu/hr

Since $\Delta q = q_1 - q_2$ and $q_1 = -q_2$

$\Delta q = 2q_1 = 2q_2$

or $q_1 = \Delta q/2$ = 8.485 Btu/hr



The transient heat transfer between the environment and the spar would be the result of the experiment instruments' exposure to the heat shield or skin temperature variations as a function of orbital position, the variation of heat dissipated from active electronic equipment, or the absorption of solar energy entering the instruments. For these reasons, the spar must be isolated from heat exchange with the experiments and the environment.

The experiments can be mounted on high thermal resistance mounts to reduce conductive transfer, and wrapping the spar with aluminized mylar will provide isolation from radiant heat transfer, as is shown above for the steady-state case.

Since the allowable transient heating rate, 8.485 Btu/hr, is greater than the allowable steady-state heating rate, 6.53 Btu/hr, the aluminized mylar shielding calculated previously for the steady-state case, two layers, is sufficient for the transient case also.

Variations in the exterior environment while ATM is pointing can result in the worst possible case for the transient bending. Reflections from the Apollo CSM to deep space can change the environment suddenly. For the steady-state case previously examined, the exterior environment was assumed to be deep space while the interior environment was assumed to have a temperature of 70°F. If the exterior environment were to change suddenly from 70°F to the -460°F of deep space, then, since it is shown above that a greater heat flow can be tolerated for the transient case, the 30 layers of aluminized mylar computed for the steady-state case is sufficient shielding for the transient case.

The calculations show that a maximum of 30 aluminized mylar shields to meet both Case 1 and Case 2 are required on the lower section of the spar that sees the exterior environment, and two shields are sufficient insulation for the upper section of the spar that sees the compartment interior. These numerical values are not to be considered as design values; they simply indicate the feasibility of the approach.

Appendix B
Section 4
CRITICAL COMPONENT THERMAL CONTROL

B.4 REQUIREMENTS

B.4.1 Solar Monitor Subsystem Filter-Solar Energy Input

Interference filters are installed between the 1 Å filter and the sun to limit the energy input to the filter to that portion of the solar spectrum in a 10 Å bandwidth centered on the Hα line. This band comprises about one percent of the total solar spectrum energy. The energy, q, strikes the 1 Å filter and is estimated by using the following equation:

$$q = (0.01)\ (E\pi D^2/4) \qquad (B.18)$$

where

E = the solar constant, 443 Btu/hr ft^2
D = aperture diameter

The factor 0.01 is for the one percent figure.

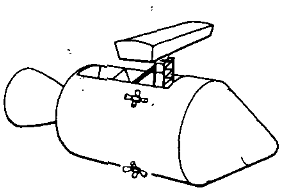
The telescope used with this system has an aperture, D, of 3.5 inches, or 0.292 ft.

These numerical values used with the above equation result in a value of q = 0.295 Btu/hr, or about 0.1 watts.

The 1 Å filter is small with an overall area of about two square inches. To radiate this energy to the environment would require a radiant heat transfer rate of 0.05 watts/in.^{2} or equivalently, 24.6 Btu/hr ft^2. At 70°F, the temperature difference between the filter and the environment required for a blackbody to radiate at this rate is on the order of 20°F. Since the 1 Å filter requires a temperature range for proper operation of 70 ± 9°F, it is necessary to provide a means of cooling the filter. Thermoelectric modules attached to the filter rim can be used to pump heat either out of the filter or into the filter (the latter for cold environments).

Thermoelectric modules can be expected to have an efficiency of about 10 percent for practical heat pumping applications. Hence, a power input of about one watt is required to pump the 0.1 watt-equivalent heat from the filter. The heat must be radiated from the modules to the local environment.

In order to maintain optimum heat pumping for the device, it is necessary to use a constant current power supply. Despite the variation of environmental temperature,



which results in a variation of the temperature difference across the device, the optimum current for heat pumping varies but little.*

The heat pumped by the thermoelectric device plus the Joule heating within the device is conducted to a radiating plate and radiated to the environment.

If the filter temperature falls below the required operating level, the current through the thermoelectric device is reversed and the device becomes a heater. A temperature sensor located on the filter edge can be used as a sensing device for the current reversing electrical network for the purpose of controlling the filter temperature.

B.4.2 Solar Monitor Vidicon

The solar monitor vidicon is located within the mounting spar, directly behind the 1 Å filter. For proper operation the temperature of the vidicon must be kept below about 120°F. Also, its electrical power input is dissipated as heat which must be channeled outside the attachment spar. Otherwise, the spar would bend if such heat were allowed to flow into the spar.

A method of channelling the vidicon heat outside the spar and maintaining the vidicon temperature well below 120°F is illustrated in Fig. B-3.

The vidicon is enclosed in a metal case: a copper sheath encloses the metal case and connects it thermally to a radiation plate which is exposed to the exterior environment, deep space, and at times to the Apollo CSM. The conductive heat transfer between the vidicon and the sheath is expressed by the equation:

$$q = C_1 A\,(T_1 - T_2) \qquad \text{(B.19)}$$

where
- C_1 = contact thermal conductance between the vidicon case and the sheath
- A = area of contact
- T_1 = vidicon temperature
- T_2 = sheath temperature

The numerical value of C_1 is about 100 Btu/hr ft^2°F for an assumed contact pressure of 100 psi.

The value of the vidicon electrical power input, q, is 5 watts, or 17.06 Btu/hr.

For an area, A, of about 0.613 ft^2 corresponding to a 7-inch length and the dimensions

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*This is shown mathematically in "Thermoelectric Materials and Devices", edited by I.B. Cadoff and E. Miller, Reinhold Publishing Corp., New York, 1960, Page 252.

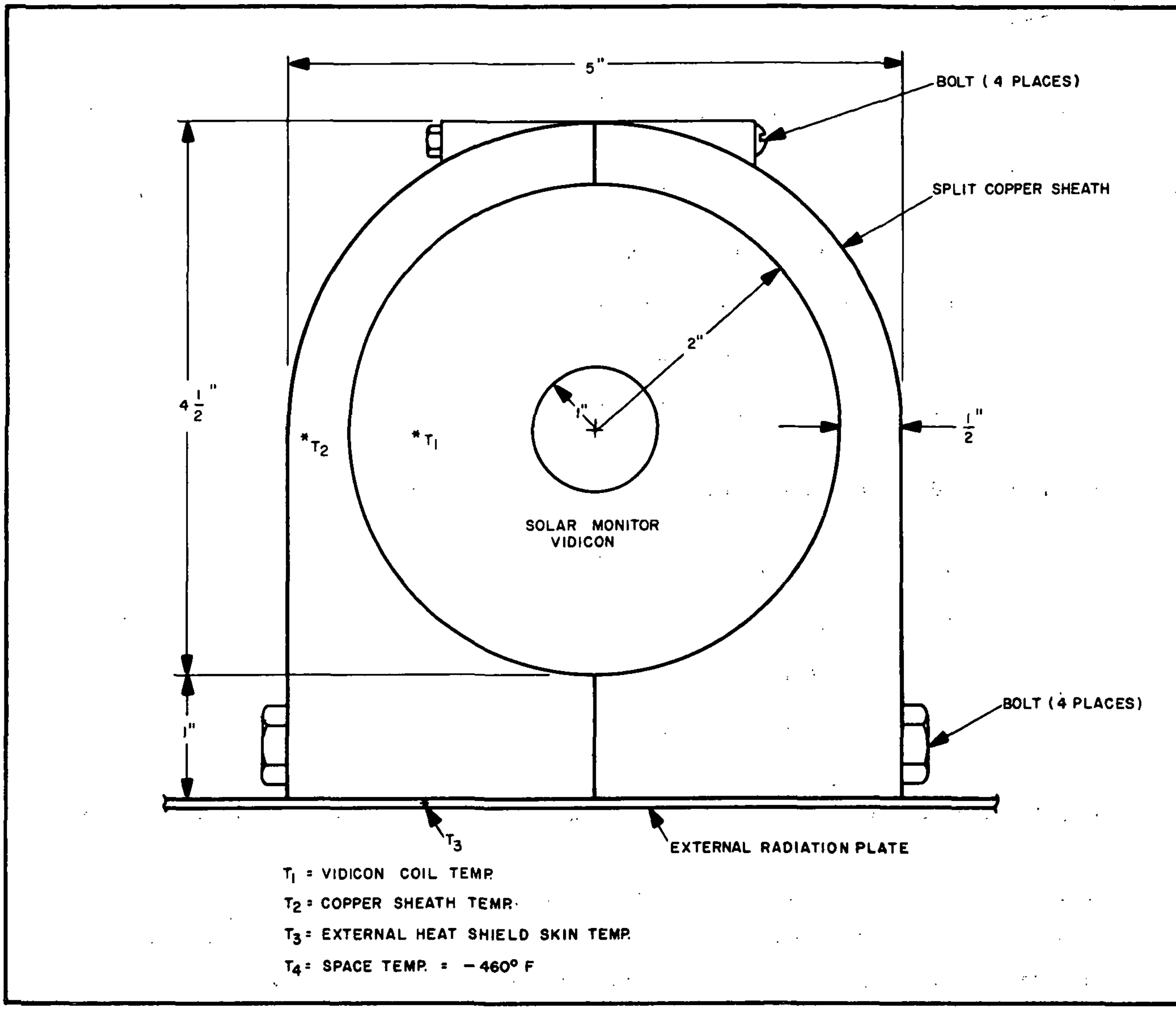
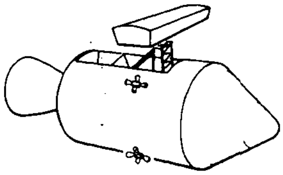


Fig. B-3 Solar Monitor Vidicon Heat Transfer

shown in Fig. B-3, and an assumed sheath temperature of 70°F, the resulting value of T_1 in the above equation is 70.28°F. Thus, the temperature of the vidicon is only slightly greater than the copper sheath temperature for the required heat transfer. The radiating plate temperature below the vidicon must be maintained at a mean temperature so that the copper sheath is held at 70°F and loses heat to the radiating plate at a rate of 17.065 Btu/hr. The heat transfer is expressed by the equation:

$$q = \left(\frac{kA_2}{(_L\bar{A}_2)} + C_2 A_3\right) (T_2 - T_3) \qquad (B.20)$$



where A_2 = mean cross section area for heat transfer in the copper sheath

A_3 = contact area between the copper sheath and the radiating plate

k = conductivity of copper = 220 Btu/hr - ft^2 - °F/ft

L = average heat transfer path length in the copper sheath

C_2 = contact conductance

T_3 = radiating plate temperature

For the above described dimensions,

$A_2 \approx 0.0486$ ft^2

$A_3 \approx 0.243$ ft^2

$L \approx 0.25$ ft

$C_2 \approx 100$ Btu/hr ft^2°F (as for C_1 above)

Using these values and T_2 = 70°F and solving the above for T_3 results in T_3 = 69.75°F.

Thus, it is necessary to maintain the radiating plate temperature at 69.75°F in order to maintain the vidicon temperature at 70.28°F. This indicates on a practical level that the vidicon temperature will be approximately that of the radiating plate. The temperature of the latter is a function of its surface thermal radiation properties and of the exterior environment. The heat transfer rate by radiation to deep space, the principal exterior environment, is expressed by the equation:

$$q = \varepsilon A_3 \sigma T_3^4 \qquad \text{(B.21)}$$

where ε = surface emissivity

σ = Stefan-Boltzmann constant

It is necessary to radiate as heat to deep space the vidicon power, q, of 5 watts, or 17.06 Btu/hr, at $T_3 \approx$ 70°F or 530°R. Using these numerical values in the above equation and solving for the required plate emissivity, ε, results in ε = 0.25. Since values of ε from 0.05 to 0.99 are easily provided, this design approach appears feasible for maintaining the vidicon temperature below 120°F.

Appendix B

Section 5

INTERIOR HEATING NECESSARY TO MAINTAIN A CONSTANT TEMPERATURE

B.5 REQUIREMENTS

To maintain an experiment compartment temperature to within ± 5°F of some absolute value requires an active heating system. In order to avoid an active heating plus a cooling system, a heating system only can be used. This system requires that the heat flow from inside to outside; the vehicle skin temperature must remain lower than the experiment compartment temperature.

The design philosophy of this system is to provide a low skin temperature and a heat flow impedance from inside to outside of the degree necessary to restrict internal heat losses to those that can be maintained with the available power.

The thermal impedance between the interior and exterior is provided principally by an inner liner shield that is attached to the inside surface of the outer radiation shield by means of insulation standoffs.

The internal power will fluctuate with experiment use; therefore, it is necessary to use heaters to supply the balance of power required to maintain the temperature level. Due to the lack of space inside the compartment, the heaters must be of low density (high surface area) and attached to the inner liner shield, again with standoffs. Thus, the heaters themselves become part of the thermal impedance between inside and outside.

The rate of radiative heat transfer between the compartment interior and the outer heat shield is expressed mathematically as

$$q_H = F_A F_\varepsilon A\sigma \left[T_c^4 - T_s^4\right] \qquad (B.22)$$

where

F_A = the geometric view factor between the interior and the shield

F_ε = the radiation exchange factor, or emissivity factor between the interior and the shield

A = the effective area of the interior surface

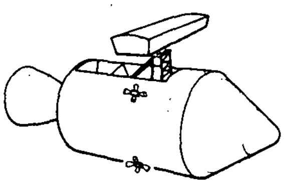
T_c = compartment temperature, absolute

T_s = outer shield temperature, absolute

σ = Stefan-Boltzmann constant

The value of F_ε is obtained from the equation

$$F_\varepsilon = \frac{1}{\left(\frac{1}{\varepsilon_1} + \frac{1}{\varepsilon_2} - 1\right) + \left(\frac{1}{\varepsilon_3} + \frac{1}{\varepsilon_4} - 1\right) + \left(\frac{1}{\varepsilon_5} + \frac{1}{\varepsilon_6} - 1\right)} \qquad (B.23)$$



where

ε_1 = emissivity of experiments and compartment components
ε_2 = emissivity of the heater surface
ε_3 = emissivity of the heater rear surfaces
ε_4 = emissivity of the inner liner shield front surface
ε_5 = emissivity of the inner liner shield back surface
ε_6 = emissivity of the outer shield inner surface

These parameters are given the following values:

ε_1 = 1.0
ε_2 = 1.0
ε_3 = 1.0
ε_4 = 0.1
ε_5 = 0.1
ε_6 = 0.4
T_c = 60°F or 520°R
T_s = 30°F or 490°R

In the absence of any other internal heat dissipation, the heaters required output would then be from Eqs. (B.22) and (B.23), q_H = 140 Btu/hr, or 41 watts. Since 65 watts are dissipated as heat if all the instruments operate simultaneously, it is obvious that the internal temperature will then rise. If, however, the shielding emissivities are adjusted to accommodate such a mode of operation, the internal heating in absence of experiment operation would then be 65 watts continuously in order to maintain the required temperature level. Thus, the variations in possible experiment operations introduce a problem concerning power and shielding requirements.

In addition to the above variations, the actual outer shield temperature can be expected to vary considerably from the 30°F temperature for which the exterior surface finishes are designed (Section 6). These variations can be due to seasonal variations of the solar constant, earth albedo variations caused by changing cloud cover, uncertainties in surface finish emissivities, and deterioration of surface finishes due to Apollo waste expulsion. The first three can be expected to result in temperature variations of the order of ±6°F, ±10°F, and ±10°F, respectively. The effect of the fourth item is unknown. Thus, the above 30°F temperature could be 0°F or 60°F; however, surface finish degradation might be expected to result in lower compartment temperatures from greater heat losses.

If the skin temperature were 0°F, the required heater power to maintain a 60°F interior would be from the above equation, 75 watts in the absence of other power sources.

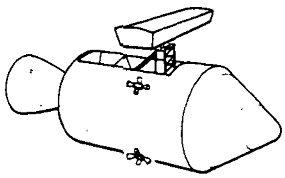
The order-of-magnitude power requirements for a 14-day mission can be estimated from the overall electrical energy (excluding thermal control energy) of 11 kw-hr required for the mission. This would amount to an average of 33 watts. At the above figure of 75 watts for a 0°F skin temperature, the average heater power required would be 42 watts for the 14-day mission. As a conservative measure, 55 watts for 14 days is specified for heater circuits. Thus, the 75-watt heaters could operate 70 percent of the time. An additional five watts is allowed for the remainder of the thermal control system.

It is apparent that the above thermal control lacks the flexibility required for a general, random environment and a random sequence for experimental operation. An additional system must be imposed for these conditions.

An additional system to reduce the heating requirement and permit variations in the external heat shield surface temperatures can be devised using mechanical heat shield louvres or shutters, though reliability now becomes a factor. This system would have a high thermal resistance like the inner liner shield when the louvres were closed and a low thermal resistance when fully open. Such a system can be designed to allow operation of all the experiments when the louvres are open, and require internal heating of the above magnitude or less when the louvres are closed. The analysis of such a system must await more detailed design studies.

8.5.1 Heater Power Control

The active heating system concept consists of eight heating zones. Each zone temperature is controlled by an independent temperature sensor mounted in the environment. The sensor is shielded from the heaters and "sees" the experiments. The sensor and control system activates the heaters when the temperature falls 5°F below the preset temperature and de-energizes the heaters when the temperature rises 5°F above the preset temperature.



Appendix B

Section 6

AN ESTIMATION OF THE HEAT SHIELD MEAN ORBITAL TEMPERATURE FOR A GIVEN ORBIT AND VEHICLE ATTITUDE

B.6 REQUIREMENTS

The mean orbital temperature of the heat shield for a given orbit and a given Apollo spacecraft orientation with respect to the earth is calculated by forming a mathematical heat balance upon the ATM system. This results in the equation:

$$\sigma(b_s + b_1)\overline{T^4} = q_{in} + \overline{\alpha_s G_s} + \overline{\alpha_A G_A} + \overline{\alpha_e G_e} + \alpha b_1 \overline{T_1^{\,4}} + \overline{G_1}(1-\alpha_1)F_1\alpha_s \qquad (B.24)$$

where

σ = Stefan-Boltzmann constant

b_s = radiation exchange factor with space

b_1 = radiation exchange factor with sector one

$\overline{T^4}$ = mean fourth power temperature of shield (averaged over the entire surface and over one orbital period)

q_{in} = internal power dissipation

$\overline{\alpha_s G_s}$ = mean solar energy absorbed by the heat shield over one orbital period. The bar indicates the weighting of the heat input by the absorptivity α_s for each portion of the shield area. G_s is the product of the area and the heat flux.

$\overline{\alpha_A G_A}$ = mean albedo energy absorbed by the heat shield over one orbital period

$\overline{\alpha_e G_e}$ = mean earthshine energy absorbed by the heat shield over one orbital period

$\overline{T_1^{\,4}}$ = mean fourth power orbital temperature of sector one (°R)+

$\overline{G_1}$ = the mean orbital solar albedo and earthshine energy impinging upon Sector I

$1-\alpha_1$ = the reflectivity of Sector I for the energy, assumed to be applicable for all three fluxes

F_1 = the geometrical view factor between Sector I and the ATM heat shield

The last term in the equation accounts for the energy reflected from the bay onto and absorbed by the heat shield.

Since $\overline{G_1} = A_1\overline{q_1}$, where $\overline{q_1}$ is the unit area heat input to the bay, then

$$A_1\overline{q_1}(1-\alpha_1)F_1\alpha_s = b'\overline{q_1}(1-\alpha_1)\alpha_s$$

where $b' = A_1F_1 = A_sF_s$

F_s = 0.2, the assumed view factor between the heat shield and the bay.

$b' = (113)(0.2) = 22.6$

The orbit assumed for this mean temperature estimation will be a circular equatorial orbit during the vernal equinox. The altitude is assumed to be 200 statute miles.

The values of the radiation exchange factors, b_s and b_1, can be calculated accurately using digital computer techniques; however, for the purposes of this order of magnitude temperature calculation their values will be estimated. The value of b_s is estimated to be:

$$b_s = E_H F_{Hs} A_H + E_F E_{F,s} A_f \tag{B.25}$$

where

E_H = the emissivity of the heat shield, constant over the entire surface, assumed to be 0.4

$F_{H,s}$ = the geometric view factor between the shield and space, assumed to be about 0.8

A_H = area of heat shield surface, assumed to be about 113 ft^2 for a cylinder 3' x 12'

$F_{F,s}$ = geometric view factor between the sun facing end of shield and space

A_F = area of sun facing end, about 7 ft^2

$E_F = E_H$

Substituting these numerical values into Eq. (B.25) results in the value:

$$b_s = (0.4)(0.8)(113) + (0.4)(1.0)(7) = 39.0$$

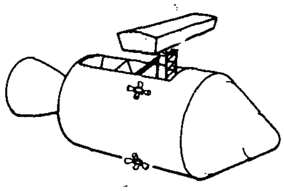
The value of b_1 is estimated assuming that the effect of multiple reflections is small.

$$b_1 = E_H E_s F_{H,1} A_H \tag{B.26}$$

E_s = the emissivity of the bay, assumed to be a diffusely radiating surface

$F_{H,1}$ = the geometric view factor between the shield and the bay, estimated to be about 0.2

and the other parameters have the same values as given for Eq. (B.25).



This results in the value:

$$b_1 = (0.4)(0.4)(0.2)(113) = 3.61$$

The value assumed for $F_{H,1}$ is probably high and, as a result, the estimated mean temperature of the heat shield may be high from that assumption alone.

It is assumed that, including absorbed solar energy, a total of 25 watts, or 85 Btu/hr, is the value of q_{in}. Most of the entering solar energy is rejected back out the entrance apertures by reflecting devices for most of the experiments.

It is assumed that the exterior surface finish of the heat shield is uniform so that α_s, α_A, and α_e are constants with respect to the surface; also, $\alpha_s = \alpha_A = \alpha_e = \alpha_1 = 0.4$.

As a result $\overline{\alpha G} = \alpha\overline{G}$.

The values of $\overline{G_s}$, $\overline{G_A}$ and $\overline{G_e}$ can be computed for given spacecraft attitudes and orbits; however, for the present calculations their values are obtained by approximate methods.

It is assumed here that the Apollo spacecraft is oriented so that the bay faces the sun while the ATM system remains pointed toward the sun according to the sketch given in Fig. B-4.

The heat fluxes (solar, albedo and earthshine) are obtained for four opposite sides of the cylindrical shield at the four positions shown in Fig. B-4 from Vidya Technical Note 9026.20-TN-1, Vidya Division of Itek Corporation, 29 October 1963. The heat fluxes to the bay are obtained from the report of North American Aviation, Inc., SID65-266, Fig. 70. p.194.

The resulting estimated orbital fluxes for the sides of the heat shield are shown in Figs. B-5, B-6, and B-7. The orbital mean heat fluxes impinging upon the sun-facing surface of the ATM heat shield are the same as those impinging upon the bay; these mean orbital fluxes are about 264 Btu/hr ft^2, 6 Btu/hr ft^2, and 41 Btu/hr ft^2 for solar, albedo and earthshine, respectively.

The value of $T_1 \doteq 80°F$, or 540°R as taken from the above North American Aviation reference, p.190. The numerical values of the terms in the heat balance equation are as follows:

$$q_{in} = 85 \text{ Btu/hr}$$
$$\overline{\alpha_s G_s} = (0.4)(7 \text{ ft}^2)(264 \text{ Btu/hr ft}^2) = 740 \text{ Btu/hr}$$
$$\overline{\alpha_A G_A} = (0.4)\,[(113 \text{ ft}^2)(9.7) + (7)(6)] = 456 \text{ Btu/hr}$$

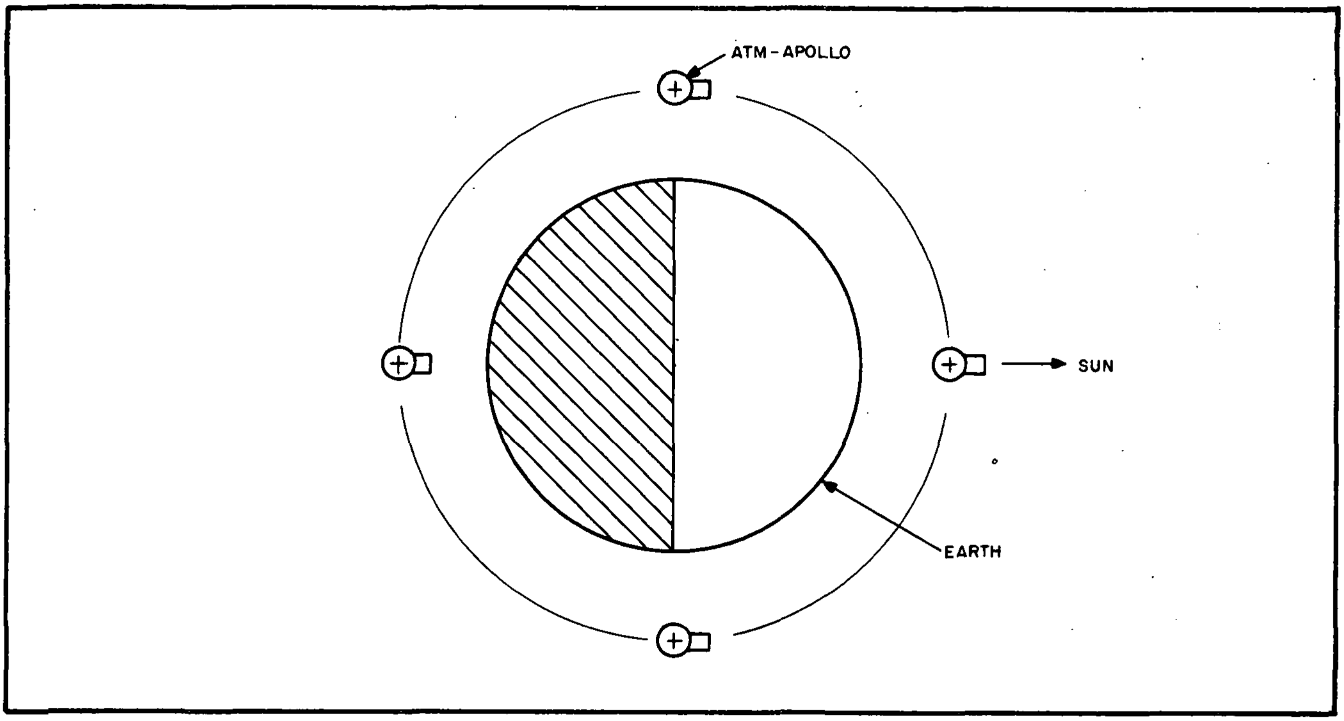


Fig. B-4 Apollo Spacecraft Orientation

$$\overline{\alpha_e G_e} = (0.4)[(113)(19.5) + (7)(41)] = 995 \text{ Btu/hr}$$

$$\sigma b_1 \overline{T_1^{\,4}} = (\sigma)(3.61)(540)^4 = 522 \text{ Btu/hr}$$

$$\overline{G}_1 \, (1-\alpha_1) b_1 \alpha_s = (311)(0.6)(22.6)(0.4) = 1685 \text{ Btu/hr}$$

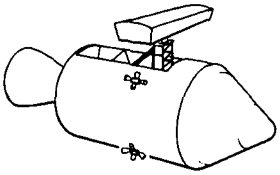
These values result in a value of $\overline{T^4}$ of

$$\overline{T^4} = 614 \times 10^8$$

Accordingly, the estimated value of the mean orbital temperature of the ATM heat shield is

$$[\overline{T^4}]^{1/4} = 496°R \text{ or } 36°F$$

It is obvious from the above results that the use of outer surface finishes to limit the heat from direct heat fluxes and from reflected heat fluxes will result in lower temperatures; and, conversely, outer surface finishes can be used to raise the mean orbital temperatures.



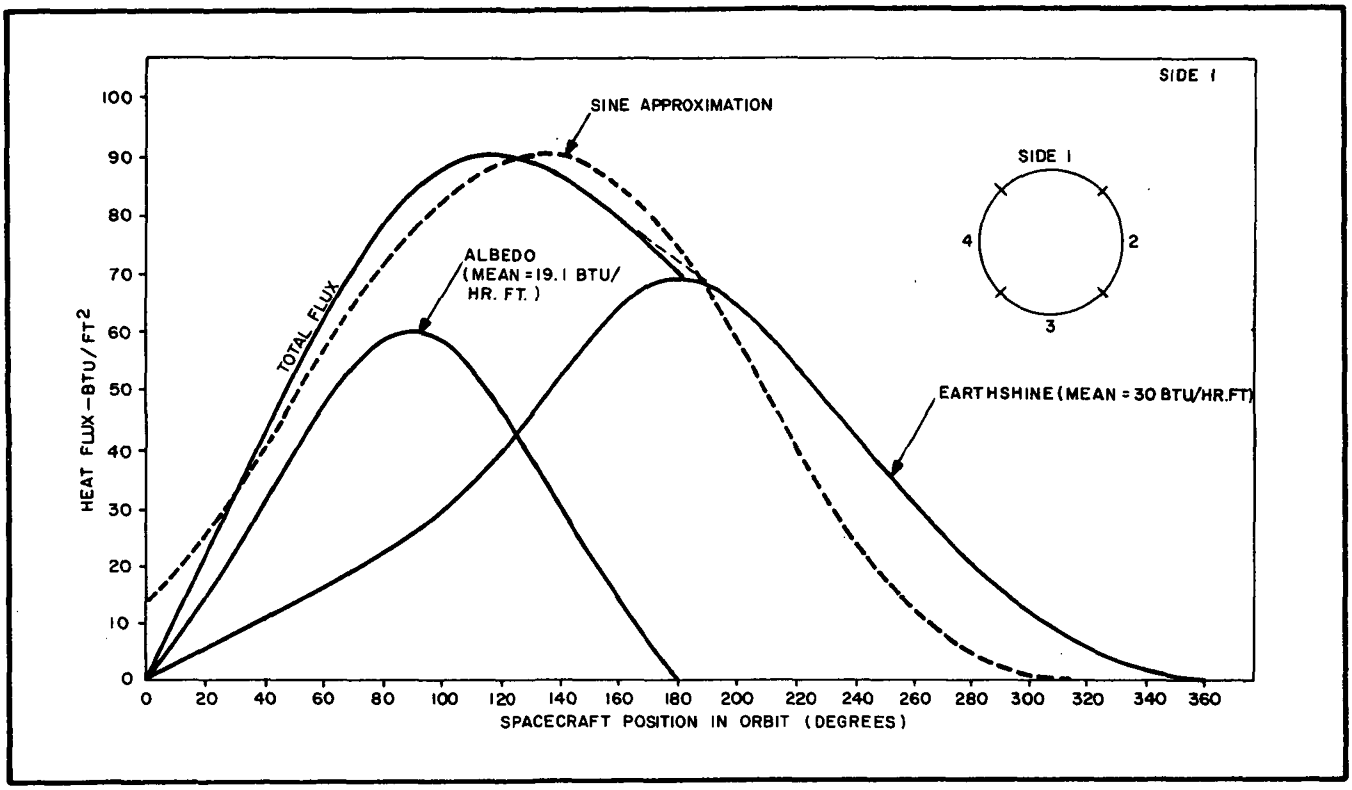


Fig. B-5 Estimated Orbital Heat Fluxes to Side One of the Heat Shield

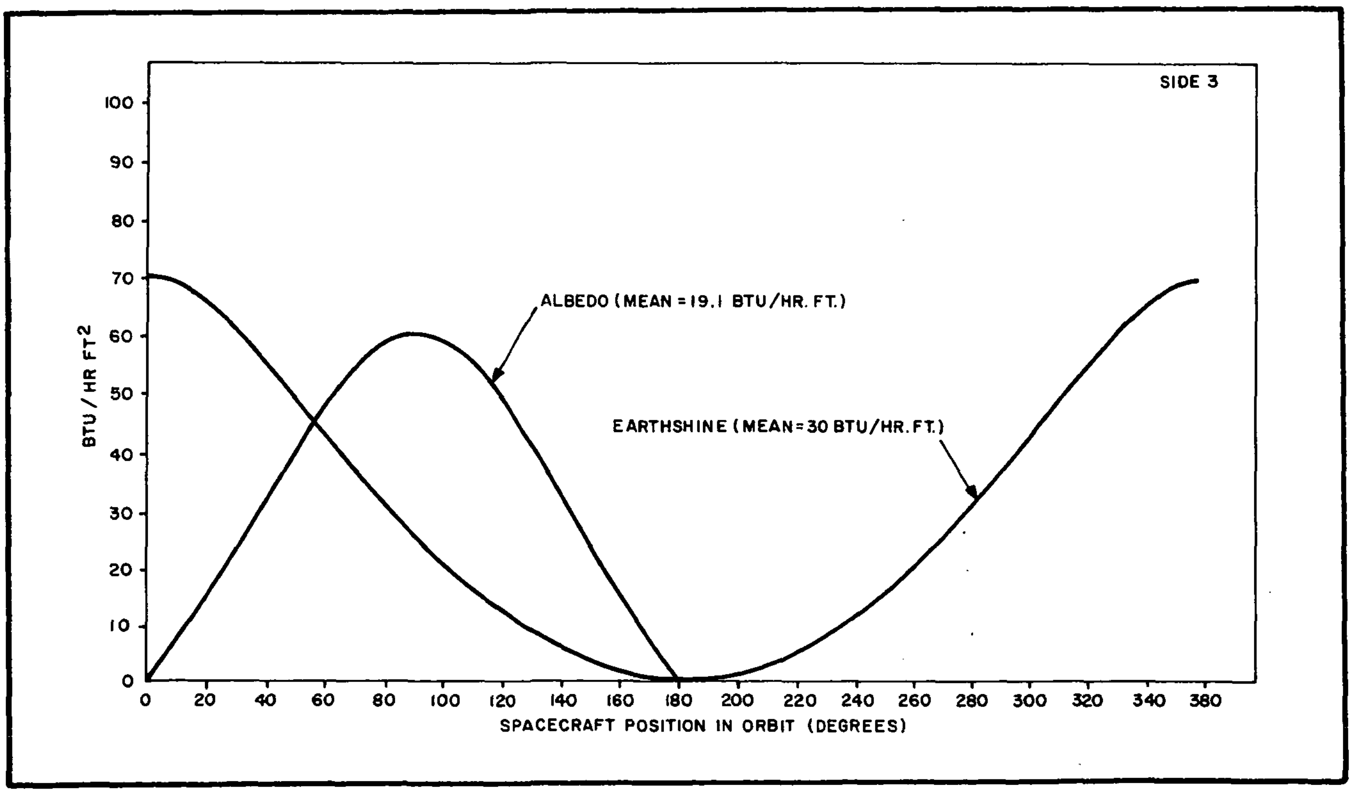


Fig. B-6 Estimated Orbital Heat Fluxes to Side Three of the Heat Shield

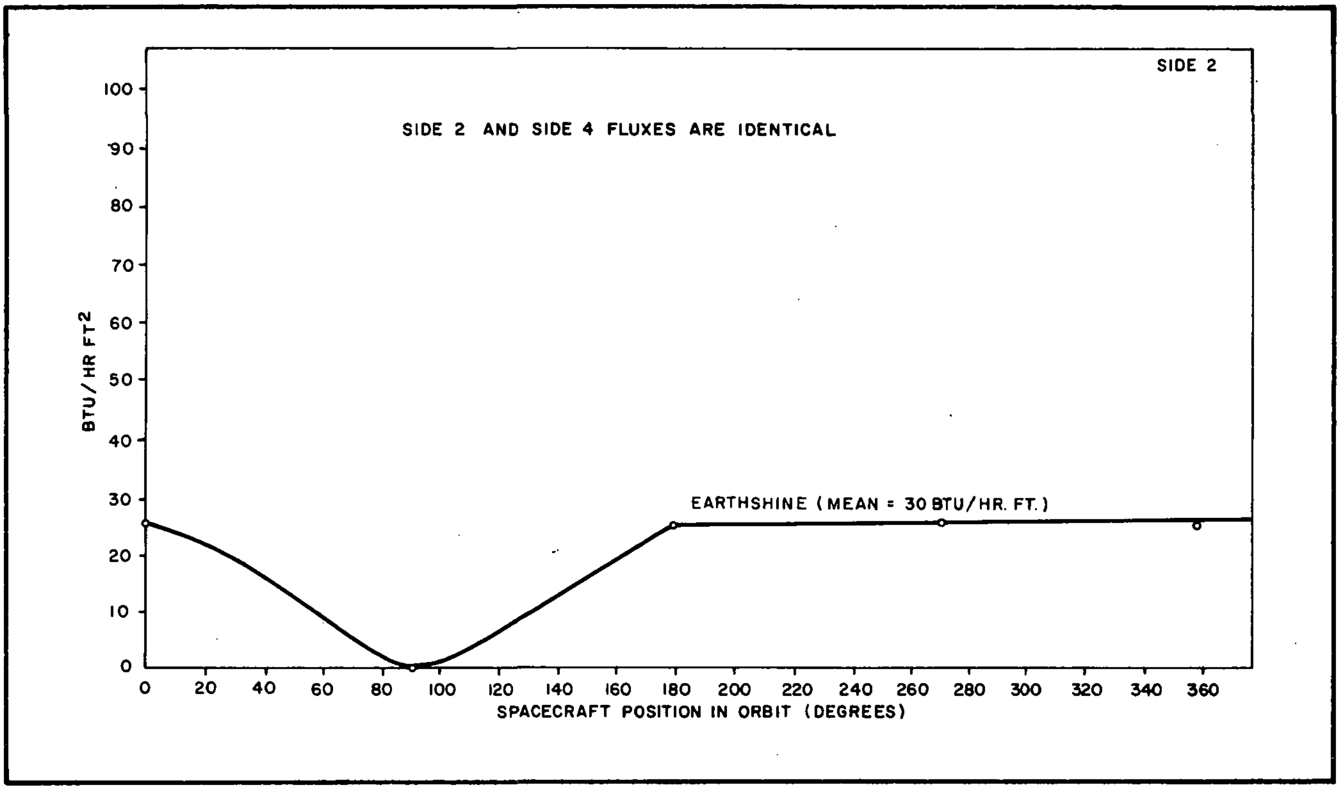


Fig. B-7 Estimated Orbital Heat Fluxes to Sides Two and Four of the Heat Shield

APPENDIX C

APOLLO COBALT 60 RADIATION EFFECTS ON ATM FILM

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C.1 INTRODUCTION

This appendix presents: (1) the calculation of the integrated cobalt 60 dose at several locations on the ATM; (2) the description and results of tests conducted during the study to determine the effect of this radiation on typical film; and (3) the correlation of film fogging at the selected ATM positions. The Apollo RCS fuel gauging system uses cobalt 60 to measure the balances in the fuel and oxidizer tanks. The cobalt 60 radiates gamma rays which may affect ATM experiments, particularly the film.

C.2 INTEGRATED COBALT 60 DOSE

The Apollo SM cobalt 60 is distributed on the eight fuel and oxidizer tanks that are used in the RCS system shown in Fig. C-1. It is distributed as a blanket which covers the outside of each tank. The cumulative total of this radiation is 25 millicuries. Isodose curves of the radial intensity for each fuel and oxidizer tank are presented in Figs. C-2 through C-7.* These curves show the measured dose from three views of both the fuel and oxidizer tanks.

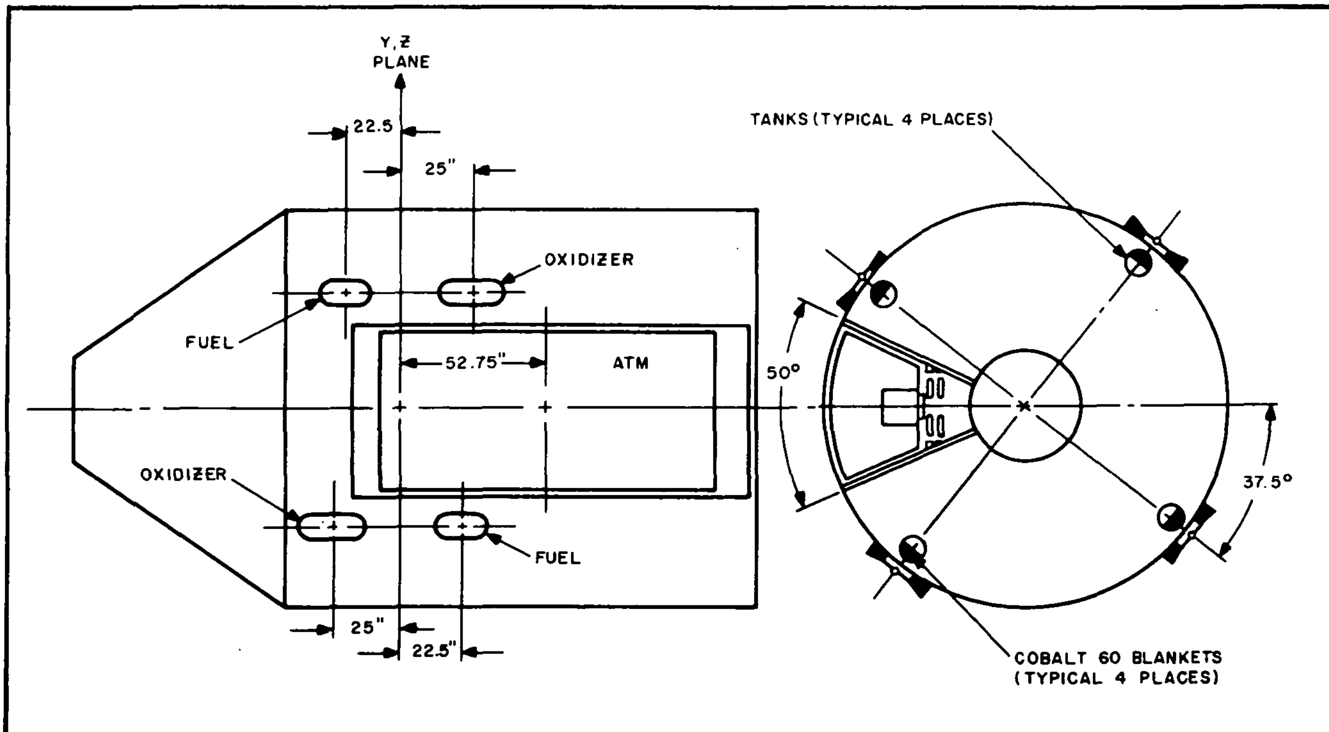
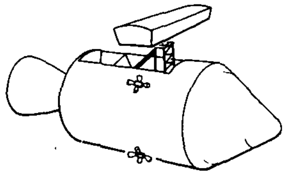


Fig. C-1 Apollo RCS Fuel and Oxidizer Tank Relationship to ATM

*Isodose curves from NAA.



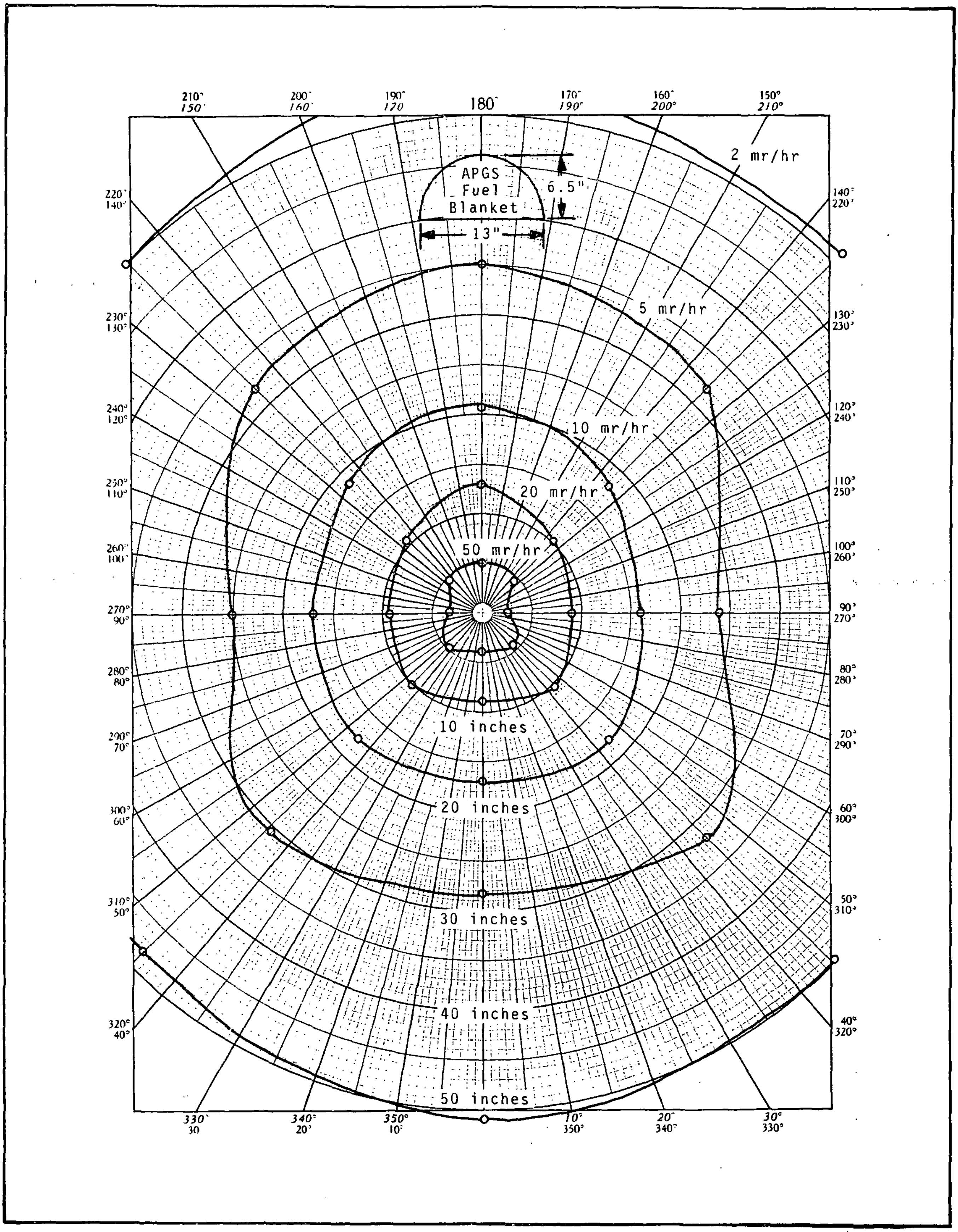


Fig. C-2 Apollo Isodose Plot - End View, Fuel Blanket

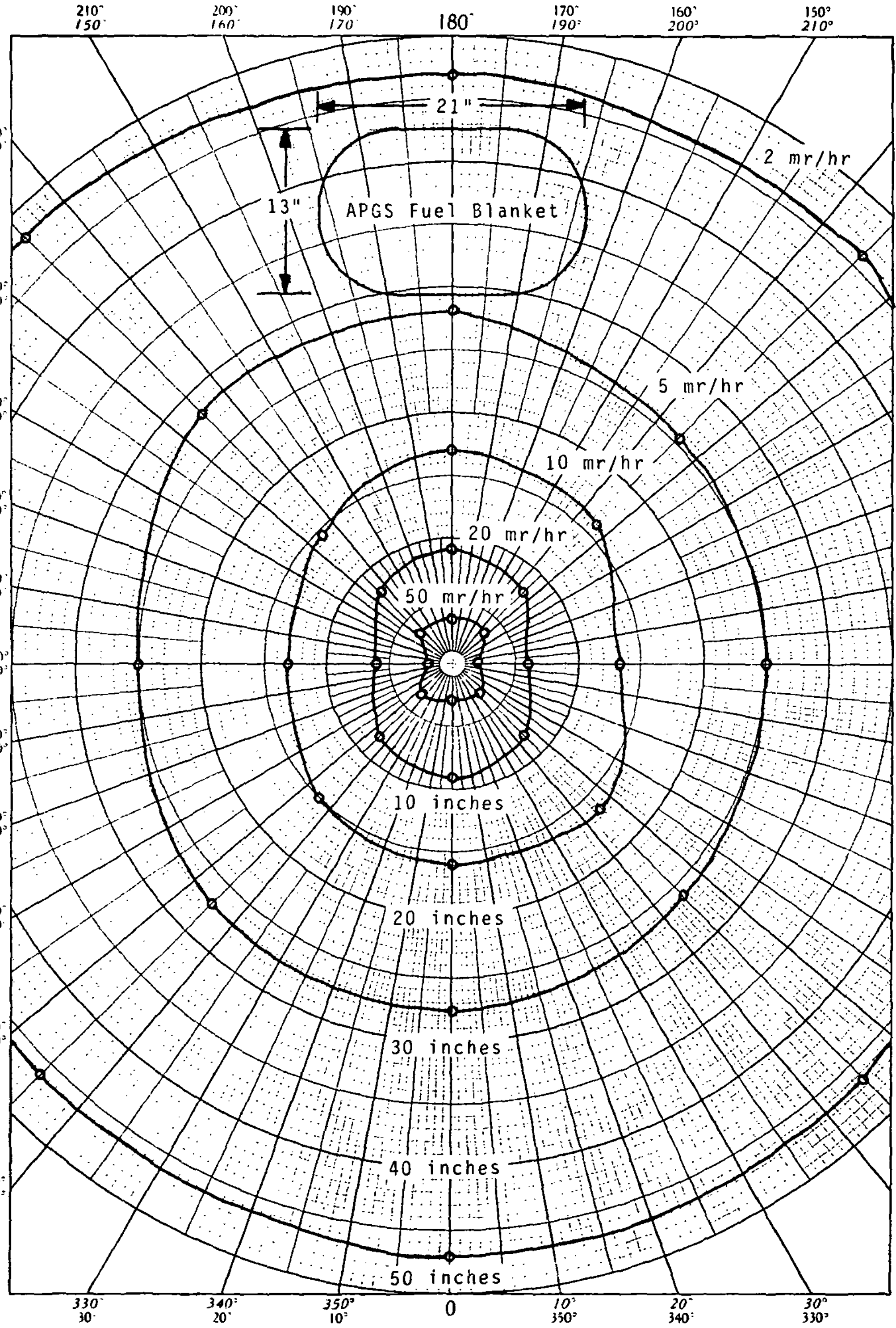
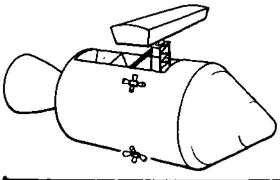


Fig. C-3 Apollo Isodose Plot - Top View, Fuel Blanket



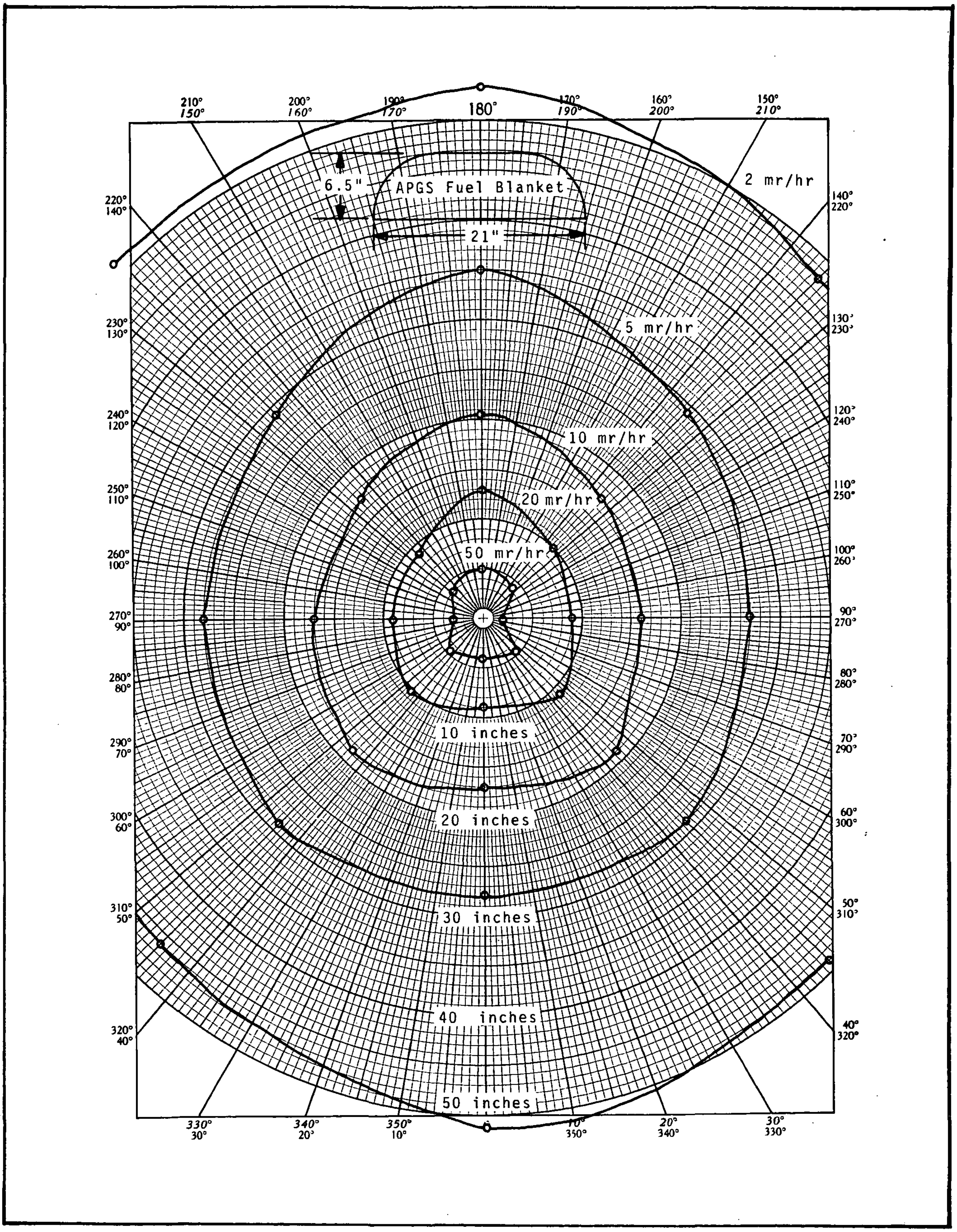


Fig. C-4 Apollo Isodose Plot - Side View, Fuel Blanket

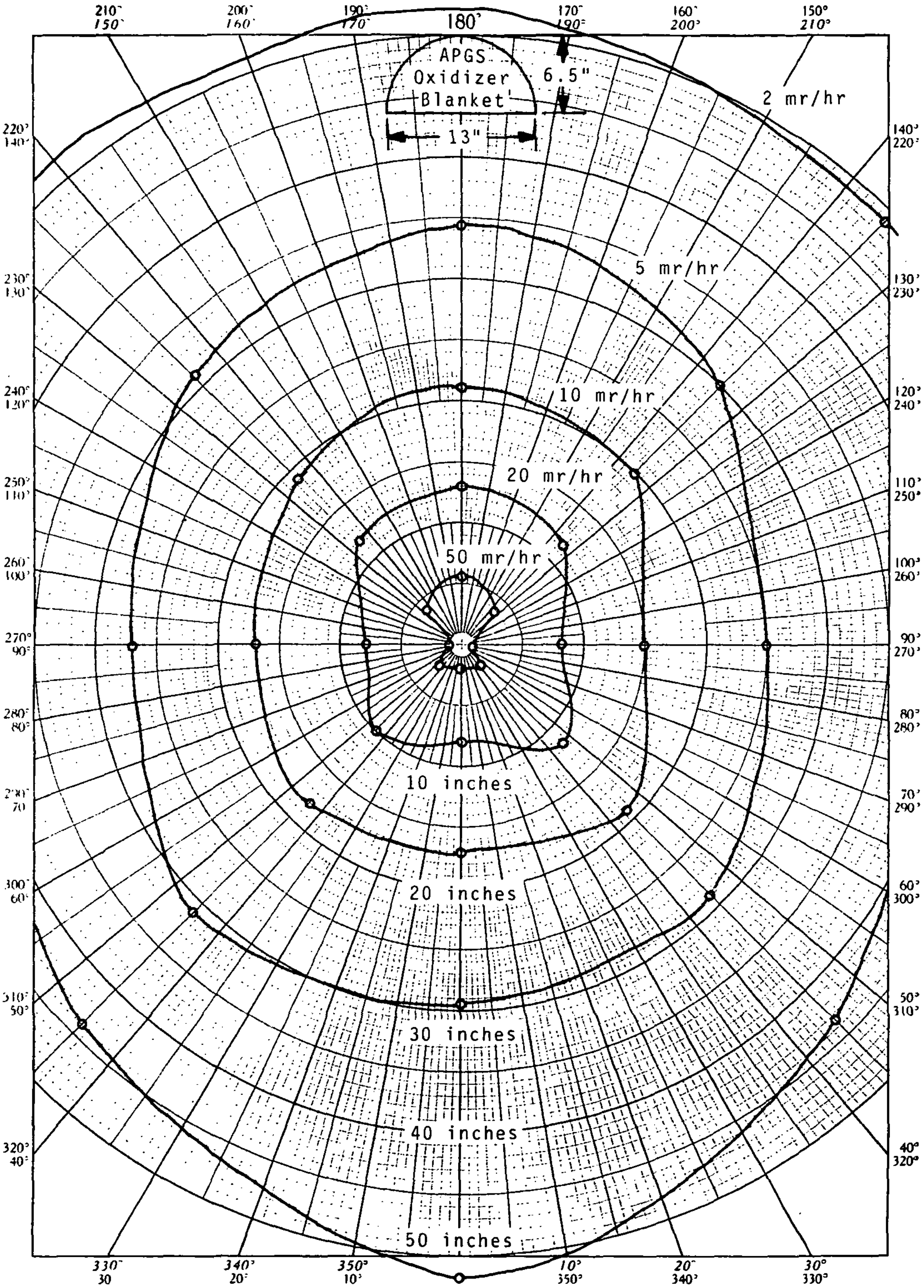
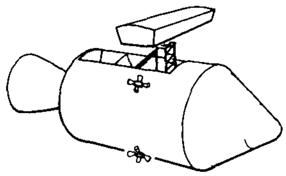


Fig. C-5 'Apollo Isodose Plot - End View, Oxidizer Blanket



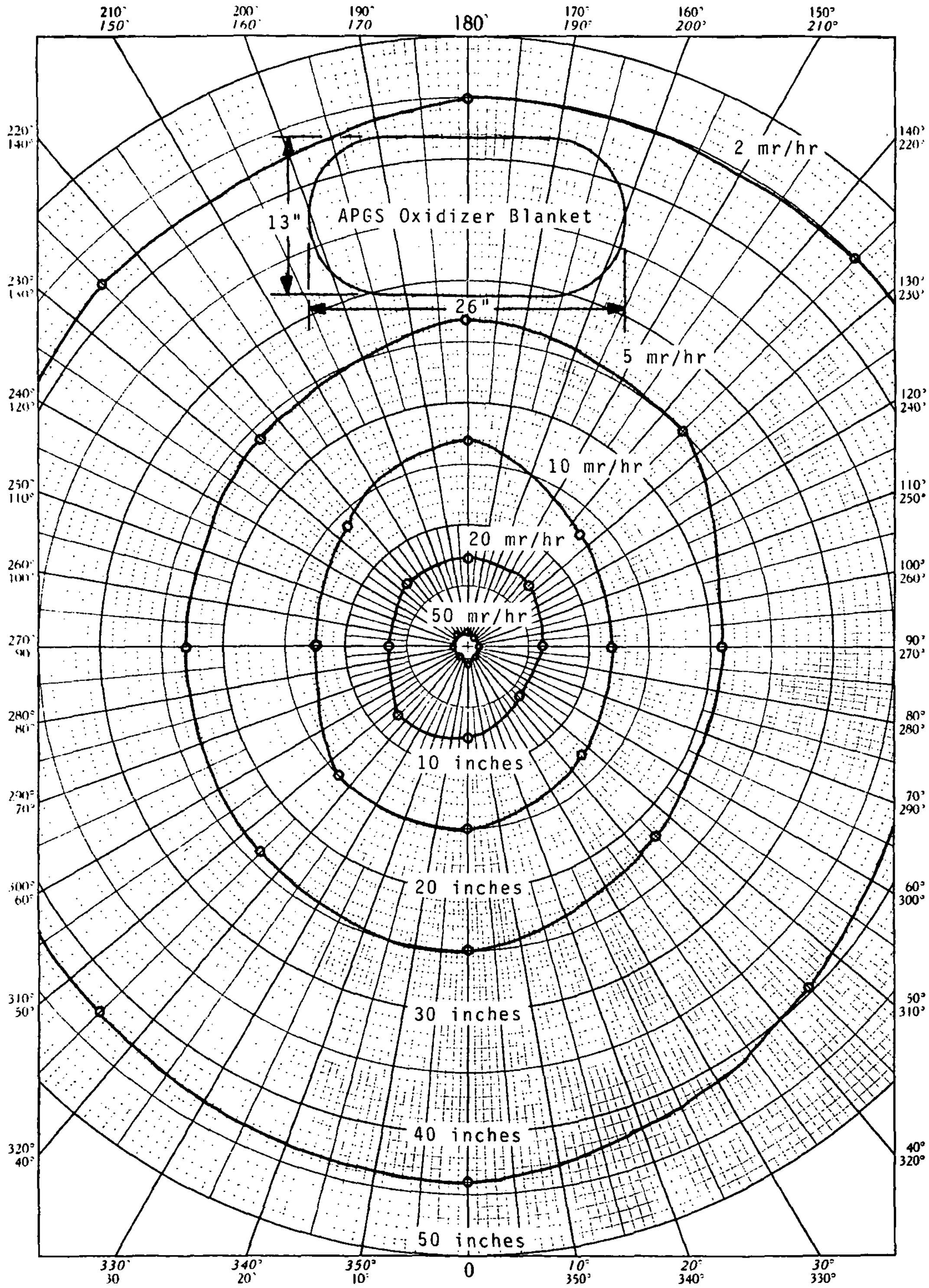


Fig. C-6 Apollo Isodose Plot - Top View, Oxidizer Blanket

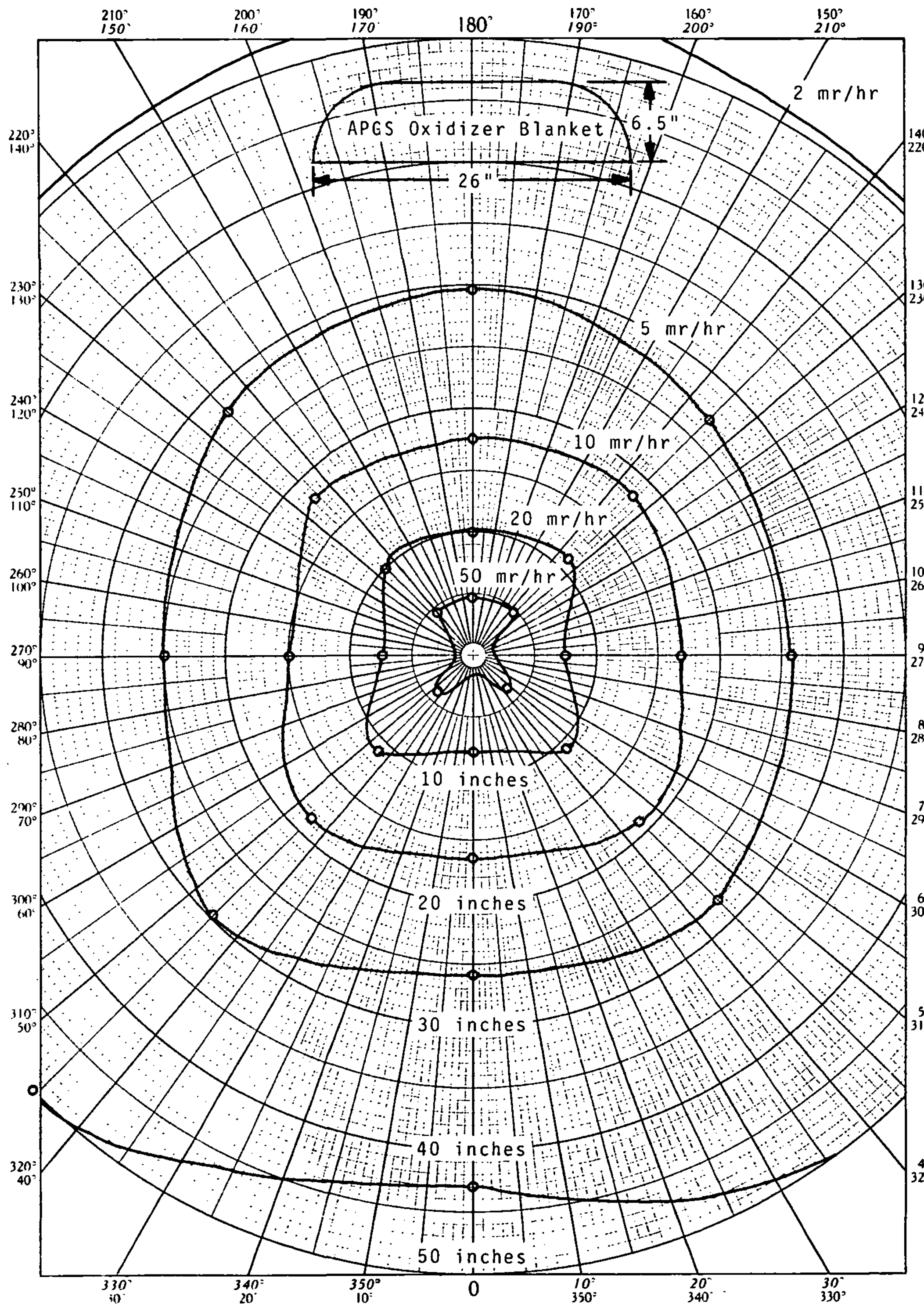
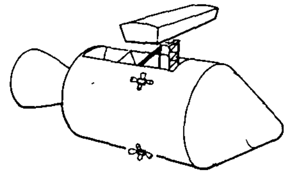


Fig. C-7 Apollo Isodose Plot - Side View, Oxidizer Blanket



Figures C-2 and C-5 show the dose in a plane through the center of the fuel and oxidizer tank, respectively, indicated as section A-A in Fig. C-8. Similarly, Figs. C-3 and C-6 show the section B-B, and Figs. C-4 and C-7 show the section C-C in Fig. C-8.

The fuel and oxidizer tanks for each thruster quad are at the same Y and Z stations, but are located at alternate X-stations between quads as indicated in Fig. C-1. The Y-Z plane passes approximately midway between a pair of tanks, and is 51.25 inches forward of the center of the stowed ATM. The ATM is located approximately midway between two quads as shown in Fig. C-1

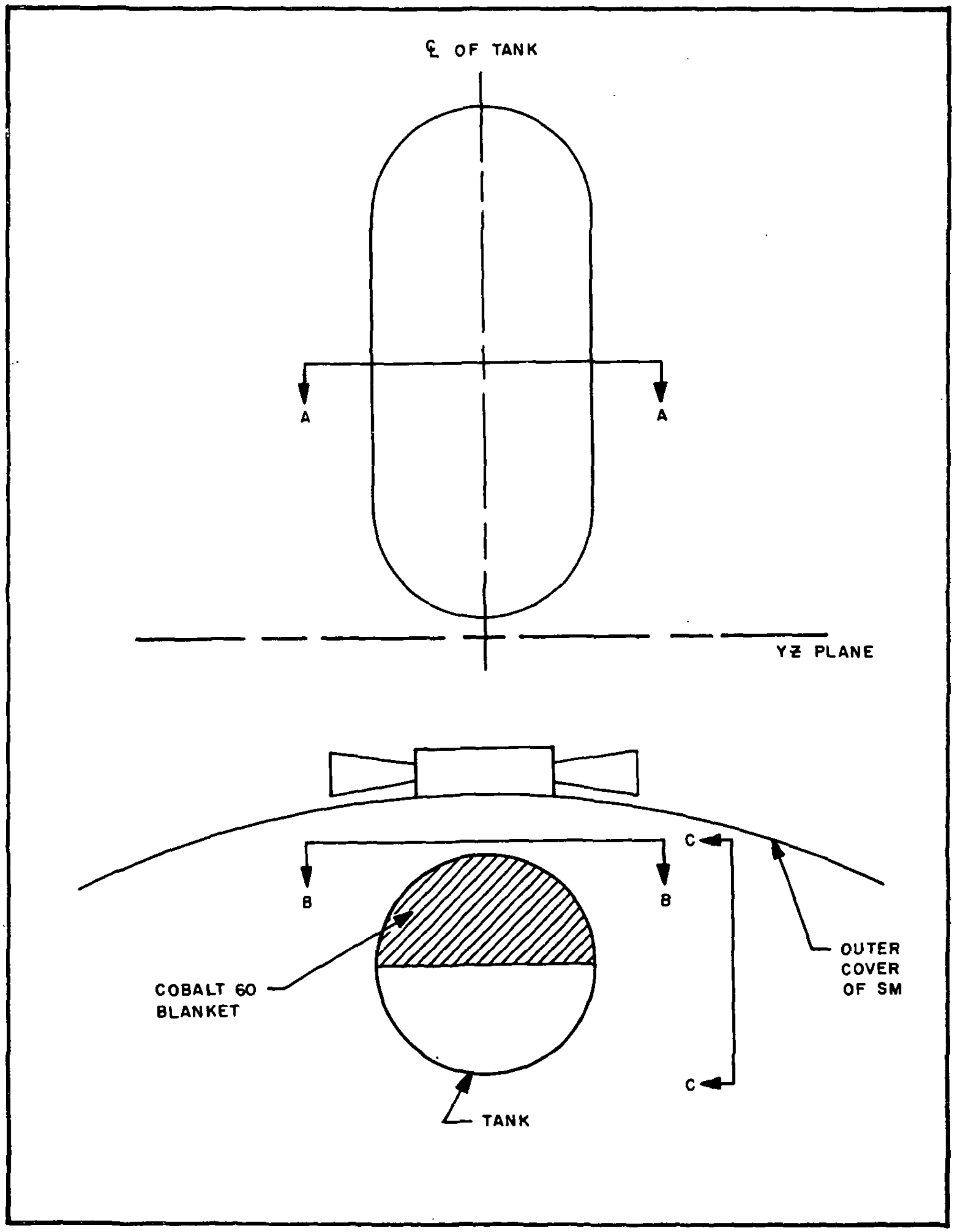


Fig. C-8 Tank Views Corresponding to Isodose Curves

In order to determine the specific radiation dose at the ATM, several locations were selected which approximate the possible positions of ATM cameras. These locations are shown in Fig. C-9. Locations P3 and P4 are on a cross section plane through the center of the ATM oriented section, and locations P1 and P2 are on a cross section plane at the aft end of the ATM oriented section, 72 inches from the center.

The integrated dose at these four ATM locations was obtained from the isodose curves. Data from these curves are plotted in Fig. C-10 for the average of four view angles from each isodose curve (0, 90, 180, and 270 degrees). An average relationship of dose versus distance from the tank was determined by a function, fitting these data as shown in Fig. C-10. This function was then extrapolated to ATM distances according to the inverse square of the distance. The distances from each of the eight tanks to the four ATM positions, stowed and extended, are shown in Table C-1. Shown in Table C-2 is the integrated dose for these distances. Note that the integrated dose is relatively high for stowed ATM locations towards the forward end; but, once extended, the dosage is approximately the same for all locations.

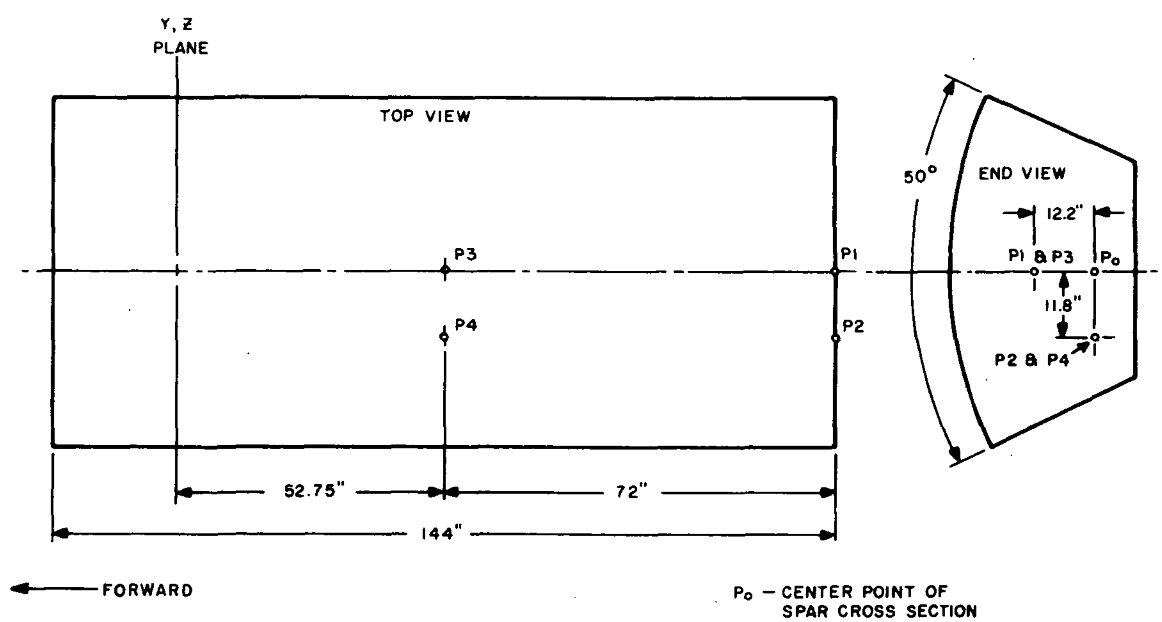
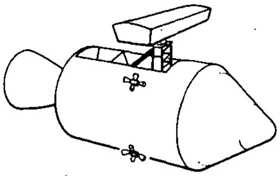


Fig. C-9 ATM Camera Locations for Radiation Analysis (Stowed)



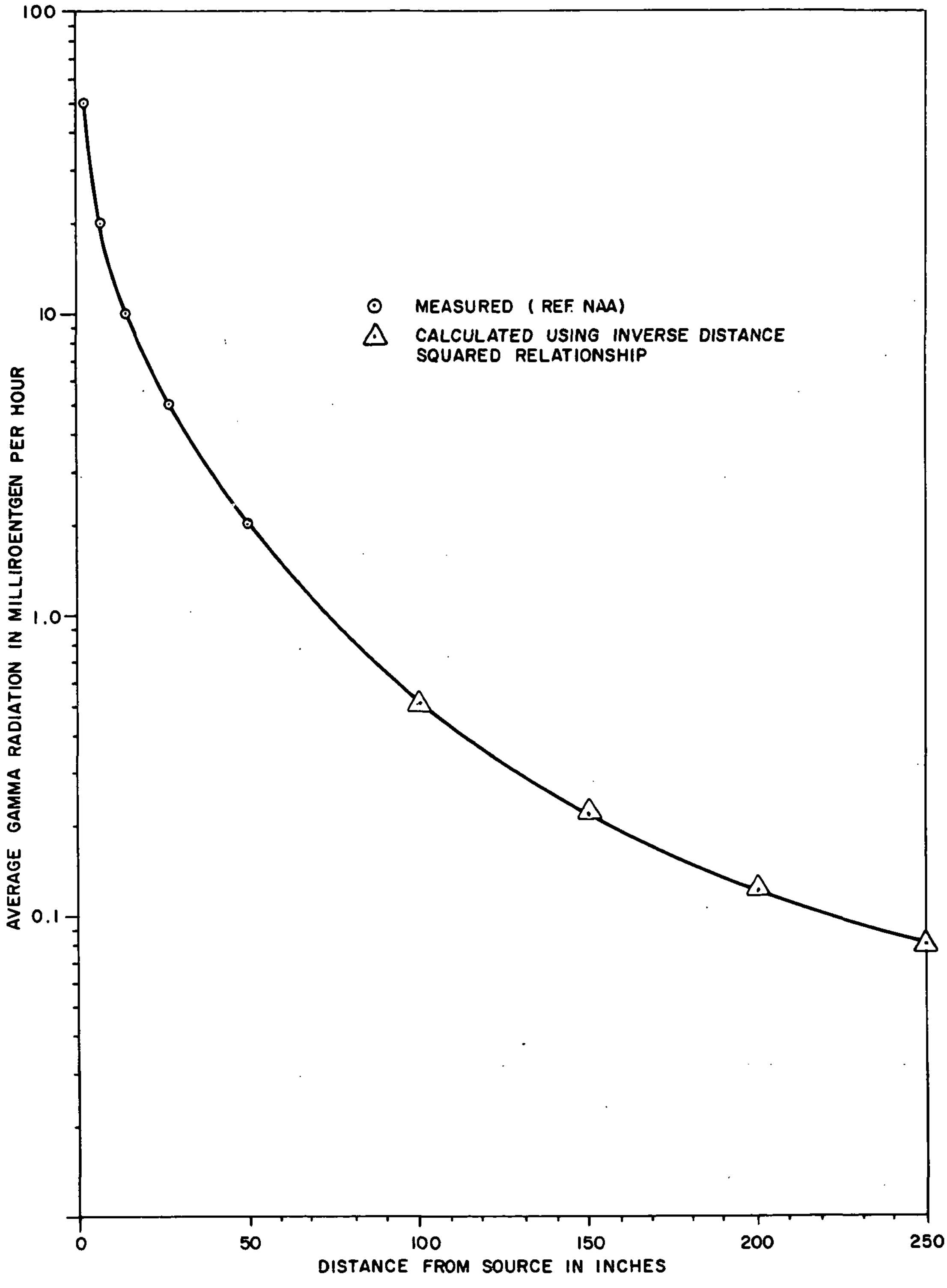


Fig. C-10 Dose Versus Distance from Source

Table C-1

DISTANCES FROM RCS TANKS TO ATM LOCATIONS (See Fig. C-9)

Tank	Distance to ATM Location (in.)							
	Stowed				Extended			
	P-1	P-2	P-3	P-4	P-1	P-2	P-3	P-4
F-1[a]	156	160	91	98	125	123	98	96
O-1[b]	112	117	56	68	102	100	107	105
F-2	160	156	126	122	191	185	201	194
O-2	236	241	145	142	204	198	188	181
F-3	230	251	144	130	203	191	188	174
O-3	157	146	125	110	190	178	193	180
F-4	114	108	59	47	102	88	119	107
O-4	158	154	93	86	126	114	98	82

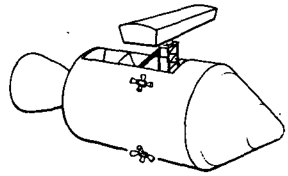
(a) F: fuel tank.

(b) O: oxidizer tank.

Table C-2

INTEGRATED COBALT 60 DOSE AT ATM LOCATIONS

Tank	Dose at ATM Location (mr/hr)							
	Stowed				Extended			
	P-1	P-2	P-3	P-4	P-1	P-2	P-3	P-4
F-1	0.20	0.19	0.61	0.52	0.33	0.34	0.52	0.54
O-1	0.41	0.39	1.65	1.21	0.49	0.50	0.45	0.46
F-2	0.19	0.20	0.32	0.36	0.13	0.14	0.12	0.13
O-2	0.09	0.09	0.23	0.25	0.12	0.12	0.13	0.15
F-3	0.09	0.08	0.23	0.30	0.12	0.13	0.13	0.16
O-3	0.20	0.24	0.33	0.43	0.14	0.15	0.13	0.15
F-4	0.40	0.45	1.50	2.35	0.49	0.66	0.37	0.45
O-4	0.20	0.20	0.57	0.68	0.32	0.40	0.52	0.75
Total Dose	1.78	1.84	5.44	6.10	2.14	2.44	2.37	2.79



C.3 COBALT 60 RADIATION EFFECTS ON FILM

The effect of the Apollo cobalt 60 gamma ray radiation on film has been investigated during this study with a series of laboratory tests. During these tests, three types of film were exposed to cobalt 60 radiation, and were analyzed for relative fogging or darkening of the developed image. The three films analyzed were: (1) Plus-X; (2) Tri-X; and (3) Royal-X Pan, with ASA ratings of 160, 320, and 1250, respectively.

The initial test of Plus-X and Tri-X was conducted using an 8.6 microcurie cobalt 60 source located 2.41 centimeters from the film. Exposure of the film to this source for 64 hours resulted in a total incident radiation of 1.21 roentgens. The exposed films were developed along with unexposed film from the same film package. All developed films were analyzed with a densitometer calibrated to zero on the unexposed film. The resulting darkening due to the cobalt 60 radiation, expressed as a relative density was:

- Plus-X: 0.18
- Tri-X: 0.36

A second series of tests was conducted at the University of Colorado isotopes laboratory using a 17.7-millicurie cobalt 60 source located five centimeters from the film. A series of exposures of different time durations was made on the Tri-X and Royal-X Pan films. Each exposed film was developed with unexposed films from the same film package, in HC-110 developer at 70°F with equal development times. Densitometer analysis was performed similar to that described in the previous paragraph. Figure C-11 displays the results of both tests.

The probable errors for these data points are: (1) time: ± 0.1 minute; and (2) distance of film from source: ± 0.1 inch. The strength of the cobalt 60 source was calculated using half-life decay and the time since its initial calibration.

From the data presented in Fig. C-11, it can be seen that film darkening proceeds somewhat more rapidly for the higher ASA rating film. However, the three films became totally fogged (film density of 1.0) at about the same time (80 minutes) after start of the exposure.

It can be concluded from these test data that film is susceptible to cobalt 60 gamma ray radiation and becomes totally fogged or darkened with a total dose of about 12 roentgens. The darkening of the test films exceeded the normally acceptable fogging limit of 0.10 to 0.15 after receiving a total radiation of about 0.4 roentgens.

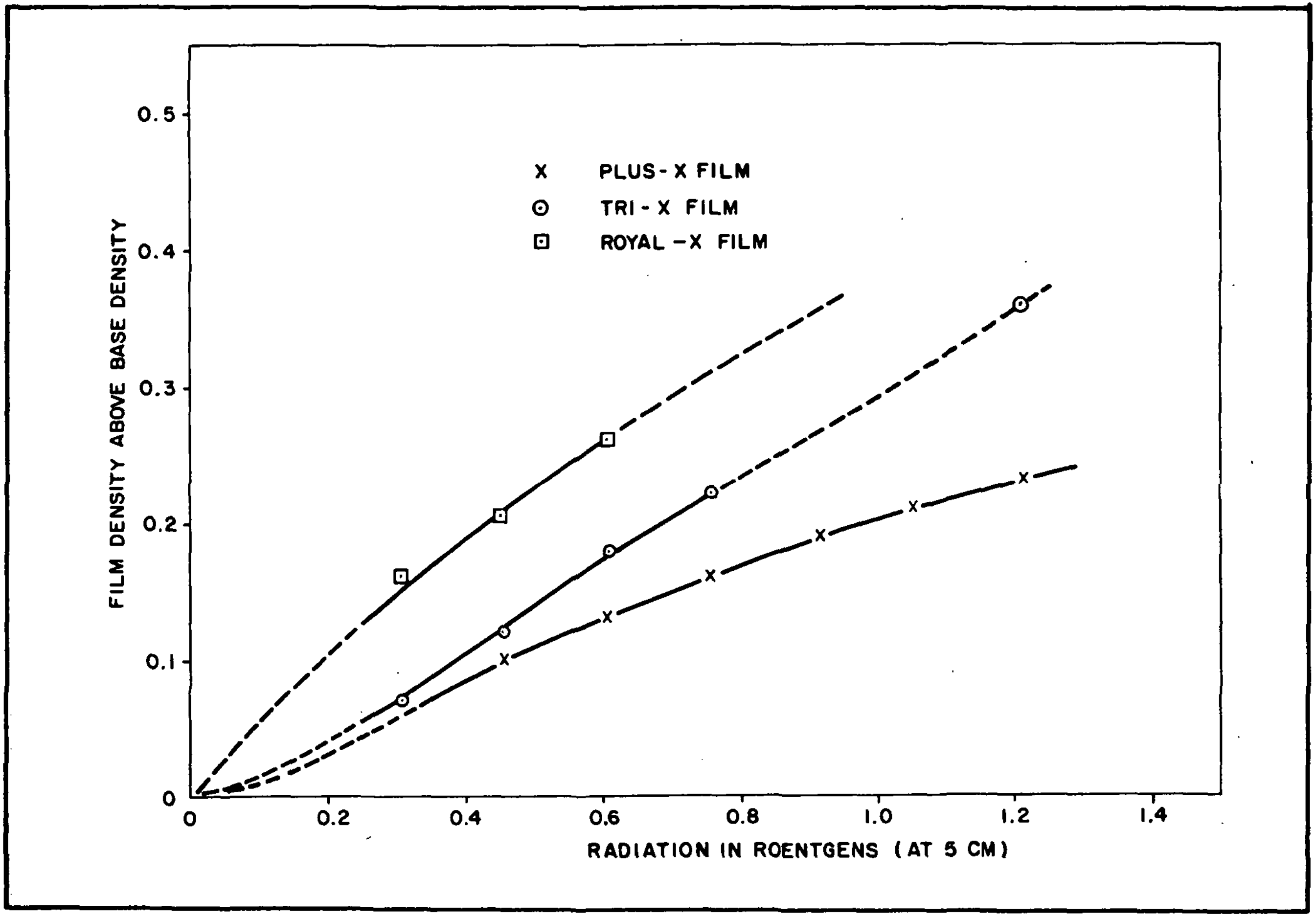


Fig. C-11 Dose Versus Radiation Plot

C.4 ATM FILM FOGGING DUE TO APOLLO COBALT 60 RADIATION

The fogging of ATM film can be determined from the fogging factors shown in Fig. C-11 and the intensities shown in Table C-2, if one assumes cumulative time durations for ATM stowed and extended. The nominal operations plan for ATM consists of film loading two days prior to launch, one day stowed after attaining orbit, and then seven days extended. The second set of film packs have at least eight days in the command module, and then six days in the extended ATM. It is assumed that the cobalt 60 radiation intensity is negligible in the CM.

The dosages for each condition and each ATM location are shown in Table C-3.

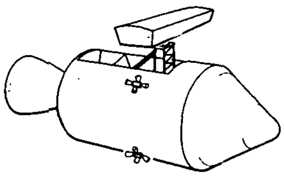


Table C-3

TOTAL COBALT 60 DOSAGE FOR ATM MISSION

ATM Location	Dose for 72 Hours Stowed (mr)	Dose for 168 Hours Extended (mr)	Total Dose For Film Pack # 1 (3 days stowed & 7 days extended) (mr)	Total Dose for Film Pack # 2 (6 days extended) (mr)
P-1	128	360	488	308
P-2	132	410	542	351
P-3	392	398	790	341
P-4	439	469	908	402

The ATM film fogging that would result at the four typical ATM locations from the cobalt 60 radiation is indicated in Table C-4 for the four time periods applicable to a typical ATM mission. It can be seen from these data that fogging of densities greater than generally accepted (0.10 to 0.15) will be encountered on ATM, if similar types of film are used.

Lead shielding surrounding the film would reduce this cobalt 60 fogging. The fogging could be reduced by one-half if the film were protected by one centimeter of lead. It can be seen that nearly all cases fall within acceptable limits with this amount of shielding.

Table C-4

POSSIBLE ATM FILM FOGGING AT SELECTED LOCATIONS

ATM Location	Mission Period											
	Stowed 3 Days			Extended 7 Days			Total After 7 Days Orbital Operation			Extended 6 Days		
	Plus-X	Tri-X	Royal-X	Plus-X	Tri-X	Royal-X	Plus-X	Tri-X	Royal-X	Plus-X	Tri-X	Royal-X
P-1	0.02	0.02	0.07	0.08	0.09	0.17	0.11	0.14	0.22	0.06	0.07	0.15
P-2	0.02	0.02	0.07	0.09	0.11	0.19	0.12	0.15	0.24	0.07	0.09	0.17
P-3	0.08	0.10	0.18	0.09	0.10	0.18	0.17	0.23	0.32	0.07	0.09	0.17
P-4	0.10	0.12	0.20	0.10	0.13	0.21	0.19	0.27	0.35	0.09	0.11	0.19

B B R C

Ball