

OPTICAL TECHNOLOGY EXPERIMENT SYSTEM (OTES)

PHASE II - FINAL TECHNICAL REPORT

LMSC Input

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Approved

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MISSILES & SPACE COMPANY

A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION

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FOREWORD

This Final Technical Report describes effort accomplished by the Lockheed Missiles & Space Company in support of the Perkin-Elmer Corporation activities on the Optical Technology Apollo Extension System (OTAES) study program. Results of the Phase II study are presented here. Material covering the optical payload and experiment studies is being prepared by the Perkin-Elmer Corporation for integration with the enclosed report and submittal to NASA/MSFC under prime contract NASA 8-20255.

This document consists of Volume I, Summary Report, and Volume II, Technical Report. Volume II is divided into two parts: Part I, Technical Report, and Part II, Resource Planning. A brief digest of the LMSC effort is contained in the Phase II Condensed Summary Report (LMSC-A848299, dated 1 August 1967).

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LIST OF ACRONYMS

AAP	Apollo Applications Program
ACS	Attitude-Control System
AGE	Aerospace Ground Equipment
AM	Airlock Module
ATM	Apollo Telescope Mount
CG	Center of Gravity
CM	Command Module
CMG	Control Moment Gyro
CSM	Command and Service Module
EMC	Electromagnetic Compatibility
EMF	Electromotive Force
EMI	Electromagnetic Interference
EI	Bending Stiffness
EVA	Extravehicular Activity
GFE	Government Furnished Equipment
GSE	Ground Support Equipment
HEPA	High-Efficiency Particulate Air
ID	Inside Diameter
IVA	Intravehicular Activity
JG	Torsional Stiffness
KSC	Kennedy Space Center
LCSM	Logistic Command and Service Module
LM	Lunar Module
LM-A	Lunar Module — Ascent Stage
LMSC	Lockheed Missiles & Space Company
LOS	Line of Sight
MDA	Multiple Docking Adapter
MOL	Manned Orbiting Laboratory
MOT	Manned Orbital Telescope

MOTEL	Manned Optical Technology Experiment Laboratory
MSC	Manned Spacecraft Center
MSFC	Marshall Space Flight Center
MSFN	Manned Space Flight Network
NASA	National Aeronautics and Space Administration
OART	Office of Advanced Research and Technology
OD	Outside Diameter
OMRF	Orbital Mirror Recoating Facility
OMSF	Office of Manned Space Flight
OSR	Optical Surface Reflectance
OSSA	Office of Space Science and Applications
OTAES	Optical Technology Apollo Extension System
OTES	Optical Technology Experiment System
OTS	Optical Technology Satellite
OWS	Orbital Workshop
P-E	Perkin-Elmer Corporation
PCM	Pulse Code Modulation
PCS	Pointing Control System
PRN	Pseudorandom Noise
RACS	Remote Analog Calibration System
RCS	Reaction Control System
SCS	Stabilization and Control System
SEARCH	Space Environmental Analysis Research Chamber
SETA	Systems Effectiveness Tradeoff Analysis
SLA	Spacecraft - LM Adapter
SM	Service Module
SPS	Service Propulsion System
STEM	Storable Tubular Extendible Member
UHF	Ultra-High Frequency
XCSM	Converted Command and Service Module
VHF	Very-High Frequency

Section 1 INTRODUCTION AND SUMMARY

Phase II of conceptual design for a manned Optical Technology Experiment System (OTES) was initiated on 16 November 1966. Perkin-Elmer Corp.(P-E), under prime contract to NASA Marshall Space Flight Center (MSFC), performed the basic conceptual design and experiment analyses required for selection of the OTES optical systems and held responsibilities for overall study program management. Lockheed Missiles & Space Company (LMSC), working closely with P-E, as subcontractor for spacecraft technology and systems integration for the optical technology experiments, assumed responsibilities in the following areas:

- Experiment selection supporting analyses
- Spacecraft conceptual analysis
- Preferred spacecraft concept refinement
- Attitude-control system analysis
- Thermal analysis
- Crew-participation analysis
- Reliability/safety analysis
- OTES model fabrication
- Program planning and support analysis

The primary objective of the Phase II OTES study program was to generate conceptual design(s) for a manned OTES. The proposed spacecraft will use a 2-m-diameter, seven-segment, active-optics primary mirror telescope system and will mate with the Apollo Telescope Mount (ATM) rack. The OTES primary mode of operation will be on the AAP Cluster X, although operating modes independent of the cluster can be performed for selected manned optical technology experiments. Both stellar and earth-based target orientation are considered for the telescope system. The resulting concept and experiments program will establish firm optical technology data that can serve as a precursor to a future 3-m telescope OTES.

Selection of the preferred conceptual design approach followed analysis of a number of concepts for a 2-m telescope implementation on the AAP cluster. Tradeoff studies of the advantages and disadvantages of the various approaches led to selection of a conceptual design that incorporates the best features of each. Upon P-E concurrence, LMSC continued its investigation of the preferred conceptual design supported by subsystem and astronaut participation analyses and definition of program planning and resource evaluation for subsequent phases of OTES development. A preliminary analysis of the OTES structural system reliability was performed and it was determined that NASA goals for AAP reliability and crew safety can be achieved with the preferred conceptual design. Reliability assurance activities for the subsequent phases of OTES development were evaluated.

Finally, the preferred conceptual design was examined to determine operational deployment techniques and compatibility with current planning for AAP cluster operations. This analysis included a preliminary examination of launch-window constraints.

Section 2

SYSTEM CONSIDERATIONS

The following overall program objectives were applied to the OTES conceptual design studies:

- Provide a 1973 OTES first-flight experiment capability
- Assure long-life, reliable spacecraft operations
- Implement the OTES program with maximum economy and probability of success by using "off-the-shelf" AAP hardware and techniques to eliminate development/qualification/testing operations associated with new designs
- Demonstrate the feasibility of the optical technology prior to commitment to the preferred 3-m operational system

Guidelines and constraints associated with the OTES system concept are as follows:

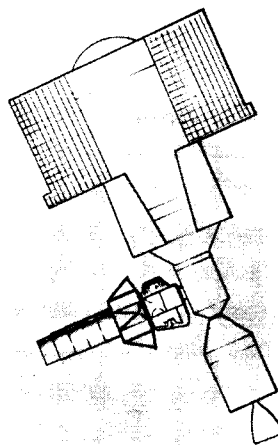
- Spacecraft. The spacecraft for OTES will be comprised of a standard Lunar Module (LM), the ATM rack, and the OTES telescope spar, and it will operate in its primary mode as an integral part of the AAP Cluster X flight system. The OTES and associated Orbital Mirror Recoating Facility (OMRF) will be launched via a single Saturn IB; Saturn-Apollo hardware and procedures will be used intact where possible. The OTES spar will accommodate an optical system employing a 2-m-diameter, seven-segment, active-optics primary mirror to be designed by Perkin-Elmer Corp., and will support the electro-optical hardware and experiments defined for the OTES Program. Appropriate astronaut maintenance tasks and accommodations will be provided for the orbit vehicle.
- Reliability. OTES life will be from 2 to 10 yr using resupply, astronaut maintenance, and crew-replacement concepts of the AAP. Reliability

goals, therefore, will be in full compliance with AAP objectives for a system (mission) goal of 0.90 and a crew survival goal of 0.999.

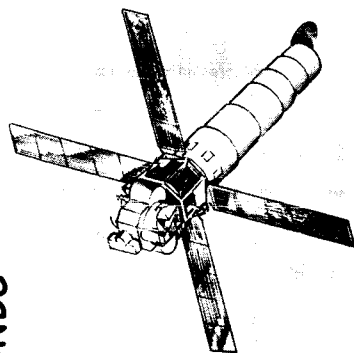
- Crew Participation. AAP cluster mission plans for crew rotation periods will apply to the OTES Mission. Crew tasks include direct or remote experiments operations, equipment maintenance and operations, mirror segment removal, surface recoating in space, and film replacement and retrieval.
- Mirror Thermal Constraints. Thermal gradients (radial) across the primary mirror will not be in excess of 1°C. The desired (optimum) operating temperature for quartz mirrors is -80°C.
- ATM Gimbal System Flexural Pivot.
 - Pitch and yaw: ± 3 deg with caging within 10 sec
 - Roll: ± 95 deg with maximum roll error of 10 arc min
 - No power for caging mode
 - Ground loads (vertical and lateral): per pivot limit = 2,500 lb, ultimate = 4,000 lb; per pivot-pair limit = 5,000 lb, ultimate = 8,000 lb
 - Launch loads with spar caged: 5 g vertical; 5 g lateral
- Attitude-Control and Pointing System. ± 2.5 arc sec accuracy.
- Saturn IB Constraints. Launch to a 240-nm circular orbit inclined 28.3 deg from the North with gross payload capability of 29,800 lb; postulated mission parameters are presented in Fig. 2-1.

ORBIT _____ 250 NAUTICAL MILE, CIRCULAR, INCLINED 28.3 DEG.
 PAYLOAD _____ 29,135 LB (SATURN 1B CAPABILITY TO 240 NM
 ORBIT = 29,800)
 LIFE _____ 2-YEAR MINIMUM; 10 YEARS WITH RESUPPLY/MAINTENANCE;
 0.90 PROBABILITY OF SUCCESS; 0.999 PROBABILITY OF CREW SURVIVAL
 GROUND STATION _____ LAUNCH: KSC
 OPTICAL: NEW OTES STATION (LOCATION TO BE DECIDED)
 MANNED: HOUSTON - MSC
 REQUIREMENTS _____ THERMAL GRADIENT: $\pm 0.5^{\circ}\text{F}$
 ATTITUDE CONTROL: ± 2.5 ARC SECONDS

OPERATION
MODES _____



AAP CLUSTER



INDEPENDENT

Fig. 2-1 OTES Postulated Mission Parameters

Section 3 SPACECRAFT ANALYSIS

As configured, the selected OTES conceptual design operates well within AAP ATM cluster guidelines and constraints.

Maximum use of the AAP cluster spacecraft subsystems provides the manned OTES concept with an off-the-shelf selection of subsystems needed in support of the mission. e.g., life cell and environmental control; structural; electrical; guidance, navigation, and control; data management; communications; recovery; orbit propulsion; and attitude control. In general, except for minor modification in some cases, this hardware remains in the AAP configuration, thus eliminating requirements for design and qualification of new hardware.

3.1 CONCEPT EVOLUTION

Tradeoff studies of a number of candidate conceptual designs for a 2-m OTES spacecraft resulted in combining a number of favorable attributes of several systems into a preferred conceptual design. Representative of these were the following:

- 1.85-m ATM Spar. Initial ATM rack configuration characteristics indicated that a 2-m primary mirror would be too large and that the maximum allowable size would be a 1.85-m mirror. The MSFC decision to move the gimbal plane from the center of the rack to about a foot from the forward end of the structure enabled incorporation of the full 2-m design. Several conceptual techniques of the design were retained.
- Pantograph Extension. A conventional lazy-tongs design for telescope erection was abandoned because of the difficulty of achieving sufficient stiffness for the large OTES spar. Favorable mass and center-of-gravity characteristics of this system were considered for the OTES design.

- Decreasing Diameter Extension. This concept is an inverted large version of a "Ship's Captain" telescope with the 2-m mirror in the smallest segment. The inverted concept facilitates integration of external attachments needed to support the magnetic suspension system hardware. It has necessary characteristics (mass, CG, etc.) in the extended position. For these reasons this approach was selected. The inverted version of the design (2-m mirror in largest barrel segment with the smallest segment at the farthest extension) was carried over to the preferred system concept.
- Quartz-Rod Mirror Supports. This design uses quartz rods as separation locators for the OTES primary and secondary optics to achieve desired thermal tolerances in critical positioning.

3.2 RECOMMENDED CONCEPT DESCRIPTION

The primary operating mode of the OTES is in the AAP cluster configuration with the telescope and associated experiments as the principal experiment. The system is configured, however, to permit independent mode operations using the LM/ATM rack/OTES as the spacecraft system for manned operations. This system would also accommodate use of the CSM docked to the LM for independent mode operations. Figures 3-1 through 3-3 show the preferred OTES concept in the launch, stowed, and extended configuration.

LMSC's contribution to the OTES Phase II activities includes the fabrication of two executive demonstration models of the preferred OTES spacecraft concept. These models are approximately 1/10 scale and are fabricated of aircraft quality aluminum and magnesium. Each integrated model consists of a representative LM/ATM rack, gimbal rings, telescope structure, and a protective transportation box. Figure 3-4 is a photograph of the OTES spacecraft model.

3.2.1 Structural Subsystem

Primary structural elements of the OTES include a support cone, fixed and movable cylindrical barrel segments, electro-optical equipment bay, CG positioner, and mirror and figure sensor supports. Design parameters and physical characteristics of the OTES are presented in Table 3-1.

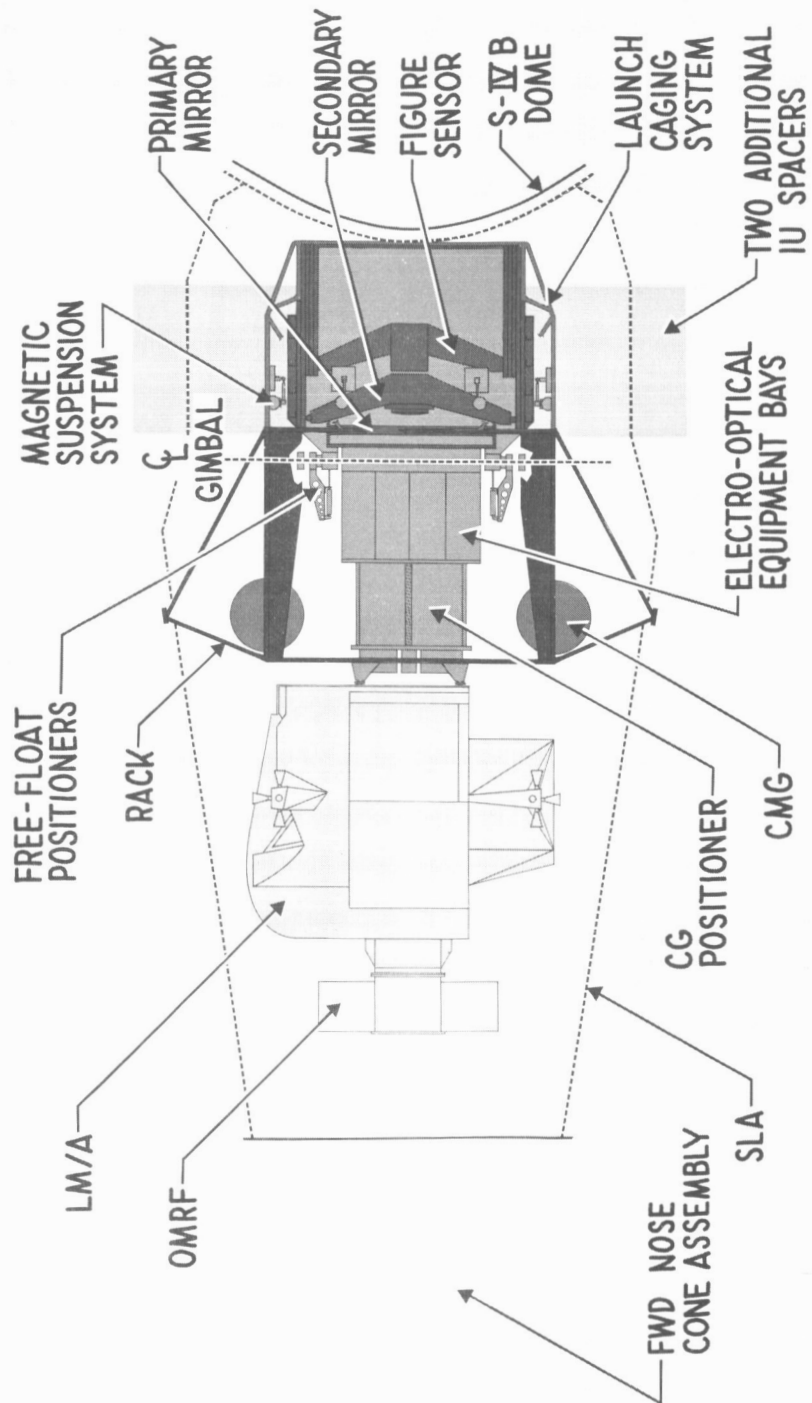


Fig. 3-1 Launch Configuration

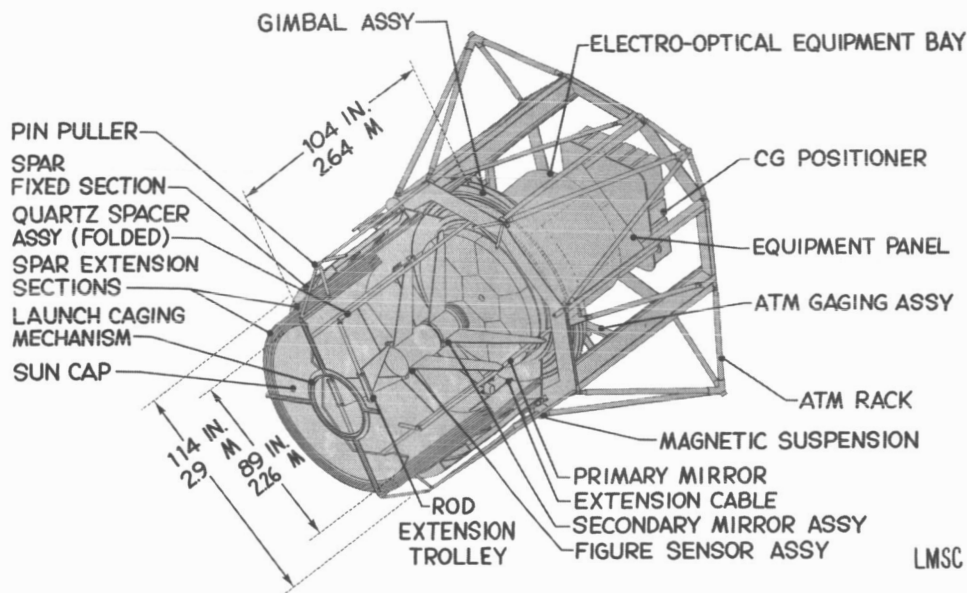


Fig. 3-2 OTES 2-m Concept, Stowed Configuration

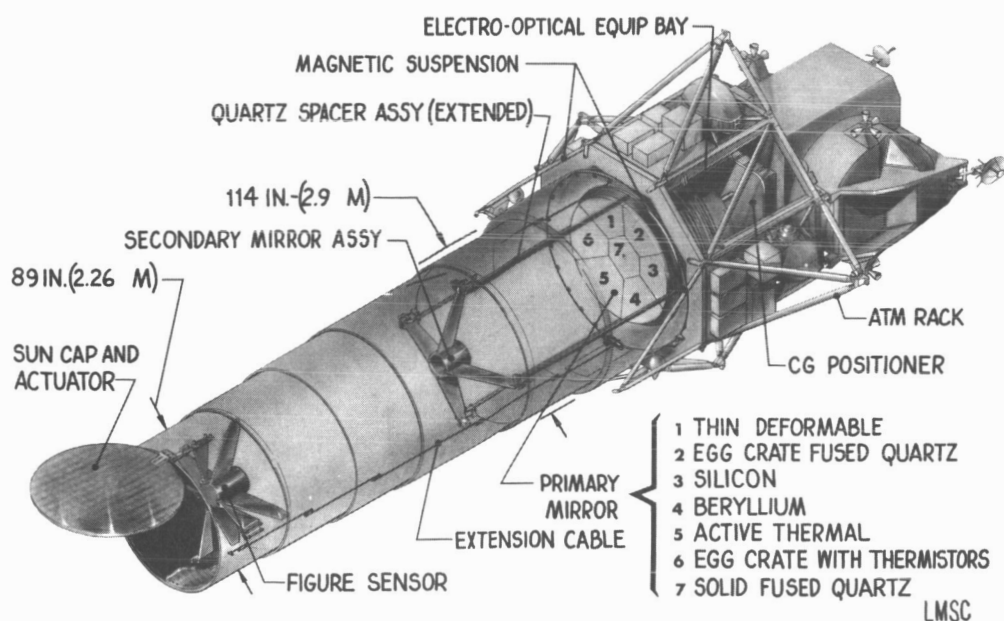


Fig. 3-3 OTES 2-m Concept, Extended Configuration

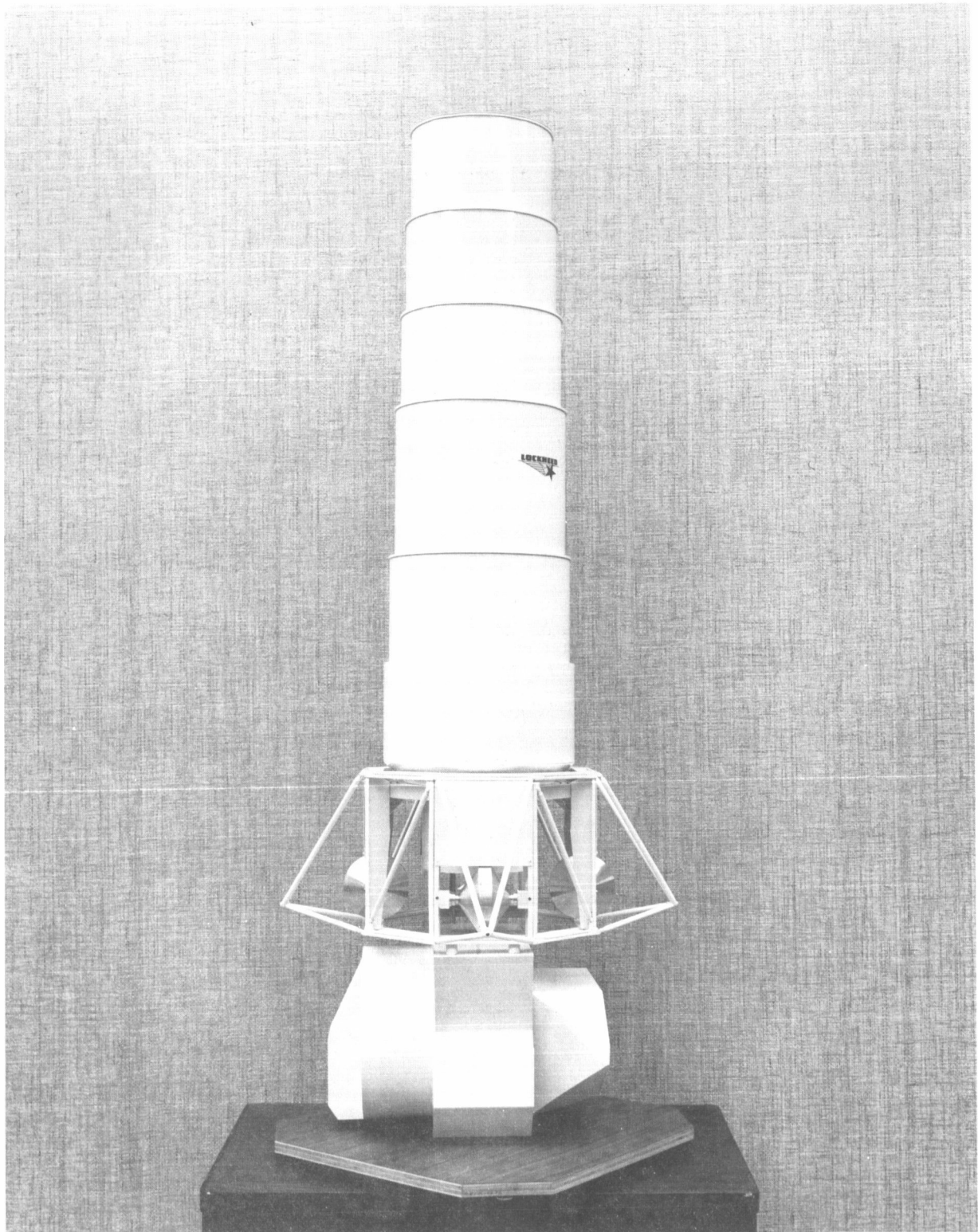


Fig. 3-4 OTES 2-m Model

Table 3-1

OTES TELESCOPE MENSURATION

Dimension	Stowed		Extended	
	(in.)	(m)	(in.)	(m)
Lengths				
Forward of Gimbal Plane	104	2.64	389	9.88
Aft of Gimbal Plane	61	1.55	91	2.31
Total	175	4.44	480	12.19
Equipment Modules			39.4	1
Diameters				
Maximum OD			114	2.90
Minimum ID			87	2.21
Built-in Gimbal Ring OD			88	2.24
Optics				
Primary to Secondary Mirrors			130	3.30
Primary to Figure Sensor			312	7.92
Movements				
CG Positioner (Maximum)			30	0.76
CG Travel			18	0.46
Telescope From Gimbal Plane			8.25	0.21
Pitch and Yaw			± 3 deg	± 3 deg
Rotation About LOS			± 95 deg	± 95 deg

Secondary structures include the magnetic suspension system, support rack and engagement mechanisms, free-float positioner mechanism, launch support cage, and cable extension system. These structures are as follows:

- Support Cone. This is a frame assembly that replaces the inner gimbal ring of the ATM gimbal system for mating the OTES telescope to the ATM rack. The large end of the support cone is riveted to the large-diameter barrel segment of the telescope spar to form the basic structural support assembly.
- Cylindrical Barrel Segments. Segments forming the spar are semimonocoque barrels spaced concentrically in the stowed position for extension in tandem fashion to the smallest segment. Upon extension of each segment, a positive-locking and light-seal joint is provided by pyrotechnic actuated pin drivers. During ascent, the segments are held in place by a launch support cage.
- Electro-Optical Equipment Bay. This equipment bay comprises a central fixed hexagon compartment containing power conditioning equipment and six replaceable modular experiment-equipment compartments. The design of experiment canisters permits extension of support-beam structure from the CG positioner to the support cone.
- Center-of-Gravity Positioner. The CG positioner comprises a movable weight with a 30-in. travel space for mechanically repositioning the OTES CG. It is used to adjust CG locations for both flexural pivot and magnetic suspension modes of operation.
- Optics Support Structure. The secondary mirror and figure sensor support structures are of 0.5-in. aluminum honeycomb panel material. Sensor supports are rigidly attached to the figure sensor housing and telescope barrel; the secondary mirror is held in position by a three-degree-of-freedom (pitch, yaw, and translating) servo system.
- Magnetic Suspension System Support Structure. This tubular aluminum structure is attached to the ATM rack and functions as a launch caging

mechanism load channel during ascent. Sufficient clearance and handholds are provided on the structure to permit astronaut access to the extra-vehicular activity (EVA) maintenance and experiment servicing areas.

- Free-Float Positioners. These devices are brackets and linear actuators and are used to disengage the telescope from the three-axis gimbal rings for the magnetic suspension system experiment.
- Cable Extension System. The ATM rack is the OTES carrier during launch and the base for the extended spar. The stowed 104-in. OTES is extended to its 391-in. operating length via the cable extension system, which consists of two drive cables (180-deg apart) that are operated by drive motors and winch assemblies (synchronized for uniform spar extension).
- Quartz-Rod Spacing Mechanism. Four 1-in.-diameter quartz rods are spaced at 90-deg intervals and are located 41.50 in. from the telescope centerline to maintain correct focus distance between primary and secondary optics. The folding rods fit within the telescope barrel sections and are enclosed in tubular magnesium structures during ascent; the rods are extended to position automatically via guides and trolley rollers and locked in position by tension-spring loading.

3.2.2 Thermal Control System

Passive thermal control techniques are used for OTES. These include exterior surface coatings having a low solar absorption, such as Optical Surface Reflectance (OSR) materials presently under development by LMSC, multilayer super insulation attached to inner surfaces of the spar, and small secondary insulators within various components. Multilayer insulation of 0.25-in. thickness is considered necessary to maintain suitable thermal gradients and levels.

3.2.3 Electrical-Power Subsystem

Maximum use of the AAP ATM power-generating hardware is made for the OTES conceptual design, i. e., solar arrays, rechargeable batteries, and fuel cells. The

estimated 300-w OTES power requirement is considerably lower than other AAP telescope/experiment applications currently in preliminary design.

During independent mode operations of long duration it will be necessary to deploy an OTES spacecraft solar array to maintain power requirements. This is conceptually considered to be a deployable/retractable solar-array design to permit multiple independent mode operations. During cluster-mode operations, the primary power source would be an array system mounted on the orbital workshop structure (S-IVB shell). Both array systems would be adjustable to permit solar orientation necessary for maximum power output.

3.2.4 Attitude-Control System

The AAP ATM attitude-control system is proposed for use on the OTES, although minor modification is required to permit attitude references to be taken using the star sensors rather than the sun sensors as used on the ATM. One gimballed star tracker would be used as an absolute roll reference, and two additional star trackers would be used as coarse and fine pointing references. Vernier gimbals between the rack and spar are caged at the center of their travel until the CSM control system brings the target star in line with the fine sensor field of view. Three control moment gyros mounted on the ATM rack and the CSM (or LM) orbit propulsion systems provide the primary OTES attitude-control system requirements.

3.2.5 Communications Subsystem

The OTES will use the Apollo unified S-band telecommunications system planned for CSM and LM AAP operations. This comprises S-band receivers and transmitters, UHF transmitters and receivers, VHF transmitters and receivers, a central processing center, the necessary antenna systems, and a mobile video system for monitoring EVA. As presently defined, the AAP communications capabilities appear fully compatible with OTES mission requirements.

3.2.6 Orbital Mirror Recoating Facility

Preliminary conceptual analysis of an Orbital Mirror Recoating Facility (OMRF) has generated a baseline approach for supporting Optical Technology Experiment 6. The OMRF would be docked separately (90-deg from the OTES) on the docking adapter, although launched on the same Saturn launch vehicle with the Cluster X-4 flight. Conceptually, the facility consists of two compartments: a pressurized chamber and a space vacuum section. The pressurized compartment is a short cylindrical section with Apollo docking ports and contains a viewport for observation of mirror segment recoating operations. The vacuum chamber contains the necessary equipment for stripping and recoating the mirror segments and provides protection against direct sunlight impinging on the mirror. Processing operations are envisioned as fully automated and remotely controlled by the astronauts. A single electrical plug - connected following docking by EVA - provides the interface between the OMRF and multiple docking adapter (MDA).

The most difficult task indicated in the OMRF design is the stripping of degraded mirror surfaces and their protective overcoating materials. Thermal evaporation is recommended as an appropriate technique for recoating the surfaces with a thin-film aluminum reflective material. A fully automatic reflectivity control system is proposed to assure material tolerances. Overcoating is considered unnecessary since the recoated segment is processed and handled completely in a vacuum environment. Consideration has also been given to initial thin-film coating in the space environment to increase mirror reflectance and to reduce the problems associated with mirror stripping requirements.

3.2.7 Preliminary Weight Performance Estimate

Preliminary estimates of OTES system weights indicate that the conceptual design is well within the 29,800-lb payload capabilities of the S-IB booster vehicle for a 240-nm circular rendezvous orbit. A gross system weight of 29,135 lb has been estimated and would leave a performance margin of 665 lb. This estimate has been based upon conservative figures and should compensate for traditional weight growth associated with space system development.

3.3 SUPPORTING ANALYSES

Supporting analyses during Phase II have included preliminary evaluation of thermal, attitude control, power, dynamic/stress, astronaut design requirements, and reliability. These analyses were performed using the latest available NASA or LMSC contractual design and study data relative to the AAP ATM. Comparatively, the OTES mission analyses indicated that OTES requirements are equivalent to or less than those already attributed to the ATM solar telescope operations.

3.3.1 Thermal Analysis

Effects of solar albedo and earthshine and additional effects of cluster heat rejection, radiation, and shielding were included in analyses of the integrated 2-m OTES cluster configuration on a low-altitude (250-nm) orbit. Two cases were selected for study which represent the extreme high and low-temperature conditions for AAP/OTES operations in the assumed space environment:

- The telescope with AAP cluster in space-oriented position for maximum sunlight reflectance ($\beta = 52$ deg), Fig. 3-5.
- The telescope with AAP cluster in space-oriented position for maximum sunlight reflectance ($\beta = 0$ deg)

Thermal models were developed for the AAP cluster, ATM rack, and OTES for mathematical definition of surface nodes and definition of the correct thermal interfaces of the OTES. An interior and exterior radiation network was developed on the basis of assumed surface finish properties. The conduction network was combined with the radiation network to form part of the system model. Where necessary, approximations were generated to define thermal conduction paths through support structures that can not be determined until the preliminary design phases of OTES development. The following conclusions were reached:

- Temperature gradients across the primary mirror segments can be minimized by passive thermal control techniques to less than 1° F.

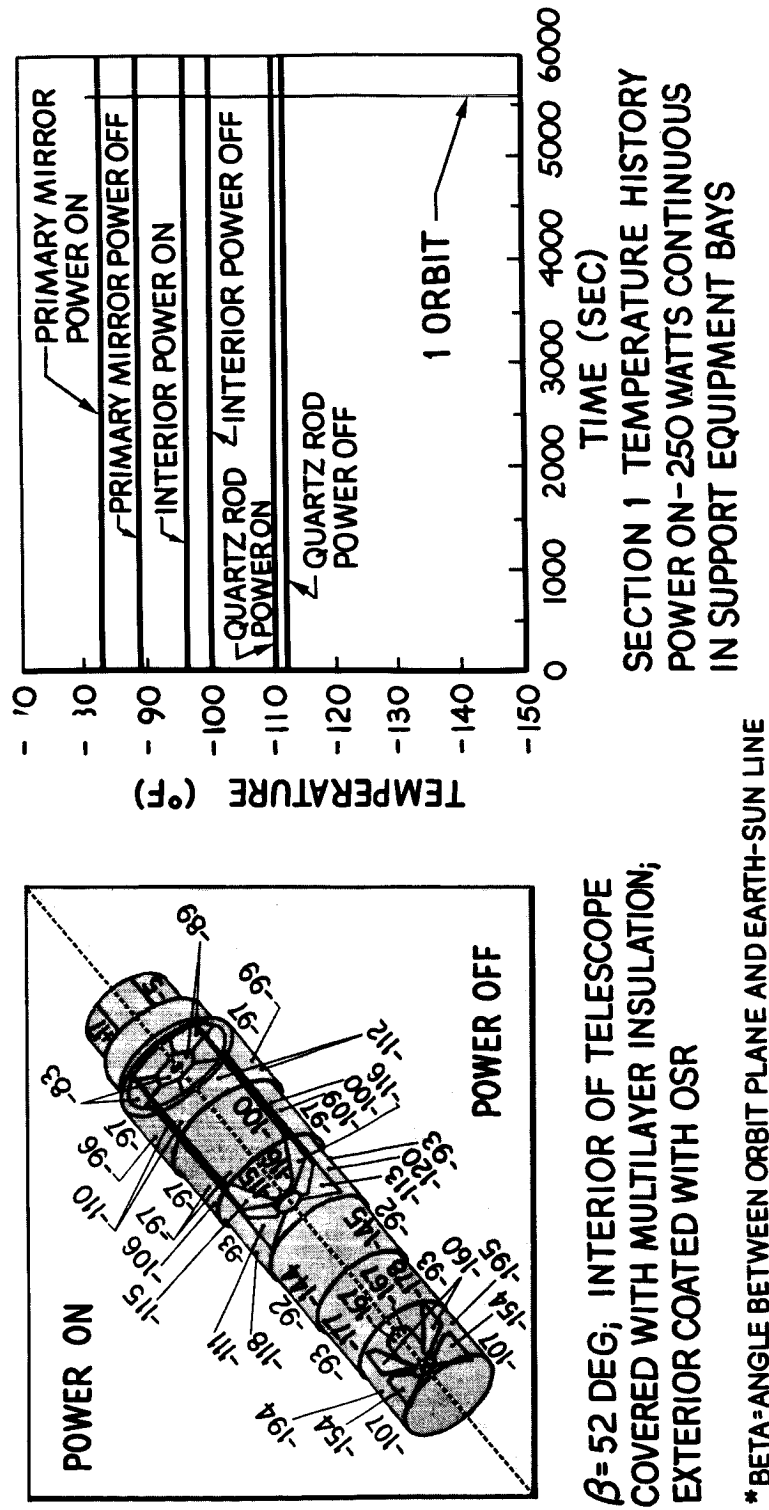


Fig. 3-5 OTES Average Temperature Profiles

- The primary mirror segments can be controlled over a range from -83 to - 107° F.
- The maximum primary mirror temperature stabilization time is about 24 hr.
- Thermal gradients are uniform along the quartz spacer rods.

3.3.2 OTES ATM Control System Performance Analysis

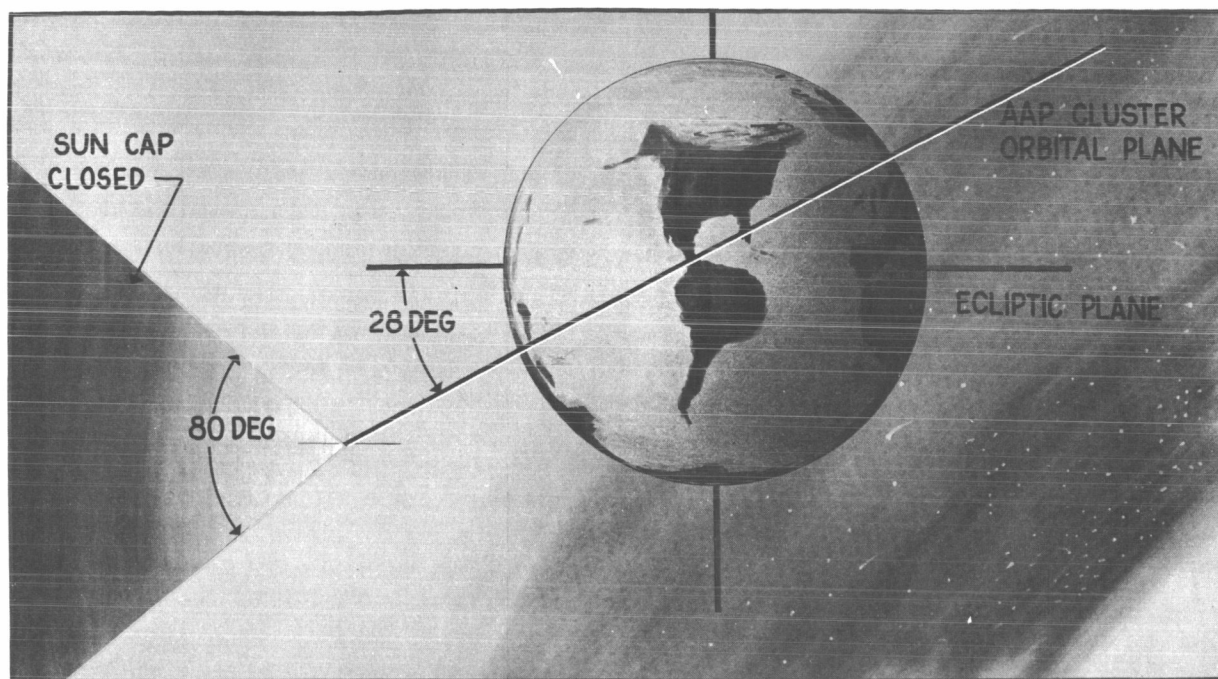
Parametric requirements for the baseline ATM control system were determined from the basic considerations of OTES telescope (spar) inertia, flexure pivot spring rate, and maximum static load-carrying capacity for ground checkout and calibration. Using these parameters, control system reaction to various conditions and disturbances (mass center offset, spacecraft motion during clustered and independent modes, and astronaut movement) were evaluated to determine functional characteristics of the selected design. The control system analysis indicates that pointing accuracies of ± 2.5 arc sec can easily be achieved for either mode of operation. Figure 3-6 illustrates the extent of star coverage for stellar photography missions. The magnetic suspension pointing system is needed for experiment 9.

3.3.3 Earth Pointing Experiments

An analysis of the OTES orbit geometry and its effect on spacecraft design requirements for telescopic observation of earth targets from the nominal 250-nm orbit was performed. Both control system and tracking error bandwidth requirements have stringent constraints for low orbit-altitude operations. Figure 3-7 illustrates the line-of-sight angular position and tracking rate geometry. At the 250-nm orbit, the target is in sight for 11.04 min (over the central angle ROQ of Fig. 3-6 during which the spacecraft velocity is 4.13 nm/sec).

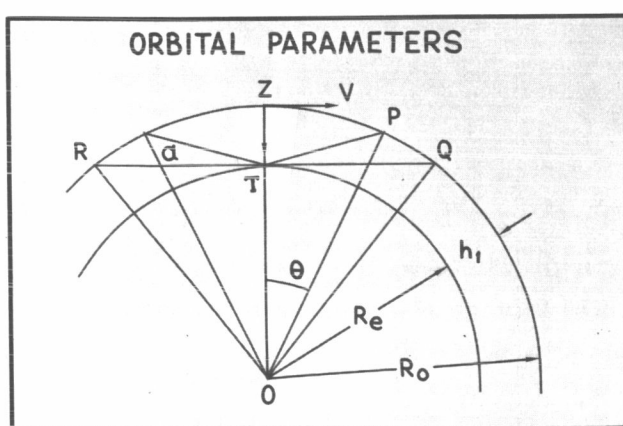
3.3.4 Electrical Subsystem Analysis

A preliminary analysis of electrical-power requirements for the cluster and independent mode operations indicates the OTES (constant load of 250 w, 300 w including



LMSC

Fig. 3-6 Typical Orbit Geometry for Stellar Photographic Mission



$$R_e = 3,440 \text{ NM}$$

$$h = 250 \text{ NM}$$

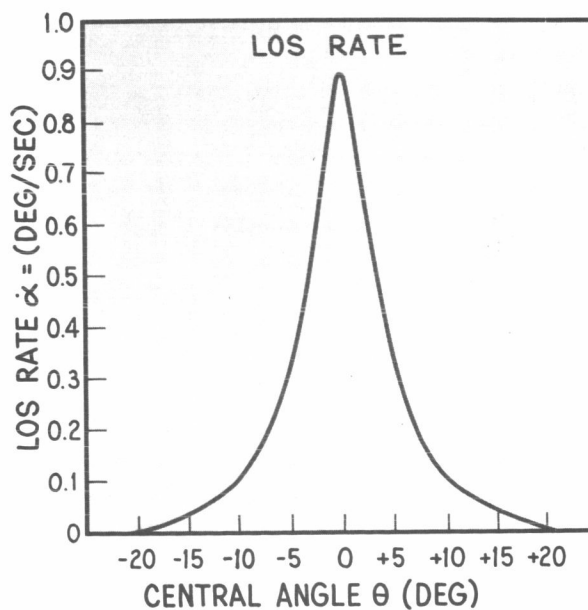
$$R_o = 3,690 \text{ NM}$$

$$P_o = 93.7 \text{ MIN}$$

$$V = 4.13 \text{ NM/SEC}$$

$$\dot{\theta} = 3.84 \text{ DEG/MIN}$$

- LOS TIME = 11.04 MIN
- LOS RANGE (MAX) = 1,336 NM
- LOS RANGE (MIN) = 250 NM



- LOS TIME FOR $\theta = \pm 5 \text{ DEG} = 2.6 \text{ MIN}$
- PITCH GIMBAL ANGLE REQ = $\pm 48 \text{ DEG}$ (FOR STABILIZED CLUSTER)

LMSC

Fig. 3-7 OTES Earth-Oriented Missions

magnetic suspension system experiment) preferred conceptual design is fully compatible with current planning for the AAP ATM. The cluster would provide primary power via a solar-array installation on the Orbital Workshop (S-IVB) structure. Although battery power would suffice for a short duration mission in the independent mode, it would be desirable to install an erectable (and retractable) solar-array system on the ATM rack structure for longer applications. Preliminary analysis of the independent mode indicates nominal electrical power loads are approximately 2.05 kw whereas the normal AAP loads are on the order of 5 to 7.5 kw and comprise the following:

- LM. Housekeeping, 600 w; other, 200 w
- Pointing control system. 1,000 w
- OTES experiments. 250 to 300 w

This load can be reduced by 25 percent by operating only the environmental control system and attitude control of the LM and conducting mode operations from the CSM. For both the AAP cluster and the independent mode, nontracking solar arrays have been selected to reduce disturbing torques.

3.3.5 Structural Dynamics/Stress Analysis

Dynamic deflections of the OTES structure were investigated to determine the rigidity associated with the six telescoping sections of the spar. This analysis employed a lumped-parameter model comprising eight mass lumps interconnected by shear and moment springs to allow for translation and rotational freedoms of motion. Astronaut motion was considered to be the only excitation causing elastic perturbations of the spar during operating modes; i. e., atmospheric drag, gravity, etc., were considered to be in a steady state (nonoscillatory) and hence not included in the analysis. Results of this analysis show the proposed design provides suitable rigidity to maintain the critical tolerances for the OTES optics. Theoretical evaluation of structural loading conditions shows essentially no critical static loading is imposed on the OTES spar during experiment operations.

Section 4

ASTRONAUT-PARTICIPATION ANALYSIS

Crew participation concepts developed during Phase I have been appropriately modified and refined during the Phase II study. This has been a result of more detailed evaluations of mission functions, representative experiments, assumed mission environments, and recently available data from other manned NASA space programs and simulations. The majority of the emphasis is placed on the EVA since the astronaut capabilities to perform onboard tasks have been adequately substantiated in these previous activities.

4.1 SCOPE

The scope of the present study has been:

- Identification of crew participation in experiments
- Specification of constraints and guidelines for crew safety
- Evaluation of mission feasibility – particularly EVA operation
- Definition of crew training requirements

4.2 INVOLVEMENT

Potential astronaut involvement in specific OTES experiments has been examined and estimates made of modes of behavior typical of each. Crew participation requirements were estimated for typical candidate experiments. Experiment support (e.g., film retrieval) and mirror recoating experiments are major candidates for evaluation of OTES EVA operations feasibility.

4.3 SYSTEM CONSTRAINTS AND GUIDELINES

The AAP cluster configuration guidelines and constraints established to maximize the probability of crew survival have been adopted for the OTES system design. This

involves total elimination of tasks that can be performed automatically or by remote control and those that could distract from primary mission success objectives. All AAP safety procedures and provisions for cluster operations will be incorporated in the OTES system design. So far as possible, normal working, eating, and sleeping habits associated with the ground environment will be maintained for functions performed in the habitable areas of the spacecraft system.

4.4 CREW TIME ALLOCATIONS

Detailed analyses of the AAP Cluster A mission tasks were available to the OTES study team and were applied as an index of the overall distribution of crew activities. Approximately 65 percent of the time allocation is given to basic system operations and maintenance, 25 percent is allocated for the AAP mission experiments, and a 10 percent contingency is provided for orbit periods of the mission.

4.5 EXTRAVEHICULAR ACTIVITIES

Where feasible, alternatives are made available and extravehicular activities are eliminated or minimized. No EVA is performed where abnormally hazardous conditions could exist, e.g., meteoroid activity or insufficient time for rescue operations. Safety and rescue provisions established by NASA for EVA operations are included in all OTES operations planning procedures. An overall mission time-line analysis can be postulated for a 1-yr mission with 90-day resupply on the basis of Cluster A planning. It is assumed that 25 percent of the crew time is available for typical AAP cluster mission experiments. The resupply concept includes a premise that 2 days at the beginning and end of each cycle are required for crew transfer and familiarization. A single EVA takes about 8 hr and each resupply cycle provides for eight EVA's.

Preliminary concepts generated during the Phase I OTES study for manned crew participation appear compatible with the current 2-m system design approach. The three areas affected are telescope erection, mirror transfer and recoating, and maintenance EVA requirements. A major change is the elimination of EVA for

telescope erection due to the limited value of astronaut participation. Moreover, the conceptual design of the telescope erection system employs techniques for extending and locking barrel segments in place with a high degree of reliability in an automatic mode. For the mirror recoating experiment, basic concepts remain essentially the same as in the Phase I study, including a carrying case for mirror handling to reduce possible damage, accident, and glare to the EVA astronauts. Figure 4-1 illustrates the astronaut EVA mirror segment removal experiment sequence.

4.6 TRAINING REQUIREMENTS AND FACILITIES

Detailed evaluation of training requirements and facilities needed in support of OTES will be generated on the basis of the defined OTES experiments and implementation requirements. Training time estimates for the AAP Cluster A mission run to a total of 430 hr/man (1,200 to 1,300 hr); Cluster B training time estimates are from 450 to 760 hr/man.

Many of the facilities required will be available from the AAP and Apollo training programs. Functional mockups and prototype hardware will be necessary during Phase B for specific training in the detailed operation of individual experiment systems. Also, presently available LMSC training facilities, such as the ZEROG and LMSC underwater zero-g simulator, will be made available for determining astronaut EVA capabilities and design refinements required to optimize mission performance during EVA operations. Cost estimates, presented in the resources planning, include development models for this purpose.

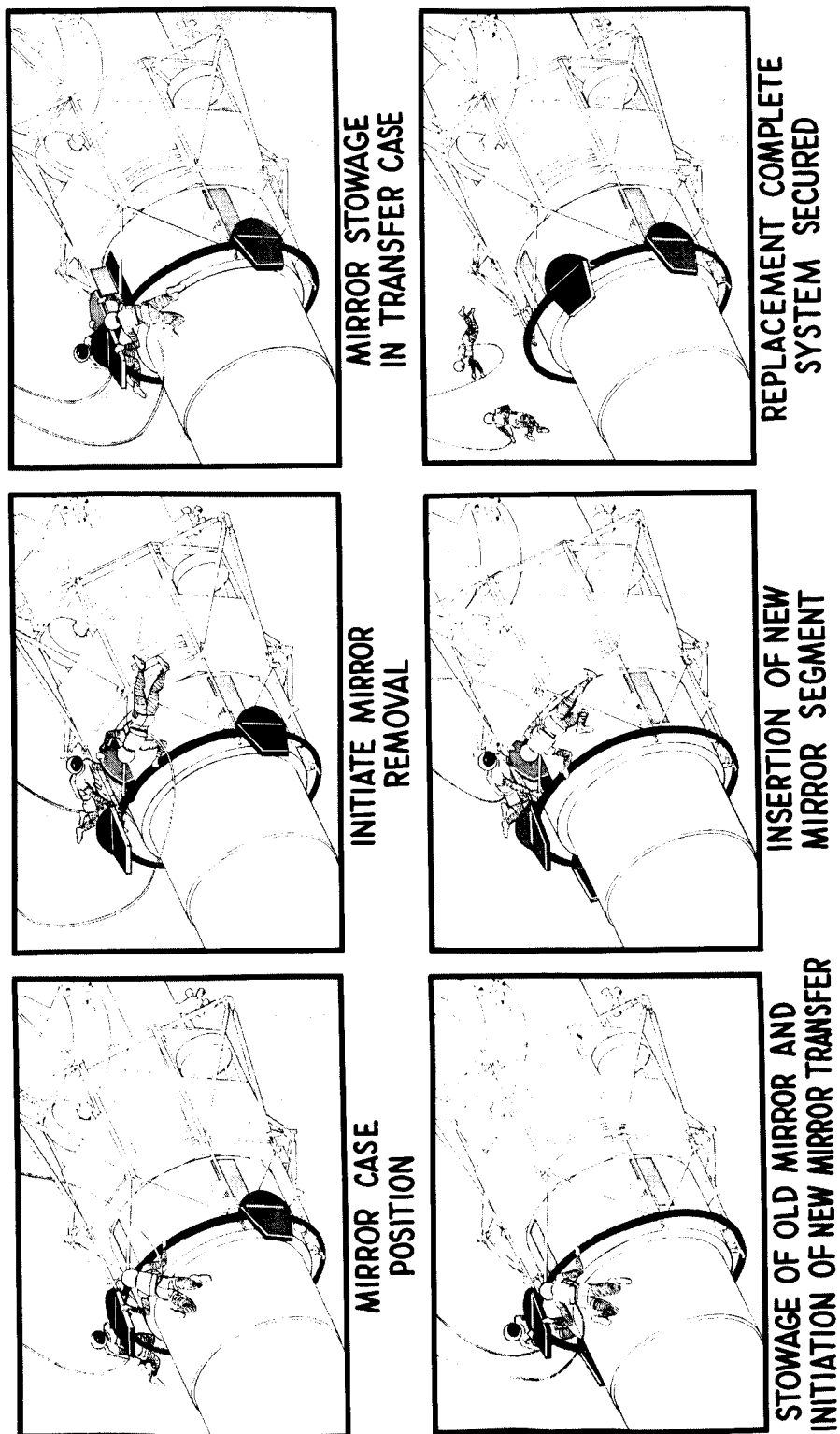


Fig. 4-1 Mirror Segment Removal Experiment

Section 5

RELIABILITY AND SAFETY ANALYSIS

The 2-yr mission OTES reliability goal of 0.90 can be attained through extensive use of redundancy and spares for the basic AAP cluster equipment supporting the 2-m telescope experiment. The astronaut safety goal of 0.999, already established for the AAP cluster application, will be maintained throughout the OTES system development program by adherence to the established AAP design and operational concepts.

5.1 SYSTEM EFFECTIVENESS TRADEOFF ANALYSIS

Systems effectiveness tradeoff analysis (SETA) is a technique through which a desired system reliability can be analytically obtained while optimizing system weight, cost, and maintenance time, and satisfying constraints for safety, gross system weight, and single point failure requirements. Each potential redundancy is evaluated by an appropriate mathematical model and computer analysis to determine reliability improvements versus weight, cost, and maintenance time penalties. Weight optimization is the key factor for conceptual design analysis. The total differential of the effectiveness equation and model distributes redundancy in an optimal manner using such inputs as component failure rate, duty cycle, type of redundancy, component weight, and criticality coefficients for determination of the program constant.

The SETA program was employed for the OTES spacecraft evaluation. However, it was not used to evaluate the electro-optical experiment equipment since sufficient design data were not available. However, the OTES Program makes maximum use of AAP and other space-qualified hardware having known or estimated failure rates. For OTES, the SETA technique will be used to update Cluster X configuration reliability apportionments with the telescope experiment included and for analysis of anticipated resupply requirements.

5.2 TELESCOPE RELIABILITY ANALYSIS

Critical components of the telescope structural system are the reel-out cables and associated drive motors. The present concept calls for redundant drive motors; however, additional redundancy could be added if subsequent analytical studies indicate a need. The telescope barrel locking mechanism (explosive force pins) have individual reliabilities in excess of 0.999. Control and relay system electronics would probably constitute the largest reliability hazard. Again, selection of appropriate circuit reliability will reduce possible failures to an acceptable level within the overall system reliability goal of 0.90. The simplified design concept for the OMRF would demonstrate a very low probability of failure. The worst case condition for the OMRF could be distribution of nonuniform energy delivery by the power system for electrochemical recoating operations. The power supply, a space maintainable unit, will be maintained at acceptable levels by close monitoring and minimization of possible operator errors.

5.3 SAFETY ANALYSIS

Hazards associated with crew survival are similar to those anticipated for AAP cluster operations. The system design, crew EVA sequences and provisions, and other requirements and constraints associated with the AAP cluster mission will be fully applied to the OTES system configuration to assure achievement of the 0.999 crew survival objectives. Figure 5-1 is a summary of OTES reliability objectives and results.

OBJECTIVES

OPERATIONAL LIFE: 2 YR (MINIMUM) TO 10 YR (WITH RESUPPLY & MAINTENANCE); 0.90 PROBABILITY OF SUCCESS, 0.999 PROBABILITY OF CREW SURVIVAL

RESULTS

SYSTEM EFFECTIVENESS TRADEOFF ANALYSIS COMPLETE (PRELIMINARY DATA)

2-YEAR LIFETIME OBTAINABLE ONLY THRU DESIGN REDUNDANCY, RESUPPLY, & MAINTENANCE

0.93 PROBABILITY OF SUCCESS FOR CURRENT AAP FLIGHT NO. 4 ATM RACK & SUBSYSTEMS (728-DAY MISSION WITH 4600 LB OF SPARES)

0.99 PROBABILITY PREDICTED FOR CURRENT OTES STRUCTURE

0.92 PROBABILITY PREDICTED FOR OTES STRUCTURE INTEGRATED INTO ATM RACK (OPTICAL SUBSYSTEMS NOT INCLUDED)

CRITICAL COMPONENTS: BATTERIES, & CMG'S

CREW SURVIVAL GOAL OF 0.999 ESTABLISHED FOR AAP CLUSTER A WILL BE EASILY RETAINED FOR OTES

Fig. 5-1 OTES Reliability Objectives and Results

Section 6

CLUSTER X MISSION SEQUENCES AND LAUNCH CONSTRAINTS

Preliminary analyses of AAP Cluster X mission tasks and configurations have been performed. They were based on tentative AAP cluster plans that are subject to change. For Phase A purposes, however, it is possible to base OTES mission descriptions on current Cluster A planning, since the Cluster X flights would be essentially a repeat of the Cluster A mission.

6.1 MISSION SEQUENCE

Cluster X flights would start with the first two Saturn IB launches of Flights X-1 and X-2. Figures 6-1 and 6-2 show the X-3 and X-4 launch sequence of events through telescope erection. The X-3 launch leaves the CSM in a circular 120-nm orbit for rendezvous with the OTES (X-4 launch) on the following day. The OTES is injected into a 244-nm circular orbit, whereupon the CSM transfers to the 244-nm orbit for docking with the LM/ATM/OTES and subsequent transfer to a 254-nm circular orbit for rendezvous with Cluster X, which has decayed from its original 260-nm orbit to 254 nm. The OTES and the OMRF are individually docked to the MDA by the LM and CSM, respectively. The CSM then docks with the docking adapter to commence orbit operations. The X-1 and X-2 spacecraft astronauts depart for earth after experiment handover to the X-3 astronauts. Resupply and crew change is performed every 90 days. Follow-on X-5, X-6, and later launches would be accomplished for follow-on mission applications as required.

6.2 LAUNCH CONSTRAINTS

The X-3 and X-4 launches are constrained by two conditions: (1) coplanar launch with the existing cluster orbit plane and (2) daylight launch conditions between dawn and noon for potential abort and crew recovery operations. Rendezvous launch periods

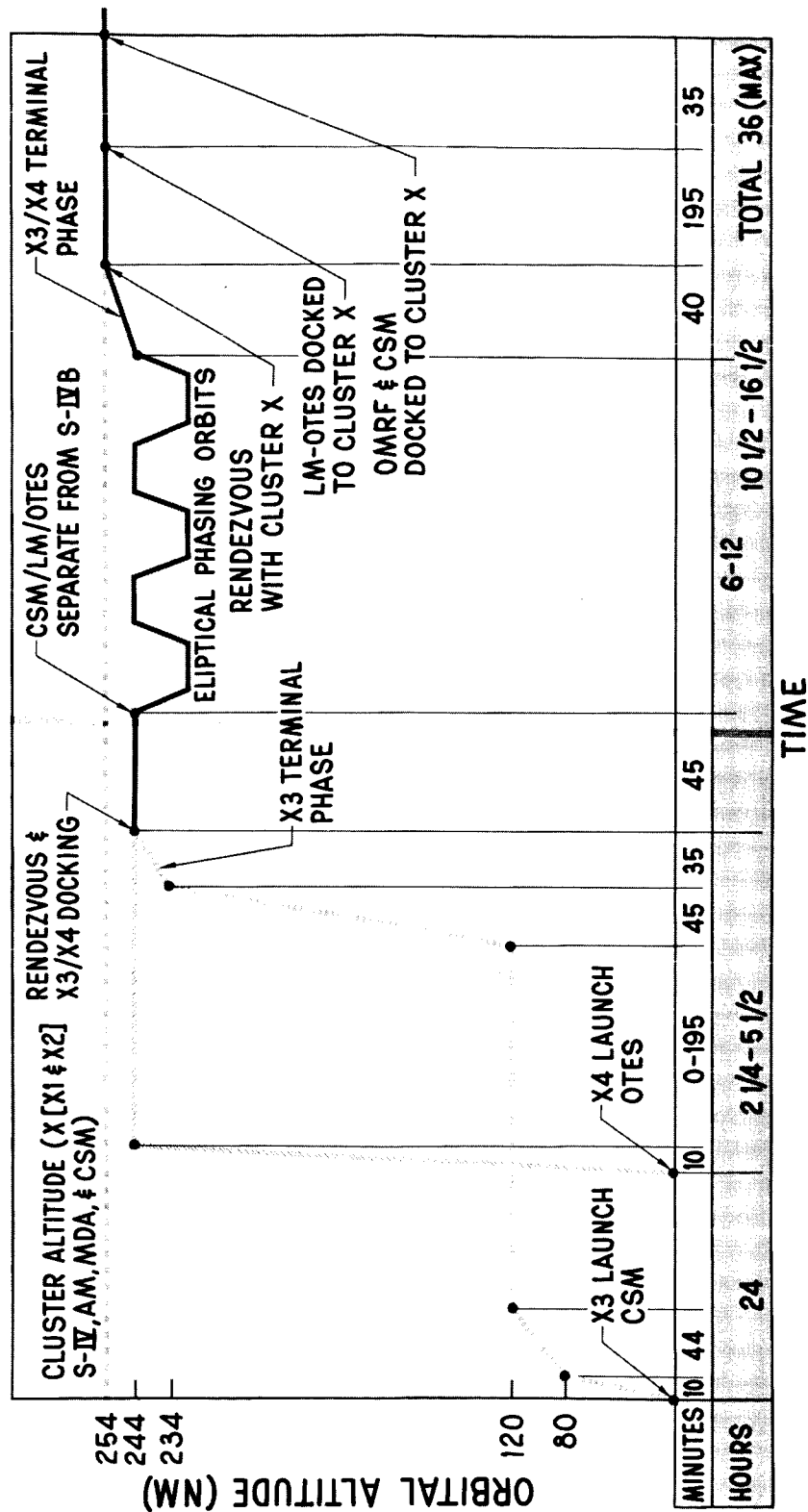
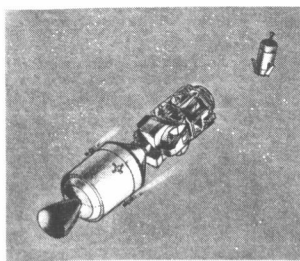
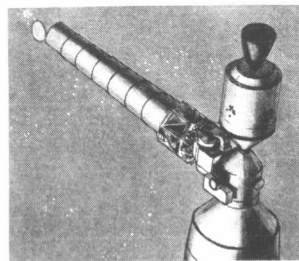


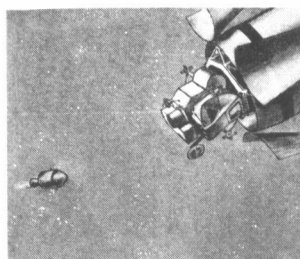
Fig. 6-1 Mission Profile



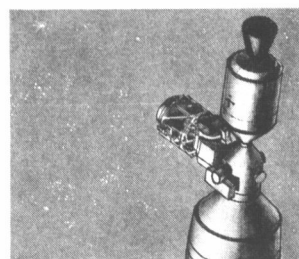
SEPARATION
FROM S-IVB



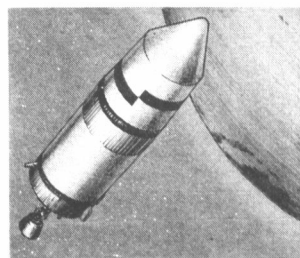
INITIATE OTES
OPERATIONS



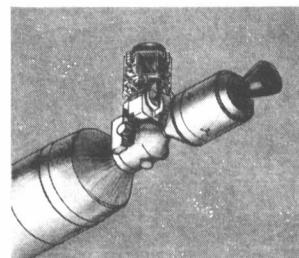
RENDEZVOUS
AND DOCKING



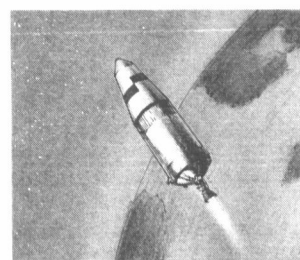
STABILIZE
CLUSTER X



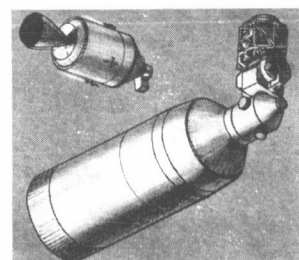
CIRCULARIZE
240 N.M. ORBIT



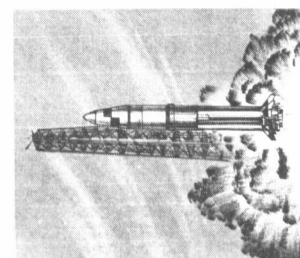
DOCK OMRF
AND CSM



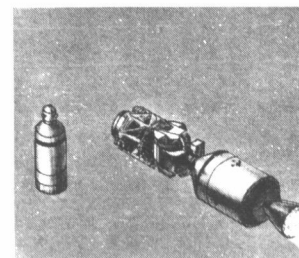
S-IVB
BURN TO ORBIT



SEPARATE AND
DOCK LM



LAUNCH
SATURN IB



ORBIT
TRANSFER

Fig. 6-2 Operations Sequence

occur every 46 days (each period lasting from 9 to 13 days) as a result of the changing orientation of the orbit plane relative to the direction of the sun. The cluster orbit plane regresses 6.8 deg/day; the earth-sunline rotates 0.10 deg/day. For midwinter launches, 5 hr of daylight are available, resulting in only a 10-day launch period.

The launch times for X-3 and X-4 must be chosen for completion of their rendezvous in an orbit position compatible with cluster rendezvous. The X-4 launch time is based on proper phasing injection with respect to the cluster, and nonplanar conditions are corrected by yaw steering during ascent. The X-3 launch time must be compatible with the planned X-4 launch. Time errors in the X-3 and X-4 liftoff are corrected by orbit phasing as shown in Fig. 6-1.

Orbit altitude decay is a function of cluster area to mass ratio. To achieve extended orbit stay times or to raise the cluster orbit, the reaction control system (RCS) or service propulsion system (SPS) on the CSM are used to impart a delta velocity to restore the desired conditions. Each orbit adjustment is individually calculated, since propellant depletion changes the cluster total mass. Use of the RCS propellant, with a lower specific impulse, is less efficient than the SPS; for orbit increases in excess of 8 nm, the SPS is always used.

Section 7

RESOURCES PLANNING SUMMARY

7.1 INTRODUCTION

During the OTES Phase II study, LMSC performed preliminary analyses to identify the resources required for implementation of the preferred 2-m OTES design. The principal objective of the effort was to provide as realistically as possible the detailed definitions of technical, test, facilities, schedules, and costs for the conceptual system design through flight operations and support. Results of the analysis are the best current estimates for use in continued program requirements definition, rather than a firm proposal for LMSC to conduct the OTES Program as presented herein.

This is not a total OTES integrated resources plan. The data presented are exclusively related to telescope development and spacecraft integration without complete definition of the optical subsystem. It is noted, however, that the resources analysis and resulting preliminary planning indicate that the critical item for a July 1973 launch is the optical subsystem development activities.

It is assumed that the principal integration and test activities at MSFC, MSC, and KSC will be performed by the AAP contractor or Government personnel. However, for convenience of presentation, AAP tasks and AAP/LMSC estimated costs are identified for reference purposes.

7.2 TECHNICAL PLAN

This investigation covers all aspects of the LMSC portion of the OTES Program to a broad conceptual level of detail. Included are the baseline assumptions, project

objectives, and a proposed development schedule; an identification of major task groups and hardware classes; and an estimated project cost summary.

7.3 FACILITIES PLAN

The facilities investigation provided an estimate of the facilities required to support all activities related directly to LMSC's spacecraft development efforts for the OTES Program. No new facilities are required beyond those currently existing or currently proposed for AAP.

7.4 TEST PLAN

The test requirements investigation was confined to LMSC test-related activities and hardware. The planning describes several types of tests, test documentation, and test prototypes. Test hardware, instrumentation, operations, and data analysis, as well as planning and coordination activities, are identified in the abbreviated master test schedule and cost summary.

7.5 SCHEDULES AND COST PLAN

This investigation provided a conceptual program schedule (Fig. 7-1) covering all remaining project phases from January 1968 to December 1974. The estimated schedule covers activities for the remaining phases of OTES development through final flight reporting.

The preliminary cost analysis covers LMSC labor, materials, support services, and hardware to be produced (breadboards, tools, test equipment, prototypes, and flight hardware) for the envisioned OTES Program. AAP payload integration and AAP spacecraft modification costs postulated for OTES Program integration activities are indicated here as additional cost reference material. Estimated OTES Program costs are presented in various levels of detail: total program yearly summary (Fig. 7-2), program phase summaries, and by gross task breakdown.

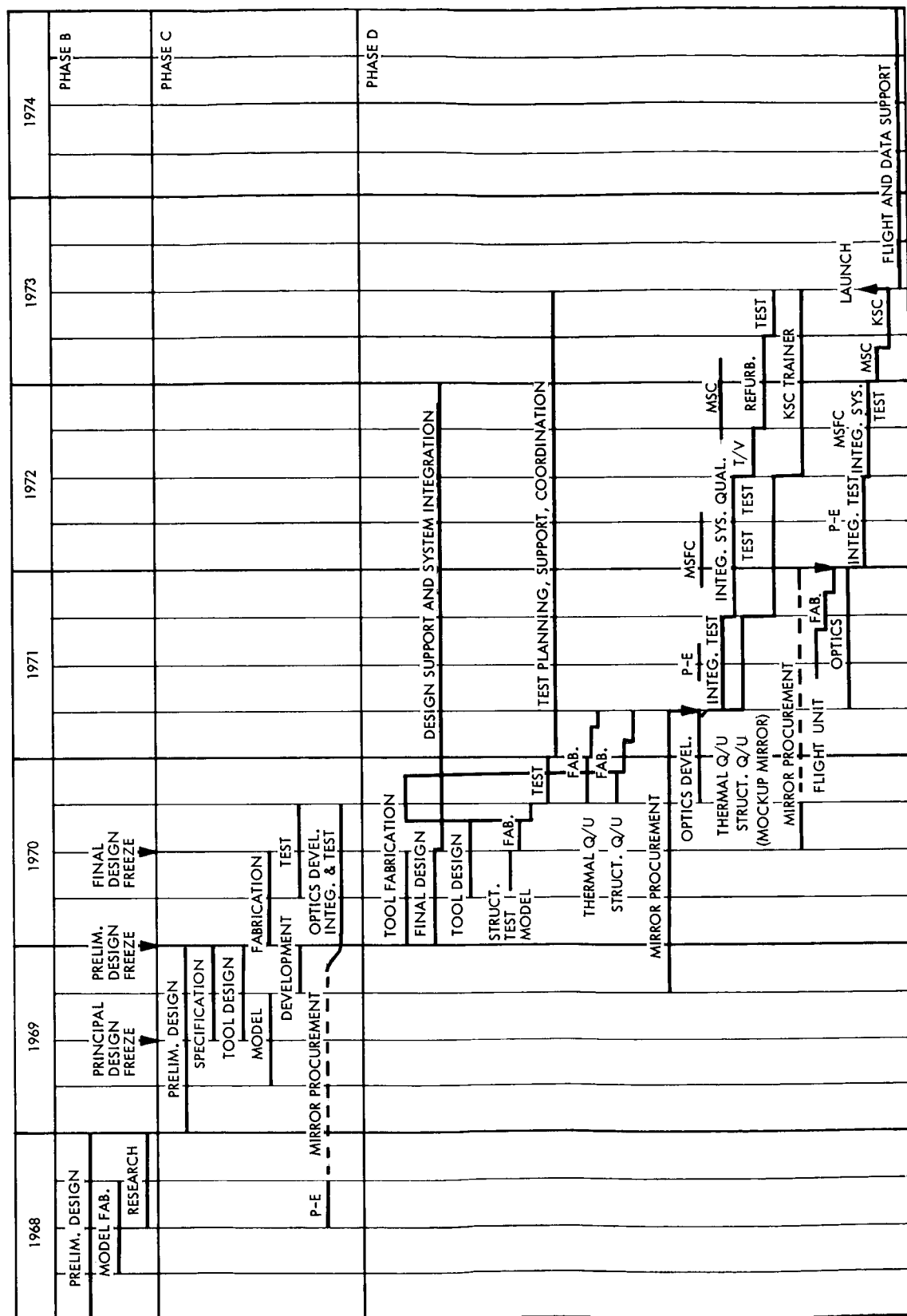


Fig. 7-1 OTES Program Schedule

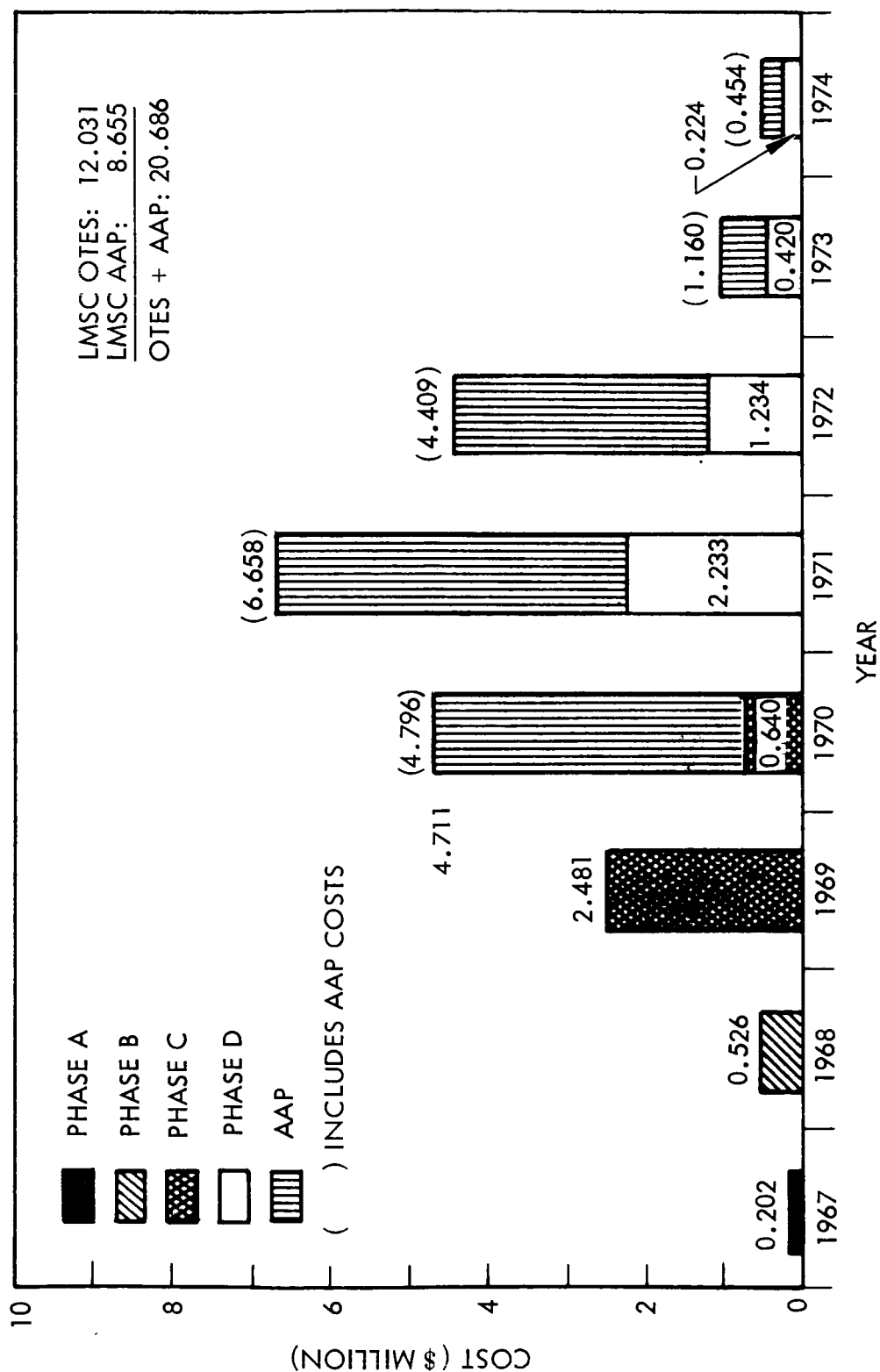


Fig. 7-2 OTES Spacecraft Cost Summary

Section 8

CONCLUSIONS AND RECOMMENDATIONS

Following is a summary of major conclusions and recommendations resulting from the Phase II OTES conceptual design studies presented in this final report:

8.1 CONCLUSIONS

- The baseline spacecraft concept fully accommodates the selected OTES experiments identified by Perkin-Elmer for the OTES mission.
- The concept facilitates integration of a full 2-m-diameter primary mirror (which may be segmented or solid) telescope into the ATM rack and into the AAP operational scheme.
- Passive thermal control system techniques will maintain the primary mirror temperature gradient to less than 1° C and provide thermal control to maintain the mirror operating environment near the desired -80° C.
- Evaluation of the AAP subsystems and performance specifications (i.e., power, communications, stabilization, pointing, etc.) indicates adequate capabilities exist to support the OTES spacecraft orbit operations in either the cluster or independent mode configurations.
- Only minor modifications to AAP subsystems are required to satisfy OTES applications.
- OTES 2-yr life goal attainment requires astronaut maintenance and resupply.
- Astronaut task development (simulation and tests) is highly desirable during Phases B, C, and D of the OTES Program.
- Existing and proposed AAP facilities are adequate for OTES Program needs.
- Mid-1973 launch can be accomplished within the OTES conceptual design approach.

8.2 RECOMMENDATIONS

- The recommended OTES concept selected during Phase II studies should be refined and developed further in preliminary design (Phase B) activities.
- In the process of advancing the design level from conceptual to preliminary design phases:
 - Major emphasis should be on electromechanical operating mechanisms, primary and secondary structures, and passive thermal-control design details.
 - Minimum efforts are required on AAP spacecraft subsystems until the AAP subsystem configurations are established firmly.
- A vigorous program of tests and simulations should be undertaken to establish firm astronaut "task" development data required for design, operational, and safety considerations.

Section 1 INTRODUCTION

This report presents the results of Lockheed Missiles & Space Company (LMSC) Phase II studies (Part II or Phase A, Conceptual Design) of spacecraft conceptual design in support of the manned Optical Technology Experiment System (OTES) Program under sub-contract to Perkin-Elmer Corporation (P-E).

Tradeoff studies have been performed for the OTES Program in the two major areas of spacecraft requirements and related astronaut participation. A preferred approach evolved from the combined attributes of a number of candidate techniques for OTES spar designs. Problem areas recognized in each of the candidate systems were eliminated or minimized in the preferred conceptual approach.

During Phase I studies, LMSC recommended development of a 3-m-aperture telescope/Manned Optical Technology Experiment Laboratory (MOTEL) conceptual approach to the OTES Program. Early in the Phase II effort, however, it became apparent that a less ambitious and more economical approach consistent with Apollo Applications Program (AAP) state-of-the-art capabilities and operational procedures would be very desirable. Therefore, an approach was implemented to include a smaller experiments demonstration model (a 2-m design) that could be readily integrated into the AAP and launched in 1973. The resulting 2-m design requires only minor modifications for integration of the OTES with the existing Apollo Telescope Mount (ATM) rack. It is also compatible with launch and operational constraints of the Saturn launch vehicle and the AAP. The system concept provides for extended growth to a fully operational 2-m telescope and, in addition, provides for an evolution to a full 3-m telescope spar after the concepts and techniques are demonstrated experimentally in the 1973 OTES flight program. Higher probability of success (resulting from minimum new development requirements) and lower overall program investment risks are considered to be advantages of this

approach. Since LMSC, under separate contract to NASA, has conducted considerable analytical work in support of the AAP Cluster A conceptual analyses, it has been possible to provide the OTES Program with a comprehensive and more detailed level of analysis than would have been possible without results of these previous efforts. In effect, the proposed OTES conceptual design is in some respects beyond the concept definition stage; i.e., some of the conceptual definition almost is as complete in detail as a preliminary design. This advanced state of development will continue to grow with LMSC's continued concurrent support of the AAP and OTES development programs.

Hardware and techniques that will be fully flight qualified and flown on the AAP cluster have been considered and will be used to a maximum extent on OTES with little, if any, modification. NASA planning, guidelines, and constraints established by midpoint of the OTES Phase II study were examined in detail to determine the extent of new development that would be required for OTES. Astronaut participation was reevaluated in light of the new concept approach for a 2-m telescope and most recent NASA data relative to manned space flight. Since astronaut capabilities for onboard experiments have been established (in a preliminary way) and the area of extravehicular activities (EVA) is so important to OTES, the primary level of effort was given to the EVA modes of manned participation, evaluation of behavioral modes, and training and facilities requirements to assure optimum astronaut performance for a representative grouping of EVA tasks.

Current NASA planning for AAP cluster missions, requirements, constraints, and guidelines are expected to be changed, modified, and improved as that program advances. These improvements will of course be reflected in the OTES design and operational considerations as the OTES Program continues.

Section 2

STUDY PROGRAM ACTIVITIES, GUIDELINES, AND CONSTRAINTS

The results of LMSC Phase I and IA studies for conceptual definition of spacecraft and astronaut activities associated with selected Perkin-Elmer Corporation optical systems conceptual designs were submitted to P-E in LMSC-A820889, "Optical Technology Experiment System (OTES), Phase I Final Technical Report, LMSC Input," dated 1 September 1966. This document recommended development of a conceptual design for 3-m telescope (primary aperture diameter) and a Manned Optical Technology Experiment Laboratory (MOTEL) spacecraft system. The Phase II LMSC OTES study was initiated on 16 November 1966 on the basis of Phase I study results and involved analyses of the constraints imposed by the Saturn-Apollo hardware on the OTES conceptual design and an assumed 1973 OTES flight program. This study was augmented by findings of other LMSC contractual studies for NASA on AAP payload integration projects, design activities, and hardware development programs to provide a realistic foundation for evaluation of OTES requirements. Overall program planning activities and conceptual guidelines forming the ground rules for the Phase II study program are presented in this section.

2.1 PROGRAM PLANNING

Early in the study program it was decided to pursue a course of evolutionary development toward the 3-m telescope concept using a 2-m, seven-segment, active-optics primary mirror (OTES) to be mated to the Apollo Telescope Mount (ATM) experiments rack. This OTES system would be flown on an AAP cluster and would perform selected manned optical technology experiments beginning in the 1973 flight period. The resulting design concept, somewhat less complex and more compatible with the compressed development period, would serve as a precursor to a future 3-m OTES system.

The principal objective of the Phase II OTES study program is to generate conceptual designs and to evaluate the degree of participation of astronauts in the manned OTES Program for stellar and earth-pointing telescope experiments. A secondary objective is to provide a preliminary analysis and plan for resources required for implementation of the OTES Program. In accordance with these objectives, LMSC and P-E negotiated a Statement of Work covering the scope of the Phase II study. Identified were approximately 40 tasks that LMSC would perform in the following general areas:

- Spacecraft design concepts
- Structural analysis
- Dynamics analysis
- Attitude-control-system analysis
- Crew participation evaluation
- Thermal design requirements
- Resources planning
- Conceptual design model fabrication

2.2 REPORTS AND BRIEFINGS

In addition to a Contact Plan and this final report, throughout the study LMSC submitted to Perkin-Elmer monthly letter reports. P-E, in turn, provided the LMSC OTES study personnel with reciprocal reports of the Perkin-Elmer optic system studies for coordination with the conceptual design study efforts. Additionally, a number of formal and informal meetings and briefings were held among P-E, LMSC, and NASA personnel upon request of P-E and LMSC. Within LMSC, technical documentation and personal briefings of OTES study personnel by members of the Lockheed personnel participating in the Phase C Apollo Applications Program (Cluster A flight program planning) were made available for coordination of OTES system design and astronaut-related design requirements. LMSC also will participate in the final report presentation and briefings to NASA subsequent to final report submittal by Perkin-Elmer.

2.3 GUIDELINES AND CONSTRAINTS

Selection of the 2-m telescope design and maximum application of Apollo Applications Program flight-qualified hardware and techniques with minor modifications were brought about by the following considerations:

- A 1973 OTES first-flight/experiment capability
- Long-life, highly reliable operation
- AAP Cluster A (and subsequent cluster development) prospects of providing fully developed and qualified flight hardware as "off-the-shelf" systems
- Demonstration of the feasibility of the optical technology prior to commitment to the 3-m operational system
- Program economy by elimination of new development/qualification/testing operations associated with new designs

Hence, the specific design guidelines selected for the conduct of the Phase II program were mapped for the spacecraft concept, reliability goals, life requirements, crew participation, Saturn IB, ATM gimbal system, attitude-control and pointing systems, etc., in accordance with presently available data regarding AAP guidelines established by NASA and resulting from Lockheed studies of the AAP ATM solar telescope. Detailed and specific design guidelines for each of the areas of study have been incorporated in the technical discussions of this document. The following paragraphs summarize critical areas of concern and guidelines adopted for the Phase II study program.

2.3.1 Spacecraft Concept

The spacecraft for the OTES Program must be designed as a manned optical technology experiment system. Technical design parameters established are as follows:

- The OTES must mate with the ATM rack with little rack modification.
- The primary mode of operations will be conducted as an integral part of the AAP cluster.

- The OTES experiment and associated Orbital Mirror Recoating Facility (OMRF) experiment will be launched via a single Saturn IB and subsequently docked with other AAP spacecraft modules in a cluster.
- Saturn-Apollo hardware and procedures will be utilized fully without modification where possible.
- The concept must provide for accommodation of an optical system employing a 2-m-diam., seven-segment, active-optics primary mirror – designed by Perkin-Elmer – as well as the electro-optical hardware necessary to support the experiments listed in subsection 2.4.
- Appropriate accommodations for astronaut maintenance of the OTES in space will be provided.

2.3.2 Reliability Goals

The OTES design reliability goals will be in full compliance with other aspects of the AAP program, i.e., a crew survival goal of 0.999 and a system (mission) goal of 0.90.

2.3.3 Life Requirements

The OTES life requirement has been established at from 2 to 10 years' orbit life, using resupply and astronaut-maintenance and crew-replacement concepts of the AAP.

2.3.4 Crew Participation

Crew rotation periods will be the same as those established for the AAP cluster missions. Crew tasks to be considered for OTES include direct or remote experiments, equipment operations, mirror segment removal and replacement, film retrieval, mirror segment coating or recoating, and specified maintenance functions.

2.3.5 Mirror Thermal Constraints

The desired operating temperature of the primary mirror has been established at -80°C ; the thermal gradient of the mirror must be no more than 1°C .

2.3.6 ATM Gimbal System Flexural Pivot

Design criteria established for the flexural pivot is as follows:

- Pitch and yaw: ± 3 deg, with caging capability
- Roll: ± 95 deg, with caging capability
- Pitch and yaw caging within 10 sec
- No power for any caging mode
- Launch loads: 5 g vertical and 5 g lateral
- Roll error: 10 arc min maximum
- Ground loads – vertical and lateral:

Per pivot: limit = 2,500 lb; ultimate = 4,000 lb

Per pair of pivots: limit = 5,000 lb; ultimate = 8,000 lb

2.3.7 Attitude-Control and Pointing System

The attitude-control and pointing system will demonstrate a pointing accuracy of ± 2.5 arc sec.

2.3.8 Saturn IB Constraints

The payload capability of the Saturn IB for a 240-nm circular orbit inclined 28.3 deg to the north is 29,800 lb. Launch load factors are as follows:

- | | | |
|---|---|--------------|
| <ul style="list-style-type: none">● Axial load 5 g● Lateral load 5 g | } | Not combined |
|---|---|--------------|

2.4 OPTICAL TECHNOLOGY EXPERIMENTS

The technology experiments listed below are those proposed for the 2-m OTES system operating in a low (250-nm) earth orbit:

1, 7, 8	Segmented active optics, including the thin deformable mirror and active thermal control of one segment
2	Super-cold telescope
3	Astronaut utilization (configuration changes)
4	Astronaut utilization (mirror figure measurement)
6	Mirror-coating facility in space
9	Verification of precision pointing limits
11, 15	Radiation effects on photosensors and optical elements
12, 13	Thermal diagnostics experiments
26	Suspension systems comparison
29	High-resolution atmospheric absorption
31	Optical heterodyning in space
32	Atmospheric effects on polarization
34	Optical scattering environment

Section 3

SPACECRAFT ANALYSIS

The primary objective of the LMSC Phase II OTES study has been to generate a conceptual design(s) for a manned OTES (2-m telescope) to be mated with the Apollo Telescope Mount (ATM) experiments rack of the Apollo Applications Program (AAP). As configured, the design makes maximum use of Saturn Apollo hardware and subsystems and operates within constraints and guidelines presented in Section 2. Since many of the items of AAP hardware are still in planning, proposed, or preliminary design phases and are subject to change, assumptions and design considerations existing as of 1 May 1967 were employed for the Phase II OTES conceptual design study.

3.1 CONCEPT EVOLUTION

Several of the design concepts were defined, evaluated, discarded, or combined during the Phase II study period before a preferred OTES system approach was selected. In each of the conceptual designs considered, the best features were retained and incorporated into a preferred approach. Inadequate or adverse features were also identified and formed a basis for design improvements that are reflected in the final selection. Some of the more significant conceptual designs that were examined in tradeoff studies to select a recommended OTES system concept are discussed below.

3.1.1 Concept A - 1.85-m ATM Spar (Fig. 3-1)

Initially, it appeared the ATM rack configuration would not be large enough to contain a full 2-m mirror. This concept, essentially replacing the current ATM spar, was selected as a compromise and featured a maximum telescope spar diameter of 80 in.

and a 1.85-m-diameter primary mirror. In April 1967, MSFC made significant ATM design changes by moving the gimbal plane from the center of the rack to a point 11.5 in. from its forward end. This change enabled LMSC to reconsider the possibility of a 2-m concept which could now be carried on the forward face of the rack. The 2-m mirror thus could be incorporated, leaving ample clearance for spar gimbaling. The 1.85-m concept was abandoned in favor of the larger mirror, although techniques used in the analysis were adapted to the preferred conceptual design.

3.1.2 Concept B - Pantograph Extension (Fig. 3-2)

Conventional pantograph structures (lazy tongs) have been successfully adapted to a number of erectable spacecraft structures for operational programs. For example, erectable booms and antenna systems using this technique are fully developed. With respect to the OTES telescope, however, the complexity of the design, lack of the necessary stiffness for stellar photography, large number of moving parts and hinges (pivots), and doubtful reliability and life expectancy for a large structure of this type led to elimination of this design. The primary advantage of the system would have been favorable mass and center-of-gravity characteristics and possible weight savings.

3.1.3 Concept C - Increasing Diameter Extension (Fig. 3-3)

This configuration concept is an adaptation of the conventional "Ship's Captain" telescope and comprises a number of cylindrical barrels which slide forward from the viewing end to telescope forward edge. The largest barrel segment would be the farthest extension from the rack with the primary mirror located at the bottom of the smallest segment. For space applications, however, unnecessarily large weights and moments would be concentrated at points far separated from the gimbal plane. The bending EI and torsional JG stiffness values would be lowest near the rack end of the telescope barrel, i.e., the inverse of the best spacecraft design characteristics. Moreover, in the stowed position, the telescoping section arrangement would

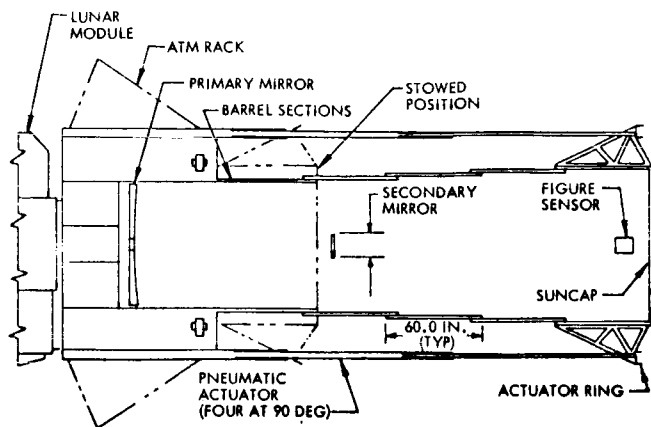


Fig. 3-1 Concept A

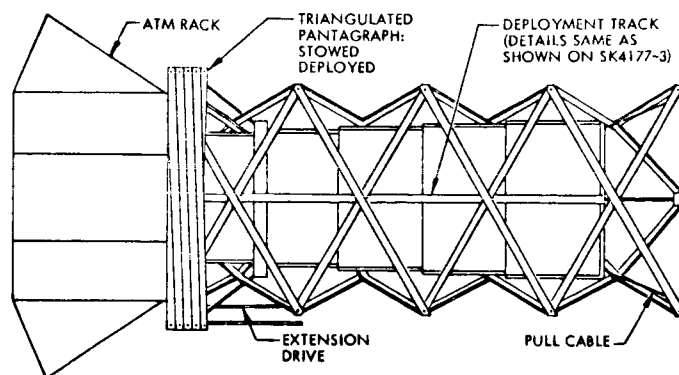


Fig. 3-2 Concept B

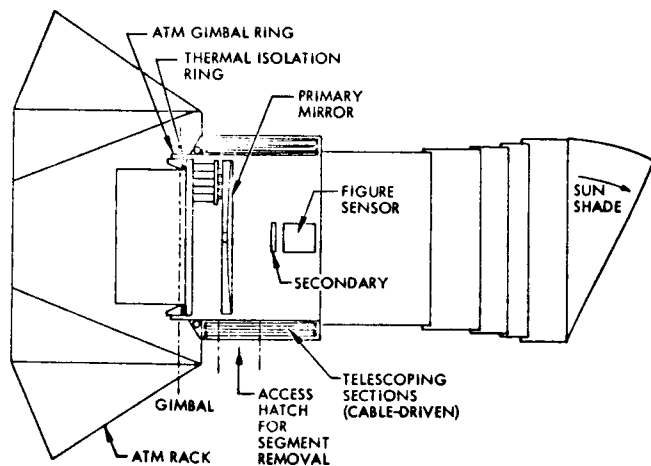


Fig. 3-3 Concept C

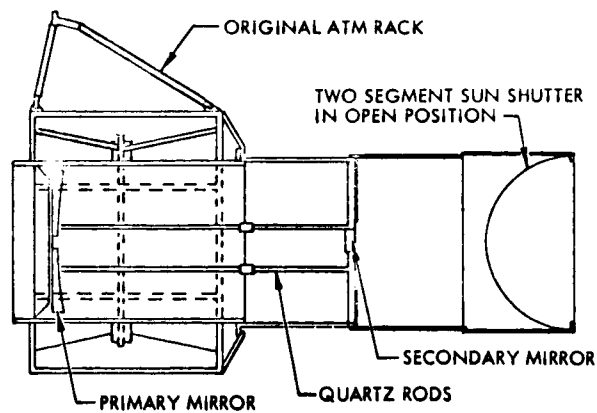


Fig. 3-4 Concept D

prevent attachment of the equipment and hardware in the most desirable areas. The reverse of this configuration, however, an inverted version running from largest to smallest barrel would provide a composite of desirable characteristics. The largest barrel section would contain the primary mirror near the spacecraft CG and the smallest and lightest segment would be at the tip of the telescope, thus providing more ideal moment and mass balance for gimbal control.

3.1.4 Concept D – Quartz-Rod Mirror Supports (Fig. 3-4)

The low coefficient of expansion and thermal stability of quartz is highly desirable for optical system applications where close tolerances are required. This concept employs quartz rods used as structural members, attached to the primary mirror support plate for supporting the secondary mirror and to support the figure sensor as a structural extension from the secondary mirror. Taking advantage of the material characteristics (at the expected range of operating temperatures) to maintain focus distances and optical element positioning, relatively high degrees of accuracy could be efficiently maintained. However, the problems of carrying loads on the quartz rods, rod storage, and installation associated with this configuration resulted in elimination of the design. The preferred system approach for OTES, however, uses quartz rods as spacers rather than as load bearing support structures.

3.1.5 Preferred Concept – Best Features Combined

The preferred and recommended OTES system concept evolved from tradeoff studies and analyses of best features and shortcomings of the conceptual designs previously described. A discussion and description of the proposed OTES design is presented in subsection 3.2 along with details of specific AAP hardware and subsystems which function as integral parts of the OTES spacecraft concept. This concept is based upon currently available data and conceptual designs for the AAP program. The concept will be appropriately refined with further definition, improvement, or refinement of AAP cluster design concepts. However, as presently determined, overall

system requirements for OTES are comparable to or less than those currently defined for the ATM solar telescope system. Therefore, the proposed concept is feasible and totally within current AAP constraints and guidelines.

Evolution of the preferred OTES system conceptual design from a combination of desirable features and techniques of several candidate systems investigated has provided the following operating features:

- Accommodation of a full 2-m-diameter primary mirror
- Employment of a movable secondary mirror and figure sensor
- Accommodation of the experiment for comparison of mechanical and magnetic suspension systems
- Provision for controllable positioning of the telescope center of gravity
- Provision for quartz spacer rods to assist in maintaining focus
- Maintenance of low thermal gradients across the diameter of the primary mirror
- Astronaut participation for both operational and maintenance tasks
- Provision for launch by a single Saturn IB launch vehicle with significantly high additional payload weight capability to assure experimental applications
- Provision for fully automated erection of the telescope, controlled by the astronaut
- Structural rigidity with high EI and JG values
- Operational flexibility for both cluster and independent modes
- Provision for development economy through maximum use of previously developed and qualified Saturn-Apollo hardware
- High reliability compatible with AAP goals for 0.92 mission success and 0.999 probability of crew survival
- Provision for an AAP cluster-compatible Orbital Mirror Recoating Facility (OMRF) for astronaut long-term maintenance experiments

3.2 DESCRIPTION OF PREFERRED CONCEPT

Adaptation of the OTES system concept to planned AAP cluster designs eliminated the necessity to design and develop a number of basic subsystems required for spacecraft missions. Cluster A, comprising the first four Apollo spacecraft launches in 1968 for AAP, combines the available subsystems of the Command Service Module (CMS), Lunar Module (LM), Orbital Workshop (a refurbished segment of the S-IVB launch vehicle, Apollo Telescope Mount (ATM), and the Multiple Docking Adapter (MDA). For the manned OTES mission, the life cell and environmental control subsystems, supporting structures, electrical system, guidance and control, communications, orbit propulsion and attitude control, data management systems, recovery system, etc., will be readily available to OTES as essentially off-the-shelf, flight-qualified hardware from the AAP. In some cases minor modification would be required to adapt the cluster or integral parts used for independent mode operations to the specific OTES stellar or earth-pointing experiments. In general, however, the qualified hardware remains in the AAP configuration.

Because many of the subsystem requirements can be met using the fully designed and qualified AAP hardware, it has been necessary in the study only to verify compatibility in gross terms to assure that they fulfill OTES orbit support requirements. For example, the environment control system, crew quarters, communications, basic electrical subsystem, and attitude control/orbit propulsion, are adequately satisfied by the CSM and LM. The following paragraphs are a description of the OTES/spacecraft concept and subsystems comprising the conceptual design. Optical systems for OTES, discussed in detail in Perkin-Elmer documentation, are discussed only to the extent necessary to define interface requirements.

3.2.1 OTES/Spacecraft Concept

For conceptual analysis, the OTES/spacecraft is considered to be similar to the Cluster A-4 (last of four AAP cluster vehicle launchings) to be docked with space

vehicles of launches A-1, A-2, and A-3 on the Multiple Docking Adapter. In the ascent configuration, the OTES spacecraft comprises a lunar module, AAP ATM experiment rack, and the attached OTES 2-m telescope (in retracted position). An Orbital Mirror Recoating Facility is carried on the A-4 launch and will be docked to the MDA of the cluster at a separate docking station adjacent to the OTES. The A-4 is launched unmanned for rendezvous with the CSM. The CSM is used also to shuttle the OMRF to its docking port. The orbit control and maneuver propulsion of the LM and CSM will be used for docking the OTES to the MDA.

Primary modes of the AAP cluster with OTES are to be augmented by an independent mode operations sequence in which the OTES spacecraft/experiment system operates as a separate spacecraft undocked from the cluster for stellar or earth-target experiments. During the study period, the majority of the independent mode studies were conducted considering the independent mode to use only the LM, ATM rack, and OTES. The OMRF and CSM were to have remained docked to the cluster. Late in the study period, however, NASA planners tended toward a policy of including the CSM as an integral part of the AAP independent mode experiments to assure astronaut safety, i. e., the LM would not permit emergency deorbit and entry. In effect, the configuration as studied is capable of performing independent mode operations. The addition of the CSM to mission planning would only enhance performance on the independent mode operations. This change, therefore, in no way affects the validity of the studies performed to date. Design parameters and physical characteristics of the OTES are presented in Table 3-1.

3.2.2 OTES Structural Subsystem

Primary structural elements of the OTES comprise the support cone, fixed and movable cylindrical barrel segments (shell), electro-optical equipment bay, center-of-gravity positioner, and support provisions for the mirrors and figure sensor. Secondary structural elements are the magnetic suspension system, support rack and engagement mechanisms, free-float positioning mechanism, launch support cage, and cable extension system. These are described in the following paragraphs:

Table 3-1

OTES TELESCOPE MENSURATION

Dimension	Stowed		Extended	
	(in.)	(m)	(in.)	(m)
Lengths				
Forward of Gimbal Plane	104	2.64	389	9.88
Aft of Gimbal Plane	61	1.55	91	2.31
Total	175	4.44	480	12.19
Equipment Modules			39.4	1
Diameters				
Maximum OD			114	2.90
Minimum ID			87	2.21
Built-in Gimbal Ring OD			88	2.24
Optics				
Primary to Secondary Mirrors			130	3.30
Primary to Figure Sensor			312	7.92
Movements				
CG Positioner (Maximum)			30	0.76
CG Travel			18	0.46
Telescope From Gimbal Plane			8.25	0.21
Pitch and Yaw			± 3 deg	± 3 deg
Rotation About LOS			± 95 deg	± 95 deg

Support Cone Structure. The support cone (Fig. 3-5) is a machined, spun-aluminum casting. A portion of the ring frame is machined integrally with the support cone; the other portion of the frame is machined separately. The two parts of the ring frame are welded together to form a complete assembly. This frame assembly replaces the inner gimbal ring of the ATM gimbal system for mating the OTES telescope to the ATM rack. The large end of the support cone is riveted to the large diameter telescope cylindrical barrel as a fixed assembly to house the concentrically located, movable barrel segments during launch and to support the extended spar on orbit. The small end of the support cone attaches to an electro-optical equipment bay and beams which support the center-of-gravity positioner assembly. A 1-in-thick, 80-in-diameter aluminum plate and insulation ring are bolted to the large end of the support cone to support the segmented 2-m-diameter primary mirror. This plate is also a location reference for the four quartz spacer rods.

Cylindrical Telescope Barrel Segments. The telescoping segments of the OTES spar are semimonocoque structures having 0.030-in.-thick aluminum skins that are flush riveted to hat section stringers. The stringers are 0.030-in. thick except for the hat (top) material which is 0.060-in. thick for greater EI values. Hollow ring frames with wedge-shaped mating surfaces are used to provide a high joint stiffness value for the extended spar. Twelve pyrotechnic-actuated pin drivers are used to positively lock the mating rings (and, therefore, the barrel segments) firmly together. The pin drivers are housed in a hollow section of the outer ring frame and drive the pins inward; pyrotechnics are contained by an expanding plastic containment casing to prevent contamination of optical equipment. When locked, the rings form light seals at each joint (Fig. 3-6).

The barrel segments are concentrically spaced in the stowed position, and during spar extension are guided by four slide mechanisms (spaced 90 deg apart around the circumference). The guides consist of Teflon pads attached to the outer surface of an inner shell section which slide on U-shaped tracks machined into special stringers on the inner surface of an outer shell section.

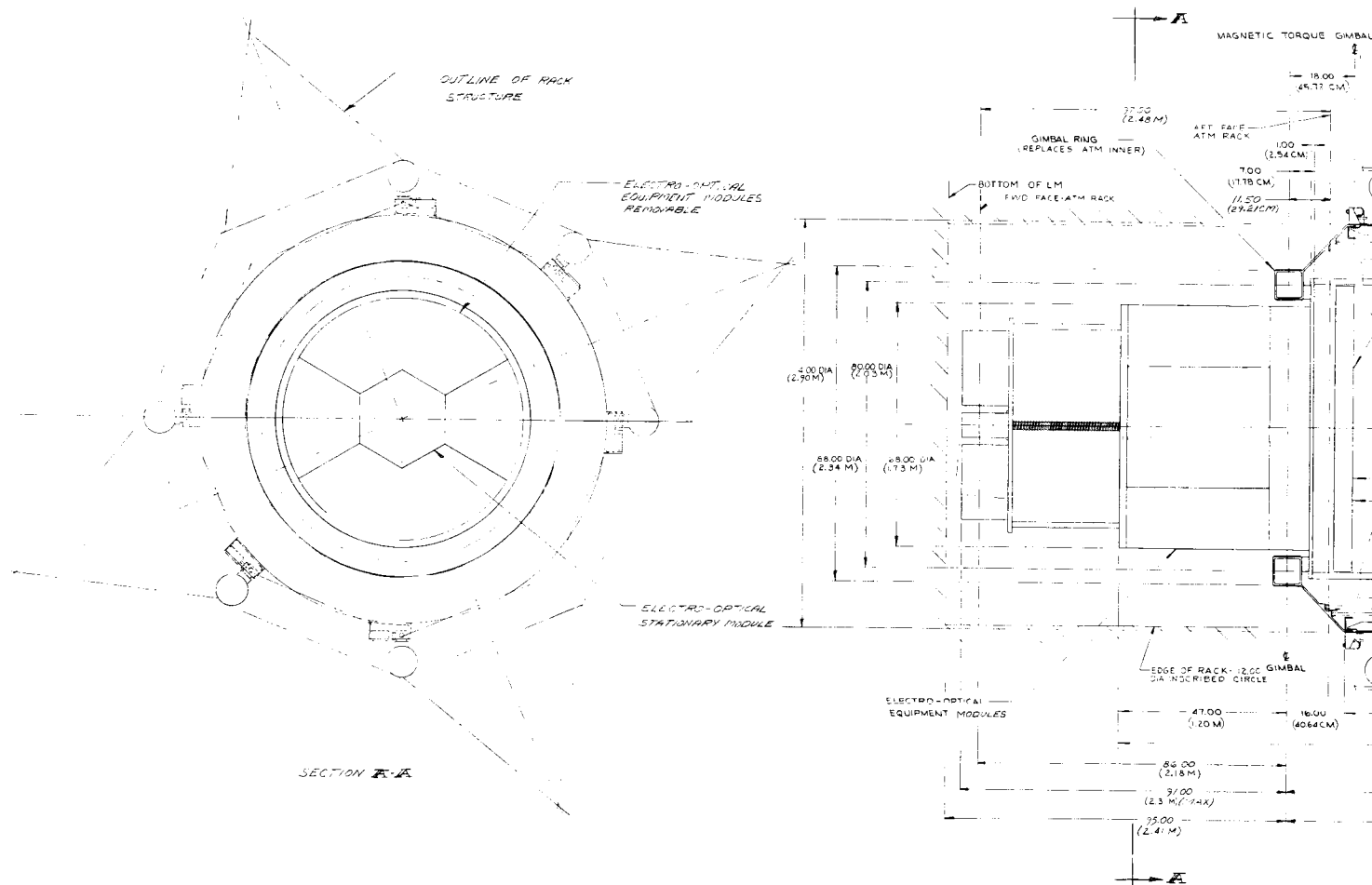
Barrel segments of the spar are arranged to have progressively smaller segments extend from the rack until the smallest diameter barrel extends furthest from the fixed barrel segment and support ring assembly. This provides the **greatest** stiffness value nearest to the support cone and lightest weight (in lb/in.) at points most remote from the center of gravity. Weight and stiffness distribution curves are presented in subsection 3.3.5.

During launch and staging operations, the barrel sections are locked in place by a launch caging structure (Fig. 3-7). Prior to extension of the spar, the caging bracket is released by four pin-puller mechanisms (holding the bracket to the support structure) and jettisoned from the spacecraft by four compression springs.

Electro-Optical Equipment Bay. The electro-optical equipment bay houses six replaceable modules that contain experiments associated with the OTES telescope system. The design features a fixed, central, hexagon-shaped compartment, containing electrical power supplies, and six pie-shaped compartments for modularizing electro-optical system equipment. The canisters are removed and replaced on orbit by astronaut EVA. The modular experiment containers are sized for ease of removal and fitted with side-mounted handles and guides to assist the astronaut during canister replacement. Separation space is provided between modules for locating three (or six beams) to support the CG positioner from the support cone ring. These beams also function as guide rails for the center-of-gravity positioner ballast weight plate.

Center-of-Gravity Positioner. The positioner (Fig. 3-8) is to change the center of gravity of the overall OTES system by movement of a ballast weight through a 30-in. travel distance. Repositioning of the center of gravity directly on the flexural pivot plane is necessary to compensate for the ATM pointing systems extreme sensitivity to CG offset. For the magnetic suspension system experiment, the positioner is used to change the system CG to the plane of the magnetic torquers.

Conceptually, the positioner consists of three telescoping rails which support a heavy plate (or, in another conceptual design approach under consideration, an



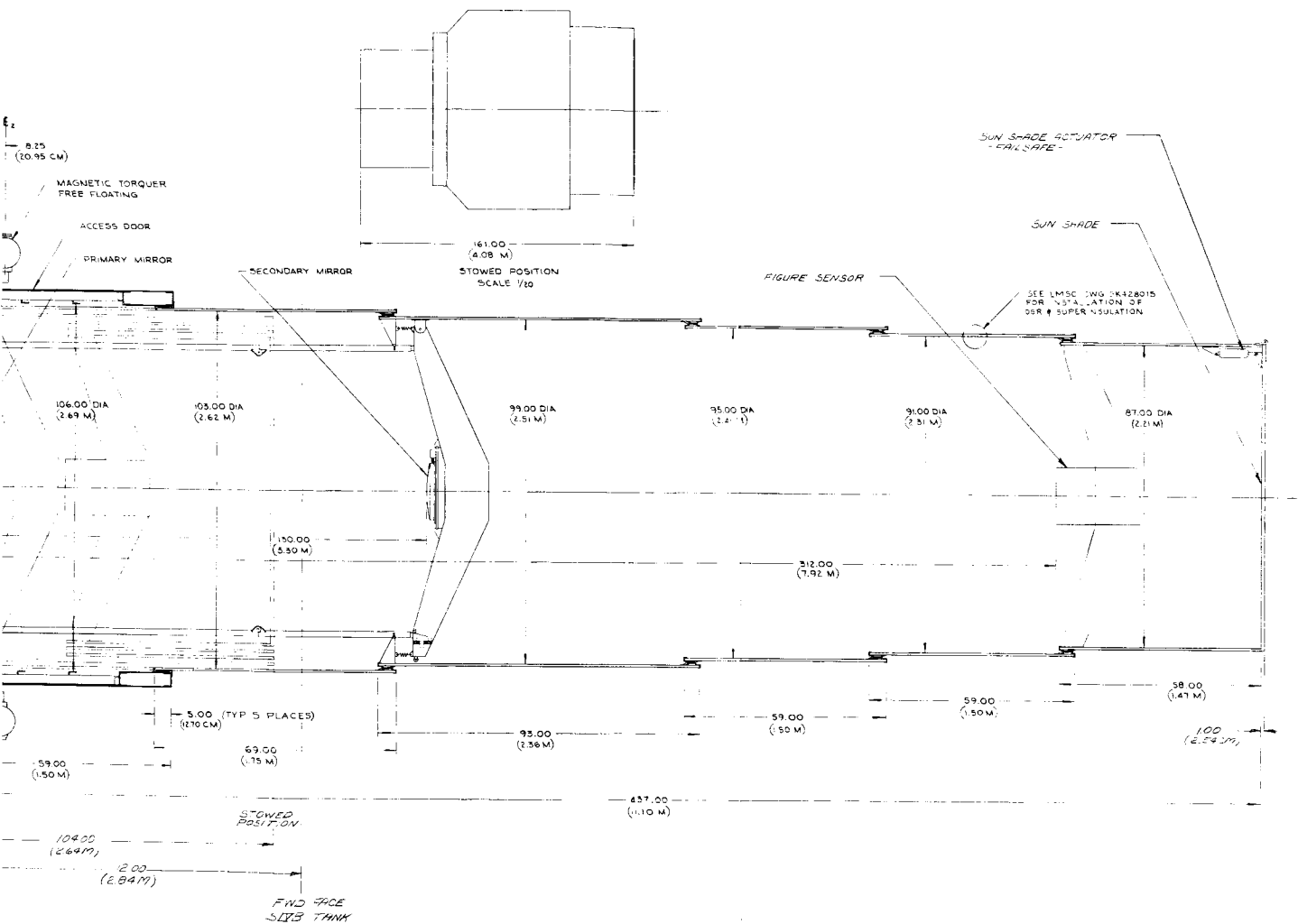


Fig. 3-5 Inboard Profile, 2-m Telescope

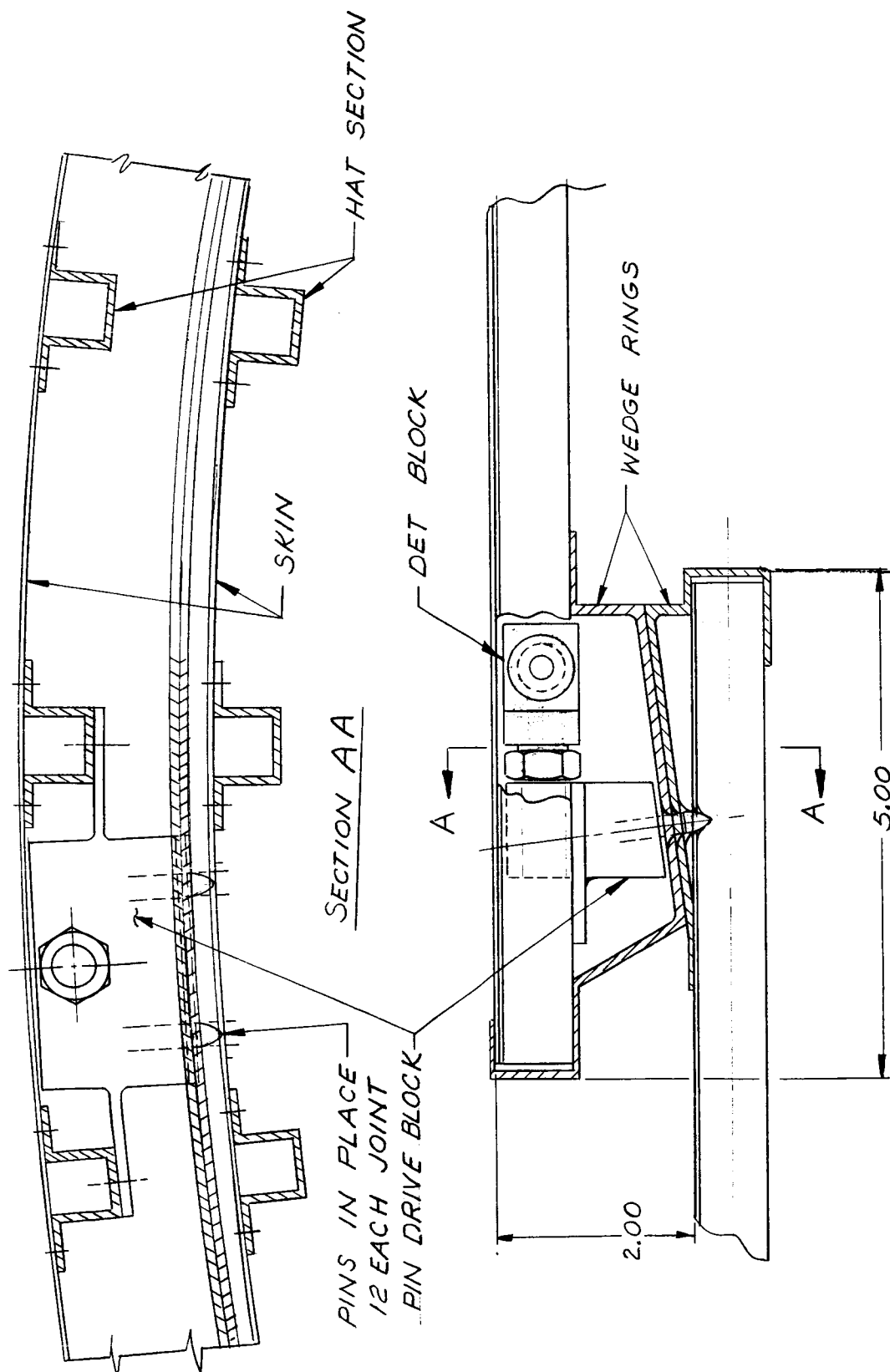
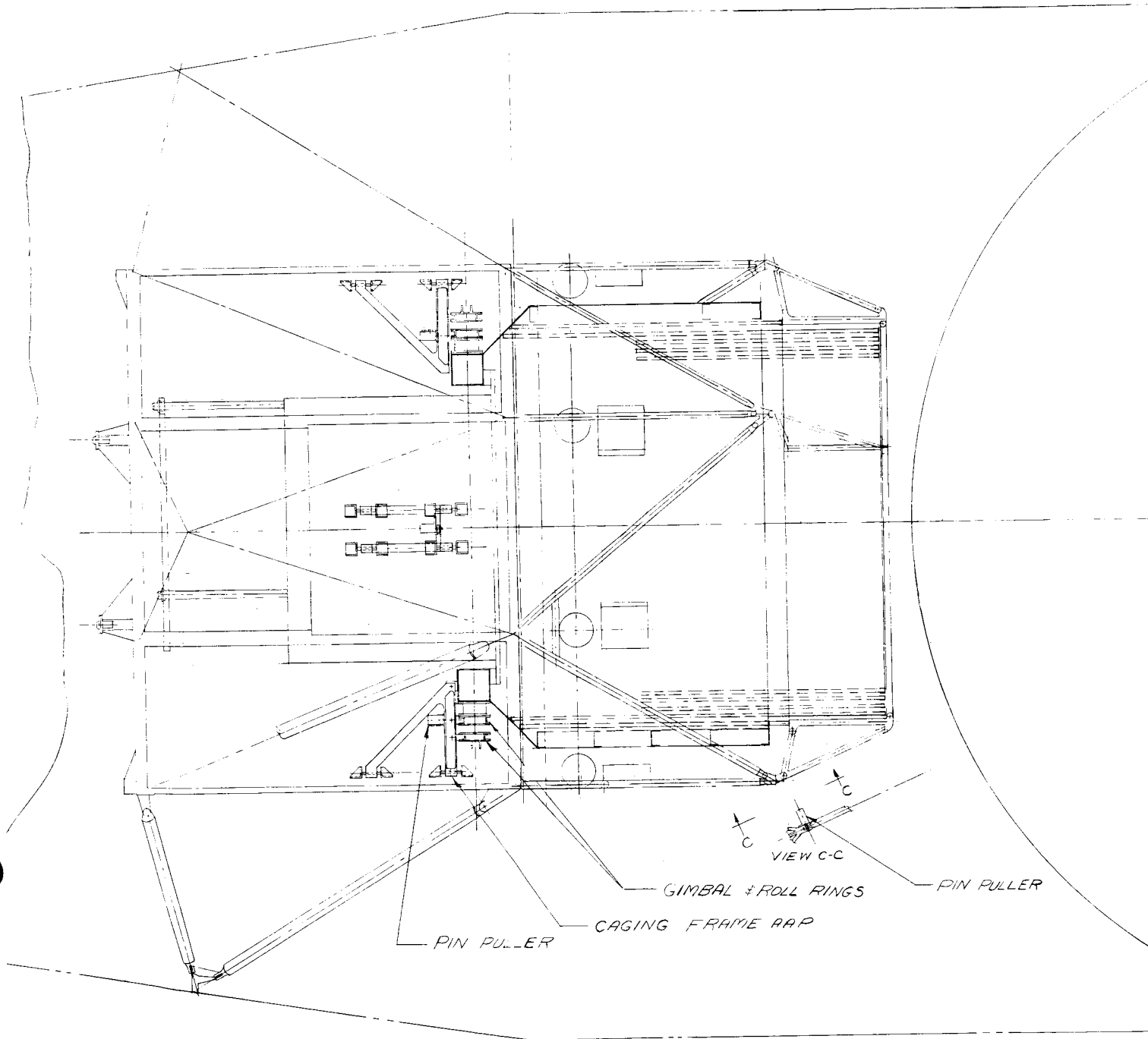


Fig. 3-6 2-m Telescope Section Locking Mechanism



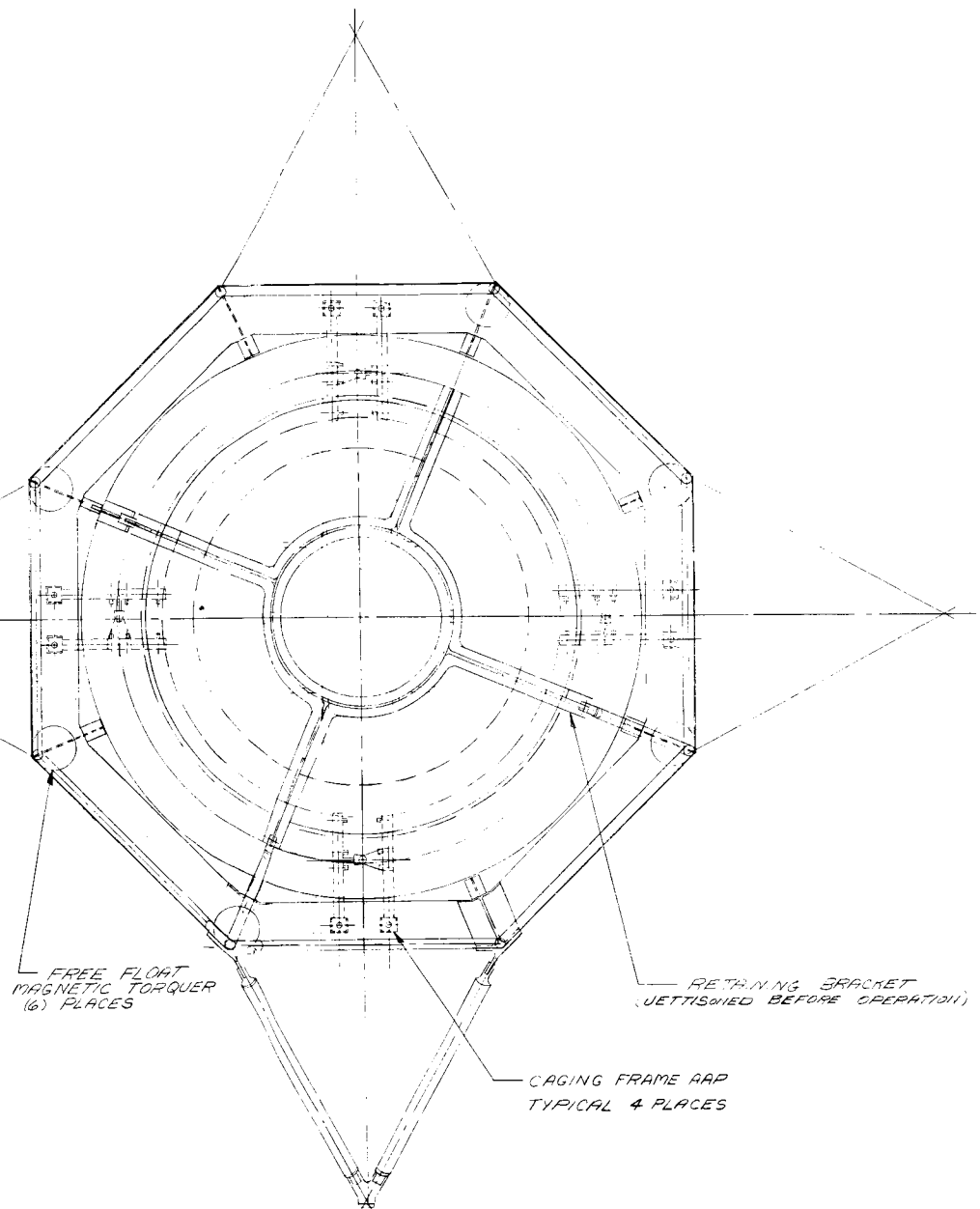


Fig. 3-7 Launch Caging Systems

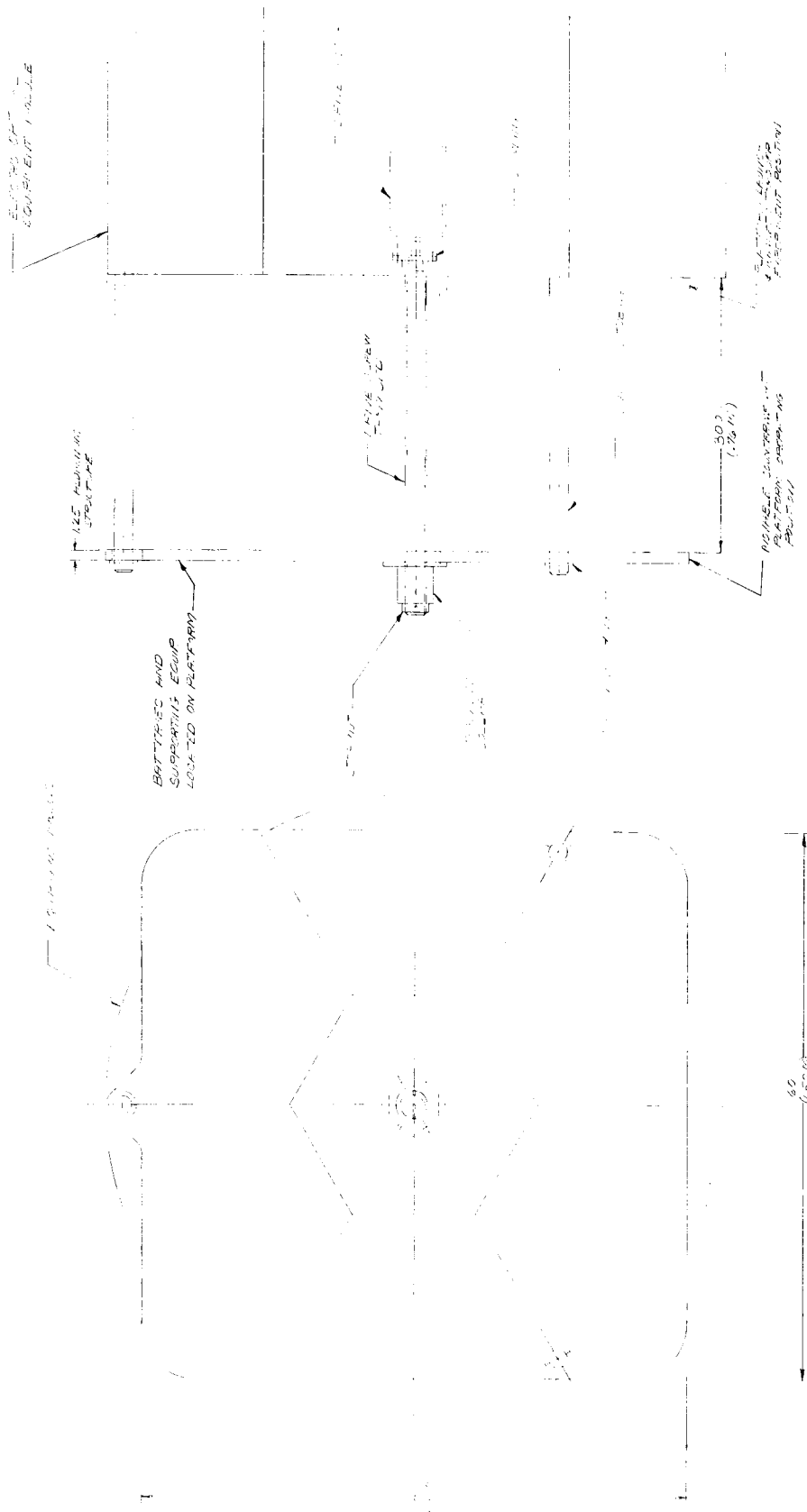


Fig. 3-8 CG Relocation Mechanism

equipment mount plate for heavy items, e. g., batteries and chargers). The positioner concept and space along with clearance availability for ballast are shown in Fig. 3-8. The screw shaft, by means of the reversible motor, forces the counterweight fore or aft. During launch, the weight is located flush against the experiment equipment modules. Before spar extension, the weight is moved for positioning of the center of gravity to the plane of the flexural position. To prepare the magnetic torquer experiment, the positioner plate is moved toward the experiment modules (same as in the launch position); this shifts the center of gravity to the centerline of the torquers.

Optics Support Structures. The secondary mirror and figure sensor support structures (Fig. 3-5) are fabricated from 1/2-in. -thick aluminum honeycomb panels. The sensor supports are rigidly attached to the figure sensor housing and telescope barrel. The secondary mirror supports are rigidly attached to the cylinder supporting the secondary mirror and actuation system but are not attached to the barrel structure. The secondary mirror is held in position by a system of springs and spring-loaded rollers (and is described subsequently as a part of the spacer rod mechanisms discussion).

Magnetic Suspension System Support Structure. The magnetic suspension system support structure is fabricated from tubular aluminum material and is attached to the basic ATM rack (Fig. 3-7). The structure also serves to channel the loads from the launch caging mechanism to the ATM rack during ascent, and to support the mirror segment transfer cases and track. The aluminum tubing is held in place at attach points (clevis brackets) by pin connectors. The structure is appropriately located to prevent interference with the three-axis gimbal system or the structural doors providing astronaut EVA access to the spar interior. Sufficient clearance and hand-holds are provided to facilitate EVA removal of mirror segments.

Free-Float Positioners. The free-float positioners concept (Fig. 3-9) consists of a pair of brackets and linear actuators (with an approximate 9-in. stroke) that are used to disengage the telescope from the three-axis gimbal rings. The brackets are

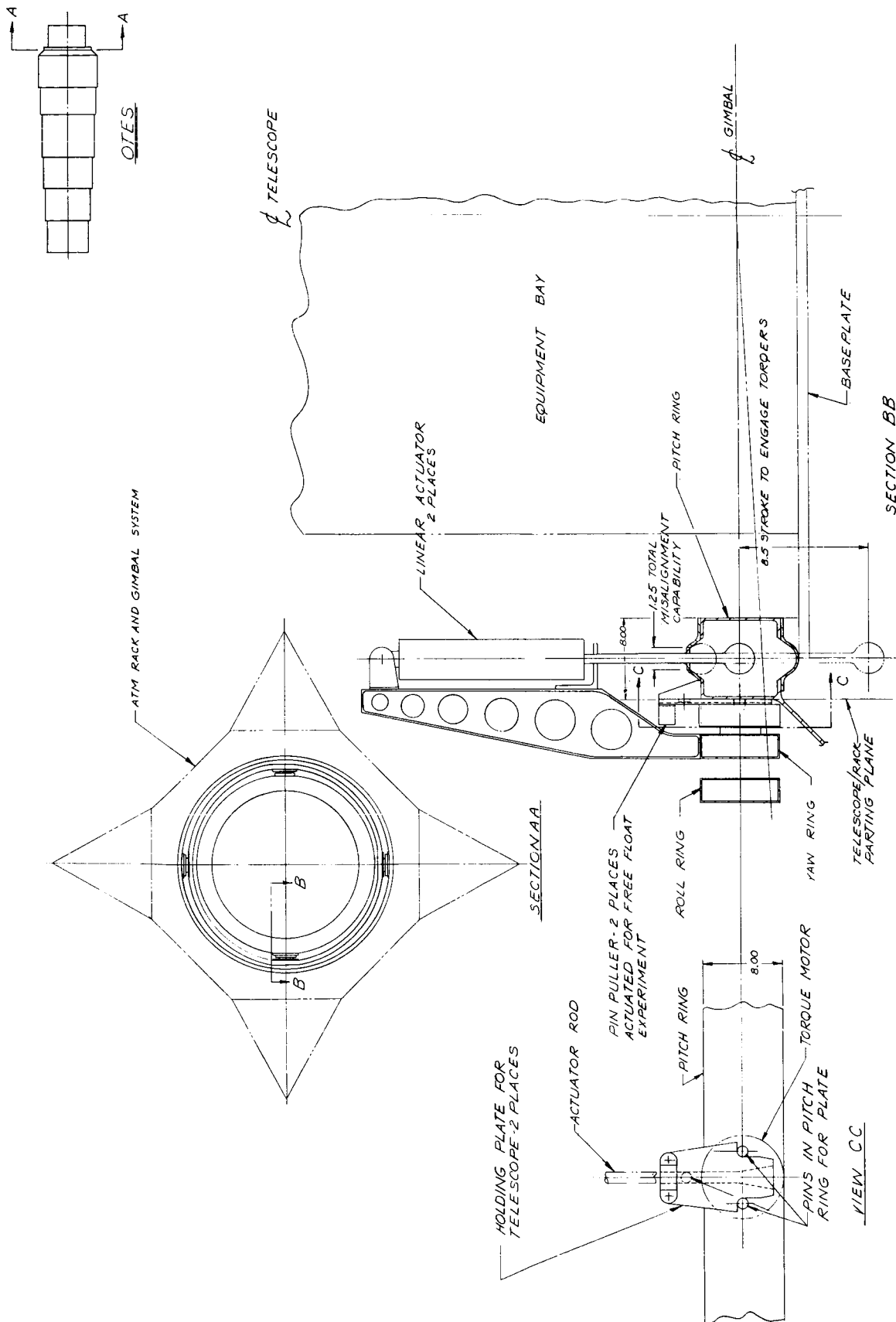


Fig. 3-9 Free-Float Experiment Positioning Mechanism

rigidly attached to the middle gimbal ring and located diametrically across from each other. These brackets support linear actuators. The actuators force the inner gimbal ring (and, therefore, the telescope) to slide forward, separating the inner gimbal ring from the middle ring. During this operation the center-of-gravity positioner operates as described earlier. The moving spar is guided by three pins which engage in slots in each of slide plates. While the magnetic suspension system is operated, the telescope is prevented from rotating about its longitudinal axis more than a fraction of an inch. Therefore, the same slots, beveled on the forward ends, are used to guide the telescope back into position in the gimbal ring. A pin (attached to the spar structure) engages the magnetic torquer arm and is locked in place by a wedge key (to prevent the pin from disengaging as shown in Fig. 3-10). The spar, therefore, remains attached to the rack only by means of the six magnetic torquers, whose fields are varied to torque (or position) the barrel. Both the magnetic suspension system and free-float positioner have redundant power supplies to assure reliable operation during these experiments.

Cable Extension System. During launch, the 2-m telescope is stowed in SLA between the LM and the S-IV tank. The ATM rack serves as a carrier platform and base for the spar. While in the primary operating modes, the spar attitude is stabilized in pitch, roll, and yaw by a three-axis control moment gyro system.

After the OTES is docked to the cluster, an astronaut in the LM activates switches to automatically deploy the spar from the stowed length of 114 in. to an extended length of 389 in. via the cable extension system. Five telescope sections are movable and are extended in concentric tandem fashion by the cable assembly. The system will be able to deploy in zero gravity (in space) or under the 1-g load conditions (for ground checkout with the telescope in a vertical position).

The two cable systems are located 180 deg apart. Each system consists of drive motors, winches, cables, and pulleys. The pulleys are operated by motors and winch assemblies. The two drive motors are synchronized for uniform extension of the

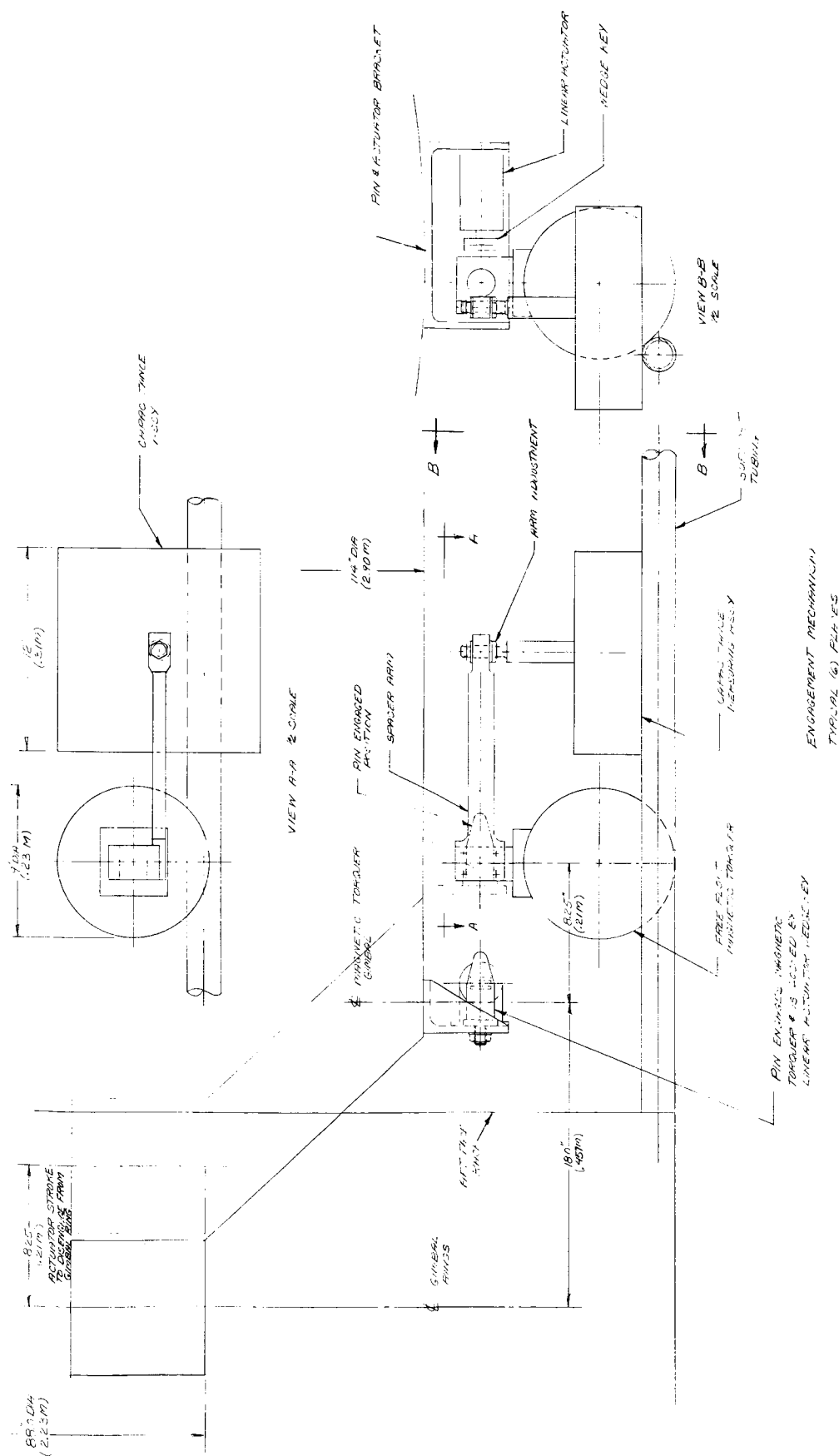


Fig. 3-10 Magnetic Torquer Engagement Mechanism

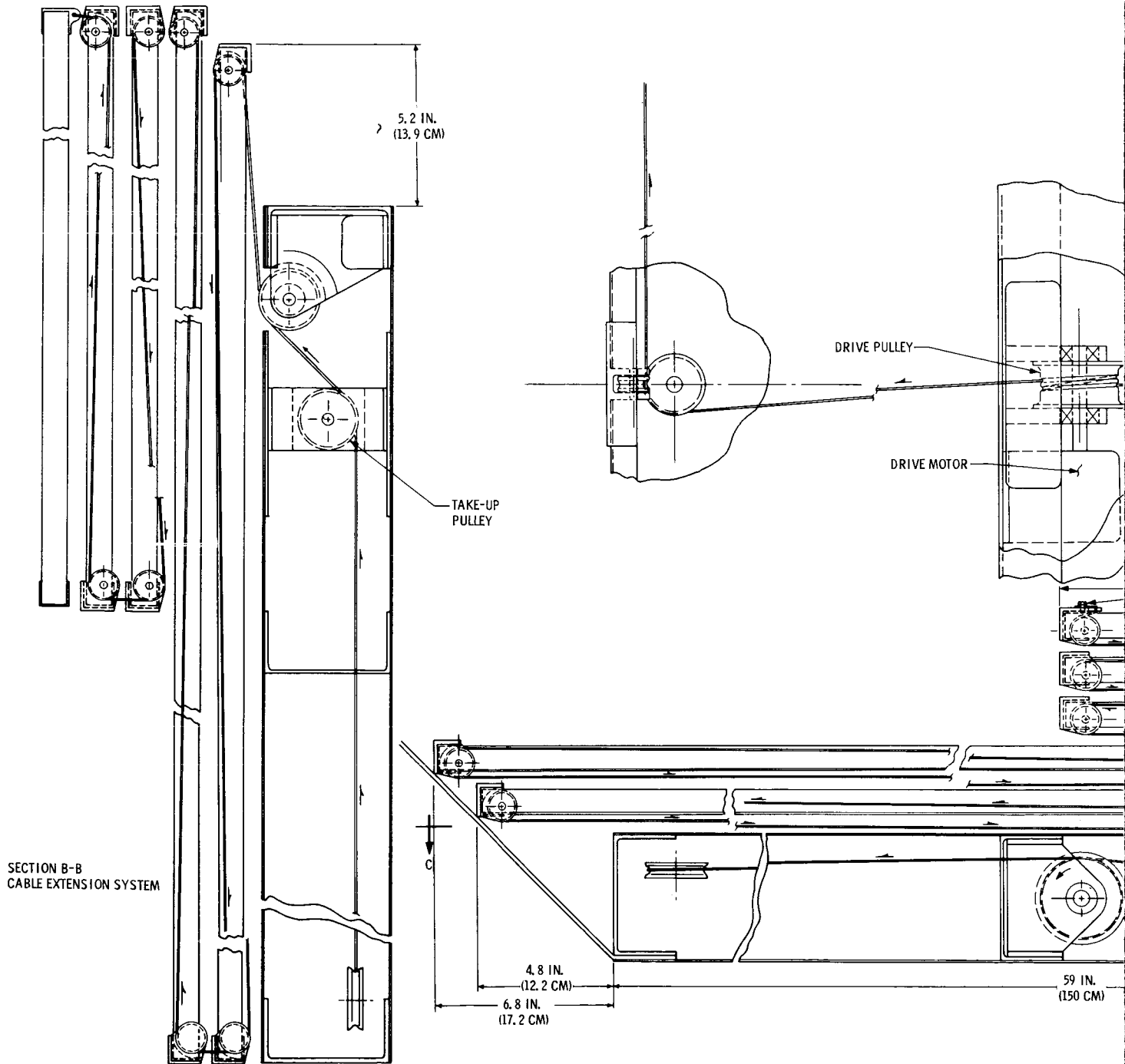
spar sections. A redundant drive motor is provided for each pulley system to add reliability. The drive cables are threaded through the pulleys of each extension segment and are clamped to the central, smallest segment (as shown in Fig. 3-11). As the motor turns, the cable length is taken up on one side of the pulley, forcing the segment to deploy outward. As each segment deploys, the cable is "played out" and equal distance. Features which led to selection of this technique for the OTES baseline design are as follows:

- Cables and pulleys. Lightweight cables and pulleys and related support structure are negligible in terms of added weight in the telescoping segments.
- Redundant equipment. A backup motor can be provided for each drive pulley to assure reliable operation.
- Design simplicity.
- Positive, light-free locking. After deployment, telescope sections are firmly fastened at the joints by locking devices with light-seal provisions.
- No further functional use is required of the extension mechanism.

Quartz-Rod Spacing Mechanism (Fig. 3-12). Four 1-in. -diameter quartz rods are spaced at 90-deg intervals and located 41.50 in. (1.055 m) from the telescope centerline to maintain focus spacing between the primary and secondary mirrors. The folding rods fit within the launch envelope of the spacecraft and are protected from loads and damage of the ascent environment by magnesium tubular structures. The rods are automatically extended to position during telescope deployment sequence.

The four quartz rods are located radially within the magnesium tubes by a series of resilient bushings. Since all known elastomeric materials have a tendency to outgas in space, built-up assemblies of spring-metal leaves have been selected for the bushing material. A damping agent between the bushings is desirable to null dynamic loading rod resonance during launch; metallic or glass wool material has been examined for this application.

In the launch configuration, sections of the spacer rod tubular housings are hinged and folded to lie flat in a position adjacent to the underside of the sun cap (Fig. 3-13) of the telescope. The tubes are held by a trolley of four rollers each and are supported



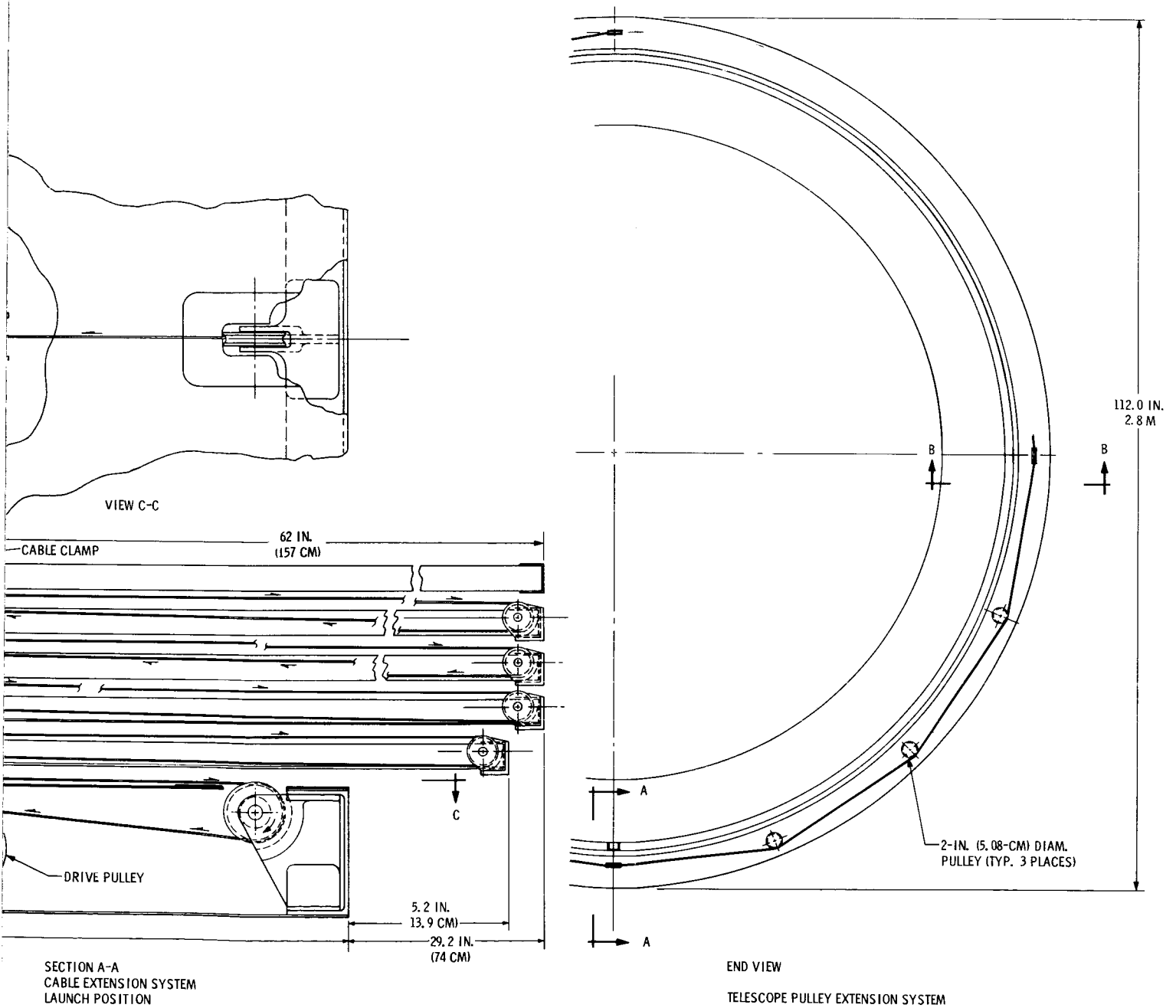
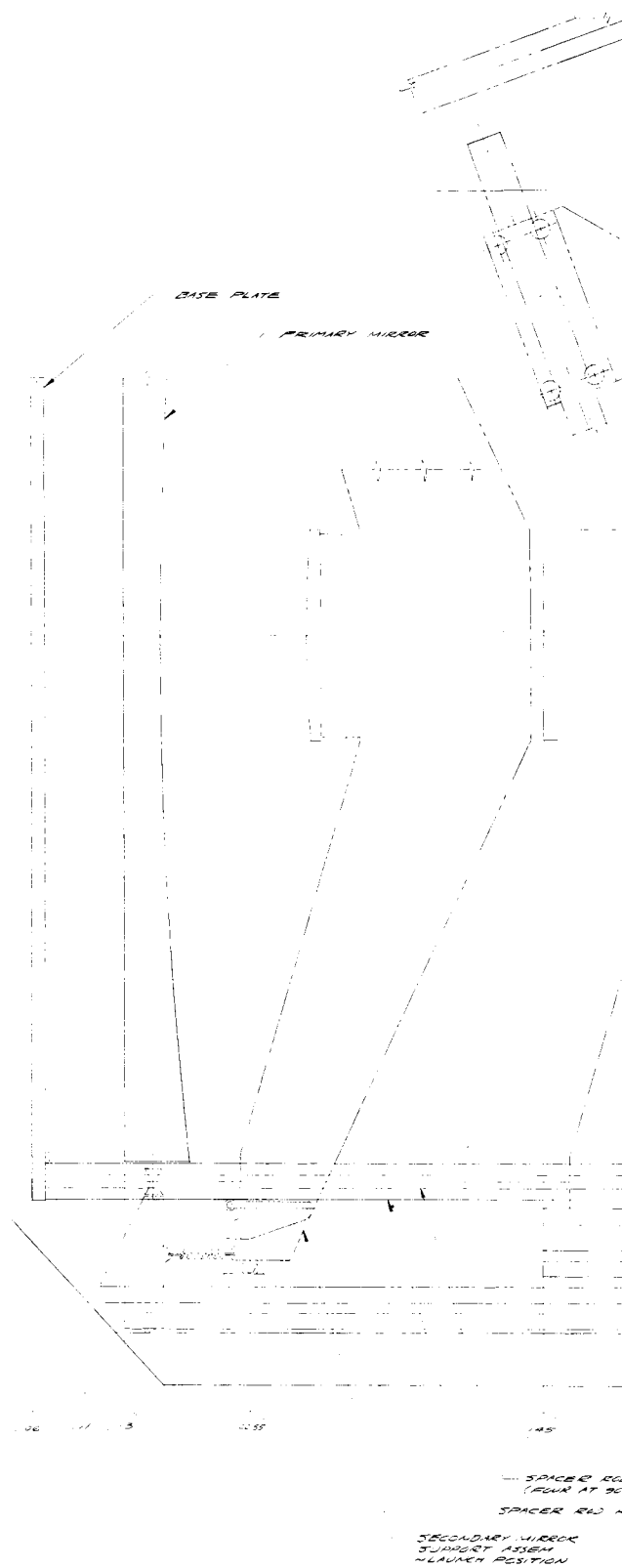
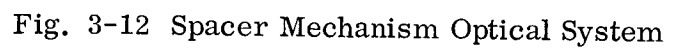


Fig. 3-11 Telescope Pulley Extension System





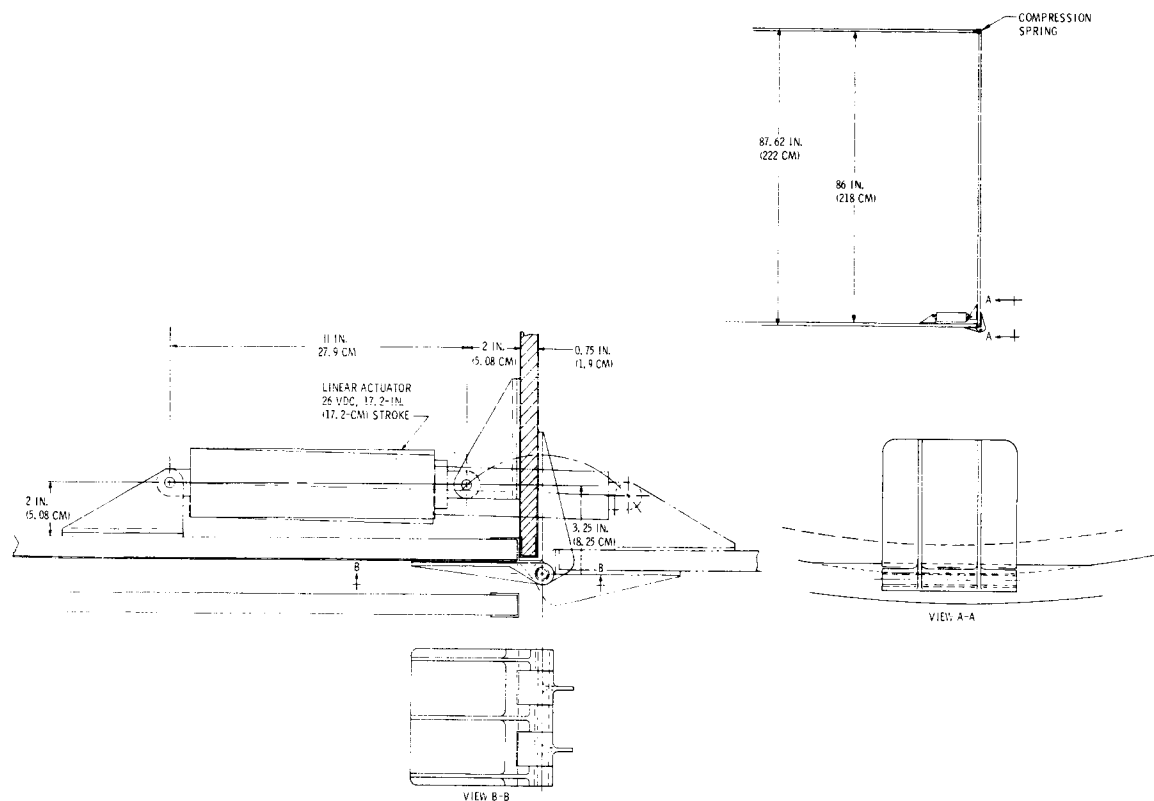


Fig. 3-13 Sun Cap Actuator

by arms pivoted from the figure sensor supports. The spacer rod sections — cut at the hinge station — are held in position radially by the bushings. These are positioned longitudinally by blocks on the sunshade inner surface.

When the telescope is deployed, the trolleys are constrained and rotate around their pivot points on the figure sensor by the relative separation of the telescoping elements. Assisted by springs, the rods rotate around and contact with the figure sensor support arms and rod housings; the integral quartz rods remain straight. Continued extension of the telescope runs the trolley rollers off the ends of the housing tubes. Four spring-loaded hinged pads mounted near the ends of the secondary mirror support arms are held in position by rollers running against the spacers housing tubes. Near the end of the telescope deployment sequence, the rollers run past the housing ends and then the rod ends are located by automatic energizing of four geared motors to rotate locating cam surfaces into position.

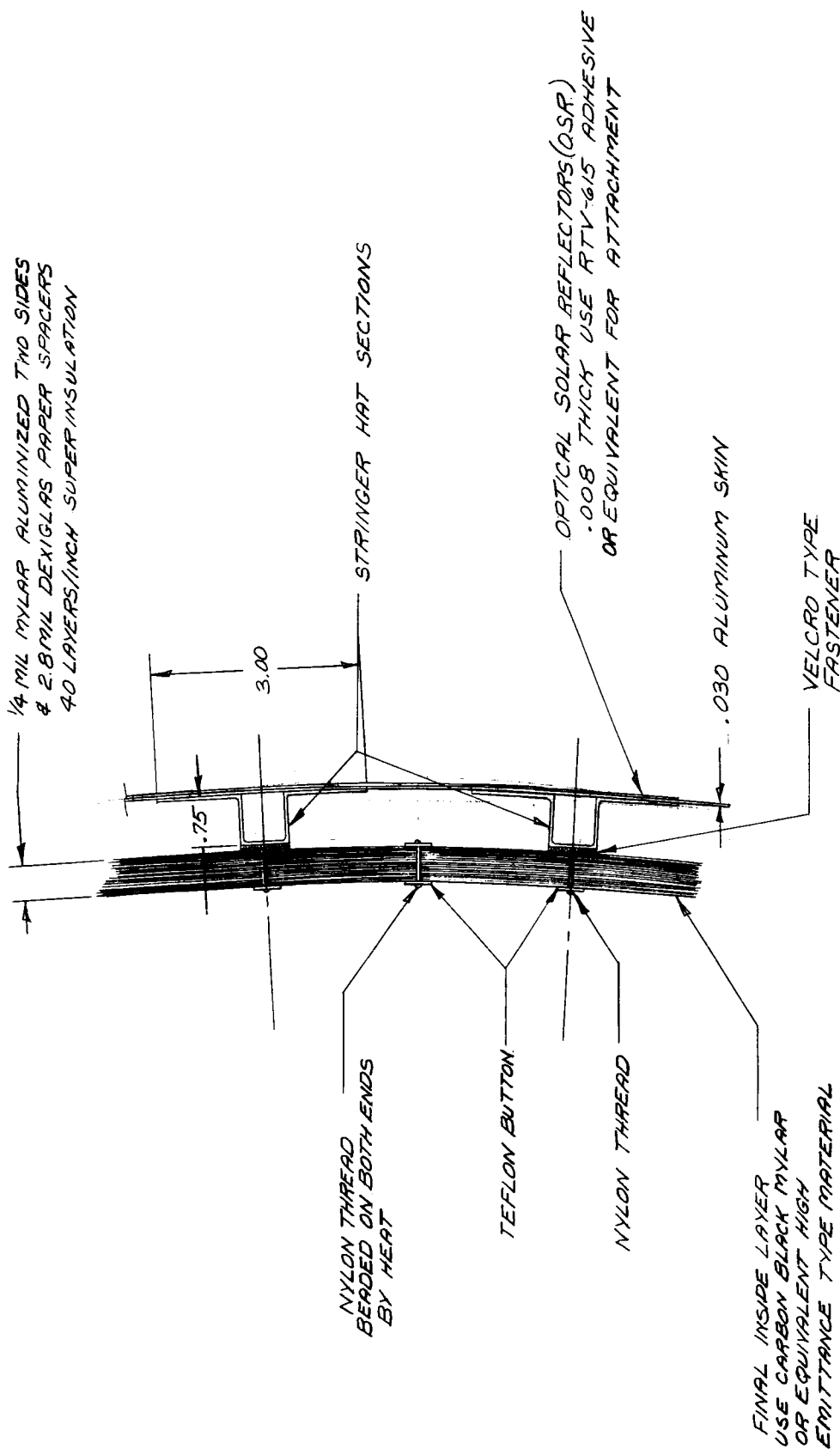
The secondary mirror support assembly, thus located laterally by four equally spaced rollers that bear on the telescope wall, is then pulled aft to load the quartz rods by four tension springs. Following deployment, springs load the secondary mirror support assembly against the spacer rods which, in turn, maintain the proper distances between the primary and secondary mirrors.

3.2.3 Thermal Control System

Passive thermal control techniques are used for the OTES to simplify spacecraft requirements and reduce torque-producing mechanisms that could degrade system operation. A number of exterior surface coatings having low solar absorption were considered. For OTES, surface finish coatings under development by LMSC are of particular interest. For instance, Optical Surface Reflectance (OSR) materials have demonstrated the ability to withstand temperatures from -320°F to over $+800^{\circ}\text{F}$ and are stable in the space radiation environment. OSR materials for the baseline OTES thermal control system are bonded to the exterior of the telescope shell. Multilayer super insulation is attached to the inner surfaces of the telescope (Fig. 3-14). The thermal control system also uses several small secondary insulators within the various components.

Basic materials selected for the OSR are vapor-deposited silver and fused silica substrates. The silver is vapor-deposited on the fused silica and overcoated with vapor-deposited inconel to inhibit corrosion and provide protection of the reflector reverse side from damage. Fused silica was chosen as a transparent cover material and has extremely high stability in the radiation environment. The fused silica substrates are of 0.008-in. thickness and are 1-in. by 1-in. square. Silver was selected for its low solar absorptance and stability.

OSR-equivalent type materials (which are less expensive than OSR) also are under consideration for OTES. LMSC is presently evaluating the use of silicon oxide deposited over flexible thin substrates with adhesive backing. Optical properties with solar absorptance values of less than 0.10 and infrared emittance values of 0.60



TYPICAL CROSS SECTION
OF TELESCOPE STRUCTURE

Fig. 3-14 Installation of OSR and Superinsulation

or higher can be obtained. The stability of the materials in space however, must still be evaluated by extensive testing. If acceptable, this system would be produced in 4- by 10-ft sheets (maximum) and used for large spacecraft surface applications.

Multilayer insulation is required on the spar interior to control thermal gradients across the mirror segments. Investigations show that a 0.25-in. insulation thickness is required. Heat leak through penetration from the telescope support structures must also be minimized in the thermal system design.

3.2.4 OTES Electrical Power Subsystem

The baseline OTES electrical power subsystem takes maximum advantage of the hybrid AAP ATM power generating hardware, i. e., solar arrays, rechargeable batteries, and fuel cells. Electrical power requirements (300 w) for OTES have been estimated to be much lower than for the AAP telescope and experiments for solar observation; hence, the electrical system needs modification only to the extent required for compatibility with stellar and earth-pointing target applications and for improved communications.

In the primary cluster configuration, it is proposed that the solar-array system used for primary power be mounted solely on the S-IVB orbital workshop structures. LMSC, under separate contract, has designed a specific AAP cluster solar-array configuration to provide maximum performance for a solar-oriented cluster that can be modified for OTES power requirements. Since tracking solar arrays would induce disturbances in the telescope experiment, only intermittent (possibly manual) repositioning of the solar array has been considered as compatible with the 1973 launch date state-of-the-art. Additional analyses of movable array designs to provide maximum power for any cluster orientation at minimum size and weight penalties have been proposed for subsequent study phases. Also, an optimum balance between rack-mounted batteries and improved fuel-cell reactant tanks for cluster operations must be determined in the Phase B study program.

For earth pointing experiments, cluster orientation requirements dictate that primary power must be taken from the battery or fuel-cell sources. For independent operating modes, it is proposed to unfurl a separate solar-array system to maintain power continuity for extended periods beyond the capabilities of the battery/fuel cell components. As planned during the study, the array would be mounted on the ATM rack; however, should the CSM be required for independent mode experiments, the exact placement of the array would be reconsidered. In either independent mode (LM or LM & CSM) configuration, the supplemental solar-array system would be needed for extended duration experiments. Also, since the array would be incompatible with the cluster configuration and since several individual independent modes could be planned for the OTES mission, it would appear desirable to design the array system for extension or retraction to stored position. These considerations would be investigated in more detail in subsequent study phases and to some degree are dependent on clearer definition of the AAP cluster configuration. Subsection 3.3.4 presents the power subsystem analysis.

3.2.5 Attitude-Control System

The attitude control system proposed for OTES is basically identical to that for the AAP and is used in conjunction with the Apollo Telescope Mount. Since the ATM, as conceived for use in the AAP, is oriented toward solar observation experiments, the attitude references are derived from star tracking (canopus) and two sun sensors — one wide-angle acquisition sensor and one fine sun sensor — for vernier control. Stellar experiments proposed for OTES dictate a change in the celestial reference sensors. A possible and likely configuration of the OTES attitude-control system, as modified from the AAP configuration, is shown in Fig. 3-15. This scheme proposes use of one gimballed star tracker as an absolute roll reference and two additional star trackers (in place of the sun sensors) for coarse and fine pointing references. The star tracker is attached to the ATM rack through gimbals to provide reference for rack attitude in roll. The coarse star sensor is attached directly to the rack and is used to reference attitude in pitch and yaw. The fine star tracker

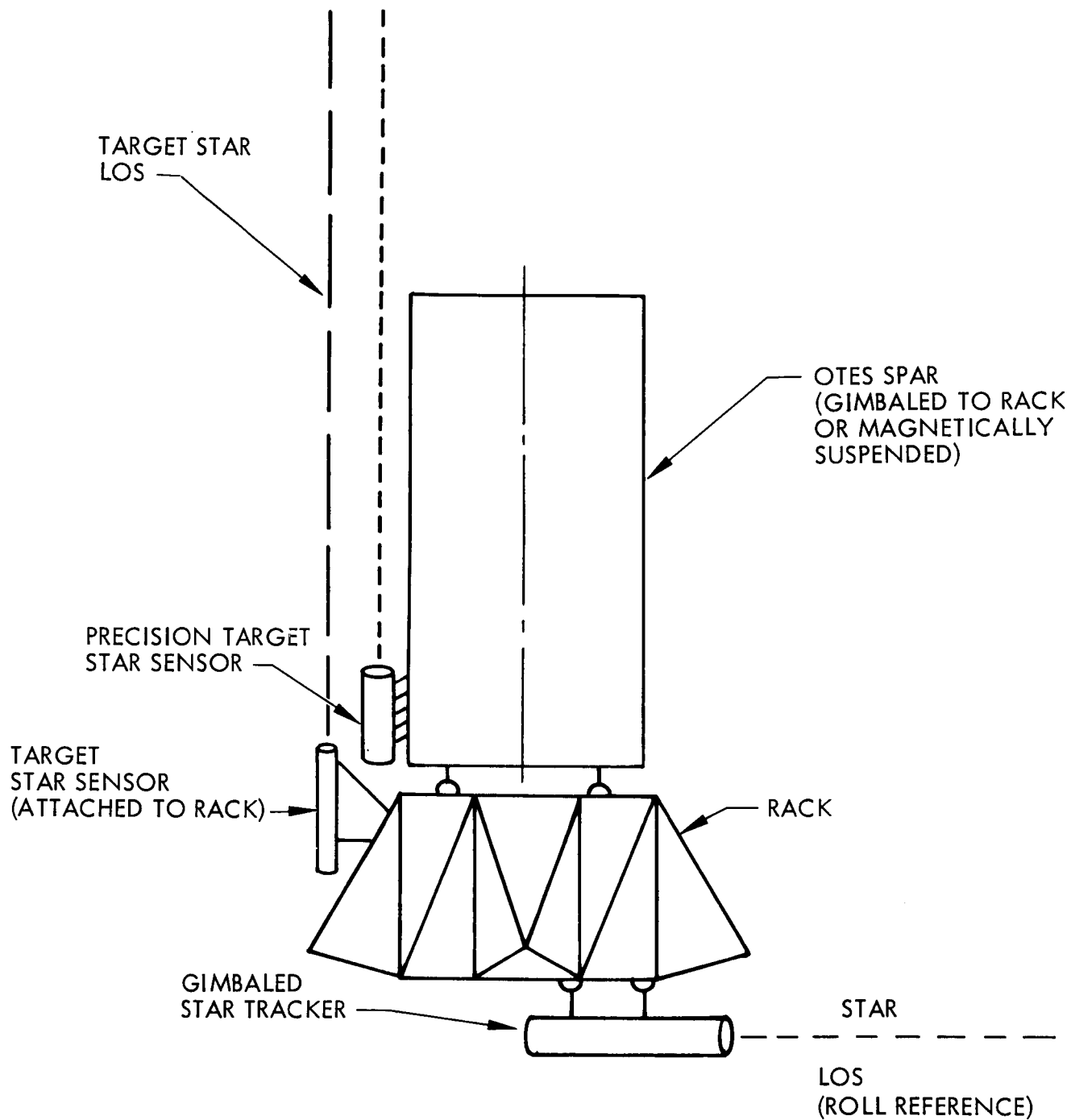


Fig. 3-15 Stellar Reference Sensor

with a narrow field of view is mounted on the OTES telescope (spar). Vernier gimbals between the rack and spar (or magnetic torques for the alternate control experiment) are caged at the center of their travel until the carrier's control system brings the target star into the fine sensor field of view. At this time the gimbals are uncaged and fine pointing is accomplished by the vernier gimbals or magnetic torquers, as applicable.

Figure 3-16 shows the complete OTES/ATM control system functional block diagram. The system comprises the carrier vehicle with a gimballed spar containing the OTES optical equipment. The carrier could be the LM ascent vehicle for independent mode operations (or, as noted, combined LM and CSM vehicles) or the full complement of space vehicles comprising the Apollo cluster. The carrier is actuated for attitude control by three Control Moment Gyros (CMG's) mounted on the ATM rack attached to the LM. Two of the gimbal axes (pitch and yaw) are hinged on flexure pivots; the third axis (roll) is not controlled except that it has the capability of movement from one fixed orientation to another. Magnetic suspension torquers would be used in substitution for the gimbal torquer motors during the "Experiment 26" suspension system comparison experiment. In this case the telescope spar would be moved forward out of the gimbal rings assembly into the magnetic suspension system for magnetic field reaction pointing experiments. Figure 3-16 also shows all three torquing systems and their appropriate controls. The various modes of operation for the three torquing systems will be described in subsequent paragraphs, assuming that an initial arbitrary attitude in space has been attained following orbital injection.

At first the spar will be rigidly locked to a neutral position in the rack by means of a mechanical gaging device. Mechanical gaging will be maintained until the rack and carrier are oriented to the desired attitude, i. e., pointing roughly at the target star for observation while maintaining a roll reference using a second reference star, e. g., Canopus. The basic (or inner loop) reference for carrier attitude is provided by three body-mounted, single-degree-of-freedom, rate integrating gyros. Each gyro provides an electrical feedback output from its signal generator to its torquing amplifier to enable the "caging for target hold" mode.

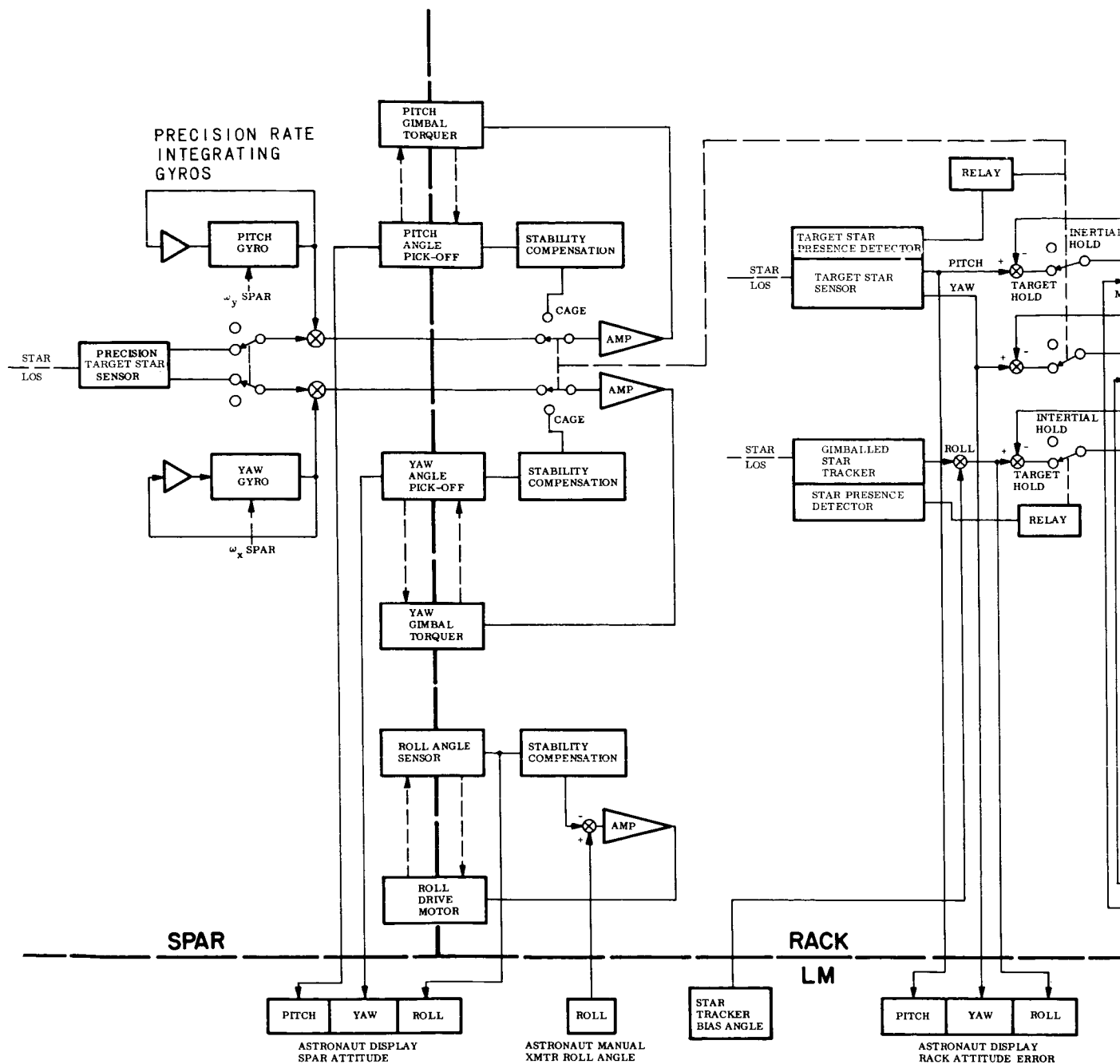
Gross orientations are performed using manual signal transmitters to torque the uncaged gyros; signals from the gyros are applied through compensation and deadband circuits to the LM reaction jet valves and nozzles. The same gyro signals are processed to generate the necessary control moment gyro control signals. Since the CMG control signals are applied directly to the CMG's, whereas the reaction jet signals are passed through a deadband circuit before application to the reaction jet valve drive amplifiers, the CMG's provide a vernier moment generation system within the deadband of the reaction system. When the CMG's are not capable of compensating for a particular disturbance, an overload condition occurs driving the gyros to their gimbal limits or to an orientation where the spin axes are parallel. When this condition occurs, the reaction control system (RCS) is activated. However, the RCS is always in operation when a given error signal is outside the RCS deadband.

Initial acquisition consists of locking the spar to the rack and manually driving the vehicle roll axis to rough alignment with the target star. When manual control brings the vehicle roll axis sufficiently near the star line, the "presence detector" in the star sensor produces a signal for actuation of the Manual-Automatic (MAN-AUTO) relay to the automatic mode. Signals proportional to the angular deviation for the star line (developed in the star sensor) are then applied through the relay to torque the uncaged gyros. This slaves the pitch and yaw axes of the ATM carrier to the star line.

During periods of occultation of the star, the absence of a signal from the "presence detector" releases the relay and the attitude reference reverts to "gyros alone." The gyros will hold the space vehicle so that the star, when next contacted, is within the field of view of the acquisition star sensor for automatic acquisition each time the star rises.

3.2.6 Communications Subsystem

Phase I-A OTES study recommendations for use of the Apollo unified S-band telecommunications system planned for use with the Command Service Module and Lunar



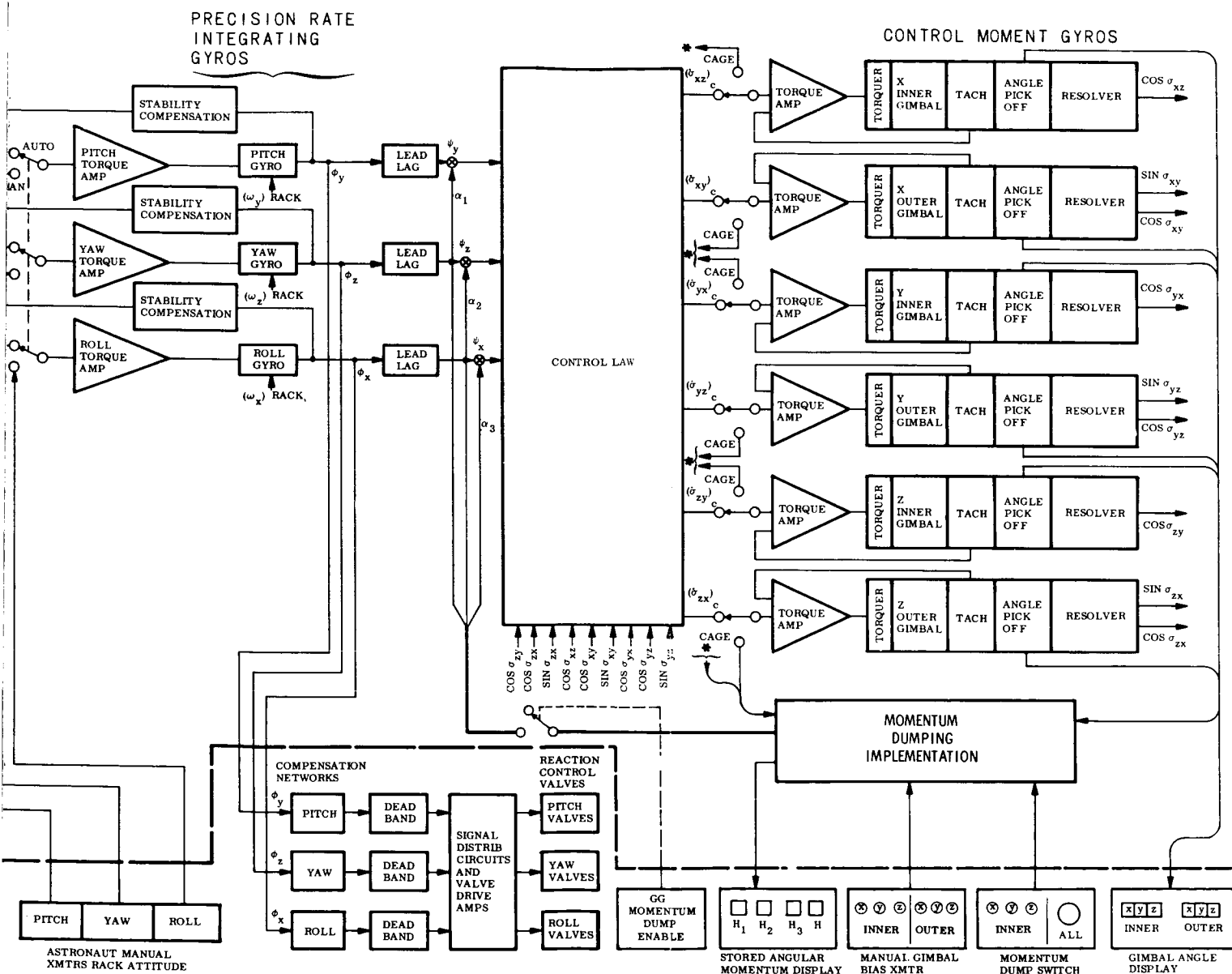


Fig. 3-16 OTES/ATM Control System

Module in the AAP have been reaffirmed during the Phase II study program. Since all OTES-related communications system requirements, as presently defined, are compatible with other AAP telescopic experimentation projects and can be adequately served by cluster electronics, new design of communications equipment is unnecessary. The proposed OTES spacecraft system for either cluster or independent modes of operation, especially if CSM participation is required for independent modes, is more than adequate without modification of presently envisioned communications equipment.

Operating Modes. The primary operating mode for the OTES mission is in the Apollo cluster configuration, i. e. , where a number of spacecraft launched separately are docked to a common Multiple Docking Adapter. This makes the Apollo-AAP communications capabilities for OTES mission applications fully available, i.e., S-band receivers and transmitters, UHF transmitters and receivers, VHF transmitters and receivers, a central processing center, the necessary antenna systems, and even mobile video cameras for visual monitoring and transmission of video data to ground stations.

For earth orbit missions of the AAP, the lunar-mission configuration of the LM has been modified to facilitate experiment applications or to remove unnecessary equipment, although essentially all communicative capabilities remain intact (lunar surface communications antennas are omitted). In the secondary, independent mode of operation, vehicle orientation requirements and shadowing effects caused by the necessity of extending supplemental solar panels for electrical power will impede antenna patterns and result in poor communications. It therefore is proposed that the S-band antennas and antenna servo-mechanisms be replaced by two S-band omni-antennas installed on extendable booms (located 180 deg apart) that are mounted on the rack structure. Comparable placement is also suggested for the UHF and VHF antenna systems. Prior to solar panel deployment the two S-band antennas could be operated in a shortened position; after deployment the booms would be extended beyond the solar panel shadowing area for improved radiation. The booms could also be retracted for docking operations with the cluster and then extended again for operation in the cluster configuration communications system. Much of

this development will be accomplished during the early AAP efforts in support of other independent and cluster modes of operations presently planned for AAP. The problem would be typical of a number of experiment applications.

Antenna compatibility problems, as noted above, also exist for OTES cluster configuration modes of operation. Each of the four launch vehicles in the cluster have separate means of transmitting data to the ground stations. However, mission orientation requirements, cluster shadowing and reflections caused by size, shape, and proximity of the four spacecraft modules, docking adapter, and cluster solar array provisions again result in communications problems. In the docked position the CSM operates as the primary communications and tracking source for the cluster and the LM functions as backup and relay for OTES experimental data to the CSM for transmission to ground stations. Some antennas would be mounted and extended from the cluster structures to achieve minimum interference and maximum coverage. Moreover, backup modes and switching capabilities will be established during the AAP development program that will permit implementation of an optimum antenna system approach.

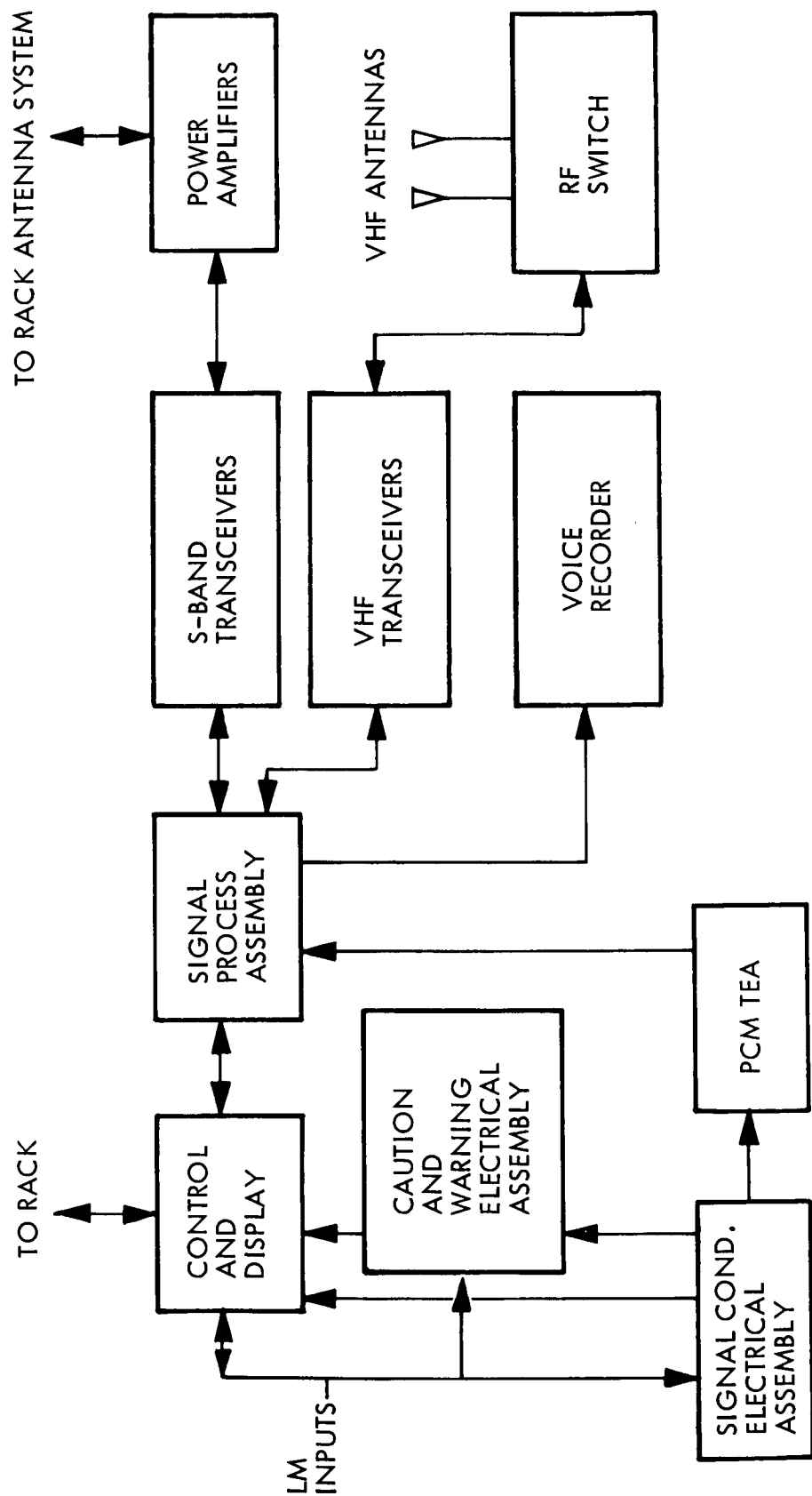
Communications System Equipment. The available communications equipment via the S-band system will include a duplex link for voice, permitting conference lines between two EVA astronauts, a simplex link for voice between LM and CSM operations, and a one-way VHF data link for telemetry data transmission from the LM in the event that S-band communications are lost (these data can also be rebroadcast from the CSM).

The S-band link carries voice from earth to the LM, as well as voice, biomedical, PCM telemetry, and television transmissions from the mobile video camera from the LM to the ground stations. Additionally, the S-band equipment functions as a transponder to receive pseudorandom noise (PRN) signals from earth for retransmission in phase coherence to the ground stations; this, in turn, is used for tracking and ranging.

Several spacecraft module combinations can use these communications links. For example, provisions are made for LM/earth/CSM and LM/earth/EVA conferences. The LM can function as a relay between EVA and earth; the Manned Space Flight Network (MSFN) ground stations can be used as the relay between the LM and CSM or, in an emergency, the LM can relay CSM transmissions to earth. Simultaneous operations of the different links can also be accomplished, e. g. , PCM telemetry to earth via S-band and two-way LM-EVA via VHF links.

LM-Related Communications. Communications provisions of the LM and rack are mutually independent. The LM data management system is used primarily for man-related operations, i. e. , crew communications, control and display, caution and warning, voice, instrumentation data (PCM telemetry), in-flight tracking and ranging, television (EVA), etc. A combination of hardwire, VHF, and S-band is used in these combination links as indicated by Fig. 3-17. Hardwire links are used for voice and video lines between EVA astronauts and the spacecraft. VHF is used for LM/CSM communications and for EVA. The S-band provides for LM/MSFN communications. Vehicle status is provided via transducers located in the various subsystems. These data (voltages or digital) are coupled to the signal conditioning equipment (instrumentation system) which conditions the data for transmittal through the communications system.

OTES-Related Communications. Rack installations are used for communications required to directly support the experimental programs. Electronics required in support of the experiments are S-band, UHF, and VHF power dividers; VHF transmitters, multicoupler, and coax switch; UHF directional coupler, command receiver, and command decoder; and remote analog calibration system (RACS) decoder adapter, and switch selector. These communications system elements are integral to the rack data management system configuration (Fig. 3-18). OTES-peculiar instrumentation (e. g. , critical thermal measurements or alignments) will be integrated with the LM or rack data management systems, depending upon specific use to be made of the experimental data.



*RACK ATTACHED TO LM

Fig. 3-17 Data Management, LM-A Configuration

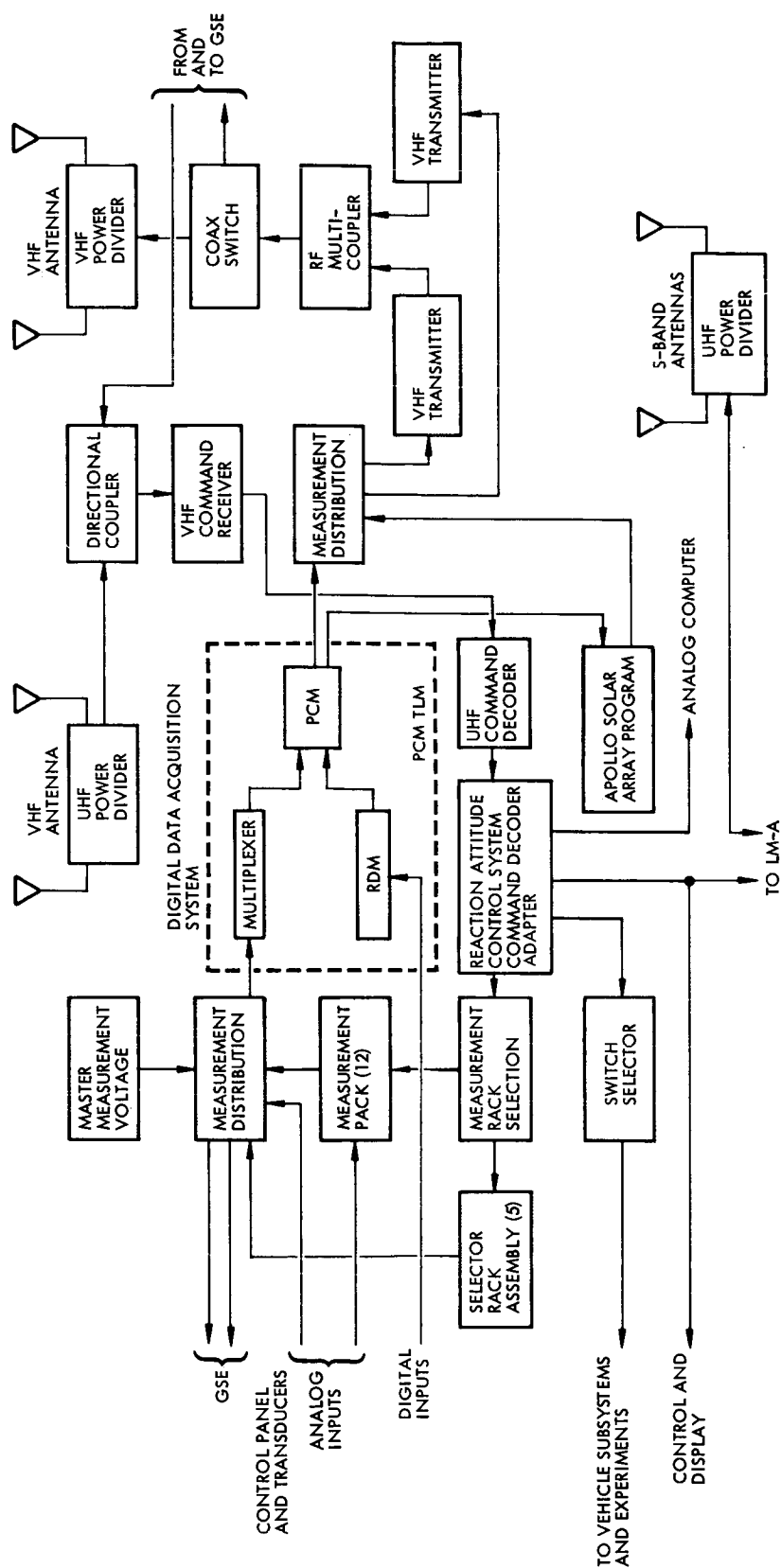


Fig. 3-18 Data Management, Rack (Installation Module) Configuration

Preliminary analyses of the Apollo telecommunications system with respect to OTES requirements indicate that new communications developments are not required. Some antenna relocation will be necessary for both independent and cluster mode operations but will be determined during development of the AAP Cluster A and will be quite similar to requirements for the ATM solar spar mission requirement. Modifications to accommodate the OTES independent mode will comprise minor equipment relocations; specific details of these modifications for OTES will be established after AAP Cluster A design details are more firmly established.

3.2.7 Orbital Mirror Recoating Facility (See Experiment No. 6)

In accordance with Phase II study guidelines a preliminary conceptual analysis of an Orbital Mirror Recoating Facility (OMRF) was conducted to generate a baseline approach for supporting Optical Technology Experiment No. 6. Assumptions, guidelines, and constraints employed in the preliminary analysis are as follows:

- Astronaut safety must not be compromised
- THE OMRF must operate on AAP Cluster X (OTES cluster experiments).
- OMRF must be launched on the Saturn IB flight carrying OTES into orbit.

Principal spacecraft considerations investigated during the tradeoff and analysis include:

- S-IVB orbital workshop utilization
- Location on AAP Cluster X
- Launch accommodations
- Mirror coating materials
- Mirror overcoatings
- Surface preparations methods
- Mirror coating methods
- Uniformity control
- Reflectivity control

OMRF Location. Since the S-IVB orbital workshop provides a large usable volume for use by the astronauts in cluster mission applications, it was first considered a possible location for the OMRF experiments. This location, however, indicated several disadvantages and a separately docked facility was investigated as a possible alternative. Table 3-2 summarizes some of the advantages and disadvantages considered and leading to the selection of a separately docked facility approach. EVA considerations, in turn, dictated the need for docking the OMRF to an MDA docking port at right angles to the OTES rack and spar.

Launch Accommodations. Launch vehicle integration parameters, such as available volume, available launch weight margins, and available means of supporting the OMRF, were investigated to establish the maximum dimensional limits available for the OMRF. Table 3-3 summarizes the OMRF preliminary weight estimate. Thus, the resulting configuration approach facilitates integration into the launch envelope of the S-IVB/LM adapter. Figure 3-19 shows the OMRF and OTES spacecraft in the aerodynamic shroud.

OMRF Concept Description. Conceptually, the OMRF (Fig. 3-20) consists of two compartments: a pressurized compartment and a space vacuum section. The pressurized compartment is essentially a short cylindrical section with Apollo docking ports (one probe and one drogue) with pressure plates on each end. This section contains viewports for observation of mirror segments being processed in the vacuum chamber and an electronic control panel for controlling process operations. The unpressurized (vacuum) compartment is a rectangular housing welded to the exterior of the pressurized section. The vacuum section contains the equipment for stripping and recoating (or possibly coating only) a mirror segment and provides an environmental protection against direct sunlight impinging on the segment. The actual mirror processing operations are envisioned to be automated and remotely controlled by the astronaut.

Mirror segments (two) are transferred from the telescope to the OMRF in a specially designed mirror transfer case. A single electrical plug provides the interface

Table 3-2
RECOATING-FACILITY LOCATION MATRIX

Facility Inside S-IVB		Separately Docked Facility	
Advantages	Disadvantages	Advantages	Disadvantages
Better inspection of work possible	Mirror segment and all equipment to perform recoating must be transferred into and through the various living modules Particle vent from recoating facility must go through pressure hull Complex assembly of equipment must be made in S-IVB Some part of the pressure hull must be opened to the space vacuum	Hard vacuum easily available All effluents or loose material can be trapped outside of living modules Easy access to EVA astronaut Assembled and tested on ground (T/V chamber) Modular concept allows for almost immediately operable systems No equipment hauled through hatches	Requires that astronaut crawl through OMRF from CSM to LM

Table 3-3
OMRF WEIGHT SUMMARY

Item		Weight (lb)
1	Docking probe	150.0
2	Docking drogue	19.0
3	Pressure hatch	13.0
4	Crawlway	22.0
5	OMRF structural (angles) members, 453 in. ³	50.0
6	Mirror mounting members	57.0
7	Magnetic mounting latches	30.0
8	Photometer (two)	16.0
9	Monochromatic light	15.0
10	Mirror positioning clamps	48.0
11	Lighting	24.0
12	Electrical connectors and cabling	20.0
13	Metal deposition tray (two)	200.0
14	OMRF exterior skin, 1, 293 in. ³	129.0
	Total	793.0

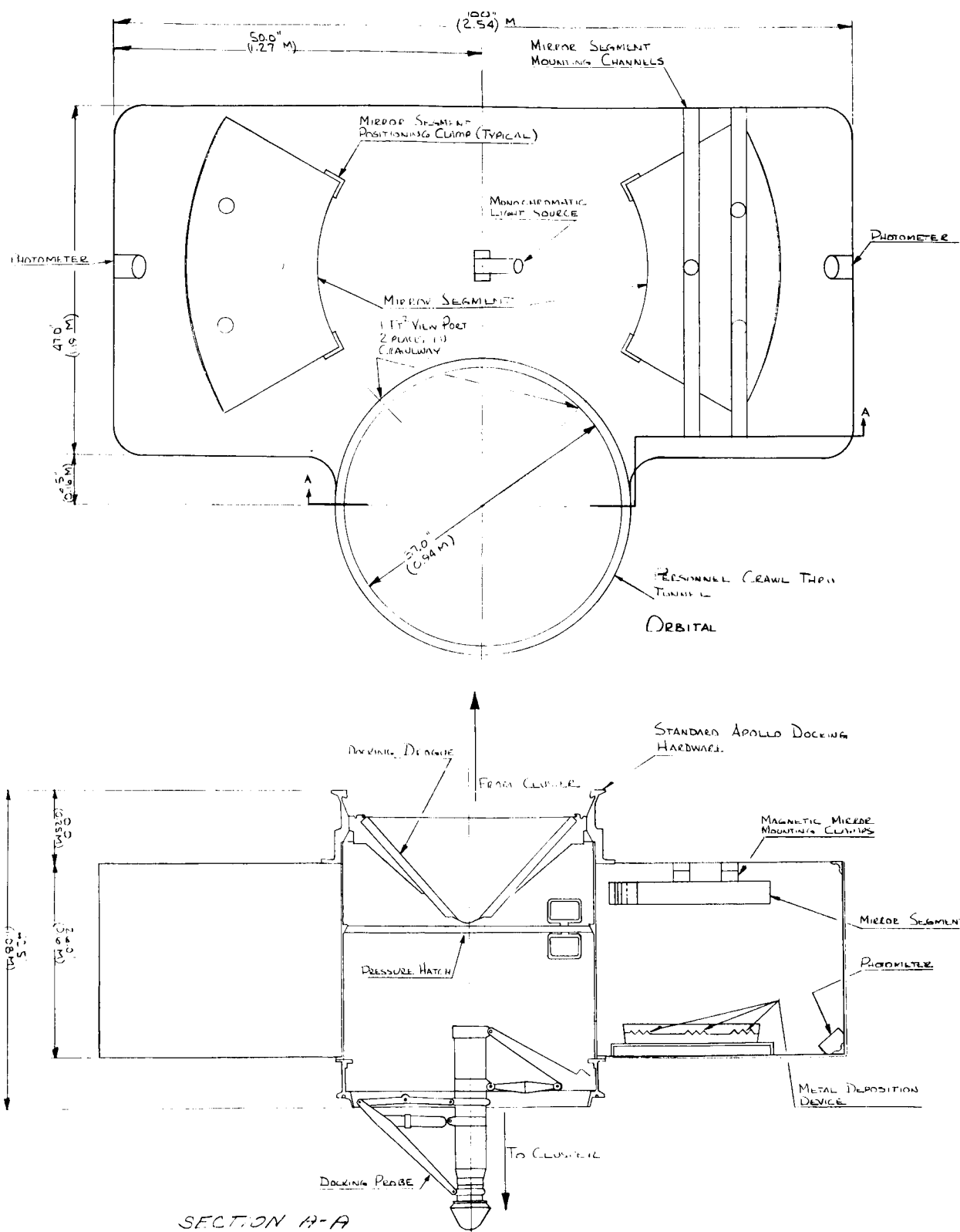


Fig 3-19 Orbital Mirror Recoating Facility

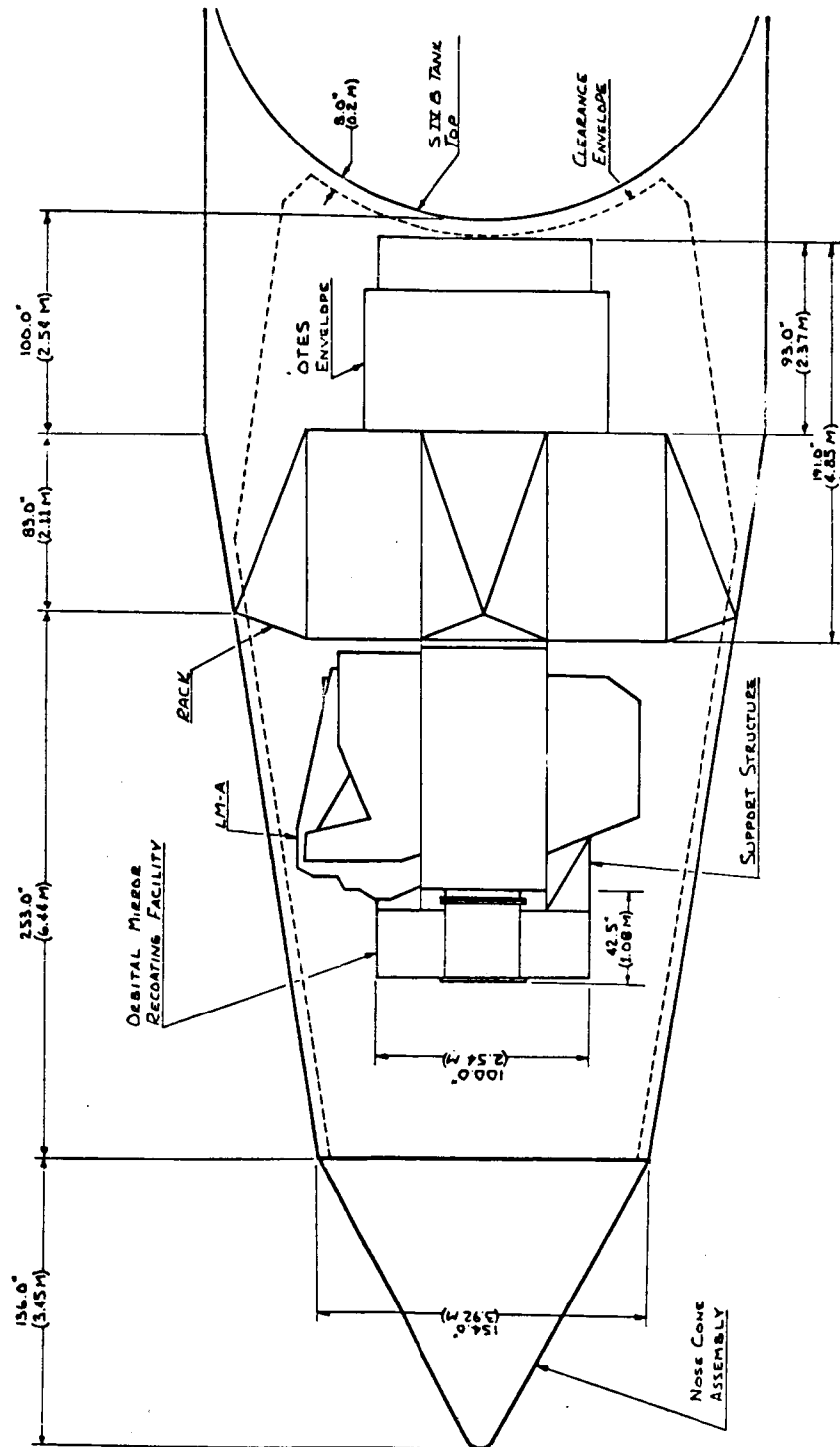


Fig. 3-20 OMRF Launch Configuration

between the OMRF and docking adapter. Current guidelines require that the electrical interface be connected by an EVA astronaut.

During launch the OMRF is docked to and supported by the LM on brackets similar to those used for attaching the ATM rack to the LM. The pressurized compartment serves as a miniature airlock and crawlway when the CSM is docked to the OMRF and LM.

Mirror Coating Material. The Perkin-Elmer team selected aluminum as the thin-film mirror coating and/or recoating material for the OTES application. LMSC fully concurs with this selection. Aluminum is the only thin film that has high reflectance characteristics in the ultraviolet, visible, and infrared regions of the spectrum. Also, the material adheres well to quartz and other substrate materials.

Mirror Overcoatings. Aluminum coatings for optical quality mirrors are normally deposited in an artificial vacuum environment to achieve an extremely high reflectance. Subsequent removal from vacuum and exposure to oxygen in the atmosphere causes the aluminum to combine and drastically degrades the high luster. This is normally inhibited in manufacturing by the application of an "overcoating" of magnesium fluoride or silicon oxide over the aluminum mirror surface. This serves to inhibit continued reflectance degradation over long periods and, therefore, is useful for earth-based applications. In the space applications, however, the overcoating tends to cause two undesired situations:

- The overcoating reduces the mirror reflectance below that obtainable by depositing the thin-film metal coating in the hard vacuum. In space, the mirror will continue to function in the vacuum environment following recoating.
- The original addition of an overcoat increases the difficulty of stripping the degraded mirror finish for refinishing in the OMRF. It is noteworthy that if the OMRF could be used to deposit the original coating on the mirror the entire system performance would be considerably improved.

Surface Preparation Methods. The task of stripping old degraded surfaces and overcoatings from a mirror segment in preparation for recoating is more difficult in orbit than actual recoating. Three basic techniques are applicable to mirror stripping in the orbit environment and were investigated for the OMRF concept:

- Mirror surface bombardment
- Ionic brush of the mirror surface in a partial vacuum
- Liquid (acid) wash of the surface in a closed environment

The acid method was discarded due to the significantly high hazards involved.

Mirror Coating Methods. Metallic films can be produced by mechanical, chemical, or physical methods. For purposes of an optical system, the mechanical method (e. g. , rolling or beating thin films or spraying) offers no suitable method for controlling thickness and reflectivity. The three candidate methods considered are chemical reduction, cathodic sputtering, and thermal evaporation. Table 3-4 is a matrix of the advantages and disadvantages of each of the three candidate techniques leading to selection of the evaporation method as a baseline conceptual design selection.

Reflectivity Control. A fully automatic reflectivity control system using a monochromatic light source and photomultiplier microphotometer pickup is presently under consideration for the OMRF concept. This device, although automatic, would permit astronaut monitoring at the central control station.

Uniformity Control. Factors affecting the reflectance of the mirror surface film are:

- Conditions under which the films are produced
- Rate of deposition
- Pressure during evaporation
- Thickness of the film
- Substrate temperature
- Vapor-incidence angle
- Purity of the material deposited

Table 3-4

RECOATING-DEPOSITION MATRIX

Item	Parameter	Chemical Reduction	Cathodic Sputtering	Thermal Evaporation
1	Best thickness control	3	2	1
2	Best reflectivity control	3	2	1
3	Least hazard to other cluster experiments	3	1	2
4	Easiest to illuminate	2	2	1
5	Least amount of manual operations required	3	2	1
6	Least power required for operation	1	3	2
7	Best astronaut access to work area	2	2	1
8	Least amount of recoating consumables required	3	2	1
9	Least amount of total EVA time required	3	2	1
10	Best uniformity of application	3	2	1
11	Requires least active temperature and pressure control systems	3	2	1
12	Best for applying different types of coatings to mirror	3	1	2
13	Estimated least manhours expended per segment	3	2	1
14	Estimated least astronaut hazards	3	1	2

Metallic films produced in the space vacuum require a high rate of deposition to attain maximum reflectance. Films should be deposited in about 1 sec and the basic material should be of the highest quality and purity (more than 99.9 percent pure). The ultimate film thickness should not exceed approximately 1,500 Å and the deposition should be accomplished with a vapor incidence angle not greater than 30 deg.

Overcoating. The reflectance of a mirror is highest upon completion of the coating process. When the surface comes in contact with oxygen the reflectance drops drastically. To preclude this, mirror segments are coated with a protective layer which slightly degrades original performance but maintains a higher reflectance for the mirror. In the space environment, this type of degradation will not be experienced; hence, the overcoating process can be eliminated from the OMRF system design. Conversely, the protective segment coatings on the mirror in ground preparation will affect the design of a suitable mirror surface stripping system.

Additional Study. The conceptual analysis of OMRF requirements has been only a cursory examination of basic parametric requirements and identification of candidate system approaches for more detailed study in subsequent phases of this study. A more detailed examination of the astronaut participation and OTES system conceptual approach and selection of appropriate resurfacing techniques will be required before refinement of a suitable OMRF conceptual approach.

3.2.8 Preliminary Weight Performance Estimate

Preliminary estimates of the OTES system weights (Table 3-5) indicate that the conceptual design is well within the 29,800-lb payload launch capabilities of the S-IB booster vehicle for the 240-nm circular rendezvous orbit. The gross system weight of 29,135 lb would leave a performance margin of 665 lb for experimental growth potential. Estimated weights are purposely conservative and should compensate for traditional weight increases associated with spacecraft development.

Table 3-5

OTES 2-M TELESCOPE PERFORMANCE AND WEIGHT SUMMARY

Item	Weight (lb)
Nose Cap	1,067
SLA	3,950
Additional IU Panels	1,100
Optical Mirror Recoating Facility	793
Launch Vehicle Modifications	650
LM Ascent Stage	4,282
LM Ascent Stage, Apollo, Dry	4,062
Ascent Propulsion System	- 516
Steerable S-Band Antenna	- 24
H ₂ O Boilers	- 20
Scientific Equipment and Miscellaneous	- 72
Stabilization and Control I/F	70
RDB Radar Transponder	20
GOX Tank Installation	60
Additional LiOH	23
Additional Hygiene	2
Water	50
Oxygen	49
Water/Glycol	37
RCS Helium and Propellant	541
Rack	11,487
Structure, Including Gimbal and Supports	2,938
CMG System	1,549
Battery System	950
Antenna Panels	400
Communications, Telemetry, Electronics	1,000
Solar Array, MSFC	4,650
Telescope	5,506
Figure Sensor, Secondary Mirror Structure, and Insulation	1,162
Mirror Base Plate	503
Mirror (2-m)	580
Electro-Optical Experiments	821
Electro-Optical Equipment Bay	200
CG Platform and Support Equipment	2,240
Film and Experiment Supplies	200
Booster Yaw Steering Allowance	100
Gross Weight	29,135
Capability (S-IB, 240-nm circular)	29,800
Margin	+ 665

3.3 SUPPORTING ANALYSES

Supporting analyses for the OTES spacecraft refinement studies during Phase II included preliminary evaluation of the thermal, attitude-control system, power system, dynamic/stress environments, astronaut design requirements, and reliability considerations relative to the preliminary and preferred OTES spacecraft system conceptual designs. Guidelines and constraints established by NASA or LMSC contractual studies for the AAP hardware designs, so far as presently defined, were used as primary parameters for definition of OTES performance and capabilities. Mathematical models of the thermal environment, ATM gimbal control system, mission geometry, and dynamic/stress environment were based upon the latest possible data relative to the AAP cluster configuration and, hence, are subject to reevaluation as required by subsequent modification or changes to the cluster system. Other subsystem requirements were evaluated, e.g., electrical and communications, with respect to experiment and operational definitions presently defined for the AAP cluster. For comparative purposes, the planned ATM solar telescope experiments were reviewed to ascertain if significant differences in parametric requirements exist that would tend to make the OTES mission more difficult or constraining than ATM applications as presently defined. As noted, in cases other than those in which features seem to be designed as an integral part of the optical technology experiment system (i.e., fine pointing accuracy and magnetic suspension), OTES requirements are equivalent to or less than those already attributed to the ATM solar-telescope operations.

3.3.1 Thermal Analysis

Phase II thermodynamic analyses were performed on the OTES concepts for an AAP cluster application using an integrated 2-m OTES with the cluster on a low-altitude 250-nm orbit. This orbit provided a temperature environment substantially higher than the 20,000-nm synchronous altitude investigated during Phase I studies, which indicated that primary mirror temperature levels could be maintained at approximately -140° F. During the study the effects of solar albedo and earthshine, and additional effects of cluster heat rejection, radiation, and shielding were included

in the investigation. The OTES thermal design in the low-altitude cluster configuration was examined in detail to determine the lowest temperature levels that could be maintained on the primary mirror segments and appropriate techniques for minimizing thermal gradients using passive thermal control-system designs. The desired temperature level of the primary mirror segments was established at -112°F (-80°C).

Missions Parameters. The space environment for the AAP/OTES cluster-configuration thermal analysis is the same as that for AAP Cluster A. Features of this environment are as follows:

- 250-nm circular orbit
- 28.3-deg orbit inclination
- 0 and ± 52 deg solar incidence angles (β), i.e., acute angle between the orbit plane and the earth-sun line represent intrinsic high- and low-temperature conditions
- Mission phases with no incident solar energy space-oriented into the telescope interior

Two cases were selected for study that represent the extreme high- and low-temperature conditions for AAP/OTES cluster configuration operations in the assumed space environment. Hence, thermal conditions for all cases would be bracketed by the two extremes, defined as follows:

- The telescope with AAP cluster is space-oriented in position for maximum sunlight reflectance, ($\beta = 52$ deg), i.e., positioned so that the incident solar input and solar reflection from cluster is maximum (Fig. 3-21)
- The telescope with AAP cluster is space-oriented in position for minimum sunlight reflectance, ($\beta = 0$); i.e., positioned so that the incident solar input to the OTES is shaded by the cluster (Fig. 3-21)

Thermal Models. Thermal models were developed for the AAP cluster, ATM rack, and 2-m OTES to allow mathematical description of surface nodes (Figs. 3-22 through

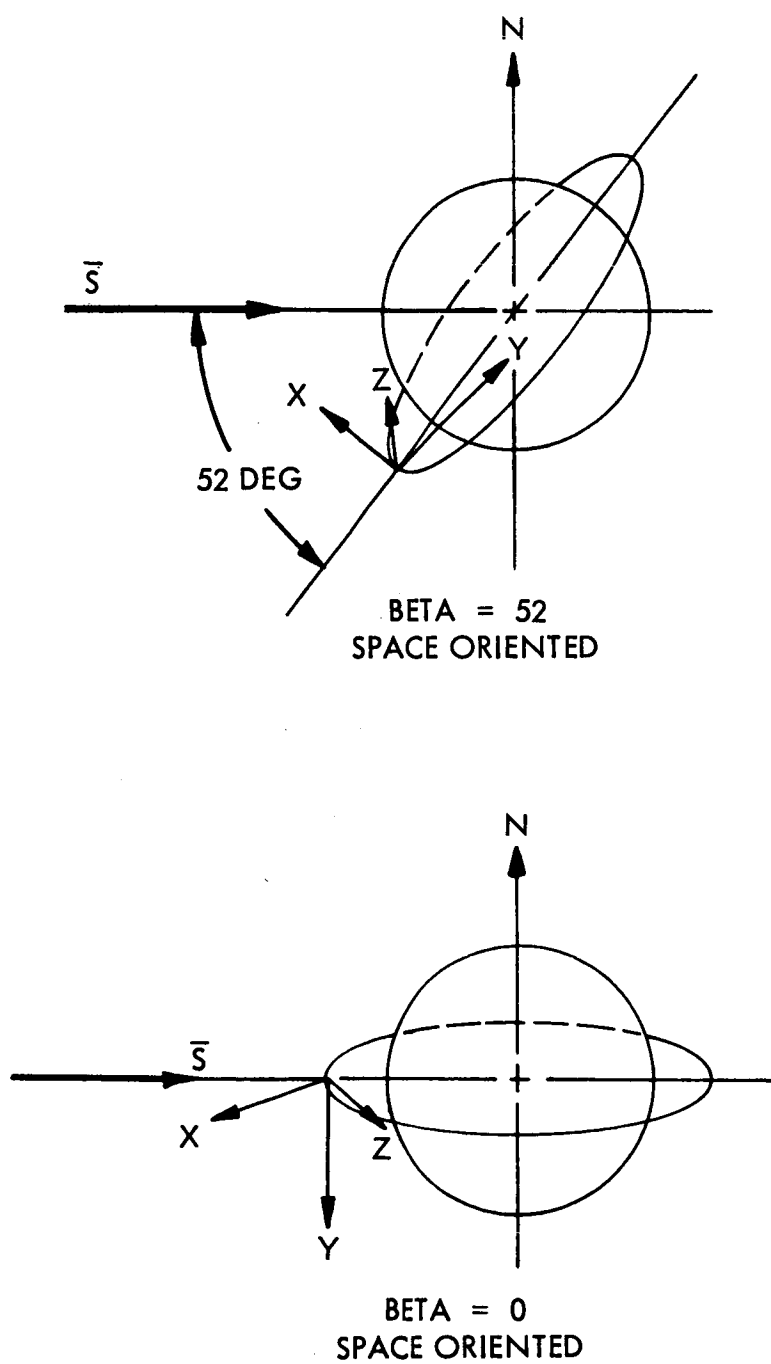


Fig. 3-21 Thermal Sun Angle Geometry

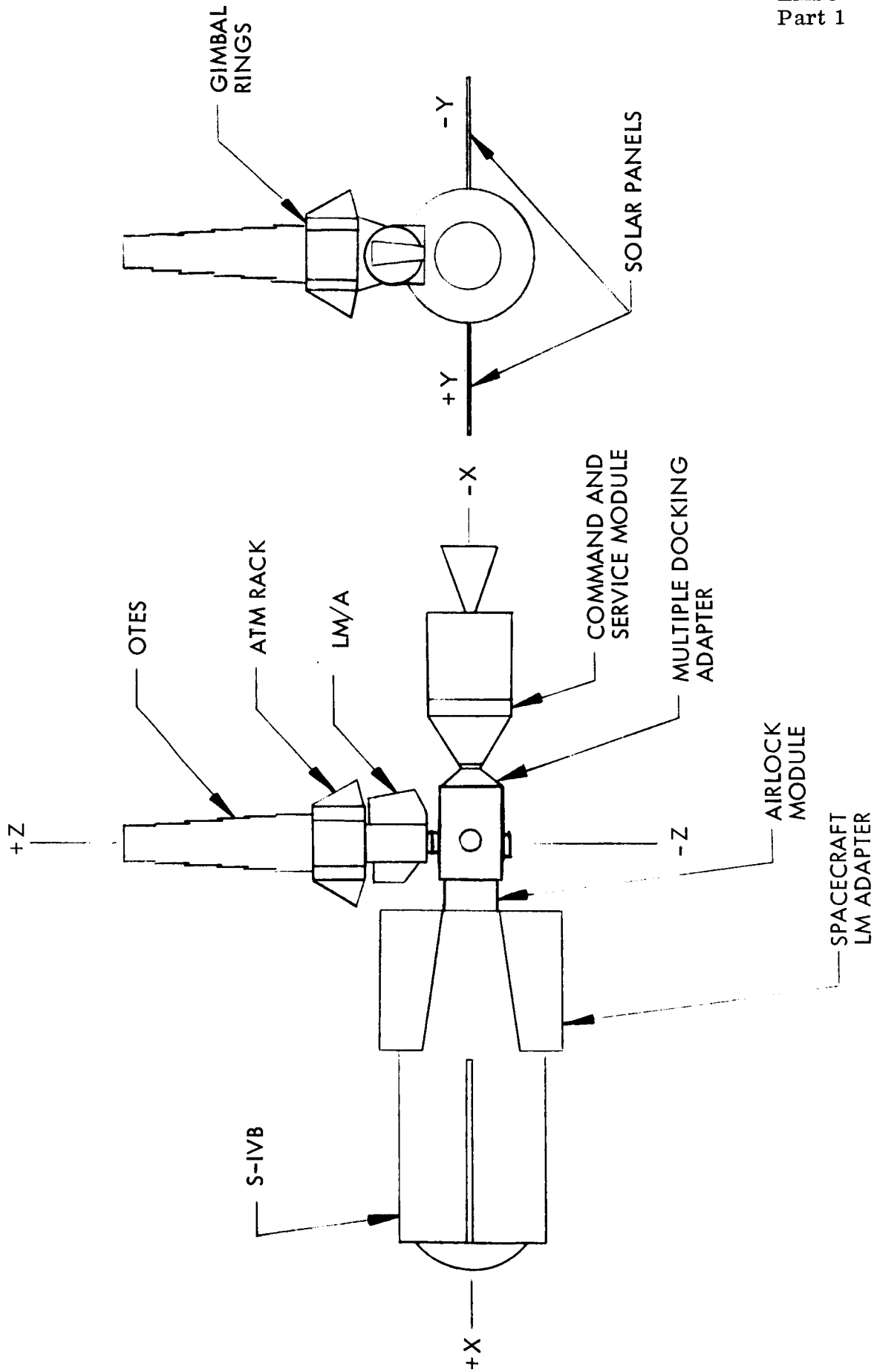


Fig. 3-22 Thermal AAP Cluster Configuration

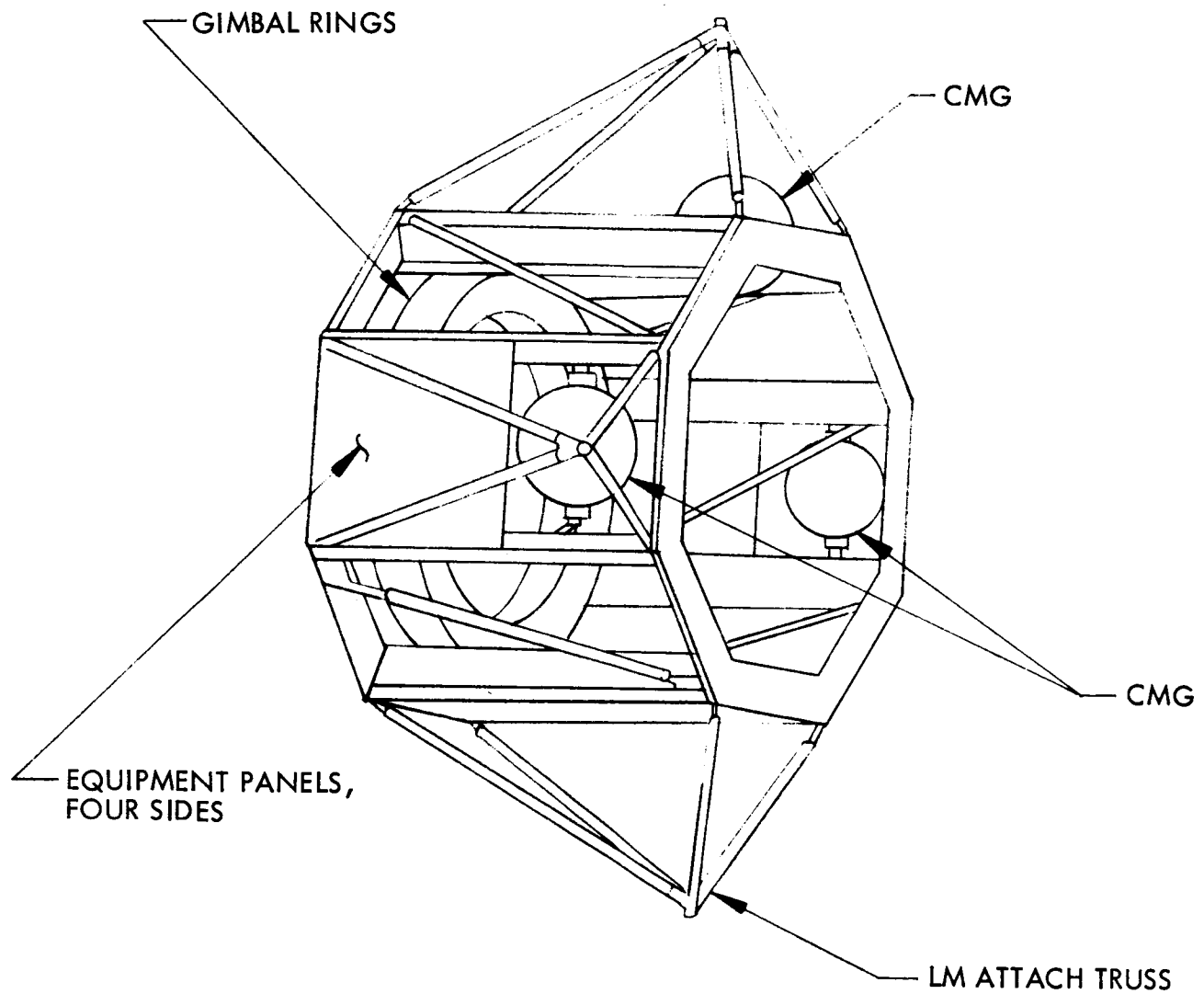


Fig. 3-23 Thermal ATM Rack Configuration

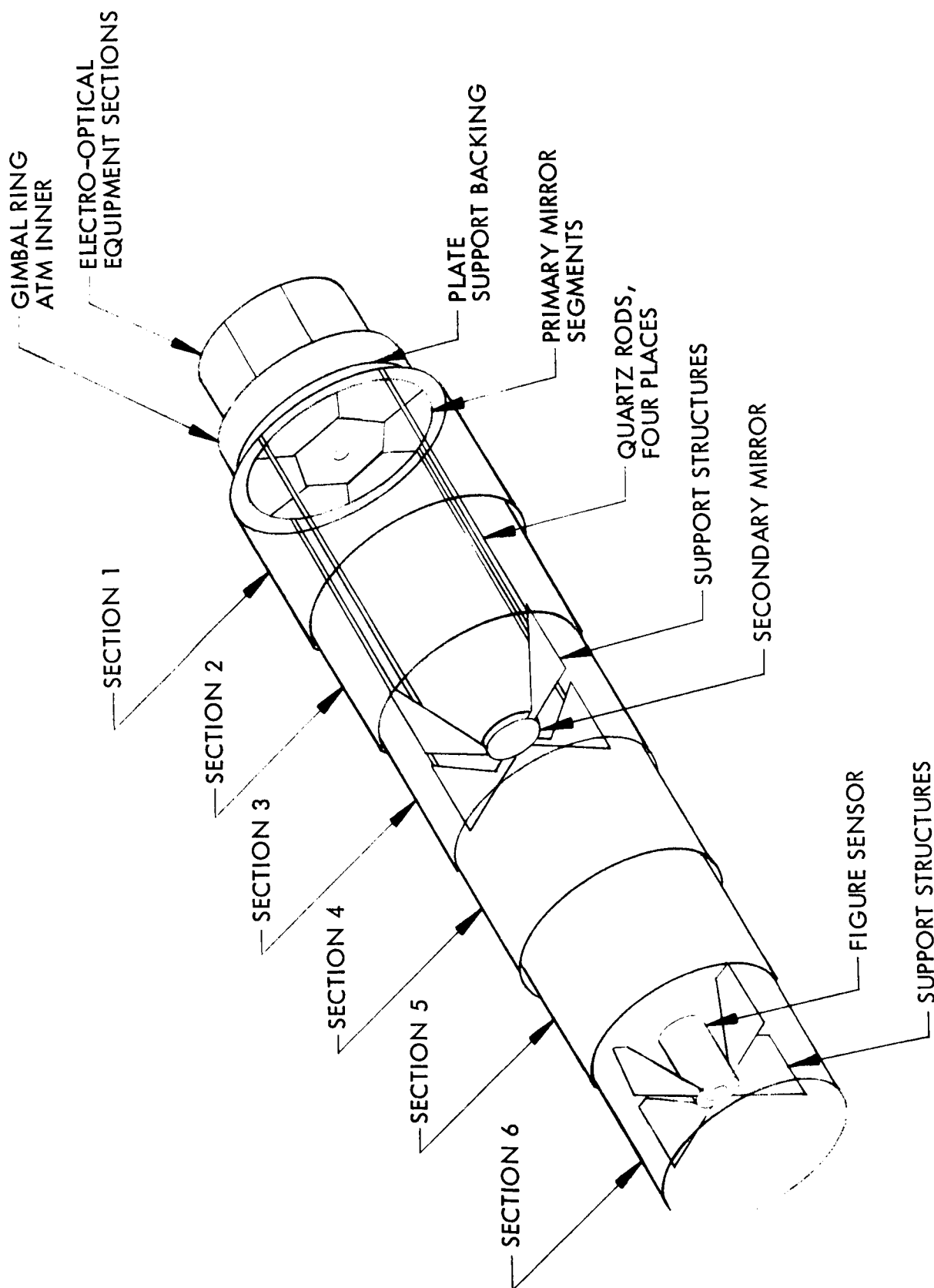


Fig. 3-24 Thermal OTES Spar

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3-24). The cluster thermal model was developed to provide information on the correct thermal interface with OTES. Effects of radiation energy exchanges from the cluster to the OTES were determined and included in the analysis. The effects of reflection and shielding from solar, light, albedo, and earth shine were determined on the basis of orientation of the OTES and cluster during orbit; these effects are included in the analytical results presented in the next paragraph. The ATM rack thermal conduction network was developed to include both the rack structure, surface characteristics, and heat dissipation of the support equipment. Surface finish properties of the rack, equipment, and cluster that were assumed for the analysis are presented in Table 3-6. An interior and exterior radiation network was developed for the OTES telescope system on the basis of the surface finish properties indicated by Table 3-7. The conduction network developed for the system was combined with the radiation network to form part of the mathematical model for the system. However, some approximations were required in setting up the thermal conduction network of the OTES model. These calculations were in the areas of thermal heat paths through the support structures, and were necessary to define thermal conduction paths that will not be determined until preliminary design phases of the OTES. Some of the approximations used for the analysis are given in Table 3-8.

Passive Thermal-Control Techniques. The investigation of passive thermal-control techniques for OTES have confirmed earlier studies during Phase I. Studies indicated that in order to maintain super-cold temperature levels on the telescope, an exterior surface coating must have a low solar absorptance and a solar-absorptance-to-infrared-emittance ratio (α/ϵ) of approximately 0.2.

The evaluation of white paints indicated that the best candidate is a Lockheed developed Thermatrol paint (titanium dioxide pigmented silicone elastomer) having α/ϵ properties of 0.20/0.93, i.e., a ratio of about 0.22. However, degradation tests at LMSC reveal that white paints are damaged extensively by ultraviolet exposure in vacuum. The increase of solar absorptance of the Thermatrol is 0.14; this changes the α/ϵ ratio to 0.34/0.93. Test results have been confirmed by comparison with flight-experience results as indicated by Fig. 3-25. Hence, these paints are not adequately

Table 3-6

CLUSTER SURFACE FINISH PROPERTIES

Cluster Item Description	Solar Absorptance, α	Emittance, ϵ	Heat Dissipation (w)
S-IVB	0.9	0.9	—
Spacecraft LM Adapter	0.9	0.9	—
Airlock Module	0.9	0.9	—
Multiple Docking Adapter	0.9	0.9	—
Command and Service Module	0.9	0.9	—
LM/A	0.2	0.5	—
Solar Panels, Cell Side	0.7	0.9	—
Solar Panels, Backside	0.9	0.9	—
ATM Rack Equipment and Panels	0.2	0.9	1,200
Control Moment Gyros (3)	0.9	0.9	225

Table 3-7

OTES SURFACE FINISH PROPERTIES

OTES Item Description	Solar Absorptance	Emittance, ϵ
Telescope Exterior	0.05	0.80
Telescope Interior Multilayer Insulation	0.9	0.9
Figure Sensor Support Structures	0.9	0.9
Secondary Mirror Support Structure	0.9	0.9
Quartz Rods	—	0.9
Plate Support Backing	0.2	0.9
Gimbal Ring, ATM	0.2	0.9
Electro-optical Equipment Sections	0.3	0.5
Primary Mirror Segments	—	0.8

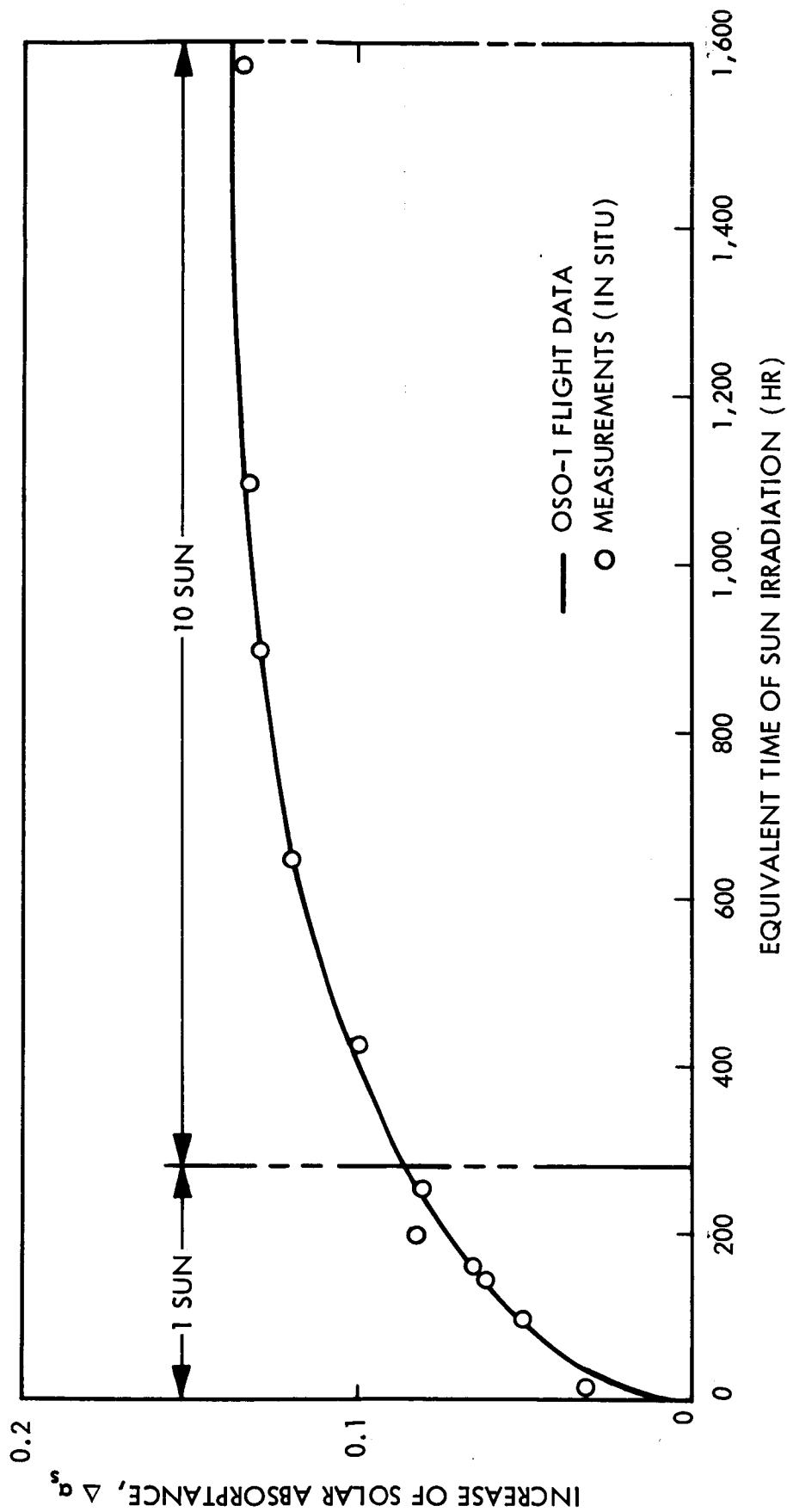


Fig. 3-25 UV Damage for Thermatrol

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Table 3-8

THERMAL RESISTANCE OF TELESCOPE STRUCTURE

Heat-Path Section	Thermal Resistance (° F-sec/Btu)
Primary Mirror Segments to Support Backing Plate	20,000
Secondary Mirror to Telescope Exterior	13,600
Figure Sensor to Telescope Exterior	7,200
Electro-Optical Equipment Section to Support Backing Plate	20,000

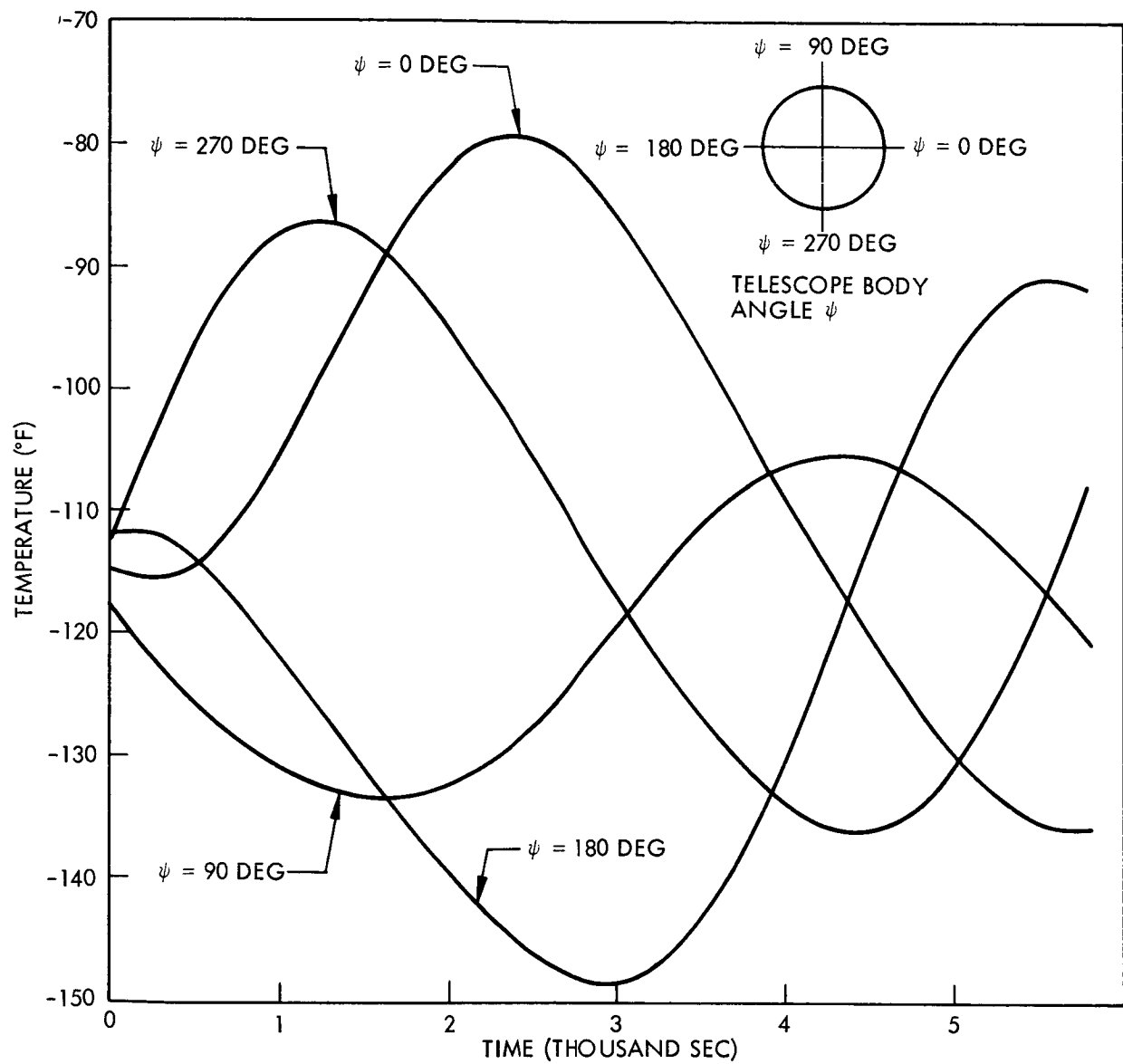
stable surface coatings for super-cold telescopes. Stable surface coatings with α/ϵ ratios less than 0.2 are definitely required for the OTES AAP cluster operations.

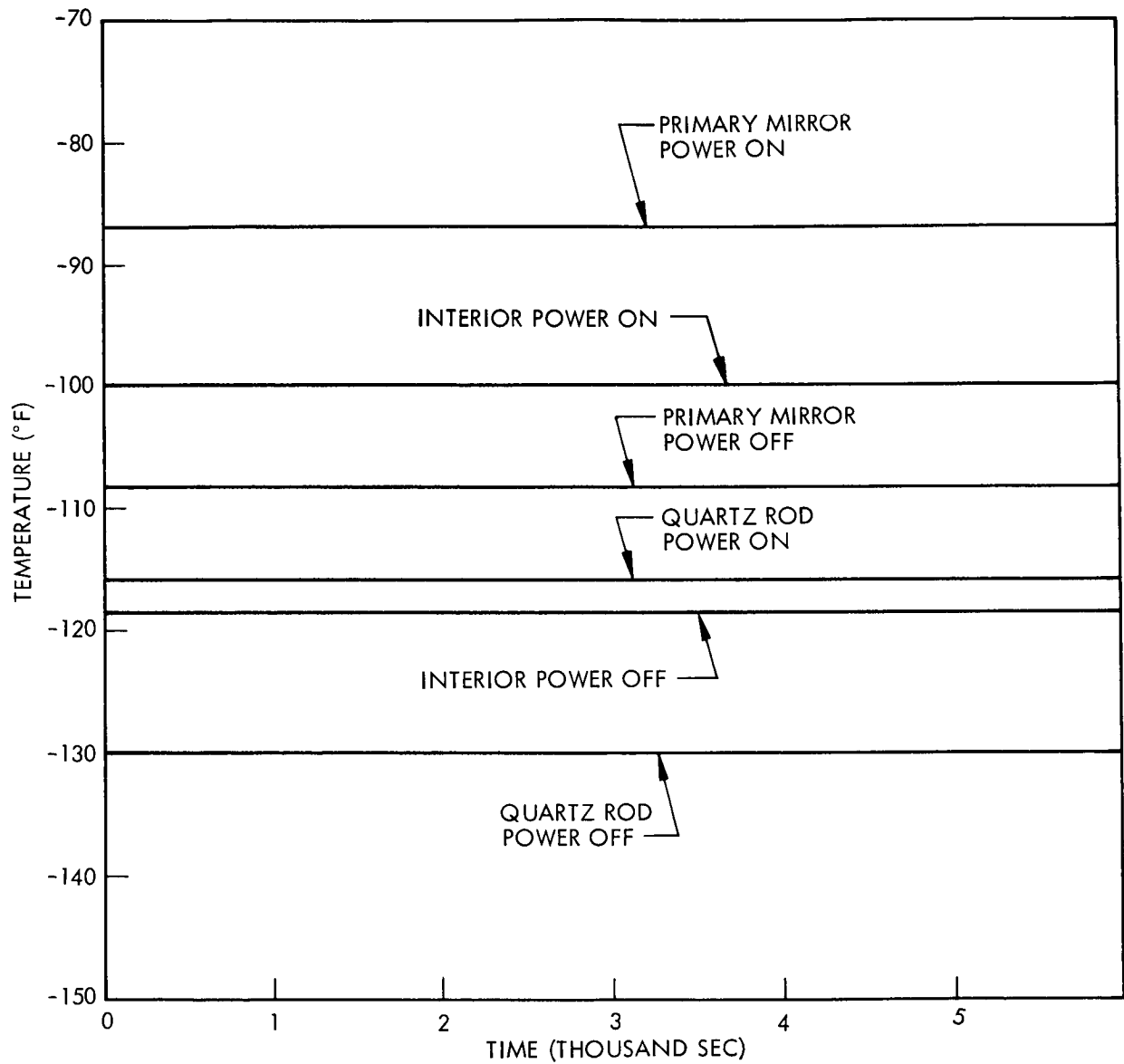
Currently in use at LMSC are surface-finish coatings developed as optical solar reflectors (OSR) that offer a low α/ϵ of 0.05/0.80; this value is three times lower than those obtainable from other surface coatings. An OSR is stable in the space radiation environment, and the basic materials can withstand temperatures from -320 to over 800° F.

As noted in the thermal control subsystem description, the basic materials selected for OSR are vapor-deposited silver and fused-silica substrates. Silver is vapor deposited on the silica, and an overcoat of vapor-deposited Inconel is added to inhibit corrosion. It was also determined that a 0.25-in. coat of multilayer insulation would be required on the telescope interior to control thermal gradients across the mirror segments.

Thermal Analysis Results and Conclusions. Results of the thermal analysis are presented in Figs. 3-26 through 3-31.* The initial temperature levels represent the overall stabilized conditions for OTES. This condition is determined by performing

*Note that the curves in Figs. 3-26 and 3-29 do not end at the same point at which they start. This is due to the fact that the computer program was not run to the point of equilibrium.

Fig. 3-26 OTES Section 1 Temperature History, $\beta = 0$ deg, 1 Orbit

Fig. 3-27 OTES Components Temperature History, $\beta = 0$ deg, 1 Orbit

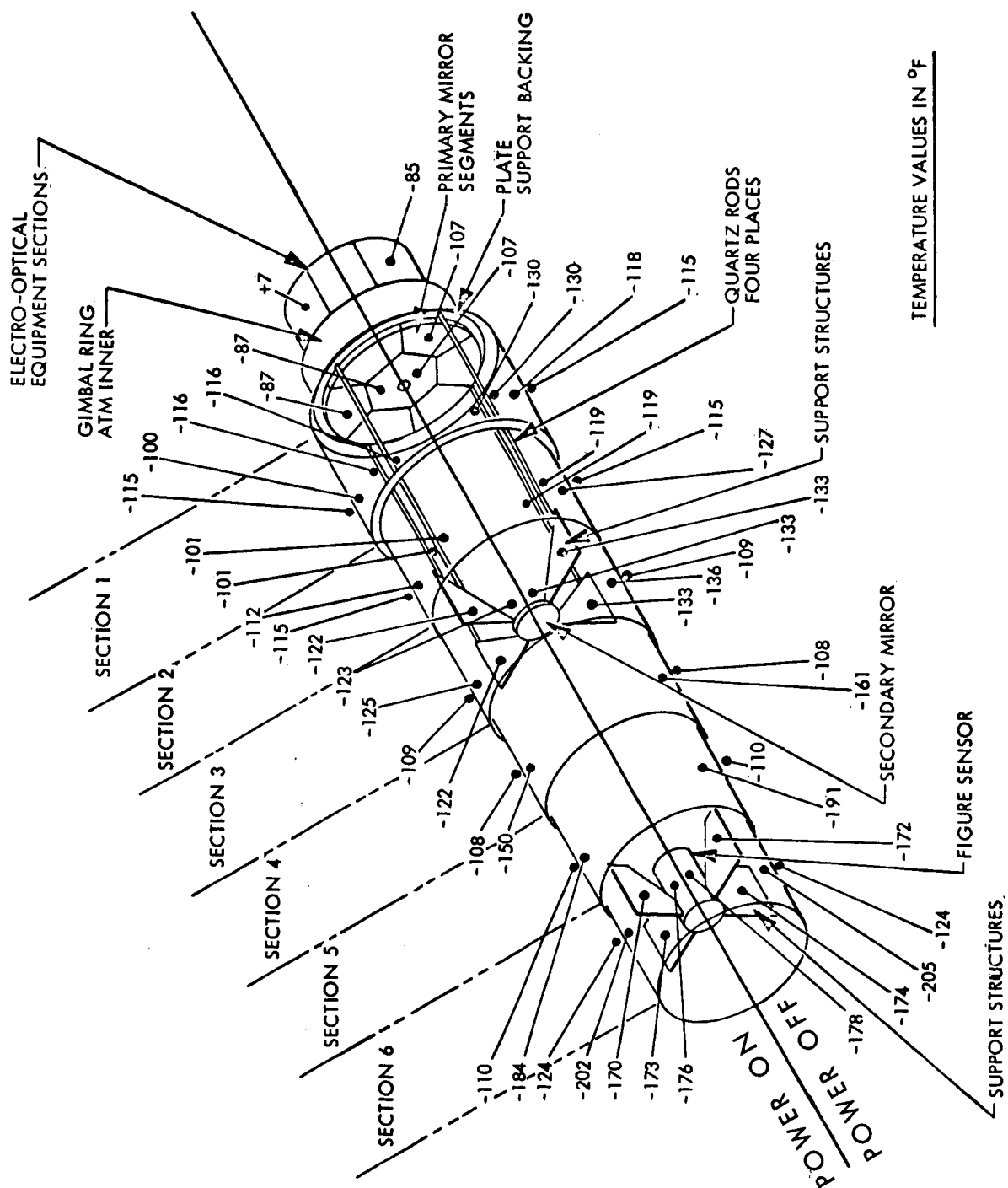
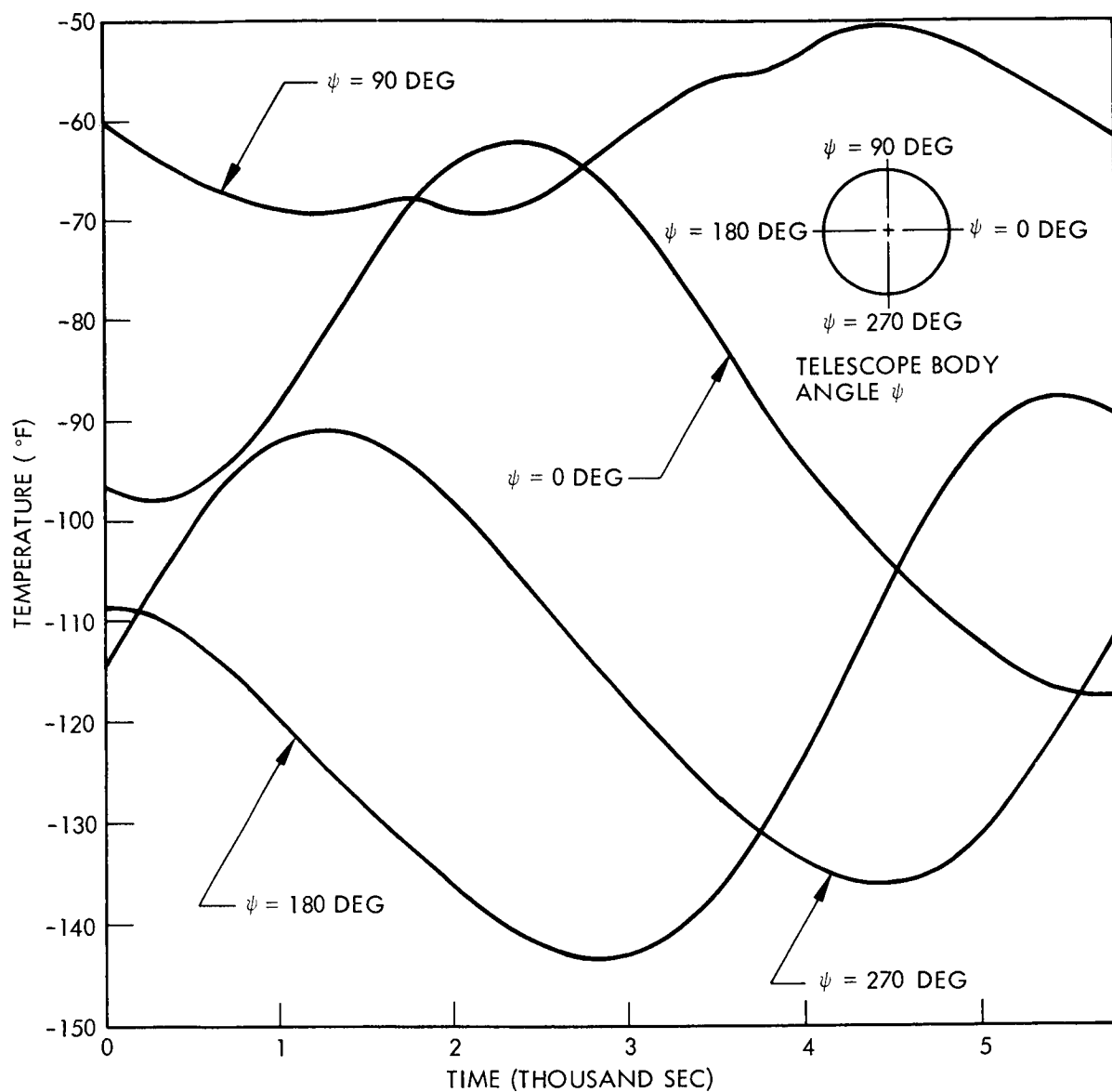


Fig. 3-28 OTES 2-m Telescope Average Temperature Profile, $\beta = 0$ deg, Power On and Off

Fig. 3-29 OTES Section 1 Temperature History, $\beta = 52$ deg, 1 Orbit

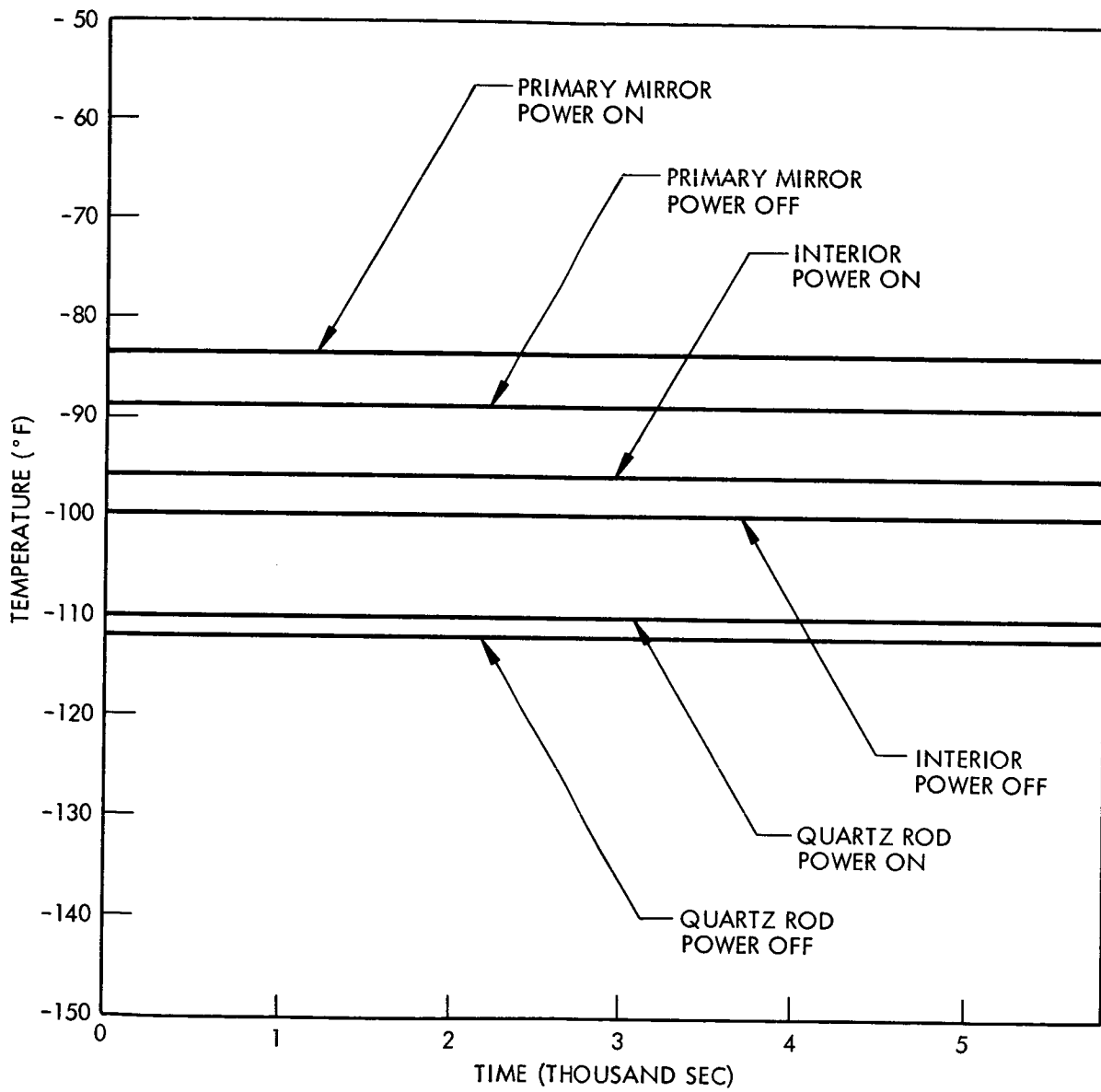


Fig. 3-30 OTES Components Temperature History, $\beta = 52$ deg, 1 Orbit

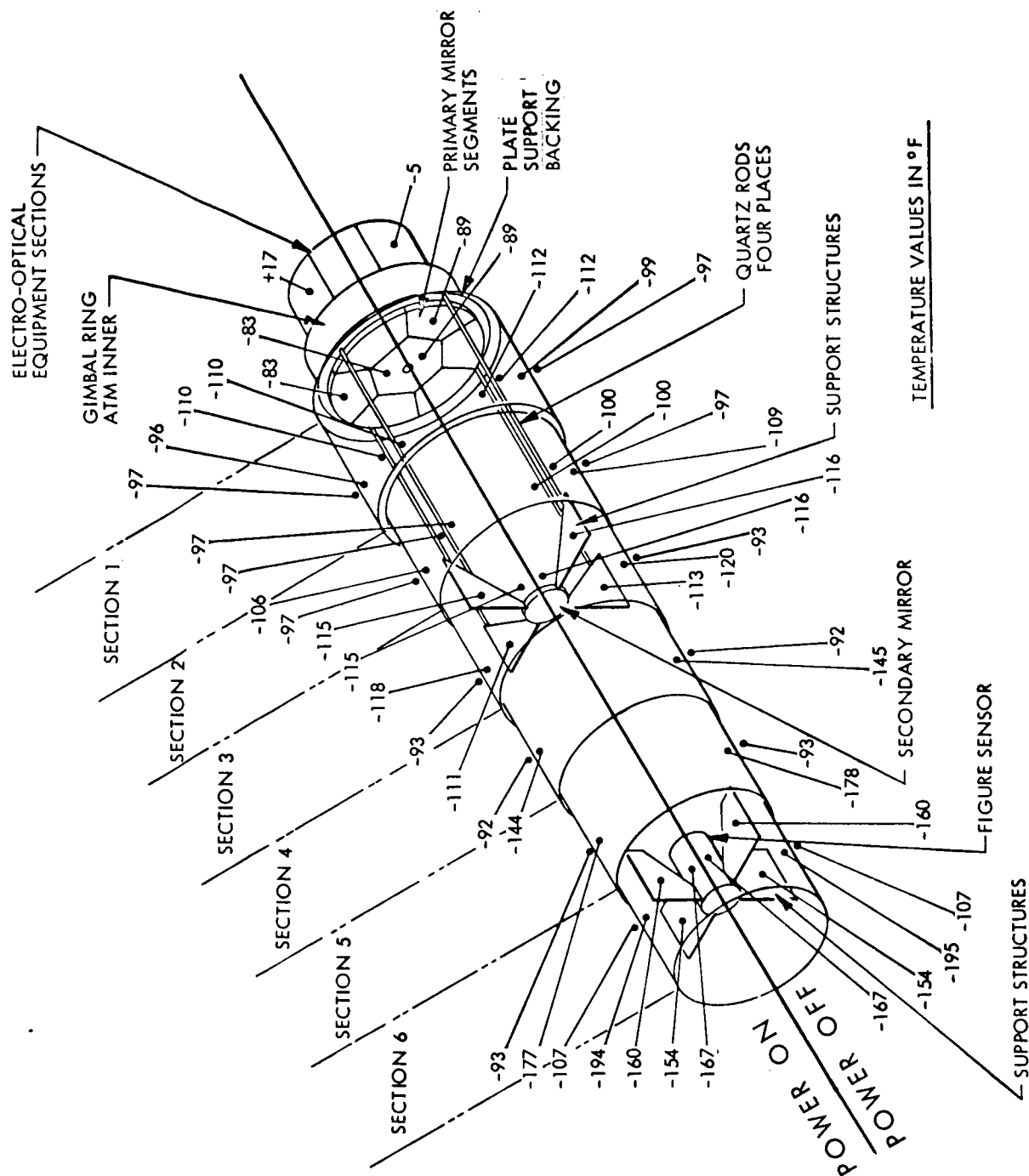


Fig. 3-31 OTES 2-m Telescope Average Temperature Profile, $\beta = 52$ deg, Power On and Off

the net average energy balance for each surface description of the spacecraft in the space environment. The orbit period of a 250-nm mission is 5,622 sec. The following four environmental cases present the overall temperature profile for the OTES application:

- $\beta = 0$. Low-temperature orbit environment with 0-w heat dissipation in the electro-optical equipment bay
- $\beta = 0$. Low-temperature orbit environment with 250-w heat dissipation in the equipment bay
- $\beta = 52$. High-temperature orbit environment with 0-w heat dissipation in the equipment bay
- $\beta = 52$. High-temperature orbit environment with 250-w heat dissipation in the equipment bay

Based on a thermal capacity of 16 Btu/°F on the primary mirror segments, the time required for attaining stabilized temperature levels from the power-off to power-on mode is about 24 hr. The beta angle shift for the AAP cluster configuration is about 1 deg/day. It can be concluded that the variation of primary mirror temperature levels due to beta shift are minimal on a daily basis. As a result of the thermal analysis of OTES in the AAP cluster configuration the following conclusions were reached:

- The temperature gradients across the primary mirror segments can be minimized by passive techniques to less than 1° F.
- The range of temperatures in the primary mirror segments can be controlled from -83 to -107° F in a space-orientation mode using passive thermal control techniques.
- The maximum primary-mirror temperature stabilization time is about 24 hr (from the power-off to power-on mode).
- The thermal gradients are uniform along the length of the four quartz spacer rods, so spacing tolerances can be maintained using passive thermal control techniques.
- Detailed thermal investigations will be required in the preliminary and detailed design phases in certain critical areas.

The recommendations for future thermal studies in the preliminary design and detailed design phases of the OTES Program are as follows:

- Determine the thermal conduction in detail for the path from the primary mirror segments to the support backing plate, secondary mirror to the telescope exterior, figure sensor to the telescope exterior, and the electro-optical equipment bay section
- Investigate the temperature range requirement for all electro-optical equipment to provide a realistic thermal control-system design
- Perform detailed thermal analysis of earth-oriented OTES experiments

3.3.2 OTES ATM Control-System Performance Analysis

Parametric requirements for the baseline ATM flexure control system were determined from the basic considerations of OTES telescope (spar) inertia, flexure pivot spring rate, and maximum static load-carrying capacity for ground checkout and calibration. Using these parameters, control-system reaction to various conditions and disturbances (e.g., mass center offset, spacecraft motion during clustered and independent OTES operational modes, and astronaut movement) were evaluated to ascertain functional capabilities of the selected design.

The effect of a center-of-gravity offset in the spar on one axis of the gimbal control system is illustrated by Fig. 3-32; as shown, spar attitude, as a function of carrier motion, is given by the expression:

$$\theta_s(s) = \frac{1}{1 + K_p} \left[\frac{\left(1 + s\tau b - \frac{s^2}{\omega_1^2}\right) \theta_{c1} + K_p \theta_R(s)}{1 + \frac{2\zeta}{\omega_o} s + \frac{1}{\omega_o^2} s^2} \right] \quad (3.1)$$

where:

A = gimbal control-loop amplifier gain

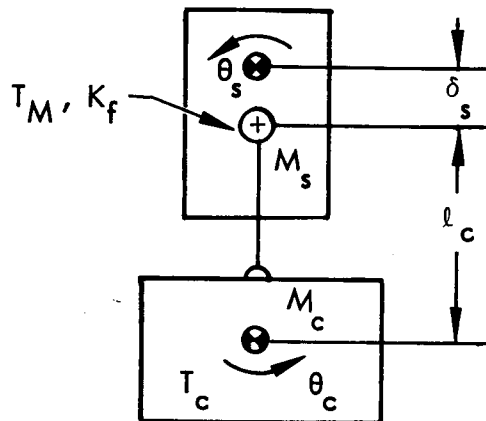
B = gimbal control-loop star sensor gain

C = gimbal control-loop rate gyro gain

E_s = spar pointing error (displacement)

J_s = moment of inertia of spar about gimbal axis

J_{sc} = mass offset inertia = $\frac{M_s}{1 + M_s/M_c} \ell_c \delta_c$



K_f = flexure spring rate per gimbal axis

K_b = back EMF constant

K_T = motor torque constant

K_p = gimbal control-loop gain parameter = $ABK_T/K_f R_m$

M_c = carrier (LM/ATM) mass

M_s = spar (telescope) mass

R_m = motor resistance

ζ = damping factor = $\frac{\tau_b + \tau_p}{2 \sqrt{1 + K_p}} \omega_f$

- θ_R = reference attitude - gimbal control loop
 τ_b = time constant - gimbal motor back EMF = $K_b K_T / K_f R_m$
 τ_p = time constant - gimbal control-loop rate = $A_c K_T / K_f R_m$
 ω_o = undamped natural frequency = $\sqrt{1 + K_p} \omega_f$
 ω_1 = resonant frequency of J_{sc} and spring at pivot point = $\sqrt{K_f / J_{sc}}$
 ω_f = resonant frequency, spar, and flexure pivots = $\sqrt{K_f / J_s}$

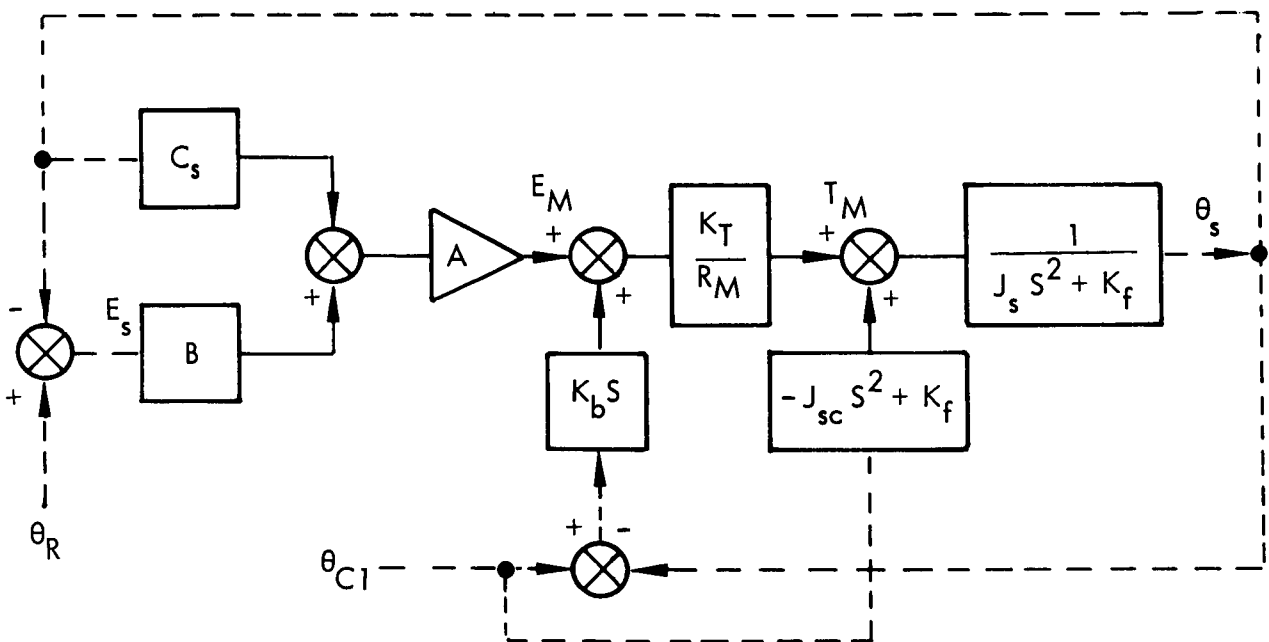


Fig. 3-32 Effect of Center-of-Gravity Offset in Spar

From the equation and figure, a frequency, ω_1 , is defined as the resonant frequency at which the offset inertia, J_{sc} , resonates with the gimbal joint spring. The square of the frequency, depending on the direction of offset in the telescope mass center, can be positive or negative.

A mathematical description of the spacecraft attitude has been used to evaluate the reaction of the flexure pivot-control system to carrier (spacecraft) motion; this

technique is equally applicable to analysis of the magnetic-suspension control system. It is assumed in the model, statistical description of carrier attitude is equivalent to band-limited white noise. This is a reasonable assumption for astronaut-induced disturbances. For this application and operating conditions, disturbances induced by astronaut motion create the critical operating condition for the design of this system. A particular displacement of mass, such as crew movements from one position to another, results in the same change in attitude regardless of the speed at which the movement is accomplished. The response is band limited because the astronaut has an upper limit of speed, and frequencies above a certain level are attenuated by the structure. For simplicity, it is assumed the band limit is a sharp cutoff at frequency ω_c and so the carrier motion power spectral density is given by

$$\Phi(\theta_{c1}) = \frac{\pi \sigma_c^2}{\omega_c}$$

where σ_c is the rms amplitude of carrier deviation from its mean attitude and θ_{c1} is an angle representing motion of the carrier from external torques, astronaut motion, or any source exclusive of the telescope reaction torques. The mechanical power required to sustain this motion is given by

$$P_a = \frac{J_c \sigma_c^2 \omega_c^3}{4}$$

counting both input and output energy as positive. Astronaut-generated power must be greater than this since most of the energy he produces is dispensed changing the kinetic energy of his mass. As an extremely conservative assumption, P_a is taken as 12 m-N/sec, i.e., about 10 percent of the power a man can produce continuously. The value of ω_c is considered to be 1.5 Hz, i.e., somewhat greater than expected for normal human response although less than a man can produce. With these assumptions, σ_c becomes 190 arc sec for the LM/ATM, and the cluster configuration yields far smaller deviations (on the order of 25 arc sec) because of the much larger moment

of inertia, J_c . For independent mode operations where the CSM remains with the OTES spacecraft configuration for crew safety, the mass offset of the CSM tends to further improve control-system performance. A detailed analysis of the latter operating concept, still under consideration by NASA, has not been performed.

The cluster configuration can not be considered as a perfectly rigid body and so application of the foregoing analysis is not strictly valid. However, the LM/ATM configuration is considered to impose more stringent conditions than the cluster configuration.

Two quantities resulting from this motion are of interest - σ_{sc} , the rms pointing error it induces in the spar, and σ_{sc}^* , the rms error rate induced in the spar. These are expressed:

$$\sigma_{sc}^2 = \frac{1}{2\pi} \int_{-\omega_c}^{\omega_c} \Phi(\theta_{c1}) G_1(j\omega) G_1(-j\omega) d\omega$$

$$\sigma_{sc}^{*2} = \frac{1}{2\pi} \int_{-\omega_c}^{\omega_c} \omega^2 \Phi(\theta_{c1}) G_1(j\omega) G_1(-j\omega) d\omega$$

where $G_1(s)$ is $\theta_s(s)/\theta_{c1}(s)$.

Performing the integration yields

$$\sigma_{sc}^2 = \frac{\sigma_c^2 \omega_o}{2(1+K_p)^2 \omega_c} \left[f_1 + \left(\frac{2\omega_o^2}{\omega_1^2} + \omega_o^2 \tau_b^2 \right) f_2 + \left(\frac{\omega_o}{\omega_1} \right)^4 f_4 \right]$$

$$\sigma_{sc}^{*2} = \frac{\sigma_c^2 \omega_o^3}{2(1+K_p)^2 \omega_c} \left[f_2 + \left(\frac{2\omega_o^2}{\omega_1^2} + \omega_o^2 \tau_b^2 \right) f_3 + \left(\frac{\omega_o}{\omega_1} \right)^4 f_4 \right]$$

where

$$f_1 = \tan^{-1} \frac{\omega_c}{\omega_o} + \frac{\omega_c / \omega_o}{1 + (\omega_c^2 / \omega_o^2)}$$

$$f_2 = \tan^{-1} \frac{\omega_c}{\omega_o} - \frac{\omega_c / \omega_o}{1 + (\omega_c^2 / \omega_o^2)}$$

$$f_3 = \frac{2\omega_c}{\omega_o} - 3 \tan^{-1} \frac{\omega_c}{\omega_o} + \frac{\omega_c / \omega_o}{1 + (\omega_c^2 / \omega_o^2)}$$

$$f_4 = 5 \tan^{-1} \frac{\omega_c}{\omega_o} + \frac{2\omega_c}{3\omega_o} \left(\frac{\omega_c^2}{\omega_o^2} - 6 \right) - \frac{\omega_c / \omega_o}{1 + (\omega_c^2 / \omega_o^2)}$$

Two systems were investigated to determine flexure pivot selection criteria — one using a soft flexure pivot and the other a hard flexure pivot. A value of 50 lb-in./radian per pivot was chosen for the flexure pivot spring rate (K_r) of the soft system. This selection resulted in a spring rate per gimbal axis (K_f) of 150 lb-in./radian, i.e., $2 \times 50 = 100$ lb-in./radian for two pivots per axis plus 50 lb-in./radian for leads and connectors required across the spar and carrier interface. The leads and connection spring rates produce an effective spring in parallel with that of the flexure pivots.

For the "hard" pivot system, K_r is given as 600 lb-in./radian, making K_f equivalent to 1,250 lb-in./radian. The spring rate of a given pivot is not constant but, rather a function of load. It has been assumed in the analysis that the pivots are operating in a weightless space environment (unloaded) and, therefore, the spring rate may be considered as constant.

Errors in pointing and jitter resulting from the expressions for σ_{sc} and $\dot{\sigma}_{sc}$ were computed for both systems and are presented in Table 3-9. Sensitivity to center-of-mass offset is particularly pronounced in the hard system. The most stringent requirements result from specification of 1 arc sec/sec for jitter.

Assumptions regarding carrier motion are conservative in both amplitude and frequency. With mild astronaut motion constraints, the value of σ_c can be reduced by a factor of two or three, and bandwidth can be reduced to 0.5 Hz (a significant improvement in jitter resulting from reduced magnitude in the coefficient f_4). The center-of-mass control shown in the table values is realizable except for jitter for the hard pivot design. This jitter represents high rates at very low amplitudes which may not be discernible within resolution of the telescope. Aside from astronaut motion, such disturbances as limit cycling of the carrier control system can cause pointing errors that occur at frequencies that are low in comparison with the bandwidth of the gimbal control loop.

The OTES ATM control system analysis indicates that pointing accuracies of 2.5 arc sec can be achieved easily in either the independent mode or cluster configuration.

3.3.3 Earth Pointing Experiments - Orbit Geometry/Tracking Rates Analysis

A low orbit altitude poses stringent demands on both control system and tracking error bandwidth requirements. Many of the earth-target tracking-system design requirements can be determined by analysis of the orbit geometry parameters. Following is a summary of an analysis of the OTES orbit geometry and its effect on spacecraft design requirements for telescopic viewing of points on earth from a nominal 250-nm orbit. A typical OTES orbit geometry is illustrated by Fig. 3-33, which represents a case where the ground target is in the orbit plane (i.e., the worst case line-of-sight [LOS] with respect to time [wrt]) although only one degree of freedom (pitch) is required of the tracking device. Two degrees of freedom would be required if the target were offset laterally from the orbit plane; however, the rates would be lower than those for the in-plane target.

Table 3-9

ERRORS DUE TO CARRIER MOTION

LM/ATM ($\ell_c = 1.5$ m)

Parameter	Soft Pivot		Hard Pivot	
	No Offset	$\delta_s = 1$ cm	Offset	$\delta_s = 1$ cm
rms pointing error, σ_{sc} (arc sec)	0.12	1.70	0.19	1.0
rms error rate, $\sigma_{\dot{sc}}$ (arc sec/sec)	0.28	11.1	0.77	7.03

LM/ATM With CG Offset ($\ell_c = 1.5$ m)

Parameter	Permissible Offset, δ_s (cm)	
	Soft Pivot	Hard Pivot
$\sigma_{sc} < 1$ arc sec	0.63	1.03
$\sigma_{\dot{sc}} < 1$ arc sec/sec	0.10	0.04

where

Parameter	Soft Pivot		Hard Pivot	
	lb-in./radian	N-m/radian	lb-in./radian	N-m/radian
K_f	150	16.9	1,250	141

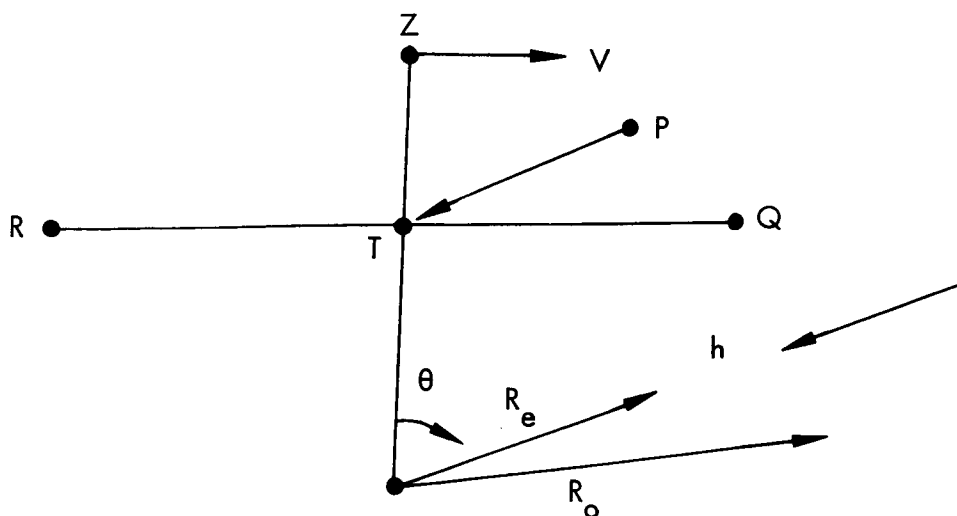


Fig. 3-33 Typical OTES Mission Orbit Geometry

Parameters shown in Fig. 3-33 are defined as follows:

$$R_e = 3,440 \text{ nm (earth radius)}$$

$$h = 250 \text{ nm (orbit altitude)}$$

$$R_o = R_e + h = 3,690 \text{ nm}$$

Considering then

- Orbit period, $P = 2\pi \sqrt{a^3/\mu} = 5,620 \text{ sec, or } 93.7 \text{ min}$
- Angular velocity, $\omega = \frac{2\pi}{5,620} = 0.00112 \text{ radian/sec} = 0.064 \text{ deg/sec}$
- Velocity, $V = \omega R_o = 4.13 \text{ nm/sec}$

The target point, T, on the surface of the earth is theoretically in sight of the orbiting OTES during time of passage from points R to Q. The central angle in which T is in sight is ROQ; hence,

- Half-central angle, $\frac{ROQ}{2} = \arccos \frac{344}{269} = 21.2 \text{ deg}$
- Central angle, $ROQ = 42.4 \text{ deg}$

- Time-of-transit, $t_{\text{RQ}} = \frac{42.4}{3.84} = 11.04 \text{ min}$
- Maximum "in-sight" range, $\text{RT} = 3,690 \times \sin 21.2 = 1,336 \text{ nm}$
- Minimum "in-sight" range, $\text{ZT} = 250 \text{ nm}$

The geometry of the line-of-sight angular position and tracking rate is presented in Fig. 3-34.

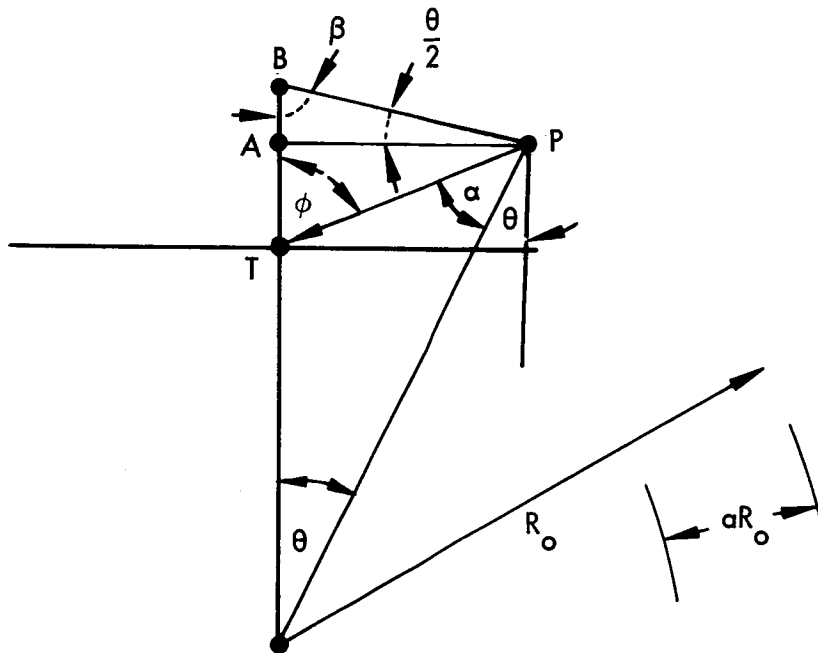


Fig. 3-34 Line-of-Sight (LOS) Angular Position and Tracking Rate (let altitude, a , be expressed as a function of orbit radius, $h = aR_0$)

The following relationships are evident in Fig. 3-34:

$$\phi = \alpha + \theta$$

$$2\beta + \theta = \pi$$

$$\beta = \pi/2 - \theta/2$$

The equation for angular position, ϕ , can be derived using standard trigonometric rules; i.e., $\phi = \arccot [a \csc \theta - \tan (\theta/2)]$, which expresses the LOS angular position as a function of a and the central angle, θ . However, ϕ is referenced to the vertical through point T. The angular position of the LOS with respect to the

spacecraft, α , is required to determine the degree of freedom of the pitch gimbal. This is the "look angle" and is referenced to the local vertical - i.e., $\alpha = \phi - \theta = \arccot [a \csc \theta - \tan (\theta/2)] - \theta$.

The rate of the LOS with respect to time at the local vertical through T is $\dot{\phi} = \dot{\theta} (W s \alpha \sin \phi / \sin \theta)$. The rate of the LOS with respect to time at the spacecraft is $\dot{\alpha} = \dot{\phi} - \dot{\theta} = \dot{\theta} [(\cos \alpha \sin \phi / \sin \theta) - 1]$.

Figure 3-34 shows the look angle, α , versus central angle, θ , through the range of interest; it is evident that when the target comes into view the angle is about 69 deg and decreases rather slowly until the spacecraft is within about 5 deg of the central angle of the target. At $\theta = 5$ deg, the look angle is 48 deg. To observe the target throughout a total central angle of 10 deg would require a pitch gimbal of ± 48 deg and a corresponding transit time of 2.6 min.

Figure 3-24 shows the rate of LOS, $\dot{\alpha}$, versus the central angle, θ . The following data can be deduced from this illustration:

- When the target first comes into view, $\dot{\alpha} = 0$ when $\theta = 21.2$ deg.
- When the spacecraft is directly over target, $\dot{\alpha}_{\max} = 0.88$ deg/sec.
- Maximum acceleration ($\ddot{\alpha}_{\max}$) is 0.021 deg/sec² and occurs at about 3 deg central angle.

3.3.4 Electrical Power-System Analysis

Analyses of the electrical power-subsystem requirements for the OTES mission were conducted for both cluster and independent operating modes. Cluster power requirements were examined for all OTES operating modes and for stellar and earth target applications. Primary consideration was given to the minimum modifications required for use of the AAP ATM Cluster A configuration as a basis for the OTES spacecraft

conceptual design. The power subsystem configuration requirements for OTES were defined to a level consistent with Phase II design efforts, and the areas requiring additional work in subsequent phases of study were identified.

For the independent mode, the spacecraft was considered to comprise the LMSC/rack/OTES combination. Power requirements were investigated and a comparison was made of the weight penalties associated with battery versus solar array power. Given independent mode operations for short durations, battery power alone would suffice; this could be extended by periodic replacement of rechargeable batteries from the cluster, e.g., from a solar-array powered battery charging unit located in the orbital workshop. For longer duration missions in the independent mode, a solar array would be installed on the ATM rack as the primary power source.

Cluster Power-System Analysis. In performing the feasibility analysis of the AAP ATM power system for OTES-mission cluster-mode operations a number of design constraints had to be estimated or assumed. Table 3-10 lists these constraints in two general categories, i.e., general constraints influencing solar-array requirements, and constraints imposed by the cluster/spacecraft interfaces and compatibility requirements.

A 2-yr power supply for a manned OTES mission in 1973 poses an unusual combination of long, reliable life and short development cycle. Stringent pointing requirements (± 0.01 sec) for stellar photography and a protracted film exposure time (6 to 12 hr) tend to relegate against use of subsystem approaches employing moving parts and associated disturbing forces to pointing accuracy. Therefore, the ideal electrical power system must employ currently flight-qualified components that will not cause disturbing torques during film exposure. Non-tracking solar arrays were selected to fulfill this requirement. Chemical electrical power systems were omitted on the basis of prohibitive weights and cost. Table 3-11 is a summary of electrical-system concepts considered and eliminated as noncompetitive.

Table 3-10

OTES ELECTRICAL-POWER CONSTRAINTS

General Constraints (Clustered Operation)

No cluster rotation for partial gravity in S-IVB workshop

No solar arrays mounted on OTES telescope rack (deployed or surface-mounted)

LM-A/rack hard docked to MDA at all times when OTES experiments are being conducted

No tracking solar arrays

Solar arrays repositioned in any manner must be locked in one position during film exposures of 6 to 12 hr

No interference by solar array on OTES telescope pointing during any orbit

Cluster always considered to roll about OTES telescope axis to achieve the most favorable attitude for use of solar arrays; only exception - acquisition, tracking, and photographing of earth

Cluster Constraints

LMSC-proposed S-IV solar array considered for Cluster X with minimum modification

Compatibility with current Cluster A vehicle designs and their power supplies achieved with minimum number of changes

LM-A/rack/OTES telescope capable of supplying power to all cluster vehicles from rechargeable batteries and to draw power from the XCSM fuel cell bus in an emergency

Power sources for Cluster X capable of being electrically isolated from vehicle loads; common ground required for vehicles using the same power source

Present Cluster A ATM gimbal system used for containing and pointing the OTES telescope; same torque motors/flex pivots used as basis for electrical loads

Table 3-11
NONCOMPETITIVE ELECTRICAL-POWER SYSTEM FOR OTES

Space Power System	Specific Disadvantage for 1973 OTES Mission								Remarks
	Low Reliab.	Long Devel.	Rad. Hazard	Complex	No Demon. On-Orbit Perform.	Poor Flex. in Sys.	Mat. Prob.	High Costs	
Radioisotope, Thermoelectric, or Thermionic Generator		x	x				x	x	Unfavorable political connotations and problematical fuel availability
Nuclear Reactor									Requires very-heavy biological shielding and radical cluster reconfiguration
Powered Systems:									
Rankine Cycle	x	x	x	x	x	x	x	x	
Brayton Cycle		x	x	x	x	x		x	
Thermoelectric			x	x				x	
Thermionic		x	x	x	x	x	x	x	
Solar Collector									Severe and continuous collector pointing requirement
Systems:									
Rankine Cycle	x	x		x	x	x	x	x	
Brayton Cycle		x			x	x			
Thermoelectric		x			x				
Thermionic		x			x		x	x	

The OTES telescope must be capable of orientation with a given star for extended periods up to 12 hr. During the entire daylight portion of its 260-nm orbit, the solar array must be capable of maximum output for any cluster orientation. At any given stellar orientation the cluster will be fixed inertially, i.e., not rotating about its own axis. The telescope is also required to acquire, track, and photograph earth targets; this would imply that the electrical system must have a tracking solar array to assure continuous solar orientation. A fixed array of a size meeting cluster electrical load requirements must necessarily be large and heavy for earth-oriented operations since the size would be selected on the basis of the most unfavorable orientation with respect to the sun. The minimum area effectivity (minimum projected area/total projected area) of a fixed array can be as low as 0.129.

Elimination of both standard fixed arrays and tracking arrays to satisfy OTES mission requirements leaves only the selection of an intermittently repositioned type of array that is not capable of maximum output during earth-pointing mission applications. A more straightforward approach, however, is to depend solely upon fuel-cell or battery power for the earth-oriented mission application. For the stellar-observation mission the array must be adjusted periodically for maximum output and could be intermittently repositioned, using manual mechanical techniques to maintain maximum electrical output for the fixed orientation of a given OTES target star and orbit plane.

In consideration of solar-powered primary electrical sources for the AAP cluster, LMSC designed a solar-array system for mounting on the orbital workshop (S-IVB) and with a maximum performance, when oriented toward the sun, of 7.6 kw. During the orbital storage mode planned for Cluster A (3 to 6 mo), the cluster is gravity-gradient stabilized with its axis pointing toward earth and the cluster remaining free to rotate about its axis. The forward and aft end wings of each half of the array, as shown in Fig. 3-22, can be rotated. Preliminary calculations show a minimum output of 800 w (average) can be obtained for any orbit inclination. (Cluster A orbit plane varies from 5 to 52 deg.)

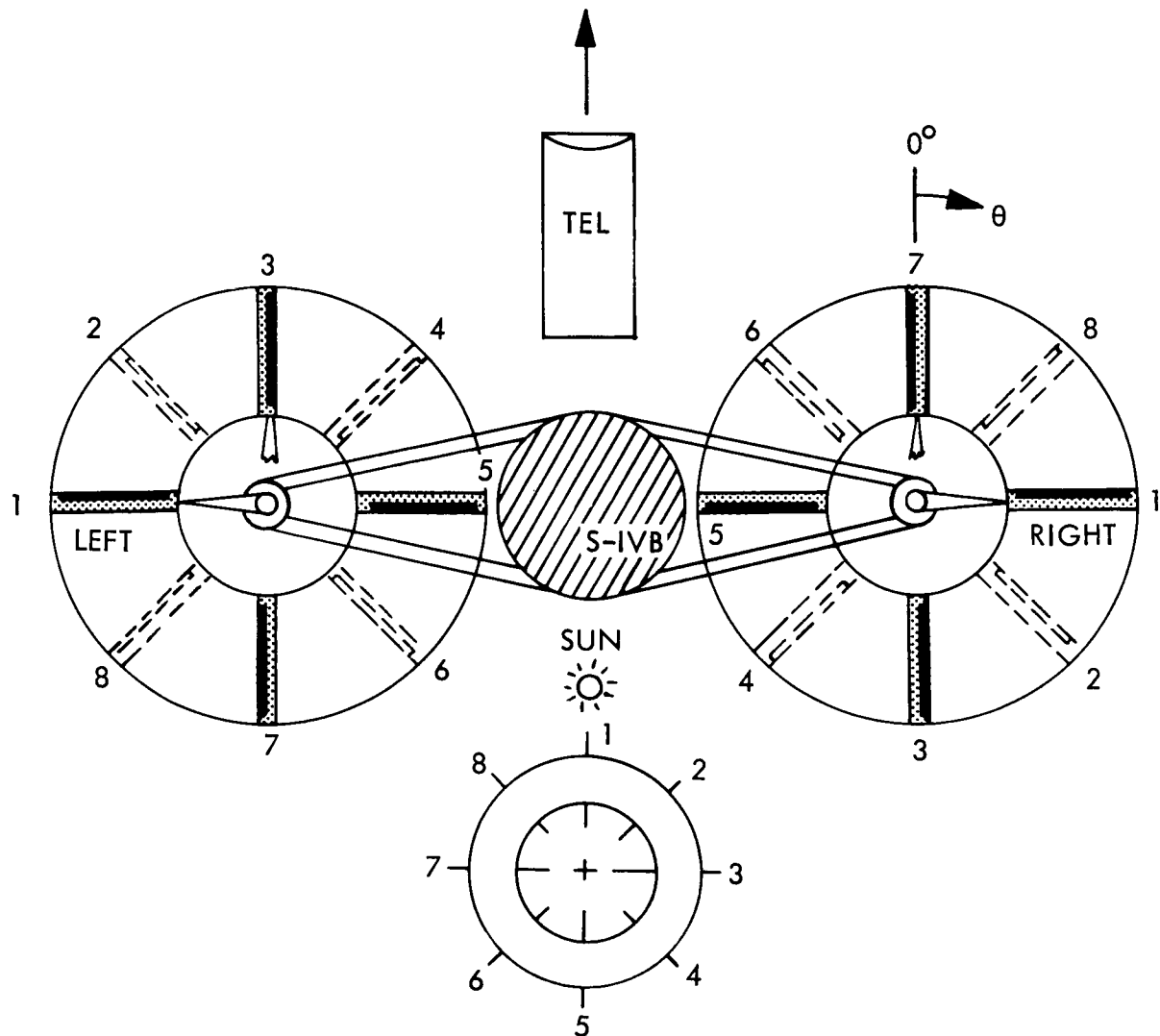
The order-of-magnitude performance variation for the LMSC proposed S-IVB solar-array design for Cluster A is intolerable for the OTES mission application. However, performance improvements can be achieved to provide satisfactory output for any stellar orientation by the following:

- Adding cells on both faces of the wings
- Increasing angular travel of the array wings
- Changing the array hinge line to be parallel with the S-IVB axis
- Using boom-mounted arrays
- Providing circular tracks on the S-IVB to permit 360-deg rotation for intermittent repositioning operations

The addition of cells to both faces of the array panels would result in significant performance change in panel output and require requalification, and may result in launch date delays. Estimates of weight penalties for this are on the order of 4,000 lb, and performance would still be inadequate for some cluster orientations. Hence, this approach is not considered a good solution. Increases in the angular travel of the wings (now set at 65 deg) would be inadequate for the same reasons. Change in the array hinge line, as shown in Fig. 3-35, is a more extensive modification and also has orientations wherein partial shading and degraded or insufficient performance may occur.

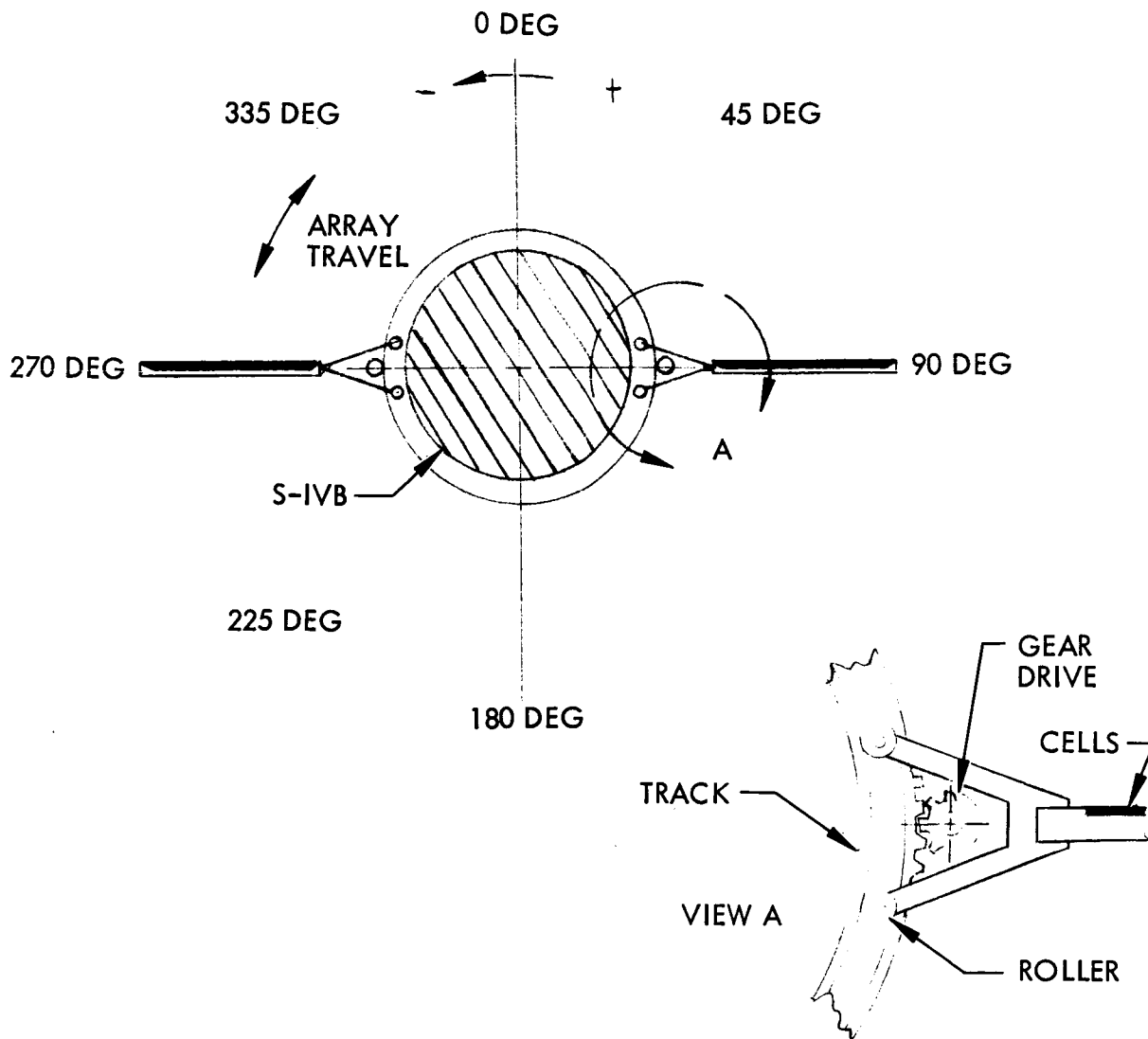
Maximum array output for any cluster orientation is achieved by the design approach shown in Fig. 3-36, i.e., providing a circular track on the S-IVB structure so the array can be repositioned mechanically following deployment; both halves of the array are always normal to solar irradiation; shading of any wing by the S-IVB cannot occur for any array position.

During the present study effort, no attempt was made to compare the minimum-area repositionable array system with simpler techniques having larger size and weight. The latter design concept, however, is presently considered the baseline candidate primary power system for the OTES mission application.



SUN POSITION	ARRAY ORIENTATION, θ (DEG)		SHADED ARRAY
	LEFT	RIGHT	
1	270	90	NONE
2	315	135	NONE
3	0	180	NONE
4	45	225	LEFT
5	90	270	NONE
6	135	315	RIGHT
7	180	0	NONE
8	225	45	NONE

Fig. 3-35 Rotatable Arrays, Boom-Mounted



SUN POSITION (DEG)	LEFT-ARRAY POSITION (DEG)		RIGHT-ARRAY POSITION (DEG)		SHADING
	INITIAL	FINAL	INITIAL	FINAL	
0	270	270	90	90	NONE
45	270	335	90	135	NONE
90	270	0	90	180	NONE
135	270	45	90	225	NONE
180	270	90	90	270	NONE
225	270	135	90	335	NONE
270	270	180	90	0	NONE
335	270	225	90	45	NONE

Fig. 3-36 Track-Mounted Arrays

Detailed definition of AAP Cluster A subsystem interfaces have not been completed at the present time. Moreover, cluster vehicle designs are not firm, and the number, type, and characteristics of selected experiments are not well defined. However, it is assumed that the AAP subsystems configurations will be developed and qualified fully at the time required for OTES applications in 1973 and that only minimum modifications will be necessary to serve the new mission application. Primary consideration for the current study has been analysis of Cluster A interfaces, as presently defined, with the OTES-mission-peculiar system requirements.

Sizing of the OTES-modified S-IVB solar array is dependent upon the ratio of fuel-cell output to battery output during dark-time operations. Vehicle loads comprise experiment loads for the OTES mission and conventional cluster equipment loads, e.g., life support, communications, attitude control, etc. A tradeoff analysis of the optimum ratio between use of the fuel cell and the battery will be required. Preliminary data from Perkin-Elmer for OTES electrical-power requirements show a constant load of 250 w with no peaks (300 w, including the magnetic suspension system experiment) following experiment activation. Other electrical loads associated with the system configuration are presented in Table 3-12, and are based upon preliminary data relative to Cluster A and conceptual data for subsequent cluster planning. Table 3-13 and Fig. 3-37 are summaries of the probable power sources that will be available for the OTES mission. A preliminary estimate of the OTES cluster electrical loads for a single orbit has been made for two cases:

- Heavy use of the CSM electrical fuel cells during dark-time cycles
- Relatively little use of fuel-cell power, i.e., sufficient use for thermal conditioning and/or use of cryo-boiloff from reactant supply tanks

Both cases are illustrated schematically by Fig. 3-38. It can be seen that the solar array output required for Case 2 is about 1,500 w above that for Case 1. Table 3-14 illustrate the weight differences and battery data for the two cases; weight savings of over 1,500 lb are gained using Case 2 for a 90-day mission.

Table 3-12
PRELIMINARY CLUSTER X ELECTRICAL LOADS

Vehicle	Load (w)				Remarks
	Minimum	Average	Maximum	Duty Cycle (%)	
XCSM	N/A	1,120	N/A	N/A	Housekeeping load
OWS (S-IVB) AM	200	1,180	1,180	N/A	Minimum load is for unmanned flight
MDA	N/A	N/A	N/A	N/A	Lighting loads will be low
LM-A	N/A	800	N/A	100	Includes housekeeping, instrumentation, Communication, and display
OTES Rack	1,020	1,020	1,020	100	Pointing control system; does not include battery recharge requirement
OTES Telescope Experiments	250	250	250	100	Data from P-E
OTES With Magnetic Suspension	300	300	300	100	Data from P-E
Total		4,370*			

*Does not include recharging secondary batteries.

Table 3-13
PRELIMINARY CLUSTER X ELECTRICAL-POWER SOURCES

Vehicle	Power Supply (w)			Remarks
	Minimum	Average	Maximum	
XCSM (Fuel Cells)	1,200 to 1,400		4,260	Data from Cluster A sub-system specification for a 56-day mission
OWS (S-IVB) Cluster A Configuration Modified for Cluster X	800*	4,700 Per Orbit	7,600	Based on design for solar-oriented Cluster A; Cluster X is <u>never</u> solar-oriented
AM	7,600	7,600	7,600	During daylight; requires movable arrays
LM-A	0	0	0	Does not include any primary batteries
OTES Rack	Variable	Variable	Variable	LM-A batteries are not included Depends upon number of secondary batteries used

*Based on an unmanned gravity-gradient stabilized mode.

DAYTIME MODE:

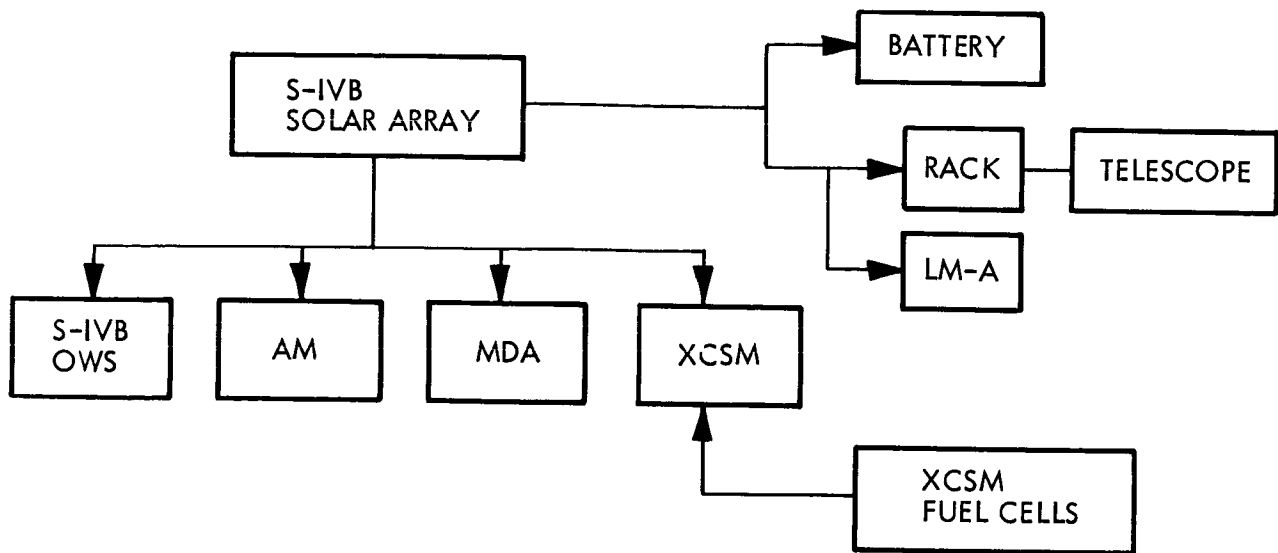
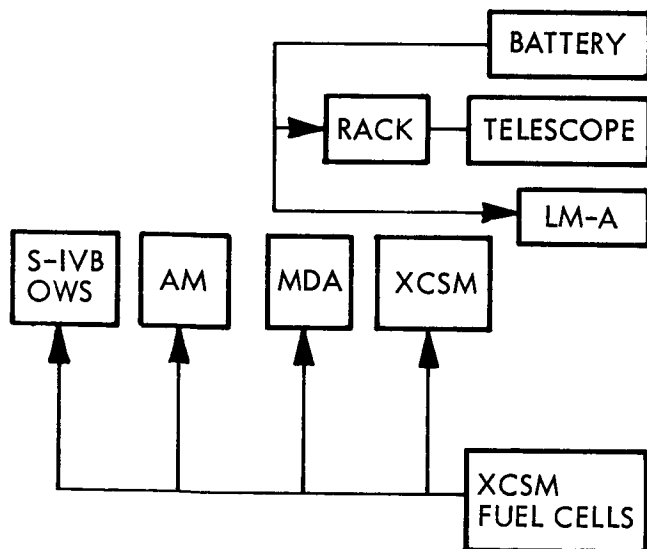
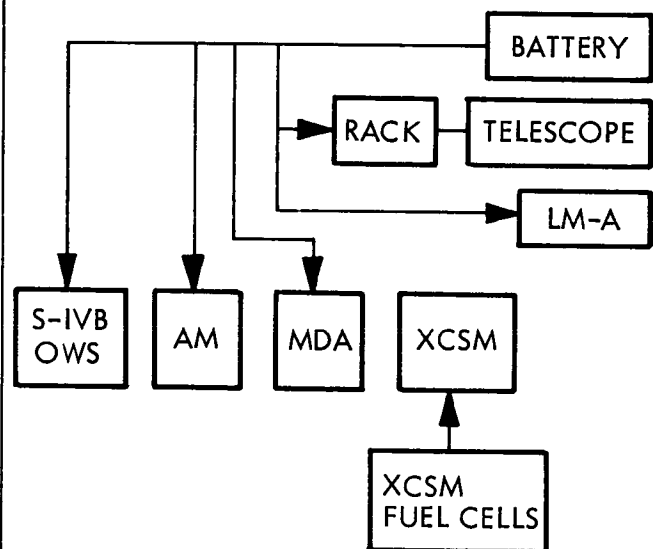
DARK-TIME MODE
(MINIMUM FUEL-CELL USAGE):DARK-TIME MODE
(HEAVY FUEL-CELL USAGE):

Fig. 3-37 Cluster B Electrical Power Interconnections

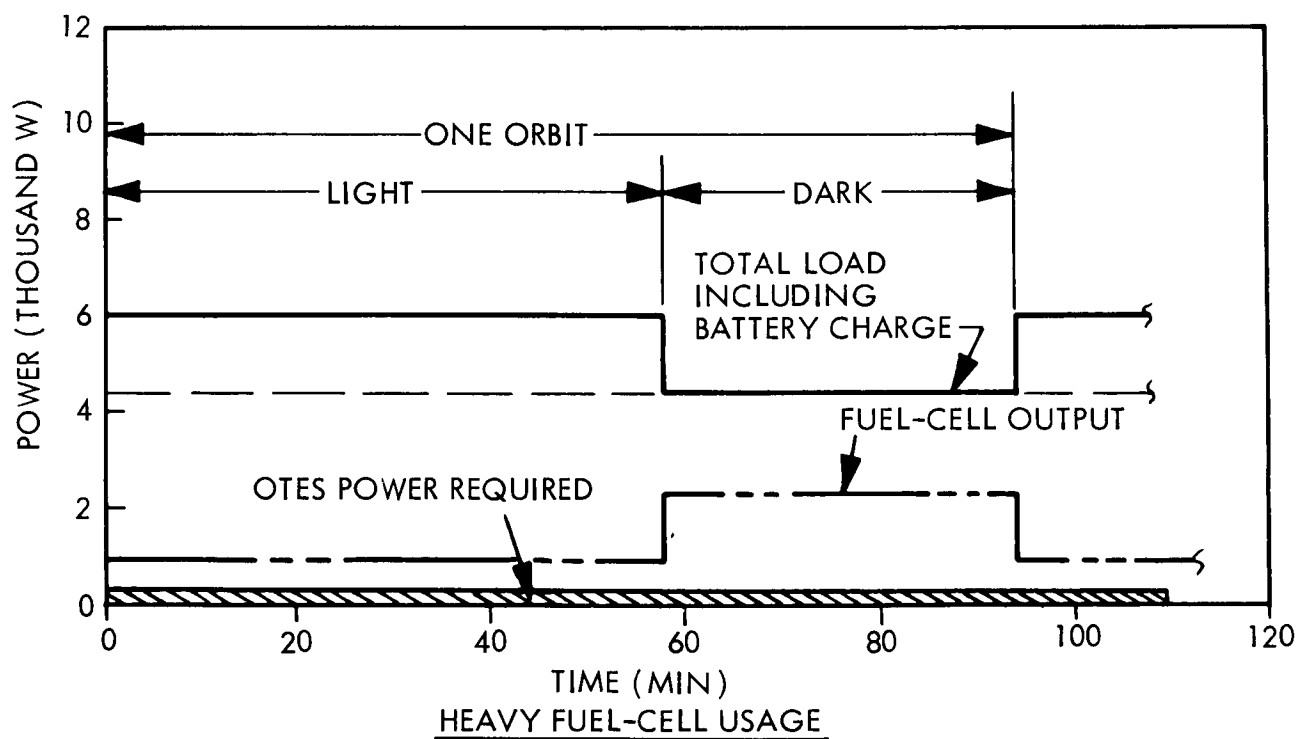
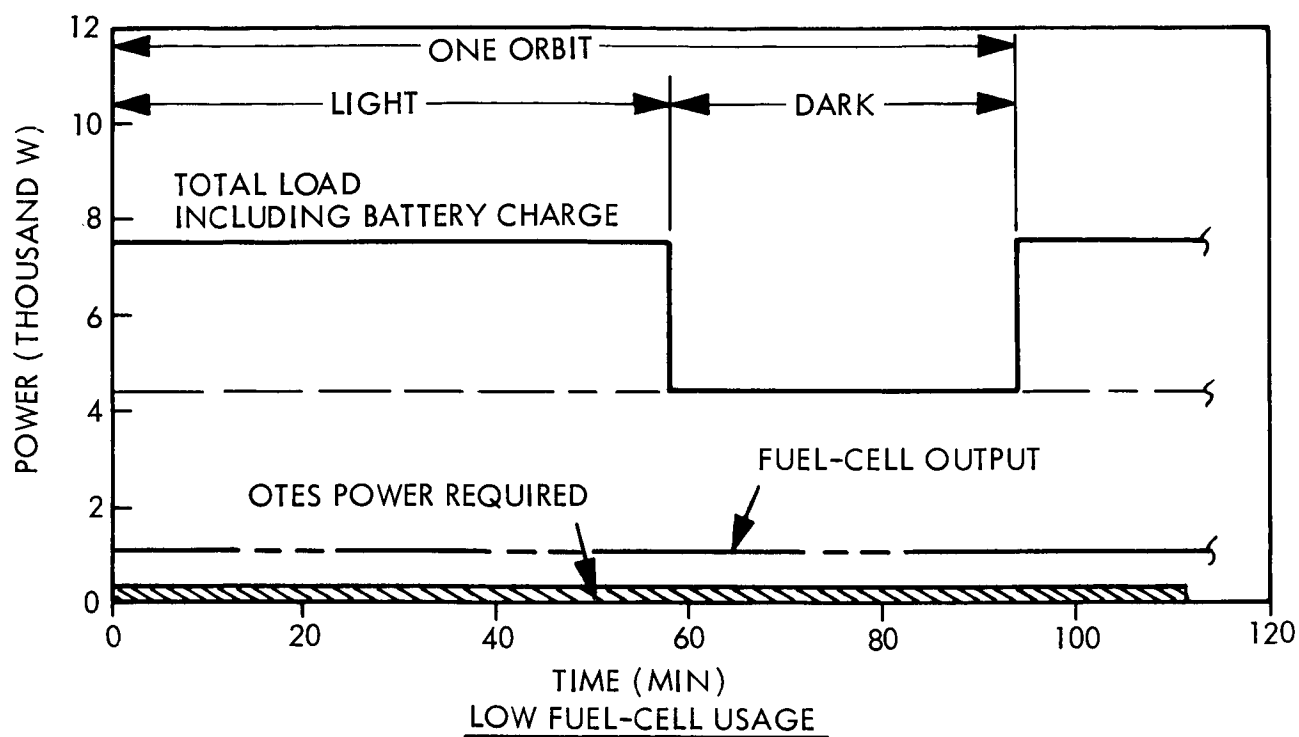


Fig. 3-38 Cluster X Electrical Loads

Table 3-14

CLUSTER X SECONDARY BATTERY AND FUEL-CELL
REACTANT REQUIREMENTS FOR 90-DAY MISSION

Case 1	Case 2
Rack-mounted secondary batteries to supply power during dark-time operations for LM-A, Rack, and OTES telescope experiments	Batteries to supply all cluster loads during dark-time operations except XCSM housekeeping loads
Battery electrical load: 2,070 w	Battery electrical load: 3,250 w
No. of MSFC modules required: 10	No. of MSFC modules required: 18
Total module weight: 750 lb	Total module weight: 1,350 lb
Fuel-cell loads Daytime: 90 w Dark-time: 2,300 w	Fuel-cell loads Daytime: 1,120 w Night-time: 1,120 w
90-day reactant weights (lb) H ₂ : 370 O ₂ : 2,210 Tankage: <u>1,800</u> Total: <u>4,380</u>	90-day reactant weights (lb) H ₂ : 130 O ₂ : 1,320 Tankage: <u>790</u> Total <u>2,240</u>
Total weight (lb) Battery 750 Reactant <u>4,380</u> <u>5,130</u>	Total weight (lb) Battery 1,350 Reactant <u>2,240</u> <u>3,590</u> (1,540 lb lighter than Case 1).

Performance data taken from current AAP data for P&W fuel cells and reactant tankage and from NASA-MSFC data on a charger/battery/regulator module were used for the comparative analysis of the two approaches. A more detailed study is required to identify overall weight tradeoff between the solar arrays and fuel-cell reactant tankage upon availability of more realistic, detailed data. The final Cluster A design may employ power conditioning methods and components quite different than those used in the current study and would lead to different qualitative results.

Location of the primary solar-array power supply system on the S-IVB orbital workshop structure appears to be the most satisfactory arrangement for OTES cluster operations; LMSC designs for this configuration have been generated in support of the AAP Cluster A missions using the ATM solar telescope experiments. For the OTES mission, orientation requirements associated with various stellar targets will require implementation of a repositionable array design that can be adjusted intermittently for proper solar orientation to yield maximum power output for each target position.

Electrical loads for the OTES Cluster X are not established but will probably be equivalent to those presently planned for Cluster A. OTES experiment power loads are much smaller than those planned for the Cluster A ATM. Preliminary tradeoff studies indicate that a 90-day mission favors minimum use of the CSM fuel cells for dark-time electrical power. Preliminary definition of interface requirements for the OTES will be performed in subsequent study efforts when Cluster A interface definitions have been more firmly established.

During the OTES Phase B study, it is recommended that the following tasks be performed with respect to further evaluation of electrical-subsystem requirements:

- Monitor AAP Cluster A electrical power-system design programs to maintain a current listing of design changes or innovations affecting design of the OTES
- Perform additional design analysis of movable arrays to provide the maximum power output for any cluster orientation while demonstrating minimum size and weight penalties

- Conduct tradeoff studies based on the latest available data to determine optimum balance between rack-mounted batteries and improved fuel-cell reactant tanks
- Refine studies of the requirements for earth-oriented telescope mission applications and examine alternate power control techniques
- Perform detailed analysis of the influence of the proposed "free floating" telescope magnetic suspension-system hardware OTES power-system loads and operation

Independent Operating Mode Power-System Analysis. Independent operation of the LM/ATM rack/OTES telescope spacecraft configuration has been investigated to determine power requirements and weight penalties for battery versus solar-array power. Preliminary analysis indicates that nominal electrical power loads for the independent mode are approximately 2,050 w comprising:

- Lunar module (housekeeping): 600 w
- Lunar module (instrumentation, communications, displays): 200 w
- Pointing control system (PCS): 1,000 w
- OTES experiments: 250 w
- Total: 2,050 w

This load can be reduced by about 25 percent by operating only the environmental control system and attitude control of the LM and conducting independent mode operations from within the CSM (as presently under consideration by NASA planners). For short-duration independent mode operations it would be possible to provide electrical power from self-contained batteries (rechargeable modules) exchanged with batteries being charged from the primary solar-array power source in the S-IVB orbital workshop. Currently available, flight-qualified silver-zinc secondary batteries (Type I-H) demonstrate a power density of 86 w-hr/lb and a 75 percent depth of discharge permissible for a moderate number of cycles. Performance for 24-, 48-, and 72-hr independent-mode operations are given in Table 3-15.

Table 3-15

INDEPENDENT MODE OPERATIONS

Type Operation (hr)	Required Power Level (w-hr)		Battery Weight (lb)	
	Minimum	Nominal	Minimum	Nominal
24	38,600	49,200	580	760
48	77,200	98,400	1,160	1,520
72	105,800	147,600	1,740	2,280

Previous work on rechargeable batteries was based on use of the MSFC charger/battery/regulator modules for high reliability over a 1-yr mission; i.e., the nickel-cadmium batteries would be cycled every orbit at a depth of discharge of 25 percent. The effective power density of the module, therefore, was comparatively low. For extended independent mode operations, however, this approach is not considered feasible. Necessarily, moreover, there would be a size and weight increase for recharging Ag-Zn batteries; this application has not been evaluated in depth.

For longer-duration independent modes of operation it would be desirable to provide an onboard solar array for electrical power. The independent mode, moreover, would be subject to much the same constraints as the cluster operations, i.e.,

- Nontracking solar-array configuration (no disturbing torques)
- No fixed array (too large, heavy, and inefficient)
- No interference with telescope operations for any stellar orientation
- No interference with communications antenna-pattern radiations

A simple deployed-array configuration with cells mounted on one side has been examined. The array, comprising two opposing wings and based upon current LMSC technology for panel fabrication and deployment, would maintain its most efficient orientation (panels normal to the sun's rays) by using a drive motor for intermittent positioning of the wings following orientation with the given stellar target. It is

assumed that the LM reaction control system could be used to roll the LM/rack/OTES about the telescope axis as required prior to commencing stellar photography. Again, for earth-pointing targets, the solar-array design would not be efficient; the continuous pitchover requirement to keep on target would cause a previously positioned array to degrade in performance as the spacecraft orbits the earth. Hence, the battery or fuel-cell electrical-power source would be required for this mission application; a comparatively short time interval of battery operation would be required for tracking specific earth-based targets.

Additionally, the design for the independent mode operations solar array must allow capability for storing or retracting the arrays during docking and docked-cluster operations. The deployment of the solar arrays for both cluster and independent mode operations also affects the radiation patterns of the communications system; as noted in a previous discussion, mounting of the communications system antennas on extendable booms has been examined as a possible method for overcoming solar-array shadow effects. Conversely, the positioning of the booms so as not to shadow the cells and degrade power must also be examined.

Summarized array size and weight data are as follows:

- Total array area: 220 ft²
- Wing area: 110 ft²
- Panels required: 124
- Panel area: 1.78 ft²
- Array weights (total): 500 lb
 - Panels: 278 lb
 - Positioning motor: 22 lb
 - Structure/mechanism: 200 lb

A plot comparing the above array weights and corresponding battery penalties is presented by Fig. 3-39. It will be noted that the secondary-battery configuration is heavier than that for the solar array for independent operations beyond 20 hr. A 3-day

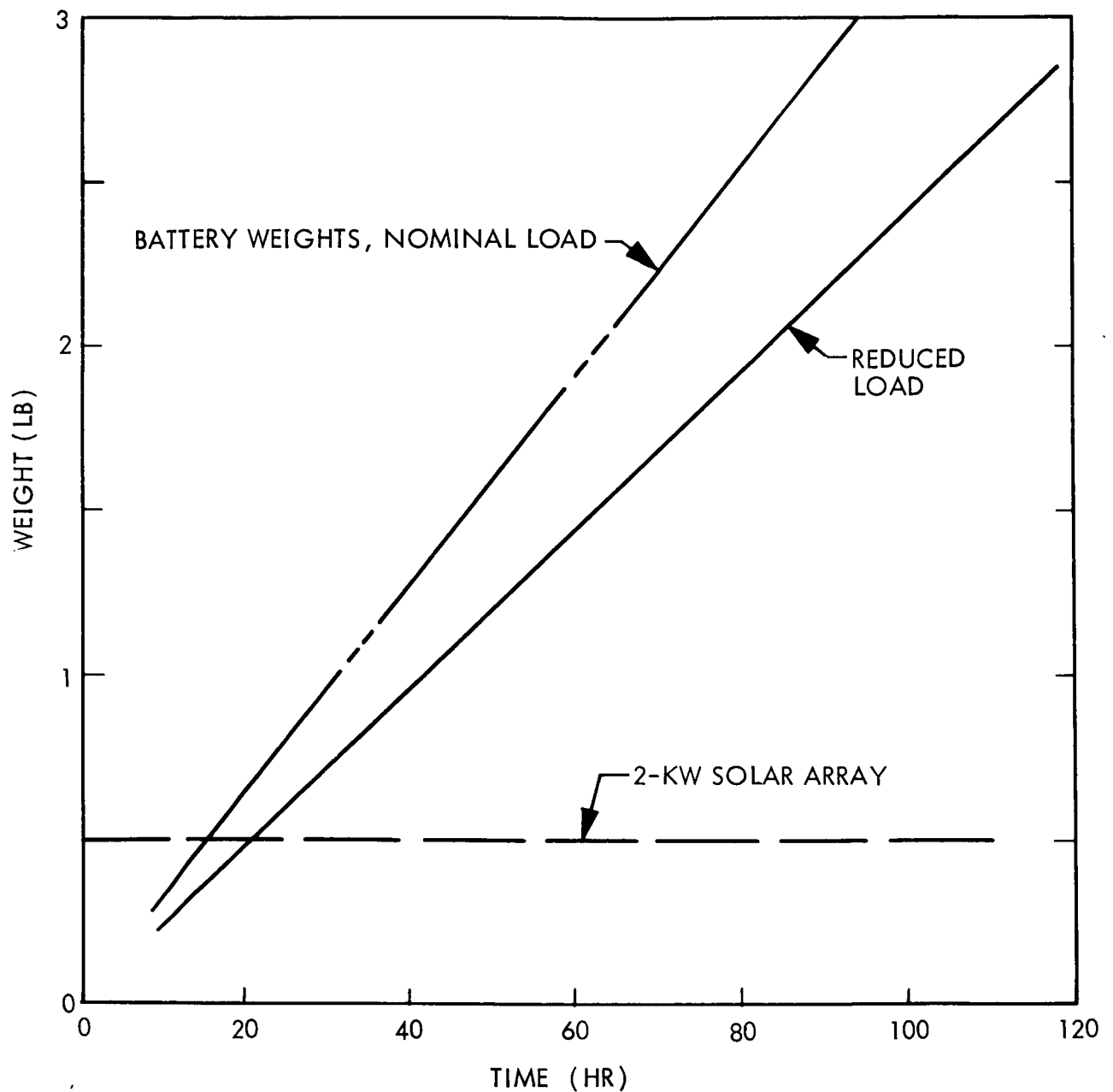


Fig. 3-39 OTES Independent Mode, Weight Penalty vs. Time

battery-powered excursion could be accomplished with 500 lb of less weight penalty if the LM subsystems operations are held to ECS and attitude-control subsystem operation.

Independent mode operations of the OTES beyond 20 hr can best be accomplished using solar-array power; a deployed rack could be mounted on the ATM rack. Beyond 20-hr missions, the use of batteries would incur a weight penalty of 24 lb/hr even with the majority of the LM subsystems inoperative.

The two-wing solar array must be capable of being repositioned intermittently by rotating each wing up to 180 deg about its own longitudinal axis. This must be accomplished between stellar photography tasks to avoid disturbing vehicle torques. The solar array must (for multiple independent mode operations with interim docking [cluster] operations) also provide for retraction or storage of the array.

Earth-pointing tracking missions are best accomplished using battery power; this is reasonable since on-target periods would be for relatively short durations.

Additional cluster-weight penalties will be incurred if independent OTES operations must be accomplished only with rechargeable batteries.

It is recommended that subsequent study phases be used for the following:

- Define more precisely the independent mode operations and perform trade studies for weight optimization
- Evaluate advanced solar-cell designs for OTES that may be applicable to the 1973 launch data
- Investigate cluster power interactions with both the S-IVB array and OTES array sources supplying power to a common bus
- Update Cluster A fuel-cell study data based on advanced reactant (cryogenic) supply systems

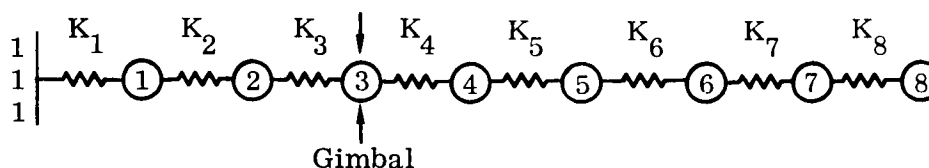
3.3.5 Structural Dynamics/Stress Analysis

The magnitude of relative deflection between critical optical components resulting from dynamic (elastic) deflections of the OTES structure was investigated during the Phase A, Part II study period. The use of six telescoping segments for extension of the OTES spar from the ATM posed a potentially critical problem of attaining sufficient rigidity between segments when the telescope was erected for use; the method of analysis of this condition was to determine modes and frequencies of the telescope structures and calculate the modal response to an appropriate forcing function.

Dynamic Model. A lumped-parameter model, representing lateral dynamic properties of the elastic structure, was derived from the mass and stiffness data presented by Figs. 3-40 and 3-41. This model comprises eight mass lumps interconnected by shear and moment springs which allow translation and rotational freedom of motion. Lateral dynamic model parameters for the eight masses are summarized by Table 3-16.

Table 3-16
LATERAL DYNAMIC MODEL

No.	CG Sta.	Mass (lb sec ² /in.)	Inertia (in. lb sec ²)	K _S (million lb/in.)	K _m (million in. lb/radian)
1	- 98	6.640	2,000	0	0
2	- 32	2.690	1,330	0.170	140
3	- 2	1.305	495	0.440	3,380
4	22	2.225	2,470	0.168	9,250
5	140	1.160	3,410	0.250	3,113
6	258	0.600	1,055	0.322	1,650
7	337	0.525	730	0.467	3,320
8	385	0.536	250	0.751	3,650



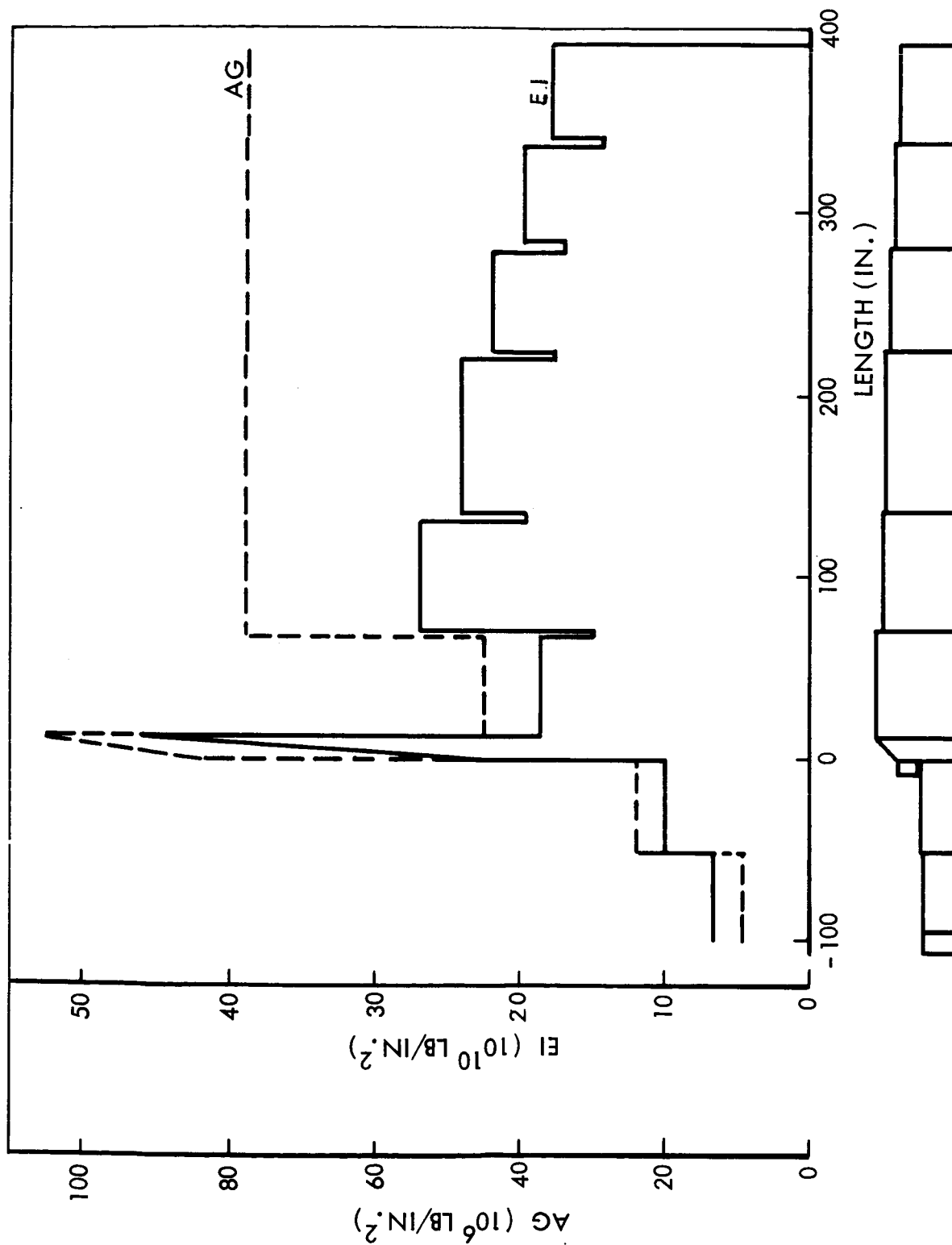


Fig. 3-40 OTES Stiffness Distribution

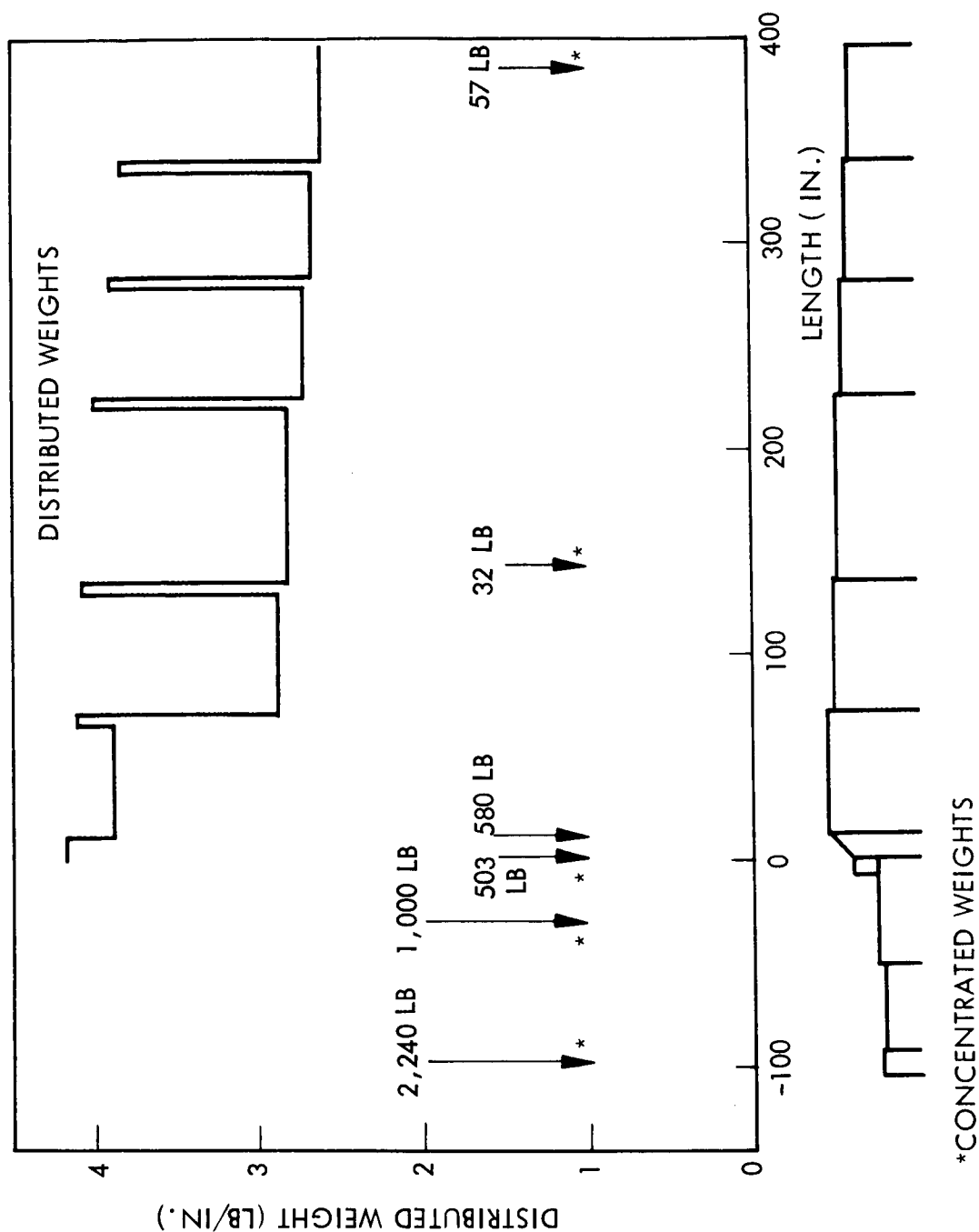


Fig. 3-41 OTES Weight Distribution

Calculated Modes. Normal modes and frequencies of the telescope were computed from the dynamic model by solving the matrix equation

$$\left[M - \frac{1}{W_n^2} C^* K C \right] \bar{q}_n = 0$$

where

M = matrix of mass elements

K = matrix of spring stiffness

C = coordinate transformation matrix

from spring deflection into mass deflections,

q_n = vector deflections (elastic mode "n")

W_n = natural frequency of mode "n" in radians/sec

The modes are plotted in Figs. 3-42 through 3-44; results are tabulated as follows:

Parameter	Lateral Mode		
	1	2	3
Frequency (Hz)	18.83	44.11	73.22
Generalized mass (lb sec ² /in.)	6.567	6.244	3.965

Forcing Function/Elastic Displacement. The response of the elastic telescope structure was calculated under an idealized random forcing function representing motion of the spacecraft cluster due to astronaut movement. It was assumed the astronaut movements result in a band-limited flat spectrum of angular motion of the cluster and can be defined by the spectral density function

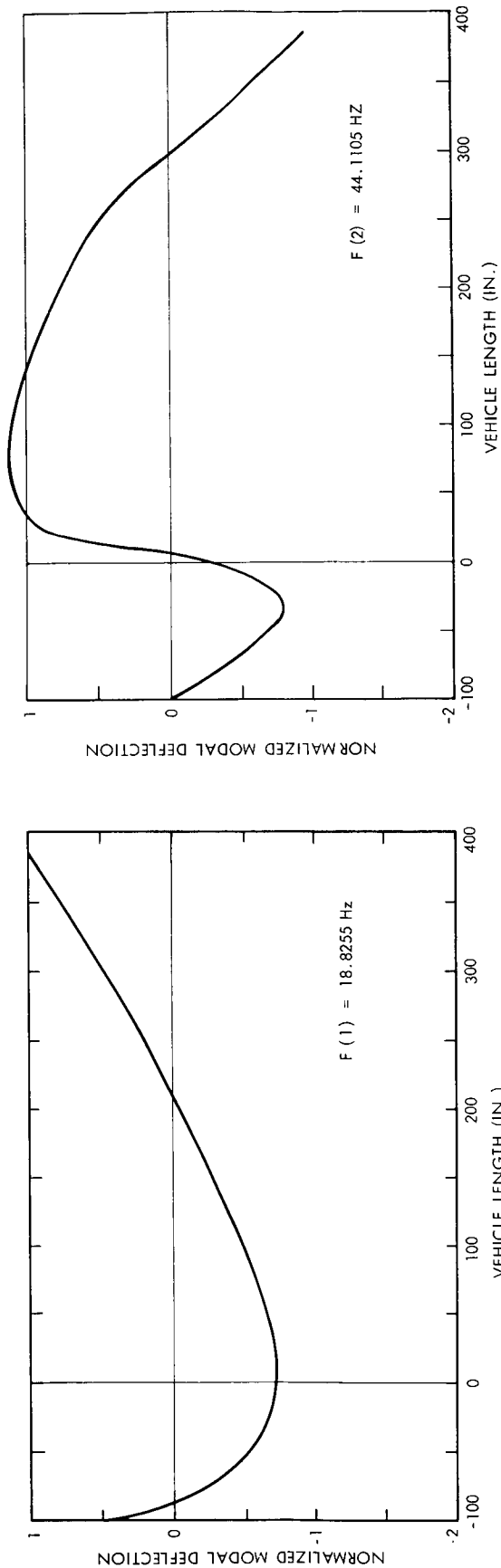


Fig. 3-43 OTES Second Lateral Mode

Fig. 3-42 OTES First Lateral Mode

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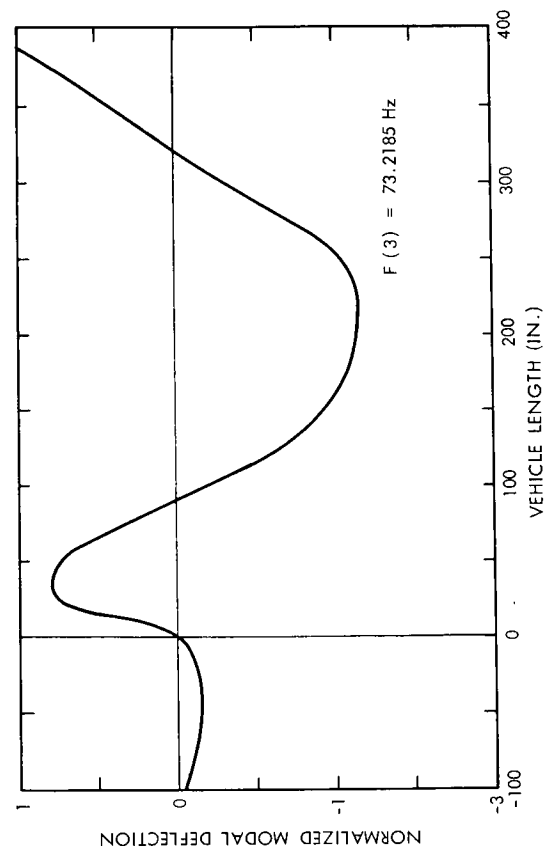


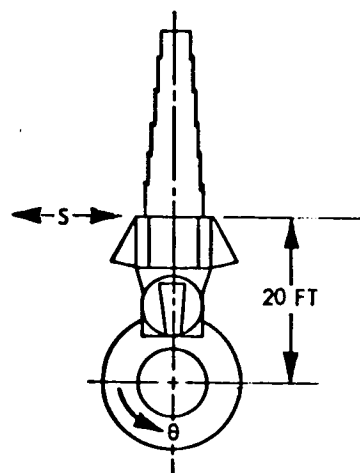
Fig. 3-44 OTES Third Lateral Mode

$$\Phi(\theta) = \frac{\pi \sigma_c^2}{f_c}$$

where σ_c is the rms angular deviation of the carrier, θ is the rotation of the carrier, and f_c is the cutoff frequency. For this study, $f_c = 1.5$ Hz and $\sigma_c = 24$ arc seconds. This is taken from the AAP/ATM control system analysis (subsection 3.3.2) in which astronaut motion was singled out as a significant dynamic excitation.* Magnitudes of f_c and σ_c were considered highly conservative. Note that 24 arc seconds corresponds to a cluster inertia which includes the LM/ATM/MDA docked to the command and service modules. Astronaut movement is taken to be the only excitation causing elastic perturbations of the telescope during operating periods. Forces such as atmospheric drag, gravity, etc., which have long periods relative to periods of structural dynamic modes, are considered to be in steady state, i.e., nonoscillatory, and were not included in the analysis.

The random spectrum of motion was assumed about the center of gravity of the clustered spacecraft, with the rotational motion transferred to the OTES through the gimbal ring located 20 ft away (see figure below). The transferred motion was taken to be translational only because of the small rotation angle and long lever arm, and because the natural frequency of rotation of the rigid OTES on the flex pivot gimbal is very low relative to the structural frequencies of the telescope. It is assumed that there is no loss or gain of translation motion through the gimbal.

Motion is transmitted directly to the third mass lump of the dynamic model (see Table 3-16), to excite the natural modes of the structure. This is indicated by the following equations of motion:



* Lockheed Missiles & Space Company, Conceptual Design and Analysis of Control System for Apollo Telescope Mount, LMSC-A842157, Sunnyvale, Calif., 17 Mar 1967.

$$\ddot{q}_n + W_n^2 q_n = \frac{F_o}{R_n} \sin wt$$

$$F_o = M_i \phi_{in} W_n^2 X_{in}$$

where

- q_n = generalized coordinate of mode n
- W_n = natural frequency of mode n , radian/sec
- R_n = generalized mass, mode n
- M_i = mass of lump i
- ϕ_{in} = modal deflection of lump i in mode n
- X_{in} = displacement of lump i in mode n

The steady-state solution of this equation is

$$q_n(W) = \frac{F_o/W^2}{\left[1 - (W/W_n)^2\right]} = \frac{M_i}{R_n} \phi_{in} X_{in} \frac{1}{\left[1 - (W/W_n)^2\right]}$$

and the deflection at any point j on the structure at any impressed frequency w is

$$X_{jn}(W) = \phi_j q_n(W)$$

Thus, for a given mode, a transfer function

$$h_j(W) = \frac{\phi_j q_n(W)}{X_{in}(W)}$$

may be derived for any location on the structure. The response spectrum $\Phi_j(W)$ is then obtained using

$$\Phi_j(W) = h_j^2(W) \Phi_i(W)$$

A difference transfer function may be obtained for two points on the structure using

$$\Delta h_{ji}^2(W) = h_j^2(W) - h_i^2(W)$$

from which a power spectrum of the differential motion between two points is

$$\Delta \Phi_{ji}(W) = \Delta h_{ji}^2 \Phi_i(W)$$

The root mean square relative motion is then obtained from

$$\Delta_{\text{rms}} = \sqrt{\int_0^{f_c} \Delta \Phi_{ji}(W) df}, \quad f = \frac{W}{2\pi}$$

Response of the OTES structure was defined using only the first mode because of the high frequency of the modes relative to the 1.5 Hz cutoff frequency of the input spectrum.

Spectral density functions defining motions of the sensor and of the secondary mirror relative to the primary mirror in the first lateral mode of the OTES are shown in Fig. 3-45. Corresponding displacements of the figure sensor and of the secondary mirror relative to the primary mirror are

$$\Delta_{\text{rms}}(\text{sensor}) = 0.0044 \text{ in.}$$

$$\Delta_{\text{rms}}(\text{secondary}) = 0.0036 \text{ in.}$$

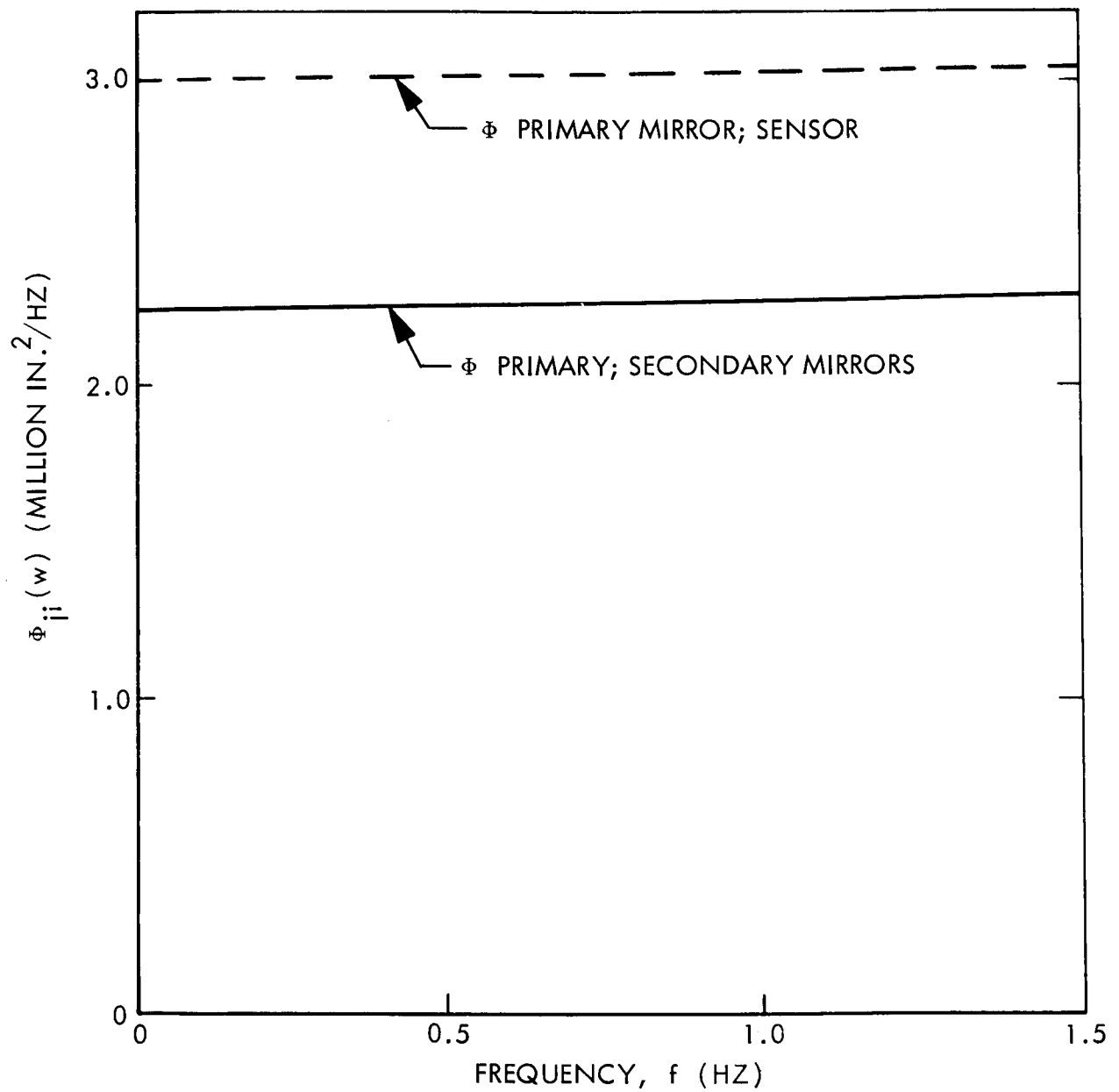


Fig. 3-45 Definition of Sensor and Secondary Mirror Motion
by Spectral Density Functions

Stress Analysis. Theoretical evaluation of the structure indicates essentially that no critical static loading conditions are imposed on the support system. This is based on the fact the design stiffness characteristics will minimize deflection of the telescope system during operation. Of significant importance is the design and development of the interface joints of the telescoping barrel segments of the spar to maintain the required degree of rigidity during the extension sequence. Additional studies of the peculiar thermal environments, when defined in more detail, will provide design data for further analysis of the structural system. The presently defined OTES stiffness values resulted from assumed joint fixity factors = 0.80. This value appears to be readily attainable.

Section 4
ASTRONAUT-PARTICIPATION ANALYSIS

4.1 SCOPE

As currently defined, the OTES system will function within the environmental constraints of the Apollo Applications Program (AAP) Cluster A or B flights. The primary mode of operation for the mission will be in the cluster configuration, although the undocked OTES vehicle (comprising a lunar module, Apollo telescope mount [ATM] rack, and 2-m telescope) will be required for individual mode operations. The Orbital Mirror Recoating Facility, although part of the ascent of the OTES vehicle, would be docked adjacent to the OTES on the centralized docking adapter of the cluster and, hence, would not be a functional part of the independent mode operations.

The mission philosophy and operational procedures for crew participation must be fully compatible with established AAP constraints and guidelines. Hence, crew-participation concepts developed during the Phase I OTES study effort have been appropriately modified and refined during Phase II as a result of more detailed evaluation of mission functions, representative experiments, assumed mission environment(s), and data derived from Gemini flights, simulations, and experimental data.

Previous study results have been reconsidered in light of revised guidelines and the modified spacecraft concepts. A clear distinction is made between intravehicular activity (IVA) and extravehicular activity (EVA). The latter requires considerably more detailed analysis and attention with regard to crew operations. Onboard experiments involve a more or less common set of functions and operations that differ primarily with respect to specific combinations of functions, accuracy requirements, and event sequences. Capabilities to perform tasks within the confines of the spacecraft have been substantiated adequately through flight experience, simulation, and

experimental data. Detailed experiment implementation during Phase D of the OTES Program will require a highly detailed level of crew function analysis. For the present study effort, however, at least a fundamental level of understanding of crew participation is necessary to permit mission planning, conceptual designing of the system, and development of a resources plan. Representative summary information has been applied for maximum definition of crew activities; the data are extracted from NASA guidelines for crew performance and from results of studies performed by Lockheed in support of AAP-class manned space missions. The scope of the present study, thus, has been confined to the following:

- Identification of crew participation in experiments
- Specification of constraints and guidelines for operations safety
- Evaluation of feasibility, with emphasis on EVA operations
- Definition of crew training and training-facility requirements

As the flight system and experimental-design characteristics and physical features are identified more clearly in subsequent phases of study, a more precise definition of the detailed tasks, operations, and sequences related to each will develop. Future detailed analyses would include identification of specific requirements for crew participation with regard to such parameters as the following:

- Experiment objectives
- Functional, hardware, and performance characteristics
- Orbit parameters (altitude, inclination, and eccentricity)
- Specific stellar-target requirements or operations
- Experiment operation requirements (seasonal, time, temperature, power, procedures, and sequences)
- Navigation, control, and orbit maneuver requirements
- Preflight, inflight, and postflight astronaut requirements
- Ground flight-operations-support requirements
- Location and mounting requirements versus ascent storage requirements

- Experiment environmental requirements (EMI, contaminant protection, radiation shielding, thermal, etc.)
- Test, checkout, and calibration (preflight, inflight, postflight)
- Data requirements (science data output, monitoring, postflight support, and data reduction)
- Display and control requirements
- GSE and facility requirements, including preflight astronaut training

4.2 ASTRONAUT INVOLVEMENT

Potential astronaut participation in specific OTES experiments has been examined and an estimate made of modes of behavior that would be typical in the conduct of each. Table 4-1 is a matrix indicating OTES experiments and possible tasks relative to astronaut participation. It appears, on a qualitative basis, that crew-participation requirements can be determined for many of the experiments, e.g., setup, checkout, monitoring, evaluation, and reprogramming. The configuration change experiment and mirror recoating facility experiment are prime candidates for evaluating the feasibility of crew participation in the OTES mission since both applications involve removal of mirror segments and performance of EVA. Coating the mirror would involve rather complex remote sensing devices and mechanization for automated refurbishment; however, these functions could be simplified considerably by using the spacecraft crew. Several other experiment-related activities involve EVA operations, such as film retrieval, or recovery of samples from the experiment modules for return to earth in data recovery capsules. Modification of the telescope configuration for the magnetic suspension system test experiment offers further opportunities for crew participation, monitoring, control, and sequencing operations. Crew contributions to the reliability and potential life of the experimental or flight system also are important weighting factors in establishing the feasibility of crew participation.

Table 4-1
NOTES MISSION ASTRONAUT-PARTICIPATION MATRIX

[illegible]

4.3 SYSTEM CONSTRAINTS AND GUIDELINES

Constraints imposed by the OTES system on crew participation and guidelines for system design and operations are as follows:

- The crew will not be required to perform any activity that would tend to distract from primary mission success or that does not significantly increase system efficiency, if automatic or remote alternatives are possible.
- At least one astronaut will be in his suit at all times during orbit operations. The crew will be in their couches and suits and will be awake during all powered maneuvers.
- Each astronaut will specialize in specific experiments, but all crewmen will be capable of performing tasks associated with all experiments although individual proficiency level will tend to vary.
- No more than two crewmen will operate in the habitable module (LM, MDA, etc.) at one time; i. e. , one astronaut will remain in the CSM at all times.
- The normal location of the crew, while not engaged in experimental tasks or pressurized IVA or EVA, will be within the CM.
- Normal eating, sleeping, personal hygiene, and other stationkeeping tasks will be performed in the CM; limited eating and personal hygiene may be required in the habitable module.
- One astronaut can perform pressurized IVA or EVA from the airlock module; a second astronaut will standby in the module.
- Pressurized suits will be worn by the crewmen upon entering the habitable area for the first time or upon reentering the area after a period of closed-hatch operation in the CM.
- Voice communications between crew members will be monitored at all times.
- Each astronaut will be assigned one 8-hr "watch" each day, during which he will control the spacecraft from the system engineer's couch.
- The first 2 or 3 days and last 2 days of a flight will generally not be utilized for conducting experiments.
- During telescope operations a minimum of additional crew activities will be necessary.

These preliminary guidelines are conservative with respect to crew activities, but they constitute a baseline set of constraints for OTES operations planning.

4.4 CREW TIME ALLOCATIONS

Detailed analysis of the AAP Cluster A mission tasks were available for the study and were applied as an index of overall distribution of crew activities on the OTES mission. With respect to the Phase I studies, refined values that are more closely aligned to the baseline (Cluster A) spacecraft design were available; hence, the summary of time allocations in Table 4-2 are considered a reasonable estimate for the OTES mission.

Table 4-2
TYPICAL AAP CLUSTER A TIME ALLOCATIONS (%)

Function	A1/A2	A3/A4
Personal Time	50.8	50.5
Housekeeping	11.6	11.6
Spacecraft Operations	3.2	2.2

This distribution allocates 65 percent of the crew time to basic system operations and maintenance and the remainder, less some contingency time, to experiment functions. Experiment times for the AAP missions have been about 25 percent, leaving 10 percent contingency during the orbit periods of the mission. These allocations are based on some of the following guidelines (abstracted from NASA); spaceflight experiments to date indicate from 6 to 8 hr/man/day for crew maintenance activity, distributed as follows:

- Three exercise periods of 10 min each: 0.5 hr
- Three hygiene periods of 20 min each: 1 hr
- Three eating periods of 1 hr each: 3 hr
- Unstowing, stowing, assembly, and breakdown: 2 hr

- Flight plan updating period (two at 15 min): 30 min
- Miscellaneous (work/rest): 1 hr

Each astronaut will spend an average of 16 hr/day performing mission and crew maintenance duties. About 8 hr/day/man remains for applications experiments and systems support; 8 hr of continuous sleep is desirable and should be as close to normal sleep as possible. A typical daily schedule of personal activities is presented in Table 4-3.

Table 4-3
TYPICAL DAILY ASTRONAUT SCHEDULE

Activity	Time (hr)
Exercise	00:00
Hygiene	00:10
Meal	00:30
Sleep (8 hr)	01:30 to 9:30
Exercise	9:30
Hygiene	9:40
Meal	10:00
Work (5-1/2 hr)	11:00 to 16:30
Exercise	16:30
Hygiene	16:40
Meal	17:00
Work (6 hr)	18:00 to 24:00

Specific time allocations are required for activities, other than housekeeping and spacecraft operations, performed on a systematic basis, e.g., 15 min for change of personnel on watch, two 15-min periods per day for flight plan updating, etc.

4.5 EXTRAVEHICULAR ACTIVITIES

Unique and critical aspects of crew participation in the OTES mission are primarily performed during the EVA periods. Potential EVA tasks would include mirror segment removal, mirror recoating, film retrieval, equipment module removal and replacement, and various maintenance tasks. The feasibility of using crewmen to accomplish specific tasks within the system as presently defined or with minor modifications for their accommodation is considered in this subsection. These tasks are considered within the operational constraints and guidelines of the system, with primary consideration given to EVA capabilities of the crewmen, time available for EVA tasks, visual work environment, and crew-safety factors.

4.5.1 EVA Ground Rules

Current AAP plans with respect to crew and spacecraft operations minimize requirements for EVA where feasible alternatives are available. Though EVA may appear feasible, a considered decision must be made before committing to EVA operations; key considerations in planning for EVA are as follows:

- EVA will be performed after exit through the airlock module (AM).
- EVA will not be programmed unless all three crewmen are available for duty. One or two crewmen will be responsible for work functions. A standby astronaut will maintain visual contact with the active EVA or will assist in the work function, as required. The standby astronaut will also be available for rescue operations.
- No EVA will be performed during hazardous space environments, e.g., meteoroid storm or radiation anomaly.
- No EVA will be performed during the first three orbits or the last 24 hr prior to reentry.
- EVA will be carried out in both daylight and darkness; lighting will be required for night operations.
- The astronaut performing EVA must be provided with restraining equipment to prevent personal injury or damage to the spacesuit or spacecraft.

- For data retrieval, it may be necessary to provide additional bails or rings along the path to the appropriate experiment carrier.
- During EVA, the spacecraft must be in a free-drift mode with all reaction control and propulsion firing modes disabled.

In support of possible rescue operations, the following procedures apply:

- One astronaut will be stationed such that the primary EVA astronaut is in view. This could require leaving the spacecraft or remaining in the airlock.
- The EVA hatch will remain open during the entire EVA.
- The backup astronaut will keep at hand the appropriate rescue device(s) for restraining or protecting an injured primary EVA astronaut.
- For two-man EVA operations, work will progress alternatively such that one astronaut (where possible) works while the other observes.
- All crewmen will be in conference communications during EVA.

4.5.2 EVA Capabilities

The proposed AAP environmental control system, as currently conceived, provides for a 65-mnhr EVA capacity between resupply visits. The EVA is performed from a 60-ft umbilical. Egress and ingress to the spacecraft is through the airlock module, which has the capacity for 15 repressurizations. The buddy system is employed to enhance safety; two crewmen are always in full suits and on umbilical lines although only one may be required to perform a given EVA task. Total EVA is cut to a period of 32.5 hr; hence, operations and tasks must be designed for completion within this available period. Each excursion requires a substantial time period for preparation, e.g., donning suit, system checkout, airlock depressurization, and pressurization. A typical EVA task analysis supplied by NASA indicates that a period of 2.5 hr is required for donning the suit and egress, preceded by a 1-hr period during which EVA equipment and spacecraft systems are checked out.

Detailed analysis of the preparation for EVA has been conducted to support the derivation of these values. The activities involve preparation for egress; setup and

checkout of onboard systems; preparation of EVA and personal equipment; donning and checkout of undergarment and pressure suit; and gathering tether lines, tools, and transport devices. Following ingress, a nominal time of 1.5 hr is occupied in depressurization, doffing the suit, stowing, etc., and must be included in the block of allocated experiment time. Considering the total time requirements involved and the present capacity of the life-support and pressurization systems, about eight excursions can be made (each taking about 8 hr of astronaut time) with an EVA period of about 3 hr. It is possible that only part of the EVA manhours in each excursion would be applied to work, particularly if the alternate work/watch concept is applied. Resupply capabilities will relax some of these restrictions, and the experience gained during cluster operations may provide for considerable optimization of EVA tasks. The light/dark cycle of the 260-nm orbit allows from 54 to 39 min, respectively, for EVA operations planning. Within the light period, the position of the sun relative to work areas must be considered; e.g., filters may be required. During dark periods, artificial illumination will be necessary, so procedures must be established for setting up lights, tethers and rescue aids, and work supports.

Table 4-4 summarizes a typical EVA for AAP film retrieval. In this case, the task time line covers egress from the airlock and retrieval of three film cassettes from the telescope equipment modules, and requires an estimated 170 min of EVA (about one entire EVA period). The analysis uses the capability to rotate the telescope in the roll ring of the ATM to afford access to three experiment modules. Alternatively, the crew could transfer positions to each retrieval site but this would require additional time.

4.5.3 Overall Mission Time Line

An overall mission time-line allocation can be calculated for a hypothetical 1-yr mission with 90-day resupply. It is assumed that 25 percent of the crew time is available for experiments on a typical AAP cluster mission. The resupply concept must include the premise that 2 days at the beginning and end of each cycle are required for crew transfer and familiarization, thus reducing total available time proportionately. A single excursion takes about 8 hr; each resupply cycle is assumed

Table 4-4
EVA FILM RETRIEVAL TIME-LINE TASK DESCRIPTION

Task No.	Task or Event	Crew Activity Time (%)	Equipment Interface	Comments	Estimated Time (min)	
					Task	Cumulative
1	Depressurize	A:100 B:100 C:100		Requires two Crewmen	4	4
2	Open LM airlock hatch; astronaut A egresses and attaches tether outside hatch		LM hatch and tethers	Astronaut A performs EVA using umbilical Astronaut B performs IVA using suit on umbilical	5	9
3	IVA astronaut B hands out experiment film cassettes with safety line attached; EVA astronaut A attaches cassettes to suit harness	A:100 B:100 C:100	Suit harness, Velcro, and safety lines	Astronaut C monitors activity and suit parameters in LM Astronaut B performs IVA in depressurized LM	6	15
4	Astronaut A releases tethers and translates to ATM hatch/work area, positions self, and attaches tethers and engages foot in stirrup	A:100 B:100 C:100	Tethers, handrail, stirrup, and tether attachment	Tethers - refers to crewmen device for securing and positioning in work space	6	21
5	Astronaut A removes film cassettes and places them in temporary storage area securing cassette safety lines to ATM rack structure	A:100 B: 50 C: 50	Safety lines	Safety lines (personal and equipment lines) prevent equipment and astronaut floating away	4	25
6	Astronaut A opens experiment access hatch and relocates and adjusts tethers for access to cameras	A:100 B: 50 C: 50	LM hatch, tethers, and tether attachments		3	28
7	Astronaut A removes and attaches auxiliary light inside ATM	A:100 B: 50 C: 5	Auxiliary lights, and attachment	Auxiliary light may be permanently positioned on rack	2	30
8	Astronaut A removes "fresh" film cassette from temporary storage and places it inside rack for cassette removal/replacement procedure	A:100 B: 50 C: 5	Storage point and Velcro		1	31
9	Astronaut A attaches safety line to exposed film cassette, releases latch(s) while holding cassette, removes expended cassette, and places it in removal/replace (R/R) storage area of the rack	A:100 B: 50 C: 5			2	33
10	Astronaut A removes fresh cassette from R/R storage, places it on camera, secures cassette latches, and requests astronaut B to verify camera will operate	A:100 B: 50 C: 5	Camera C/D (S)	Complete remove-and-replace procedure as fast as possible to reduce contamination of camera	3	36

Table 4-4 (Cont)

Task No.	Task or Event	Crew Activity Time (%)	Equipment Interface	Comments	Estimated Time (min)	
					Task	Cumulative
11	Astronaut A removes cassette safety line from fresh cassette and attaches it to expended cassette, removes other cassette, places expended cassette in temporary storage outside experiment module, and removes auxiliary light and attaches it to suit harness	A:100 B: 50 C: 5	Safety lines, auxiliary lights, and auxiliary equipment	Assume all cassettes have safety lines attached in all sequences	5	41
12	Astronaut A releases and relocates tethers allowing him to be in position for closing and securing access hatch; closes and secures hatch	A:100 B: 50 C: 5	Tethers, access hatch, and latch device		3	44
13	Astronaut A positions self for OTES rotation	A:100 B: 50 C: 5			1	45
14	Astronaut A requests astronaut B to rotate OTES until experiment module access hatch is in position; verifies rotation and positioning with astronaut B	A:100 B:100 C: 5			1	46
15	Astronaut A repeats tasks 5 through 12 for removal of cassette				23	69
16	After completion of cassette removal, astronaut A secures the expended cassettes to his suit harness, removes the cassette safety lines from storage attachment, and attaches them to his suit	A:100 B: 50 C: 5			6	75
17	Astronaut A releases his personal tethers, secures them, and translates to LM ingress hatch	A:100 B:100 C:100			3	78
18	Astronaut A attaches tethers, removes cassette from suit harness, and hands it inside to astronaut B	A:100 B:100 C:100	Tethers and cassettes		3	81
19	Astronaut B stows first cassette in temporary storage, releases the safety line, attaches the safety line to second cassette, releases the cassette, and passes it outside to astronaut A who attaches it to his suit harness	A:100 B:100 C:100	Cassette, safety lines and suit harness	Cassette safety lines are attached to cassettes and anchored during all sequences	3	84
20	Astronauts A and B repeat steps 18 and 19 for other cassettes				6	90
21	Astronaut A repeats steps 4 through 19 for removal and replacement of other film cassettes				69	159
22	Astronaut A removes cassette and hands it to astronaut B, releases his tethers, enters the airlock hatch, and seals the hatch	A:100 B:100 C:100	Tethers and hand holds or hand rail		5	164
23	Begin post-EVA procedures					

to provide for eight EVA excursions. Each of the four resupply cycles offers 86 days during which experiments can be performed, most probably within a single working day, unless event scheduling requires reprogramming of crew duty cycles. Each resupply cycle offers eight EVA periods, which leaves 78 working days during which about 25 percent of the crew time is available to support experiment functions.

Figure 4-1 shows this time-task allocation; a more detailed time line-analysis for the film retrieval activities is presented in Table 4-4.

4.5.4 Extravehicular Illumination Environment

Illumination encountered during EVA is of critical importance to astronaut performance capabilities. Whereas current manned space-flight operations have involved relatively small spacecraft (Gemini/Agena target vehicle) and constant surface areas, the cluster configuration of the S-IVB/docking-adaptor/LM(OTES)/CSM/mirror-recoating facility is a composite of irregular shapes. Characteristics of solar illumination (unattended by the atmosphere) and the configuration peculiarities may result in marked differences from flight experience to date.

Direct solar illumination in space is intense (1.24×10^4 lu/ft²), demonstrates different spectral characteristics, and will be collimated to about 32 min of arc. Uniform illumination caused by scattering in the earth's atmosphere will modify the visual field considerably, with extremes between high-intensity reflected light and shadows of the cluster. In illuminated areas, the astronaut may require filters to reduce blinding glare that scatters off the irregular surfaces of the cluster as the astronaut EVA moves from point to point on the OTES. Functional characteristics of the astronaut's visual system also will change in sensitivity as a result of the radical contrasts between shadow and high-intensity light.

Thus, the illumination environment of the flight system, with respect to EVA, must be considered in the design of equipment and establishment of operating techniques for space operations. These effects and capabilities for simulation of solar illumination in ground test facilities are discussed in more detail in subsection 4.7.

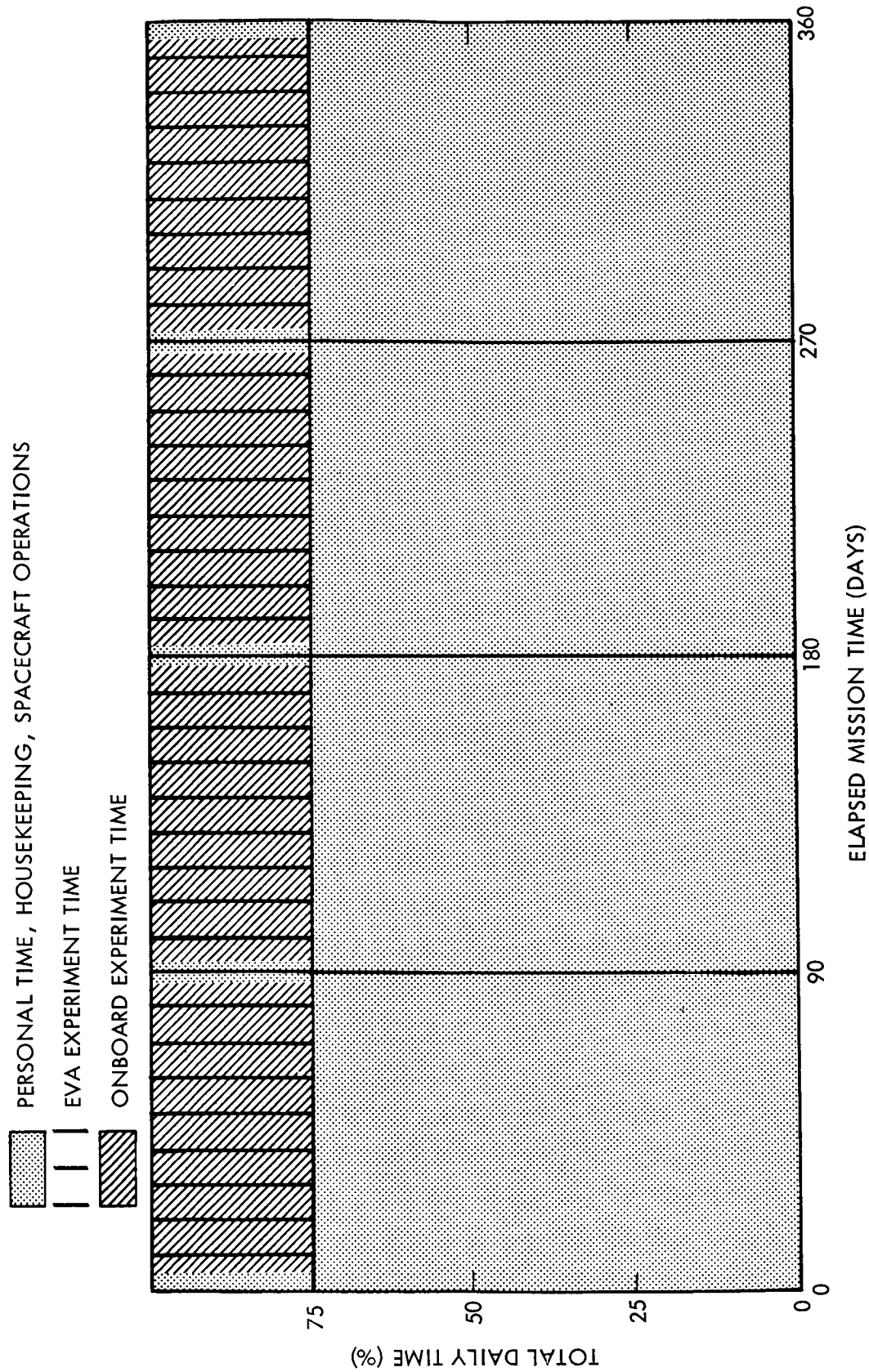


Fig. 4-1 Overall Mission Time Allocation for Primary Mission Functions

4.5.5 EVA Work Systems Concept

Preliminary concepts for crew participation generated during the Phase I study effort appear to be compatible with the current 2-m OTES design approach. The three areas affected are those operations associated with telescope erection, mirror transfer and recoating, and maintenance.

One of the major changes is elimination of crew EVA activities by use of automatic telescope erection. A primary reason for this is the relatively short, useful EVA time that is available. Revised estimates of crew activities show long preparation times for EVA and rather extensive requirements for tethers; this would tend to reduce crew participation to relatively simple operations. Hence, since the erection process would be essentially automated, it appears wasteful to employ EVA time for this application.

The basic concept for extracting a mirror segment and transferring the segment to the Orbital Mirror Recoating Facility (OMRF) remains unchanged from the earlier studies. However, it appears that the addition of a transfer case would eliminate possible problem areas. This case would serve to protect the mirror from physical or thermal damage during transfer, protect the astronaut from injury on mirror edges, and prevent blinding glares from the mirror surface. The mirror would be guided into the transfer case on a set of rails and would be locked in place during transfer. The case would be designed to mate with the mirror access hatch on the telescope barrel and on the recoating facility. A system would be erected by the crew for tethering and guiding the mirror in transport. One EVA would be used for erection of the transfer unit and the second for mirror movement.

Details of the mirror recoating process are still under consideration. The present concept is for the OMRF to be docked to the docking adapter adjacent to (90 deg from) the OTES spacecraft. Once the mirror is transferred to the recoating facility, processing would be accomplished and monitored by the crew via a viewport/control-panel

system compatible with processing requirements. The EVA would include mirror transport and preparation of the segment for controlled reprocessing.

The use of astronaut EVA for maintenance of the spacecraft and experimental systems is a distinct possibility. Conceptually, the maintenance function would be quite similar to operations associated with mirror segment removal and replacement; i. e. , it would involve removal and replacement of plugin modules. For a 1-yr orbit mission, the time for EVA modular replacement operations must be traded off with other EVA applications such as film retrieval. Improved resupply capability, such as increased storage space on the docked cluster configuration, would increase considerably the prospects for EVA.

4.6 CREW SAFETY

A major consideration in the design of the OTES system is maximum crew safety for all phases of the mission. NASA is particularly concerned about safety with respect to EVA; the following list summarizes identifiable factors that would result in EVA abort or crew rescue operations:

- Personnel Equipment Malfunction
 - Communications
 - Life Support
 - Tethering
 - Auxiliary equipment (lighting, safety)
 - Suit
- System Equipment Malfunctions
 - Internal vehicle subsystems (SCS, ACS, power)
 - Hatches and ports
- Environmental Hazards
 - Meteoroid punctures (suit, back, etc.)
 - Solar irradiation (visual damage)
 - Radiation (solar activity)

- Vehicle Hazards
Contamination
Crew entrapment or separation from the vehicle
Personal equipment failures
- Physiological Hazards
Fatigue
Hyperventilation
Heat shock
Carbon dioxide poisoning

Table 4-5 is a list of hazardous conditions, methods of hazard reduction, and emergency procedures. Conditions under which EVA rescue would be conducted are as follows:

- At command of the astronaut operating outside of the spacecraft
- After a reasonable period following loss of communications, if visual status information cannot be obtained
- After a reasonable period following loss of oral and visual contact and if monitored physiological data on the astronaut indicates satisfactory physical condition
- Upon loss of oral, visual, and physiological data
- After a reasonable time if the astronaut will not reenter the CM on command
- Immediately after out-of-tolerance physiological data are detected

Several techniques are possible for crew rescue. These are as follows:

- Direct retrieval by the standby astronaut for EVA, using emergency oxygen supply bottle
- Retrieval with a pressure box that encloses the astronaut completely
- Remote retrieval through tether or umbilical connections
- Use of a stowable tubular extendible member (STEM) – a collapsible tape reel used as a rigid rod for astronaut retrieval

Table 4-5

HAZARDOUS CONDITIONS, AND METHODS FOR HAZARD REDUCTION AND EMERGENCY PROCEDURES

Condition	Method of Hazard Reduction	Emergency Procedure
Environmental		
Solar radiation	Use visor and shielding afforded by structure	Wait for blindness to pass or wait for rescue
Particle radiation	Avoid regions of high flux density	Withdraw to craft
Micrometeoroid flux	Use shielding afforded by structures	Return to craft
Vacuum	Suit maintenance and checkout	Use EOS and/or crew rescue bag
Spacecraft discharge	Avoid attitude changes or jettisoning waste during EVA	Remove particles from face plate
Electrical potential	Provide electrical path among structures touched by astronaut	(Unknown)
Garment/Life Support		
Tears	Perform maintenance, checkout, and short missions; avoid sharp objects and narrow passages	Rescue if trapped; avoid self-release
Condensation on face plate	Short missions and frequent rest	Rest and wait for plate to clear; return to craft
Loss of communication	Check out communications frequently	Return to craft
Crew Morphology/Health		
Nausea	Select and train crew; control diet; avoid fatigue and return to craft at first symptom	Reschedule EVA so man not required (to be supplied)
Vertigo	Avoid sudden movements; train crew	Rest or rescue
Fatigue	Train crew; rest frequently	Rest; return to craft
Bends	Perform denitrogenization procedure; slow change in pressure	Increase pressure then reduce pressure slowly
Heat exhaustion	Monitor physiological variables; perform short missions; rest	Rescue
Operating Procedures		
Tangled umbilical	Train crew; monitoring of procedure by standby astronaut	Stop movement; allow standby to free lines
Caught between moving structures	Communications with other crewmen; train crew; improve design to avoid EVA near moving structures	Rescue; standard medical treatment

4.7 TRAINING REQUIREMENTS AND FACILITIES

A complete list of astronaut training requirements and facilities will be generated when OTES experiments and experiment-implementation requirements have been defined. For the present, however, it is possible to identify in general terms some of the personnel functions and various items of equipment that will be required. From these, it is possible to identify some of the skills that will be required, and the facilities needed for skills development.

Training requirements will be derived from an analysis of crew functions and skill requirements, identification of training modes, and an assessment of required training time. Crew functions may be identified by examining the mission or experiment functions; these functions, in turn, can be analyzed for specific skill and knowledge requirements. The following training techniques may be used to provide this capability:

- Familiarization. Training to make the operator aware of the goals, techniques, and equipment needed to perform the experiment or task (training manuals, visual aids, mockups, and prototypes)
- Part-task training. Training to acquire skill in the performance of one set of procedures and functions under realistic operating conditions
- Full-task training. An extension of part-task training in which part tasks are integrated in correct sequence
- Mission simulation. Overall mission training using full-scale simulators of actual spacecraft or experiment equipment

In general, the skills required for inflight experiments can be identified as follows:

- Scientific and technical background for proper experiment performance
- Ability to recognize, identify, and act with respect to significant natural phenomena
- Specific skills required for the operation of scientific devices and equipment, including discrete switching procedures, continuous control, and contingency operations

Training sequences that are generally applicable to OTES involve the following steps:

- Academic training in relevant scientific and technical areas
- Ground training involving the operation of similar equipment under dynamic conditions
- Procedural training on part tasks or integrated training devices
- Training under simulated mission conditions

A typical training schedule for an inflight experiment would be as follows:

- 20 hr – discussion of experiment goals, probable conditions, and choices of actions involved
- 40 hr – specific instruction on details of mission experiment events
- 40 hr – operation of part and full tasks on simulated mission equipment

Training equipment will include the following items:

- Training aids, e. g. , charts, maps, texts, photos, movies, sample data and materials, functional mockups
- Part-task trainers that duplicate functional, physical, and dynamic characteristics of actual equipment
- Mission simulation wherein a complete, integrated static and dynamic representation of vehicle systems is used to duplicate mission phases

OTES training must be planned such that the total number of training hours are compatible with available astronaut time, and so that the sequences and timing factors are compatible with those during which the desired skills are applied. An integrated set of training facilities would help to optimize training effectiveness.

The training course must be developed in recognition of common elements among mission tasks, required skills, and knowledge. Certain skills can be acquired by the astronaut prior to the time-critical portions of the OTES Program, and only brief refreshers may be needed. Mission effectiveness and training time requirements can benefit from careful operations planning and engineering of tasks and equipment to simplify operations to the maximum reasonable extent. Overall training must

recognize commonalities which, once a basic level of skill is reached, can be generalized to apply to several experiments and hence reduce training time to essential new-skill training.

4.7.1 Training Time Requirements

Only an order of magnitude estimate of training time requirements can be made at the present time because insufficient detail is available concerning OTES experiments. However, prior training on other AAP cluster experiments may demonstrate commonality and thus reduce training needs. In addition, crew selection specifics will tend to modify training plans, and mission experience gained on Clusters A and B will influence later training requirements.

Estimates for ATM Cluster A missions run to a total of 1,200 to 1,300 hr (430 hr/man). This can be modified by grouping so as to divide the training among crew members. In the Cluster B missions, training time estimates range from 450 to 760 hr/man. It is probable that these times can be reduced by design of the training program, grouping and dividing of skills among crew members, and crew selection.

4.7.2 Training Facilities Requirements

Many of the facilities required for OTES mission training will be available from the AAP and Apollo training program. No new facilities will be required for academic, scientific, and technical instruction although course curriculums and training aids will be needed. Existing equipment will be readily available for ground training in various technical areas. Functional mockups and prototype hardware will be needed for specific training in detailed operation of each experiment. These represent dynamic simulation gear in a variety of configurations. For EVA training, full-scale mockups will be necessary, for example, to develop skills in mirror transfer and recoating functions. Functional structural models of the OTES, mirrors, transfer cases, guide systems, and the Orbital Mirror Recoating Facility would be provided in existing facilities for this type of training.

Lockheed has developed simulation techniques to evaluate manned systems performance in situations involving major crew participation. These techniques are directly applicable to OTES and involve such facilities as: (1) reduced gravity simulation, (2) control display simulation for ATM orbital telescope operations, and (3) solar illumination simulation. These facilities are described in the following paragraphs.

Reduced Gravity Simulation. The LMSC Biotechnology organization has three facilities available for simulation of reduced gravity. These facilities are used to help determine man's EVA capabilities. Two of these use mechanical suspension techniques; the third, under water neutral-buoyancy techniques.

The ZEROG II was the first zero-gravity simulator to be developed by LMSC and is particularly useful for EVA simulations requiring little or no free space translation. The device is a very low-friction simulator and has proved quite effective in simulating zero- or partial-g loading. The device consists of two counterbalanced beams, a yoke, and a personnel cradle - all gimbaled on ball bearings. It affords six degrees of freedom and can be maneuvered within a 32-ft-diam. circle horizontally and 11-ft vertically. Filtered shop air or bottled air can be supplied to a pressure-suited subject; the cradle also will accommodate a backpack. Figure 4-2 illustrates an application of the ZEROG II for evaluation of hand rail effectiveness on the LM/ATM configuration. Figure 4-3 shows the simulator used for evaluation of the ATM film-retrieval task.

LUNARG is a more recent simulation device development and is used for simulating reduced gravity effects on the moon. The unit relieves five-sixths of a man's earth weight by means of nine negator spring motors integrated with a cable-reeling system. The negator motors apply a constant torque to a shaft (cable reel) regardless of the rotational displacement of the shaft. Each motor applies about 30 lb of tension to the cable supporting the man. Individual motors may be added or subtracted to vary the cable tension in 30-lb increments. A 180-lb man would require five motors to simulate the one-sixth gravity condition. The effect can be achieved over a spatial

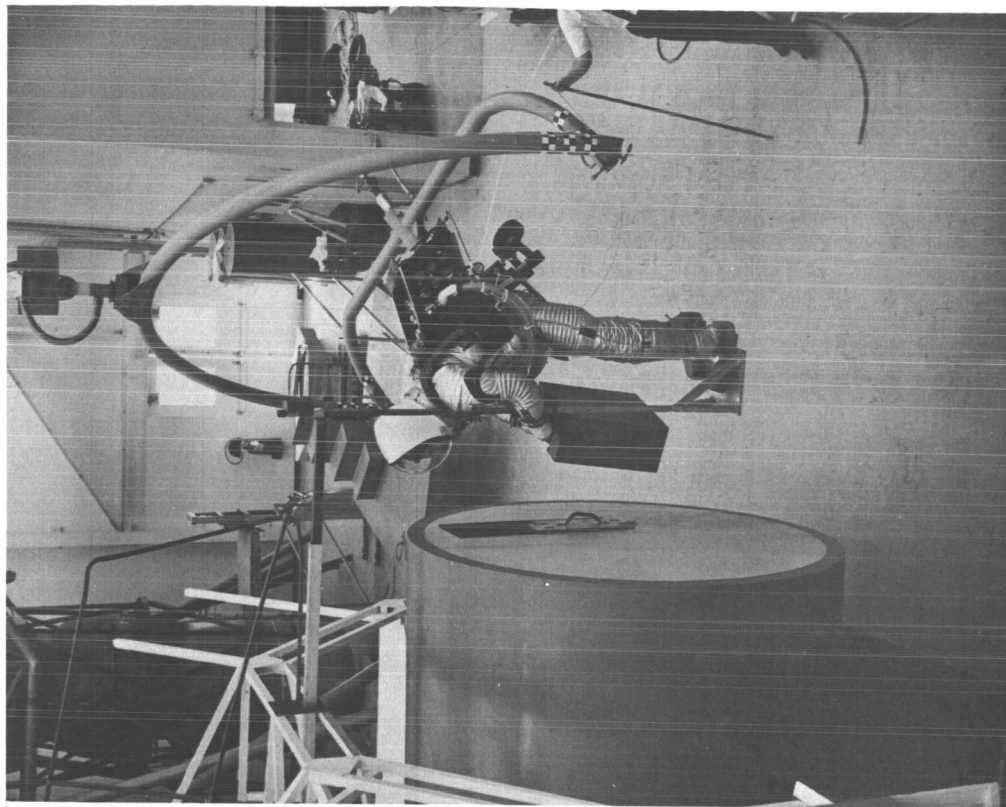


Fig. 4-3 ATM Film Retrieval

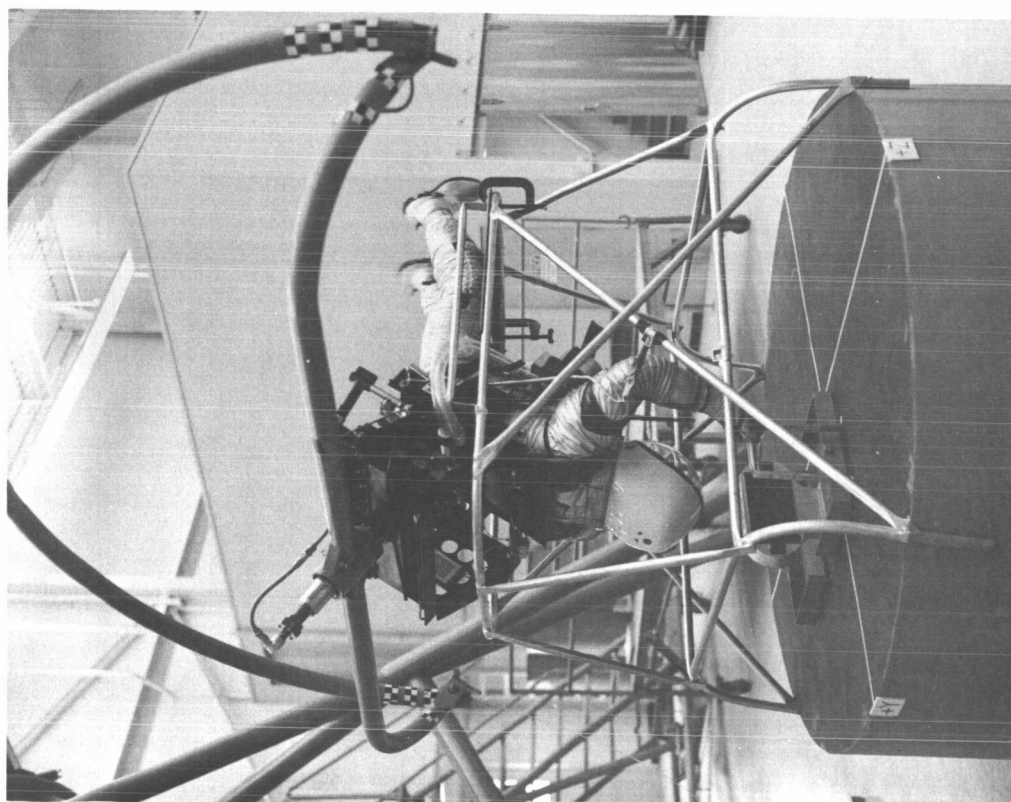


Fig. 4-2 ZEROG LM Handrail

rectangle 40 ft long, 16 ft wide, and 10 ft high. This is accomplished by allowing the spring motor to travel with the man forward, backward, and laterally, and up and down in an x-y plotter manner, as shown in Fig. 4-4. The system is designed for operation either at ambient conditions or within the confines of an altitude chamber. The LUNARG is also adaptable to zero-gravity simulation. Because of its low inertia characteristics, as compared with ZEROG II, LUNARG would be applicable to EVA simulations involving extensive free-space translation. Since LUNARG is compatible with vacuum-chamber operations, vacuum environmental conditions can be included as an additional EVA simulation factor.

In recent years, however, water immersion has been demonstrated as an effective technique for simulating some of the physical characteristics of the weightless state encountered in orbital flight and for assessing man's ability to perform useful functions under zero gravity. Weightlessness is approximated by a state of neutral buoyancy. The mass and volumetric displacement of the subject and the objects that he uses are manipulated so that their effective density is equivalent to the density of the water. Thus, they are able to move or be moved within six degrees of freedom in response to applied forces in the fluid medium. Two LMSC facilities are available for neutral-buoyancy simulations - the Ocean Biotechnology Research Tank and the Biotechnology Neutral Buoyancy Simulation Facility (Fig. 4-5).

The Ocean Biotechnology Research Tank originally was developed for evaluation of manned underwater tasks. It comprises a cylindrical tank 15 ft in diameter and 12 ft high, and is equipped with a service and access stairway and poolside deck at the water entry level (about 10 ft above the floor). Four 12-in.-diam. viewing ports are equally spaced around the tank perimeter for visual or photographic monitoring. Water clarity is maintained by a pump and filtration system which recirculates one full tank of water in less than 8 hr. A 0.5-ton-capacity monorail hoist is installed above the tank to permit installation of full-scale mockups of spacecraft sections or modules. A two-way lock recompression chamber is available to permit immediate treatment of air embolism, if required. Subjects used in simulation tests wear open-or closed-circuit SCUBA and air-pressurized full-pressure suits.

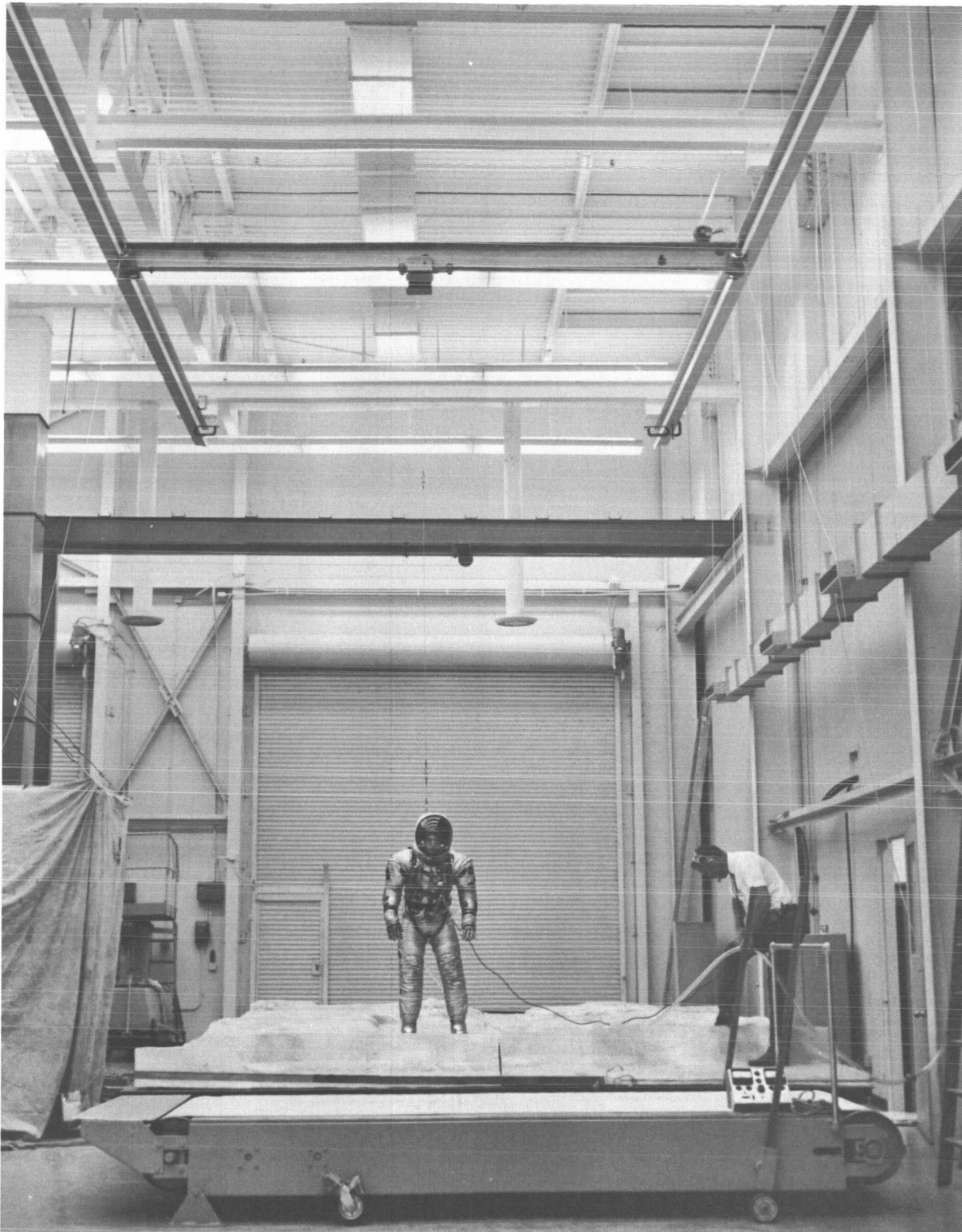


Fig. 4-4 LUNARG Chamber



- TWO-MAN DECOMPRESSION CHAMBER
- FACILITY MODIFICATIONS
- SELECTION AND TRAINING OF OPERATIONAL CREWS
- PRETEST OPERATIONS
- CREW STATION SIMULATIONS IN SMALL TANK
- EVA AND MAJOR CREW TASKS IN LARGE FACILITY

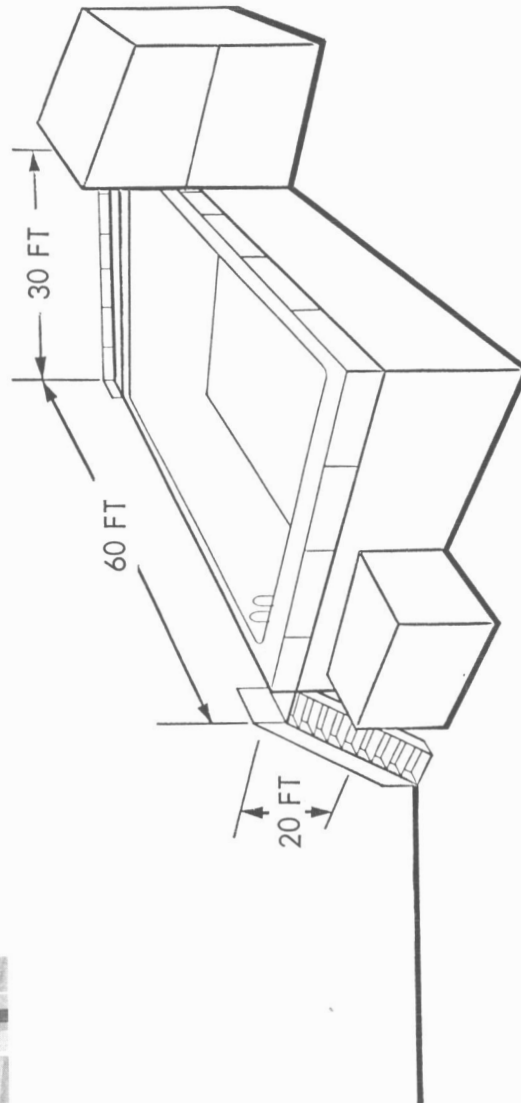


Fig. 4-5 Neutral Buoyancy Simulation Facilities

The Neutral Buoyancy Weightlessness Simulation Facility was initially developed for hydrostatic test tank applications with the Lockheed Deep Quest submersible vehicle and was completed in early 1967. This tank will have essentially the same capabilities as the Ocean Research Tank, but will provide a much greater volume for evaluation of full scale mockups of major space-station modules.

Control/Display Simulation - Orbital Telescope Operation. An independent development program being conducted by the LMSC Biotechnology Organization is obtaining feasibility data and system design criteria for astronaut control of orbital telescope operations and experiments. The candidate astronaut tasks of primary interest in this program are those associated with the control functions of attaining and maintaining the fine pointing accuracy required of a manned orbiting telescope. Secondary consideration is given to astronaut tasks associated with gross vehicle orientation prior to commencing a fine pointing operation, and with procedures for activating and controlling the astronomical data acquisition systems after required fine pointing accuracy is attained. The objectives are accomplished through man-in-the-loop simulation of the control/display dynamic relationships. These dynamic relationships are modeled from descriptions of the control-system configurations. Test data are then acquired with the manual tasks in the control loop.

The system block diagram is shown in Fig. 4-6. Manual control outputs from the crew station, shown in Fig. 4-7, are accepted by an analog computer that models selected control-system configurations and, in turn, controls the simulation equipment that creates the crew-station display dynamics. Thus, test operations are conducted at a simulated real-time rate with respect to control response and information display. Control system configurations are modeled to include selected fundamental control system concepts (such as aiming axes coupling and manually controllable or fixed slewing rates) and selected dynamic parameter values within the selected control-system concepts (such as maximum and minimum slewing rates and response lags). Errors and response times for manual control tasks are measured during simulated operations. Thus, an engineering development and evaluation tool is provided to test the effectiveness of a manual control element within a selected control

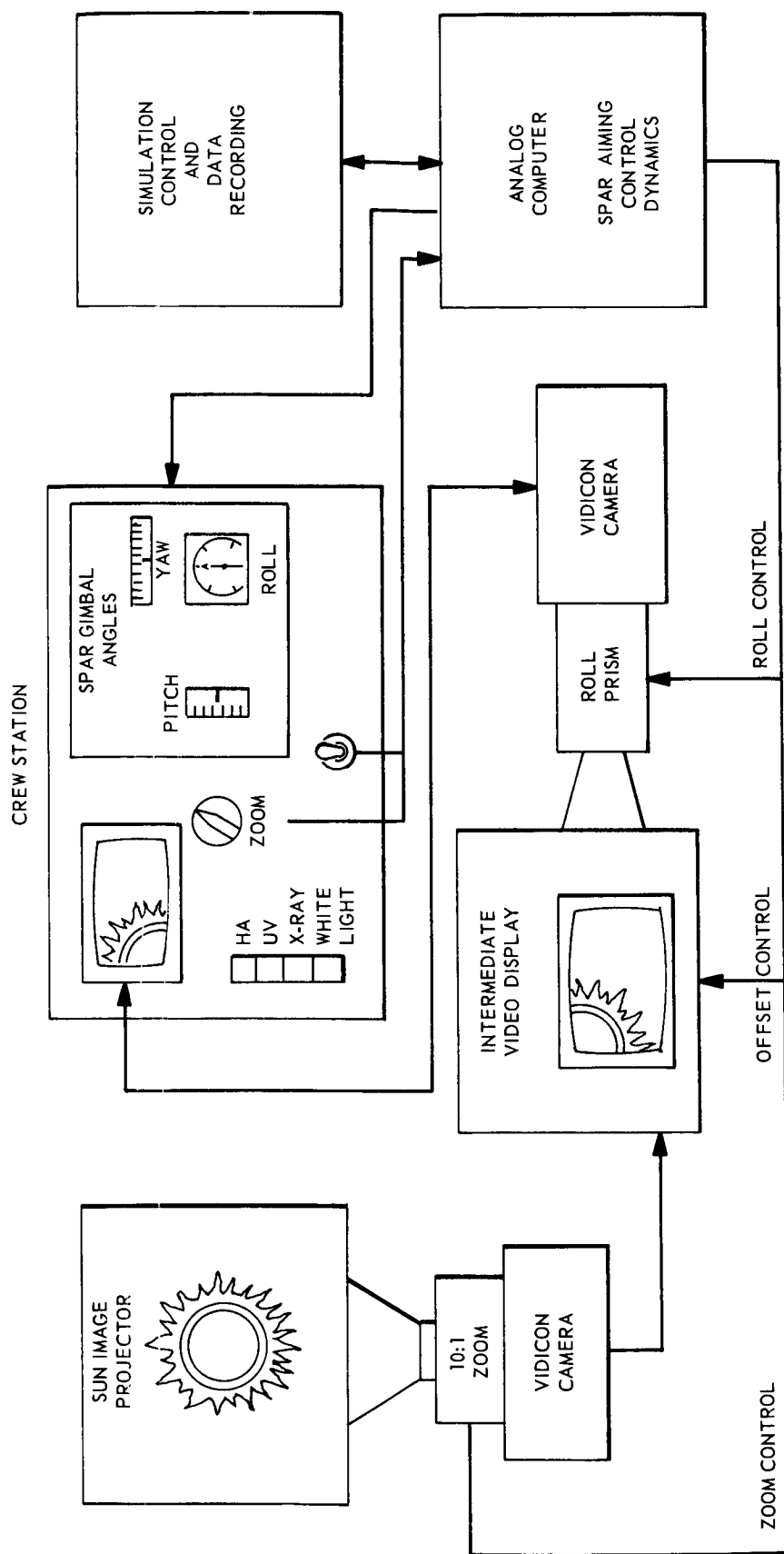


Fig. 4-6 ATM Control/Display Simulator

ATM POINTING CONTROL SYSTEM DYNAMICS AND CONTROL/DISPLAY REQUIREMENTS ARE THE SUBJECT OF A CURRENT LMSC-FUNDED SIMULATION PROGRAM. ITS OBJECTIVES ARE TO DETERMINE THE TIME REQUIRED TO ACCOMPLISH MANUAL SUN-CENTERING, SLEWING AND POINTING USING VARIOUS SOLAR IMAGE PRESENTATIONS AND CONTROL MODES, AND TO CONDUCT EMPIRICAL EVALUATIONS OF CANDIDATE CONTROL/DISPLAY CONFIGURATIONS

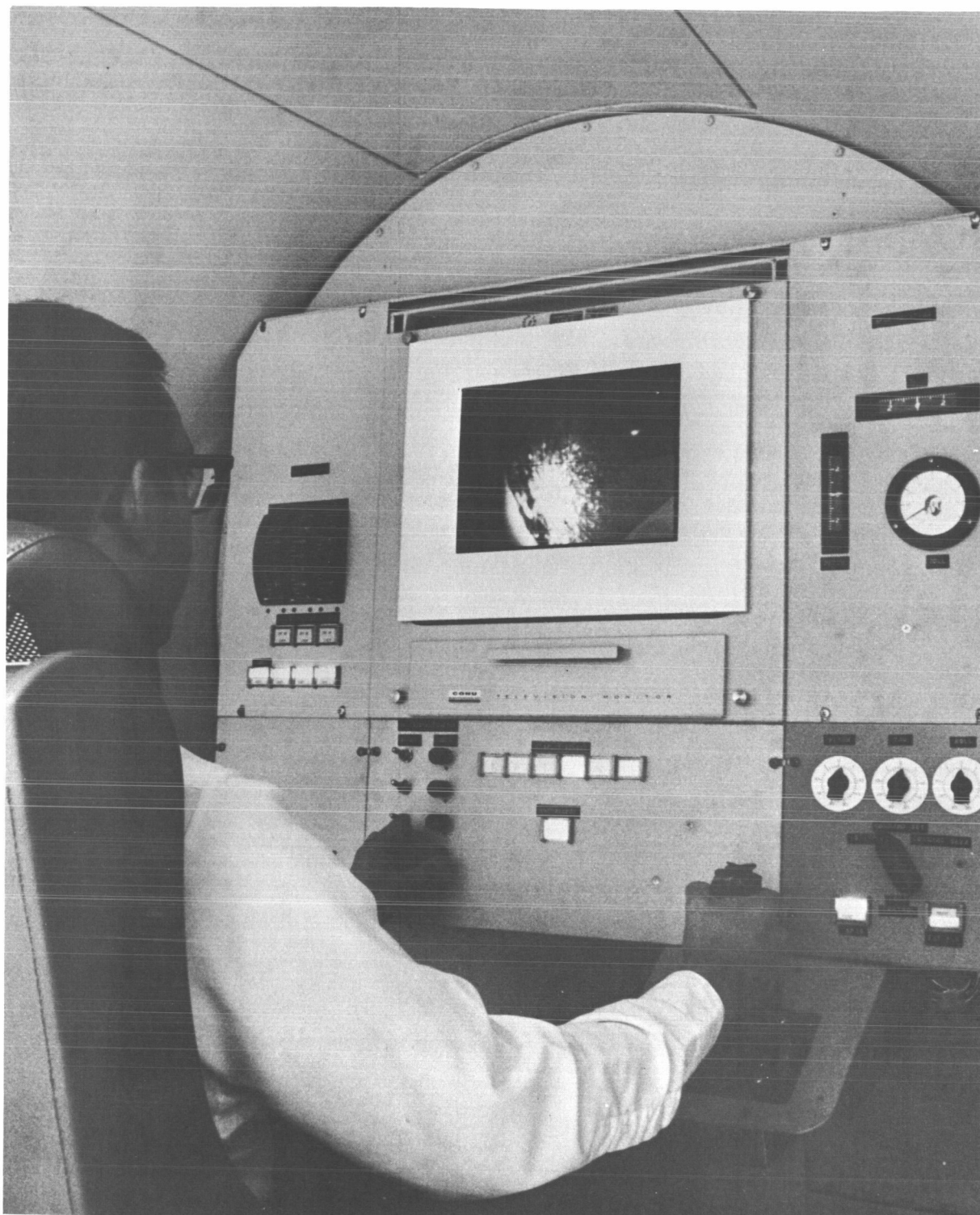


Fig. 4-7 ATM Crew-Station Simulation

system configuration. These simulation techniques might be adaptable for evaluation of crew-station control/display components and arrangements, and for use in training simulations, if more realistic crew-station physical configurations were provided, although such fidelity is not necessary to satisfy the engineering development and evaluation objectives of this program.

The current telescope configuration being simulated and tested is the solar Apollo Telescope Mount. The primary astronaut display is a cathode-ray tube upon which video outputs are displayed from sensors or from cameras boresighted with the optical aiming axis of the telescope. Based upon simulated solar images in selected electromagnetic bands appearing on this display, the telescope is manually aimed in the direction of a solar event at which it is desired to bring a selected sensor to bear. The tightest manual aiming requirement in the present testing is ± 2.5 sec of arc. Stabilization after attaining the desired aiming accuracy is automatic, although provision is being made to introduce the effects of crew induced perturbations on both the aiming operation and subsequent stabilization. This application has offset (pitch and yaw), roll, and zoom degrees of freedom reflected in the solar disc image dynamics on the primary display and in the readouts on the secondary display that include roll, pitch, and yaw gimbal angles.

This simulator could easily be transitioned into other orbital telescope applications, as long as a cathode-ray-tube display or the aiming field image is satisfactory. Stimulus material could be developed for earth, lunar, or stellar observation. Control-system configuration changes reflected in the control-display dynamic interrelationships are accomplished through analog computer mechanizations. Other than stimulus material changes, no alteration in the dynamic display generation equipment would be required as this equipment has been designed to accommodate the full range of manned orbital telescope aiming problems.

Solar Illumination Simulation. Characteristics of solar illumination unattenuated by the atmosphere are important factors for designing equipment and appropriate

operational techniques for docking, EVA, and observations. The most outstanding feature is the high illumination level from solar radiation outside the earth's atmosphere (approximately 1.24×10^4 lu/ft² – about 50 percent greater than that encountered on earth). Other major sources of illumination consist of earth- and moon-reflected sunlight, and from each of the vehicles of the cluster. The portion of the solar spectrum of interest lies in the visible range of 0.38 to 0.75 nanometers. The hue of the earth's surface observed from space will differ from the hue observed from within the earth's atmosphere, and will be characterized in the violet to yellow-green range but will occur throughout the visible spectrum.

In space, blue will be more dominant than green or red – this must be taken into account in color-coding instrument displays and equipment used in space under lighting conditions that alternate between solar and artificial illumination. Since the earth-sun (or moon-sun) distance is so great, the light rays emanating from points on the sun's surface are essentially parallel when they reach the earth and the moon. The only beam spread (decollimation) encountered is due to the angular size of the sun – about 32 min of arc at the earth-sun distance. This relative collimation, combined with the absence of a scattering media, eliminates fill lighting; therefore, there is no visibility in shadowed areas. Another phenomena attributable to the great distance is the apparent uniformity of illumination on a plane oriented at any angle to the incident sunlight. The variation of intensity over these planes is well below man's brightness discrimination threshold, and it is generally assumed that sunlight is 100 percent uniform in intensity over the illuminated surfaces observed by the astronaut regardless of sunlight incidence angle or size of the illuminated area.

Collimation of sunlight results in significant differences in visually sensed surface characteristics such as shape, luminescence, and even color, as an angle of incidence varies from 0 to 90 deg. The differences are direct functions of this angle of incidence and of the angle from which the surface is viewed.

Spacecraft will experience the effect of reflected sunlight from the lunar surface and the earth. Although intensity levels are far less than those from direct sunlight,

there is adequate illumination for performance of some tasks. The decollimation of light incident on a spacecraft in a 200-nm earth orbit would be about ± 70 deg for earthlight and ± 0.5 deg for moonlight. Uniformity of illumination for both incident earthlight and moonlight throughout the earth-moon space is such that, as in the case of sunlight, variations would be imperceptible. These sources of illumination usually act in combination.

Initially, the orbit position establishes the number and types of possible sources of illumination and certain basic angular relationships such as the angle between the incoming target illumination and the line of sight. In general, the brightness of a surface that reflects light is a function of the area, shape, albedo, and specularity of the surface and incident illumination. Thus, target brightness is a direct function of both illumination angle of incidence and viewing angle. This functional relationship is confined to angles in the range of 0 to 180 deg, depending upon the relative position of the light sources, target, and rendezvous vehicles as defined by the orbital position of the vehicles.

"Glare" is the effect of stray light from extremely bright surfaces and tends to veil the visual field by superimposition of brightness over the line-of-sight field of view. The quantitative effects of glare are not fully known. Research, such as that by Dr. Richard Haines of Ames Research Center, is attempting to quantitatively describe the effects of extended glare sources on the visibility of objects near the source. The expression empirically derived from experimental data for veiling brightness (E_v) when the eye is fixated on a point θ deg from a steady light source providing E foot-candles of illumination at the eye is $E_v = 10(E)/\theta = \text{footcandles}$. Gemini astronauts reported difficulty in sighting stars near the edge of the sunlit lunar disc due to the glare effect. Another source of visual field veiling is the spacecraft window, even though rather sophisticated attempts have been made to reduce this effect. When direct or reflected sunlight or earthlight is incident on the windows, even a small percentage of scattering is sufficient to decrease contrast to a point where external objects are not visible or are just barely so.

The number of reflective surfaces in the astronaut's field of view during docking with the cluster makes it imperative that the various relationships between the illumination sources and docking vehicle be thoroughly evaluated to minimize possible degrading effects of glare.

The intensity of an illuminated target is directly proportional to its reflectance (or albedo) and to specularity of its surface. Reflectance is generally defined as the percent of light reflected by the surface and (except for a few isolated types of materials) is independent of wavelength and angle of incidence of the illuminating rays. Specularity is the amount of surface roughness or measure of deviation of tangency of a microscopic "facet" of the surface with respect to the average slope of the surface. Attempts have been made to analyze these relationships for a sphere, assuming the surface to be covered with small mirrors whose deviation from tangency with the slope of the surface is some Gaussian distribution function. Results of the study were restricted to a specific viewing angle but did, however, indicate a marked variation in intensity per unit area of the target as a function of the specularity and the point on the target surface being examined. Total energy reflected to a given point by the surface was not evaluated. Since there was such a large variation in the intensity per unit area, as a function of the variables discussed above, a further analysis using a more general approach would be helpful in choosing the target surface coating and shape. This coating could be chosen to satisfy requirements for visibility at long ranges, e.g., rendezvous and guidance information, and those for size shape perception occurring during docking phases of the mission.

During docking and close-in inspection of a target, it is quite difficult to perceive the geometry of a specular surface illuminated by collimated sunlight. Since diffuse surfaces reflect impinging rays back in many directions, some sunlight directed back to the observer from all points of the illuminating surface will enter the pupil. For specularly reflecting surfaces, only the rays of light having appropriate incidence angles will be reflected in the directions required for entering the pupil of the observer's eye. Portions of the spacecraft thus may be effectively invisible or their

presence recognized only by silhouette against known or surmised visible objects in the field of view.

The Solar Illumination Simulation System currently available at Lockheed has the following characteristics. An Osram 6,500-w xenon compact arc lamp is used as the light-energy source and produces 325,000 lu from an arc 2.4 mm by 9 mm in size. The lamp employs forced-air cooling at a rate specified by Osram and must be kept within certain limits to maintain the lamp life. The bulb has a guaranteed life of 350 hr at a 6,500-w power consumption rate; however, most are capable of a nominal 400 to 500 hr. The source requires a power supply and starter, cooling apparatus, and a lamp status console to continually display power consumption, hours of run time, and number of starts. It can be operated at 8,000 w for short durations and can be tilted 10 deg from the vertical without degradation of performance.

A searchlight reflector 5 ft in diameter is used as the primary light-gathering and collimating optics. A secondary reflector behind the lamp is used to collect some of the light that would normally be lost by the system and to help correct nonuniformity in the beam. The reflector is equipped with a mounting ring that has a number of clamps around the periphery for adjustment and maintenance of sphericity of the reflector. The assembly is mounted on a vertical test stand to permit ± 10 deg tilt from the horizontal.

The primary test enclosure is a 10-ft cube constructed of epoxy-coated honeycomb cardboard painted with 3M Company Black Velvet Coating. This enclosure is designed to alleviate the back scattering problem of reflected light. The velvet coating reflects only 1 to 2 percent of the incident light that escapes from the light trap of the open honeycomb surface. The light enters and is absorbed by the light trapping coating during successive reflections from the labyrinth walls. On the open side of the chamber is a continuous 20 ft by 10 ft by 10 ft dark room. At the end of this secondary enclosure is a 6-ft-diam. aperture through which the source light enters the test chamber. The aperture serves to block the entry of unwanted portions of the light beam.

Light scattering caused by air molecules and dust is a problem since it will spread the beam and decrease contrast ratios by scattering light into shadowed areas. Preliminary calculations indicate that the contrast ratio decrease is about 1 percent in normal air. The beam will be visible even at this low scattering percentage unless an ambient pressure, clean-room environment is maintained.

Ideally, all testing should be accomplished in a vacuum environment for maximum fidelity and to eliminate atmospheric light scatter. Absorption by the atmosphere at the distances considered (100 ft) should be so small that the effects are negligible. A dehumidifier or molecular-sieve moisture filter should be incorporated to remove to the maximum extent water vapor from the room. The available facility at Lockheed does not at present provide for a sophisticated clean-room atmosphere. However, the test cube is equipped with positive pressure blowers and mechanical filters to reduce airborne particulate matter.

Performance characteristics of the facility are currently being determined by an equipment checkout and test program, and enough data are available to estimate its capability. The intensity at a power consumption of 6,500 w will be 0.4 solar constant (one solar constant is equal to 130 w/ft^2 outside the earth's atmosphere) on a 6-ft-diam. area in the test volume. At the 8-kw power setting, the incident intensity is about 0.5 solar constant. The uniformity of intensity should be within the ± 15 to 20 percent limits normally encountered with systems of this type. There should be a relatively close match between the simulated solar visible spectrum and the Johnson solar curve. The collimation angle estimated using simple ray tracing should be at most 0.8 to 1.0 deg. Because of the asymmetry in the arc, the collimation angle will vary slightly around the periphery of the beam. A clear glass ultraviolet filter will be used to remove undesired wavelengths for protection of test personnel.

Section 5

RELIABILITY AND SAFETY ANALYSIS

The OTES reliability goal for a 2-yr mission period is 0.90. This goal can only be attained through extensive use of redundancy and spares for the equipment in the basic AAP cluster which supports the 2-m telescope experiment. A further goal is to extend the mission life to 10 yr with additional resupply and maintenance. The safety goal for the AAP cluster is 0.999, and the telescope experiment is to be designed so as not to degrade this goal. While the specific maintenance and resupply schedules are beyond the scope of this feasibility analysis, the analysis of the AAP-ATM (Cluster A, Flight 4) presented below indicates that the 0.90 goal is feasible with reasonable allowance for redundancy, spares, and/or resupply.

5.1 SYSTEM EFFECTIVENESS TRADEOFF ANALYSIS

System effectiveness tradeoff analysis (SETA) is the basic reliability and analysis tool used to evaluate alternate designs and to assist design in meeting the reliability goals in an optimal manner. Through this technique, the desired system reliability is obtained analytically, and at the same time: (1) the system is optimized with respect to weight, cost, maintenance time, or any combination of these parameters, and (2) other design constraints are satisfied (such as safety, gross system weight, and single failure points) to perform the analyses. A mathematical model has been formulated and the equations reduced to a computer program that evaluates each potential redundancy in terms of reliability improvement versus the weight, cost, and maintenance time penalties. Redundancy types that can be accommodated include active, standby, multithread, binomial, and phantom.

LMSC experience has shown that the effects of cost and maintenance time on the reliability tradeoffs are relatively small, once a basic concept has been selected. Optimizing

with respect to weight becomes the key factor in a conceptual design. For the SETA computer program, the program constant is obtained by taking the total differential of the effectiveness equation, and the model distributes redundancy in an optimal manner.

Inputs are as follows:

- Component failure rate, duty cycle, and type of redundancy
- Component weight
- Criticality coefficients and the program constant

The computer calculates a $\Delta W/\Delta R$ ratio for each addition of a redundant component, compares the ratio with the program constant, and rejects the redundancy if the program constant is exceeded.

Output information generated and printed by SETA for application to reliability and design comprise the following:

- For each component – reliability, $\Delta W/\Delta R$ for each redundancy tested, optimum redundancy, and reliability
- For each subsystem – initial and final reliability and weight, and a reliability block diagram with optimal redundancy

An example of the subsystem output data is shown in Fig. 5-1 with a sample calculation. Figure 5-2 is the AAP ATM system (Cluster A, Flight 4) evaluated for a 2-yr mission not including the telescope experiment. The SETA program has not been used to evaluate the telescope experiment thus far because of the lack of specific design data. For instance, explosive force pins are proposed to lock the extended spar segments, but the number of pins required to attain the required structural rigidity will be determined by actual test data of a particular design. With these data and the known reliability of 0.9996 for a single force pin, SETA can be used to compute an optimal number of spare pins.

The OTES Program makes maximum use of existing AAP and other space-qualified hardware which have known or estimated failure rates. The SETA technique apportions

to the component level, and automatically considers the factors of criticality, complexity, weight, and reliability. The system reliability generated by SETA is an apportionment as well as a predicted reliability; that is, it is based on known or estimated failure rates. The SETA technique is used to generate new apportionments and predictions throughout the design cycle as changes are proposed and/or made. The rapid response of SETA permits its use as a real-time design tool, and it facilitates design evaluation. As test and operational data become available, predictions are revised and updated to reflect current data.

For the OTES Program, SETA will be used to update to the Cluster X configuration with the telescope experiment included. The analysis of Fig. 5-2, for instance, does not contain failure sensing and isolation devices, the inclusion of which would reduce the attainable reliability. The SETA approach would also be used to analyze expected resupply requirements.

5.2 TELESCOPE EXPERIMENT RELIABILITY ANALYSIS

Analysis of the telescope reliability could be considered to consist of a deployment subsystem and an imaging subsystem, with the Orbital Mirror Recoating Facility (OMRF) included as a "recoating subsystem."

The critical components of the deployment subsystem are the reel-out cables and their associated drive motors. The present design incorporates two sets of motors and cables. In the event of the failure of one set, it is probable that the remaining set could complete deployment. However, if subsequent analysis indicate additional sets are required, the design can permit installation of two additional sets.

The locking of the deployed telescope is accomplished by explosive force pins implanted around the periphery of each telescope tube section. The design can accommodate a large number of these devices which have an individual reliability of 0.9996. The present configuration calls for 12 devices per tube section, while it is expected

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PROGRAM TO COMPUTE OPTIMAL REDUNDANCY AND SYSTEM RELIABILITY, WEIGHT, COST, AND EXPECTED MAINTENANCE

REDUNDANCY TYPE, S = STANDBY, A = ACTIVE, P = PHANTOM, X/Y = BINOMIAL MISSION DURATION IS

ANALYSIS NO. 4054

DATE 03 MAR 67

ELECTRICAL POWER REGULATION AND CONTROL

EQUIP	(1) FAILURE RATE	(2) WEIGHT	COST	(2) OP TIME MISSION	(3) DOWN TIME	RELIABILITY	(4) WT/REL	COST/REL	DT/REL
1	3.00E-06	17.00	-0.	1344.	1/ 1 1P -0.00 2P -0.00	.99597611 .99999189 .99999998	42.3 20998.5	-0. -0.	*****
	INVERTER								
2	25.00E-06	90.00	-0.	1344.	21/24	.99254306			
	BATTERY ASSEMBLY								
3	.01E-06	47.00	-0.	1344.	39/40	.99999986			
	SOLAR ARRAY PANEL								
4	2.00E-06	35.00	-0.	50.	1S -0.00	.99990000 .99999999	3500.3	-0.	-.
	POWER DISTRIBUTOR								
5	9.85E-06	10.50	-0.	50.	1S -0.00	.99950761 .99999987	213.3	-0.	-.
	AUX POWER DISTRIBUTOR								
6	8.37E-06	25.50	-0.	50.	2/ 2 2/ 3 -0.00	.99911338 .99999940	287.8	-0.	.
	CONTROL DISTRIBUTOR								
7	.80E-06	35.00	-0.	1344.	1S -0.00	.99892537 .99999941	325.9	-0.	-.
	SOLAR ARRAY DISTRIBUTOR								
8	24.00E-06	23.30	-0.	56.	1S -0.00	.99865690 .99999909	173.6	-0.	-.
	COMMANDERS CB PANEL								
9	24.00E-06	22.70	-0.	56.	1S -0.00	.99865690 .99999909	169.1	-0.	-.
	SYSTEM ENGRS CB PANEL								
10	1.00E-06	15.10	-0.	200.	1S -0.00	.99980001 .99999997	755.2	-0.	-.
	RELAY JUNCTION BOX								
11	4.76E-06	7.00	-0.	1.	1/ 2	.99999999			
	PYRO BATTERY								

INITIAL SUBSYSTEM RELIABILITY WITHOUT REDUNDANCY = .983183

INITIAL SUBSYSTEM WEIGHT WITHOUT REDUNDANCY = 4023.599900

INITIAL SUBSYSTEM COST WITHOUT REDUNDANCY = .000000

FINAL SUBSYSTEM RELIABILITY WITH OPTIMUM REDUNDANCY = .987147

FINAL SUBSYSTEM WEIGHT WITH OPTIMUM REDUNDANCY = 4040.599900

FINAL SUBSYSTEM COST WITH OPTIMUM REDUNDANCY = .000000

FINAL SUBSYSTEM EXPECTED MAINTENANCE TIME = .000000

ANCE TIME

56 DAYS

(4) TEST/REL	(5) OPTIMAL REL
42.3 20998.5	.99999189 .99254306 .99999986
3500.3	.99990000
213.3	.99950761
287.8	.99911338
325.9	.99892537
173.6	.99865690
169.1	.99865690
755.2	.99980001 .99999999

NOTES:

- (1) Failure rate is expressed as failures per million hours ($E - 06 = \times 10^{-6}$)
- (2) Weight is in pounds and operating time is in hours
- (3) RED = amount and type of redundancy
- (4) $WT/REL = TEST/REL = \Delta W/\Delta R$
- (5) OPTIMAL REL = reliability with optimum number of redundant units
- (6) Program constant = 67.0

Sample Calculation. With a program constant of 67.0, the computer will add redundant units as long as the ratio of $\Delta W/\Delta R$ (lb/%) is less than 67.0. With item 1, the inverter, addition of the first redundant unit gives the following:

$$\Delta W = 17.0 \quad \Delta R = (0.99999189 - 0.99597611) 100 = 0.401578$$

$$\Delta W/\Delta R = 17.0/0.401578 = 42.3$$

Since $\Delta W/\Delta R < 67.0$, a redundant inverter is added. The program then recycles and tests for a 2nd redundancy

$$\Delta W = 17.0 \quad \Delta R = (0.99999998 - 0.99999189) 100 = 0.000809$$

$$\Delta W/\Delta R = 17.0/0.000809 = 20998.5$$

Since $\Delta W/\Delta R > 67.0$, the 2nd redundancy is rejected and the optimum status is one redundant inverter with a net reliability of 0.99999189 for the inverter function.

Fig. 5-1 SETA Printout of the A4 Electric Power Regulation and Control Subsystem

SUBSYSTEM	WEIGHT		EXPTD D/T	COST		RELIABILITY	
	INITIAL	FINAL		INITIAL	FINAL	INITIAL	FINAL
ENVIRONMENTAL CONTROL SYSTEM	733.0	968.6	.00	.0	.0	.816631	.995600
STABILIZATION, NAVIGATION, AND GUIDANCE	1743.6	3470.9	.00	.0	.0	.049872	.975069
PROPULSION REACTION CONTROL	194.2	219.5	.00	.0	.0	.984519	.999966
ELECTRICAL POWER REGULATION AND CONTROL	3783.6	5115.6	.00	.0	.0	.000100	.982064
DATA DISPLAY	307.5	421.3	.00	.0	.0	.839535	.999142
DATA ACQUISITION (INSTRUMENTATION)	641.1	1427.9	.00	.0	.0	.001686	.986318
DATA TRANSMISSION (COMMUNICATION)	228.5	447.2	.00	.0	.0	.334464	.991661
DATA PROCESSING	40.0	185.0	.00	.0	.0	.441719	.998458
ARRANGEMENTS, INSTALLATION AND STRUCTURE	5438.3	5438.3	.00	.0	.0	1.000000	1.000000
726 DAY SYSTEM	13109.8	17694.5	.00	.0	.0	.000000	.930215

Fig. 5-2 SETA Printout of the X-4 OTES/ATM System, 2-yr Mission

that no more than 6 would suffice. With this amount of redundancy, the section reliability would be in excess of 0.999999 when computed using the binomial distribution, i.e.,

$$R = \sum_{r=7}^{12} \binom{12}{r} p^r (1-p)^{12-r}$$

The control and delay system electronics have not been defined and would probably contribute the largest reliability hazard. These circuits will be made redundant to eliminate these hazards insofar as possible.

The imaging subsystem contains an imaging surface with positioning servos and the secondary lens with servos and the control electronics. The servo drive motors are extremely reliable as is the imaging surface. The electronics presents the greatest possibility of reliability hazard but when finally configured will contain the necessary redundancy to minimize the hazard.

The preliminary concept for an OMRF depends upon manual segment insertion and electrochemical deposition. This simplified design contains an extremely small probability of failure. The tungsten active elements rely upon an electrochemical phenomenon to perform their function, and, in addition, there are several more elements than are absolutely necessary. The major equipment failure probability would be in the case of nonuniformity of energy delivered from the electrical power supply to the elements. However, in addition to the close monitoring of the power supply, the supply is also space maintainable and resuppliable as are the tungsten elements themselves. Provided that the probability of operator error can be minimized by provision of automatic processing controls and an instinctive control arrangement and rigorous training, it is extremely unlikely that the unit would fail in space.

5.3 SAFETY ANALYSIS

During the deployment phase of the telescope, the astronauts will not engage the EVA. The EVA operations required of the astronauts in support of the OTES experiment are those associated with replacement of failed equipment section modules, film retrieval, and recoating of the segmented mirror. The design will be so configured that sharp protuberances do not exist. Access to removable devices is adequate in all respects, and, therefore, the hazards affecting crew survival are entirely similar to those associated with other AAP cluster experiments. It follows, therefore, that the safety provisions and design objectives applicable to the AAP and the astronaut suit loop are applicable to the OTES. Therefore, since these have been considered during the AAP safety studies and no further hazards have been discovered at this stage of the design, it is anticipated that the overall safety objective of 0.999 will not be degraded by inclusion of the telescope experiment.

5.4 CONCLUSIONS

At the present stage of conceptual design, it has not been possible to conduct a truly meaningful quantitative analysis. However, preliminary review of the interacting reliability values has permitted the conclusion to be drawn that the expected OTES reliability objective can be met and will be in the order of 0.92. Safety conclusions, as stated above, appear to assure attainment of the goal of 0.999 established for crew survivability.

5.5 RELIABILITY ANALYSIS TASK SUMMARY

5.5.1 Study Phase B Tasks

During the Phase B Project Definition, the systems effectiveness tradeoff analysis technique, together with supporting reliability analyses to estimate failure rates, will be used to assist in design tradeoffs leading to the selection of a single design concept. The reliability of the resulting preliminary design will be assessed, and any remaining

problem areas will be identified for further Phase C analysis. Failure mode effects and criticality analyses will be performed to identify critical failure modes which would degrade mission success or astronaut safety.

Preliminary reliability, safety, maintainability, and quality assurance program plans will be prepared for the Phase C and Phase D efforts, together with estimated manpower loading.

5.5.2 Study Phase C Tasks

Detailed reliability, safety, maintainability, and quality assurance program plans will be prepared and updated as required. The reliability tasks will include, but not necessarily be limited to, the following:

- Systems effectiveness tradeoff analysis
- Reliability apportionment
- Reliability prediction
- Failure modes and criticality analyses
- Design reviews
- Parts application analyses
- Parts procurement support
- Subcontractor and vendor control and liaison
- Preliminary test analysis
- Preparation and review of definitive specifications
- Review and sign-off of drawings (all levels)

Program plans will be updated and defined for Phase D.

5.5.3 Study Phase D Tasks

The Phase C reliability efforts will be continued during Phase D in support of the final design effort. The quality assurance and inspection plans will be implemented per the specific requirements of the hardware. All test data will be analyzed and reliability assessments updated.

Section 6

CLUSTER X MISSION SEQUENCES AND LAUNCH CONSTRAINTS

Although no firm Cluster X flight plans are currently in effect, preliminary analyses of Cluster A mission tasks have been conducted for which configurations have been tentatively specified by NASA. It is recognized that these plans are subject to gross changes. However, in order to show how the OTES could logically replace the ATM and function in the AAP cluster configuration, a description of tasks and pertinent mission elements for an OTES cluster application has been prepared on the basis of present knowledge. The description presented by this section is based, in part, on Cluster A plans because the initial Cluster X flights would, in effect, be a repeat of the Cluster A mission.

6.1 MISSION SEQUENCE

Nominally, Cluster Flights X-1 and X-2 will have orbited the earth for 90 days, during which time the orbital workshop (OWS) and multiple docking adapter (MDA) will have been established with a habitable, shirtsleeve environment, suitable to the subsequent experimental purposes.

The OTES flights of Cluster X begins with the X-3 launch (Figs. 6-1 and 6-2) when the S-IVB and CSM are placed in an initial elliptical parking orbit of 81 by 120 nm. The CSM separates from the S-IVB and circularizes in orbit at 120 nm. One day later, the X-4 launch, carrying the OTES, is injected directly into a 244-nm circular orbit. The X-3 spacecraft transfers to a 244-nm orbit for rendezvous with the X-4. The X-3 CSM docks to the LM/rack. The LM/rack, containing the OTES 2-m telescope, is then placed in a 254-nm circular orbit by the SPS to rendezvous with Cluster X. The cluster by this time has decayed from its original 260-nm orbit to 254 nm. Two astronauts transfer to the LM/rack and independently dock the spacecraft to the MDA. The CSM completes its docking with the cluster. The X-1 and X-2 spacecraft astronauts, after experiment handover, depart for earth.

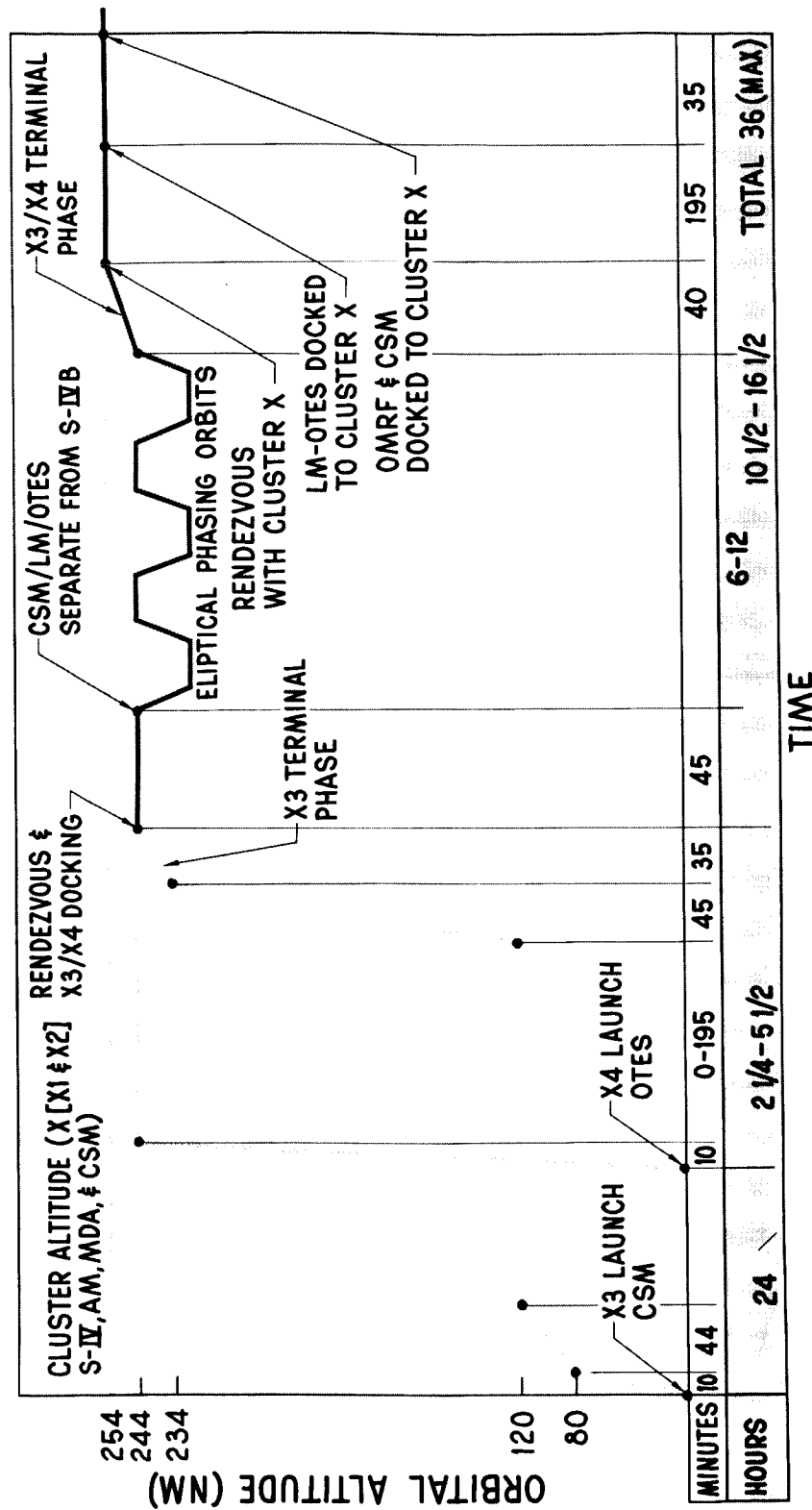
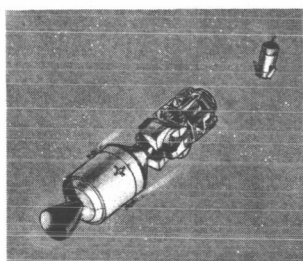
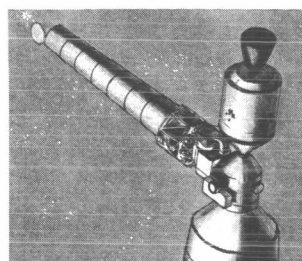


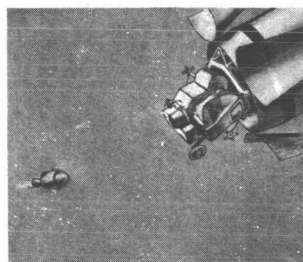
Fig. 6-1 Mission Profile



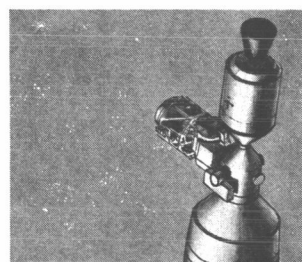
SEPARATION
FROM S-IVB



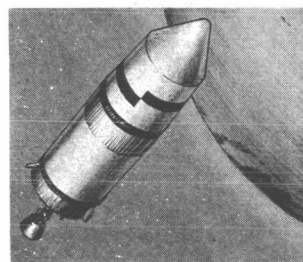
INITIATE OTES
OPERATIONS



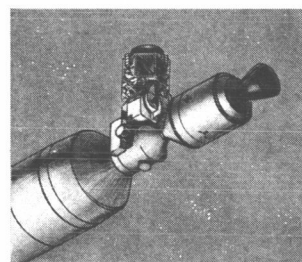
RENDEZVOUS
AND DOCKING



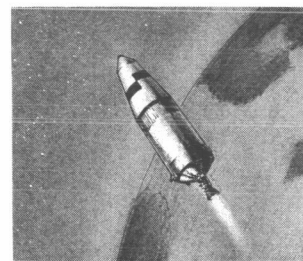
STABILIZE
CLUSTER X



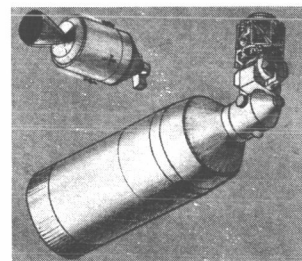
CIRCULARIZE
240 N.M. ORBIT



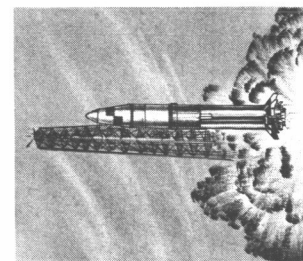
DOCK OMRF
AND CSM



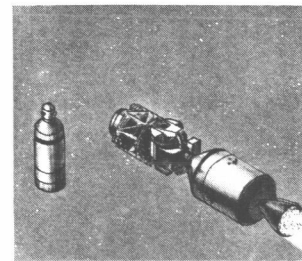
S-IV B
BURN TO ORBIT



SEPARATE AND
DOCK LM



LAUNCH
SATURN IB



ORBIT
TRANSFER

Fig. 6-2 Operations Sequence

Resupply and change of Cluster X operating personnel are effected by subsequent launches of a CSM (capable of 90-day, three-man life support) every 90 days. Present plans indicate that X-5 and X-6 launches may be made with possible follow-on missions as required.

6.2 LAUNCH CONSTRAINTS

Launch periods for the X-3 and X-4 flights are established by two constraining conditions: (1) coplanar launch with the existing cluster orbit plane, which occurs daily, and (2) daylight launch conditions between dawn and noon for potential crew recovery under an aborted launch. With a daylight launch constraint, rendezvous launch periods occur approximately every 46 days with each period lasting 9 to 13 days depending upon the amount of time from dawn to noon, i.e., season of the year). The 46-day cycle is caused by changing the orientation of the orbit plane relative to the direction of the sun. The cluster orbit plane, inclined about 28.5 deg regresses 6.8 deg per day due to earth oblateness. The earth-sunline rotates 0.99 deg per day due to earth orbital motion. The combined effect is a changing orbit plane-sunline geometry that repeats every 46 days. With the orbit plane regressing at 6.8 deg per day, the launch site on a rotating earth becomes coplanar, each successive day, 0.5 hr earlier. For X-3 and X-4 launches in mid-winter, 5 hr of daylight are available from dawn to noon; this results in a launch period of 10 days with coplanar launch time occurring about noon on the first day followed each day 0.5 hr earlier and ending about dawn on the tenth day.

The cluster position in orbit is randomly located with respect to time when the launch site is coplanar. This requires that the launch times of X-3 and X-4 be chosen for completion of X-3/X-4 rendezvous in an orbital position compatible with cluster rendezvous. A choice of X-4 launch time is based upon proper phasing injection with respect to the cluster, and any nonplanar conditions at liftoff are corrected by yaw steering during the ascent trajectory. A choice of the X-3 launch time must be compatible with the planned X-4 launch time with corrections of nonplanar conditions after liftoff by yaw steering. Time errors in X-3 and X-4 liftoff will be corrected by on-orbit phasing.

Flights X-5 and X-6 will be launched at 90-day intervals, extending the mission by resupplying consumables and interchanging crews.

As the Cluster X mission progresses, orbital altitude decays at a rate depending upon cluster area to mass ratio. Figure 6-3 shows approximate orbit decay characteristics. To achieve extended orbital stay times or to raise the orbit of the cluster, additional velocity may be imparted to the cluster by utilizing the RCS or the CSM SPS. Figure 6-4 shows the total velocity change, in feet per second, required to restore cluster orbit altitude. Since mass of propellant required to effect orbital altitude change varies with the Cluster X total mass, individual calculations must be made for each case. Figure 6-5 shows the mass of propellant required based upon a typical cluster weight of approximately 105,000 lb. The figure shows both methods, RCS or SPS, of changing the orbit. The use of RCS propellant is less efficient than SPS propellant because of lower I_{sp} . When SPS is used, some RCS propellant is also used for ullage control and guidance dispersions; the SPS advantage is realized for orbit increases in excess of 10 nm.

Follow-on flights, after the X-6 flight, although not currently scheduled, could be made to meet additional mission requirements.

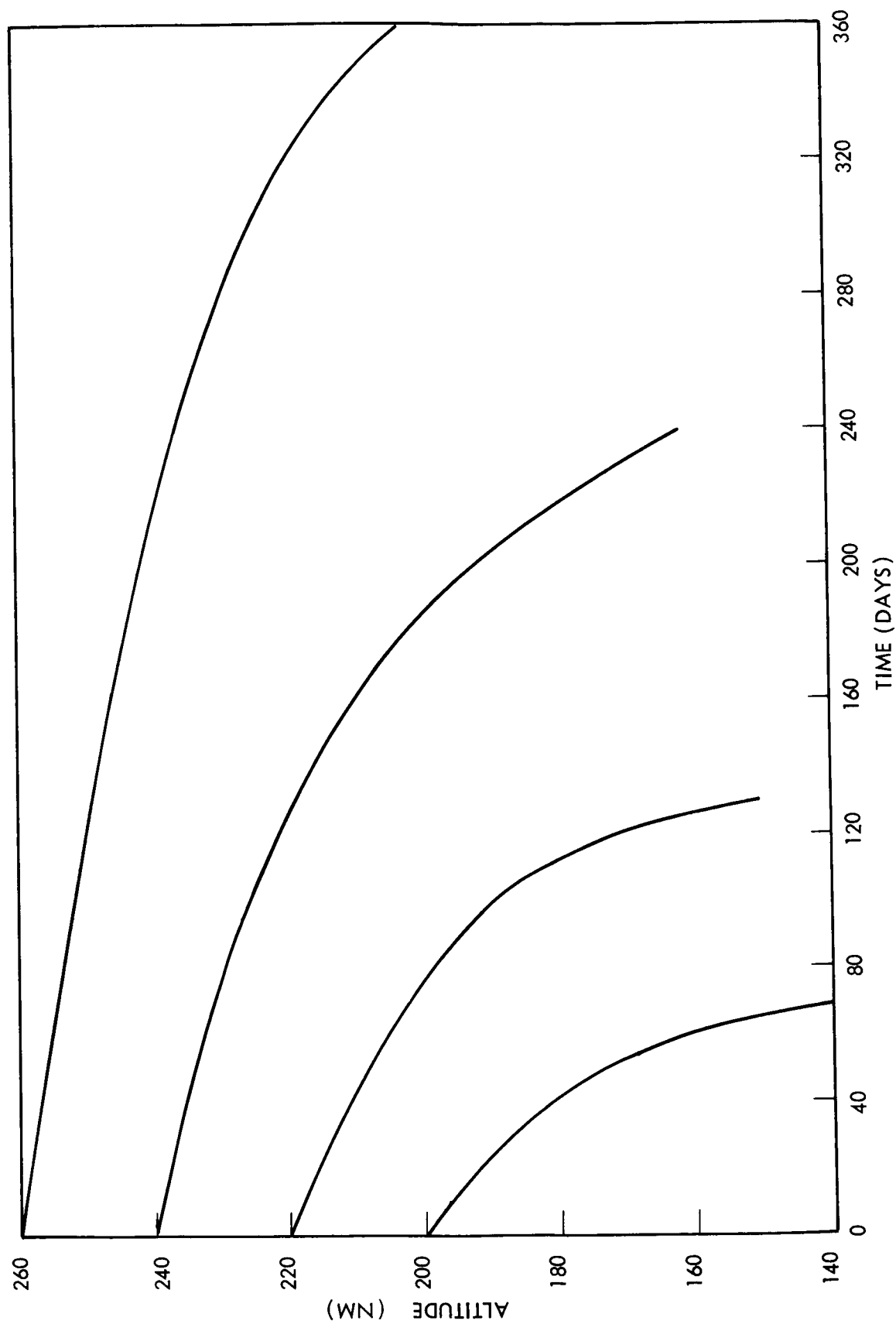


Fig. 6-3 Cluster B Orbit-Decay Characteristics

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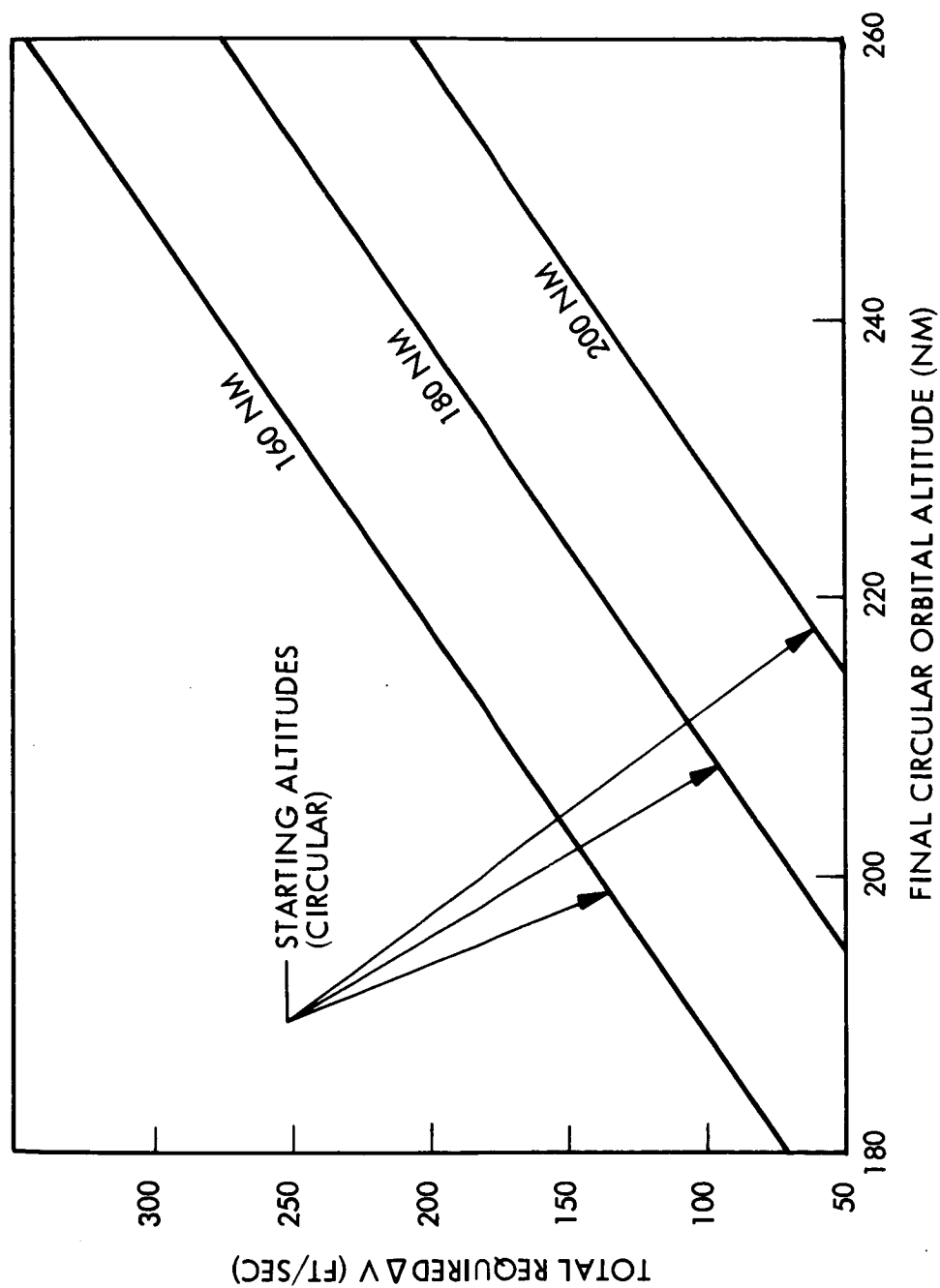


Fig. 6-4 Total Velocity Change Required to Restore Cluster Orbital Altitude

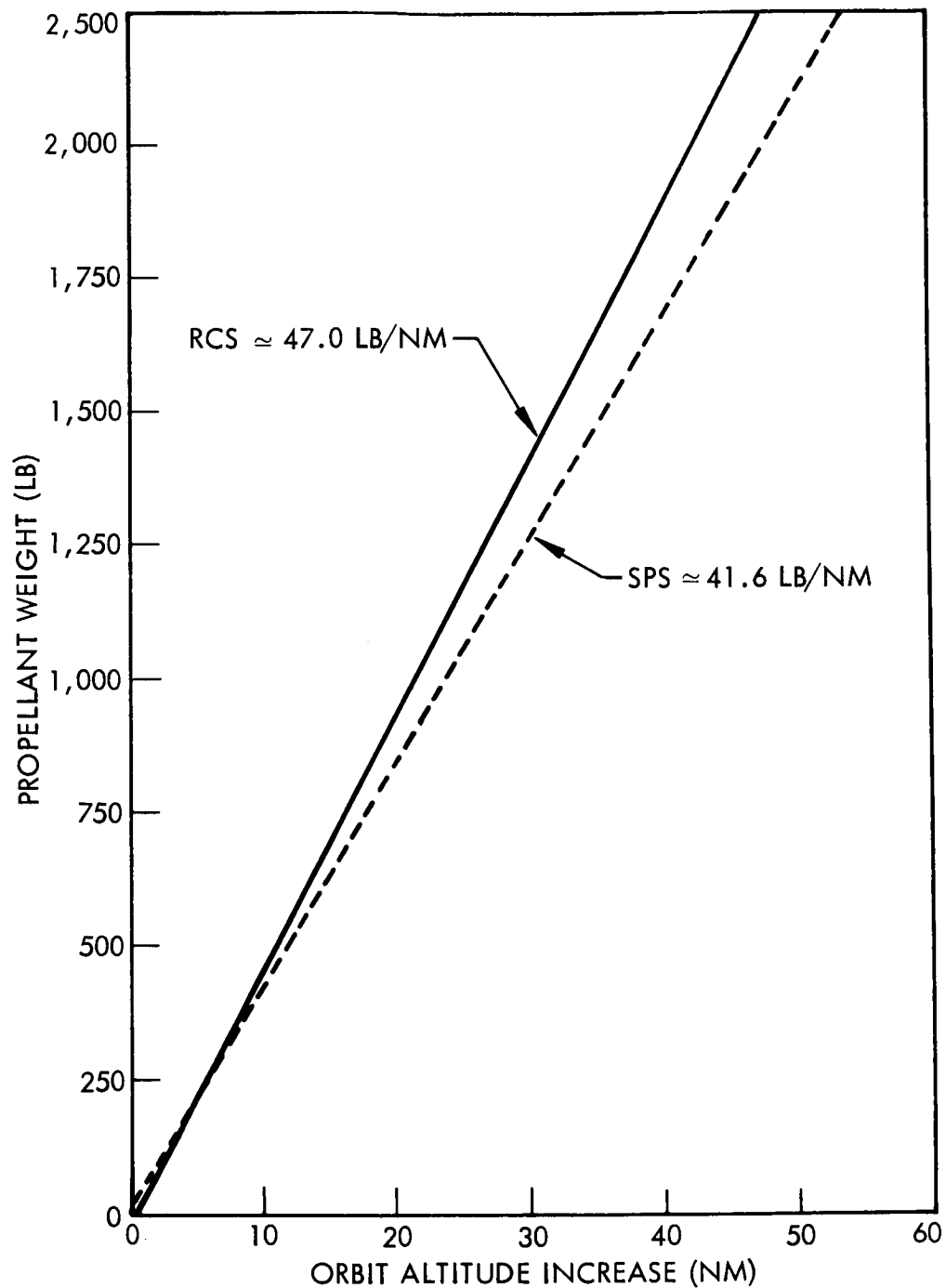


Fig. 6-5 Propellant Required for Orbit Adjustment

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Section 7
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Section 1
RESOURCES PLANNING SUMMARY

During the 3-yr Phase A of the OTS and OTES studies, Perkin-Elmer alone and with LMSC investigated eight different optical technology system spacecraft concepts:

- 1-m Unmanned Optical Technology Satellite (OTS) (Reports 7846 and 7924)
- 3-m Concept Without Life Cell (Report 8319)
- 3-m Concept With Life Cell (MOTEL) (Report 8500)
- 2-m Concept in Autonomous Mode (Report 8319)
- 2-m Concept in Cluster X (Report 8900)
- 1-m Concept Integrated With Lunar Module (Report 8319)
- 0.5-m Concept Integrated With Lunar Module (Report 8319)
- 0.4-m Concept Called Laser Communication Satellite Experiment (LCSE) (Reports 8399 and 8399A)

The basic variations between the eight concepts evolved out of emphasis on crew participation in the optical technology experiments system and the emphasis on optical technology as applied to large telescopes in contrast with optical technology as applied to optical laser communications. Thus, the initial OTS study (NAS 8-11408) concentrated on an unmanned spacecraft using the Saturn I-Centaur boost vehicle for laser experiments. With the introduction of the AAP concept in NASA, emphasis of the study shifted to the larger optical systems utilizing the Saturn V vehicle system hardware and, in the current study, emphasis of the experiment system again shifted. The current 2-m OTES study is directed at the optical technology experiments that can be accomplished using the present NASA cluster concept. This system uses the multiple launch Saturn IB boost vehicles.

One of the most difficult problem areas in the current OTES study was the broad question of recommendation to MSFC and the Office of Advanced Research and Technology

of an optical technology plan with appropriate flight experiments at suitable times. The plethora of concepts had to be sifted and sorted to search for the recommendation that would represent the best buy for NASA. The recommendation would have to stand the test of acid scrutiny to satisfy the future needs of NASA in the most economical manner. The interdependence of the proposed flights and the data that they would bring to the prospective national goals for NASA are inherent in the recommended NASA Space Optical Technology Plan.

During the current Phase A work on OTES, a basic NASA Space Optical Technology Plan was developed with four different options. Of the four distinctly different approaches detailed in this plan, the Perkin-Elmer Corporation recommends Option 4 to NASA. The key elements of this option are the two separate flights - one in synchronous orbit and devoted to laser communications and one at 260-nm altitude and designed for experiments in large optics. The resources planning that is discussed in this volume of the Phase A report is detailed for the recommended Option 4.

Before proceeding into the resources details for this option, it is important that NASA management understand the choices that are available and the characteristics that distinguish one option from the other.

Time-oriented PERT charts were developed for the four approaches (Figs. 1-1 through 1-4), and the OTES Program Budgets for each option precede the PERT plans.

1.1 SPACE OPTICAL TECHNOLOGY PLAN OPTION 1 (FIG. 1-1)

1.1.1 Key Features

Individual payloads developed to support different missions are as follows:

- Laser satellite experiment launched to synchronous orbit in 1970 and oriented for experiments in laser propagation
- 2-m OTES telescope launched into low orbit in 1973 as part of the AAP cluster and oriented for experiments in large space optical telescopes; the 2-m telescope is a precursor for the 3-m telescope
- 3-m OTES telescope launched into synchronous orbit (assumption of results of 2-m telescope experiments) in 1976; the 3-m telescope is used for space experiments directed toward space-station observatory missions

The 3-m OTES telescope residual hardware is used with the apparatus from following flights to modify it to become a part of the 3-m MOT-class astronomical space observatory.

1.1.2 Equipment Budget Recommendations

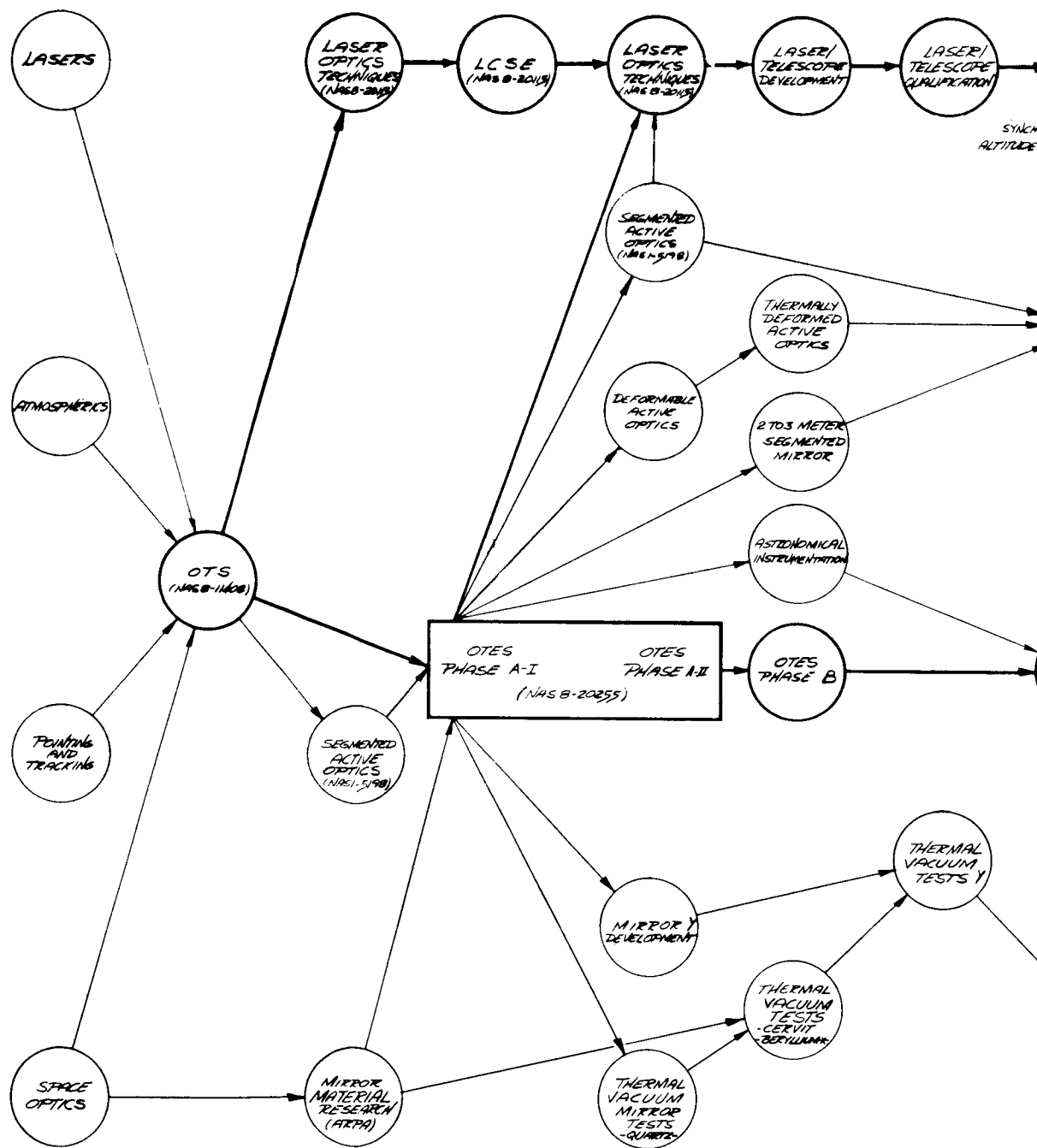
<u>Item</u>	<u>Amount (10⁶ \$)</u>
Supporting technology (1968-1972)	8
Laser satellite experiments in 1970 as part of AAP cluster at synchronous altitude	7.5
2-m OTES as part of AAP cluster at low altitude in 1973	42
3-m OTES telescope as part of AAP cluster at an assumed synchronous altitude in 1976	105
3-m OTES MOTEL (life cell for astronauts) launched with telescope to synchronous altitude in 1976	107
Telescope astronomical instrumentation for 3-m OTES to be used following optical technology experiments and launched to mate with the 3-m telescope in 1976	<u>22</u>
Total	291.5

1.1.3 Basic Advantages and/or Disadvantages to NASA

- The optical technology development will occur in a timely manner to support the anticipated National Space Goals of Interplanetary Missions in the early 1970's and the National Space Observatory Mission in the late 1970's.
- The optical technology development will require substantial program commitments during the next 2 yr.

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63	64	65	66	67	68	69
FOUNDATION OPTICAL TECHNOLOGY R & D	INITIAL SYSTEMS STUDY	LABORATORY DEVELOPMENT AND EXPERIMENTS	SPECIFIC SPACE EXPERIMENT STUDIES	EQUIPMENT DEVELOPMENT AND GROUND TESTS		



70

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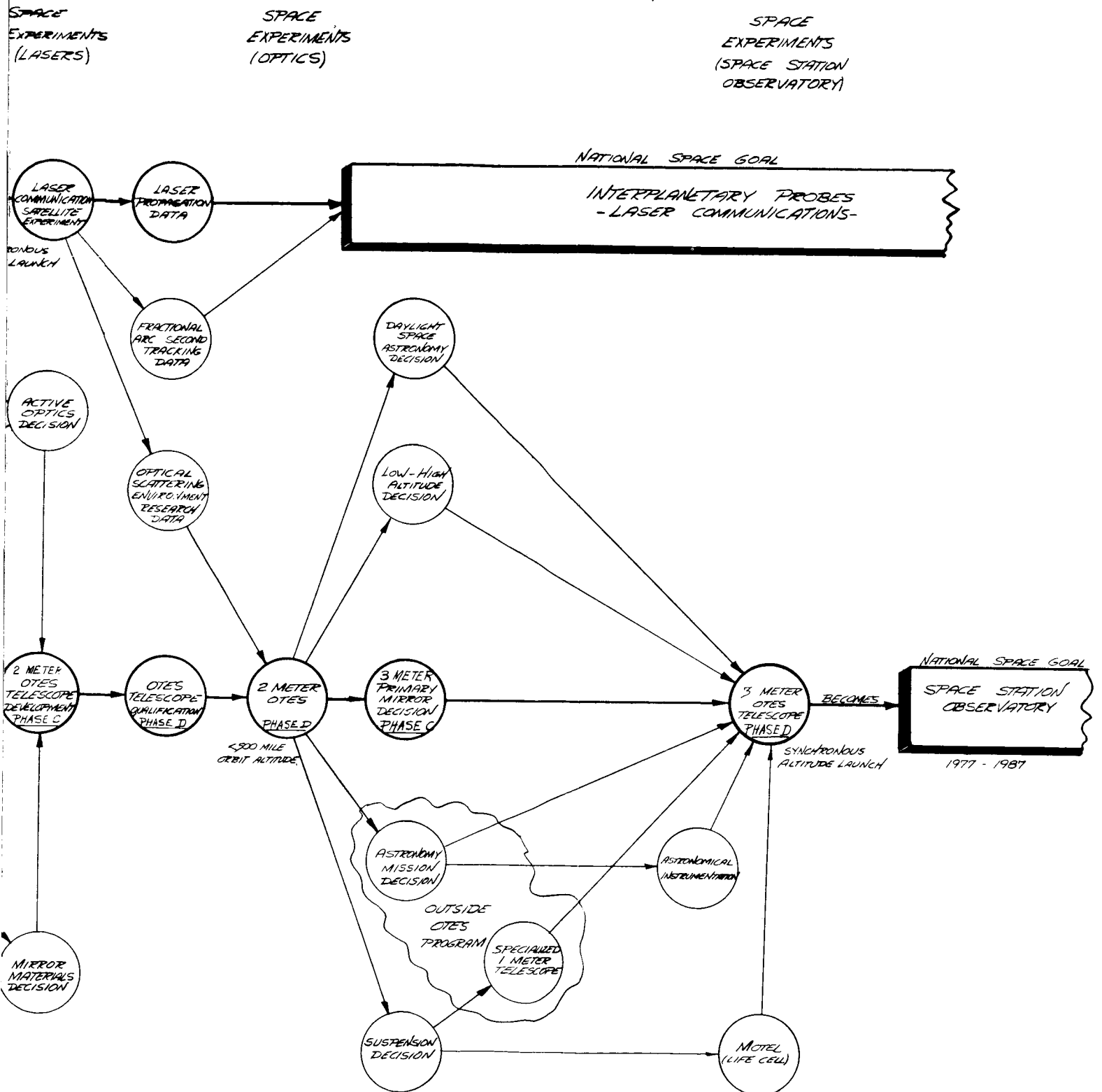


Fig. 1-1 NASA Space Optical Technology Plan Option 1

1.2 SPACE OPTICAL TECHNOLOGY PLAN OPTION 2 (FIG. 1-2)

1.2.1 Key Features

- A combination of the laser propagation experiments with the optical technology experiments into a single 2-m telescope launched into a synchronous orbit in 1973
- The 3-m OTES telescope deleted from the OTES Program

1.2.2 Equipment Budget Recommendations

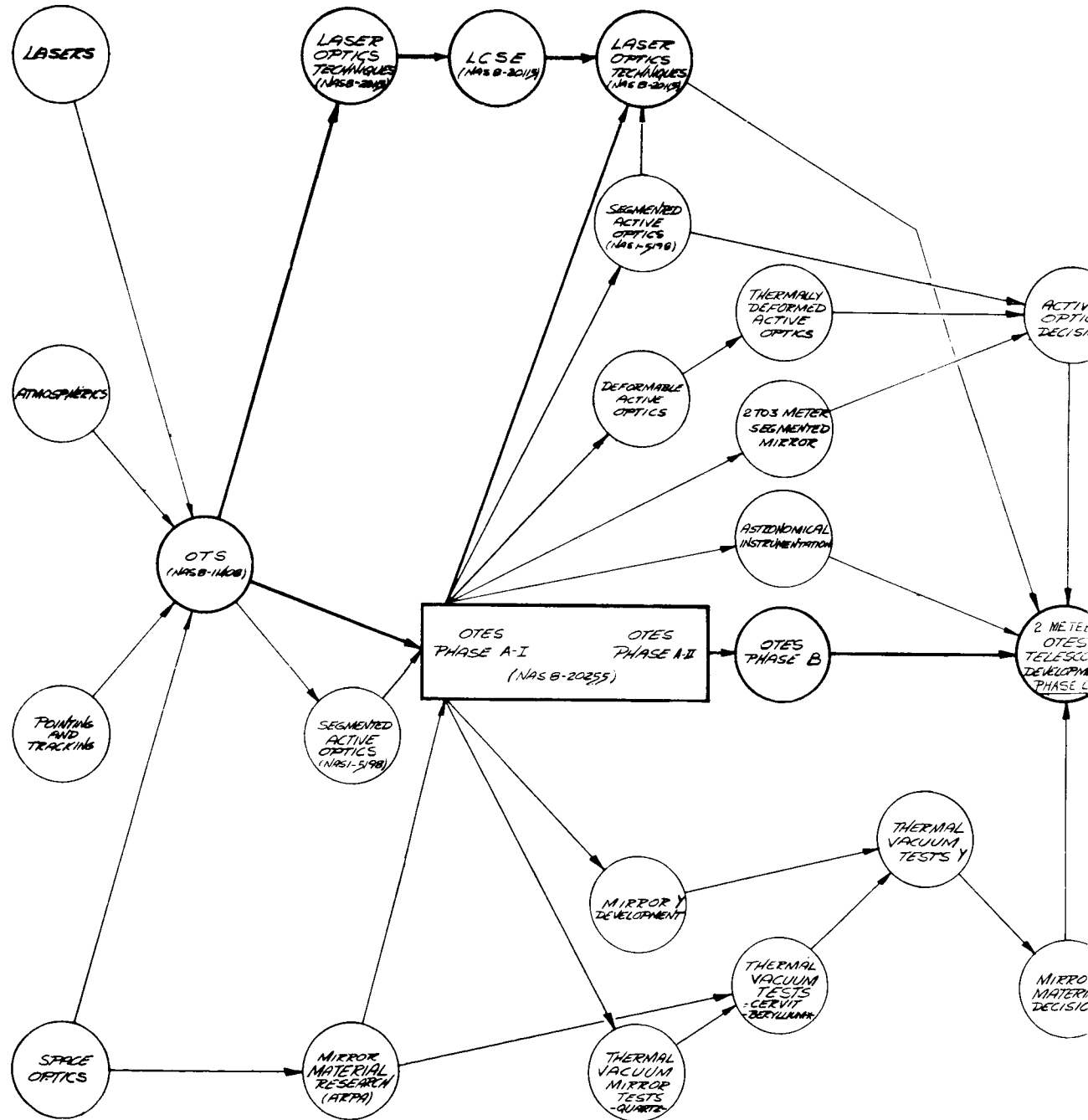
<u>Item</u>	<u>Amount (10⁶ \$)</u>
Supporting technology (1968-1972)	8
2-m OTES as part of the AAP cluster at synchronous altitude and ready for launch in 1973	<u>48</u>
Total	56

1.2.3 Basic Advantages and/or Disadvantages to NASA

- This option is the least costly of technology development approaches.
- Laser communications from deep-space probes will not be ready for Voyager spacecraft until 1975 and beyond.
- The most suitable orbit for the National Space Station Observatory will not be established. The questions that must be answered before an optimum mission orbit is selected will not be answered by a single synchronous OTES launch even when the data from the ATM experiments are included.

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63	64	65	66	67	68	69	70
FOUNDATION OPTICAL TECHNOLOGY R & D	INITIAL SYSTEMS STUDY	LABORATORY DEVELOPMENT AND EXPERIMENTS	SPECIFIC SPACE EXPERIMENT STUDIES	EQUIPMENT DEVELOPMENT AND GROUND TESTS			



71

72

73

74

75

76

77

SPACE
EXPERIMENTS
(OPTICS
&
LASERS)

SPACE
EXPERIMENTS
(SPACE STATION
OBSERVATORY)

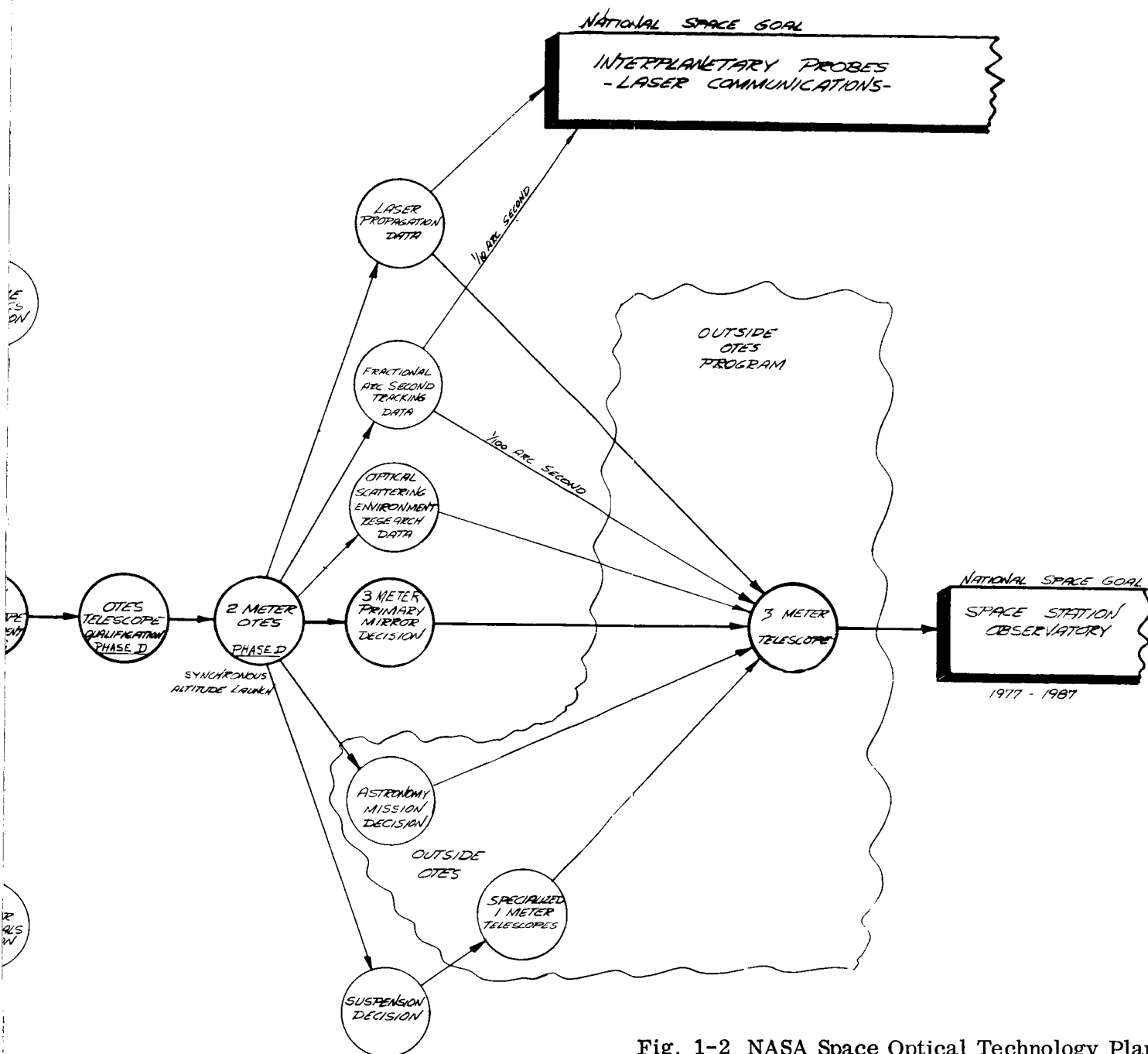


Fig. 1-2 NASA Space Optical Technology Plan
Option 2

1.3 SPACE OPTICAL TECHNOLOGY PLAN OPTION 3 (FIG. 1-3)

1.3.1 Key Features

- Combines the laser flight experiments and the optical experiments associated with the 2-m OTES into a single payload launched into synchronous orbit in 1973
- Develops the 3-m OTES telescope within the OTES Program but has the astronomical instrumentation equipment developed outside the OTES Program or dropped
- Uses the S-IVB orbital workshop or the MOL as a life cell for the telescope instead of the 3-m MOTEL

1.3.2 Equipment Budget Recommendations

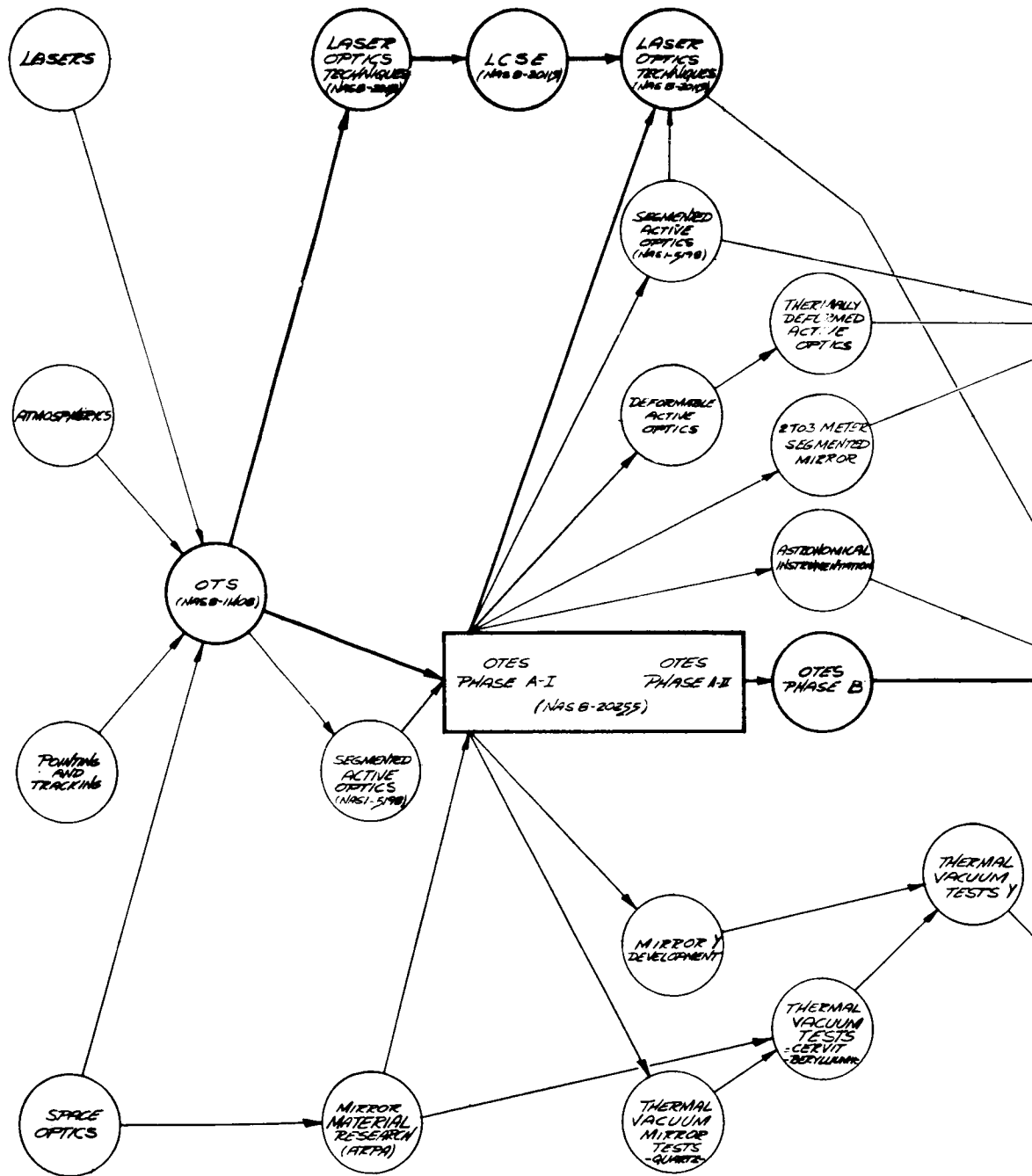
<u>Item</u>	<u>Amount (10⁶ \$)</u>
Supporting technology (1968-1972)	8
2-m OTES telescope with both laser and optical experiments launched into synchronous orbit in 1973. It is mated with the AAP cluster and fits into the space provided for the ATM solar experiments	48
3-m OTES telescope without life cell and astronomical instrumentation. Ready for launch into synchronous orbit by 1976 assuming that the 2-m OTES experiments provide satisfactory data on daylight astronomy, 1/100 arc sec pointing, and active optics	<u>107</u>
Total	163

1.3.3 Basic Advantages and/or Disadvantages to NASA

- Requires OART to develop the optical technology, OSSA to develop the astronomical mission and then the appropriate astronomical instrumentation, and OMSF to develop the astronaut/astronomer life cell simultaneously
- Laser communication technology will not be ready to support Voyager 1973 (or other interplanetary probes) until 1975

PERKIN-ELMER/LOCKHEED

63	64	65	66	67	68	69
FOUNDATION/ OPTICAL TECHNOLOGY R & D	INITIAL SYSTEMS STUDY	LABORATORY DEVELOPMENT AND EXPERIMENTS	SPECIFIC SPACE EXPERIMENT STUDIES	EQUIPMENT DEVELOPMENT AND GROUND TESTS		



70 71 72 73 74 75 76 77

SPACE
EXPERIMENTS
(OPTICS
&
LASERS)

SPACE
EXPERIMENTS
(SPACE STATION
OBSERVATORY)

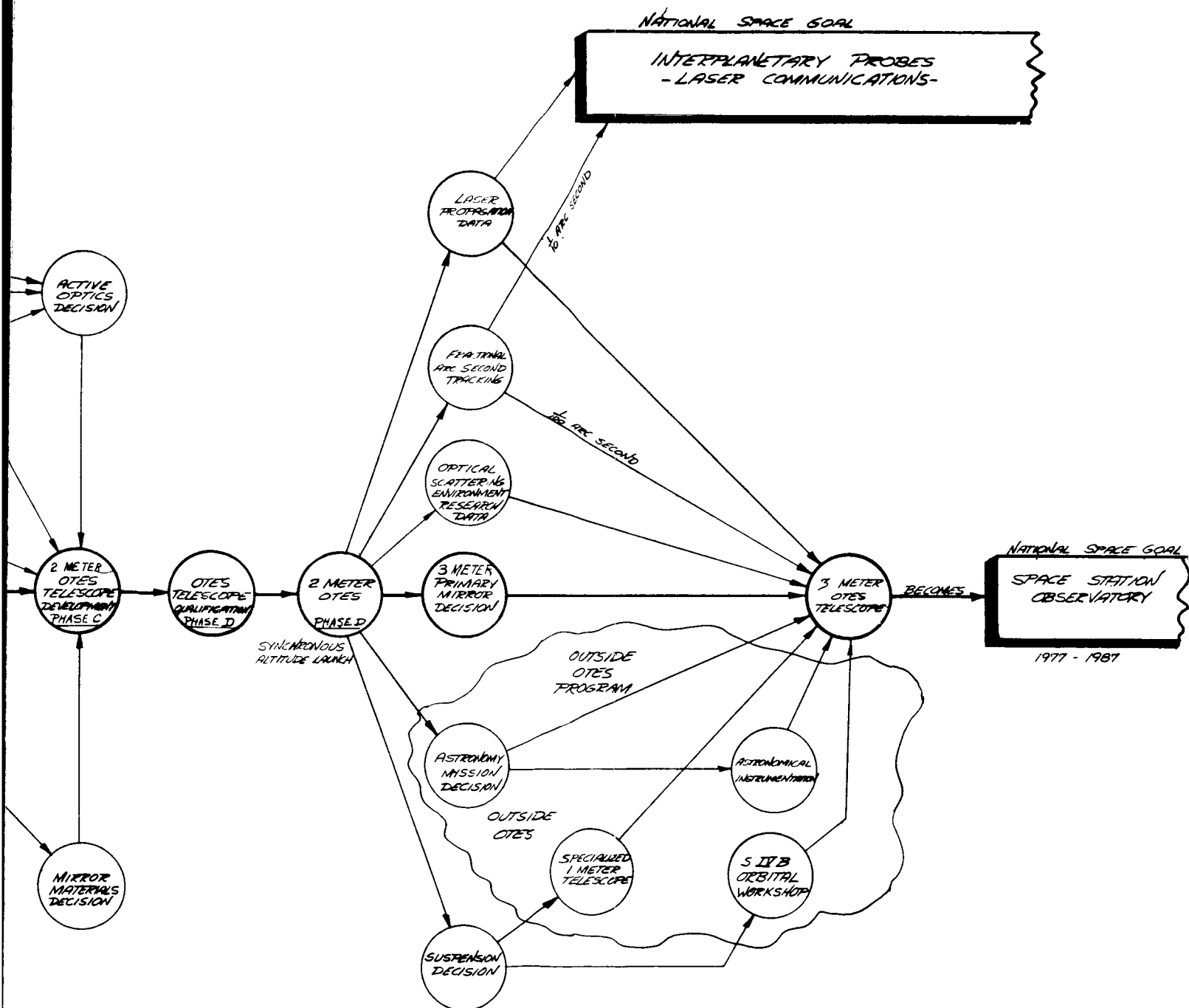


Fig. 1-3 NASA Space Optical Technology Plan Option 3

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1.4 SPACE OPTICAL TECHNOLOGY PLAN OPTION 4 - THE RECOMMENDED PLAN
(FIG. 1-4)

1.4.1 Key Features

Two flight experiments only as follows:

- Laser satellite experiment will be launched to synchronous orbit in 1970 for the development of laser propagation data.
- 2-m OTES telescope will be launched into low orbit in 1973 as part of the AAP cluster. This instrument would be designed to conduct experiments in large optical technology and act as a test bed before committing significant funds to the development of a 3-m telescope.
- The 3-m telescope is deleted from the OTES Program.

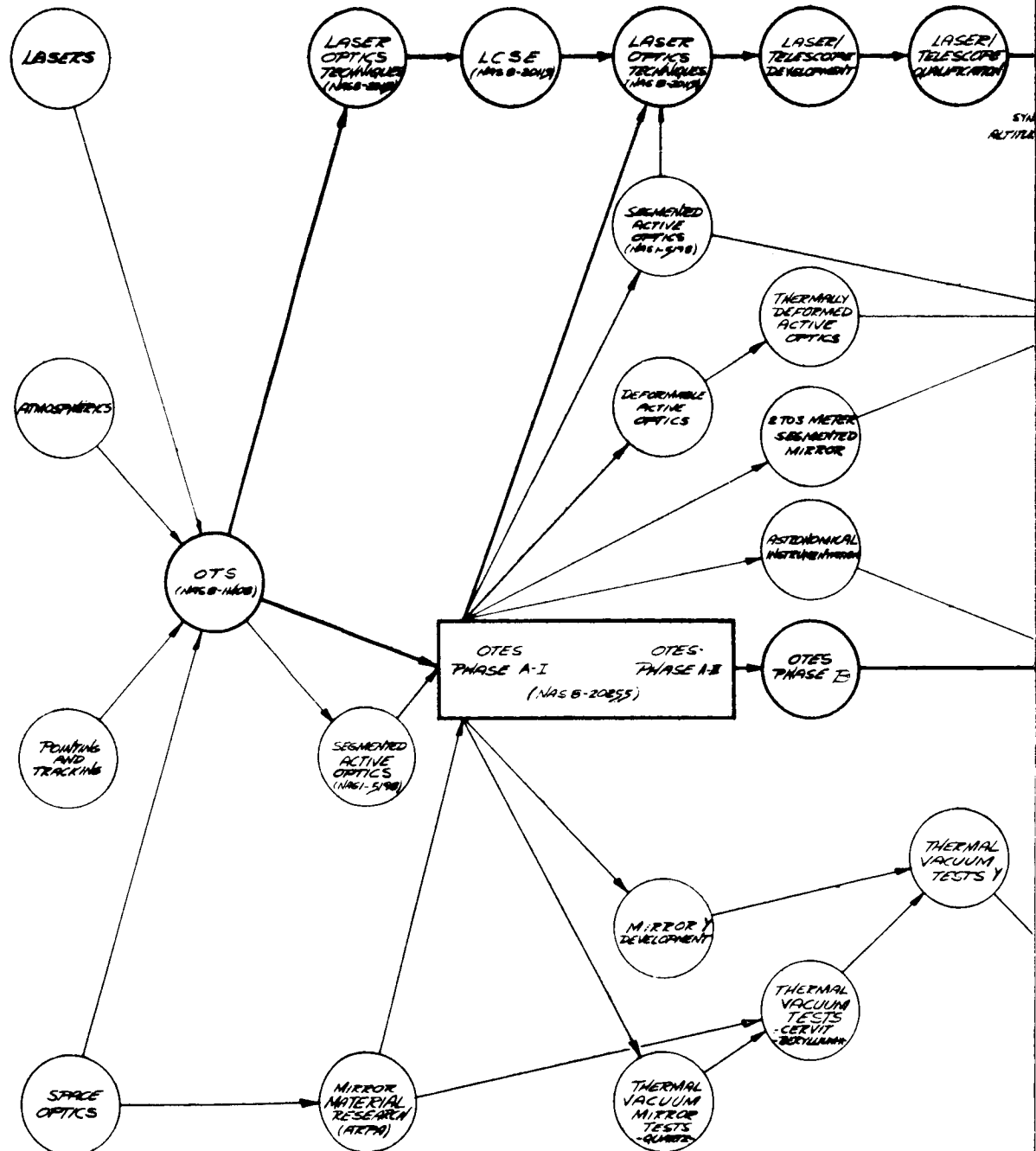
1.4.2 Equipment Budget Recommendations

<u>Item</u>	<u>Amount (10⁶ \$)</u>
Supporting technology (1968-1972)	8
Laser satellite experiments as part of AAP cluster at synchronous altitude in 1970	7.5
2-m OTES telescope as part of AAP cluster at a low altitude in 1973	<u>42</u>
Total	57.5

1.4.3 Basic Advantages and/or Disadvantages to NASA

- Timely technology development
- Conduct appropriate space experiments to develop the necessary data on the following subjects:
 - Up- and down-going laser propagation data
 - Daylight astronomy feasibility
 - Low-high orbit for space observatory
 - Optimum primary mirror for space observatory
 - Optimum suspension technique for future telescopes

63	64	65	66	67	68	69
FOUNDATION	INITIAL	LABORATORY	SPECIFIC	EQUIPMENT		
OPTICAL	SYSTEMS	DEVELOPMENT	SPACE	DEVELOPMENT		
TECHNOLOGY	STUDY	AND	EXPERIMENT	AND GROUND		
R & D		EXPERIMENTS	STUDIES	TESTS		



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70

71

72

73

74

75

76

SPACE
EXPERIMENTS
(LASERS)

SPACE
EXPERIMENTS
(OPTICS)

SPACE
EXPERIMENTS
(SPACE STATION
OBSERVATORY)

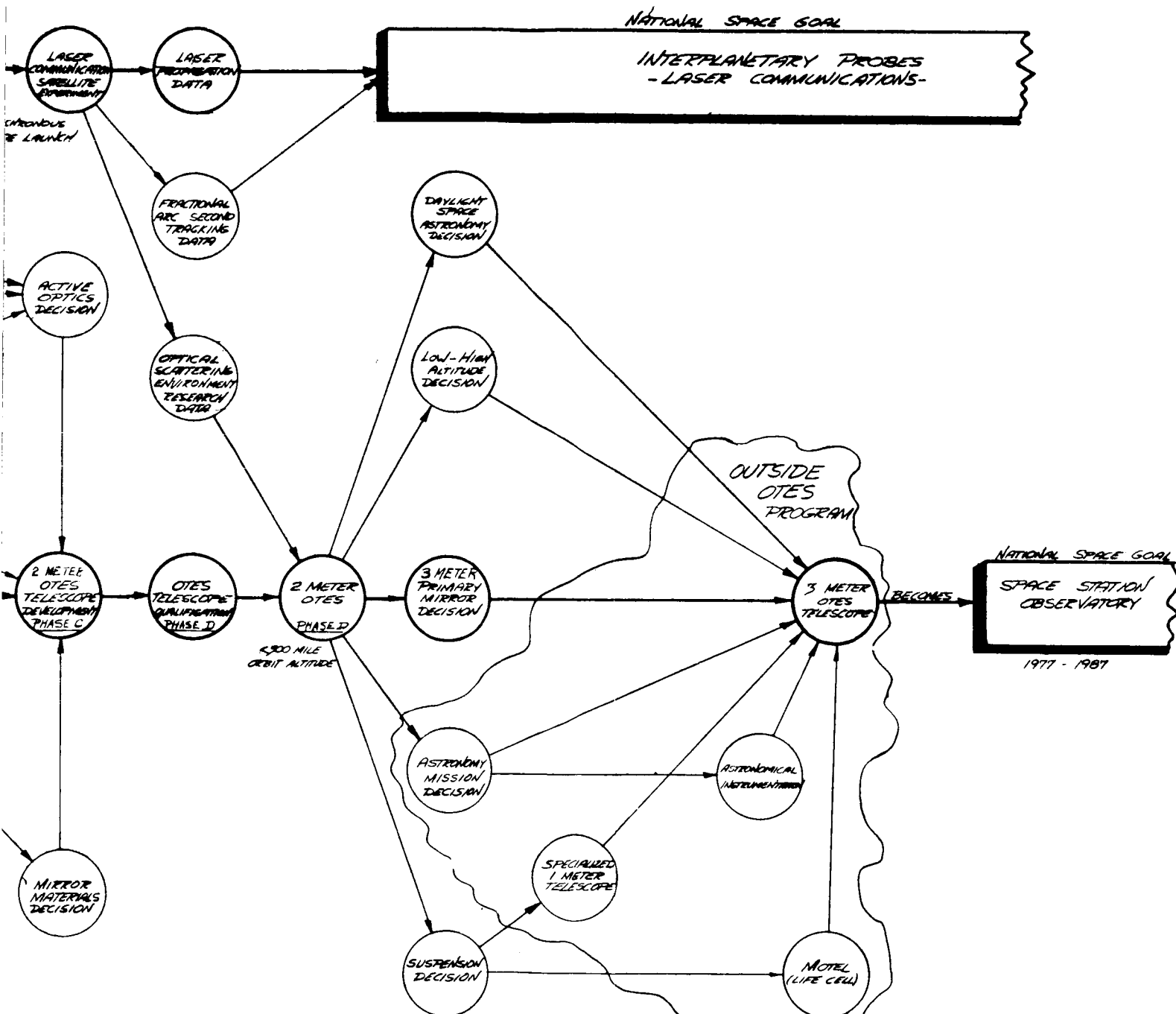


Fig. 1-4 NASA Space Optical Technology Plan Option 4

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1.4.4 Option 4 Resources Summary

Following the identification of Option 4 as the recommended option, preliminary analyses were performed to identify what resources would be required to implement the preferred conceptual design for the 2-m telescope system. Primary areas of planning included technical program, test program, facilities requirements, and schedule and cost requirements. Results of these efforts are presented in the following sections of this Phase A Report. The results presented herein are based upon conceptual design and preliminary program definitions. Hence, in spite of the fact that considerable effort has been expended to provide realistic cost and schedule estimates, the resulting information must be considered to be the "best available" estimates for use in continued program definition.

The technical or program plans developed during the final phase of the conceptual design studies are based upon a 1973 OTES launch, and recognize the optics and electro-optical equipment as the critical pacing items for program development. As a result, the program planning identifies some overlap of the phases (Phases C and D) of the standard NASA phased program development scheme.

Test requirements and test planning activities are fully compatible with standard NASA policies. In general, testing is sequenced for maximum use of NASA facilities and testing procedures. Research simulation and test activities identified for development of realistic astronaut task data are considered to be in the research rather than development-testing phase of development. Therefore, the topic has been treated in the crew-participation analysis activity and is not included under test planning.

Facilities planning investigations indicate that existing NASA Apollo and proposed AAP facilities are adequate for spacecraft development, test, checkout, and operational phases of the OTES Program.

Conceptual level schedule and cost summaries for the presently envisioned spacecraft portion of the OTES Program are presented in Figs. 1-5 and 1-6. The cost summary

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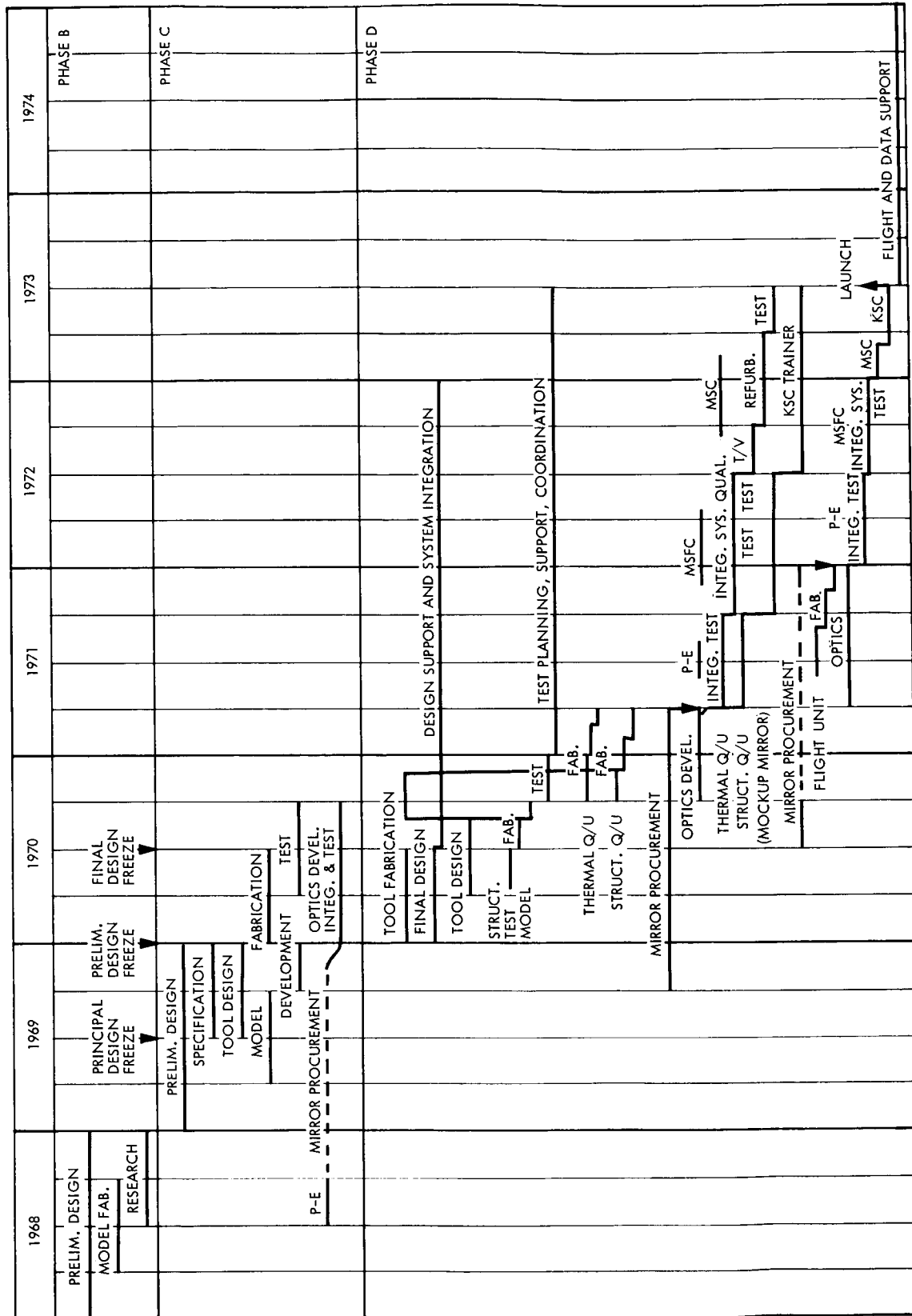


Fig. 1-5 OTES Program Schedule

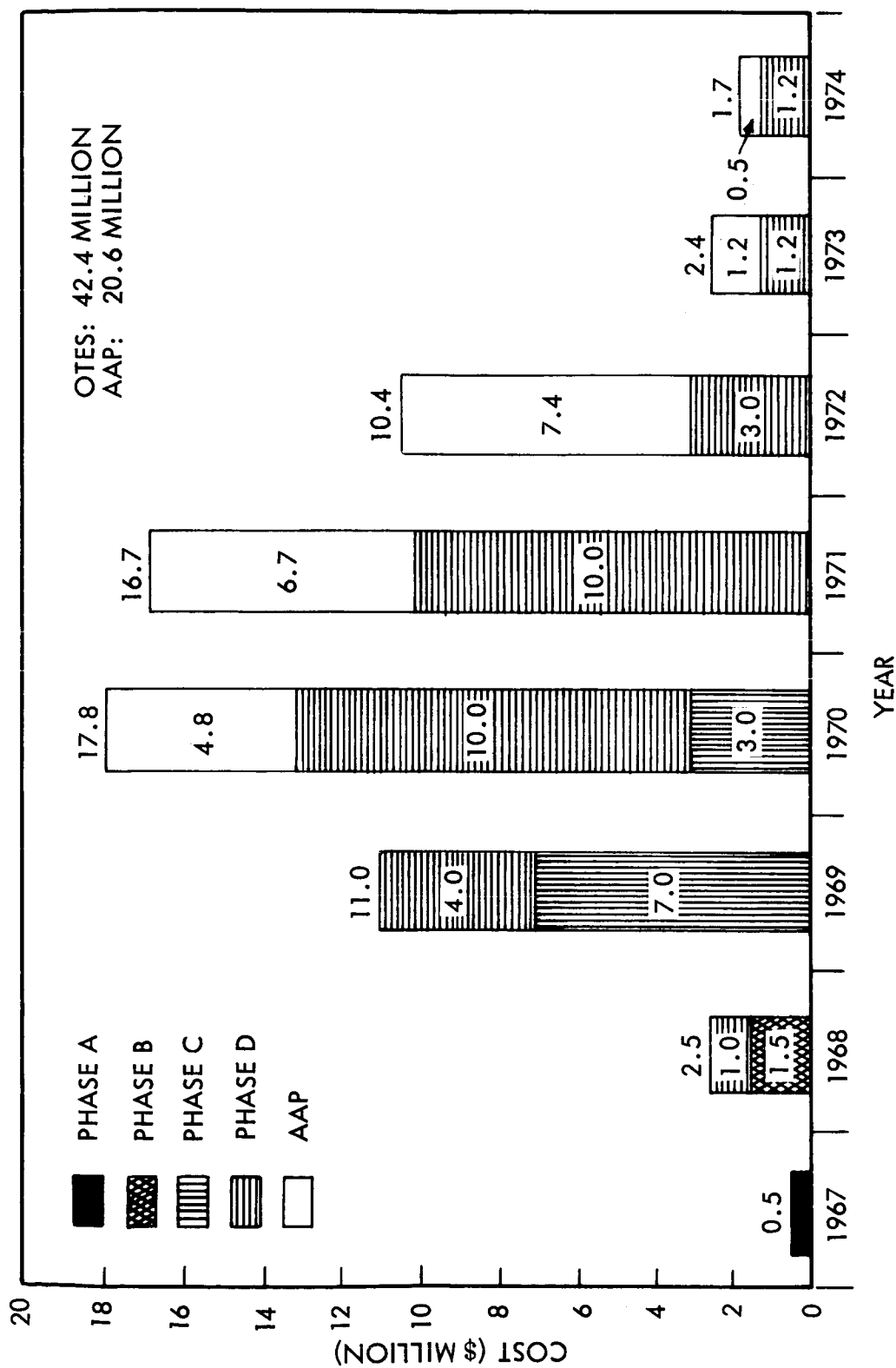


Fig. 1-6 OTES Costs

includes costs for those functions necessary to design, manufacture, test, and check-out the spacecraft portions of the OTES Program. The following items are specifically excluded from cost considerations:

- NASA inhouse costs
- AAP payload integration
- Launch vehicle
- Launch range
- NASA test facilities
- Operations
- Related supporting technology

In some cases, portions of these excluded costs are presented for information or comparative purposes but are not included in the OTES Program cost summary.

Section 2

PRELIMINARY TECHNICAL PLAN

The Technical Plan for OTES embodies all of the basic contents of the other four plans (Facilities, Test, Schedule, and Cost), but to a broader level of detail. It identifies the major classes of engineering, manufacturing, and testing activities; the major classes of flight and support hardware; and the schedule and cost increments appropriate to each class. The plan provides a discussion of the assumed baseline and concurrent inputs to the OTES Program, as well as the assumed NASA guidelines, ground rules, and constraints. Statements of the objectives to be achieved in each program phase and major activity class are followed by discussion of the methods recommended for achieving them. As a summary of other plans, the Technical Plan displays identifiable activities and outputs from the assumed January 1968 go-ahead for Phase B, through the assumed July 1973 launch, until termination of flight-support activities at the end of 1974.

2.1 APPROACH

Optical Technology Experiment System development poses many unique challenges to total systems management and to spacecraft development resulting from the complex technical and administrative interface between spacecraft and optical disciplines and the complexities of coordinating OTES activities at Huntsville, Houston, and Cape Kennedy with a plethora of other AAP major experiment payloads.

The Technical Plan and related subordinate plans propose to pace all spacecraft development activities and, to the extent possible, all payload integration activities to the optical subsystem development. To this end, data pertaining to launch and orbital environments and to astronaut EVA capabilities are provided when optical support requirements are still being crystallized in Phase B. Likewise, a full-scale

model of the preliminary design is provided to support optical experimental investigations. Test data on flight-level spacecraft hardware will be provided before completion of optical subsystem preliminary design.

The Perkin-Elmer/LMSC approach to coordinating with other AAP payload integration activities is to provide ample spans of time at the NASA centers for performing all OTES tasks and to divide activities into segments which can be shifted forward or backward interchangeably by increments of several weeks to accommodate unforeseen AAP schedule conflicts.

2.1.1 Assumptions

All data presented, conclusions reached, and recommendations made in the Resources Plans are founded on the following assumptions:

- All designated Government facilities, equipment, and services, as well as designated AAP engineering, manufacturing, and test support to OTES, will be available on a no-cost basis.
- The OTES will be physically integrated into an Apollo Telescope Mount (ATM) rack assembly which, with its flexure-pivot three-axis gimbal system, will be GFE hardware.
- Although the proposed program for the 2-m telescope concept will develop technology applicable to the development of a 3-m or larger diameter telescope system, resource requirements for a larger system are not detailed in this report.
- The OTES telescope/rack assembly will be mated to a Lunar Module (LM) which will be physically modified to conform with OTES drawings and specifications at no cost to OTES. (One of the OTES design objectives is to minimize such modifications.)

- The OTES telescope/rack/LM will be carried to orbit (e.g., 250 nm) in July 1973 by a Saturn IB launch vehicle as a principal experiment of a multi-spacecraft "cluster" similar to Cluster A (currently being defined by AAP). For purposes of this plan, the OTES mission cluster is identified as Cluster X.
- Spares provisioning and planned logistic resupply during orbital operations is consistent with a 2-yr operational life requiring resupply and maintenance.

2.1.2 Objectives

This subsection presents some typical objectives of the OTES Program phases (Table 2-1). These are developed from the broad, technical, and management goals of the total program down to detailed requirements for individual subsystems. From these few examples a pattern can be inferred that properly identifies OTES as a challenging technological undertaking.

Payload integration and qualification test operations will readily lend themselves to schedule shifts and interchanges by spans of several weeks in order to mesh flexibly with AAP master schedules.

It is assumed that the AAP programmed support equipment, facilities, and operations will be employed in the OTES Program. Included in this category of collateral support capabilities are the AAP assembly and test areas at MSFC, MSC, and KSC; conjunctive support of mission control, experiment control, and data management operations; and collective planning of logistic resupply systems.

2.1.3 Methods

Methods designed to achieve the preceding objectives are discussed in this subsection (Table 2-2). As before, to present a revealing cross section of the LMSC approach, the methods shown range from broad to specific.

Table 2-1

OTES PROGRAM OBJECTIVES

Level	Objective
OTES Program	Provide an experimental flight demonstration of the technical and operational feasibility of a large space optical system capable of long-term (several orbits) 0.01 arc sec, diffraction-limited, star observation. This optical system will have sufficient operational lifetime for its performance to be confidently extrapolated over a 10-yr period.
Program Definition (Phase B)	From the various feasible concepts studied in Phase A, the single most promising concept must be sufficiently refined in Phase B to produce a Preliminary System Specification if the launch schedule is to be met. A complete set of preliminary functional plans will be provided in Phases C and D to outline all project activities.
Design (Phase C)	Design of the selected concept will be carried forward, supported by analysis and experimental testing, sufficiently to produce a complete set of design drawings and a firm System Specification. These design documents, together with updated Functional Plans for Phase D, will clearly justify and support commitment of major government funding for the development, qualification, and operation of OTES in the period from 1970 to 1975. With respect to the development of mockups and models, a subordinate objective is to provide, at the end of the third quarter of this Phase, representative flight structures to be utilized in subsequent experimental investigations. At the time of model fabrication, design analysis will be 60 percent complete. Another subordinate objective in this phase is to provide a sufficiently complete detailed set of tool designs, except for minor subsequent revisions, so that flight hardware jigs and fixtures can be completed in the first 6 mo of Phase D.
Development/Operations (Phase D)	To the extent that the program launch date provides the time to do so, test results will be reviewed and corrective designs formulated before extensive work is accomplished on successive generations of hardware. A development and qualification program will be presented that will demonstrate, within the capability constraints of ground test facilities, that the flight hardware produced will achieve its operational performance objectives. The qualification program will provide for piecemeal generation of test results through the succeeding levels of component, subsystem, and system tests, culminating in complete integrated system tests.

Table 2-2

OTES PROGRAM METHODOLOGY

Objective	Method of Achieving Objective
0.01 arc-sec, long-term observation	The spacecraft contribution to achievement of this objective resides principally in the promised capabilities of the ATM attitude-control subsystem with its CMG's and three-axis gimbal. The optical fine pointing techniques will evolve out of the utilization of the transfer line, the free-float system, and the image movers described in Experiment 9.
Refinement of a single concept design	After NASA approval, the concept definition of a telescope and the experiments will be further developed and refined by intensified analysis and design efforts. Complete concept definition of all spacecraft subsystems will be dependent on and paced by the selected experiments and primary mirror development. In the last half of Phase B, all-metal mockups of portions of the spacecraft structure related to astronaut EVA tasks will be fabricated from conceptual drawings and utilized in the LMSC Biotechnology Laboratory for 0-g (dry) and underwater simulations. The human factors data thus derived will be introduced into the design of Phase C.
Development of a complete set of preliminary functional plans for Phases C and D	After first preparing a detailed master schedule identifying all principal activities and interfaces (to the extent possible in Phase B), the following plans (others may be added) will be drafted: • Management • Engineering • Configuration Control • Quality Assurance and Reliability • Integrated Test • Transportation • Safety • Schedule and Cost • Manufacturing • Facilities • Payload Integration • Logistics • Training • KSC Operations • Mission Operations • Flight Data Operations
Complete set of design drawings and a firm system specification	Achievement of this objective will be brought about by instituting a principal design freeze at the midpoint of Phase C which is paced to a simultaneous freeze of the optical subsystem interface. The design work performed in the last 6 mo, supported by hardware demonstrations of certain design features, will then provide the firm design definition in support of Phase D commitment.
Completion of a functional engineering model by August 1970	By developing the 2-m mirror in 1968 through the SRT and commencing fabrication ahead of the principal design freeze and then retrofitting the minor concept changes, this model can be completed as scheduled. After the design changes brought about by design-analysis results and interface feedback from Perkin-Elmer are finalized, Perkin-Elmer will update the models in their shops for the remaining year of their optical subsystem experimentation.

Table 2-2 (Cont.)

Objective	Method of Achieving Objective
Complete tool design definition at the end of Phase C	By commencing tool design at the time of principal design freeze and pacing the definition of tool configurations to finalization of the spacecraft design, this activity will be essentially completed by the end of 1969.
Minimal overlap of testing and fabrication of next-generation hardware	This principle is emphasized to the maximum extent possible in the early portion of Phase D. For example, structural and thermal test results of the design engineering model (non-tooled) will be available before fabrication of the first (tooled) structural model. Likewise, test results of the latter model are available before final tool modifications leading to fabrication of the qualification prototypes. As development progresses, confidence in system reliability increases and greater overlap is acceptable.
Flight-hardware qualification program	Because of the particular criticality of the optical subsystem performance and its dependence on precisely known and controlled spacecraft parameters, the OTES qualification program provides full-scale verification testing to every level of Phase D hardware generation, including the flight unit itself.
Provision for flexible scheduling of integration operations	To satisfy AAP requirements for full and unbroken utilization of integration/test facilities and personnel, their master schedule will show, at least by 1970, all the operations required to support OTES. Because of the complexity of coordinating the large number of spacecraft and operations increments, AAP will probably find it necessary to reallocate relatively large blocks of time and support to maintain launch schedules. Therefore, each major OTES integration or test operation will be separately planned, specified, and implemented to permit drastic schedule shifts. For example, if the MSFC structural test facility becomes loaded, work can commence instead on instrumentation of the thermal qualification unit.

2.2 PRODUCT

The deliverable products of the OTES Program are generally defined by NASA Phased Project-Planning Guidelines. That is, the Preliminary Analysis Phase (A) delivers intermediate and final reports of feasibility and resources; the Project Definition Phase (B) delivers a complete set of preliminary functional plans and a preliminary design and analysis report; the Design Phase (C) delivers a firm System Specification and a complete set of design drawings and functional plans; the Development/Operations Phase (D) delivers the flight, qualification, and support hardware, as well as a set of flight reports which provide spacecraft performance and experiment results.

2.2.1 Reports

Reports to be provided during OTES Program are summarized as follows:

- Phase A. Feasibility Study Report, Multi-Concept Study Report, Concept Definition Report, Resource Plans
- Phase B. Research Report, Design and Analysis Report, Preliminary Functional Plan, Preliminary System Specification
- Phase C. Design Report, Analysis and Test Report, Functional Plans, Firm System Specification
- Phase D. Design and End-Item Specifications, Project Status Report, Qualification Report, Operations and Training Manuals, Flight Reports

2.2.2 Flight Hardware

Flight hardware comprises not only the OTES telescope/rack/LM but also the launch vehicles, orbital cluster spacecraft elements, and the periodic logistic resupply

flights. Some flight hardware will be developed under the OTES contract; other items will be provided GFE from other contract outputs; still other hardware items will be modified from GFE.

Table 2-3 identifies flight hardware by title and function, and indicates whether it is OTES new development, Government furnished, or OTES modification of GFE.

Table 2-3
FLIGHT HARDWARE

System	Function	Class*
OTES	Experiment Payload	P-E/LMSC
OTES Spacecraft	Structural and Mechanical	LMSC
OTES Rack	Telescope Support	MOD
OMRF	Orbital Mirror Recoating Facility	LMSC
OTES Optics	—	P-E
LM/A	Telescope Life Cell	GFE
LCSM	Logistic Vehicle	GFE
OWS	Orbital Workshop	GFE
AM	Airlock Module	GFE
MDA	Multiple Docking Adapter	GFE

*GFE = Government Furnished Equipment
MOD = Modified GFE
LMSC = New Development by LMSC
P-E = New Development by Perkin-Elmer

A further definition of the OTES flight system is provided in Table 2-4.

Table 2-4

FLIGHT SYSTEM BREAKDOWN

System Level				System Level 3 Item Descriptions	System Level 4 Item Descriptions
1	2	3	4		
Optical Technology Experiment System	Telescope Subsystem	Structure and Mechanisms Subsystems	Primary Mirror Support Fixed Shell Section Telescoping Sun Shade Sun Shutter Assembly Equipment Compartments Secondary Mirror Support Quartz-Rod Spacer Assembly Gimbal Disengage Assembly CG Positioning Assembly Launch Caging Assembly	<u>Gimbal Subsystem.</u> Pitch/yaw rings and roll track are actuated in three axes to provide ± 2.5 arc sec pointing; rate gyro pickoffs provide feedback to attitude-control subsystem.	<u>Primary Mirror Support.</u> Supports the mirror and its actuators; provides thermal and dynamic isolation.
			Optics Subsystem	<u>Attitude-Control Subsystem.</u> Within the ± 0.5 -deg deadband of the CSM reaction control system, the control-moment gyros absorb from and retransmit angular momentum to the entire AAP cluster to reduce the deadband to ± 1.0 min; three star trackers provide coarse and fine error signals.	<u>Fixed Shell Section.</u> Ties the mirror support to the rack gimbal; supports the telescoping sun shade.
	Rack Subsystem	Gimbal Subsystem	Structural Assembly Pitch Actuator Assembly Yaw Actuator Assembly Roll Actuator Assembly	<u>Electrical-Power Subsystem.</u> Designed for either dependence on cluster power when docked to the MDA or separate solar panels for power generation when in an independent mode; the EPS contains sun-tracking solar panels, rechargeable batteries, and power switch gear.	<u>Telescoping Sun Shade.</u> Five nested tubular sections extend on orbit to provide visual and thermal protection from sunlight.
			Attitude-Control Subsystem	<u>Rack Structure.</u> The basic rack structure is that of the ATM. Additional structure is added for launch caging and the magnetic suspension assembly.	<u>Sun Cap Assembly.</u> A hinged circular lid on the end of the tubular sun shade; prevents sunlight from entering the tube, when closed, and during some telescope orientations, when open.
		Electrical Power Subsystem	Solar Panels Electromech. Actuators Charge Controller/ Battery/Reg. Modules Control Distributor Solar Power Distributor	<u>Thermal-Protection Subsystem.</u> The insulation assembly is a new development keyed to the requirement for a 2-m mirror.	<u>Equipment Compartments.</u> Six modular wedge-shaped containers around a central cylinder; contain electro-optical subsystems.
			Rack Structure Thermal Protection Subsystem	<u>Optical and Laser Subsystems.</u> See Section I-3 of report.	<u>Secondary Mirror Support.</u> Structure that supports mirror and its servo actuators.
		Magnetic Suspension System	Basic Structure Struct. Mod. Kit Assembly Insulation Assembly	<u>Spectrometers.</u> See Section I-3 of report.	<u>Quartz-Rod Spacer Assembly.</u> Mechanism that supports, unfolds, protects, positions, and spring-loads the rods.
			See Experiment 9		<u>Figure Sensor Support.</u> A structure attached to outer tube section which supports the figure sensor package.
	Experiments Subsystems	Various	See experiments Section I-2		<u>Gimbal Disengage.</u> A mechanism to slide the fixed shell section 8 in. (approx.) out of the gimbal until engaged by the magnetic suspension assembly.
	Orbital Mirror Recoating Facility				<u>CG Positioner.</u> A mechanism to shift a heavy battery rack longitudinally; shifts nominal telescope CG 3 ft from the plane of the gimbal flexure pivots to the plane of the experimental magnetic suspension system.
					<u>Launch Caging.</u> An ejectable structure that transmits launch-induced forces into the SLA, bypassing the lightweight, nested telescope.

2.2.3 Support Hardware

This subsection describes classes of hardware other than OTES flight hardware and the primary AAP cluster spacecraft listed in subsection 2.3.2.

Table 2-5 lists the classes of support hardware and identifies certain major equipment items in each class. Equipment functions can be deduced from their titles. For each item the table indicates whether it is newly developed, already existing, or Government furnished.

2.3 PLAN

In this section, broad classifications of tasks, schedules, and costs are delineated. For greater detail, see Section 5.

2.3.1 Tasks

Phase A.

- Part I – Experiment Study and Feasibility Analysis
- Part IA – Conceptual Definition
- Part II – Conceptual Refinement

Phase B.

- Research
- Spacecraft Modification and Preliminary Analysis
- Preliminary Design
- Development Planning

Phase C.

- System Integration
- Preliminary Design
- Design Analysis

Table 2-5

OTES SUPPORT HARDWARE

Class	Item
Research and Engineering	Zero g and underwater metal partial mockups for EVA simulations Mirror tests
Manufacturing Tooling	Master tool assembly fixtures and jigs for: • Six tubular shell sections • Mirror base • Seven equipment compartments • Rack structure modifications Various jigs, clamps, fixtures, frames, templates, and special tools
Development Test Prototypes and Equipment	Full-scale wood mockups (one for LMSC, one for P-E) Early (Preliminary Design) full-scale engineering model (one for P-E) Mechanism breadboards Thermal insulation and partial structure breadboards Test instrumentation, apparatus, data recorders, computer LMSC shake and shock tables, acoustic chamber, thermal/vacuum chamber
Qualification Test Prototypes and Equipment	Early (Final Design) full-scale engineering model (one for LMSC) Structural test model; the first tooled production unit (one for LMSC) Structural test model; the second tooled production unit (one for P-E) Thermal test model; third tooled production unit (one for P-E) GSE checkout console to be used for integrated system runs in conjunction with the RCA-110 computer at MSF Various test instrumentation, apparatus, and fixtures MSFC shake and shock tables and acoustic chamber MSC Chamber A (thermal/vacuum) with its associated operations control and data processing equipment
Acceptance Test and Checkout Equipment	Vertical stand used at MSFC for static balance (same as the first vertical stand listed below under Handling and Transportation Equipment) GSE checkout console to be used for integrated system runs in conjunction with RCA-110 computer at KSC ACE computer for launch pad system runs
Handling, Transportation, and Launch Equipment	Vertical support stand for deployment tests in Sunnyvale; also used at MSFC for weight, inertia, and balance determination of deployed telescope Vertical support stand for use in MSC Chamber A thermal/vacuum testing of telescope/rack/LM

Table 2-5 (Cont.)

Class	Item
Handling, Transportation, and Launch Equipment (Cont.)	Various support stands and dollies used in manufacturing testing Various containers, frames, and dollies used for transcontinental shipment Guppy or Super Guppy airplanes; ocean-going barge and tugboat Cranes, jibs, gantry, hoists, umbilical masts, etc., at KSC
Crew Training Equipment	Metal structural mockups used in Phase B research will also be used for EVA astronaut training at Sunnyvale or Houston OMRF test model for EVA training use Structural qualification model used for KSC ground crew training after qualification is complete Thermal qualification model used for manned mission simulation in Chamber A and for general flight crew familiarization
Launch Vehicles and Flight Support	Twelve S-IB vehicles needed; four for initial cluster activation and one every 90 days thereafter for 2 yr Instrument Unit (IU) structural shell spacers (for payload launch) Ten CSM spacecraft needed; two for initial cluster activation and eight for crew rotation and resupply during the 2-yr operational period Logistic spares; such as star trackers, CMG's, rate gyros, gimbal actuators, film cassettes, batteries, etc. Logistic resupply of crew, food, atmosphere, reactants, propellant, etc. Recovery fleet
Mission and Experiment Control	Kennedy tracking and control (launch) MSC Mission Control Manned Space Flight Network (MSFN) MSC Operations Control
Data Management	MSC Data Center

- Design Support
- Tool Design (Primary)
- Tool Manufacturing Plans
- GSE Design
- Model Fabrication
- Experimental Testing
- Development Testing
- Development Planning
- Report Preparation

Phase D.

- System Integration
- Final Design
- Design Analysis
- Design Support
- Tool Fabrication (Primary)
- Tool Design (Secondary)
- Tool Fabrication (Secondary)
- Model and Prototype Fabrication
- Quality Assurance and Reliability
- Development Testing
- Qualification Units Fabrication
- Flight Unit Fabrication
- Handling and Transportation
- Spacecraft Integration Liaison
- Qualification and Acceptance Testing
- Training Liaison
- Logistic and Spares Operations
- Prelaunch Checkout
- Launch Operations
- Flight Operations
- Data Acquisition and Reduction
- Data Processing and Reporting

OTES Associated AAP Integration (assumed not chargeable to OTES).

- Fabricate OTES Racks
 - One development model
 - Two qualification units
 - One flight unit
- Integration Engineering
- Spacecraft OTES Modifications
- Assembly and Installation to Saturn Launch Vehicle
- Qualification and Acceptance Testing

2.3.2 Schedule

The Master Schedule in Fig. 1-5 illustrates the salient features of the OTES Program.

In Phase B, the "model fabrication" refers to the metal partial mockups utilized during the "research span" for zero-gravity and neutral-buoyancy EVA simulation.

In Phase C, the three development models (Engineering, Structural, Thermal) needed by Perkin-Elmer to support optics subsystem development are shown being fabricated before and after the principal design freeze. This design freeze signals the start of primary tool design and development of the firm system specification. The model fabrication in 1970 is essentially the LMSC engineering model, fabricated before tooling becomes available, but after the preliminary design freeze. The separate periods of "development test" are for mechanism design verifications and structural thermal testing of the engineering model, respectively.

Fabrication of primary tools used for major structural assemblies) begins immediately after Phase D go-ahead. To facilitate this, principal tool production orders are written in Phase C. As the final design crystallizes, secondary tool design (for mechanisms and subordinate structure) begins. After fabrication of the LMSC structural test model, which is the first tooled unit, the secondary tooling is incorporated in fourth quarter

1970. Subsequently, the completed tooling is utilized in fabrication of the two qualification prototypes.

The qualification program carries the structural and thermal units in parallel to provide maximum scheduling flexibility at the AAP Integration Facility. Upon completion of qualification, the thermal model is refurbished in Building 36 at MSC and shipped to KSC as a flight backup unit. The structural model, which is provided with a mockup mirror, is shipped to KSC for use as a ground crew trainer. The thermal model will already have served as a flight crew indoctrination model at MSC.

Flight article fabrication is started as late in the program as its nominal acceptance and checkout spans allow, to take maximum advantage of qualification testing experience.

2.3.3 Costs

Costs for the spacecraft portion of the OTES Program are presented in Section 1 of this report. These costs cover all activities necessary to design, produce, and quality the 2-m telescope, optics, structures, and mechanisms. The laser experiment module and the f/10 film imagery module have been included. The Echelle module, Rowland circle module, f/50 electronic imagery module, and "spare" module are not costed in Phase A.

Types and numbers of support equipment, development, test, and flight hardware units envisioned as necessary for a complete program are identified in the cost and schedule summaries and are included in the various cost summaries.

The costs shown do not include any procurement or hardware cost for Apollo/AAP items. These are assumed to be GFE to the OTES project. Similarly, these costs do not include labor and services provided by government employees at any of the NASA centers or installations, or labor or services of other Government/contractor personnel except as indicated in the preceding paragraph. All costs are gross estimates consistent with

a Phase A conceptual definition and do not in any way constitute an offer to perform the effort indicated for the costs shown.

Figure 2-1 illustrates graphically the profile of P-E and LMSC man effort in the OTES Program. This figure illustrates the programmed overlap of Phases C and D. The anticipated sharp peaks of model and prototype fabrication activity, and the lesser peaks of development and qualification testing activity, have been averaged out to facilitate overall manpower requirements integration.

Costs for hardware produced in Phases B and C do not include research, design, or test labor, but only material and direct-manufacturing labor cost.

Cost of the test prototypes fabricated in Phase D include a pro-rata amortization of final design, design analysis, system integration design, quality assurance, and handling. Test prototype cost also includes the total cost of the testing performed on each model. The refurbishment and retest cost involved in preparing the thermal qualification unit as a flight backup is not included in the thermal model cost. Refurbishment effort at MSC is assumed to be GFE.

Flight unit cost includes its pro-rata share of final design, analysis, etc. (like the test prototypes), but also includes prelaunch checkout and launch operations. Other Phase D costs not charged to hardware include design support, logistics, flight, and data operations.

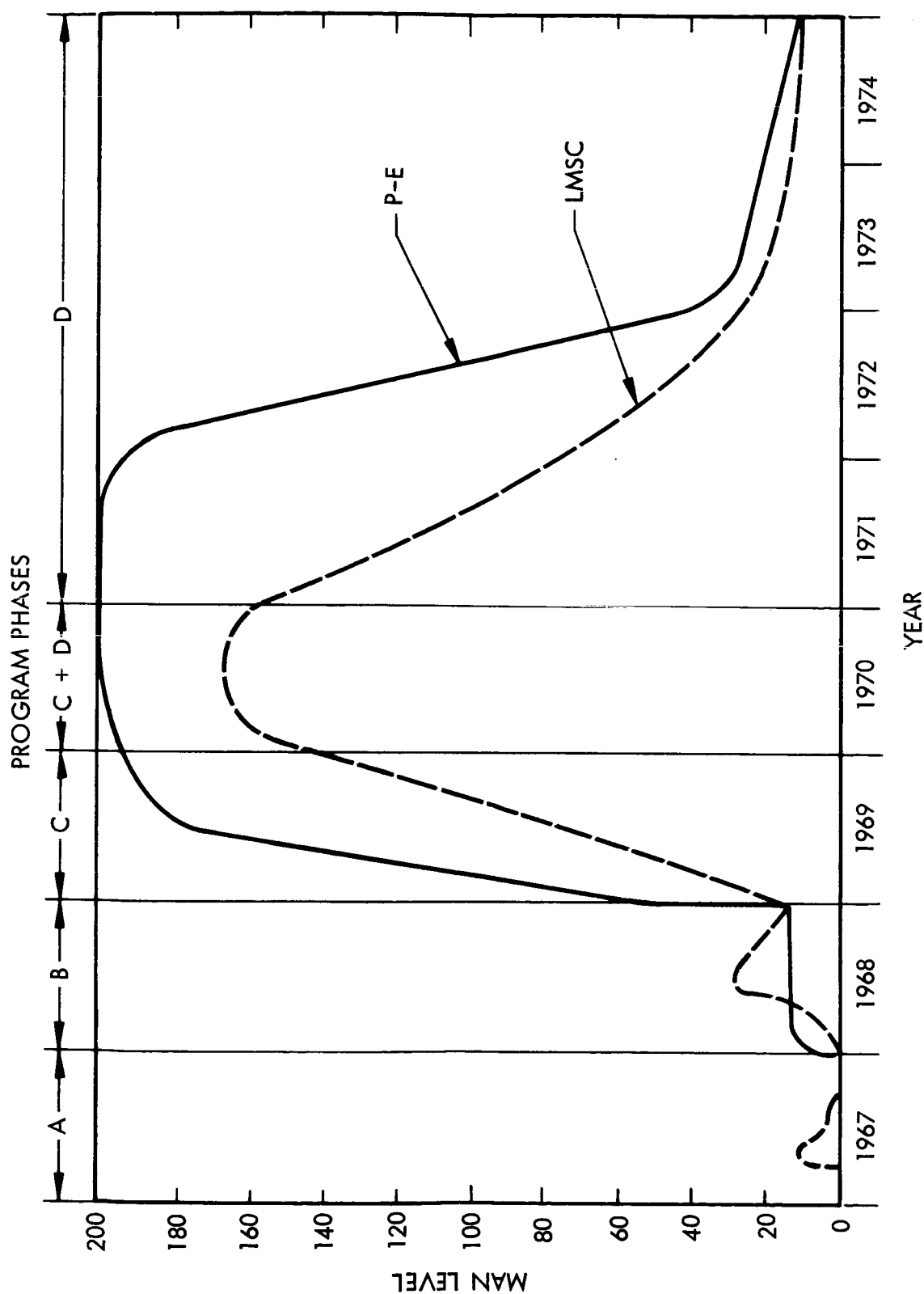


Fig. 2-1 P-E and LMSC OTES Manpower

II-210a

Section 3

FACILITIES PLAN

3.1 SCOPE

A preliminary investigation was made of the facilities required to support the OTES experiment and spacecraft integration. Analyses were made to:

- Identify general and special OTES facility requirements
- Determine which existing government, Perkin-Elmer, and Lockheed facilities are capable of effectively supporting OTES functions
- Establish preliminary cost and schedule requirements for facility augmentation and activation

The study approach considered the accommodations necessary to support the following general OTES activities:

- Engineering – office, desk and board, and full-scale mockups
- Fabrication and subassembly – OTES and associated hardware buildup and subassembly
- Carrier preparation – rack modification and OTES subsystem installation
- OTES integration – final installation and integrated systems test
- Manufacturing operations, quality assurance, and other general support

The results of this analysis provided the foundation for the facilities plan described in the following paragraphs.

3.2 BASIC OTES FACILITY REQUIREMENTS

The basic space requirements to support Perkin-Elmer and Lockheed OTES activities represent an analysis of current program schedules and manpower forecasts. Facility

requirements are categorized by organizational areas and include space required at the AAP Payload Integration Facility as well as support areas in Huntsville, Sunnyvale, and Norwalk, and sites near KSC and MSC. Various levels of environmentally controlled areas are provided at the AAP Payload Integration Facility, and should be adequate for the OTES Program.

3.2.1 Specialized Test Capabilities

Requirements for specialized test capabilities to support the OTES Program include vertical vacuum optical tunnel; class 100,000, class 10,000, class 100 clean rooms; large capacity thermal/vacuum chamber; vibration testing; and anechoic chamber. It is currently planned to utilize the Space Optics Facility at Perkin-Elmer in its entirety and specialized test facilities recommended by LMSC and summarized in Table 3-1.

The Perkin-Elmer Space Optics Facility was designed and built specifically for the OTES Program (previously designated OTS) as a model space optical project that would require specialized and unique facilities. The special capabilities of the OTES facility are shown in Table 3-2.

The breakaway concept of the OTES facility is shown in Fig. 3-1. The costs for this facility are not included in the project costs on the assumption that this building could be cleared of the projects by the time the OTES is ready to go into Phase C (i.e., in 1969).

3.2.2 OTES Integration Facility Capabilities

Since the recommended 2-m OTES concept is envisioned as an integral experiment of the AAP program, AAP supporting facilities were considered in the OTES facility planning effort.

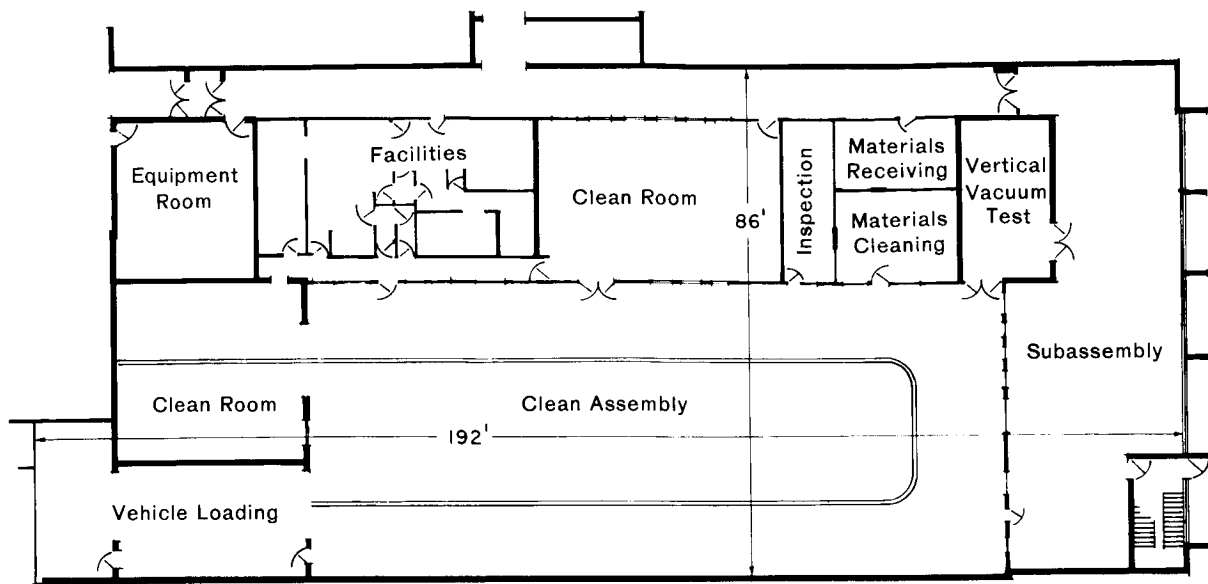
The P-E/ LMSC plan provides for final integration of the OTES into the AAP spacecraft to be accomplished in the AAP Payload Integration Facility at Huntsville. It

Table 3-1

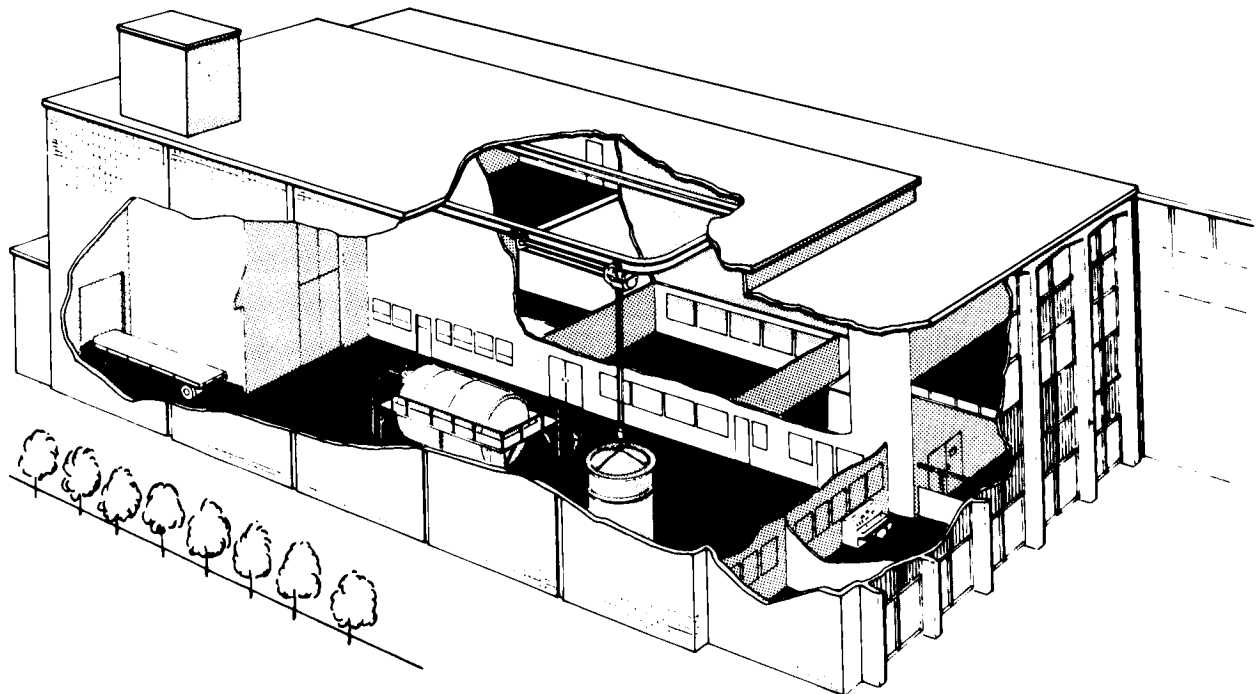
SPECIALIZED TEST FACILITIES

Facility	Specimen Size	Description
<u>THERMAL/VACUUM CHAMBERS</u>		
<u>Large Capacity</u>		
• Arnold Engineering Development Center, Tullahoma, Tennessee ②	20 ft dia × 50 ft ht	10 ⁻⁹ torr, solar and cold simulation
• NASA Goddard Space Flight Center, Greenbelt, Maryland	30 ft dia × 40 ft ht	10 ⁻⁹ torr, solar and cold simulation
• NASA Manned Spacecraft Center, Houston, Texas ①	40 ft dia × 75 ft ht (150,000 lb)	10 ⁻⁵ torr, solar and cold simulation - man-rated
• Lockheed HIVOS Chamber, Bldg 104, Sunnyvale, California ③	12 ft dia × 15 ft ht	10 ⁻⁶ torr, solar and cold simulation
• Lockheed STARS Chamber, Bldg 152 Sunnyvale, California	18 ft × 35 ft	10 ⁻² torr, vacuum - man-rated
• Lockheed Cryogenic Space Flight Simulator, Santa Cruz Test Base, California	15 ft × 25 ft	10 ⁻⁶ torr, cold simulation
<u>Medium Capacity</u>		
• MSFC, Bldg 4708, Huntsville, Alabama ①	15 ft dia × 20 ft	10 ⁻⁶ torr, vacuum
• Arnold Engineering Development Center, Tullahoma, Tennessee	7 ft × 12 ft	10 ⁻⁸ torr, solar and cold simulation
• Lockheed HATS, Bldg 102, Sunnyvale, California ③	180 cu ft	10 ⁻⁵ torr, solar and cold simulation
• Lockheed TASC, Bldg 104, Sunnyvale, California ②	6 ft × 25 ft	10 ⁻⁶ torr, solar and cold simulation
<u>VIBRATION TESTING</u>		
Manned Spacecraft Center, Houston, Texas ②	65 ft × 85 ft × 30 ft ht	22,000-force-lb shaker; 30,000-force-lb shaker (can be coupled)
MSFC, Huntsville, Alabama ①	60 ft × 60 ft × 90 ft	Six 10,000-force-lb thrusters
	100 ft × 100 ft × 360 ft	Thruster for dynamic testing of Saturn V
	Subsystem and component	28,000-force-lb shaker
Lockheed, Sunnyvale, California	16 ft dia × 25 ft	Two 30,000-force-lb shakers (can be coupled)
Wyle Laboratories, Huntsville, Alabama ③	160 ft × 50 ft × 60 ft	Eight 50,000-force-lb hydra shakers (can be coupled) Three 200,000-force-lb hydra shaker systems
<u>LARGE ACOUSTIC REVERBERATION CHAMBERS</u>		
Wyle Laboratories, Huntsville, Alabama ①	25 ft × 55 ft 45 ft × 55 ft	Frequency range 20 to 10,000 cps, maximum overall sound pressure: 155 db, 20,000 acoustical w
Manned Spacecraft Center, Houston, Texas ②	30 ft × 85 ft	Frequency range 30 to 10,000 cps, maximum overall sound pressure: 171 db, 160,000 acoustical w
MSFC, Huntsville, Alabama ③	18 ft × 15 ft × 20 ft (est), 4846 cu ft	40,000 acoustic w
<u>ANCHOIC EMC CHAMBERS</u>		
Manned Spacecraft Center, Houston, Texas ②	55 ft × 55 ft × 150 ft	Reflection of RF fields from -30 db to -55 db at 3000 Mc and above
MSFC, Huntsville, Alabama ①	33 ft × 33 ft × 120 ft	Assumed to satisfy MIL-I-26600
Lockheed, Sunnyvale, California ③	30 ft × 30 ft × 60 ft	Meets MIL-I-26600

○ Order of Preference Recommendation



Overall Facility View



Floor Plan

Fig. 3-1 OTES Facility

Table 3-2
BASIC OTES FACILITY CHARACTERISTICS*

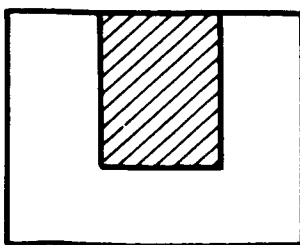
Area	Characteristics
Clean Assembly Areas	Class 100,000 – 28 ft high, 6,000 ft ² Class 10,000 – 28 ft high, 900 ft ² Class 100 – 10 ft high, 900 ft ²
Vehicle Loading Area	Class 100,000 – 35 ft high, 600 ft ²
Vertical Vacuum Test Tunnel	Capacity to Test Optical Systems up to 10 ft diameter
Equipment Control Area	Remote operation of equipment in clean room Windows for monitoring operations

* Facility is adjacent to existing engineering, manufacturing, and environmental facilities.

also provides for an OTES/AAP interface management and liaison engineering office in the Lockheed-Huntsville facility and for the primary spacecraft engineering and manufacturing in the Lockheed-Sunnyvale Plant. A concept of the MSFC-Lockheed Huntsville Site Plan recommended for the AAP program is shown for reference in Fig. 3-2.

Concepts of the buildings comprising the Payload Integration Facility in Huntsville (at the time of this report writing) are as follows:

- Receiving, Receiving Inspection, and Storage – Building 4752. The north end of MSFC Building 4752, to be used for receiving and storage of AAP hardware,



is capable of handling the OTES requirements for receiving, inspection, and shipping container storage. Existing facilities in this building provide capability for comprehensive inspections, analyses, and tests.

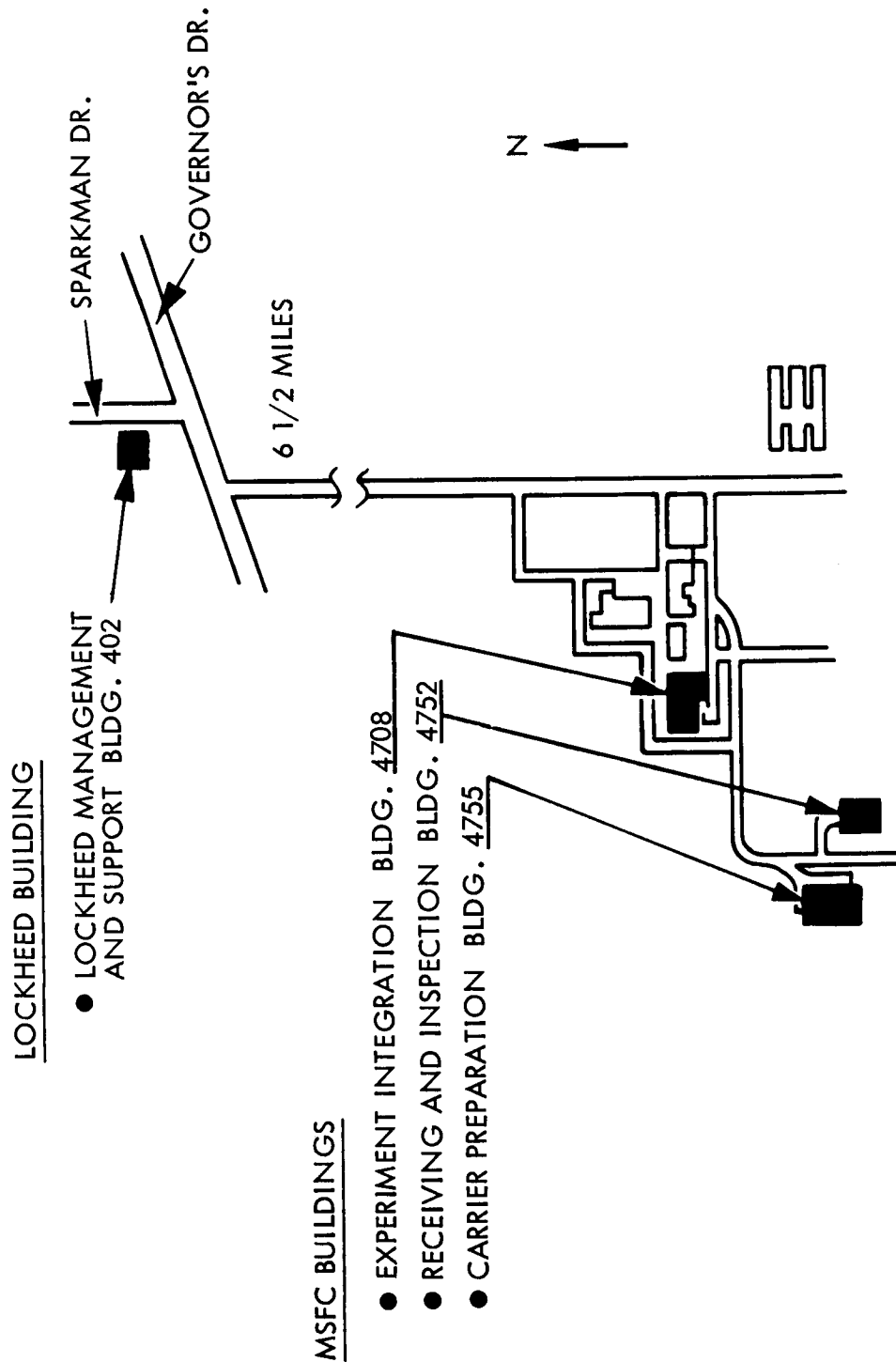


Fig. 3-2 MSFC/Lockheed Huntsville AAP Facility Concept

Alternate Facility Plan is selected. Two high bays provide 56,250 ft² for carrier modification, subsystems and experiment installation, and integrated systems test. Complete overhead crane coverage is provided with 25-ton capacity and 40-ft hook height. The bay used for experiment installation and integrated systems test is adjacent to the checkout equipment area and is environmentally controlled to Lockheed Class O-M cleanliness level. Building 159 includes a two-story bay adjacent to Integrated Systems Test which will provide 26,000 ft² for housing checkout equipment, test engineers, experimenter, and reliability and operations personnel. Design engineering and indirect support will be housed in nearby existing buildings at Sunnyvale. Existing Lockheed facilities that will be used at Sunnyvale include:

- Building 102: computer center
- Building 103: machine shop and processing
- Building 104: component testing
- Buildings 151 and 152: electronic fabrication and test

3.3 FACILITIES SUPPORT PLAN

3.3.1 Spacecraft Fabrication and Assembly

Fabrication of tools and hardware components for the OTES Program will be provided by the Lockheed Sunnyvale general support shops in Buildings 103, 151, and 152. Assembly of OTES models and storage of project tooling and fabricated parts will be accommodated by utilizing a portion of Lockheed-Sunnyvale Building 152. This assembly area will be activated in three increments in conformance with the OTES hardware development plan.

Initial assembly area activation in the third quarter of 1968 will provide 2,000 ft² for assembly of full-scale models of sections of the scope assembly required during Phase B. The second phase activation will provide a cumulative total of 5,600 ft² beginning the second quarter of 1969. This activation will accommodate assembly

of up to three OTES models and provide approximately 2,000 ft² of storage for fabricated parts and tooling. The third and final assembly area activation in the third quarter of 1970 will satisfy the peak OTES requirements for assembly, storage of fabricated parts and project tools, manufacturing support, and shipping and receiving. Figure 3-4 illustrates the use of this area to accommodate the engineering model as well as assembly of the thermal and structural test units during this peak period.

Lockheed structural testing is planned to be performed in Lockheed-Sunnyvale Building 102 Test Bay. This existing facility is completely equipped with a 60-ft high bridge crane and required test equipment. Thermal development test on segments of the OTES engineering model will be performed in the Lockheed-Sunnyvale Biotechnology Laboratory vacuum chamber.

3.3.2 LMSC R&D Support Facilities

Several equipment facilities of the LMSC Research & Development Division are available for use on the various Lockheed spacecraft development programs. For the OTES thermal similitude study, for example, the Thermophysics Group has developed a Space Environmental Analysis Research Chamber (SEARCH) that will provide OTES with a capability to verify experimentally predicted orbital temperature distributions using scale models.

The chamber (Fig. 3-5) is a horizontal steel cylinder 8 ft in diameter by 10 ft long. Four 6 in. diameter view ports are located at the horizontal centerline, two on each side. Liquid nitrogen cold wall sections are spaced out 4 in. from the chamber wall. Both ends are equipped with lapped doors on rolling frames. A spare door is available for instrumenting new test specimens while the chamber is in operation. Each door is provided with a central 12-in. diameter quartz view port which can be replaced with an insulated feed-through plate. In addition, each door is equipped with five 6-in. diameter feed-through plates. Each door is also equipped with a vacuum-sealed actuator rod which provides up to 6 in. of axial translation or bidirectional

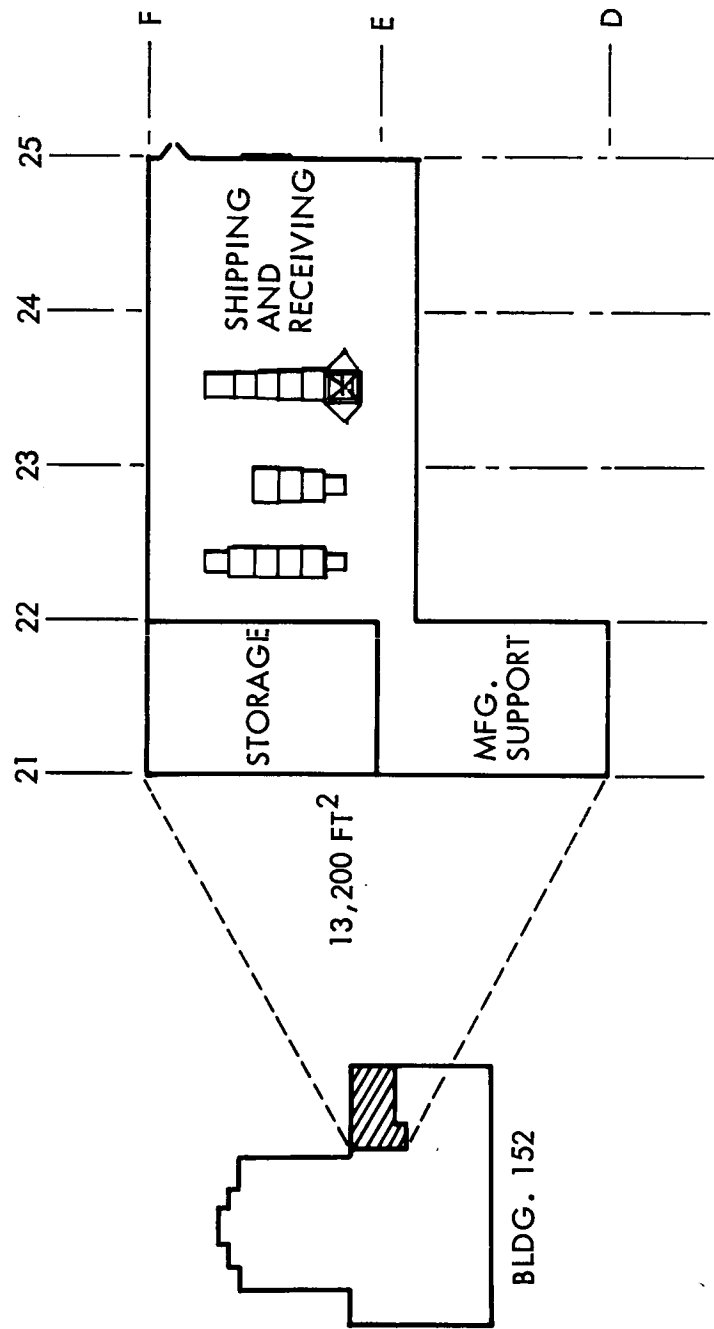


Fig. 3-4 Sunnyvale Building 152 OTES Assembly Area

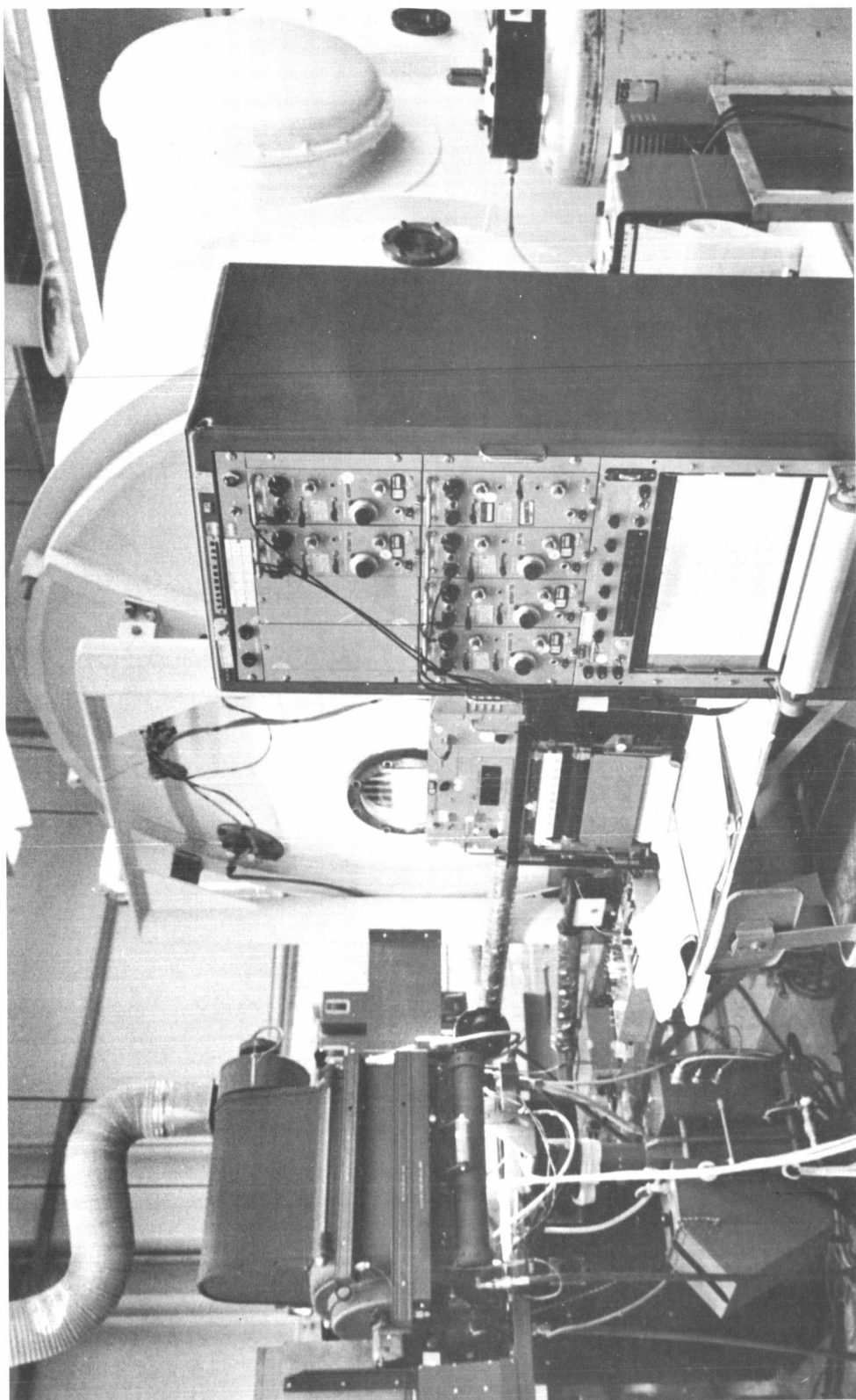


Fig. 3-5 Overall View of Search Chamber Showing Instrumentation in Place

rotation up to 200 rpm (limited by seal friction). The actuator rod incorporates a slip ring with connections for 40 pairs of instrument leads. Test specimens are supported entirely by the doors.

The chamber is equipped with a single 6-in. exhaust valve which can be connected alternately to either a 250-cfm oil lubricated mechanical pump or a 178-cfm blower. There are plans for future installation of a second 6-in. valve so the roughing pumps can be operated in parallel. There are also two finishing pumps. One pump is a 350 l/sec oil ejection pump incorporating two diffusion stages and one ejector stage. It is capable of pumping down to 10^{-4} torr. The second finishing pump is a 36-in., 50,000 l/sec oil diffusion pump valved in series with the ejector pump. With the ejector acting as a fore pump, the diffusion pump produces 10^{-7} torr.

The test specimen can be irradiated by tungsten-filament quartz lamps mounted inside the chamber, with total input up to 25 kw. However, the principal source of thermal radiant energy is an optically focused carbon arc lamp (on a separate rolling stand) which produces a 3-ft diameter spot 50 in. from the door. The lamp is equipped with a masking frame behind the objective lens to shape the beam to the configuration profile of the test model so as to reduce back scattering from chamber walls. The output at spot focus is equivalent to one sun irradiance, or 0.14 w/cm^2 .

The SEARCH facility will soon be prepared to perform a thermal similitude verification for NASA-MSFC under contract NAS 8-20411. In considering the selection of a candidate spacecraft to be modeled for study in the chamber, the MSFC project office expressed the desire to use a telescope configuration, preferably one already under study at that time (June 1966). It was natural that attention should be drawn to the OTES Phase I study and to the four telescope concepts then defined (0.5-m, 1-m, 2-m, and 3-m). After a period of study of the four configurations and the modeling problems which each entailed, the 2-m concept was selected as the subject.

3.4 OTES OPERATIONS SUPPORT PLAN

3.4.1 Manned Spacecraft Center

Technical support of the OTES program will be provided to assist the Manned Spacecraft Center (MSC) in astronaut training, conducting thermal-vacuum tests, and experiment data analysis. These activities will be in support of the MSC Mission Control Center as well as the anticipated MSC Experiment Control Center and the MSC Data Center. Office space for Lockheed personnel will be provided in the Lockheed Clear Lake facility on El Camino Real, approximately 1.5 miles from MSC. It is recommended that the large thermal-vacuum chamber, located in MSC Building 32, be employed for qualification and flight readiness testing of the integrated OTES carrier. The chamber is capable of accommodating the proposed OTES configuration and current schedules indicate the chamber will be available at the times required.

3.4.2 Kennedy Space Center

Technical support will be provided to assist Kennedy Space Center (KSC) during the following operations:

- Receiving, inspection, and interim storage of the OTES and associated hardware
- GSE preparation, ground crew training, and prelaunch checkout of the module

Office space for Lockheed personnel will be made available at the Lockheed facility on North Atlanta Avenue, Cocoa Beach. Figures 3-6 and 3-7, respectively, illustrate the KSC site locations and support facility layout.

Major OTES activities at KSC will occur in the Manned Space Operations Building (MSOB) and at Launch Complex 37. MSOB activities will include receiving-inspection, mating of OTES/rack with the Lunar Module Ascent Stage, and complete systems checkout to verify integrated module operation. The entire integrated module will then be mated with the Saturn LM Adapter (SLA) prior to transport to the launch pad complex.

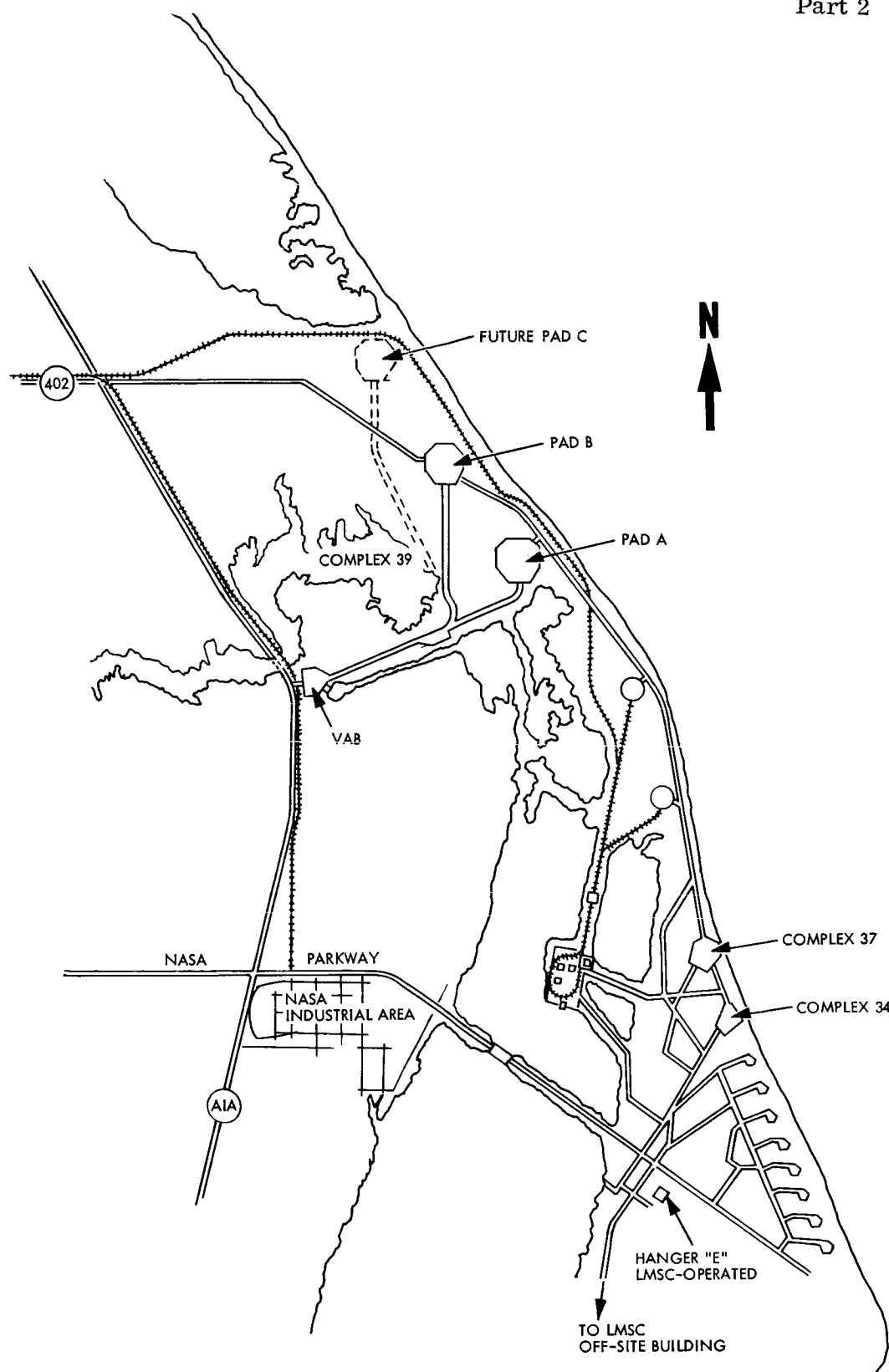


Fig. 3-6 Location Map of Kennedy Space Center

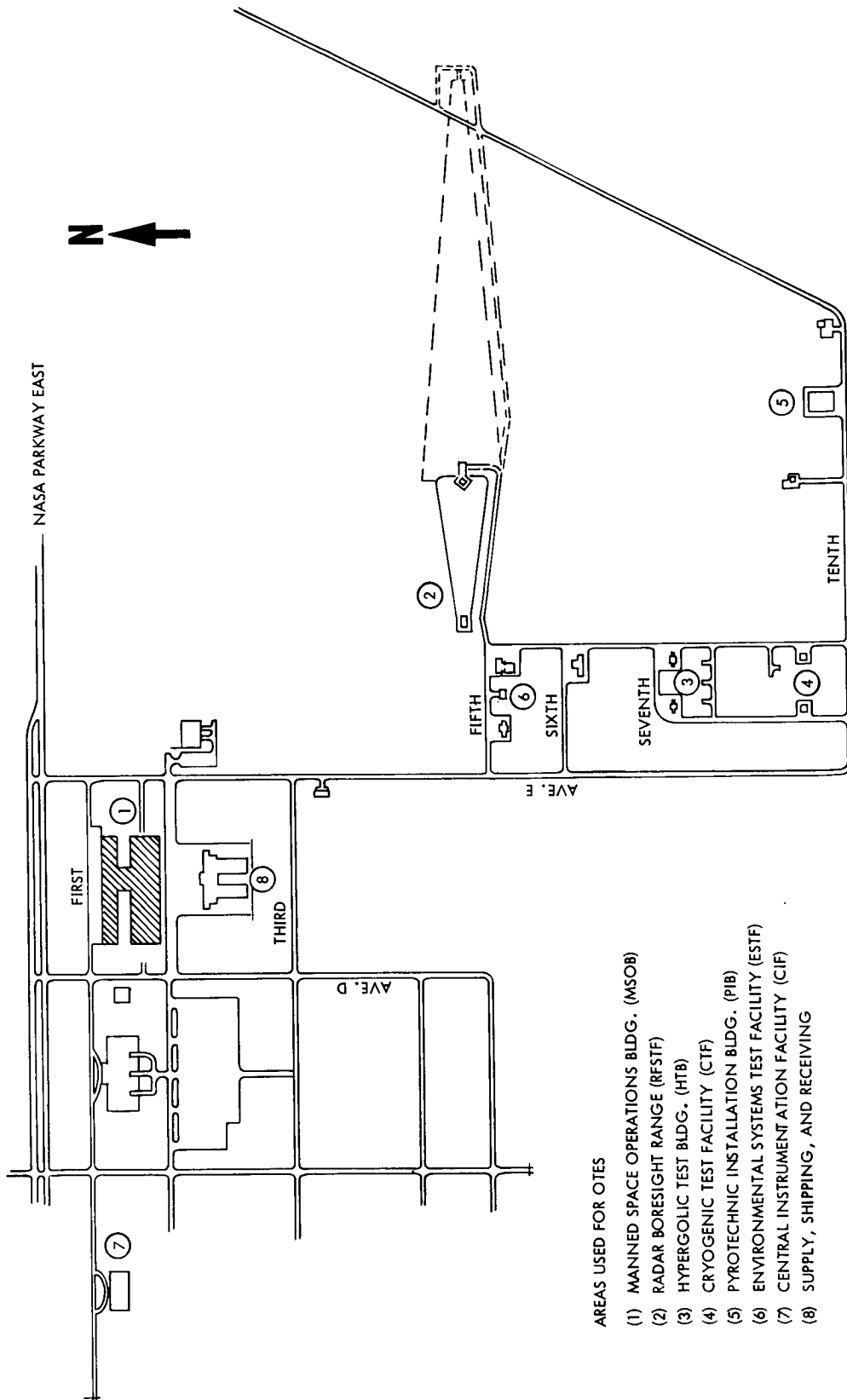


Fig. 3-7 OTES Support Facilities at KSC

In addition to the MSOB, Launch Complex 37, and supply shipping and receiving, OTES support may be required from the following additional KSC facilities:

- Radar Boresight Range (RFSTF)
- Hypergolic Test Building (HTB)
- Cryogenic Test Facility (CTF)
- Pyrotechnic Installation Building (PIB)
- Environmental Systems Test Facility (ESTF)
- Central Instrumentation Facility (CIF)

3.4.3 Modification Requirements

Preliminary analysis of basic OTES facility requirements indicates that no major modifications or new construction will be required beyond that already planned for the AAP Program. Current planning indicates that the Sunnyvale Building 152 Assembly Area equipment access door will require widening from its present 12-ft width to accommodate OTES hardware and tooling. Approximately \$1,500 will be required for this modification. Figure 3-8 summarizes the planned activation and utilization of facilities to support the OTES Program.

REQUIREMENT	1968				1969				1970				1971				1972				1973				1974							
	1	2	3	4	1	2	3	4	1	2	3	4	1	2	3	4	1	2	3	4	1	2	3	4								
SUNNYVALE OFFICES (LEASED BUILDINGS)	1,800 FT ²				2,200 FT ² 2,700 FT ²				3,800 FT ²				3,100 FT ²				2,100 FT ² 1,900 FT ²				1,600 FT ²											
HUNTSVILLE OFFICES (LOCKHEED ENGINEERING BUILDING)					1,000 FT ² 1,700 FT ²				3,700 FT ²				3,300 FT ²				3,000 FT ² 1,800 FT ²				1,000 FT ²											
PAYLOAD INTEGRATION FACILITY (AAP OFFICE TRAILERS)																																
MSC (LOCKHEED BUILDING, CLEAR LAKE)	300 FT ²				700 FT ²												1,000 FT ²								600 FT ²							
KSC (LOCKHEED BUILDING, COCOA BEACH)					600 FT ²								900 FT ²												300 FT ²							
ASSEMBLY AREA (SUNNYVALE, BLDG. 152)	2,000 FT ²				5,600 FT ²				13,200 FT ²				5,600 FT ²																			
CARRIER PREPARATION AREA (MSFC BLDG. 4755)													4,000 FT ² (MODELS 3 & 4)				2,000 FT ² (F/M NO. 1)				2,000 FT ² (MODEL 3 REFURBISH)											
INSTALLATION AREA (MSFC BLDG. 4708)													4,000 FT ² (MODELS 3 & 4)				2,000 FT ² (F/M NO. 1)				2,000 FT ² (MODEL 3)											
SYSTEMS TEST AREA (MSFC BLDG. 4708)																	2,000 FT ² (F/M NO. 1)															

Fig. 3-8 Facilities Activation Schedule

Section 4

PRELIMINARY TEST PLAN

The OTES spacecraft master test planning document is based generally on current NASA standards and practices relative to major system hardware qualification and acceptance testing. Consistent with the early conceptual study phase, this preliminary plan provides a guide to later development of a refined master test plan and establishes requirements for the test prototypes and test activities needed to verify launch and mission readiness for the OTES spacecraft in either the cluster or independent modes.

4.1 SCOPE

This preliminary test plan defines and describes the types of tests, test techniques, and assumed guidelines and constraints associated with the OTES spacecraft development program. The plan outlines the major activities and interrelationships required for design verification, development, qualification, and acceptance testing of nonflight hardware and the prelaunch system tests and pad checkout of flight hardware. Hardware items at all levels of assembly from piece parts through the complete OTES are included. The master schedule covers the period from the beginning of Phase C in January 1969 to the assumed launch date in July 1973.

4.2 GUIDELINES

Ground rules for hardware and test requirement development assumed for the OTES program are summarized in the following subsections.

4.2.1 Hardware

The OTES spacecraft subsystems (as distinguished from optical subsystems) include the telescope spar/structure and associated mechanisms, ATM rack, ATM gimbal, LM-A, and Orbital Mirror Recoating Facility (OMRF).

With the exception of battery modules, minor power distribution, and open-loop controls for actuator drives, the telescope spar structure and mechanisms contain no electrical black boxes.

The electrical subsystem arrangement of the OTES rack differs from its progenitor only in the arrangement of the boxes, the deletion of certain unneeded signal and power leads, deletion of several unneeded electrical subsystem components, addition of a third star tracker, and the attendant minor cable harness changes. Most electrical hardware is assumed to have been previously qualified in Apollo or post-Apollo programs. All optical subsystem electrical components are assumed to be completely contained in the electro-optical equipment compartments of the telescope.

Structural additions to the ATM rack required to produce an OTES rack are primarily confined to the addition of support framework for the Perkin-Elmer magnetic suspension assembly and the telescope launch caging assembly. A support cone and ring structure assembly replaces the ATM rack gimbal ring.

4.2.2 Test Activities

One central test organization will be established. It will be responsible for all test planning, including coordination with the various design groups in establishment of integrated test program requirements and their dissemination in the form of test plans.

All testing will be performed under the cognizance of this organization except for receiving inspection testing, manufacturing inprocess testing, and some prelaunch checkouts. This organization will, however, have the responsibility of reviewing all fixed procedures employed during the aforementioned activities and approving the original issue and any subsequent changes to those documents that deal specifically with the OTES Program.

Test requirements in all categories emphasize testing at the highest practical level of hardware assembly. Integrated system testing will, in most cases, supersede component or subsystem level testing; rack subsystem components will have been previously qualified.

Manned participation in the chamber during thermal vacuum testing at MSC is assumed. Trained operators and/or test technicians will perform the initial debugging runs; astronauts will assist in the qualification and flight-hardware demonstrations as a part of their familiarization training.

Ground support equipment (GSE), particularly for use with electrical testing, is subject to the same acceptance test philosophy as the flight hardware in cases where it interfaces directly with the flight hardware. GSE at the system level, such as a system test complex, is validated with comprehensive simulators by the using test organization. Use of prototype and flight hardware for checkout of the test equipment will be kept to an absolute minimum.

Whenever practical, development and qualification testing will use flight prototype hardware. This approach does not prohibit breadboard development testing before flight prototype hardware is available, but requires that planning include its use at the earliest possible point in the test program to minimize the "similarity gap" between what is tested and what is flown.

Every functional black box requires certified evidence of qualification by analysis and test. In the case of rack components, certification is assumed to be available from AAP. Qualification testing and/or certification is required at each successive level of assembly up to and including the integrated OTES.

Reliability data requirements will be satisfied through development, qualification, and acceptance test data from OTES and other programs. In special cases, where the quantity or quality of existing test data are inadequate, additional reliability testing will be conducted.

Government furnished equipment (GFE) is considered qualified and acceptance-tested at the original contractor's facility. Receiving inspection will consist primarily of identification and damage checks. Critical electronic hardware, however, will be subjected to power application tests to verify polarity, phasing, and internal routing, and to check for gross defects such as short circuits or broken wires.

Manufacturing inprocess testing is limited to those tests necessary to verify the integrity of the fabrication and assembly process. Inprocess tests supplement the final acceptance test of the flight hardware when the final test cannot adequately verify functions below the top assembly level.

Environmental (thermal-vacuum) testing planned for flight prototype hardware at the system level is essentially an integrated function test of the flight article. Integrated system testing of the assembled OTES consists of a mission sequence test at ambient conditions and a repeat of this test under simulated thermal vacuum environmental conditions. Electromagnetic interference (EMI) compatibility of the integrated system may be evaluated at MSC before and/or after the thermal-vacuum tests.

Flight or flight prototype hardware is used when possible for cluster-element mechanical and electrical compatibility testing. Master tools and mockups are used only when the flight level hardware is not available.

Prelaunch checkout is conducted at the integrated system level. Checkout procedures developed during operations will be employed during integrated systems test. Correction of integrated test failures is by replacement of the malfunctioning black box.

4.3 TEST CONTROL DOCUMENTATION

Documentation controls are applied in varying degrees of sophistication depending on the test category. In general, they progress in depth of detail and rigidity of control from breadboard testing.

4.3.1 Test Documents

Master Test Specification. This specification is subordinate to, and derived directly from the OTES system specification. In outline format, it lists the several levels of tests to be performed (experimental, developmental) at each system level (integrated system, spacecraft system, subsystem assembly, and component) and provides the performance limits to be demonstrated in each applicable test.

The established limits on test levels for each parameter will be determined jointly with NASA test specialists and will receive written NASA certification before the qualification program is initiated. The specification will be drafted by systems engineers in cooperation with the design group. It will serve as a reference and a guide in the preparation of all lower-level test specifications and procedures and will govern the sequence in which the tests are conducted.

Subsystem Test Specification. Each subsystem test program is covered by a separate specification document derived from, and subordinate to, the master test specification. In addition to reiterating the performance limits, the subsystem specification defines the overall test objective and general approach for the subsystem as a whole and for the major assemblies (if tested separately). It defines the locations and type of test instrumentation, the expected scale range for test readings, and the amount and general format of data records and test reports.

Component Test Specification. In those few cases in the OTES test program where components will be tested individually (because of prior qualification of ATM and LM-A subsystems), two types of component test specifications apply. One type, applicable to newly developed or significantly modified purchased components, is covered by an LMSC Design Control Drawing (DCD) and a detailed Acceptance Test Specification (ATS). The other type, applicable to off-the-shelf purchased components for which qualification certification exists, is covered by Specification Control Drawings (SCD) which describe the principal features and refer to the appropriate test specification, i. e., either an LMSC-approved vendor specification or an LMSC-generated specification.

Test Procedures. Test procedures are written by test engineers. The procedures reiterate the purpose of the test: the detailed performance limits to be demonstrated and the data requirements, the specific instrumentation and test equipment, how it is to be set up, procedures to be followed, the frequency of data logging, and other instructions and guides. Test procedures are reviewed by, and subject to, written approval of System Engineering. In many cases, procedures specify that a system engineer must witness the test performance.

4.3.2 Test Support Documentation

Some test control documents are not as formal as a specification, but rather serve as paperwork necessary to the overall control and support of the test program. These are described below.

Log Book. This is a summary document, initiated by Quality Engineering, which provides a complete history of each component, assembly, and subsystem. This type of document has been in effective use by LMSC for years and is currently used as the hardware and operational control document for space systems.

Calibration and Certification. Test equipment is calibrated and certified to ensure test integrity. The proficiency of test personnel is also reexamined at regular intervals. Quality Assurance audits the organization records and certification status of all test equipment and personnel. Test and support equipment is validated to approved procedures on initial installation and use. Periodic recalibration requirements are determined and so noted on the equipment by application of a coded placard indicating the necessary recalibration date. The using organization maintains records of all equipment requiring this maintenance and rotates equipment as necessary. Quality Assurance personnel provide surveillance of records and equipment to ensure that only calibrated and certified equipment is used. Personnel assigned to critical operations, such as thermal-control surface preparation and manned-system testing, will receive special training and certification. The using organizations maintain certification records and periodically recertify personnel. Quality Assurance personnel audit these records and verify certification.

FACI/DD 250. This is the presentation and delivery of completed equipment for acceptance, and is the responsibility of the Quality Assurance organization. All affected organizations are required to provide support to the Configuration Management Office for the First Article Certification Inspection (FACI) evaluation and to provide all background and engineering records on actions taken. Flight worthiness of the end-item hardware is certified as a part of this action.

Qualification Certification. Qualification is required of all flight hardware items. When qualification testing is required to determine design acceptability, test procedures developed by the test organization and approved by engineering are witnessed and verified by Quality Assurance. The Reliability organization prepares the qualification certificates and maintains the program qualification records.

Failed Equipment Data Report (FEDR). The FEDR system was devised to accomplish, with minimum paperwork, the maximum dissemination of information on failed components or assemblies and the redesign correction of any trend or classification of failure on like or similar systems. It is both an informational and redesign tool and serves an additional function of ensuring good shop practices by reporting all anomalies on components or assemblies.

4.4 TESTS AND TEST EVENTS

4.4.1 Experimental Testing

Experimental testing may be compared to development testing except that test hardware has not progressed to the prototype phase. They are used to prove concept and feasibility of a general design rather than to determine if there is fault with a specific design.

Typical of experimental testing, the following will be performed to demonstrate feasibility during this phase.

Thermal Materials. Models of various thermal materials in various layer combinations will be devised and tested in an attempt to establish the proper thermal control early in the design phase.

Pyrotechnic Lock Ring. A method is needed which will firmly lock in place the five extended sections of the telescope body. Present design plans are to fabricate pyrotechnically operated locking wedge assemblies which will rigidize the telescope body at each interconnecting joint. Test models of this type of joint will be fabricated and tested, on the breadboard level, at LMSC's Santa Cruz Test Base.

Sun Cap. A sun sensor, actuator, and activation control system should be devised to cause the sun shade to cover the telescope end to preclude exposure of the optics to direct sunlight. Breadboard models of the system will be built to demonstrate feasibility.

4.4.2 Development Testing

These tests identify design weaknesses, establish performance characteristics, and evaluate areas of potential performance degradation under simulated mission environmental conditions.

Free-Float Positioners. The OTES will be contained in a three-degree-of-freedom gimbal mount in the ATM rack. During the OTES mission, gimbal control will be shifted to a magnetic suspension system with the mechanical gimbal system disconnected by means of the free-float positioners. A development model will be devised to demonstrate this capability.

Cable Extension System. A system must be devised to deploy upon command the five sections to the OTES telescope barrel. The development system will consist of pulleys and cables driven by electrical motors. Manual override may be used to permit astronaut deployment of the spar in case of redundant system malfunction.

Center-of-Gravity Positioner. When the telescope is shifted from the fixed gimbal to the magnetic suspension, an undesirable CG shift occurs. A system which is capable of shifting the CG to the new center of rotation of the magnetic suspension system will be tested.

Quartz Spacer Rods. Quartz rods will be utilized to locate the secondary mirror in the OTES. These rods are folded and protected during launch. They are unfolded and spring-loaded mechanically before operation. A deployment test will be performed.

4.4.3 Receiving Inspection and Test

Receiving inspection, analysis, and testing will be conducted by Quality Assurance on all material and piece parts procured and stocked. The quality requirements are dictated by the current contracts, purchase orders, schedules, process controls, and prior source inspection and tests. All items are identified to reflect results of inspections and tests performed and final disposition. Data packages are prepared for part control and end-item acceptance. These data become an integral part of the hardware log book at the next assembly level and are used for reference as required at any point in the test program.

4.4.4 Manufacturing or Inprocess Testing

Inprocess testing is the sequence of activity which takes place during fabrication and assembly of end-item hardware and GSE. The Manufacturing Engineering organization performs this activity in accordance with Manufacturing Test Procedures approved by Engineering and monitored by Quality Assurance. For end-item hardware at the black-box level, inprocess testing ends with the acceptance test of the finished unit. For manufactured or modified GSE, the final inprocess testing is generally the acceptance of the equipment at the console level. Typical examples of inprocess tests are:

- Alignment checks and functional operation of extension, erection, are powered mechanisms

- Electrical and electronic circuit, high potential, and continuity tests
- Major section testing of structure before final mating of the sections
- Qualitative tests of the assembled OTES to determine conformity to design requirement, installation integrity, and readiness for system test

4.4.5 AGE and GSE Testing

AGE and GSE must be tested and certified prior to use against any flight system. For certification, the AGE and GSE will be tested against a procedure rather than building checkout equipment.

The following tests are required as applicable, but are not to be construed as limiting:

- Visual and Mechanical Inspection
- Handling and Transportation GSE Proof Tests
- Fluid Proof Tests
- Environmental Tests
- EMI Testing
- Interface Verification Testing

4.4.6 Subsystem Installation and Compatibility Checkout

All subsystems and other GFE received are inspected for identification and damage for compliance with applicable documentation and proper gross functioning requirements for checkout, servicing, storage, calibration, and installation of each subsystem. Each subsystem is received and assembled into an OTES qualification unit in a functional configuration. Tests are conducted to establish that compatibility exists between the subsystem and its support equipment. The subsystem/data-system interface is established and verified.

4.4.7 Qualification Testing

Qualification tests demonstrate the capability of hardware not previously qualified to meet the requirements established. Tests are designed to provide a high level of

confidence so that all items of hardware will survive the expected mission environment and will perform successfully. Test articles used for qualification will be manufactured by the same method and to the same tolerances as the flight hardware. Qualification testing of components and subsystems is at the highest level of assembly consistent with previous development testing experience and applicable past flight performance.

Qualification testing is applicable to flight prototype hardware only. The qualification overstress percentage levels of 10 to 30 percent, as applied to specific test conditions, will be specifically defined in the hardware specifications. Typical areas of concern, other than the standard techniques of structural load application and proof-pressure testing, are as follows:

- Temperature extremes in excess of design operating envelopes
- Energy input levels in excess of boost phase environment for vibration, acoustic, and acceleration testing
- Simulated solar radiation levels in excess of design operating envelopes for thermally controlled systems
- Input voltage and current level extremes in excess of design limits for critical electrical equipment
- Application of design limit test conditions for times in excess of design requirements (vibration, pressure, current limiters, etc.)
- Exposure to radiation levels in excess of design limits for radiation-sensitive equipment

4.4.8 Integrated System Testing

Integrated system testing is a simulation of the mission profile as closely as the ambient environment permits. A simulated countdown test is performed to verify that the OTES and the associated support equipment are properly conditioned electrically and mechanically for the simulated flight. The OTES is then placed in the prelaunch mode, and a simulated flight is performed. The test sequence proceeds from pre-launch through launch, ascent (rendezvous and docking), orbital transfer, activation,

operations, deactivation, and storage, with certain functions simulated to present as realistic a flight sequence as possible. Performance is evaluated through the airborne telemetry data system. A crew is used during orbit simulation to pattern the astronaut activity and obtain maximum utilization of onboard checkout systems for stimulation and response. Vehicle interrogation and data readout modes are exercised. The test ends with an evaluation of normal operations and backup modes. The data from this test are part of the basis for vehicle acceptance through the DD-250 format.

4.4.9 Mass Characteristics Test

Final testing for weight, CG, and moment of inertia is performed on both the thermal qualification unit and flight unit just before they are shipped to the thermal-vacuum facility. Data from this activity are used as verification of previous engineering calculation and provide certification for the weight statement input to launch and orbital operations. All flight hardware must either be installed or adequately simulated.

4.4.10 Environmental Test (Thermal Vacuum)

The thermal qualification unit and the flight OTES are both subjected to thermal-vacuum environment. The environmental level for this test is lower than the design limits or operating envelope for the hardware. The purpose of this test is to substantiate the math-thermal analytical model. In addition, any defects will be revealed which might have occurred in the fabrication, assembly, and other processes, such as marginal electrical connections, improper thermal-surface application or control, and system level defects or interactions that by their nature cannot be discovered in an ambient or static environment.

4.4.11 Anechoic Chamber Testing

Electromagnetic compatibility (EMC) tests are conducted in an anechoic chamber in conjunction with the final systems tests on the flight prototype configuration. Typical

requirements of EMC testing are:

- The mission configuration is compatible within itself in all operating modes and flight sequences.
- A minimum safety margin of 6 db in voltage or current exists at all critical points within the vehicle and at the critical interfaces of the launched vehicle.
- The flight vehicle configuration is compatible with all anticipated electromagnetic environments.

Specific EMC requirements for the mission and flight as dictated by mission operations and hardware design will be contained in individual detail test plans and specifications.

4.4.12 Prelaunch Tests

The launch-pad sequence follows the standard Apollo test plan consisting of premate checks, simulated flight, flight readiness test, and countdown demonstration tests. The OTES special tests and confidence checks will be performed in a manner similar to that used for the ATM spar. The payload and the spacecraft LM adapter are installed on the launch vehicle, and premate tests ascertain compatibility of the two units with the launch complex GSE before final mate. This phase also checks the communications systems. The combined readiness tests furnish the final all-up verification of space vehicle readiness. The simulated countdowns and flight operations ensure that the system will perform properly in the sequences planned for the mission.

4.5 TEST DATA AND REPORTS

Formal test reports and other test documents (such as procedures) serve as an historical record of component or subsystem development and test. Test data, signed off by Quality Assurance, certify proper performance of the various checkouts and tasks assigned to the hardware by Engineering. Development test records will be maintained in Sunnyvale.

Based on the assumption that LMSC will be the OTES System Manager and AAP Integrator, free use of the AAP reporting system is planned. Reporting, in AAP, is required for all testing performed, and the information (data and reports) is stored and maintained in a test program data bank, with cross-referencing and indexing by flight, test category, and hardware identification. The information storage and retrieval techniques used are adaptable to computer application at a later date.

4.5.1 Qualification Test Report

This report summarizes the results of qualification tests at all system levels. Raw data from qualification and acceptance testing are recorded on log sheets or magnetic tape during the test, and are retained for processing and record purposes. Onboard telemetry data, processed and analyzed, are used as the primary basis for OTES acceptance, with hardline data for readout of test instrumentation. The data processing function is performed by the test organization and the data analysis function is performed primarily by the test organization, but in close conjunction with engineering personnel. At MSFC and KSC the RCA 110 and at MSC the ACE automatic checkout systems will be used for computerized data processing and analysis.

4.5.2 Other Reports

Test reports containing test results on a specific test or phase of testing will be periodically released. Some major milestones are:

- Completion of all development-level demonstration tests
- Completion of dynamic and static development testing of the OTES structural model
- Completion of the OTES engineering model thermal-vacuum test
- Completion of the integrated systems tests

4.6 TEST HARDWARE AND COSTS

Below are listed the costs for LMSC-utilized test hardware. The test hardware includes the models and prototypes which are fabricated for test purposes, as well as

the special test equipment designed and fabricated solely to support test performance. This listing does not include standard, minor test equipment which is either GFE available onsite or is GFE equipment available on a lease-free basis. Although these costs are included in the OTES Program cost summaries, they are isolated here for future test planning considerations.

4.6.1 Models and Prototypes

<u>Equipment</u>	<u>Cost</u> <u>(\$ thousand)</u>	
<u>Experiment Test</u>		
Pyrotechnic test model	21	
Thermal insulation models	11	
Sun cap sun-alarm model	4	
Subtotal		36
<u>Development Test</u>		
Mechanism demonstrators	64	
Orbital mirror recoating facility	224	
ATM rack modifications	42	
Engineering model	312	
Subtotal		642
<u>Qualification Test</u>		
Structural test model (LMSC)	538	
Structural test model (No. 3) (LMSC/ P-E)	890	
Thermal test model (No. 4) (LMSC/P-E)	1,238	
OMRF model	269	
Subtotal		<u>2,935</u>
Total		<u>3,613</u>

4.6.2 Test Equipment

<u>Equipment</u>	<u>Cost</u> <u>(\$ thousand)</u>
GSE checkout consoles (2)	100
Vertical support stand (balance and T/V tests)	<u>138</u>
Total	<u>238</u>

4.7 TEST SCHEDULING AND COSTS

4.7.1 Test Scheduling

The complete OTES master schedule for testing is presented by Fig. 4-1. This schedule shows gross functions from which more detailed schedules can be produced in subsequent phases of program development. Phase C, ending in December 1970, is primarily devoted to AGE/GSE and OTES system design, mockup fabrication, and development model testing. Late in Phase C, an engineering handmade model is produced for further development tests. Six months later a structural test model and a thermal test model are fabricated to hard tooling for qualification testing.

The models are then shipped to P-E for integration of the optical subsystem prior to qualification test. These models, with optical system or mass equivalents, are to be mated to the rack at MSFC. The design integrity of these two flight prototype models is qualification tested at MSFC. The flight unit construction and assembly are initiated in mid-1971 - approximately 18 months after the start of development testing. This period provides time for design updating and improvement prior to flight model fabrication.

The flight model will be fabricated at LMSC-Sunnyvale, shipped to Perkin-Elmer in Connecticut for optics installation and checkout, and then moved to Huntsville for mating to the rack and LM-A, and then it is instrumented and checked out. Following DD-250 at MSFC, the flight unit is shipped to MSC to undergo a manned thermal vacuum test for verification of flight worthiness. Following this test, the unit will be shipped to the Eastern Test Range to undergo prelaunch tests and countdown to launch in July 1973.

PERKIN-ELMER/LOCKHEED

	1969												1970											
	J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A				
MASTER TEST SPECIFICATION WRITTEN																								
EXPERIMENTAL TESTING																								
GSE DESIGN																								
ENGINEERING MODEL FABRICATED																								
STRUCTURAL SUBSYSTEM TEST SPECIFICATION WRITTEN																								
MODIFY ATM RACK																								
RACK SUBSYSTEM TEST SPECIFICATIONS WRITTEN																								
COMPONENT TEST SPECIFICATIONS WRITTEN																								
STRUCTURAL TEST OF ENGINEERING MODEL																								
THERMAL TEST OF ENGINEERING MODEL																								
MECHANISM DEMONSTRATIONS																								
STRUCTURAL DEVELOPMENT MODEL FABRICATED																								
STRUCTURAL DEVELOPMENT MODEL TESTED																								
STRUCTURAL QUALIFICATION UNIT FABRICATED																								
THERMAL QUALIFICATION UNIT FABRICATED																								
OMRF QUALIFICATION UNIT FABRICATED																								
QUALIFICATION UNITS SHIPPED TO P-E																								
QUALIFICATION UNITS RECEIVED AT MSFC																								
QUALIFICATION UNITS INTEGRATED TO LM/RACK																								
QUALIFICATION UNITS FOR TEST INSTRUMENTED																								
Q/U INTEGRATED SYSTEM TEST CONDUCTED																								
Q/U STRUCTURAL/THERMAL TESTED																								
STRUCTURAL QUALIFICATION UNIT SHIPPED TO KSC																								
THERMAL QUALIFICATION UNIT SHIPPED TO MSC																								
THERMAL QUALIFICATION UNIT INSTRUMENTED																								
MANNED THERMAL/VACUUM PERFORMED																								
ACOUSTIC TEST PERFORMED																								
THERMAL Q/U RETESTED AND REFURBISHED																								
TQU AS FLIGHT BACKUP SHIPPED TO KSC																								
FLIGHT TELESCOPE STRUCTURE FABRICATED																								
OMRF FLIGHT UNIT FABRICATED																								
FLIGHT ARTICLES SHIPPED TO P-E																								
FLIGHT ARTICLES RECEIVED AT MSFC																								
FLIGHT ARTICLES INTEGRATED																								
FLIGHT ARTICLES INSTRUMENTED																								
INTEGRATED SYSTEM TEST PERFORMED																								
FLIGHT UNITS SHIPPED TO MSC																								
T/V MANNED SIMULATION PERFORMED																								
FLIGHT UNITS SHIPPED TO KSC																								
PRELAUNCH CHECKOUT PERFORMED																								
FLIGHT UNITS LAUNCHED																								

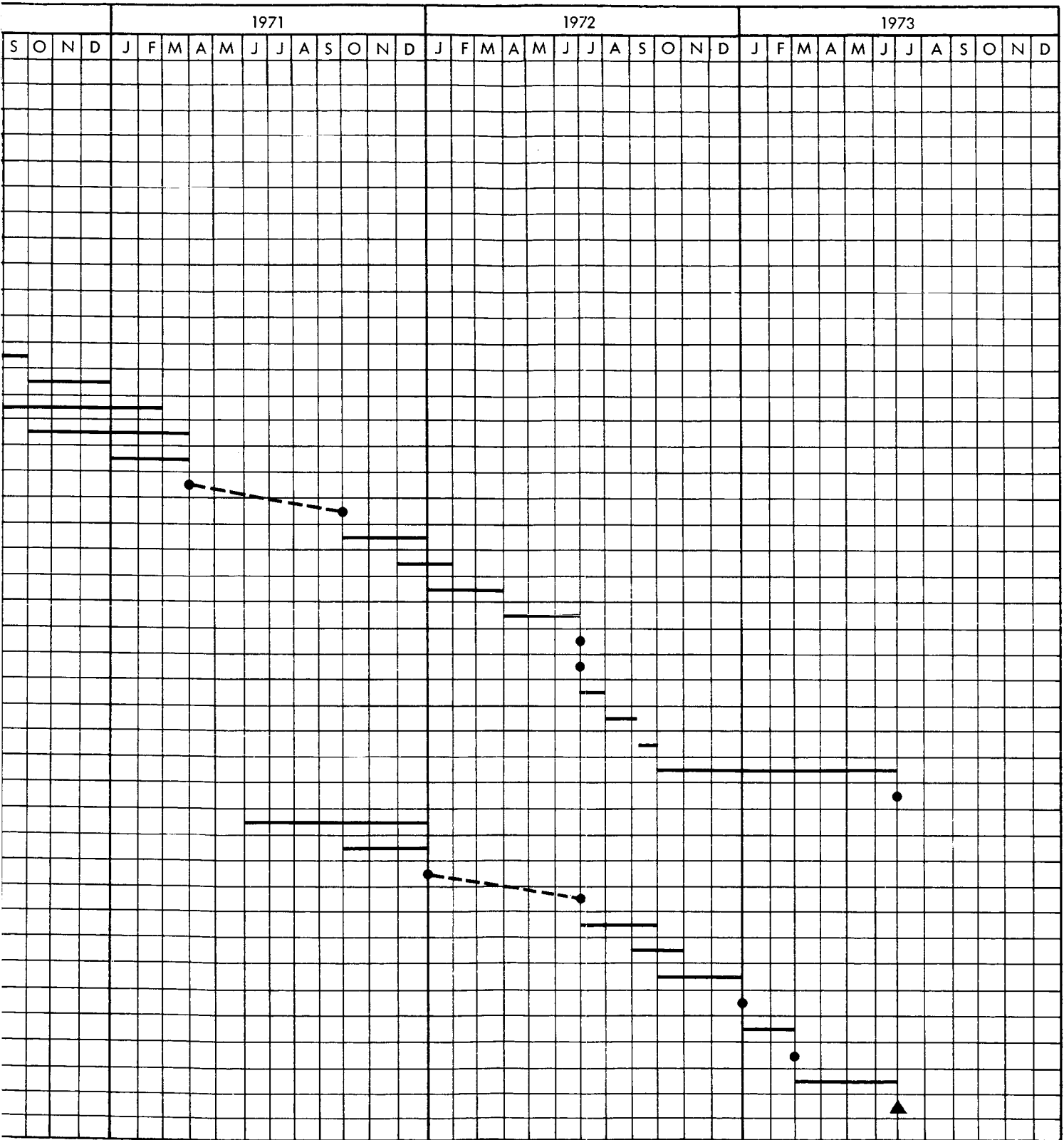


Fig. 4-1 Test Program Schedule

4.7.2 Test Activity Costs

<u>Activity</u>	<u>Cost</u> <u>(\$ thousand)</u>	
<u>Test Planning and Specifications</u>		
Phase C	26	
Phase D	<u>32</u>	
Subtotal		58
<u>Experimental Test (Phase C)</u>		
Pyrotechnic lock ring	26	
Insulation materials	21	
Sun cap demonstration	4	
EVA simulations	<u>21</u>	
Subtotal		72
<u>Development Test (Phase C)</u>		
Mechanism demonstration	20	
Engineering model		
Deployment	48	
Structural	64	
Thermal	<u>64</u>	
Subtotal		196
<u>Qualification Test (Phase D)</u>		
Test coordination and control	74	
Structural model test (LMSC)	93	
OMRF structural/thermal	<u>56</u>	
Subtotal		223
<u>Testing Liaison and Coordination</u>		
Qualification tests (MSFC)	216	
Qualification tests (MSC)	64	
Acceptance tests (MSFC/MSC)	72	
Prelaunch checkout (KSC)	<u>160</u>	
Subtotal		512
<u>Qualification Report</u>		<u>32</u>
Total		<u>1,093</u>