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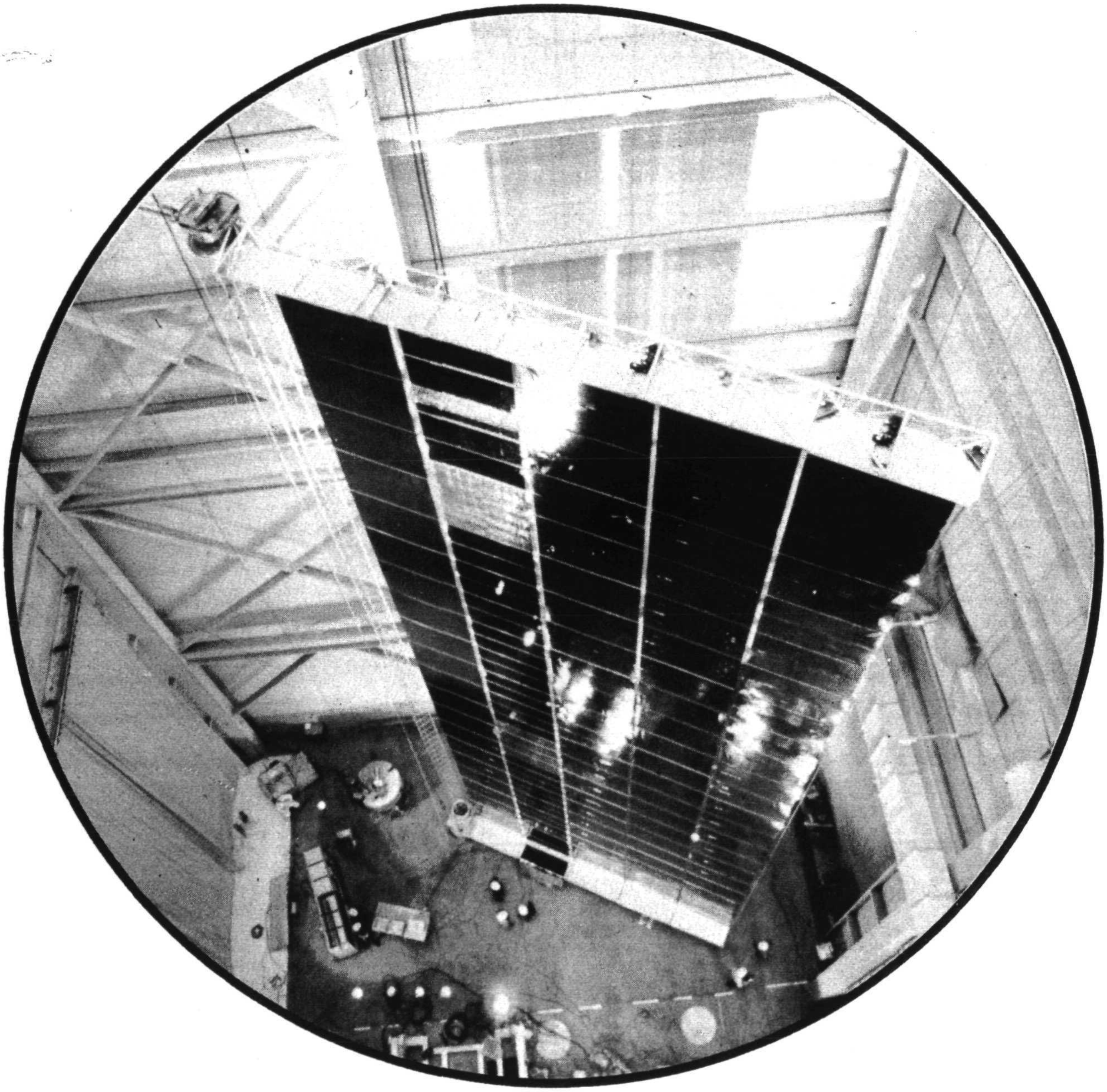
SPACE STATION SOLAR ARRAY PROGRAM

FINAL REPORT



CONTRACT NAS 9-11039

LOCKHEED MISSILES & SPACE COMPANY, INC.
A SUBSIDIARY OF LOCKHEED AIRCRAFT CORPORATION

MSC-07163
LMSC-D159744
February 1973

FINAL REPORT

SPACE STATION SOLAR ARRAY TECHNOLOGY EVALUATION PROGRAM

Contract NAS9-11039

Prepared for
Manned Spacecraft Center
Houston, Texas

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FOREWORD

This last report of the Space Station Solar Array Technology Evaluation Program documents the final results of all the major program phases. The program was conducted with the NASA Manned Spacecraft Center (MSC), Houston, Texas under Contract NAS9-11039. All goals of the program, which are listed in Section 2.1, were successfully accomplished and are briefly described in this document.

A complete list of all drawings generated under the contract by LMSC, Ball Brothers Research Corporation, and Astro Research Corporation is included as Appendix A. All of the listed drawings were microfilmed (punch card mounted) for convenient storage and reference at MSC, Houston.

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ILLUSTRATIONS

ILLUSTRATIONS (Contd)

ILLUSTRATIONS (Contd)

TABLES

1.0 INTRODUCTION AND SUMMARY

Under the direction of NASA Headquarters, conferences were held with NASA and Industry power system specialists in 1968 and 1969 with the objective of determining what technology programs should be initiated to insure "technology readiness" of solar photovoltaic power systems in the 25-100 KW range for application to Manned Orbital Space Stations. In these conferences, large flexible solar arrays were almost unanimously identified as an important pacing technology for Space Station development. As a result, in mid 1970 the Space Station Solar Array Technology Evaluation Program (SSSATEP) was initiated under NAS9-11039 with the Manned Spacecraft Center, Houston, Texas. The principal goal of the SSSATEP program was to determine the feasibility of designing and building extremely large area solar arrays. Therefore, a great emphasis was placed on reaching the point where a representative amount of full scale hardware could be fabricated and evaluated. The program was divided into three major phases which were:

(1) Technology Evaluation Phase - A complete review of all analysis, studies, and test results related to lightweight solar array technology up through and including 1970. Conclusions drawn from this review were presented and recommendations made as to where additional research and development work was required to support development of the 10,000 ft^2 Space Station Array having a design lifetime of 10 years.

(2) Design and Analysis Phase - Using the work accomplished to date on other R&D studies, and the groundwork performed by the Space Station Prime Contractors as a base, a detail design of a 10,000 ft^2, 100 KW, continuous sun tracking system was completed. The analysis used to arrive at this design was based on the most rigorous set of design requirements gathered from NASA and the Space Station prime contractors.

(3) Development Test Program Phase - Using the above design, full scale hardware representing one quadrant (2,500 ft^2) of the 100 KW array design was fabricated and subjected to a test evaluation program.

2.0 PROGRAM OBJECTIVES AND DESIGN REQUIREMENTS

2.1 Program Objectives

The Space Station Solar Array Technology Evaluation Program, conducted under NAS9-11039, with the Manned Spacecraft Center, Houston, Texas, was a pre-development program to evaluate the feasibility of using sun-tracking solar arrays as the prime power source for larged manned earth orbital Space Stations. The program was initiated July 1, 1970, and was extended through February 1973. Program activities were divided into three major phases which are shown in the schedule (Figure 2.1-1).

The goals for the Space Station Solar Array Technology Evaluation Program, listed below were originally established by NASA MSC and subsequently modified by the NASA Ad-Hoc Committee on Space Station Power.

1. Establish design requirements which are typical of space station missions.
2. Perform technology evaluation of existing techniques for meeting such requirements.
3. Identify specific areas where technology advancement is needed.
4. Prepare a plan for development of the technology defined in item 3.
5. Design components capable of meeting requirements imposed by a representative space station.
6. Verify, by test, that the technology for item 5 is available--or establish the problems and deficiencies therewith--and that the analytical tools used in such design are adequate.
7. Prepare a representative system Development Plan for system and mission evaluations.
8. Develop a Design Handbook of Parametric Data based on the analyses and tests above.

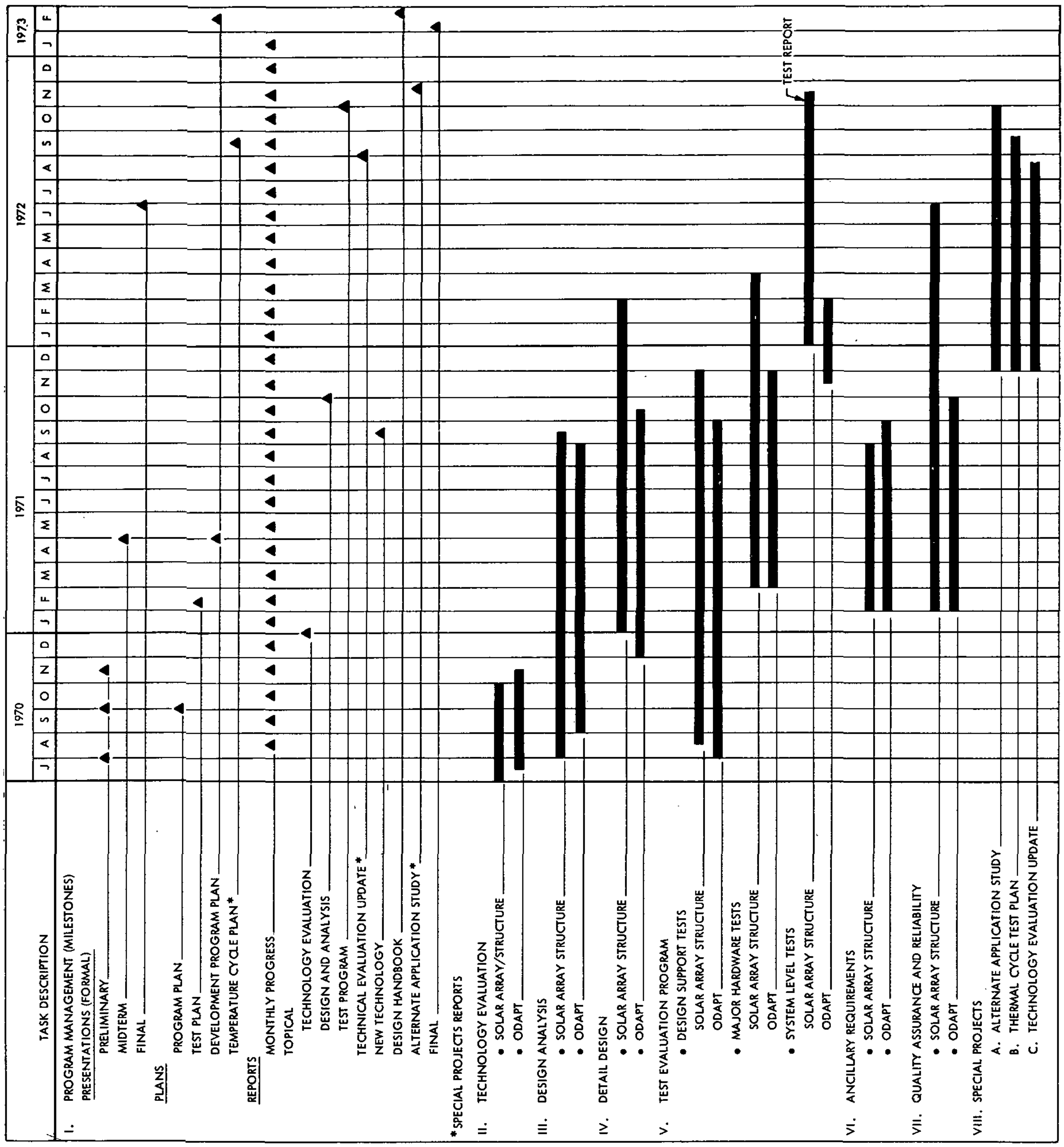


Figure 2.1-1 Program Schedule

2.2 Program Outputs

The results of the three major phases of the program were reported in a series of three topical reports. The first of these, the Technology Evaluation Report which reviewed and evaluated all solar array and drive system data, was delivered in December 1970. The second topical report, Baseline Design and Analysis, was issued November 1971. It covered all design/analysis data including the tradeoffs leading up to the selection of the baseline solar array design. The last topical report, published in November 1972, covered the extensive test program results. In addition to the topical reports and the final report, two other major reports were delivered. One was a Development Program Plan which described the steps required to provide an operational space station solar array in time for the planned space station operational launch program. The second was the Design Handbook which contained all analysis and data developed for trade studies and hardware definitions under the program. The design and analysis data and the capability of the baseline hardware design were verified through testing of full scale major component hardware including a full scale solar array quadrant fabrication and test and fabrication and testing of the major components of the orientation and power transfer system.

PROGRAM OUTPUT SUMMARY

Hardware Fabrication and Test

- Design support component fabrication and test
- Full scale array quadrant fabrication and test
- Full scale orientation and power transfer fabrication and test

Reports

- Topical Reports (3) - Results of major tasks
  - Technology Evaluation Report - December 1970
  - Design Analysis Report - November 1971
  - Test Program Report - November 1972
- Development Program Plan leading to final Space Station power system hardware - February 1973.

- Design Handbook containing analytical methods used and parametric data developed for trade studies and hardware definition - February 1973

- Final Report describing program activities and achievements - February 1973

2.3 General System Design Requirements

2.3.1 Background

The initial requirements which formed the basis for the solar array design and analysis were derived as an iterative process by MSC, MSFC, LMSC, McDonnell Douglas, and North American Rockwell. Hardware requirements were then derived from these general requirements based on specific loads, packaging constraints, and performance goals.

The initial design activity was based on a 33-foot-diameter single launch station. Later design activities responded to the requirements of a shuttle-launched modular Space Station. Fortunately, the power boom design remained generally the same throughout the station design iterations, and the solar array design was therefore not perturbed significantly. General familiarization with overall configurations and introduction to major design-induced requirements can be obtained by reviewing Section 1 of the Design and Analysis Topical Report (Red Book - LMSC A995719).

The early Space Station studies which concentrated on the 33-foot diameter single-launch module included an artificial-g experiment as shown in Figure 2.3-1. The artificial-g mode represented a worst-case maximum loading condition for solar array structure design. This design requirement was retained throughout the contract in spite of the fact that it was later dropped as a Space Station requirement.

Because the overall Space Station design was continuously being modified and in addition varied between the two parallel contractor/NASA efforts, it was necessary in many instances to assume design requirements or to choose between conflicting sets of design requirements.

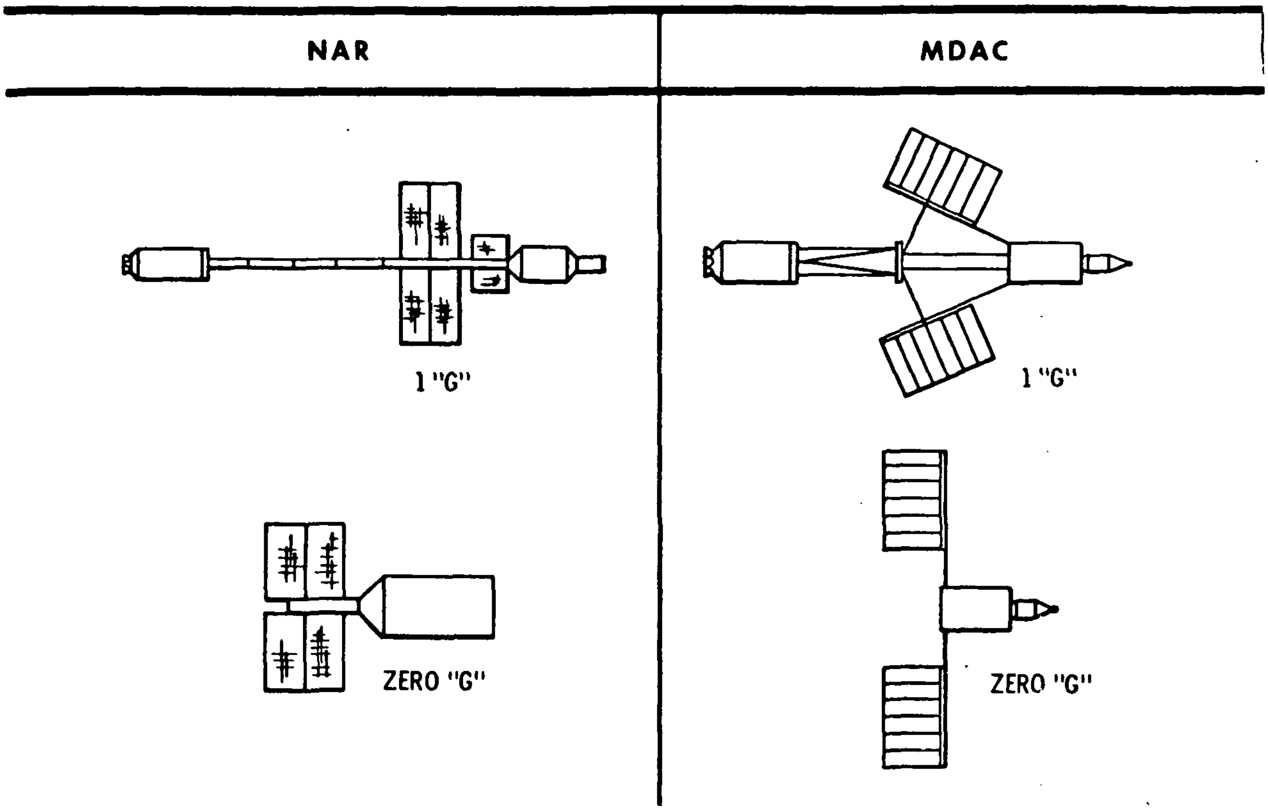


Figure 2.3-1 Saturn Launch Configuration Studies

For these reasons, it was necessary to define a requirements philosophy and design selection methodology that would provide a valid demonstration of available component technology in this climate of changing requirements. To do this, two basic ground-rules for requirements definitions were established:

a. Make maximum use of modularity to provide for accommodation of changes at the component and subcomponent levels and to accommodate straight-forward scale-up or scale-down in size.

b. Where a conflict existed between two requirements or potential requirements, adopt the more difficult of the two to achieve, in order that any subsequent changes or firming up in requirements will result in a less difficult design or fabrication problem and thus not compromise the validity of the technology demonstrated in the program.

Space allocation changes impacted the packaging design of the array considerably. On the basis of the most recent shuttle launch configurations, the 5-foot tunnel clearance requirement was retained for the localized area of the orientation drive and power transfer, while the maximum diameter was set at 14 feet and the maximum length at 38 feet. For the last MSFC configuration, the overall constraints were the same, with the exception that the inside diameter of the tracking system mated with a 40-inch-diameter boom.

As requirements were defined or derived, the applicable background and evaluation associated with the selection of the given requirement was published in bulletin form, with these bulletins being compiled into a requirements and constraints document, an up-to-date copy of which was kept by both LMSC and MSC. In this way, a continuing well defined set of design groundrules was maintained throughout the program.

The last baseline Space Station configurations considered were the NAR cruciform and barbell designs and the MDAC radial and linear designs shown in Figure 2.3-2. NAR placed more emphasis on the barbell configuration and MDAC on a "hybrid" radial configuration.

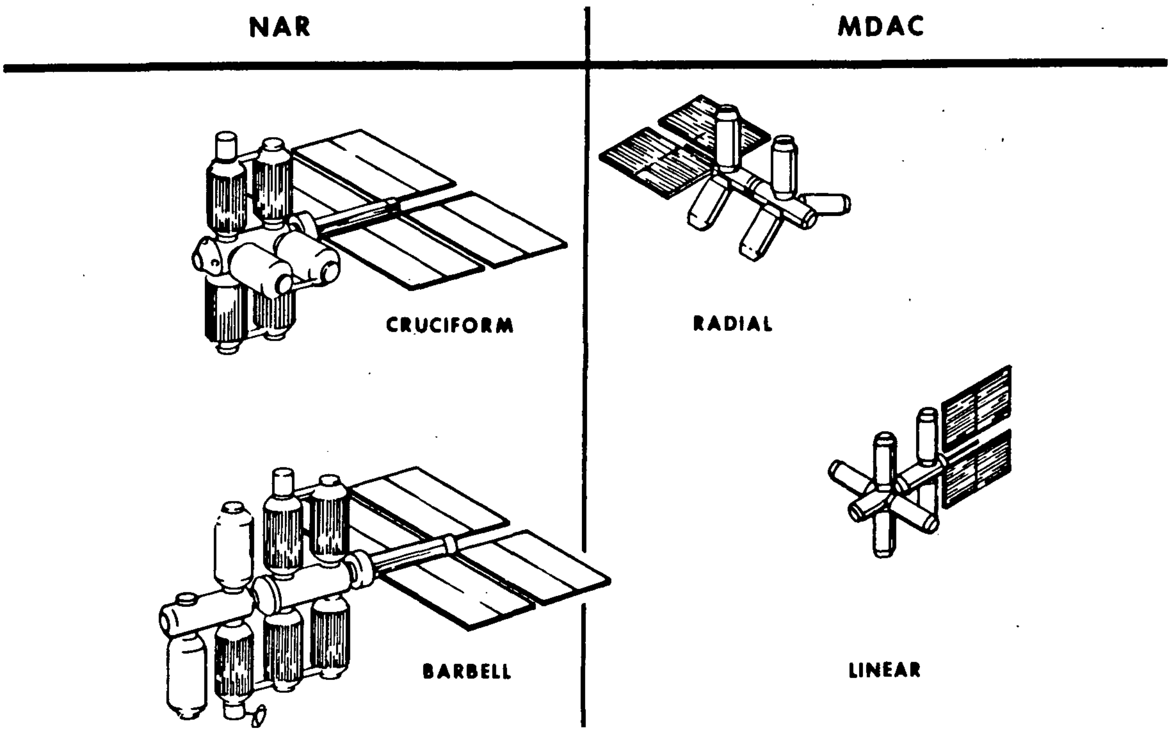


Figure 2.3-2 Shuttle Launch Stations

2.3.2 General Requirements Definition

In addition to the RFP, the major source for initial design requirements was MSC-03696, modular Space Station guidelines and constraints document. As indicated in Table 2.3-1 four requirements were adopted from this document directly without modification. These include the module weight of 20,000 pounds, the shuttle launch mode, the launch parameter of 55 deg. inclination, and the 240-270 nautical mile altitude. All other requirements of this document were met or exceeded. Although not stated in the guidelines document, one derived requirement was that the complete power module should be replaceable with no EVA and that individual solar array strips should be replaceable with EVA.

TABLE 2.3-1

DESIGN BASELINE REQUIREMENTS

ITEM \ SOURCE	MSC - 03696	DERIVED	ASSUMED
CARGO			
MODULE SIZE	14' X 58'	14' X 38'	
ART. "G" MODE	NOT ON 1ST STATION	CAPABILITY FOR ALL STATIONS	ART. "G" AT START OF FLIGHT ONLY
POWER LEVEL	15 KW AVG (MIN)	25-KW AVG 100-KW MAX	
MODULE WEIGHT	20,000 LB		
LAUNCH MODE	SHUTTLE		
RESUPPLY LEVEL		COMPLETE POWER MODULE/ NO EVA	REPLACE STRIP WITH EVA
INCLINATION	55°		
ALTITUDE	240 - 270 NM		

2.3.3 Electrical Power Requirements

At the beginning of the design phase in the Space Station Solar Array Program, the 24-hour average power requirements indicated by three different Space Station studies varied between 15 and 24.1 kilowatts for the initial Space Station and up to 38 kilowatts for the growth Space Station, as shown in Table 2.3-2. It should be pointed out that the MSFC growth Space Station provides the 38 kilowatts through two solar arrays, one located at each end of the Space Station (19 kilowatts each). Each array power requirement therefore is less than the 25 kilowatts selected for the system being developed under the present program.

TABLE 2.3-2

SPACE STATION ELECTRICAL POWER REQUIREMENTS

Build up Status	Incremental	Initial Space Station	Growth Space Station	LMSC Array Capability
Manning Level / Config.	3	6	12	Kilowatts (minimum)
MSC (MSC-03696)	-	15 KW	-	25
NAR	-	19.6 KW	30.3 KW	25
MDAC	13.7 KW	17.3 (5 yr EOL)	32.1 (10 yr EOL)	25

The conservative 25-kilowatt minimum capability was selected for the initial solar array design. Furthermore, since the 10,000-square foot array size called out in the original RFP was retained, the actual predicted end of life power output for the array, excluding battery charging capability, was estimated to be 40 kilowatts. It is highly probable that this single solar array design could accommodate all requirements listed in the table, including the growth stations.

3.0 MAJOR PROGRAM PHASES

Topical reports were published describing the three (3) major program phases: the Technology Evaluation, the Design and Analysis Task, and the Test Evaluation Program, which resulted in a feasible design for a 10,000 ft^2 solar array for the Space Station. These program phases are summarized in this section with photographs and data taken from the Topical Reports.

3.1 Technology Evaluation Phase

This major task was initiated at the beginning of the contract (July 1970) and continued through November 1970. The results of this survey and evaluation were published in the first (1st) Topical Report (LMSC A981486), December 1970. Two hundred fifty (250) copies of this document were printed and distributed throughout Government Centers and industry. This document, for the first time, assembled and evaluated all the known solar array technology developed from 1965 through 1969. A supplemental publication, which updated the information through 1971 (LMSC/D159124), was completed and published in July 1972.

3.1.1 Data Acquisition and Evaluation

The major sources for technical information for the technology evaluation task were Lockheed in-house independent development reports, conference proceedings--particularly the photovoltaic, power, and mechanisms oriented conferences. Contract reports from previous NASA and Air Force flexible solar array programs and personal contacts with over 100 specialists in the various technical fields contributing to solar array design were also utilized. These materials were all compiled and evaluated to determine the availability of approaches, data, and components for the design of a ten thousand (10,000) sq. ft. Space Station solar array. This information was compared to the baseline design requirements for the space station array and tracking system to determine applicability of this background towards the specific design problem at hand. From this evaluation the feasibility for a scale up of the various concepts, techniques and approaches was evaluated and the major output of this evaluation was the data employed directly in design analysis of the array, tracking and power transfer systems, and recommendations for emphasis for further analysis and tests during the present program.

At MSC direction, the Technology Evaluation reports were designed in such a way that updating of the technical information could be accomplished conveniently. Further, the two reports were designed in a handbook format so that it could be used as an easy reference for solar array designers.

The basic report format is shown in Figure 3.1-1. The bulk of the report consists of a matrix type presentation of data on the major components of the space station solar array system. Text was held to an absolute minimum in order to maximize the utility of the tabulated data to the designer without causing him to leaf through volumes of narrative matter. An extensive bibliography is included which summarizes the content of each of the references used in the evaluation and indicates the applicability of the material to the space station solar array design problem. This allows the reader to research a given subject more thoroughly without have to re-read all the references in order to do so. The gaps in technology which were identified during the technology evaluation are presented in the form of recommended R&D

programs which should be conducted to augment the effort being conducted under the present program. In the update document the technology gaps were summarized in tabular form to identify status and provide guidelines for agencies undertaking work in these areas.

FORMAT	CONTENT
o INTRODUCTION	o SCOPE AND OBJECTIVES o GUIDELINE REQUIREMENTS o DATA SOURCES
o SUMMARY	o DISCUSSION OF RESULTS
o EXISTING TECHNOLOGY • SYSTEMS • ARRAY • STRUCTURE • ORIENTATION DRIVE • POWER TRANSFERS	o METHODOLOGY OF DATA ACQUISITION o SURVEY, CATEGORIZATION, AND EXPLANATION OF TECHNOLOGY RELATED TO LIGHTWEIGHT TRACKING ARRAY SYSTEMS
o APPLICABLE TECHNOLOGY	o EVALUATION, IDENTIFICATION, AND REVIEW OF USABLE, PREVIOUSLY CONDUCTED WORK
o TECHNOLOGY ADVANCEMENT REQUIRED	o SUMMATION OF DEVELOPMENT, STUDIES, AND TESTING RECOMMENDED
o APPENDIX	o CROSS-REFERENCE BIBLIOGRAPHY

Figure 3.1-1 Technology Evaluation Report

3.1.2 Recommended Technology Projects

The Recommended Projects' summary charts appearing in the Update document (LMSC-D159124) which were organized in three categories, are included here for easy reference (Charts 3.1-1, 3.1-2 and 3.1-3). They describe the present status of all the technology gaps uncovered during the first phase of the Program.

Chart 3.1-1

RECOMMENDED TECHNOLOGY PROGRAMS

Category I - Projects Conducted on the Space Station Solar Array Program

RECOMMENDED STUDIES	SUMMARY OF RECOMMENDATIONS	CURRENT OR COMPLETED ACTIVITY	PLANNED ACTIVITY
1. SPACE STATION SHADING STUDIES	Vehicle on array shade factors are used as basis for radiator location and electrical design. Shading patterns, as a function of inclination angle and vehicle position in orbit, along with shade sweep direction should be determined. Photographic shading and computer analyses should be made for selected configurations to determine power output losses.	Preliminary studies at LMSC (L. 4-48 Appendix B. 4). Shadowing study by JPL Venus Mercury Fly-By-Solar Panel (N. 4-24). TRW Array Shading Studies for ATM, OWS.	Additional shading study will be performed when Space Station configuration is fully defined. Shading model and computer program available for this purpose.
2. BASIC SUBSTRATE MATERIALS EVALUATION	Mechanical properties determined over required temperature ranges. Changes in properties due to UV, vacuum, and prolonged temperature cycling. Specimens must include laminates and module joints. Determines parametric life data.	Creep, Tensile, and Tear Tests at LMSC (L. 4-37 and L. 4-57). Creep properties compilation by Allied Chemical (A. 7-1). Creep behavior of polymers (I. 1-2).	NASA MSFC contract NAS8-28432 with LMSC to study basic materials involved, and optimize flexible substrate design.
3. SUBSTRATE PACKAGING EVALUATION	Large area flexible array packaging designs, including rollup, flat foldouts, and others requiring either integral or separate padding techniques require full scale feasibility and performance demonstration and evaluation. Mechanical complexity added to the automatic repackaging during retraction requirement set the pace here.	Lockheed flatfold studies and test (L. 4-47, L. 4-48, L. 4-57). General Electric final rollup tests (G. 2-9 and G. 2-21).	Possible shuttle-launched experiment under consideration by MSFC.
4. DEPLOYABLE STRUCTURE TEST AND EVALUATION	The central mechanical component of any packaged flexible solar array system is the extendible boom which deploys and retracts the panels. All types of booms should be fabricated and tested (small and full scale) to determine characteristics for future designs. Data should include unloaded alignment, stiffness, buckling (lateral and column loads) and packaging values.	Lockheed tests on Astromast (L. 4-57). General Electric tests on SPAR (G. 2-9 and G. 2-21).	NASA MSFC/MSC CVT program will perform additional evaluation testing of Space Station Solar Array hardware.
5. DEPLOYMENT DRIVE AND TENSIONING MECHANISM EVALUATION	These devices control boom deployment, array segment retraction, and uniformity of substrate tensioning under variable orbital load conditions (0 to 1 "g"). Substrate length variations due to thermal growth and creep must be considered. Scale-up data and definition of cost in weight and complexity for candidate systems should be obtained.	Design support and major hardware tests by Lockheed (L. 4-48, L. 4-47).	(Same as above)
6. FULL-SCALE ARRAY ASSEMBLY TESTING	To evaluate design concepts and analytical procedures used in design and to assess problems in fabrication, assembly, and test, full scale tests of the major array components should be conducted. These tests would provide simultaneous evaluation of ground handling methods and mechanical/electrical acceptance test techniques.	Lockheed Array Quadrant tests (L. 4-57). General Electric rollup tests (G. 2-9 and G. 2-21).	(Same as above)
7. LUBRICATION TESTS	No one lubricant can perform properly for all required space applications. Tests must be conducted to select the best lubricant for each requirement. Operating modes which cause cold welding, increase viscous drag, increase start and running torques, and cause surface degradation of bearing elements must be determined and solved.	Design support tests by BBRC (L. 4-57)	May be included in CVT life testing.
8. DRIVE MOTOR EVALUATION	Ten-year vacuum operation, periodic maintenance, and component replacement requirements, coupled with the large stall torques of the tracking system demands thorough evaluation and performance testing of current commercially available motors before final design selection.	Design support tests by BBRC	(Same as above)

Chart 3.1-1 (Cont'd)

RECOMMENDED STUDIES	SUMMARY OF RECOMMENDATIONS	CURRENT OR COMPLETED ACTIVITY	PLANNED ACTIVITY
9. FULL SCALE DRIVE SYSTEM EVALUATION	No automatic 2-axis orientation system of this size with manual override provisions has ever been tested. No such system has sustained loads in a high artificial "g" field. A test model should be tested under simulated docking and spin mode stresses and launch loads. Tracking and gear dynamics should be studied.	Interaction Study by Fairchild-Hiller (F.1-13, F.1-14, F.1-15). Major hardware fab and test by BBRC (L.4-48, L.4-47). NASA Goddard work (N.2-19).	NASA MSFC/MSC CVT program will perform additional evaluation testing on Space Station Solar Array hardware.
10. MAINTAINABILITY: BEARINGS, BRUSHES, MOTORS	Drive systems to date have light loads and are of small size. The long duration (10 year) space station application makes an effective maintenance/replenishment philosophy mandatory. Major hardware endurance testing coupled with degradation failure mode analyses and design complexity assessments should produce required results.	No activity identified except the information obtained and reported in the Space Station Solar Array Program Topical Reports, NAS9-11039.	No activity planned. Should be done in conjunction with overall Space Station studies.
11. TRACKING SYSTEM DUTY CYCLE ANALYSIS	Determination of the total travel and rate requirements of the two-axis drive system for various possible Space Station flight modes (computer analysis). Evaluation of the impact of these on drive and power transfer hardware design and design complexity.	Preliminary analysis and design by BBRC (L.4-48). Work by Hughes Aircraft for Air Force (H.6-29). Lockheed/MSC Gimbal and Drive Study (NAS9-11874)	No activity planned.
12. SLIP RING MATERIAL EVALUATION	This test series will determine the effect of static and slow speed sliding performance of power-type brush/slip ring combinations in vacuum. Tests should be conducted over a range of brush pressure, current density, lubricant types, and speeds from 0 to 6 degrees per minute to determine friction and wear rate.	Hardware design and tests by BBRC (L.4-48, L.4-57).	May be included in CVT life testing.
13. FLEXIBLE CABLE EVALUATION	Test information relative to flex cables alone or in combination with various power transfer devices is critical to final design. Maximum life limiting factors, stiffness of cabling carrying 100,000 volt-amperes with minimizing torque, power consumption, and cooling requirements is desired. Vacuum cycling ±180 degree flexure tests on cable configurations is required.	Weight and volume tradeoffs conducted on NAS9-11039 by BBRC (L.4-48).	No activity planned.
14. FULL SCALE POWER TRANSFER TESTS	Key problems associated with space station solar array power transfer involve scale-up of present capability and physical unit size, coupled with 10 year life. A power transfer model with full scale current density and thermal configuration should be thermal vacuum tested to determine friction drag, temperature rise, power dissipation, wear, and electrical noise.	Tests on Space Station Major Hardware components by BBRC (L.4-57).	No activity planned.

Chart 3.1-2
RECOMMENDED TECHNOLOGY PROGRAMS
Category II - Additional Projects to Ensure Technology Readiness

RECOMMENDED STUDIES	SUMMARY OF RECOMMENDATIONS	CURRENT OR COMPLETED ACTIVITY	PLANNED ACTIVITY
1. TEMPERATURE CYCLING PROGRAM	Survival of the Space Station Solar Array over 58,000 temperature cycles in low earth orbit over ten year life requires comparative data (non-existent) on candidate cell/substrate assemblies to determine designs which will withstand this environment. To demonstrate this capability, a high test sample capacity facility with "in situ" output measurement capability should be built. Test results should reveal basic failure mechanisms.	Temperature Cycling Plan by Lockheed (L. 4-58), published in August 1972. Other activities as described in Table 4.1.16 of this report.	Temperature cycling program set up to standardize method of temperature cycle testing and to evaluate flexible array modules– follow-on to NAS9-11039. In-house cycling of advanced modules planned at MSFC.
2. FLEXIBLE INTERCONNECT DEVELOPMENT	Tests in this program should include high and low temperature fatigue and tensile tests of candidate materials (copper, kovar, molybdenum, aluminum, aluminum-40% copper and silver), and geometry combinations applied both to adhesive-bonded and integral-pointed circuit flexible substrate assemblies. Promising approaches using appropriate joining techniques and solar cell assemblies should be fabricated for inclusion in temp. cycling test program, (1) above.	Integral substrate-interconnect laminates by Lockheed (L. 4-48). Development work at TRW under Air Force contract (T. 3-36). Also, Ion Physics (I. 3-14).	Interconnect materials study and evaluation by testing planned for NAS8-28432 contract to be performed by LMSC with MSFC.
3. SOLDERLESS JOINING TECHNIQUES	No standardized test programs exist to develop and compare solderless interconnection methods. Solar cell assembly techniques including brazing, welding, ultrasonic bonding, and thermocompression should be investigated and comparatively evaluated. Promising methods for flexible cell assemblies should be incorporated in temp. cycling test program. Cost reduction and ease of repair should be emphasized.	Being developed by Lockheed under MSFC Huntsville contract (final report in April 1973) NAS8-28432. New contract, NASA-LeRC, with TRW. Spectrolab contract with COMSAT Labs.	Solderless joining technique development will be included in NAS8-28432, described above. Testing to be done at MSFC.
4. UV AND IRRADIATION TEST – FLEXIBLE SUBSTRATES	Long term effects of combined vacuum, ultra violet, and penetrating radiation on the structural and thermal properties of the polymeric substrate materials, used as major structural components, should be determined by a test program. Post-radiation tensile and creep testing (-250°F to +200°F range) should be included on candidate substrate materials such as Kapton, FEP Fiberglass, and laminates of these materials.	Tensile, tear, and creep tests by Lockheed (L. 4-57) – temperature only – no environments.	Classified projects working on effects of irradiation on flexible substrate assemblies.
5. FLEXIBLE ARRAY THERMAL PROPERTIES DETERMINATION	Test data on thermal and optical properties (emissivity, absorptivity, transmissivity, reflectivity, specific heat, coefficient of expansion, and thermal conductivity) of the flexible substrate materials and laminates, solar cells and solders, as a function of temperature down to -300°F, are required for both design and the thermal cycling program. This program should be combined with the Radiation Program.	Being compiled by Lockheed under MSFC Huntsville contract (Final Report in April 1973) contract NAS8-28432. (Reference Category I – Items 2 & 3)	Work to be performed on NAS8-28432 through 1972.
6. STORAGE/LIFE TESTING OF ERECTION/ RETRACTION COMPONENTS	Flexible solar cell arrays require preloads for ascent protection. These preloads are provided in drum rollup systems by substrate tensioning, and in flat-fold systems by compression between structural covers. Both systems employ padding material for cell protection. Permanent set (edge curl for drum configurations and creases for flat-fold) and environmental effects under long storage could result and should be determined. Structural components such as springs, cables, and bearings should be included as to long exposure to space environment effects.	No activity.	Will be included in NASA MSFC CVT program to evaluate Space Station hardware.

Chart 3.1-2 (Cont'd)

RECOMMENDED STUDIES	SUMMARY OF RECOMMENDATIONS	CURRENT OR COMPLETED ACTIVITY	PLANNED ACTIVITY
7. ARRAY-STATION INTERACTION STUDY	Determination of dynamic compatibility based on a model simulating Space Station structure, solar array structure, and Space Station guidance and control system. Present program does not include artificial gravity model. Development of dynamic model (continuous improvement) using inputs of test program results (boom stiffness, mass properties, substrate tension, and actual tracking drive properties is required.	Computer programs being developed and used by Fairchild-Hiller (F.1-13, F.1-14, F.1-15).	Additional work to be performed by NASA-MSC when results of additional structural characterization testing are available.
8. ALTERNATE LARGE BEARING SYSTEM TEST	Bearing or roller system could transmit total dynamics loads between two large cylinders (solar array boom and space station power boom). Little known in the area of large ball bearings and small rollers. A full scale bearing and drive structure must be fabricated for testing and evaluation of rolling friction starting torque, wear for either roller or ball systems.	No activity identified.	No activity planned.
9. LIFE TESTING – DRIVE SYSTEM	Continuation and extension of Category I – Program 9 testing. Replaces the idea of evaluating long term effects by accelerated tests which could give misleading results. Hardware from Category I – Item 9 with slight modification can be used here.	BBRC test program (L.4-57).	May be a part of NASA MSFC-CVT program.
10. ENVIRONMENTAL LIFE TESTS – POWER TRANSFER ASSEMBLY	Continuation and extension of Category I – Program 14 testing. Will increase the accuracy in operational performance prediction.	No activity identified.	No activity planned for full scale testing.

Chart 3.1-3
RECOMMENDED TECHNOLOGY PROGRAMS

Category III - Projects Providing Significant Downstream Improvement

RECOMMENDED STUDIES	SUMMARY OF RECOMMENDATIONS	CURRENT OR COMPLETED ACTIVITY	PLANNED ACTIVITY
1. TEFLON COVER EVALUATION	Inadequate process and production technique optimization and environmental testing for this newcomer as coverglass material. Applicable to any size power system with great weight and cost savings potential. Tests measuring degradation of teflon covers by particle and UV radiation and determination of thickness to application/environment are required.	• Lockheed investigation for NASA-LeRC (L. 4-34 and L. 4-42). • NASA LeRC contract with TRW initiated February 1972 to fabricate and test heat-bonded teflon covers for solar cells.	Continuation of TRW/LeRC work. LMSC ID work on spray-on teflon solar cell covers.
2. INTEGRAL SOLAR CELL COVERS	Integral covers can be 1-2 mils, as compared to 6 mil minimum for conventional coverglasses. Significant weight reduction and elimination of the adhesive would result. Development of processes and material for use with standard cell manufacturing techniques and of production capability is required. Heavy process development expenditures should not be made until this approach is compared with Program 1 (above) results.	• Development work by Heliotek (H. 3-21 and H. 3-24) and Texas Instruments (T. 2-1). • Solar cell coverglass development by Ion Physics (I. 3-16, I. 3-17). • In-house development by NASA Goddard and NASA LeRC (N. 2-28, N. 6-27, N. 6-40 and N. 6-43).	GE funded by JPL for spray plasma deposition of ultra pure fused silica without stress problem.
3. IMPROVEMENT OF EOL SOLAR CELL EFFICIENCY	Investigations state theoretical attainable efficiencies up to 22%. These higher efficiencies can be achieved only by a better understanding of the physical phenomena governing solar cell performance. Electrical degradation in the cell due to UV and particle radiation, as well as repeated temperature cycles, should also be reduced. Testing to evaluate improved cells should be carried out at one central facility to better control conclusions.	• Lithium doping (H. 3-20, H. 3-25, C. 3-12, C. 3-16, A. 1-8, R. 1-25, R. 1-26, R. 1-31, N. 4-22, J. 1-1). • Efficiency improvement (P. 1-4, P. 2-7, N. 7-12, N. 7-13, N. 4-2, N. 4-16, N. 4-34, C. 9-1).	• Centralab and Heliotek will continue development work with NASA LeRC to improve cell efficiency to 20%. • IBM will continue development efforts in Gallium Arsenide cells to verify performance of 18%.
4. WRAPAROUND CONTACT SOLAR CELLS	Development of backside contact cells would result in cost reductions of up to $200/ft^2 by reducing the complexity of panel assembly. Present series connection to the top electrode calls for generous stress relief series tabs and increased cell spacing complicating assembly. Whereas backside contact cell will allow fully automated assembly, reduce series spacing, and padding thickness and weight.	• Heliotek development work (H. 3-26 and H. 3-19. • Centralab development work (C. 3-13 and C. 3-17). Under above contracts wraparound contact cells were developed for both LMSC and Lewis Research Center.	Evaluation of wraparound contact cell application will be performed on NASA MSFC contract NAS8-28432 and on NASA MSC contract NAS9-11039, both with LMSC continuation of LeRC work.
5. STANDARDIZATION OF SOLAR CELL SPECIFICATIONS	Some cell procurement specifications are directed at cell appearance (cosmetic) rather than proven performance criteria. There is a need for specific performance data as function of contact or ohmic strip width, chips and nicks, contact pinholes, and color variations. A joint NASA/industry study team should review the case and prepare a standard cell procurement specification.	• Effort by JPL cell calibration on high altitude bellows (N. 4-50).	An industry and government agency meeting was held by JPL on July 17 & 18 to discuss industry and government viewpoints on standardization. Results not yet published at time of writing. A serious effort is underway.
6. COST EFFECTIVE CELL AND COVER PROCUREMENT	Cell production spans should be pre-programmed to increase production personnel competence (eliminate reassignment and layoffs) and to improve vendor facility and personnel use. This would result in reducing cell costs and in higher quality production. Solar cell production should be administered by a central NASA-Air Force procurement office to a common procurement specification (5 above).	(Same as above)	(Same as above) Terrestrial low cost studies directed out of NASA Lewis (part of A. D. Little team).

Chart 3.1-3 (Cont'd)

RECOMMENDED STUDIES	SUMMARY OF RECOMMENDATIONS	CURRENT OR COMPLETED ACTIVITY	PLANNED ACTIVITY
7. INTEGRATED POWER MODULES WITH ON-ARRAY ELECTRONICS	Multiple electronic modules for voltage regulation and limiting voltage and for fault isolation, mounted directly on the solar array modules, should be thoroughly investigated. Could replace present by-pass diodes and zener diode voltage limiters. Effort should concentrate on comparative testing of circuit elements and of alternate concepts. Electrical testing should be conducted on the panel module level and be directed at heat rejection.	Boeing studies of Hi-voltage solar arrays (B. 3-28 and B. 3-29). Hi-voltage array work by Hughes (H. 6-32 and H. 6-33).	No activity planned for Space Station Systems.
8. OPTICAL FILTER EVALUATION	Determine the exact degree of "blue" filter protection to cell and adhesives and find a pure "red" filter that does not degrade in space environment and which would reflect all light energy about 1.2 microns in wavelength. A study of all AR coating should be conducted to find one with low reflectance, high transmission, with low or no degradation. Blue filter elimination would save $0.32 per coverglass. Good red filter would produce 8% increase in power output.	JPL Boeing work on solar cell filters (B. 3-31 and B. 3-32) for Mercury/Venus mission for control of solar array temperatures.	OCLI Studies on reflective filters aimed at rejection of 30% additional solar energy, which could lower temperature by 50°F.
9. COMPOSITE STRUCTURE MATERIAL AND JOINING TECHNIQUES	Generation of basic properties of advanced composites, such as graphite/organic materials, is required for flexible solar array deployment structure. Creep fatigue effects from temp. cycling and bonding methods should be investigated. Fabrication of shapes, and testing of these structurally and thermally, should be considered. There is a high potential payoff in weight and stiffness.	Graphite/epoxy truss members and flexible lenticular sections fabricated and tested at Lockheed under in-house funding.	Full scale truss sections to be built and demonstrated at Lockheed.
10. TEST/EVALUATION OF ALTERNATE POWER TRANSFER DEVICE	Slip rings or flex harness are the only flight demonstrated methods. Power clutches and rotary transformers are relatively new. Power clutch needs development of face plate materials and lubricants. Rotary transformers require scale-up and experimental work for high power applications. Development models of each should be constructed and tested similar to slip ring tests.	Contract effort identified for INTELSAT Programs.	Comsat investigating rotating transformer at Philco Ford and rolling bi-stem type elements at Spar Aerospace. Liquid metal power transfer considered for Canadian Tech.

3.2 Design and Analysis Phase

The Design and Analysis Program took place between December 1970 and November 1971. The first four months were devoted to trade studies of various solar array configurations and the packaging, deployment, retraction and orientation problems associated with each design approach. The remaining eight (8) months after selection of the Baseline were spent completing the design analysis, design support testing and drawings for fabrication of the major components.

3.2.1 Configuration Studies

Two general launch modes--integral launch and shuttle launch--were investigated for the Space Station during the contract. In the earlier study phases, where a single launch 33 foot-diameter station was under consideration, there was a design requirement to incorporate an artificial gravity experiment. This was to be accomplished by rotating the Space Station at 4 rpm about an expended boost vehicle stage counterbalancing mass coupled to the station by a telescoping tunnel or cable system (Figure 3.2.1). This experimental operational mode would be in the initial phase of the mission and be of limited duration. The dominant long-term mode would then be zero-g for the remainder of the mission with the Space Station three axis stabilized. Even though the artificial-g experiment was dropped as a firm requirement for the shuttle-launched station studies, it was retained as a requirement in this technology evaluation program. Space allocation, total array size, and artificial-g loading were the prime design criteria in the configuration tradeoffs.

The major goal of the initial configuration tradeoff studies was to evolve an array configuration that would:

- Package within the power boom envelope
- Deploy the 10,000 ft^2 array blanket
- Accommodate the artificial-g loads while providing at least 40 percent power
- Provide an effective zero g two-axis tracking configuration
- Contain a high degree of modularization and component versatility in order to be adaptable to alternate manned spacecraft configurations.

Early in the program an effort was made to evaluate the general spectrum of configurations (par. 3.1.1) in parallel with refinement efforts on the baseline design (par. 3.1.2). It became apparent that the deployable boom employed to provide the basic structural support for the tensioned flexible solar array system represented the single most critical structural component selection.

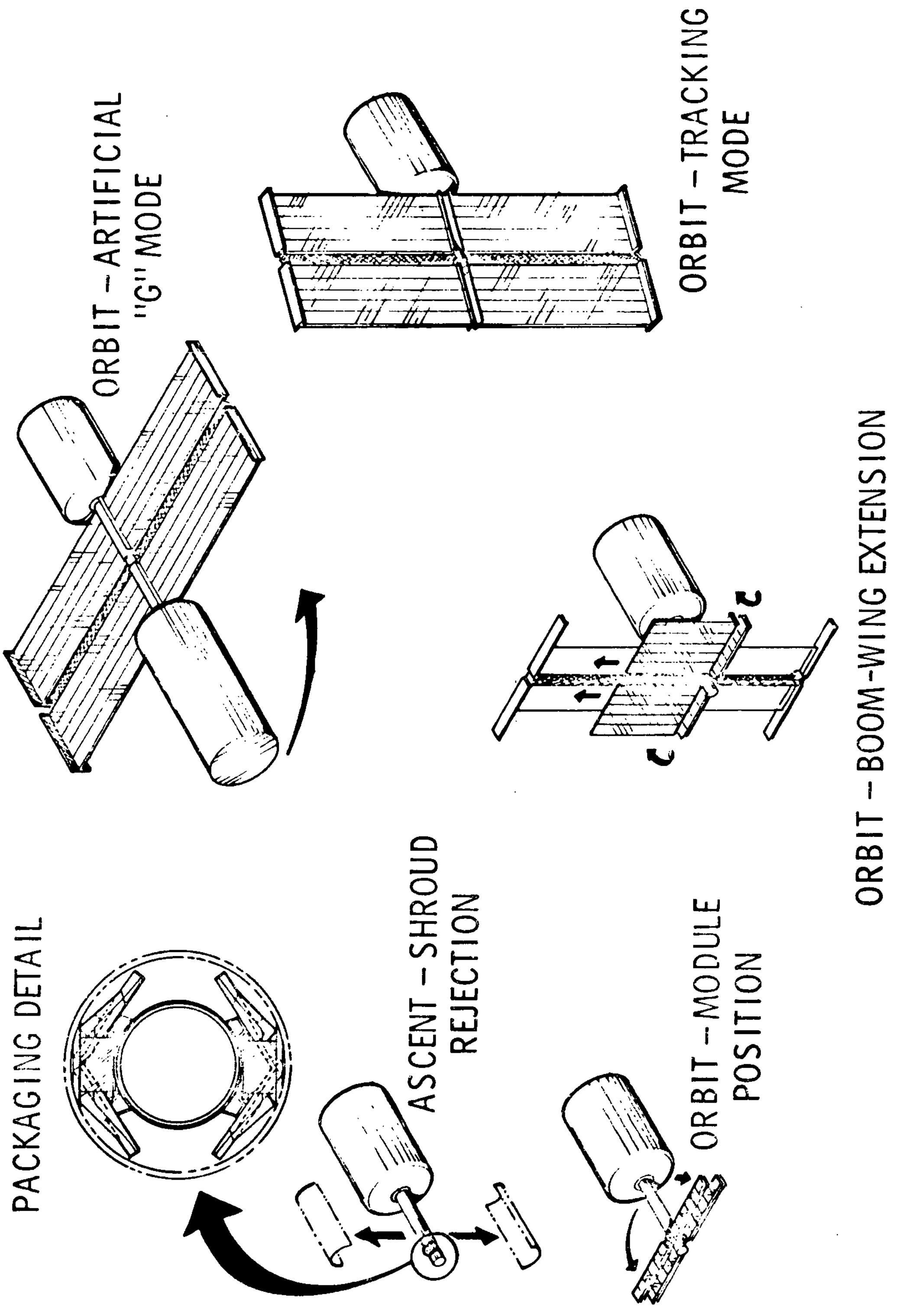


Figure 3.2-1 Proposed Baseline Solar Array and Structure

The general applicability of several candidate boom designs was evaluated. From these general configuration candidates, several alternate baseline designs were selected and studied in sufficient depth to perform a first order evaluation of weight and design complexity in comparison to the proposal baseline system.

In parallel with these efforts, final refinements were made to the proposal baseline design, final selection of artificial- and zero-g deployment modes was made on the basis of an evaluation of sixteen (16) alternates, and a final system configuration selection was made.

The initial task of the design and analysis phase of this program was a general evaluation of 30 different solar array configuration approaches for the combined zero-g and artificial-g modes.

Two basic structural arrangements of the main boom system were considered; main boom parallel to the Space Station axis or main booms normal to the Space Station axis. For each of these boom approaches, a number of basic methods for accommodating artificial-g were considered. The following basic modes for taking structural advantage of partial power operation during artificial g were considered.

- o Partially extend all array strips
- o Extend only a portion of the array strips
- o Partially extend inboard and outboard array strips to variable lengths
- o Leave all array strips fully extended

For each of these modes, the following basic methods for accommodating the shift in center of rotation of the station were applied:

a. Rotating the arrays so that the centrifugal forces are symmetrical with respect to the array structure

b. Translating the array assembly towards the center of rotation

c. Leaving the array center of gravity in a fixed position and design for the high loads.

For any of the thirty (30) possible combinations, a spectrum of boom types, number of booms, guy wire configurations, and aspect ratios were possible. It was therefore necessary to reduce these possibilities to a manageable number by early qualitative evaluation. Most of the combinations considered, especially those requiring extendible beams parallel to station E, variable length partial extensions and translation of the array, represented mechanical complexities and were, therefore, eliminated from the initial studies. Primary study emphasis was placed on the following general configurations:

- Fully extended, fixed cg (proposal baseline)
- Fully extended, rotated
- Array strips nearest boom extended, fixed cg
- Only forward array strips extended, fixed cg
- All array strips partially extended, fixed cg

In parallel with the overall baseline solar array configuration evaluation, a specific detailed evaluation was made of the various candidate booms for purposes of determining versatility of the various candidate boom technologies for a variety of potential configurations and loadings. A detailed evaluation was reported on in the first topical report LMSC-A981486 and the candidate system configurations are described in the second topical report LMSC-A995719.

On the basis of reasonable weight and volume capabilities, coupled with the ability to retract, excellent scale-up and scale-down properties of the basic mechanism, and well established design techniques, the Astromast ETB (Extendible Truss Beam) was selected for further development and the baseline design was changed accordingly.

3.2.2 Baseline Definition

Following completion of the Technology Evaluation and the Configuration Trade Studies, the second major task (Baseline Definition) was initiated. It had as objective the selection and definition of the Space Station Solar Array Baseline configuration. By mid-April 1971 the baseline concept was sufficiently determined so that it could be described in detail in Volume II (same document number). In the ensuing nine months drawings of major components and subassemblies were refined, modified and eventually released to manufacturing and vendor shops for fabrication and assembly. During this same period the Second Topical Report (Red Book: LMSC-A995719) was generated and published in November 1971. It incorporated both of the volumes identified above along with copies of some major layout drawings and photographs of the already completed components and assemblies. The appendices of this volume contains the analyses used in completing the baseline selection.

Baseline Design Study

The original proposal baseline (Figure 3.2-1) was a two-boom system, which remained fully deployed during artificial g, relying on boom stiffness and auxiliary guy-wire support to carry the loads within a reasonable weight. However, it was desirable to be able to retract the entire structure and it was determined that partial deployment would provide adequate power, and the selection of the Astromast provided both the possibility of partial array deployment and a retraction capability.

The three alternate artificial "g" approaches considered are illustrated in Figure 3.2-2 (a), (b) and (c).

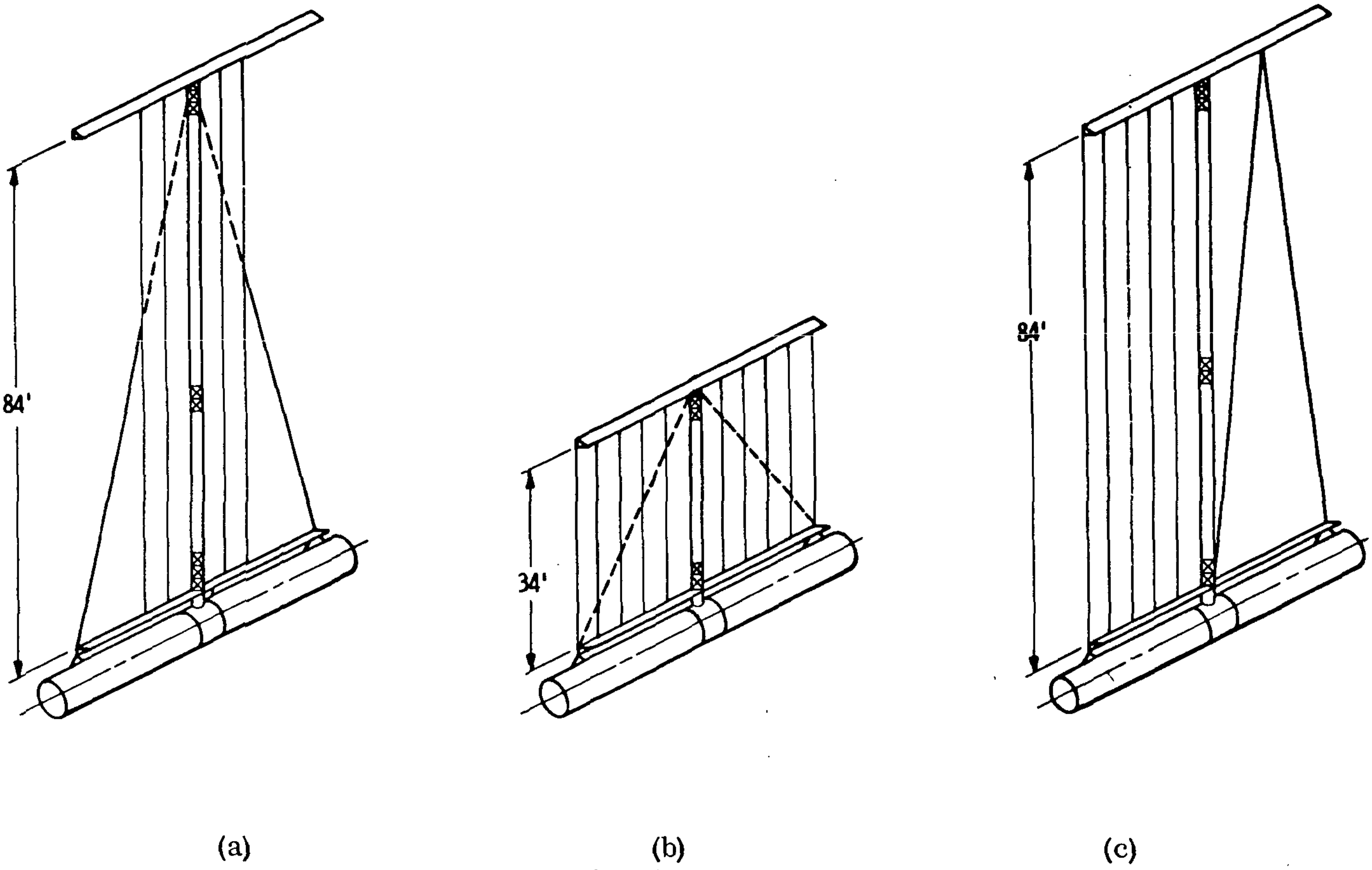


Figure 3.2-2 Artificial g Configuration Candidates

Configuration (a), the configuration selected as baseline, greatly reduces the complexity of the array strip tensioning system for configuration (c) and is also less complex than configuration (b) in mechanization of the array and guy-wire packaging.

Alternate System Design Studies

In addition to the baseline refinement efforts previously discussed, five other completely different baseline approaches were studies in sufficient detail to allow evaluation of loads, weight, and relative complexity.

These designs and their characteristics are presented and discussed in detail in Appendix A of the Design/Analysis document 2nd Topical Report, LMSC-A995719.

Of the five approaches, the lazy tong four-mast array was considered to be the most promising alternate to the baseline. The major advantages of this approach are as follows:

- o No auxiliary guy wires required
- o Capable of supplying 3/4 of array power with one boom completely failed
- o Light weight, all zero-g alternate available by employing all zero g mast construction

System Selection

In the final evaluation between the Four-Mast Lazy-Tong Array and the baseline system, the baseline system was retained for detailed design on the basis of the following considerations:

a. Capability of retraction demonstrated
b. Only a single boom type required
c. Less complex launch packaging of array
d. Lower development and test costs
e. Less complex tensioning system
f. Essentially equal weight
g. Greater versatility of the boom technology

The final selected concept is briefly described in the following Section 3.2.2. The design fabrication and test of the major components of this system are summarized in Section 3.3.

Artificial-g Configuration

The selected artificial-g configuration and the assumed spin-up mode are shown in Figure 3.2-3. The initial station spin up starts at R_1, which is 6 feet toward the station from the center line of the solar array system, and progresses to a final point at R_2, which is 44 feet outboard of the array centerline. A rotational rate of **4** rpm was assumed.

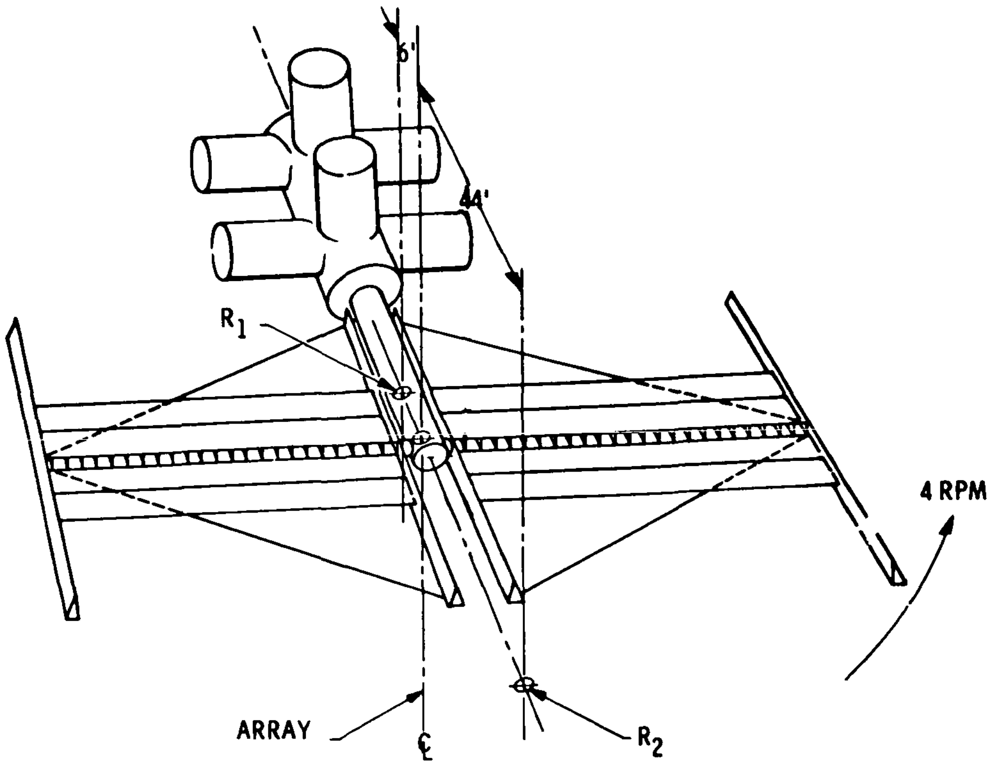


Figure 3.2-3 Selected Aı tificial-g Configuration

Zero-g Configuration

For the zero-g operation, three additional solar array strips are deployed in each array quadrant to bring the total number of deployed solar array strips to 20 for the entire array resulting in 2.5 x the power output available during artificial-g (Figure 3.2-4). The guy cables employed for artificial-g are retained during this mode, but the inboard supports are released from the power boom to allow sun tracking.

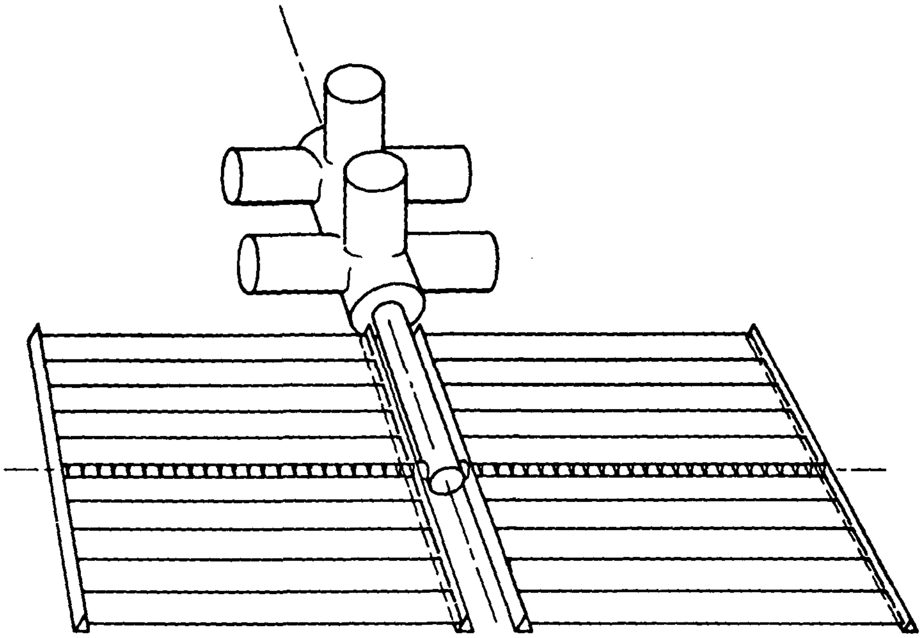


Figure 3.2-4 Selected Zero-g Configuration

3.2.3 Final Baseline Design

Structures/Mechanisms

The initial deployment sequence of the solar array is shown in Figure 3.2-5 starting with the position of the stowed quadrants which are packaged within the 14 ft. maximum envelope which is a basic requirement of the design. The major structural elements of the solar array are shown in Figure 3.2-6 which also depicts the second step in the deployment sequence. The two inboard solar array strips on either side of the boom deploy in this initial sequence to provide power for the artificial-g mode which is the initial mode assumed to be employed in the operation station. The inboard and outboard supports, which also form the upper and lower supports for the packaged array during launch, contain the tensioning mechanisms required for proper support of the arrays. Figure 3.2-7 sketch shows the deployed arrays with guy wires in place and the inset shows further details of the packaged array prior to deployment. The boom selected for this application was the astromast boom shown schematically in Figure 3.2-8. Details of the technical characteristics of this type of boom as opposed to other candidate deployment structures are given in the technology evaluation report "LMSC-A981486" and the Design and Analysis Report "LMSC-A995719".

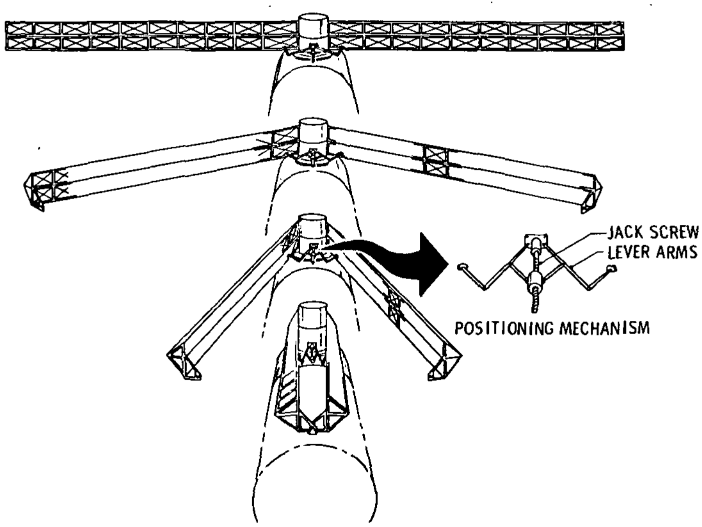


Figure 3.2-5 Initial Positioning Stowed Quadrants

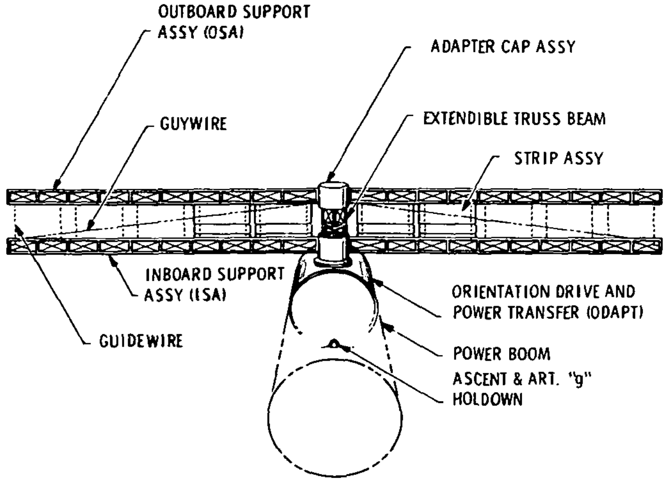


Figure 3.2-6 Baseline Structural Elements

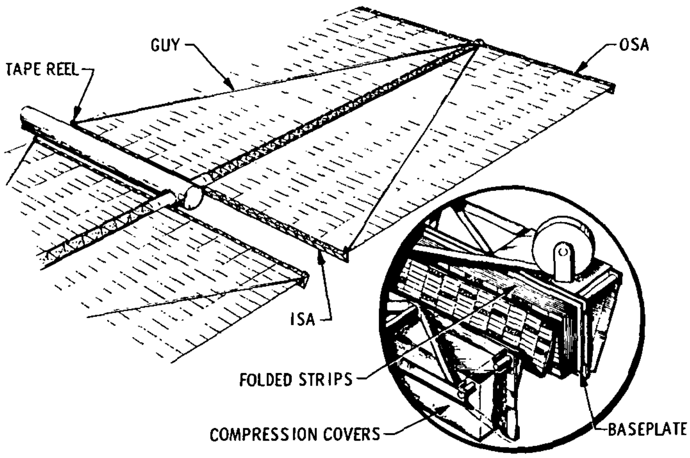


Figure 3.2-7 Array Wing

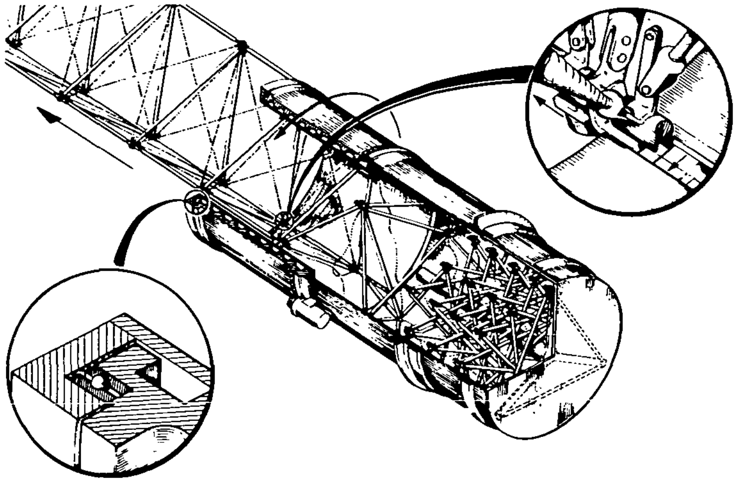


Figure 3.2-8 Cutaway of Astromast (ETB)

Solar Array Substrate/Cell Assemblies

The nomenclature adopted for the baseline solar array strip and module designs are identified in Figure 3.2-9 and 3.2-10. Details of the 6 ft x 84 ft array strip are shown in Figure 3.2-11, including elements of the laminated printed circuit flexible substrate. Producibility was an important consideration in the evolution of these designs and completely automated solar array substrate assembly fabrication techniques were developed. Using wraparound contact solar cells, all soldered or welded joints are inspectable from the rear of the module. The individual module concept (similar to rigid arrays) is retained for ease of repair or replacement by merely unsoldering flex cable attachments (Figure 3.2-12 and sliding off the hinger retainer Figure 3.2-13 at each end of the module).

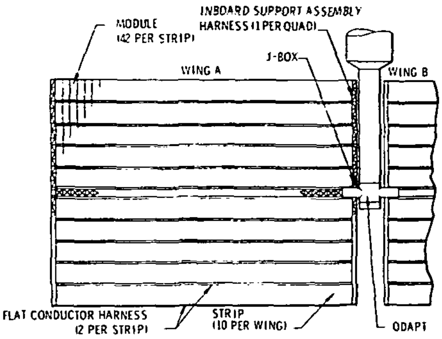


Figure 3.2-9 Power System Nomenclature

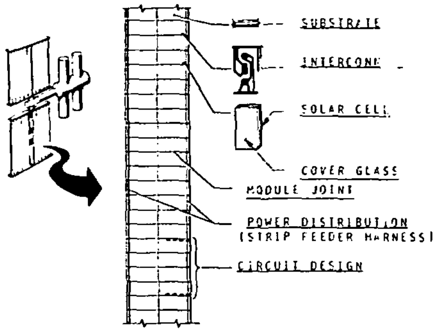


Figure 3.2-10 Solar Array Baseline Electrical Elements

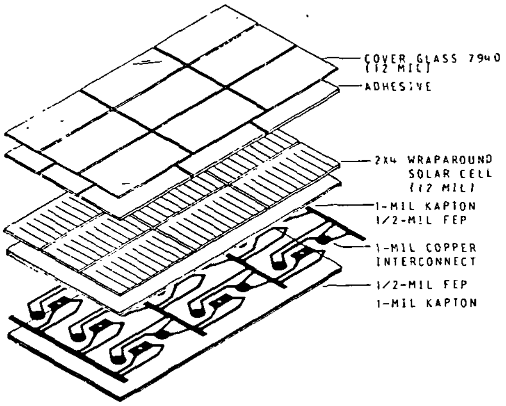


Figure 3.2-11 Substrate Assembly Exploded View

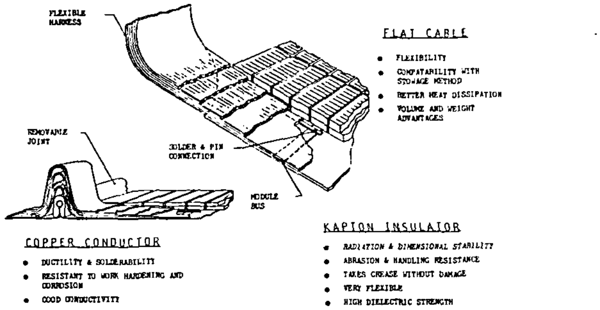


Figure 3.2-12 Flexible Feeder Harness and Connection

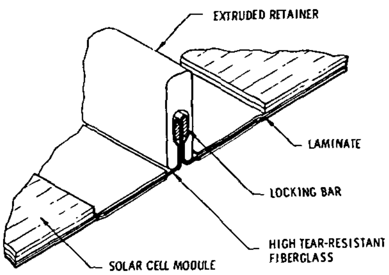
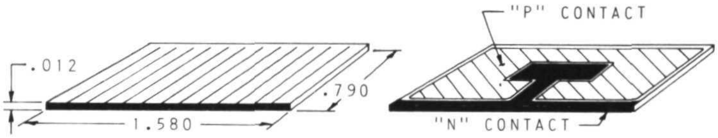


Figure 3.2-13 Hinge Joint

Solar Cell Selection

For the baseline solar cell, sizes up to 2 x 5 cm and thickness from 8-14 mils were considered. Other choices which had to be made included soldered vs solderless cells, reflective coating type, base resistivity and contact configuration. The result of these tradeoffs are shown in Figures 3.2-14 and 3.2-15. Perhaps the two most controversial aspects of the baseline selection are the selection of a wraparound contact and the selection of a 12 mil cell thickness. The fact that the most frequent failure modes on present solar panels are associated with the series tab connection, which this design eliminates, is well substantiated by numerous investigators. This particular step in the operation is also the most expensive part of the substrate assembly process. It involves the most handwork and is therefore most prone to error. It is anticipated that a higher degree of production control can be maintained in an all automated assembly employing back contact cells.



2X4 CM/BACKSIDE CONTACTS-

60% OF 2X2 COST/UNIT
40% ASSY LABOR SAVINGS.
ONE STEP SOLDERING.

BASE RESISTIVITY-2 Ω CM

5% > EOL PWR THAN 10Ω
1.5% < CELLS/UNIT POWER

THICKNESS-12 MIL

4% > EOL PWR THAN 8 MIL
COST SAVINGS OVER 8 MIL UP TO 10%.
STRENGTH ADVANTAGES NEEDED FOR FLEXIBLE ARRAYS.

CONTACTS-PASSIVATED, SOLDERLESS

30% < WEIGHT THAN SOLDERED
HUMIDITY STABILITY

Figure 3.2-14 Silicon Solar Cell

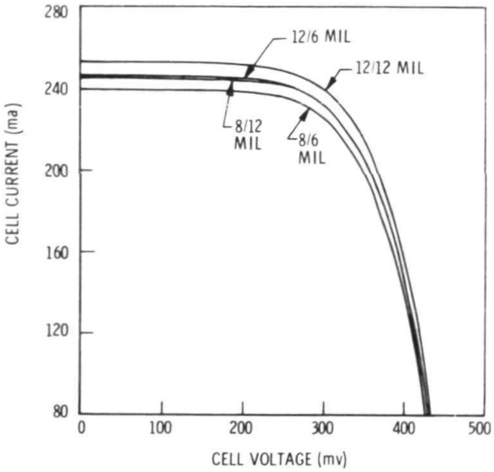


Figure 3.2-15 EOL Solar Cell Characteristics - 71°C

If such advanced cell covers as the teflon systems currently being developed become practical, wraparound contact cells will be much more adaptable to the covering of multiple cell assemblies by teflon sheets. This will provide additional edge protection from natural radiation and a lower overall assembly processing cost. The relative power output for the various combinations of cell and cover thicknesses studied is shown in Figure 3.2-16. Since 12 mil cells with 6 mil covers are significantly more expensive than the 8 mil cell with 12 mil cover, this particular combination was eliminated from consideration early in the study, since the difference in weight between the lower curve and upper curve is approximately 1000 lbs. For the total space station solar array system, this choice was an extremely difficult one. However, the 12 mil cell with 12 mil cover was selected as baseline on the basis of cost and reliability considerations. A summary of module and strip cell configuration is presented in Figure 3.2-17.

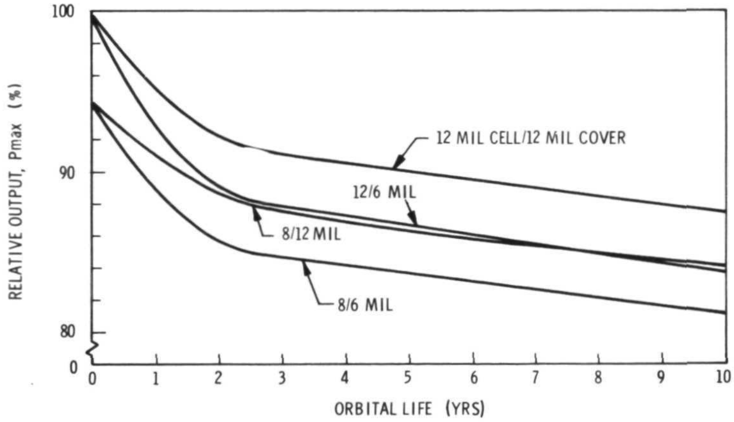


Figure 3.2-16 Relative Power for Combinations of 8 and 12 Mil Cell/6 and 12 Mil Covers

- 7 CELLS IN PARALLEL FOR SUBMODULE.
- 320 CELLS IN SERIES FOR 112 VOLT SYSTEM.
- 47,040 CELLS PER ARRAY STRIP.
- 940,800 CELLS PER VEHICLE.
- APPROXIMATELY 7 WATTS ELECTRICAL POWER PER SQUARE FOOT OF ARRAY AT END OF 10 YEARS.

Figure 3.2-17 Cell Configuration Summary

ODAPT (Orientation Drive and Power Transfer) Selections

The baseline orientation and power transfer drive assembly which mounts on the five foot station boom and interfaces with the solar array structures is shown in Figure 3.2-18. It is composed of 2 gimbal and slip ring assemblies to provide orientation in two axes. The control system is an on-off type capable of orientation in 2 axes to within $\pm 12^{o}$ of normality to the sun. The slip ring assemblies are sized to transfer up to 100 kilowatts of power at 112 volts from the solar array to the vehicle across sliding interfaces. Access for limited maintenance of the lubricant systems is provided through the power boom which as planned will be maintained in a shirtsleeve environment.

The baseline selections for major components and lubrication are:

Major Component	Lubrication
Large Thin Line Ball Bearings	VAC Kote (Oil)
Slip Rings	Niobium Diselenide (Solid)
DC Torque Motor, Brush	VAC Kote (Oil)
Harmonic Drive	VAC Kote (Oil)
Pinion and Ring Gear	VAC Kote (Grease)

Although they represent a step up in both size and design life, the baseline components have an extensive flight history upon which to base their selection.

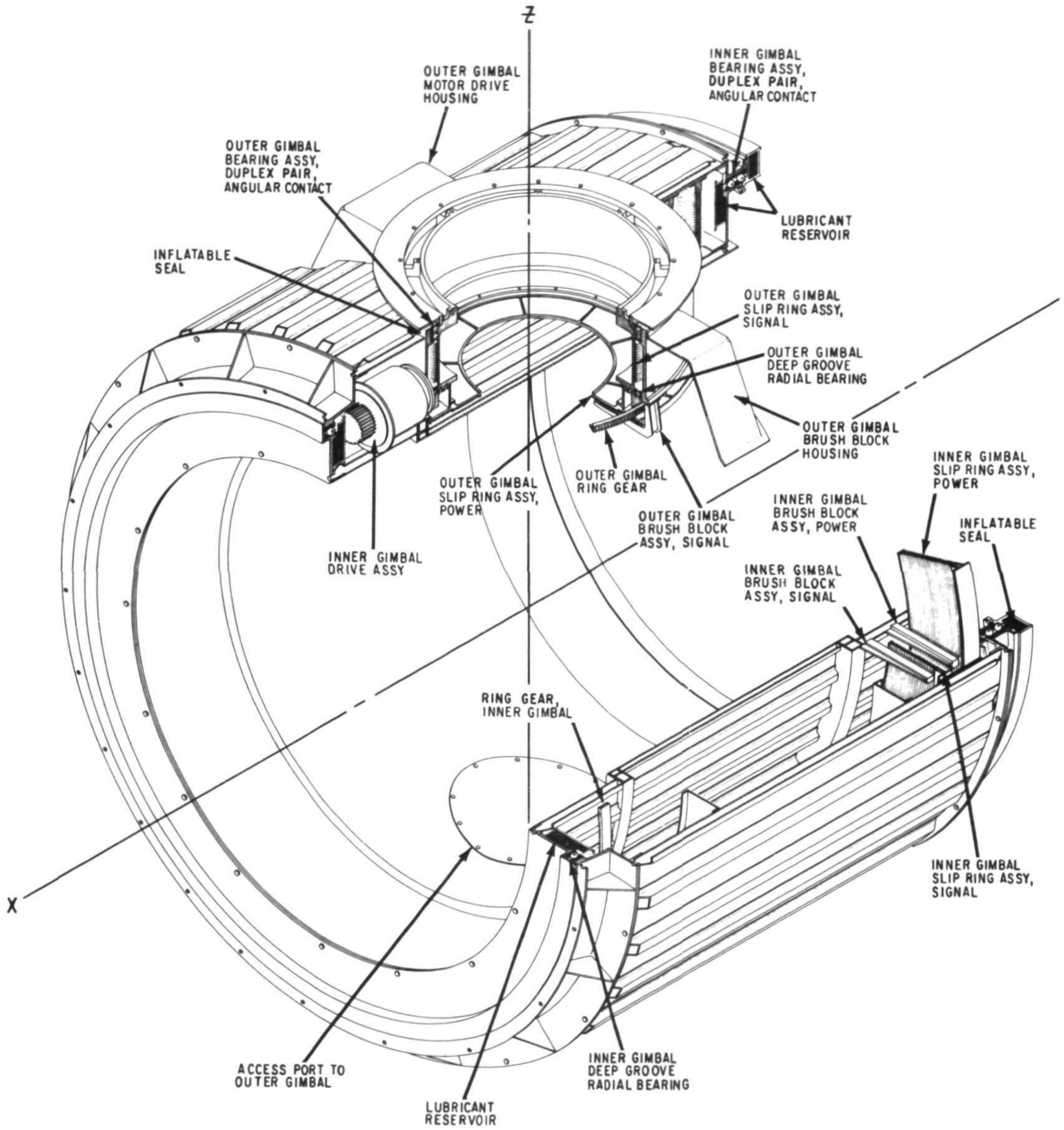


Figure 3.2-18 Cutaway of ODAPT

3.3 Test Evaluation Phase

The Test Evaluation program included all hardware fabrication and the subsequent testing of that hardware. It was therefore by far the most significant portion of the program and required approximately two thirds of the program resources. The program was broken down into the two (2) main tasks of "Hardware Fabrication" and "Hardware Test" which are described in Sections 3.3.1 and 3.3.2.

3.3.1 Hardware Fabrication and Assembly

As drawings were completed and approved, they were released for purchase and/or fabrication of system parts, components and assemblies to be used in various tests. This section identifies and describes major components and assemblies with photographs showing various stages of fabrication.

3.3.1.1 Structures - Mechanisms

The key structural element of the Solar Array, the Astromast ETB (Extendible Truss Beam) deploys the array, supports it under loads, and retracts the system. A subcontract for the design, fabrication and assembly of this unit was awarded to Astro Research Corporation (Santa Barbara, Calif.) in the last week of February 1971 with a delivery time of 18 weeks. It was shipped to LMSC in less than the allotted time on 21 June 1971.

Acceptance testing of the ETB was successfully completed on 13 July 1971. The ETB provides deployment and retraction mechanization and structural strength to sustain orbital loads. Figure 3.3-1 shows the next stage of assembly for the deployment system wherein a tip cap and adapter are added to tie the beam to the inboard support and outboard support assemblies. The four pins shown form the pivot points for these assemblies. The three drive motors employed for beam deployment are shown on the canister adapter wall.

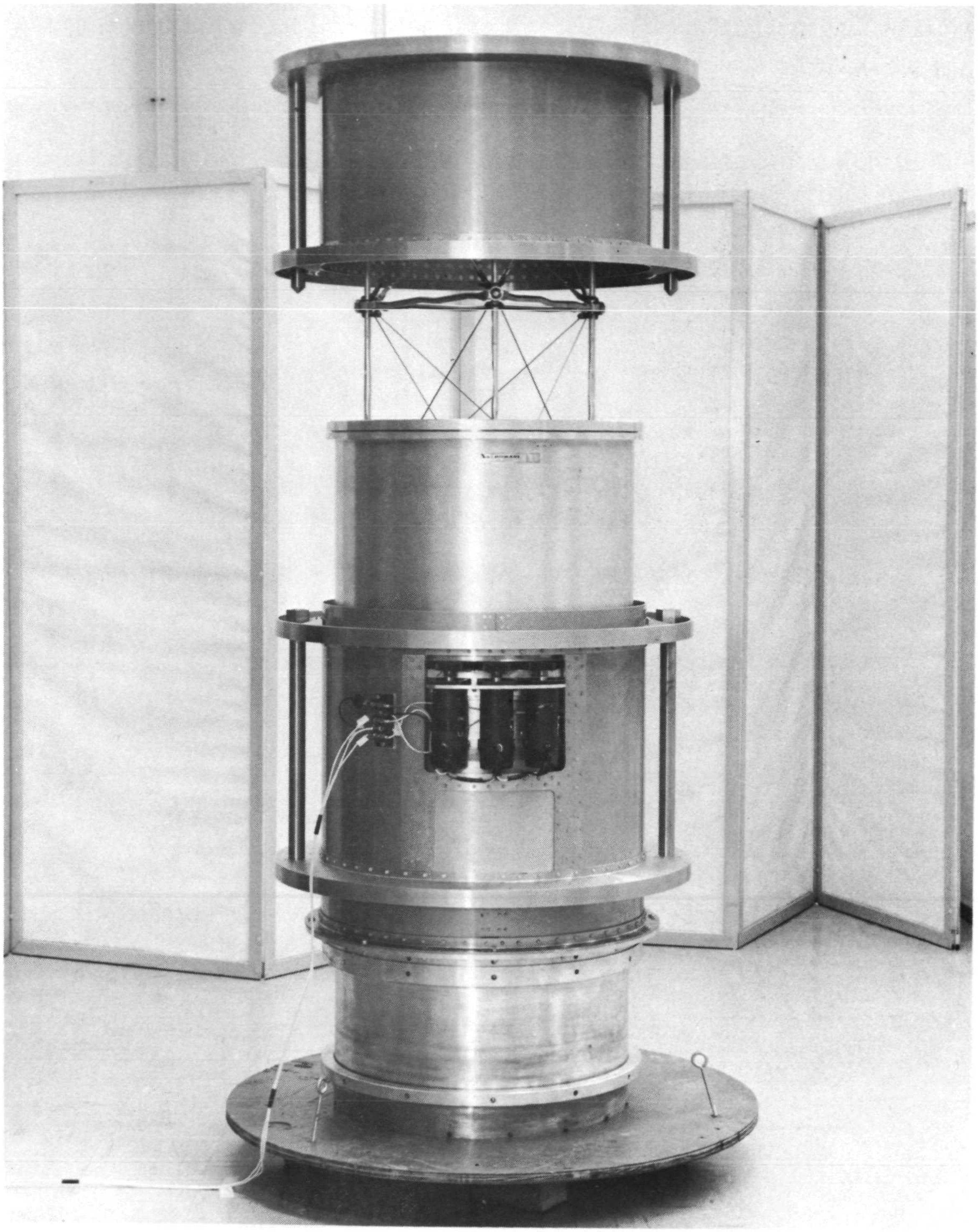


Figure 3.3-1 ETB Assembly Deployed One Bay

The complete major structural assembly and the method of deployment of the inboard support assembly and outboard support assemblies are shown in Figures 3.3-2 and 3.3-3.

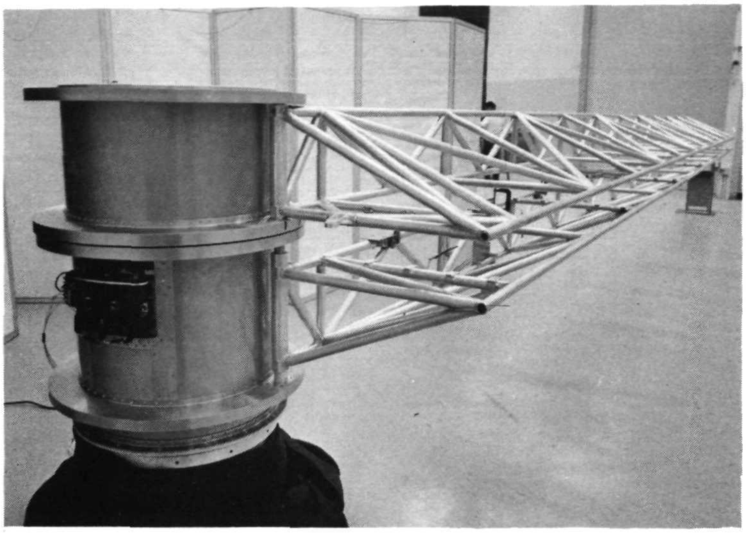
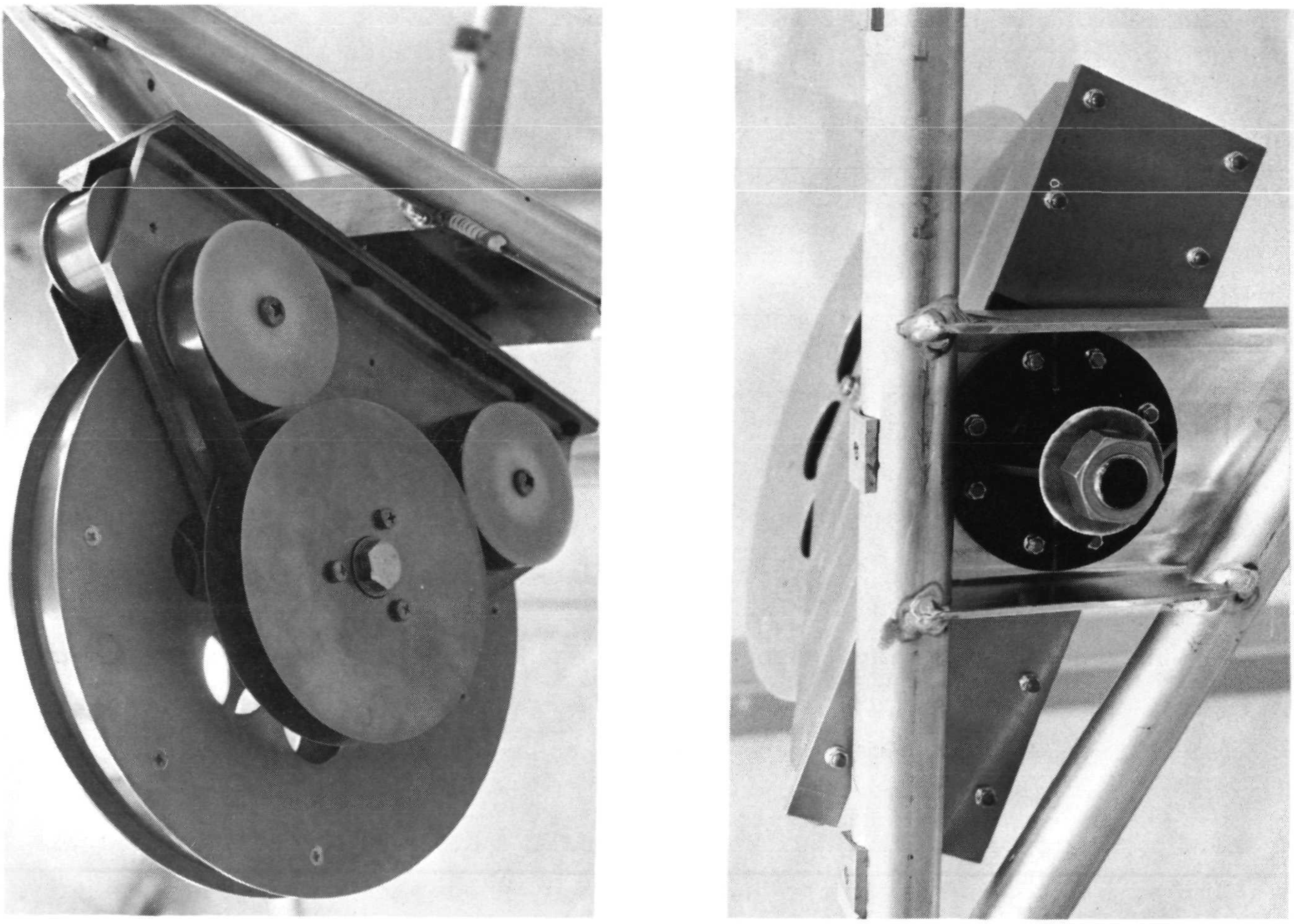


Figure 3.3-2 ETB Cap-Adapter/ISA-OSA Pivot Pin Fit Check

Figure 3.3-3 Same Assembly - Deployed Two ETB Bays

The guy tape mechanism (Figure 3.3-4 (a) and (b)) is mounted at the outboard end of the inboard support assembly and runs to the tip of the deployable beam. This provides additional beam tip support which is required to take anticipated artificial "g" loads.



(a) (b)

Figure 3.3-4 Guy Tape Stowage Mechanism

At least two levels of tensioning are required for the two inboard solar array strips to accommodate the zero "g" and artificial "g" loads. This is accomplished by the tensioning mechanism (Figure 3.3-5) which acts as a passive tension device for the zero "g" mode when unpressurized. By pressurizing the bellows during station rotation the sheet tension can be multiplied by a factor of over 20. The cable is shown attached to the bottom of the solar array strip.

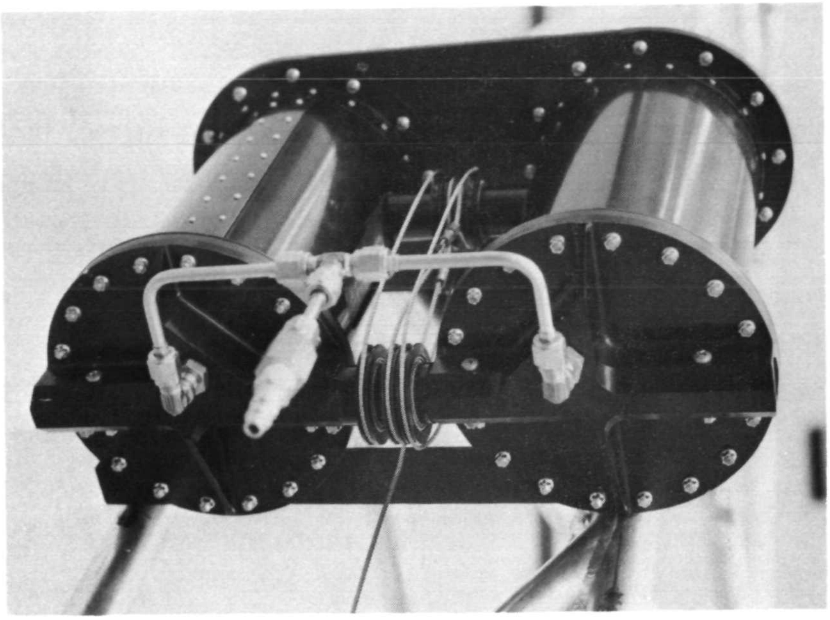


Figure 3.3-5 Artificial-g Tension Mechanism Mounted on ISA

A cross section of the array strip folded into its container assembly is shown in Figure 3.3-6. The cover end base plates are 5/32 thick aluminum honeycomb with 0.020 in. face sheets. These pallets are the main structural support for the stowed strip assemblies. The sides of the container are 0.050 in. aluminum sheet. There are six preload screws for each container which can be adjusted by torque wrench to the correct value. Cushioning pads between layers of the cell assembly are two one mil thick embossed Kapton sheets. When assembled the pad is 0.15 in. thick.

The packaging container, Figure 3.3-7 (a) and (b), stows the 6 ft x 90 ft array strip. The base plate shown in the previous figure can be seen at the bottom of the assembly. The negator springs at the bottom of the package provide the spring force for guide wires which deploy with the outer support assembly as guides for array strip deployment and retraction.

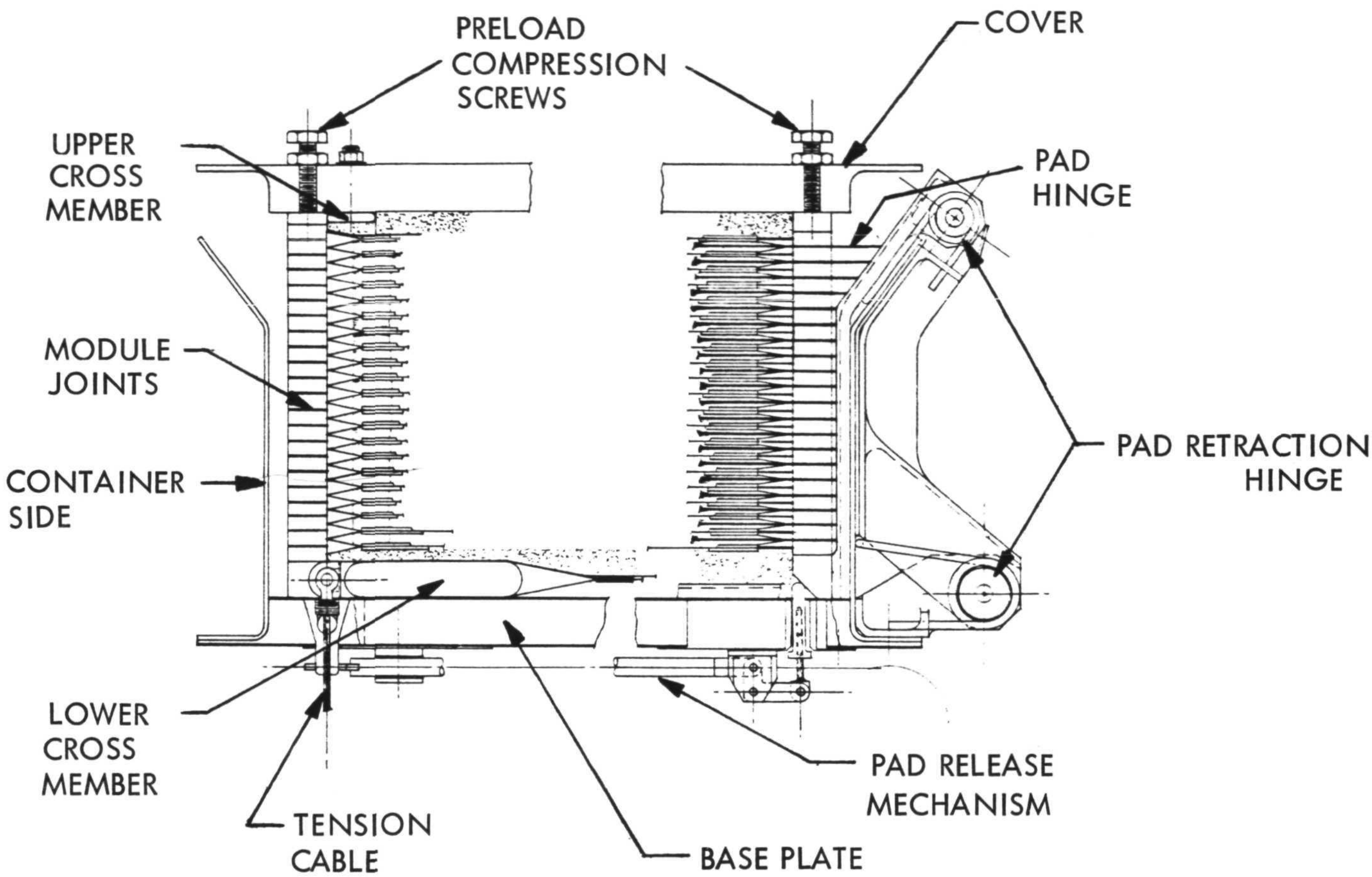
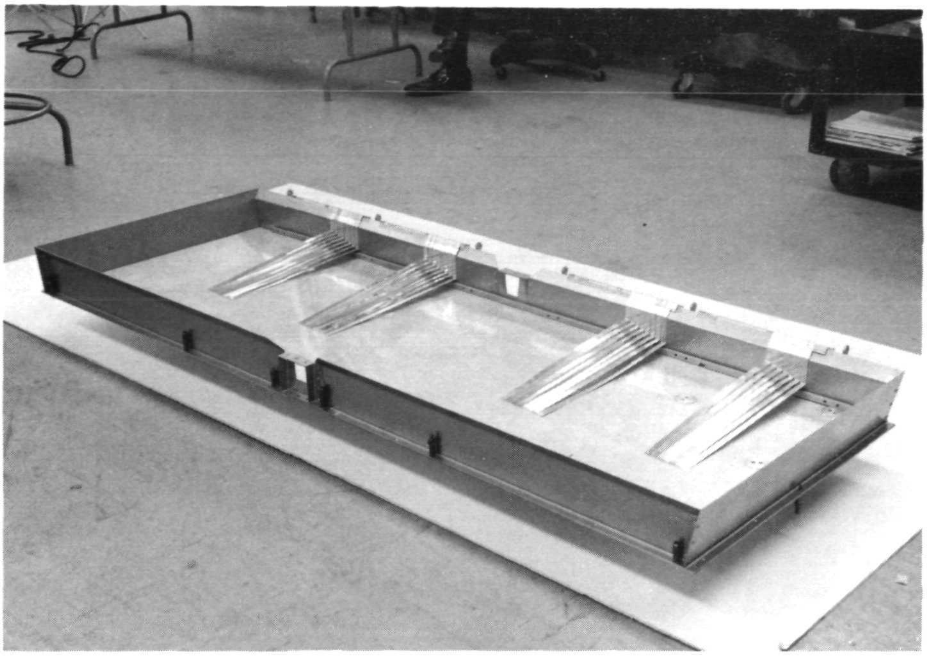
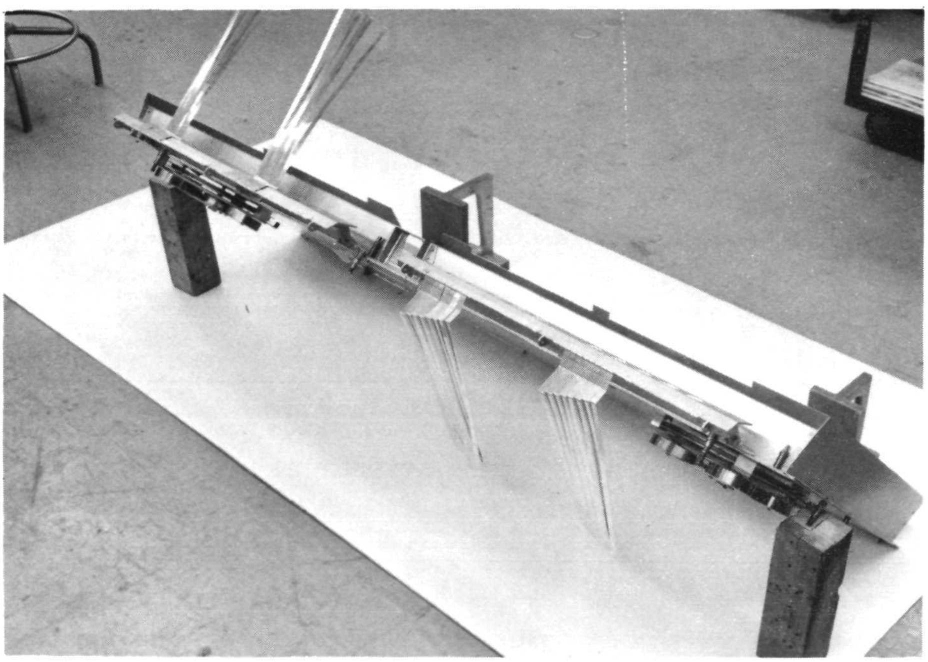


Figure 3.3-6 Strip Assembly Package - Cross Section



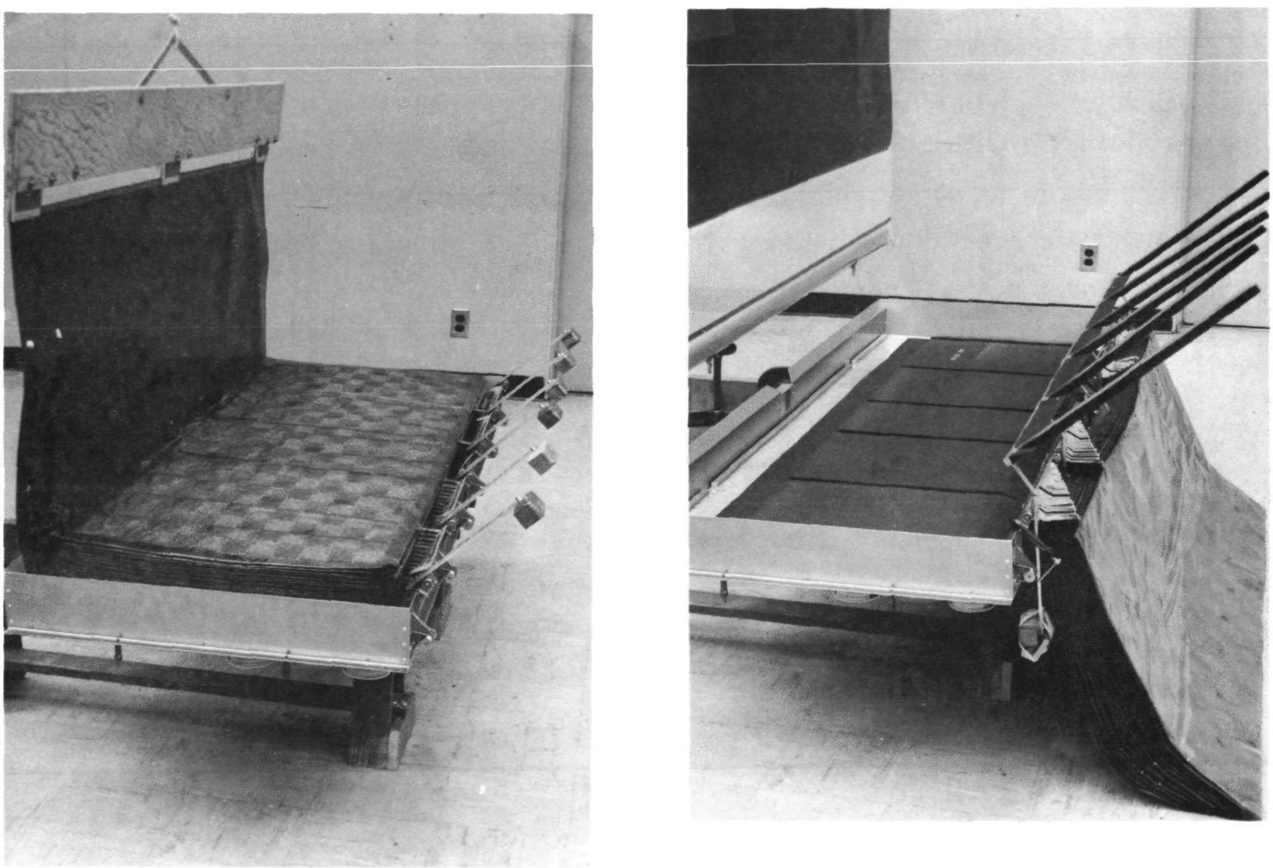
(a)



(b)

Figure 3.3-7 Packaging Container

Figure 3.3-8 shows the same array strip packaging unit as finally configured. The padding is embossed Kapton employed to protect the cells during launch. The layers of padding flip out of the container as each strip module deploys. The array strip can be retracted back into the retainment box with indexing of folds provided by guy wires. The container counterweights shown were used only as a test aid for removal of padding strips in a one-g test environment.



(a) Packaging Setup Before Release (with counterbalance)

(b) Package Setup After Pad Removal

Figure 3.3-8 Array Strip Packaging Assembly

3.3.1.2 Substrate-Cell Assemblies

The baseline flexible solar array strip (5 per quadrant) each 84 ft. long by 6 ft. wide is composed of 42 individual modules approximately 2 ft. long by 6 ft. wide. These are mechanically joined at the six foot edges by extruded-grooved aluminum bars with flat locking rods. As an aid in following the fabrication process, Figure 3.3-9 shows the assembly sequence of the flexible foldout panels. A 2 ft. x 6 ft. "F" film sheet is laminated to a 1.28 mil copper sheet after which the interconnect circuits are etched using a riston process. A second "F" film coverlay with holes punched at soldering locations is then bonded to this assembly. The circuit is then solder beaded, shaved, and inspection holes are drilled in the solder beads to provide inspectability for fillets. Seven 2 x 4 cm cells are soldered at one time to these interconnects using induction soldering techniques. The completed panels are then mechanically connected using an extruded metal jointso that each module can be easily removed by de-bonding a panel bus connection only.

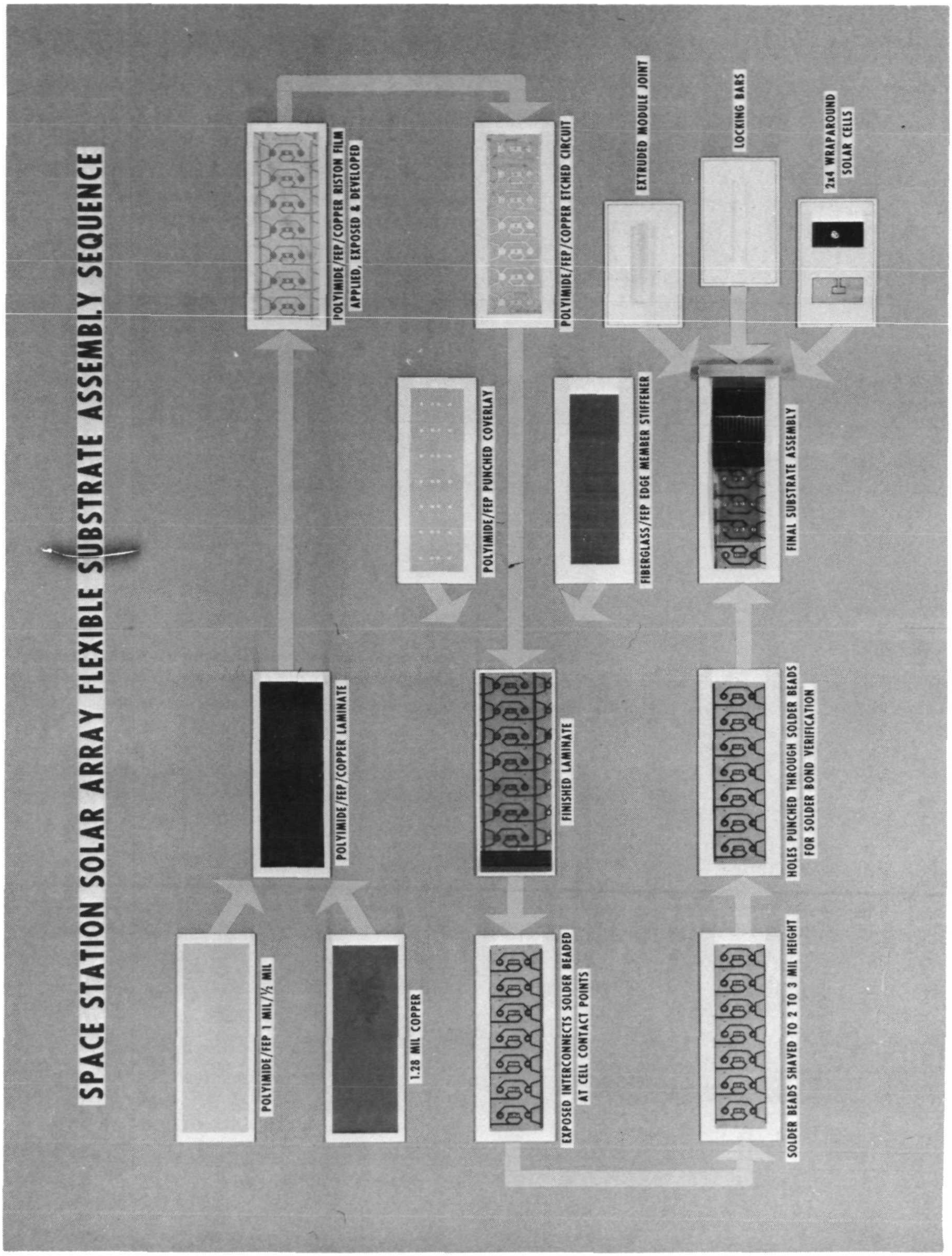


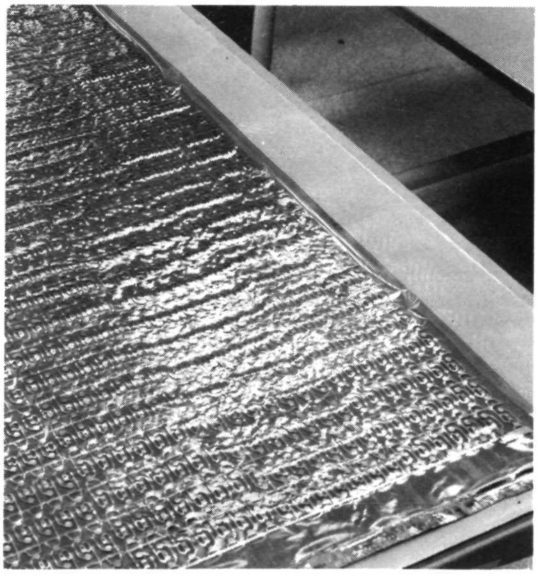
Figure 3.3-9 Solar Array Substrate Module Assembly Sequence

Figures 3.3-10 (a, b, c and d) show some of the equipment used in flexible printed circuit panel production. The space station interconnect pattern is shown on the substrate in the lower right illustration.

(a) Kapton-FEP Overlay on Copper Foil

(b) Developer Machine

(c) Etchant Machine



(d) Partial Substrate Laminate

Figure 3.3-10 Solar Cell-Substrate Fabrication Equipment

Subcontracts were awarded on 10 June 1971 to the two solar cell suppliers--Centralab and Heliotek--to develop the 2 x 4 cm 12 mil wraparound solar cell selected for this program. The two cell design approaches are shown in Figure 3.3-11. Hundreds of cells, meeting specifications, were received from both companies by closure of contracts in March 1972.

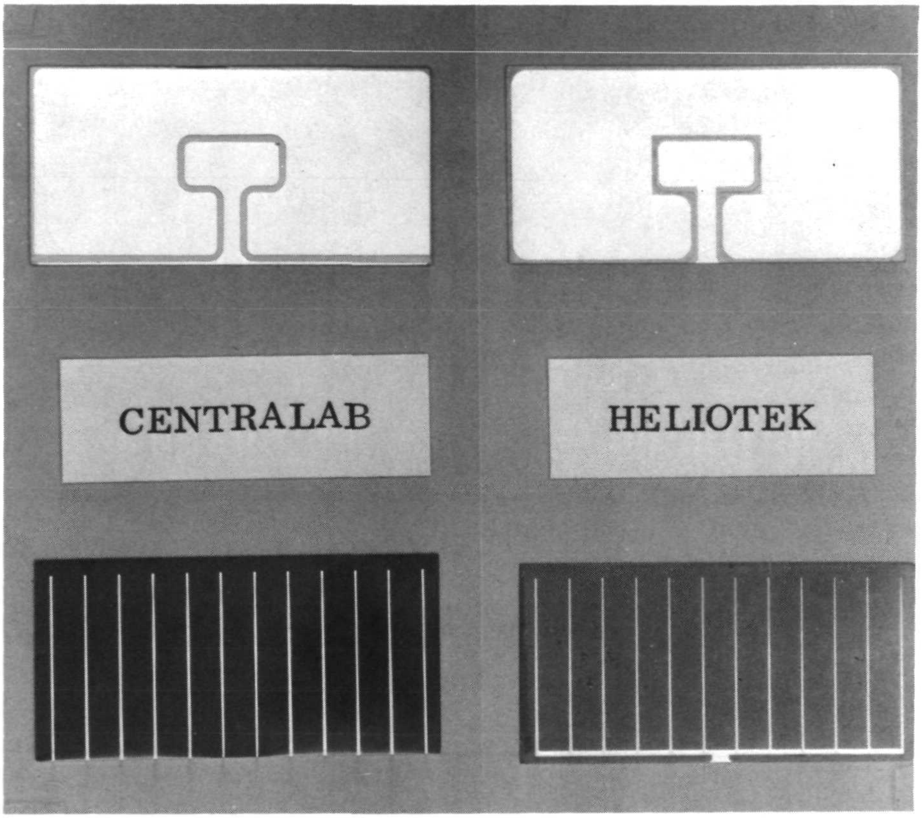
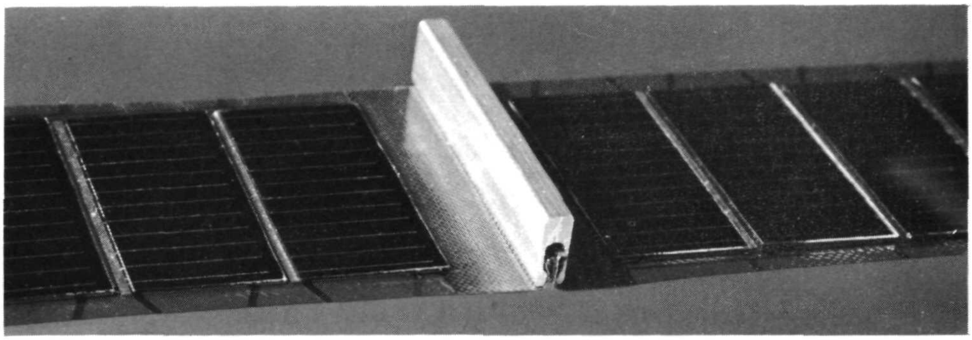
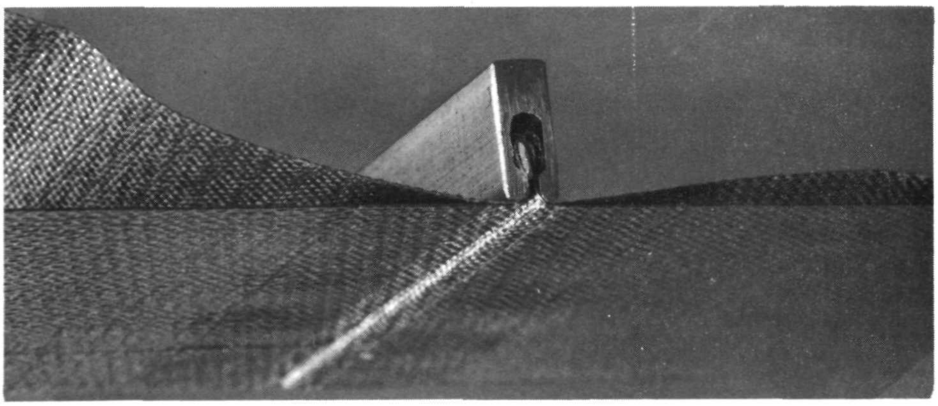


Figure 3.3-11 Solar Array Wraparound Cells

To take the high artificial "g" inertial and tensioning loads and at the same time provide modularity, a high strength extruded mechanical panel joint was selected as shown in Figure 3.3-12. The retainer slides over the locking bar contained inside the fiberglass end strip (bonded to the Kapton substrate) which provides added tear strength. This fiberglass strip is laminated into the panel as an integral part of the panel assembly sequence.



(a) Solar Cell Specimens with Extruded Joints



(b) Fiberglass Edge Specimens with Extruded Joints

Figure 3.3-12 Panel-to-Panel Joint

In addition to the mass simulated strip, four other strips are required for quadrant deployment to demonstrate the sequential deployment ability of the array and to provide a means of tensioning the quadrant. Mylar strips were employed for the purpose. These strips are joined together with riveted joints and assembled (Figure 3.3-13).

Packaging base plate and compression cover, prior to acquiring drill holes and fittings, are also shown in the figure.

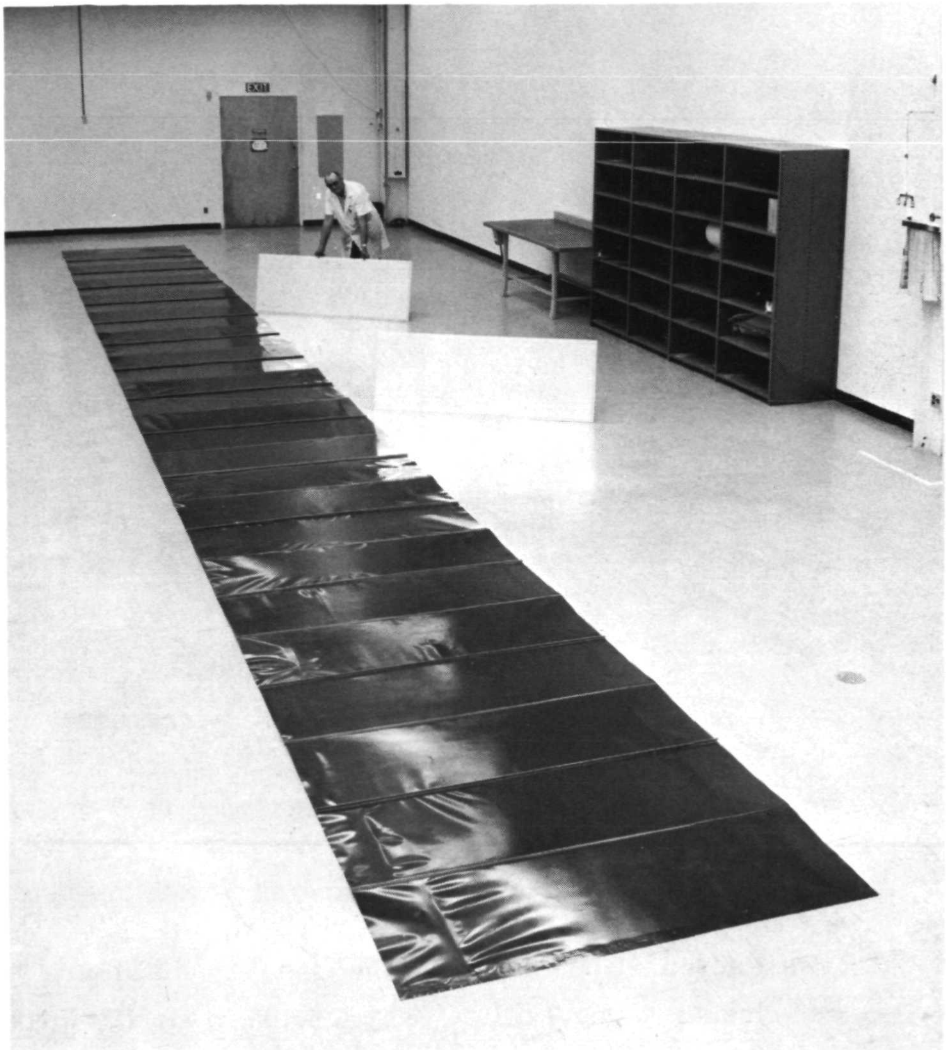


Figure 3.3-13 Array Strip Mylar Mockup

The first completed printed circuit substrate module with 2 x 4 wraparound cells is shown in Figure 3.3-14. This module employed early development cells from both vendors. Many of these early cells were below the electrical specification but were mechanically sound. This panel fabrication provided an opportunity to test and modify soldering and fixturing techniques for use in later electrical modules.

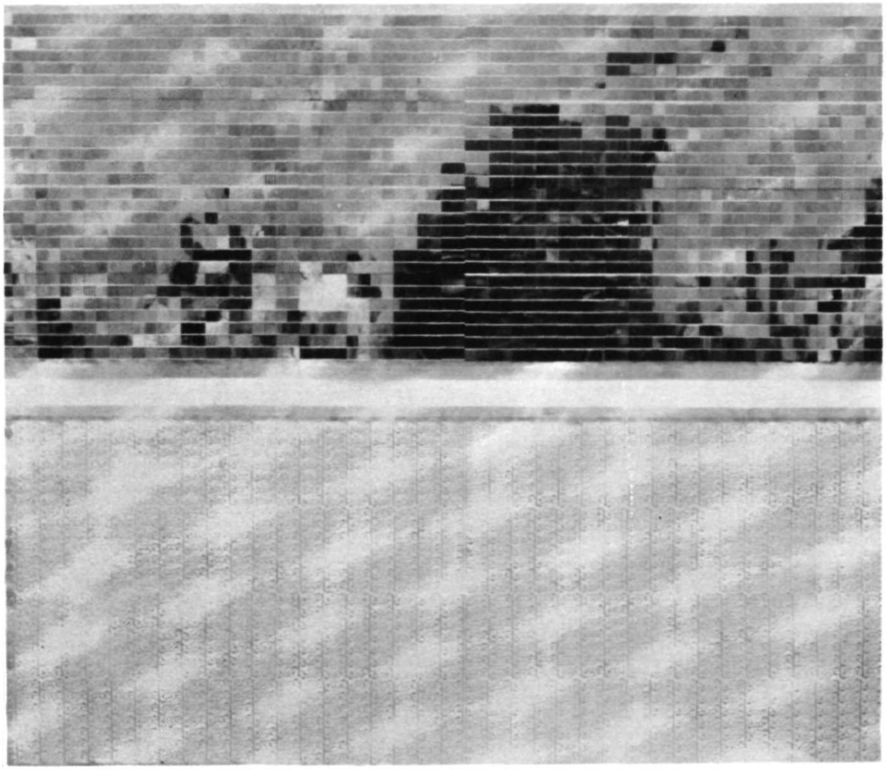


Figure 3.3-14 Solar Cell (Mechanical) Substrate Module

Figure 3.3-15 shows a seven module portion of the mass simulated strip (20 mil glass chips, painted blue, in place of cells). The final strip had 42 of these modules packaged in a 5" thick stack which deploy to a 6 ft x 84 ft array strip. Five of these strips form an array quadrant.

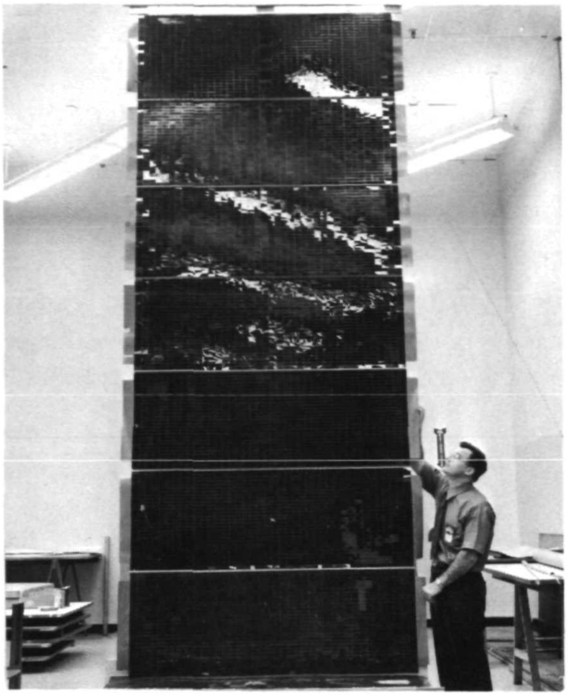


Figure 3.3-15 Seven Module Strip

3.3.1.3 ODAPT (Orientation Drive and Power Transfer)

Ball Brothers Research Corporation (BBRC) of Boulder, Colorado was a team member with Lockheed in the proposal response for this contract. In this role, BBRC was responsible for the design, fabrication and test of the ODAPT system. The BBRC contract extended from July 1970 through March 1972. Monitoring of contract schedules, reports, and budgets, along with technical coordination of this portion of the contract, were carried out by Lockheed as prime contractor. The major hardware elements produced by BBRC and delivered to LMSC after test are described in this section.

The complete ODAPT assembly with the exception of the cover to the outer gimbal drive is shown in Figure 3.3-16. The base mounting plate simulates the vehicle attachment. The inside cylinder forms the extension to the power boom and is fixed to the Space Station during tracking operations with the outer cylinder rotating around the inner cylinder. The other axis of tracking is provided by the smaller diameter outer gimbal. The solar array deployment mechanism attached to the outer gimbal bracket.

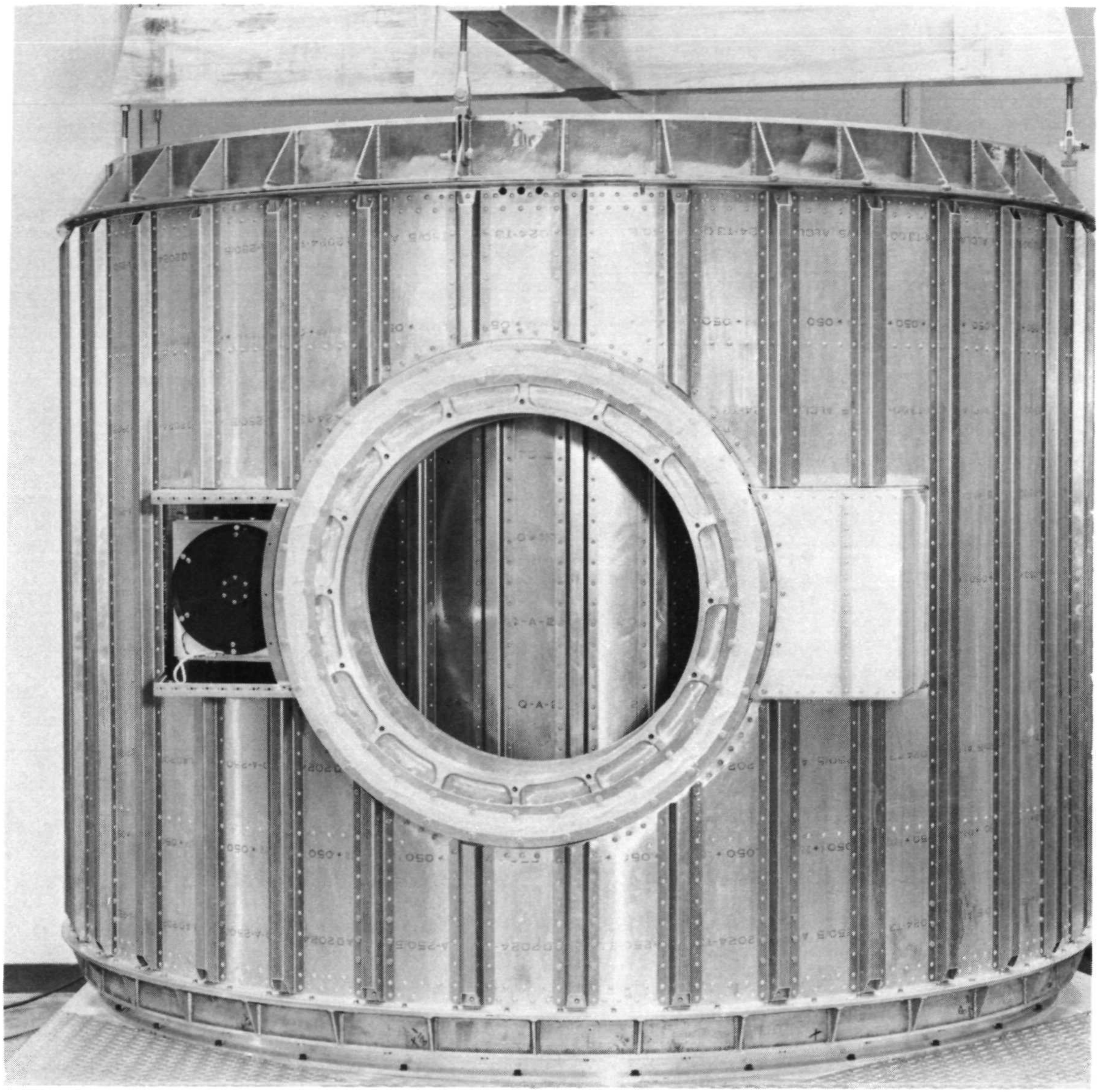


Figure 3.3-16 Final ODAPT Assembly

Technicians are installing (Figure 3.3-17) the inner gimbal bearing duplex pair retainer. This hat section of the inner cylinder is mounted to the remainder of the inner gimbal cylinder shown in Figure 3.3-18. The 76" bearings can be seen inside the retainer.

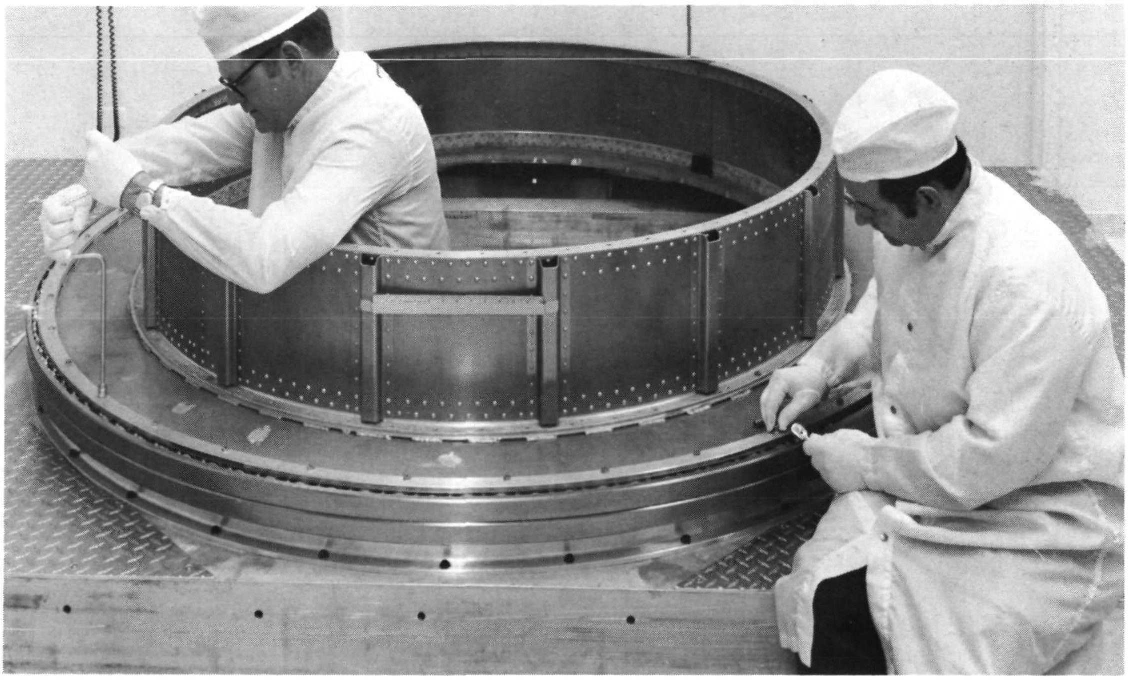


Figure 3.3-17 Inner Gimbal Bearing Installation

Figure 3.3-18 Inner Gimbal Cylinder

The assembly in Figure 3.3-19 shows the hat section mounted to the inner gimbal cylinder with the outer bearing races now contained by a structural ring at the bottom. The outer cylinder is attached to this ring. The ring immediately below provides the structural interface between the ODAPT and the Space Station power boom.

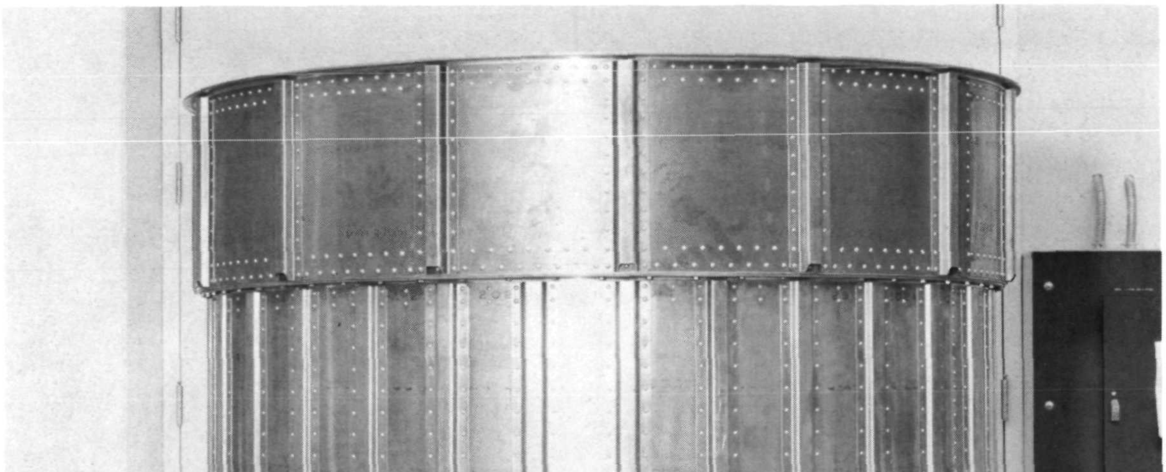


Figure 3.3-19 Inner Gimbal Structural Ring

The inner gimbal cylinder (Figure 3.3-20) includes outer gimbal ports and the outer gimbal drive motor housing at the top. At the forward end can be seen the two slip ring assemblies which are employed in the test to conduct power to the outer gimbal drive motors. Although a complete set of flight type slip rings is not included in this assembly, their mass was simulated in this test unit.

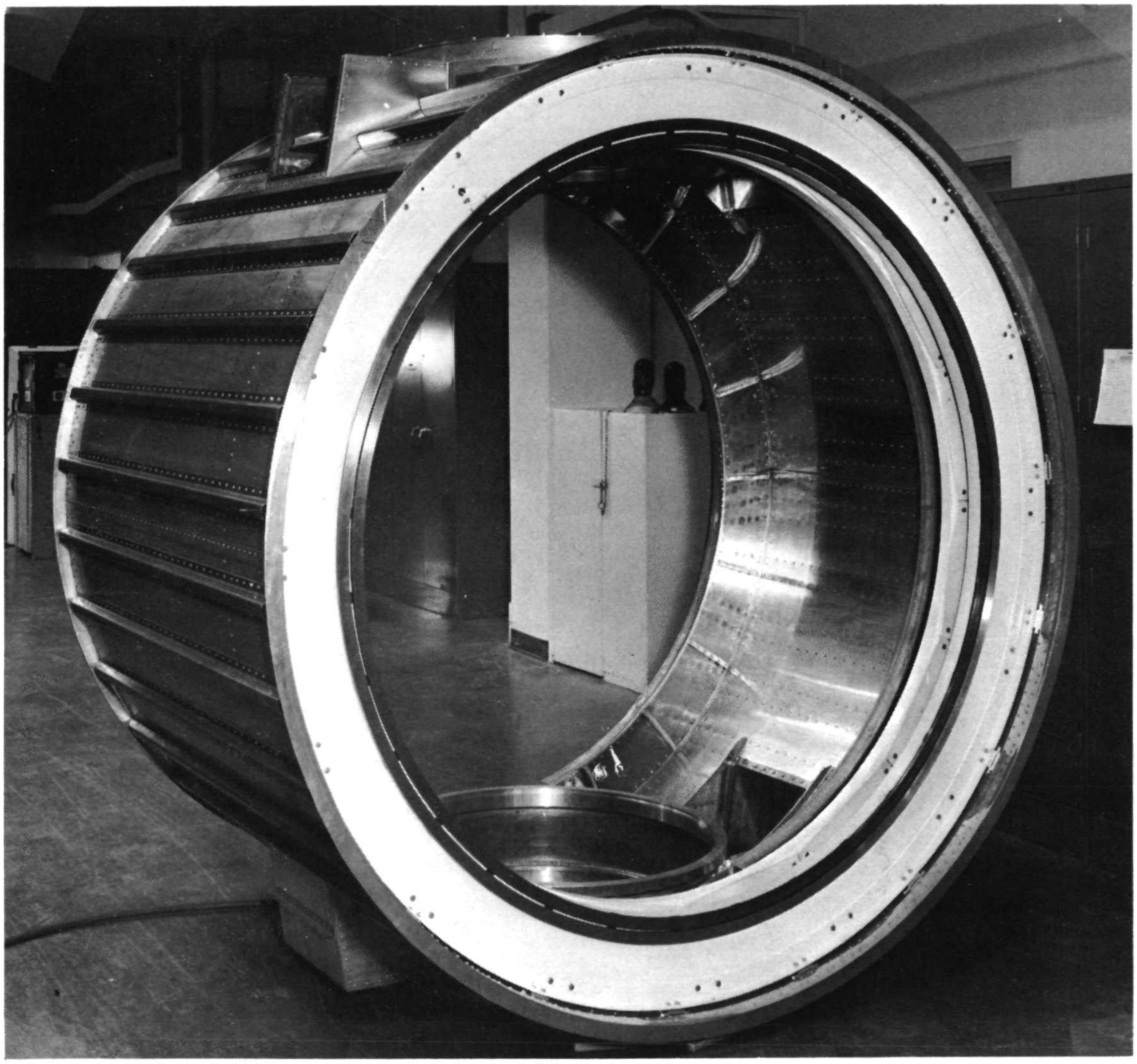


Figure 3.3-20 Inner Gimbal Cylinder

The outer gimbal motor drive is being installed in Figure 3.3-21. The drive pinion and bull gear shown are used for the outer gimbal drive only.

Figure 3.3-21 Outer Gimbal Motor/Driver

Figure 3.3-22 shows the inner cylinder with the 100:1 harmonic drive gear motor installed. This motor drives the inner gimbal. The drive motor assembly can be removed as a unit by the astronaut from inside the inner cylinder for servicing and replacement. The astronaut access port to the slip ring brush assemblies is also shown at the bottom.

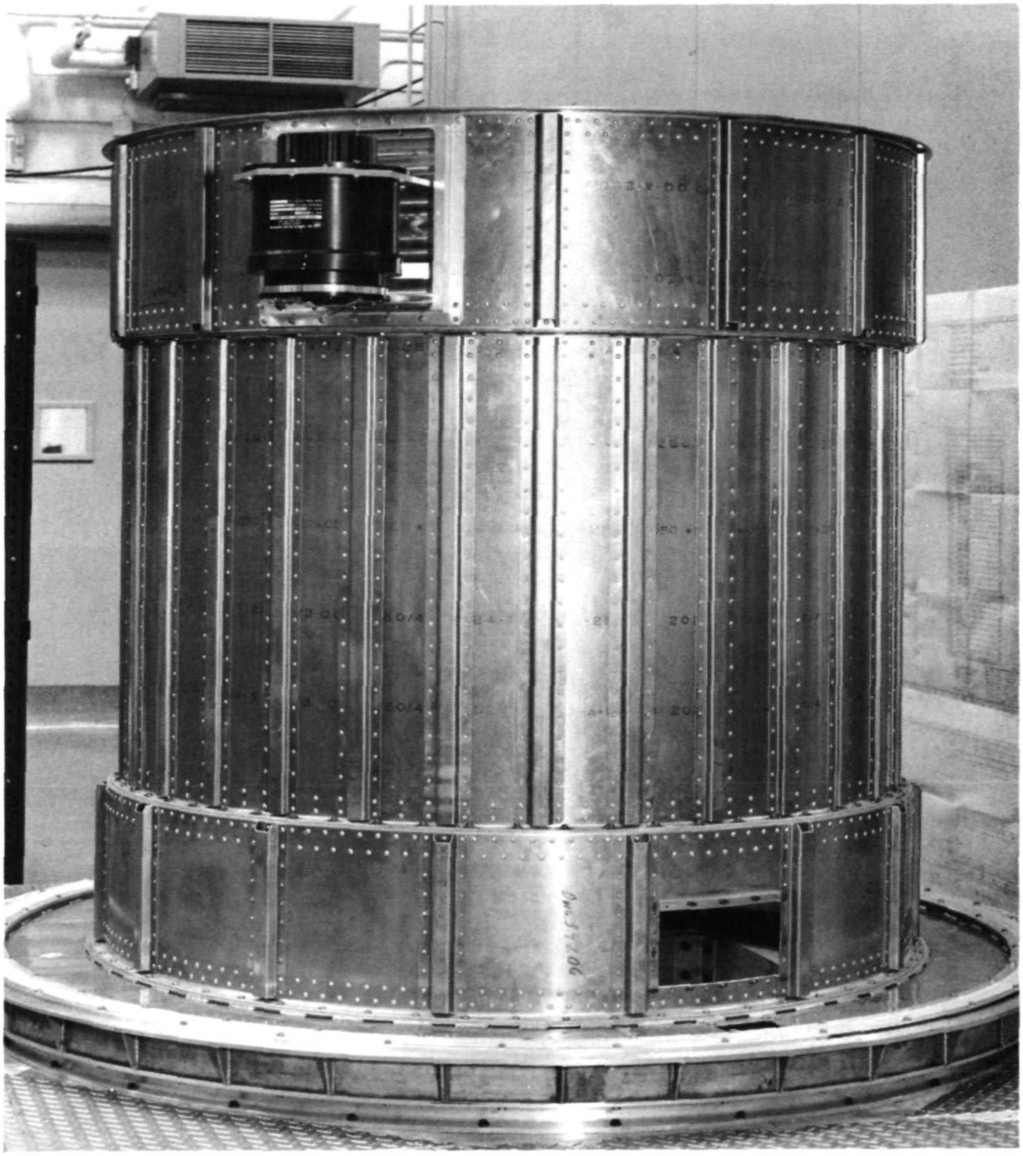


Figure 3.3-22 Inner Gimbal Motor/Driver Installation

Figure 3.3-23 shows the final assembly of the inner cylinder being lowered into the outer cylinder. The large bull gear which drives the outer gimbal is visible at the top of the assembly.

Figure 3.3-23 Installation - Inner Cylinder/Outer Cylinder

3.3.2 Test Program

Three categories of tests were identified as necessary for successful completion of the program.

- Design Support Tests
- Major Hardware Tests
- System Level Tests

The Test Evaluation Plan document (LMSC/A984133), submitted and approved in February 1971, contains a description of test objectives and requirements for these categories. Budget constraints and changes in priorities modified and deleted some tests later in the contract. All of the testing conducted and the test results are completely reported in the 3rd Topical Report, Design Support and Major Hardware Tests, LMSC/D153526, released in November 1972. Only some highlights of the lengthy test program, in condensed form, are presented here.

3.3.2.1 Design Support Tests

Engineering design decisions and baseline component selections were mostly based on laboratory and bench type tests (small scale as required), since little or no information for large (10,000 ft^2 area) flexible arrays existed. In the structural members and mechanisms area all three members of the contract team conducted numerous tests related to load capability, deployment, retraction, packaging constraints, and endurance (10 year life time in orbit). Some component materials such as lubricants, packaging protective padding, substrate laminates, ETB canister rollers, and ODAPT slip rings and bearings, were selected on the basis of comparative tests.

In the structural mechanisms area of effort many simple, small scale tests were conducted concurrent with the design activities. One such test (Figure 3.3-24), as an example, was the determination of the slip resistance under various compressive loadings between the protective padding and the solar cell assemblies in the packaged condition. These data were used to select compressive loads for the array launch packaging system.

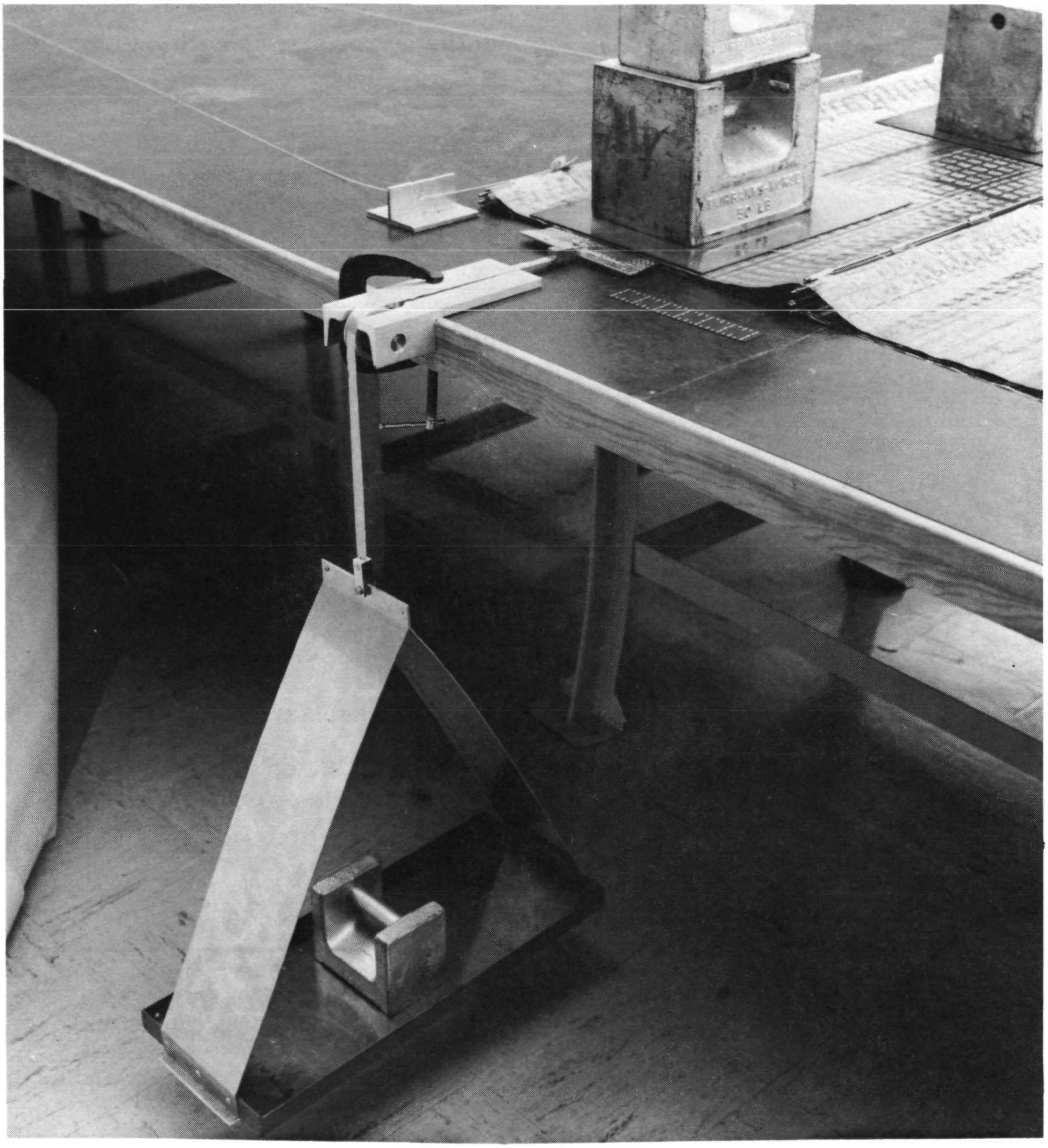


Figure 3.3-24 Cell-to-Protective Padding Slip Resistance Measurement

As an example of mechanism tests Figure 3.3-25 shows the test setup for the artificial "g" variable tensioning mechanism. This test consisted of pressurizing the artificial "g" mechanism with the gauges shown simulating array strip position to test load distribution capabilities of the tensioning system.

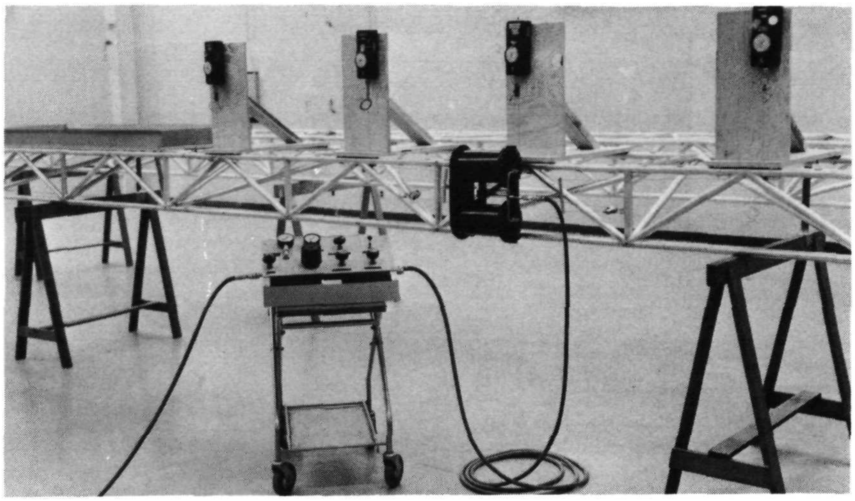


Figure 3.3-25 Artificial-g Tension Mechanism Checkout Setup

In the substrate/cell assembly three important test series were conducted:

(1) Substrate Test and Tensile Tests
(2) Substrate Creep Tests
(3) Steady State Temperature Tests

These tests had a general objective to gather information for selection of materials to design and fabricate the baseline array strip substrate/cell modules and assemblies.

During the technology evaluation phase it was found that little tensile test data existed on flexible laminates such as those proposed for this program, particularly at the high loads anticipated for artificial "g". Table 3.3-1 summarizes the tensile testing conducted under this program.

TABLE 3.3-1

SOLAR CELL-SUBSTRATE TENSILE AND TEAR TEST SUMMARY

Preliminary Test Program (Room Temperature)

- o Tensile - 103 Specimens, 17 Combinations of Materials, Laminations and Joints

- o Tear - 25 Specimens, 8 Combinations of Materials and Laminations

Final Tests

- o Tensile - 111 Specimens, 12 Combinations of Materials and Configurations Tested at $-80^{O}F$, $25^{O}F$, $+170^{O}F$

- o Tear - 23 Specimens, 3 Combinations of Materials and Laminations Tested at $75^{O}F$, $170^{O}F$

An Instron testing machine with its temperature chamber was employed for all tensile tests (Figure 3.3-26).

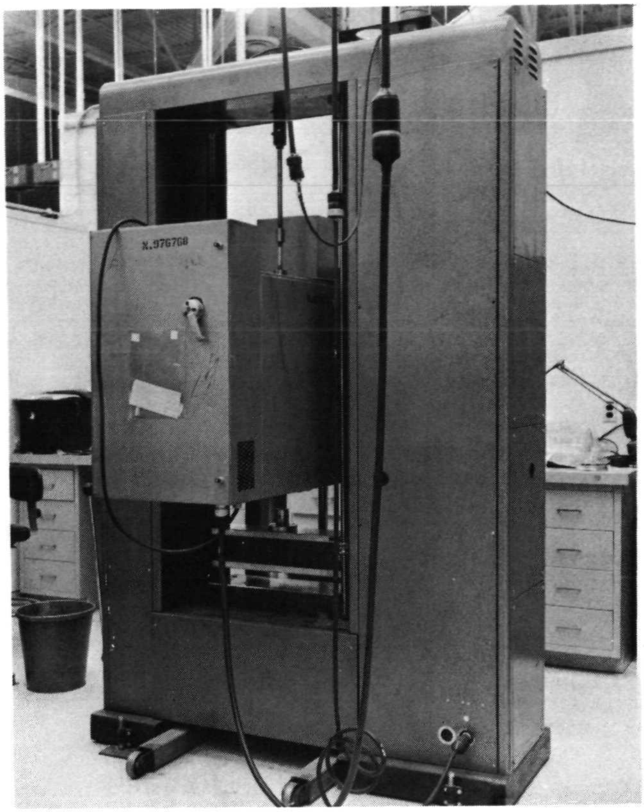


Figure 3.3-26 Instron Machine for Tensile/Tear Test

A typical set of specimens, as tested both with and without solar cells installed is displayed in Figure 3.3-27. All seven of these specimens were cut from the same production substrate to provide direct comparative data on the effect of the presence of cells.

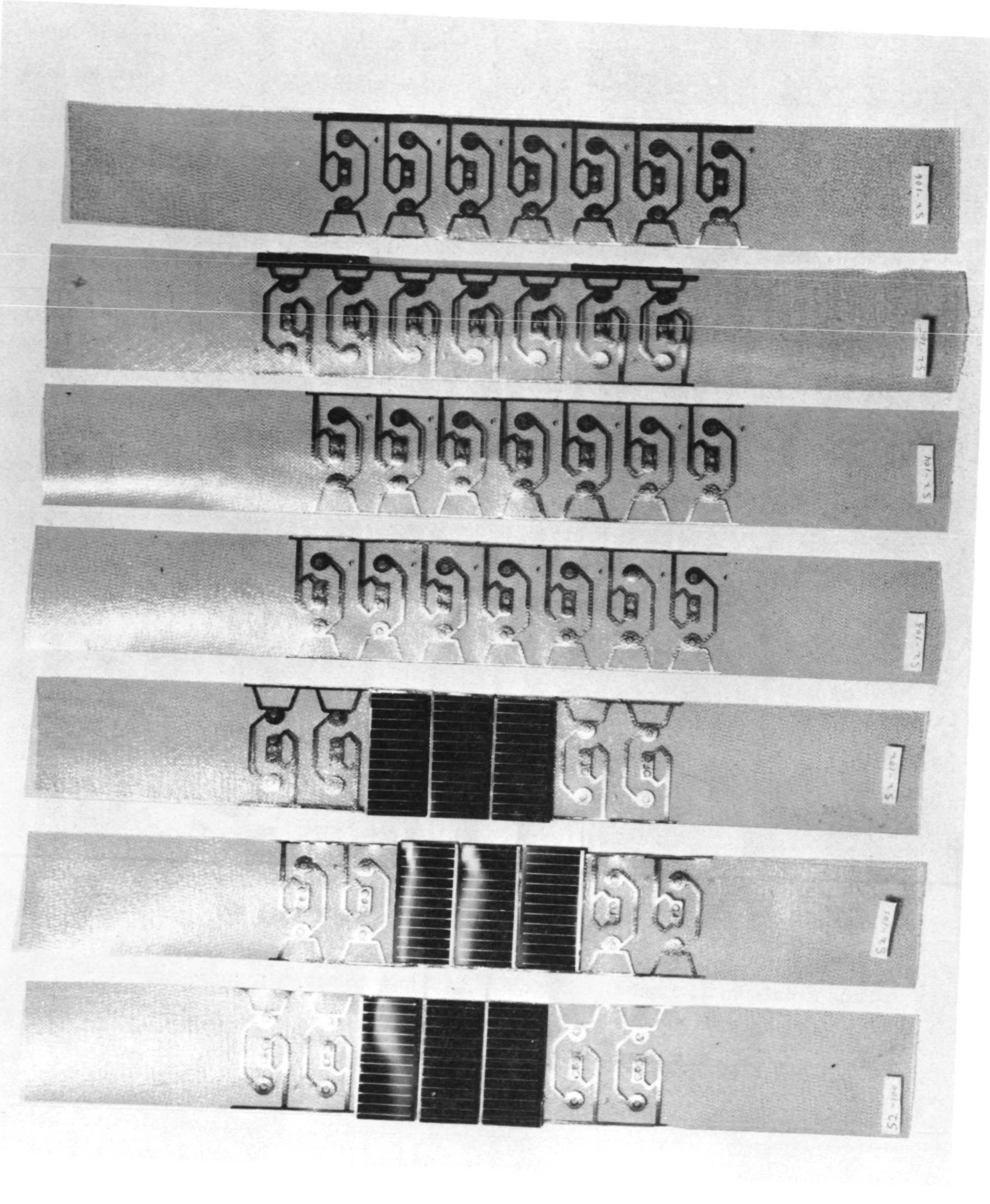
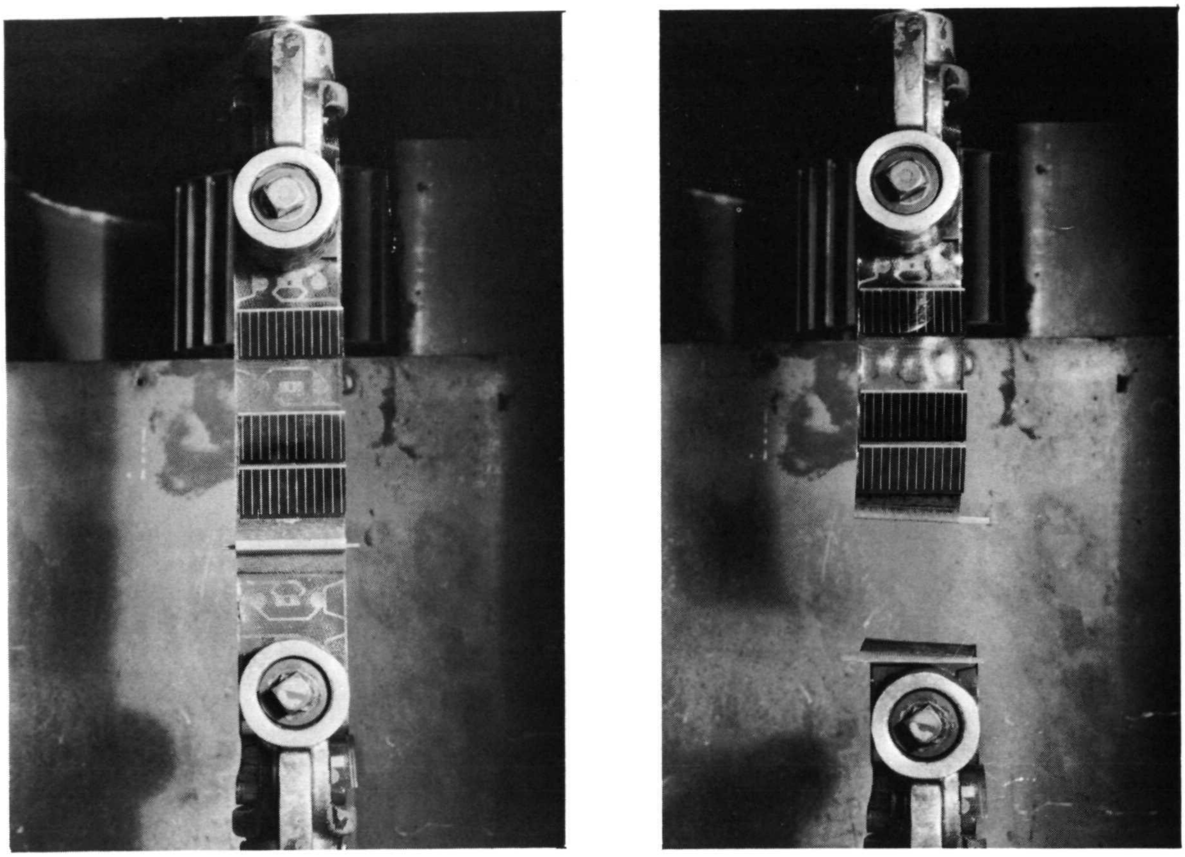


Figure 3.3-27 Substrate Specimens with Circuit

Assemblies with module to module joints were also tested and are shown in Figure 3.3-28 (a and b). Failures consistently occurred at the joint at approximately 30 lbs/in rather than in the cell assemblies. Since the maximum expected load is less than 10 lb/in these mechanical joints were deemed more than adequate for worst case space station loadings.



(a) Substrate with Circuit, Cells and Joint-Small Load

(b) Substrate with Circuit, Cells and Joint-Failed

Figure 3.3-28 Tensile Test Closeup

The creep test specimens were selected largely based on tensile test results plus other preliminary evaluations. The 14 lb/in load (Table 3.3-2) represents a factor of safety of 1.5 over maximum anticipated artificial "g" loads and a factor of safety of 300 for the long term zero "g" tension loads; the added built-in safety factor results from the constant orbital maximum test temperature of 140°F which is far in excess of the orbital average. Under these extreme conditions the total substrate creep for the selected system would be less than one inch over the 90' array strip span for a complete 10 year mission.

TABLE 3.3-2

SOLAR CELL-SUBSTRATE CREEP TEST SUMMARY

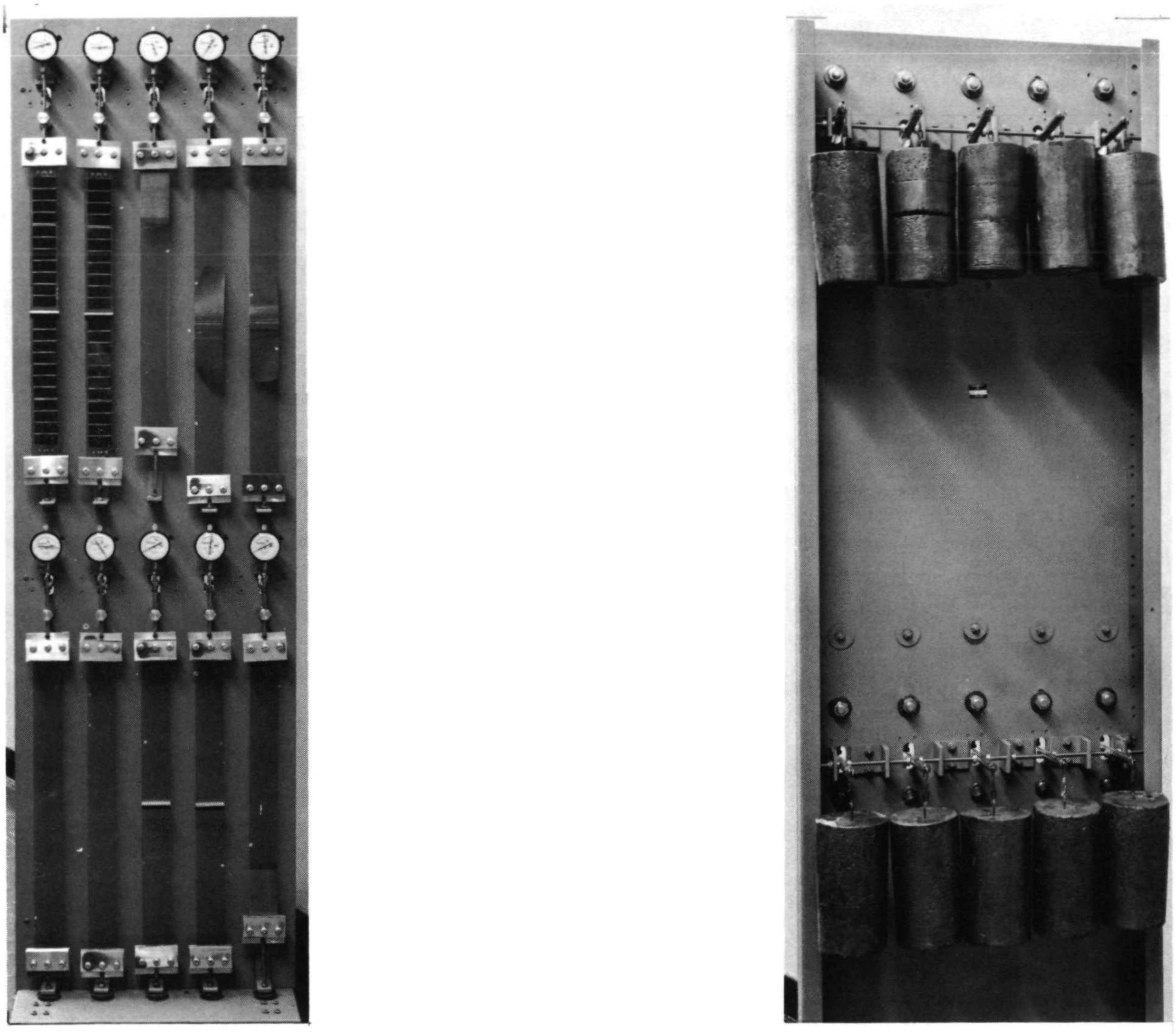
Preliminary Tests

- 10 Specimens
- 5 Materials, Laminations, and Joints
- 2 Loads (1/4 Lb. per Inch, 4 Lb. per Inch)
- 75°F, 140°F, 176°F

Final Tests

- 10 Specimens
- 5 Materials, Laminations, and Joints
- 14 Lb/Inch, 140°F

Figure 3.3-29 (a and b) shows the test setup employed for final material configuration selection. The 14 lb/in. loads were applied from the back of the panel, Figure 3.3-29 (b) and the entire test fixture was installed in the chamber as in the preliminary tests. No failures were experienced during this testing.



A. Front B. Back

Figure 3.3-29 Final Creep Test Setup

ODAPT Tests

One of the first design support tests performed for the ODAPT concept at Ball Brothers Research Corp. had as an objective the selection of lubricant characteristics suited to Space Station environments and a 10 year life time.

Prior to final lubricant selection two test programs were performed to augment the data gathered on candidate lubricants during the technology evaluation phase. Final design selection was based on these data.

Lubricant Tests

In order to select a brush material, a preliminary material evaluation was run using the test setup shown in Figure 3.3-30. Both static and 3 in/min oscillatory motion was applied at brush pressures between 4 and 16 psi and current densities of 50-100 amperes/in^2. The brush materials employed were silver/graphite, silver/niobium diselenide and silver/molybdenum disulfide/copper. The ring material used in this test was copper.

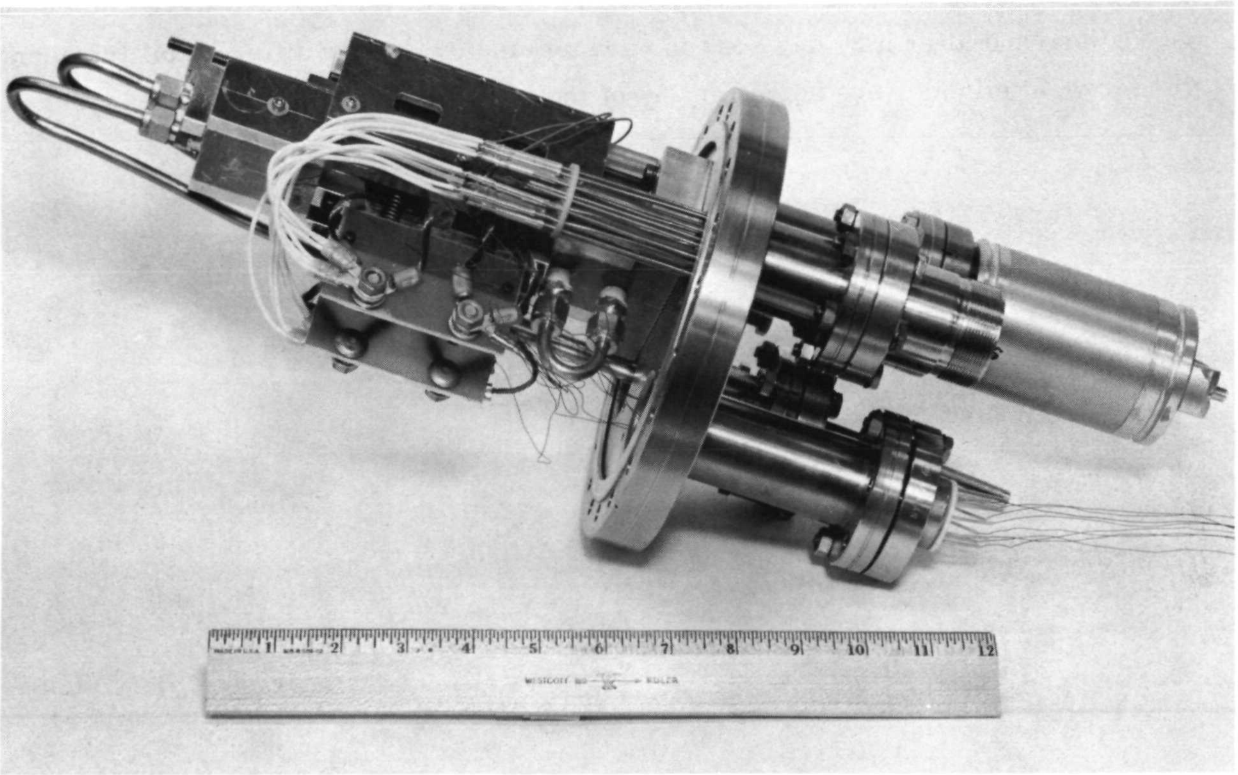


Figure 3.3-30 Brush/Slip Ring Test

Tests Performed

1. Weight Loss Rates of Oils - Vapor Pressure vs Temperature from 290°F to 370°F for Kryton 143-AB and Mobil XRM 217D.

2. Wear Tests of Candidate Gear and Bearing Lubricants
   - o 7 Oils and 6 Greases
   - o 100, 158, and 212^{o}F
   - o 45 Ft/Minute Sliding Speed
   - o 210,000 psi Initial Hz Stress

A bearing test was conducted to determine the effect on oil lubricated bearings during extended static periods in vacuum. The test setup, Figure 3.3-31, is capable of testing two lubricants simultaneously. Starting and running torques were determined both before and after a 60 day soak in vacuum employing four different oil lubricants. No cold weld effects were noted for any of the lubricants.

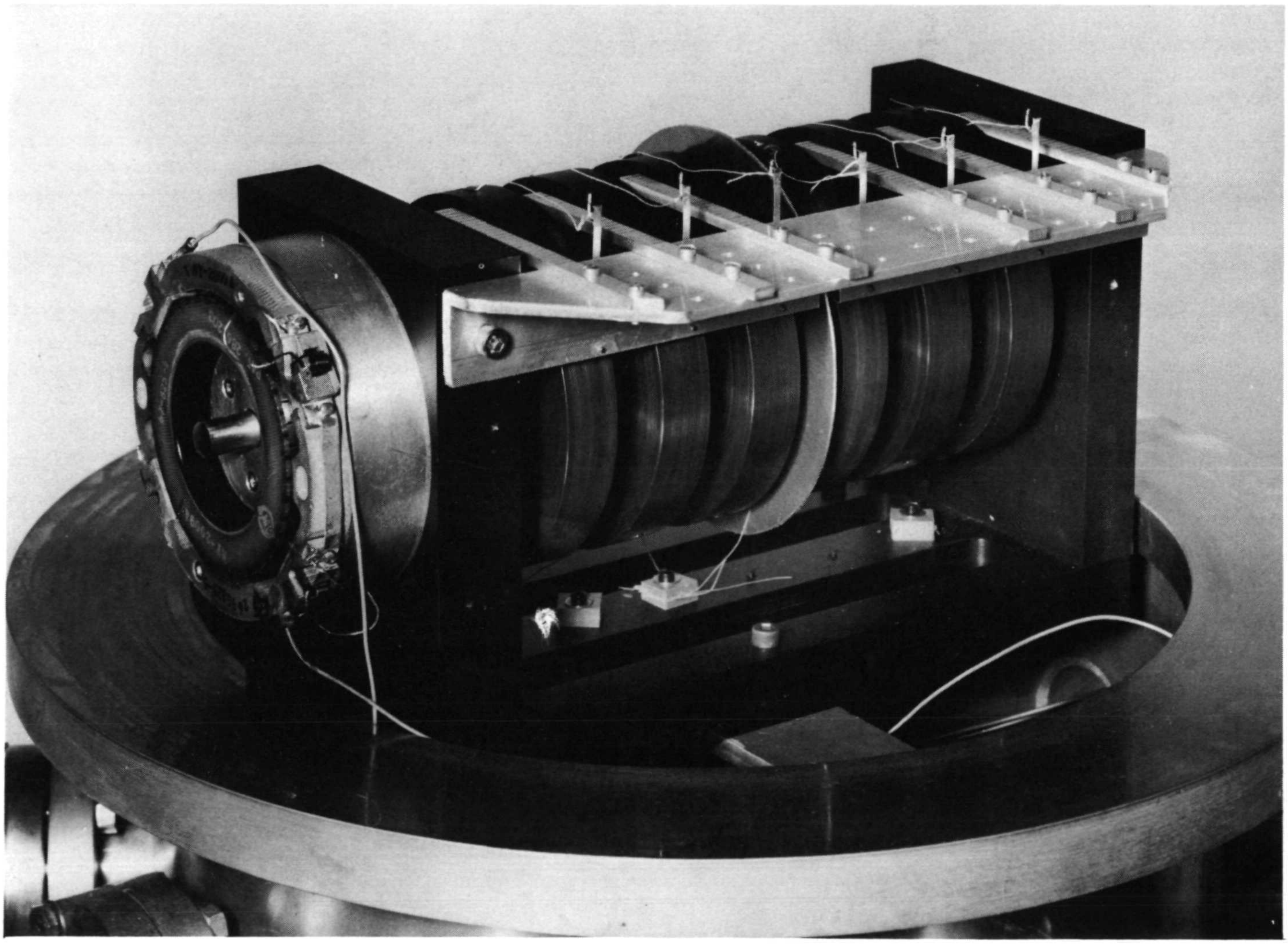


Figure 3.3-31 Bearing Lubricant Test Setup

A prototype slip ring and brush test fixture was fabricated as shown in Figure 3.3-32. Coin silver was selected for slip ring material with silver/niobium diselenide as brush material. No life testing has been accomplished on this unit to date.

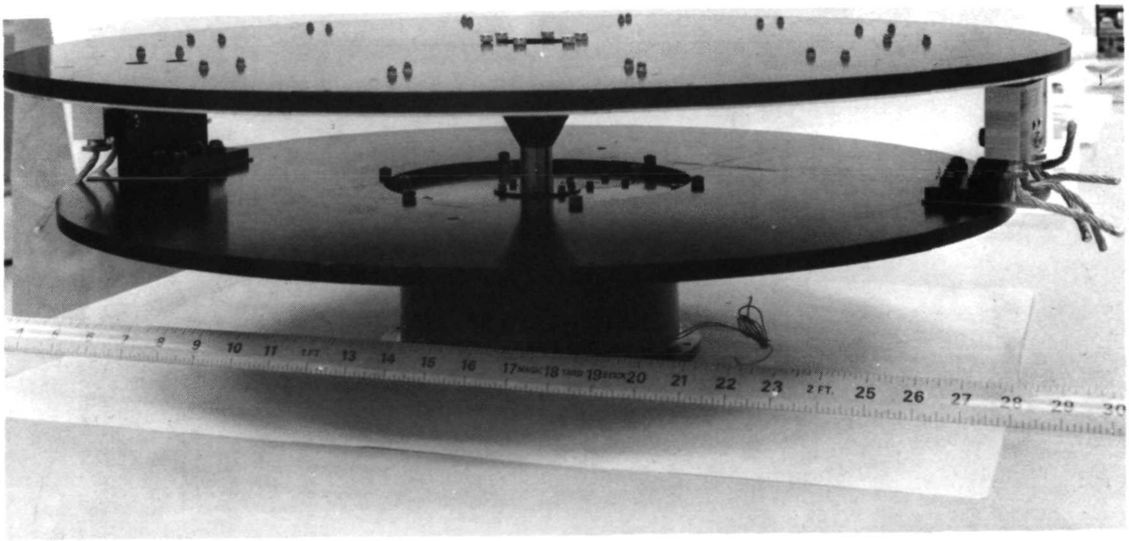


Figure 3.3-32 Prototype Slip Ring Test

3.3.2.2 Major Hardware Tests

The Astromast (ETB) is shown in Figure 3.3-33 at various stages of deployment during acceptance testing, successfully performed at LMSC in early July 1971. Dimensional measurements were taken after completion of the acceptance tests runs which established complete compliance with the Lockheed specification.

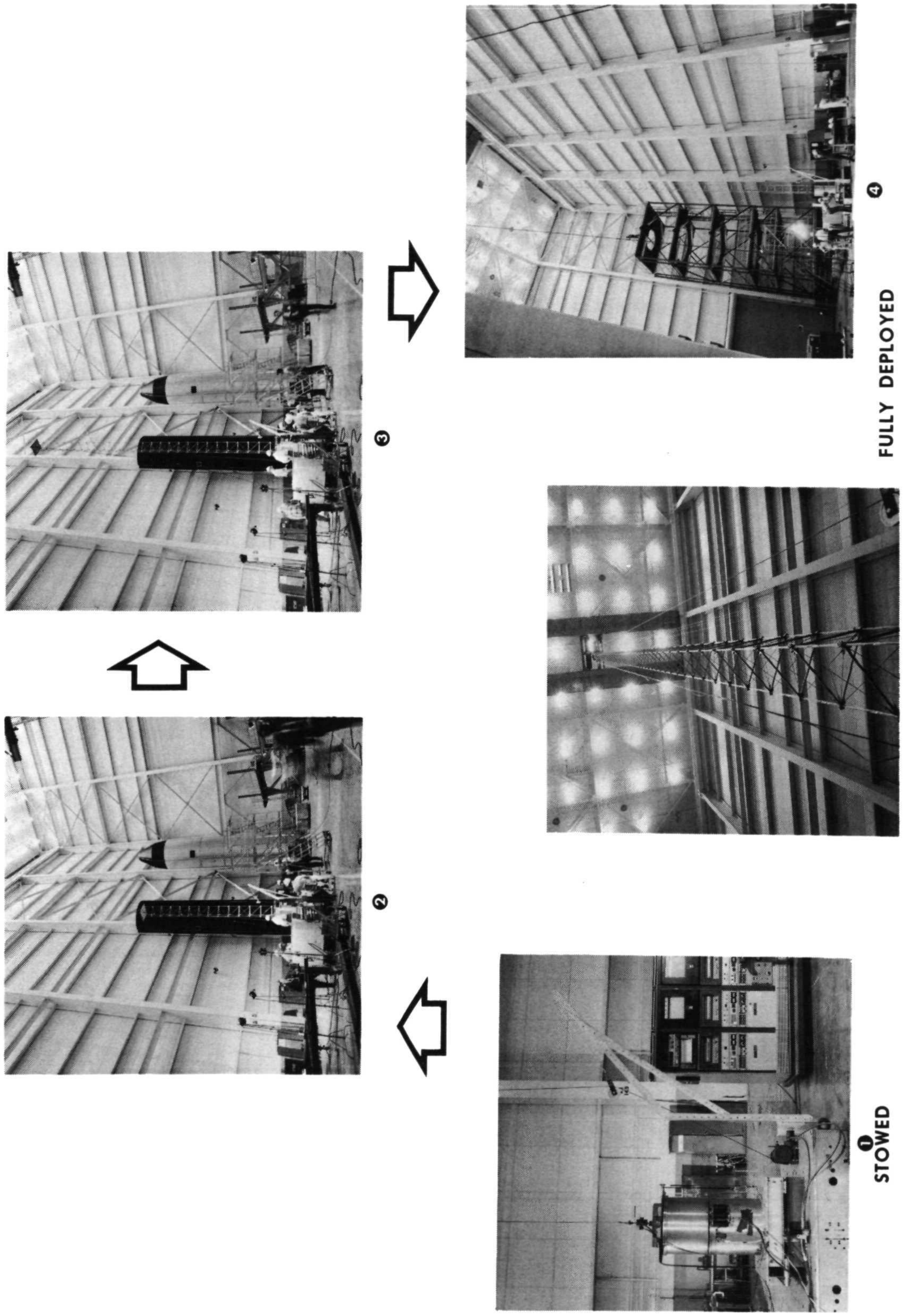


Figure 3.3-33 ETB Acceptance Test

The quadrant inboard and outboard support assemblies (ISA-OSA) in and from which the array strips are stored and deployed were given a simple proof load test prior to final system assembly. For this test the beam tip cap was used as a test fixture with the ISA and OSA mounted to it (Figure 3.3-34). Various distributed loads were applied as fixed weights and differences in deflection were measured with respect to the floor. The warping of the end cap diameter which accounted for more than half the tip deflection was also measured.

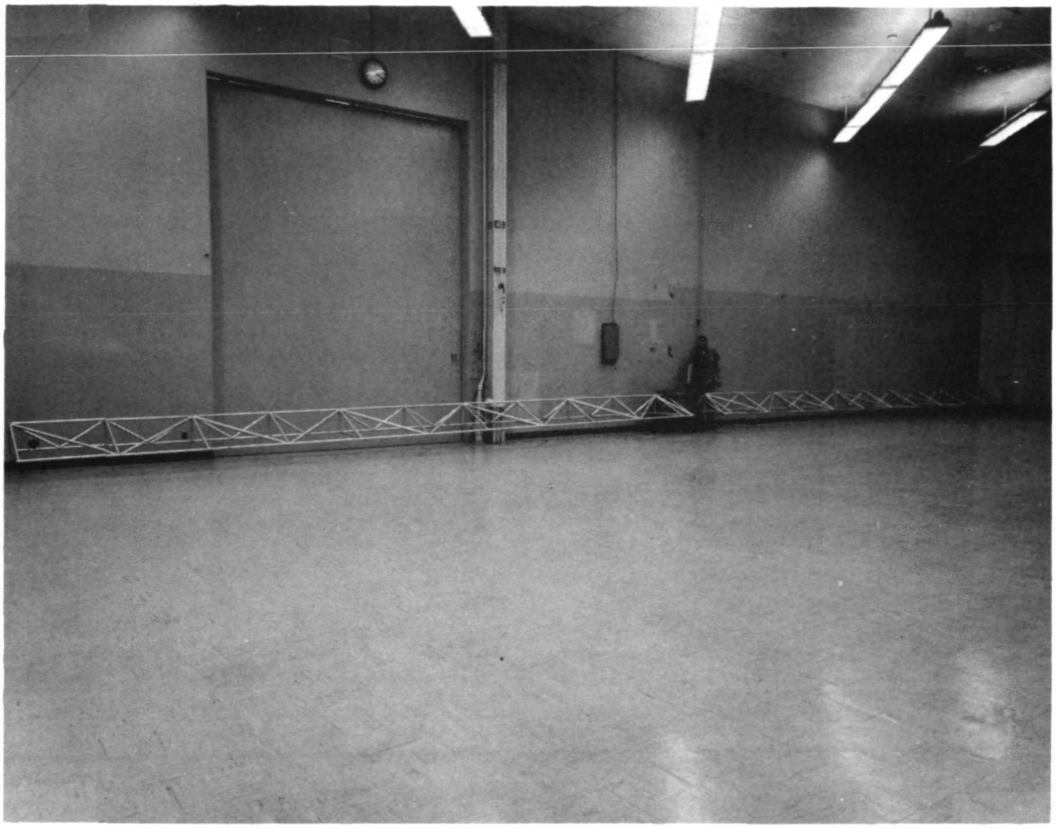


Figure 3.3-34 ISA/OSA Static Load Test Setup

The Astromast (ETB) Load Evaluation Tests took place in October-November 1971 as follows:

(1) The first tests run on the ETB were to check vertical alignment accuracy during repeated extensions and retractions. The transits shown on the load platform were used to determine these alignments (Figure 3.3-35).

The large structure surrounding the beam was used as a datum for later deflection measurements. Access to the top of the beam was by ladder from outside the structure.

(a)

(b)

Figure 3.3-35 ETB Alignment Test

(2) A test series was performed to determine torsion and damping characteristics. The setup at the upper tip of the beam for each of these tests is shown in Figure 3.3-36. The accelerometers were used to indicate tip motion and rate of decay after an initial tip deflection. The bar shown in Figure 3.3-37 was used to apply torsional loads to the beam with deflections measured by deflectometers tied to the tip plate.

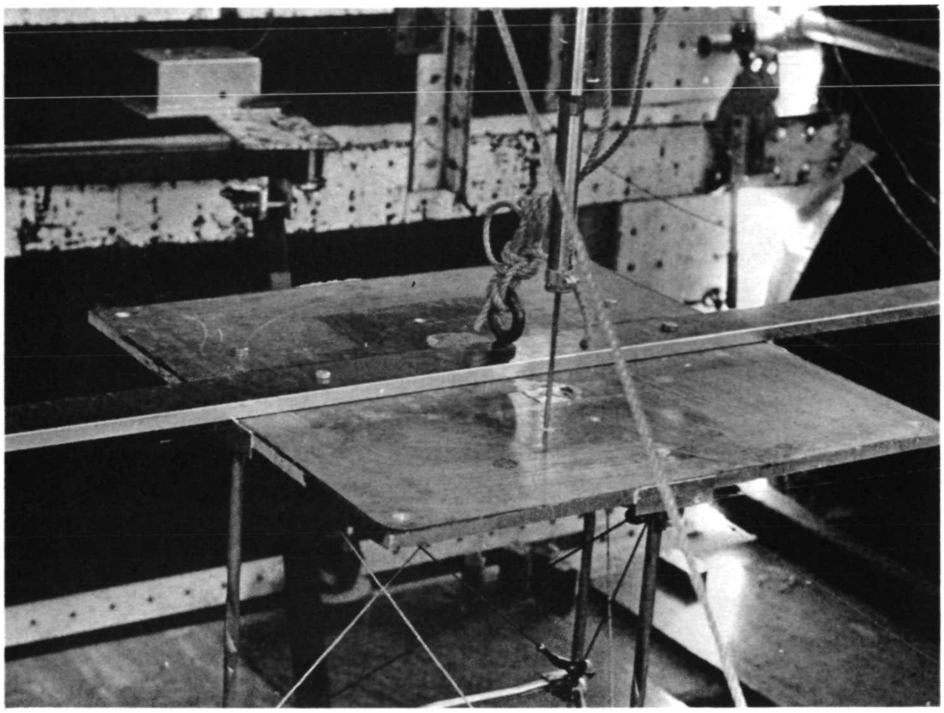


Figure 3.3-36 ETB Tip Torsion Bar and Potentiometers

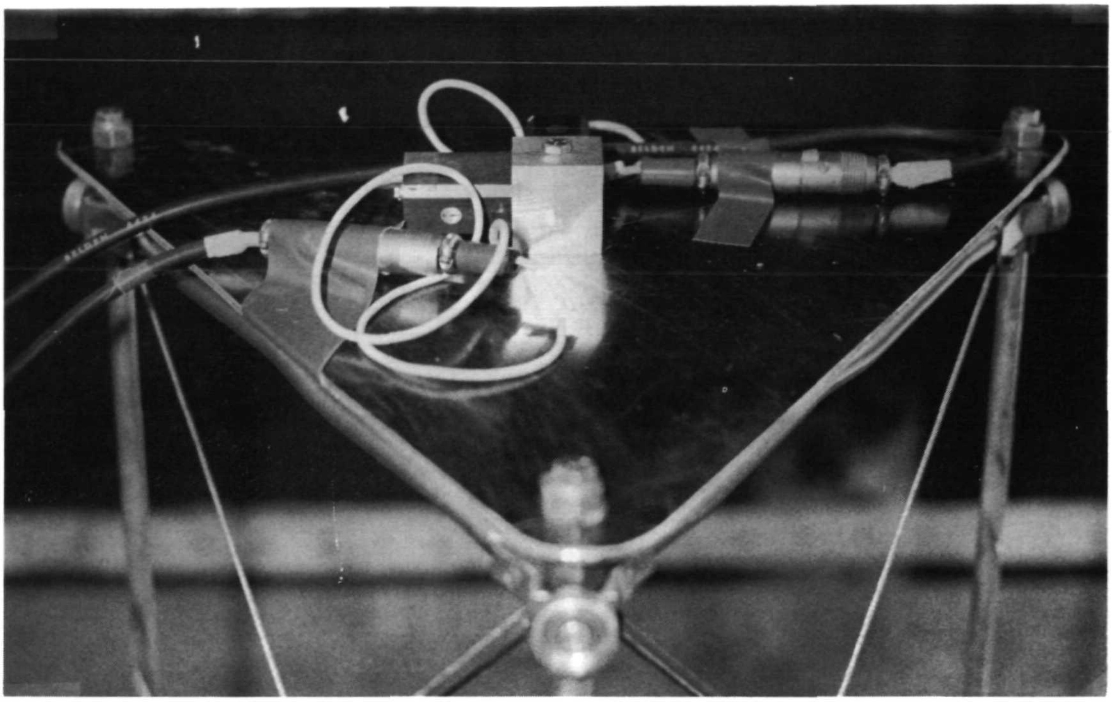


Figure 3.3-37 Two Axis Accelerometer - Damping Test

(3) Figure 3.3-38 shows the deflected beam during the bending and shear tests and indicates the method used to align the team tip after each deflection. The bar shown in Figure 3.3-39 was used to apply axial loads to the beam. This test program involved parametric measurement of beam bending and shear characteristics under a series of compressive preloads and represented the most complex of the beam tests conducted.

Fig. 3.3-38 ETB Bending/Preload Test

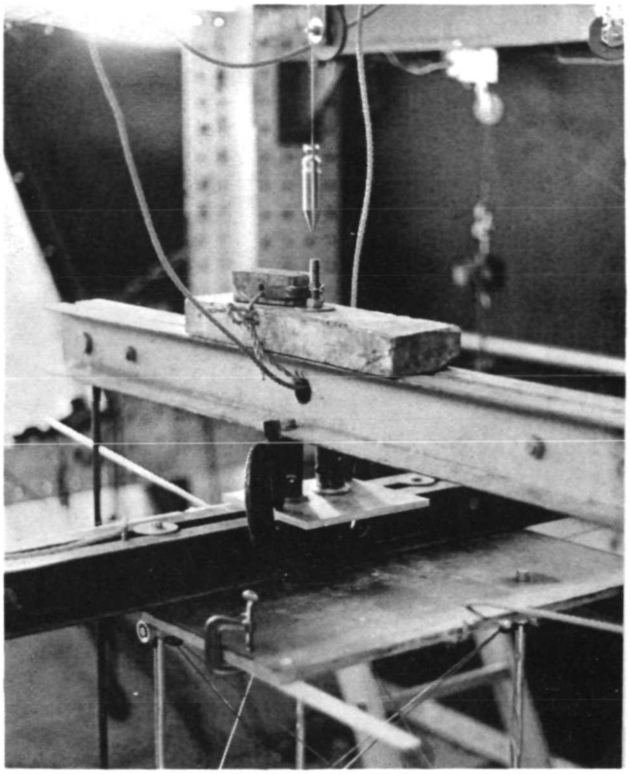
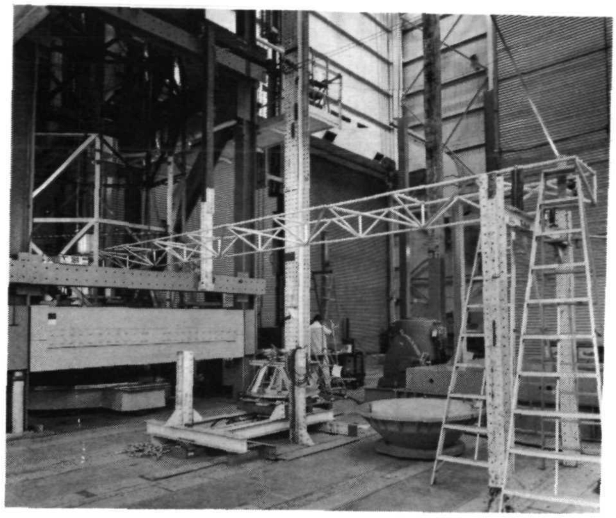


Fig. 3.3-39 ETB Tip Axial Load Beam

(4) The final test series was an artificial "g" load simulation with the inboard support assembly installed (Figure 3.3-40a) and the guy tape employed as a portion of the structure. The guy tape is shown (Figure 3.3-40 a & b) in its fully extended position. This test represented the most complex deployed structure test conducted prior to final deployment.

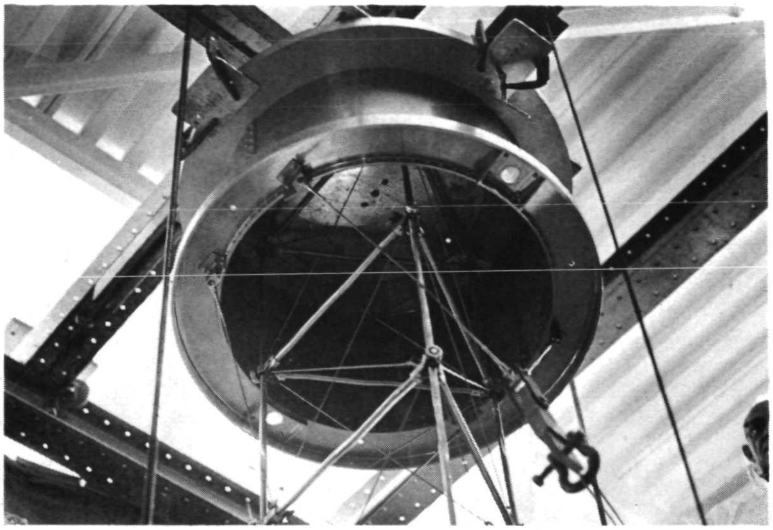


(a)

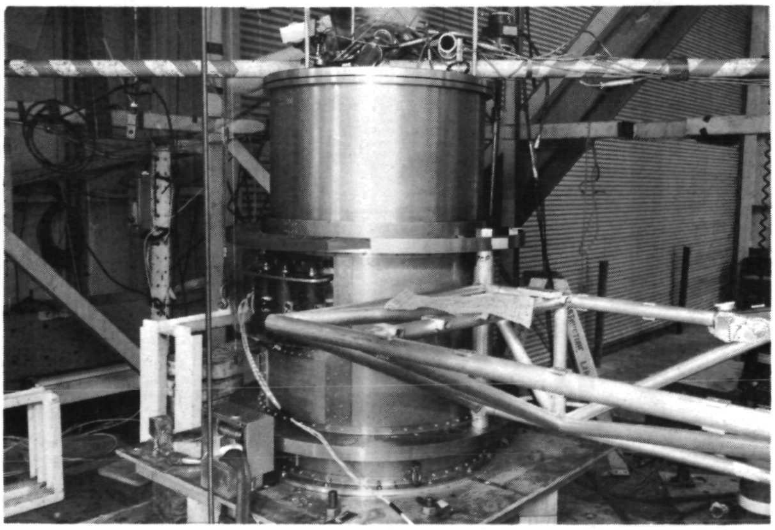
(b)

Figure 3.3-40 Structure Assembly Test Setup

Figure 3.3-41 (a & b) shows a closeup view of the artificial "g" test setup including (a) the cables used to apply tip compressive loads and (b) the interface between inboard support assembly and the beam canister.



(a)



(b)

Figure 3.3-41 Structure Assembly Test

This ETB test series was completed in November 1971. Test results and evaluation are discussed in LMSC/D153526 (White Book).

Before delivery to Lockheed in March 1972, several hardware performance and evaluation tests were conducted at Ball Brothers Research.

For example, preliminary evaluation tests were run on the inner gimbal drive with one motor used as load and test fixture for the other Figure 3.3-42. Life tests were not run although the hardware is available for this purpose, if required.

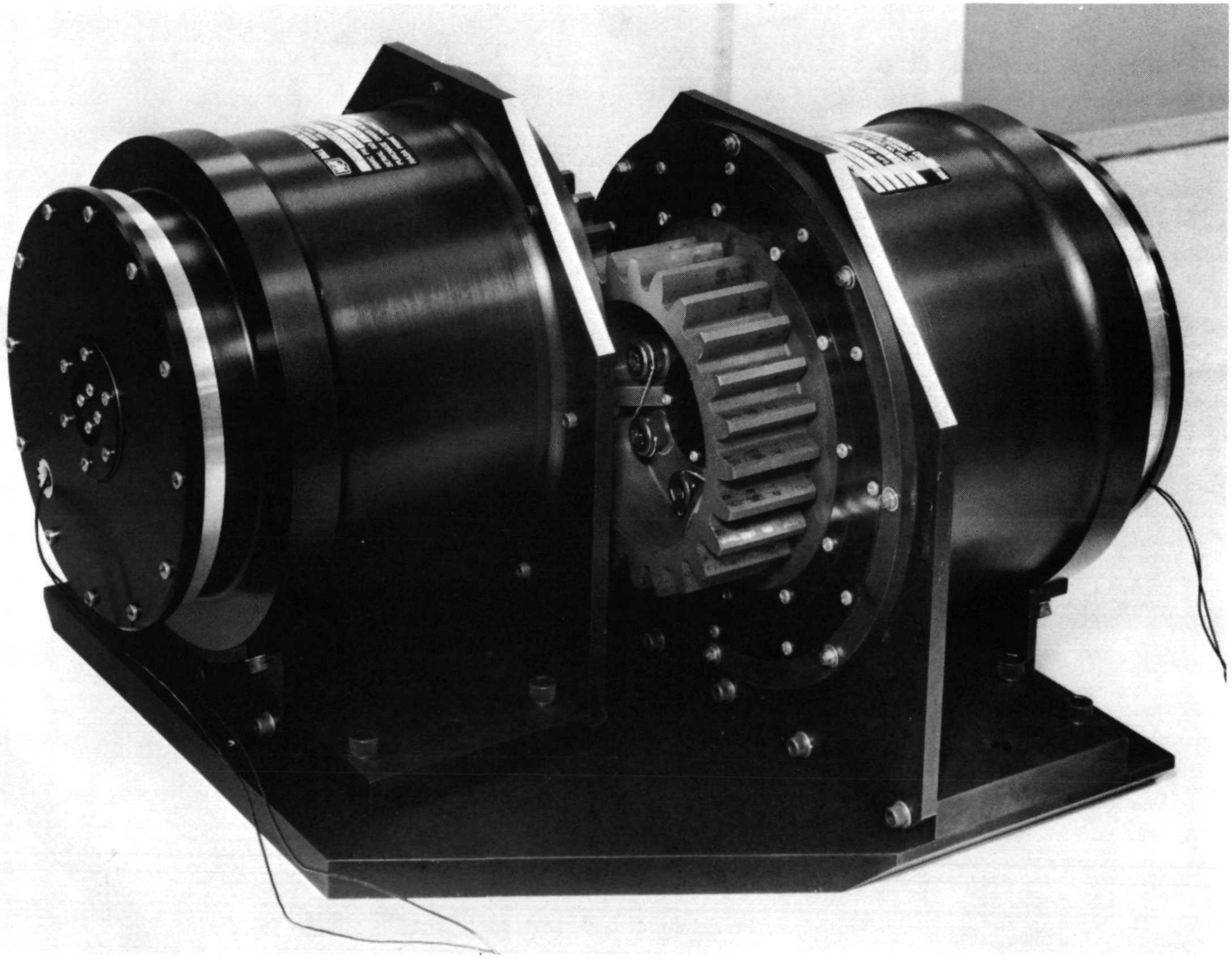


Figure 3.3-42 Motor Driver Test

3.3.2.3 System Level Tests

To demonstrate compliance with Space Station launch and orbital requirements, full scale array/structure and orientation/power transfer assemblies (the former at Lockheed, the latter at Ball Brothers) were ground tested during the second quarter of 1972. During these tests ground handling, storage, checkout and test procedures were evaluated in addition to the functional performance of the system and its components. Recommendations for altering design features were made to improve system performance.

Array Quadrant Test

By the last week of March 1972, the Array Quadrant Specimen and the Suspension Counterbalancing Test Support system were satisfactorily mated and checked out for testing.

One-fourth (5 strips) of the total 10,000 ft^2 array systems along with the ISA-OSA and ETB structures formed the final Test Specimen.

Figure 3.3-43 illustrates how the test equipment was utilized to simulate a zero gravity environment for the full scale 2500 ft^2 specimen during deployment and retraction. This testing evaluated operational capabilities and handling techniques.

The solar array assembly was supported by a 3000 pound load plate at the point where the extendible beam assembly attached to the ODAPT and also at two points along the inboard support assembly. A forty (40) ft. long reaction or counterbalance beam assembly was suspended above the test specimen from a mechanical hoist, specially installed on the building for this test. This beam assembly included pulley assemblies for counterbalance systems and a reaction point for the moment reaction system. The reaction beam was leveled and positioned by a cable from each end of the beam to large concrete blocks on the floor. A guy tape counterbalance was positioned to eliminate the lateral effects of the specimen guy tape during its deployment.

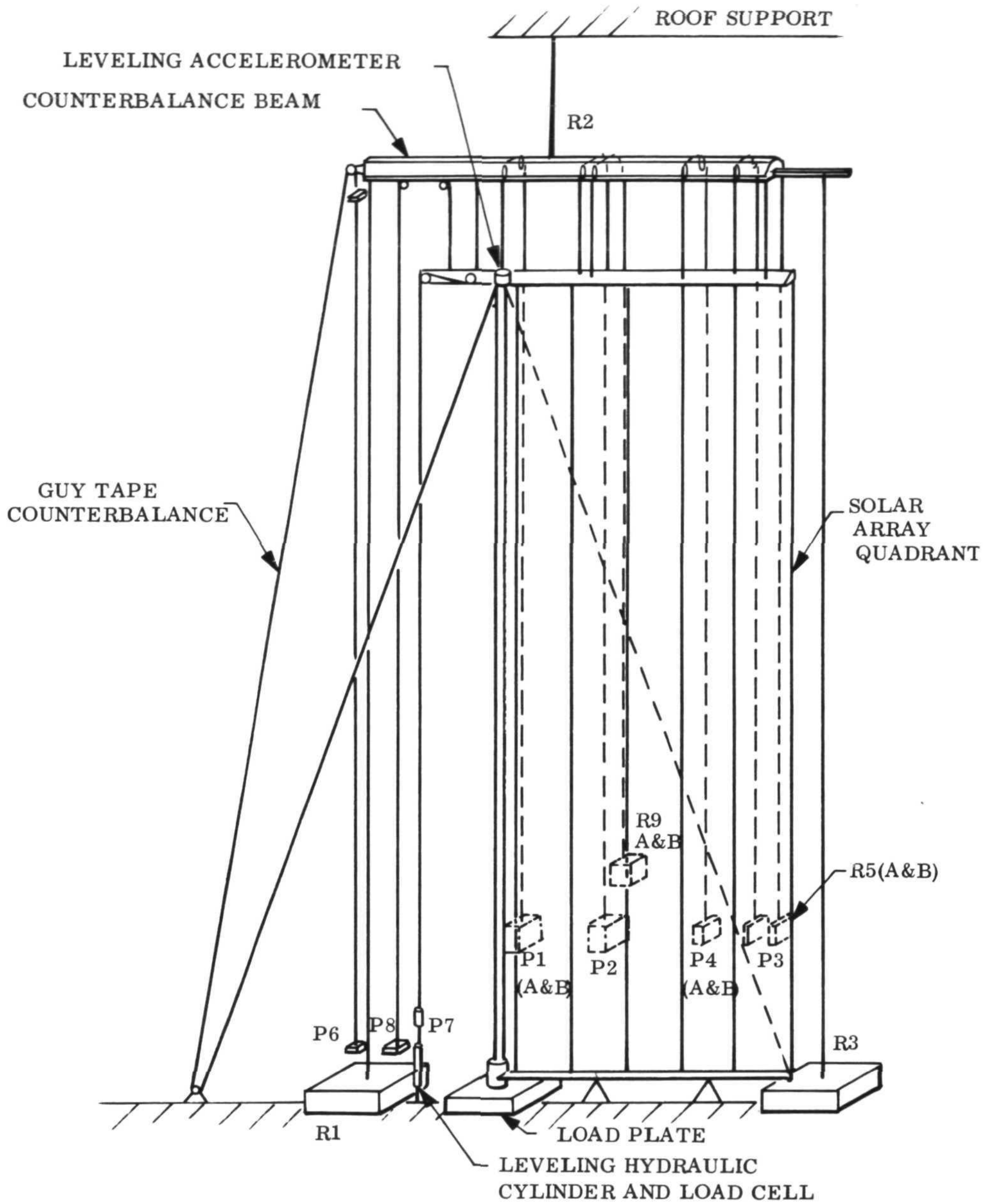


Figure 3.3-43 Full Scale Quadrant Deployment Test Schematic

As the beam deployed, P_1 (A & B) counterbalanced the fixed weight of the cap and the variable weight of the beam. P_2 and P_3 counterbalanced the fixed weight of the OSA, the zero "g" mechanism, and strip 1 and 2 covers. P_9 counterbalanced the variable weight of strips 1 and 2, the guide wires, and the pullup tapes. P_4 (A & B) counter-balanced the fixed weight of strip 3 and 4 covers and the variable weight of strips

3 and 4. P_5 (A & B) counterbalanced the fixed weight of the strip 5 cover and the variable weight of the strip. P_7 was a cap leveling system which counterbalanced the movement about the cap due to strip, guide wire, and pullup tape tensions. A single axis leveling accelerometer was used to sense quadrant tilting and provide the necessary input to the hydraulically controlled counter-mass to correct the load level.

Figure 3.3-44 shows three sizes of flexible solar arrays which have been manufactured and assembled at LMSC. In the left foreground is the 100 ft^2 array designed and built on LMSC independent development funds; to its right the LCS (Lockheed Communication Satellite) fold-out array; at right center the fully deployed 2500 ft^2 Space Station solar array quadrant. The Ball Brothers Research Corp. ODAPT unit appears at lower center--this unit mounts the solar array deployment beam to the Space Station power boom.

A view of the fully deployed solar array quadrant taken from the crane platform in the 120 ft. high building where the deployment/retraction test was conducted is shown in Figure 3.3-45. Beam cap counterbalancing arm and OSA are clearly visible.

Figure 3.3-44 Full Scale Array Quadrant Test

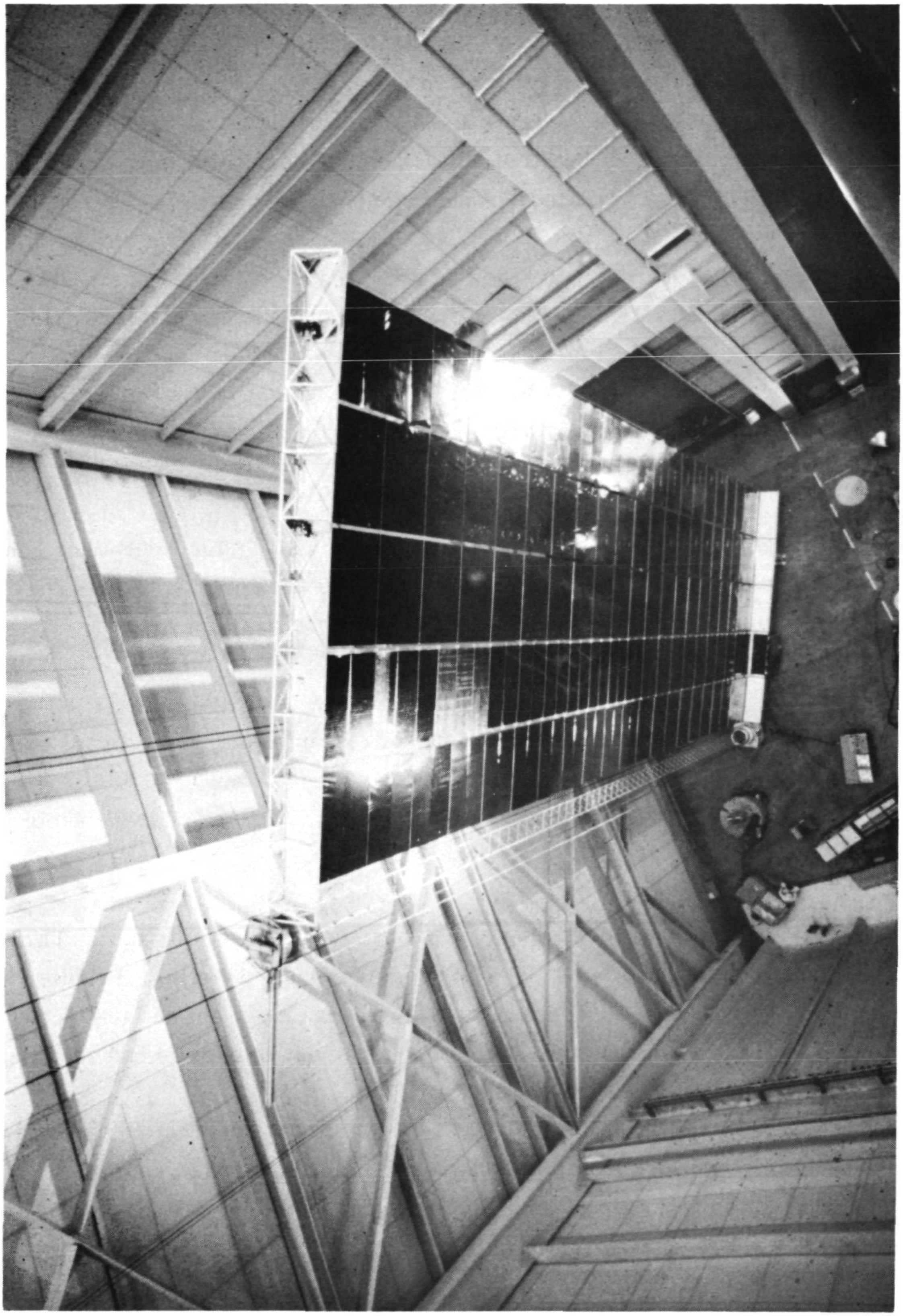


Figure 3.3-45 Full Scale Array Quadrant Test (Top View)

ODAPT Functional/Static Load Tests

Ball Brothers Research Corporation conducted functional and mechanical tests on the ODAPT assembly to evaluate its operating and structural stiffness characteristics. These characteristics in turn can then be used for determining overall space station interaction characteristics. The purpose of the functional test was to perform an end-to-end checkout of the ODAPT Assembly drive system, inner and outer gimbals, and to demonstrate that it was operational. The functional testing consisted of measuring the friction, stiction, acceleration, velocity, stopping transients and motor parameters. The frictional characteristics were very close to those estimated in the Design Analysis Topical (LMSC A995719). The inner gimbal estimates were 1250 ft-lbs as compared to 1207 ft-lbs measured. The outer gimbal was estimated to be 40 ft-lbs and measured at 47 ft-lbs.

The results of the testing for the other characteristics indicated some eccentricity in the inner gimbal, and slight "overshoots" in outer gimbals acceleration tests. The stopping transients measured before and after the mechanical tests indicated no damage to the bearing systems.

In the stiction test (setup shown in Figure 3.3-46) for example, the inner gimbal was driven at the slowest rate possible with no inertia load (1 revolution in 57 min. 14-1/2 sec) and the angular displacement vs time required for the angular displacement was recorded.

The test data indicates that the inner gimbal drive has some eccentricity in it. The high point or eccentric area is located 120° CW from the inactive gimbal. Results of the functional test were used for comparison with the results of similar tests conducted after the mechanical evaluation tests to determine that no damage had occurred to the ODAPT assembly.

The purpose of the mechanical evaluation tests was to determine the structural stiffness or spring rate of the ODAPT assembly structure.

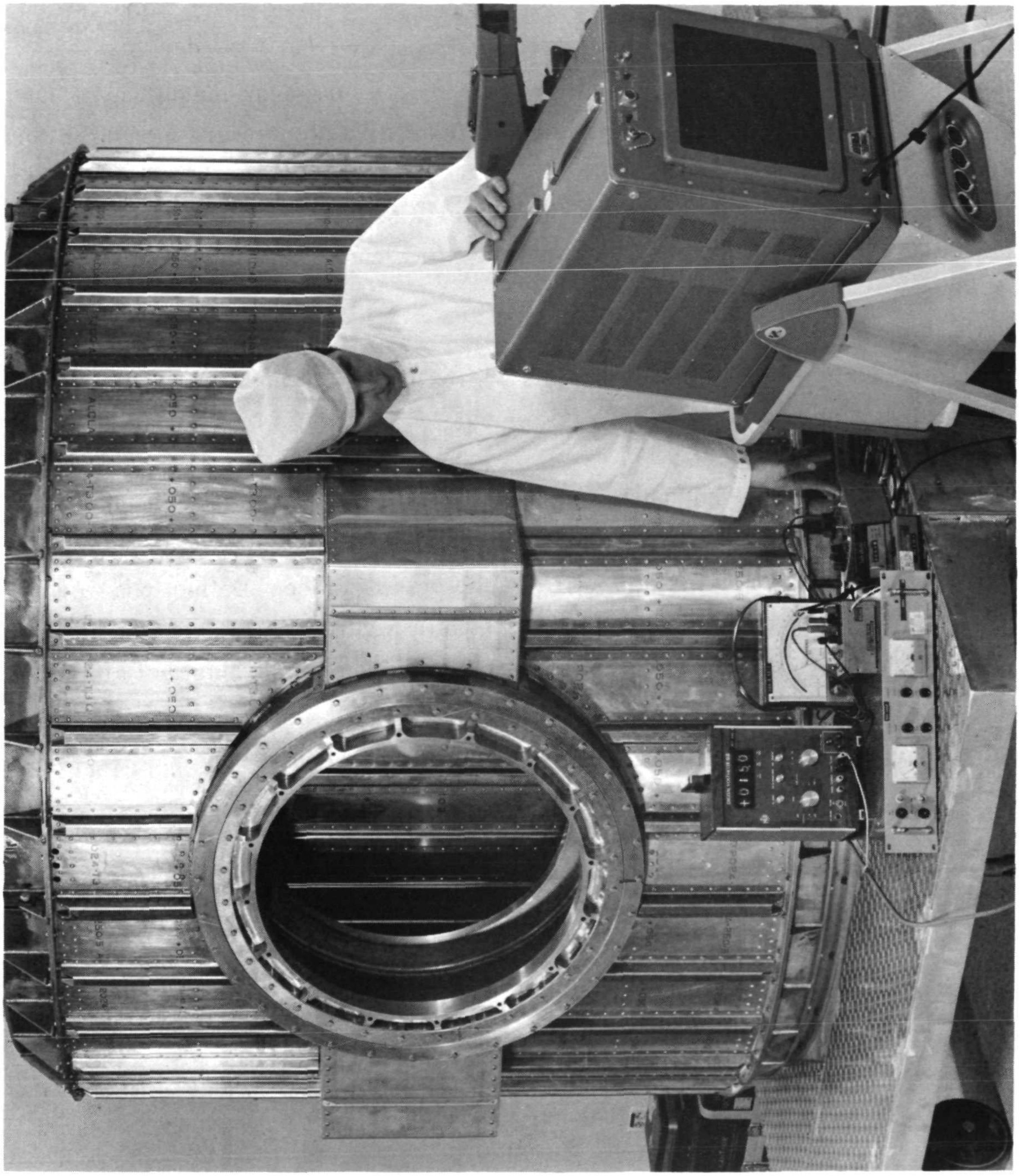


Figure 3.3-46 ODAPT Stiction Test Setup

The mechanical testing consisted of static loads applied longitudinally, laterally, and torsionally to the outer gimbals. The data recorded for this test was not analyzed but is presented in the test report.

In the longitudinal test the direction of load was up at a maximum magnitude of 5000 lbs, 2500 lbs to each outer gimbal outside housing. Five points were measured for deflections. In the lateral pull test the direction of the applied load was lateral as shown in Figure 3.3-47. Its maximum magnitude was 5000 lbs and deflections were measured at 5 points. The load simulated the worst case shuttle abort loading on the Space Station.

The load was applied in 1000 lb increments approximately 30 inches from the base with deflection readings taken at each increment.

The ODAPT assembly was subjected to a longitudinal torque that was 40,600 ft-lbs maximum in magnitude. The loads were applied vertically upward to the outer gimbal front face and vertically downward on the opposite face. Deflections were measured on both sides of the structure as shown in Figure 3.3-48. The applied load creating the torque was applied in five 1000 lb increments. This test simulated the worst case artificial "g" loading on the ODAPT.

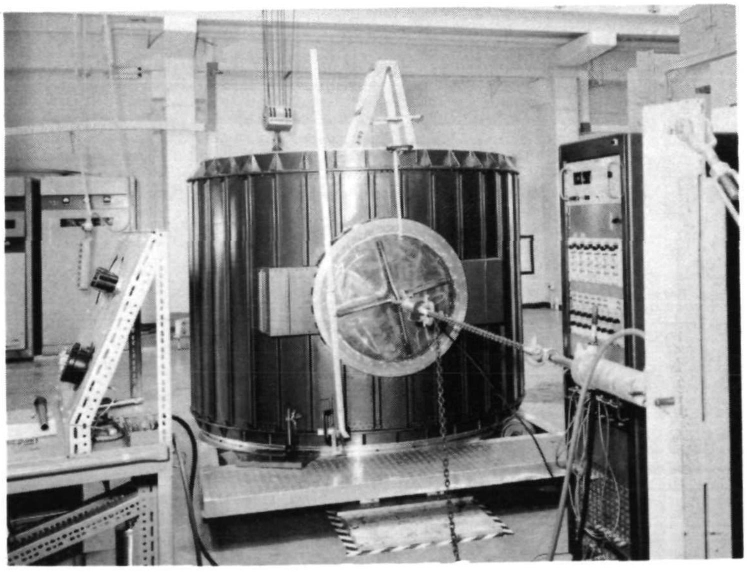


Figure 3.3-47 Shuttle Abort Loading (ODAPT)

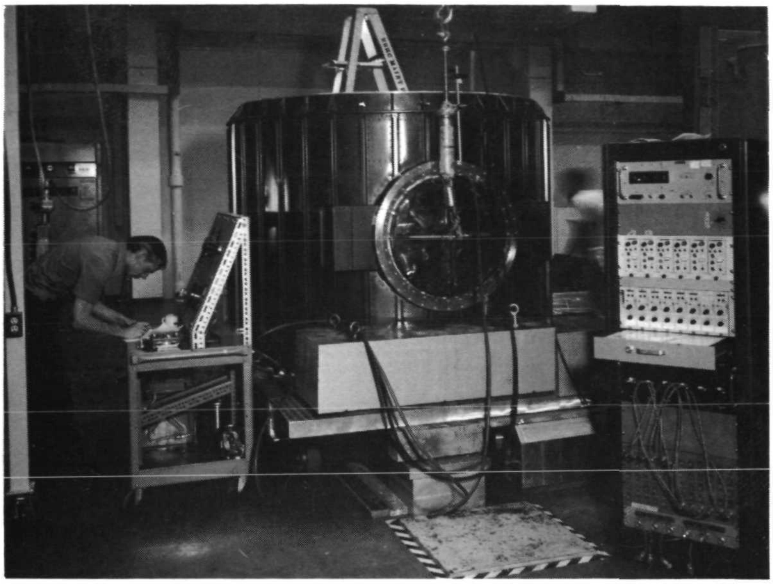


Figure 3.3-48 Artificial "g" Loading (ODAPT)

4.0 SPECIAL PROJECTS

Several revisions were made to the original eighteen (18) months contract. Reasons for these changes are self-explanatory. The extension, March 1 to August 1, 1972, was a contract add-on with increased funding to perform three additional tasks, while completing those required by the contract. The three special projects, in the order of their completion, were:

- 1st Topical Report Update - Technology Evaluation
- Temperature Cycling Test Plan
- Alternate Applications Study

Using excerpts from each of their reports, these three tasks are briefly described in the following sections.

4.1 1st Topical Report Update

The first topical report (LMSC/A981486) published in December 1970, was a review and evaluation (five years - 1965 to 1969 inclusive) of available technology which might be applied to the design of a 10,000 ft^2 Space Station Solar Array. It was published in a loose-leaf notebook so that it could be updated periodically. The update report which followed had as an objective to summarize the work performed since 1969-70 concerning lightweight solar array assemblies and to include any work that was missed in the original survey and search.

The new update document (LMSC/D159124), published in July 1972, reviewed and evaluated applicable portions of some 296 documents (reports, papers, articles, etc.) in the following categories:

•	Array-Design, Development and Test	- 54
•	Deployment and Orientation Systems	- 86
•	Solar Cell Development and Improvement	- 156
	Total	296

In addition to the review and evaluation of available solar array technology, the original document contained two other significant sections; (1) a list of studies recommended to be performed to fill technology gaps or provide a beneficial weight or cost advantage and (2) an annotated bibliography which abstracted and categorized every technical report reviewed during that task. With respect to those items, the update report provides (1) a summary of the work accomplished and the work planned on the Recommended Studies and (2) a supplement to the bibliography abstracting all documents reviewed and abstracted for this updated report. The publication of both the original and the updated reports was a major goal of this program, to identify solar array technology areas where additional development should occur and to recommend specific studies which could be conducted to fill in the technology gaps. A total of 34 studies were recommended in three categories:

Category I - Projects conducted on the Space Station Solar Array contract

Category II - Additional Projects Recommended to Ensure Technology Readiness

Category III - Projects Recommended to Provide Down-Stream Improvement

As examples, four of the 10 Category II items and their current status are described in Table 4.1-1.

TABLE 4.1-1
SAMPLE CATEGORY II RECOMMENDED PROJECTS

RECOMMENDED STUDIES	SUMMARY OF RECOMMENDATIONS	CURRENT OR COMPLETED ACTIVITY	PLANNED ACTIVITY
1. TEFLON COVER EVALUATION	Inadequate process and production technique optimization and environmental testing for this newcomer as coverglass material. Applicable to any size power system with great weight and cost savings potential. Tests measuring degradation of teflon covers by particle and UV radiation and determination of thickness to application/environment are required.	• Lockheed investigation for NASA-LeRC (L.4-34 and L.4-42). • NASA LeRC contract with TRW initiated February 1972 to fabricate and test heat-bonded teflon covers for solar cells.	Continuation of TRW/LeRC work. LMSC ID work on spray-on teflon solar cell covers.
2. INTEGRAL SOLAR CELL COVERS	Integral covers can be 1-2 mils, as compared to 6 mil minimum for conventional coverglasses. Significant weight reduction and elimination of the adhesive would result. Development of processes and material for use with standard cell manufacturing techniques and of production capability is required. Heavy process development expenditures should not be made until this approach is compared with Program 1 (above) results.	• Development work by Heliotek (H.3-21 and H.3-24) and Texas Instruments (T.2-1). • Solar cell coverglass development by Ion Physics (I.3-16, I.3-17). • In-house development by NASA Goddard and NASA LeRC (N.2-28, N.6-27, N.6-40 and N.6-43).	GE funded by JPL for spray plasma deposition of ultra pure fused silica without stress problem.
3. IMPROVEMENT OF EOL SOLAR CELL EFFICIENCY	Investigations state theoretical attainable efficiencies up to 22%. These higher efficiencies can be achieved only by a better understanding of the physical phenomena governing solar cell performance. Electrical degradation in the cell due to UV and particle radiation, as well as repeated temperature cycles, should also be reduced. Testing to evaluate improved cells should be carried out at one central facility to better control conclusions.	• Lithium doping (H.3-20, H.3-25, C.3-12, C.3-16, A.1-8, R.1-25, R.1-26, R.1-31, N.4-22, J.1-1). • Efficiency improvement (P.1-4, P.2-7, N.7-12, N.7-13, N.4-2, N.4-16, N.4-34, C.9-1).	• Centralab and Heliotek will continue development work with NASA LeRC to improve cell efficiency to 20%. • IBM will continue development efforts in Gallium Arsenide cells to verify performance of 18%.
4. WRAPAROUND CONTACT SOLAR CELLS	Development of backside contact cells would result in cost reductions of up to $200/ft^2$ by reducing the complexity of panel assembly. Present series connection to the top electrode calls for generous stress relief series tabs and increased cell spacing complicating assembly. Whereas backside contact cell will allow fully automated assembly, reduce series spacing, and padding thickness and weight.	• Heliotek development work (H.3-26 and H.3-19). • Centralab development work (C.3-13 and C.3-17). Under above contracts wraparound contact cells were developed for both LMSC and Lewis Research Center.	Evaluation of wraparound contact cell application will be performed on NASA MSFC contract NAS8-28432 and on NASA MSC contract NAS9-11039, both with LMSC continuation of LeRC work.

This type of summary of recommended projects (completed or currently planned) measured the effectiveness of the program and determines the degree of "Technology Readiness" at the time of publication.

Most of the technical data for the Update Report is presented in Section 4.0 which discusses and evaluates the available technology. Section 4.2 (the orientation and power transfer section) was omitted since Ball Brothers Research Corporation provided the information for the Blue Book and their subcontract was completed prior to preparation of this update. However, the extensive work performed by BBRC on this program represents a major portion of the applicable technology work accomplished since 1969 in the drive system and power transfer areas and this work was reported in detail in the Second and Third Topical Reports, LMSC-A995719 and LMSC-D153526, respectively.

Technology advancements and what may turn out to be major breakthroughs were accomplished in the 1970-71 time period with the advent of 18% gallium arsenide solar cells, wraparound contact solar cells, lithium doped solar cells, extendible-retractable structures, array packaging techniques, adhesiveless solar cell array assemblies, large area array testing methods, slipring-brush material development, lubricant evaluation, and solar cell assembly fabrication techniques.

All of the above items are discussed in this report as well as flexible array concepts now being developed by European Satellite teams.

Typical of current flexible solar array systems which were designed, fabricated and ground tested are the four summarily presented in Table 4.1-2. The successful flight testing of the Hughes array FRUSA (flexible rolled-up solar array) in 1971 was a major milestone in flexible array technology.

A summary of major flexible array ground test programs completed during the update period is presented in Table 4.1-3.

Table 4.1-2

FLEXIBLE ARRAY SYSTEMS SUMMARY

Contract	Title	Company or Agency	Dimensions Size	Power/Wt Watt/Lb	Program Status	Application (Mission)	Retractable	Stowage Method	Bibliography No.
1) NAS 9-11039 MSC-Houston	Large Space Station Solar Array	LMSC	10,000 ft^2 (2500 ft^2 Quadrant – 90 ft by 36 ft Fabricated)	18 watts/lb	Fabricated/quadrant for feasibility Ground Test Final Report – Aug 1972	Shuttle Modular Space Station	Yes	Flat Foldout	L.4-47 L.4-48 L.4-56
2) Contract 952314 NASA-JPL	Rollup Subsolar Array	GE (Valley Forge Space Center)	250 ft^2	30 watts/lb	Ground environmental and performance tests completed Oct 1970 Final Report – Feb 1971	Interplanetary Probes	Yes	Drum Roller	G.2-9 G.2-21
3) F33615-68-C-1676 Air Force – APL	Flexible Rolled-Up Solar Array	Hughes Aircraft	166 ft^2	22 watts/lb	Flight tested on Agena Oct 1971 Final Report – Jul 1972	Experimental Air Force Vehicle	Yes	Drum Roller	H.6-7 through H.6-13 H.6-37 through H.6-40
4) ESRP	–	AEG – Telefunken	108 ft^2	22 watts/lb	Design study and demonstration model only. No reports available.	Planned for Direct Broadcast TV Satellite and CTS Satellite	Yes	Drum Roller	Aviation Week 24 Apr 1972 Communication Forestieri-Lott

Table 4.1-3

FLEXIBLE ARRAY – MAJOR HARDWARE TESTS

Company	Specimen	Description of Tests	Results
LMSC Contract NAS 9-11039 MSC-Houston	1 Quadrant of 10,000 ft^2 Space Station Solar Array 4 Strip-Mylar Mockup 1 Strip-Three Modules Solar Cells	• Astromast static load tests for zero and artificial g space station requirements • Array Quadrant Series of deployment and retractions	• No mechanical failures during demonstrations • Proof of packaging techniques
GE Contract JPL-952314	2 Blanket/Cylindrical Drums for Each Total Array Area 250 ft^2 Each Blanket 33.5 x 4 ft Dimensions	• Bi-Stem Thermal Bending Tests • Systems Level Tests: Deployed dynamics pyrotechnic shock, thermal-vacuum, stowed dynamics • Throughout tests array blanket and mechanical inspections for damage or breakage	• Dominant test problem accommodating gravity forces so as to eliminate interference with tests • Achieved major objectives conceived new test techniques
Hughes Contract F3365-68-C-1676	2 Blanket/Common Cylindrical Drum Total Array Area 165 ft^2 Each Blanket Dimension 14.8 x 5.5 ft	• Flown on Space Test Program (STP) 71-2 Vehicle – Completed 8 full months of operational service to date • Successfully performed 10 complete rollups and rollouts and 2 partial extensions/retractions (1/3 and 1/6)	• Generated the required 1465 watts • Telemetry data system failure limited flight performance information

4.2 Temperature Cycling Test Plan

A major problem area identified during the technology evaluation phase of this program was the lack of comparable solar cell assembly temperature cycling test data. The effects of temperature cycling have been identified as the principal cause of failure of interconnected solar cell assemblies. Although this is a well-established fact and many temperature cycling programs have been conducted, there have been no successful attempts to standardize this testing or to control test variables to the extent that data can be evaluated comparatively with other test or flight results. Also the testing that has been done cannot be used to predict extended lifetime cycling results.

Based on past experience in qualification testing of solar arrays the most critical environment affecting solar panel performance is the temperature variation associated with alternate sun/shade cycles in earth orbit. This criticality has been intensified in recent years with the emphasis on lightweight flexible (low heat capacity) solar arrays and increased mission lifetime. For example, the Space Station Solar Array must survive more than 58,000 temperature cycles over the temperature range of $-118^{\circ}C$ to $+70^{\circ}C$ during the ten year mission.

Because of the low mass of flexible arrays, temperature rates-of-change during cycling will be three or more times the temperature rates-of-change of present honeycomb panels. Recognizing this problem, many companies and agencies have conducted temperature cycling programs (normally to their own program-peculiar test conditions and on their own designs) with the result that almost no valid comparative data is available after years of testing.

The investigation conducted under this special project had as primary objective to review present solar array test facilities and testing methods and to evolve from these findings a recommended plan for standardizing future thermal cycling tests. The secondary objective was to propose a facility design for implementing the plan.

Some of the more important goals of the test program outlined in the test plan are:

- o Evolve and select life-test candidate designs
- o Demonstrate feasibility of proposed designs
- o Understand the basic failure mechanisms, and, ultimately, develop the analytical models to predict failure
- o Check predictions of temperature cycle range and rates-of-change of temperature
- o Determine methods to accelerate testing

Twenty (20) thermal cycling tests, conducted by thirteen (13) different companies, were identified during the investigation. The majority of these tests were not performed in a vacuum environment, and among these, only two consisted of more than 300 cycles.

The weak areas in thermal cycling state-of-the-art prompted the requirement for a new test plan. These deficiencies are outlined in Section 2.0 of the report LMSC/D159198 published in September 1972. The tests do point out that the most troublesome parts of the solar panel assembly are the series interconnects, the solder, and the cell-to-substrate adhesive. Therefore more design emphasis should and is being placed on other techniques such as welding to replace soldering and wraparound contacts to eliminate stresses on series interconnects. New adhesives or adhesive-less systems must be developed for the cell-to-substrate problem.

The tests revealed that many different temperature ranges, environmental conditions, test methods, numbers of cycles, etc. made comparison of data very difficult. Therefore one integrated, comprehensive approach to thermal cycle testing is provided by the plan generated from this study.

Perhaps the most important temperature cycling testing to be done on a large variety of solar cell assemblies is the work which is being performed by Lockheed and NASA/MSFC. Lockheed is under contract (NAS8-28432) to study improved flexible-substrate solar array designs, fabricate approximately 50 specimens for temperature cycling testing at MSFC, and evaluate the test data. The program greatly augments and complements a standardized temperature cycling program and will provide comparative temperature cycling results from the MSFC testing and much of the required specimens material properties data. Specifically, the MSFC program is developing an optimum flexible array design using temperature cycle testing as one of the evaluation parameters whereas the proposed program would standardize temperature cycle testing of all types of solar panel designs.

The study report also presents the design of a temperature cycling test facility that is capable of performing comprehensive tests on solar array samples. These facilities are either operational or under construction at the present time. They include:

- o Four separate thermal vacuum chambers for high quality environmental simulation and for short or long-term testing
- o A convective cooling (gaseous N_2), radiant heating test box with the capability of accelerating the temperature cool-down rate
- o A liquid N_2 dip and radiant lamp heating enclosure for fast screening of new designs.

The four vacuum chambers provide different temperature cycles representing near earth, mid-range and synchronous orbit environments can be performed simultaneously. This can be done with stainless steel bell jar type chambers of 28-in. diameter by 30-in. high. Sixteen standard samples can be mounted in a single chamber as shown in Figure 4.2-1. These chambers are arranged as shown in Figures 4.2-2 and 4.2-3 on an arc whose center contains the solar simulator. Thus the solar simulator can be rotated as required to obtain I-V curves from any of the possible 64 samples.

Assuming this new facility, it is estimated that approximately 1 to 2 months will be required to de-bug the equipment and to establish and test its capabilities after installation and before the start of specimen testing.

Table 4.2-1 summarizes the recommended tests to meet the immediate goals of the testing program described above. Figure 4.2-4 shows the test scheduling.

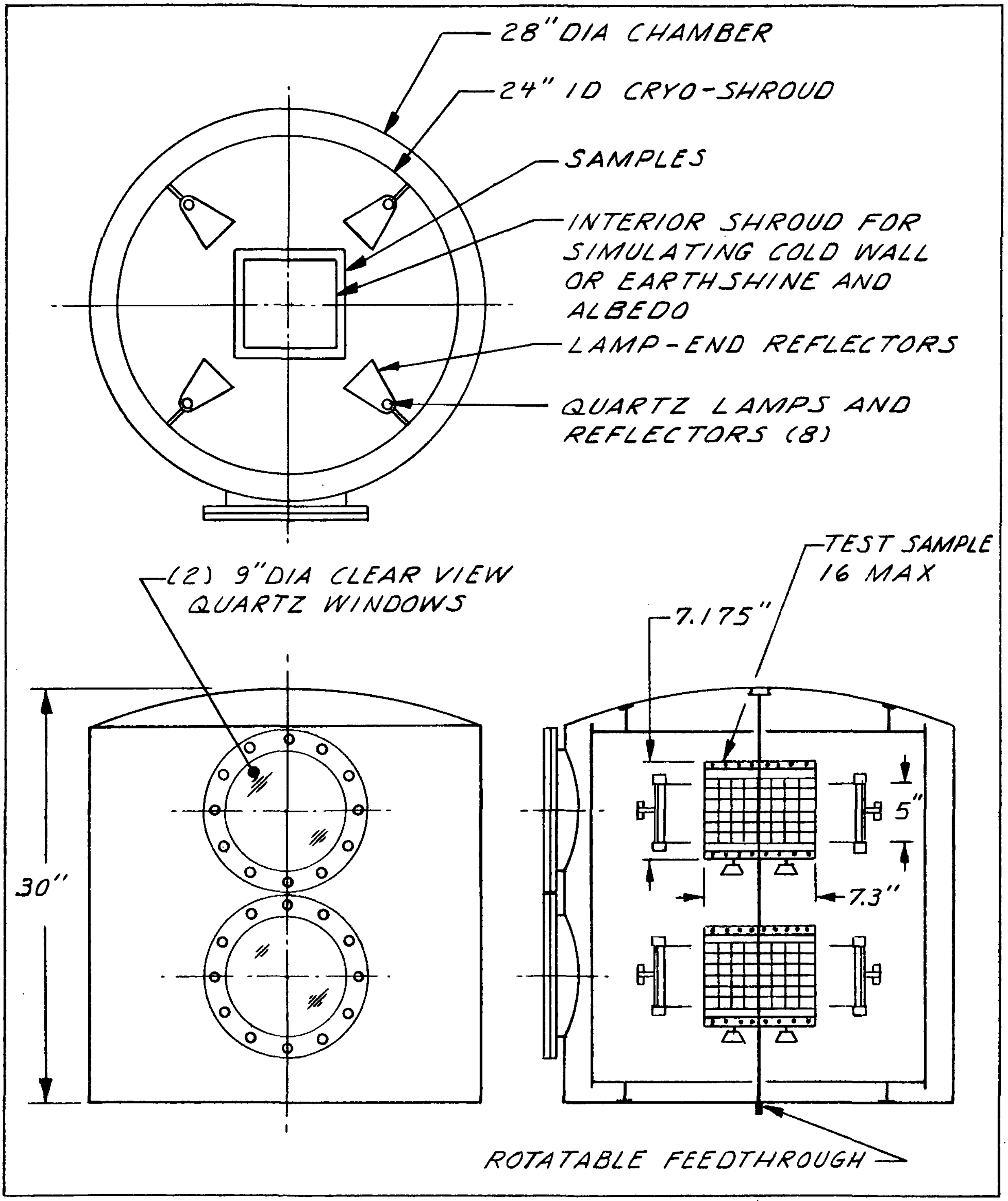


Figure 4.2-1 Sample Arrangement in Chamber

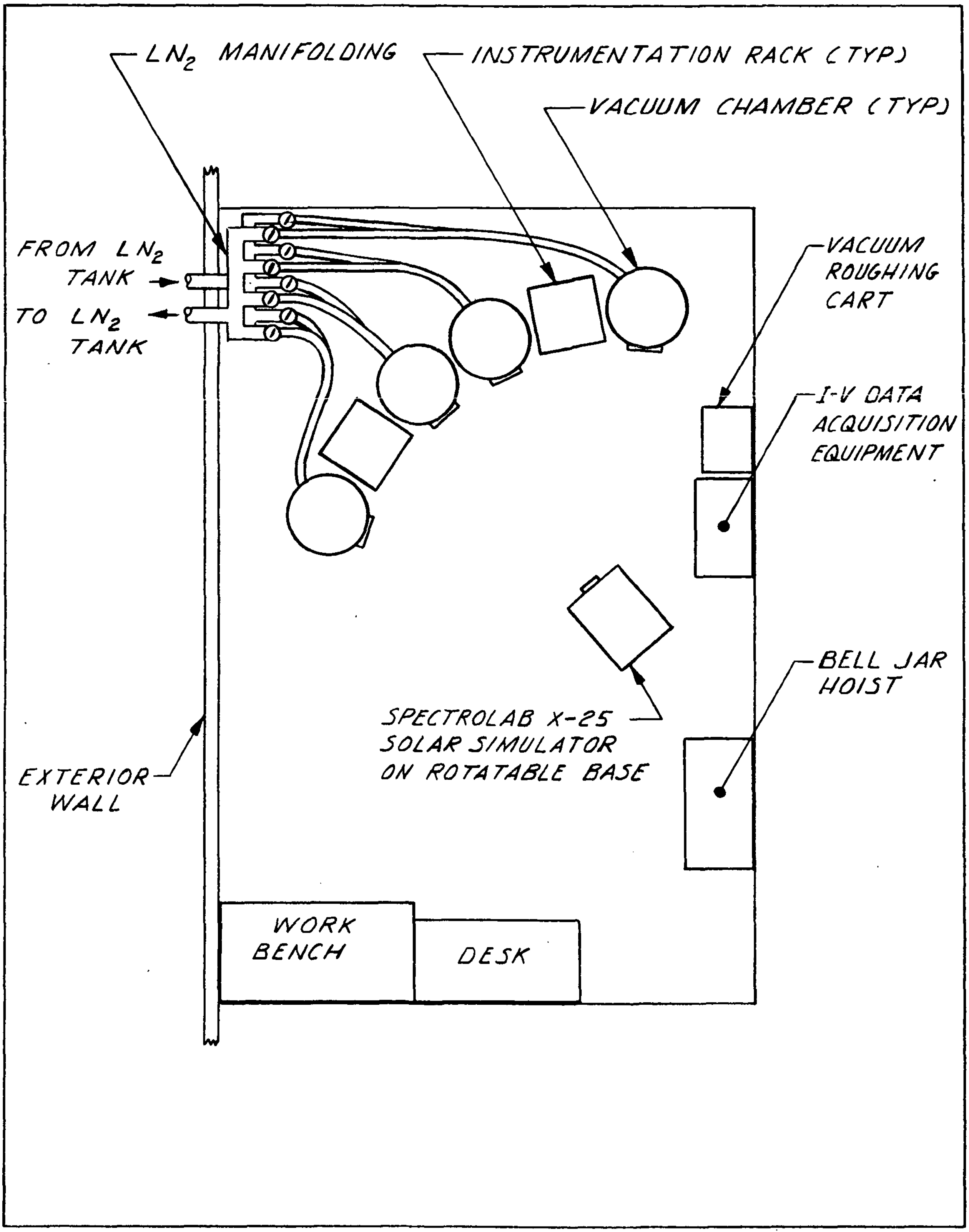


Figure 4.2-2 Layout LMSC Thermal Vacuum Test Facility

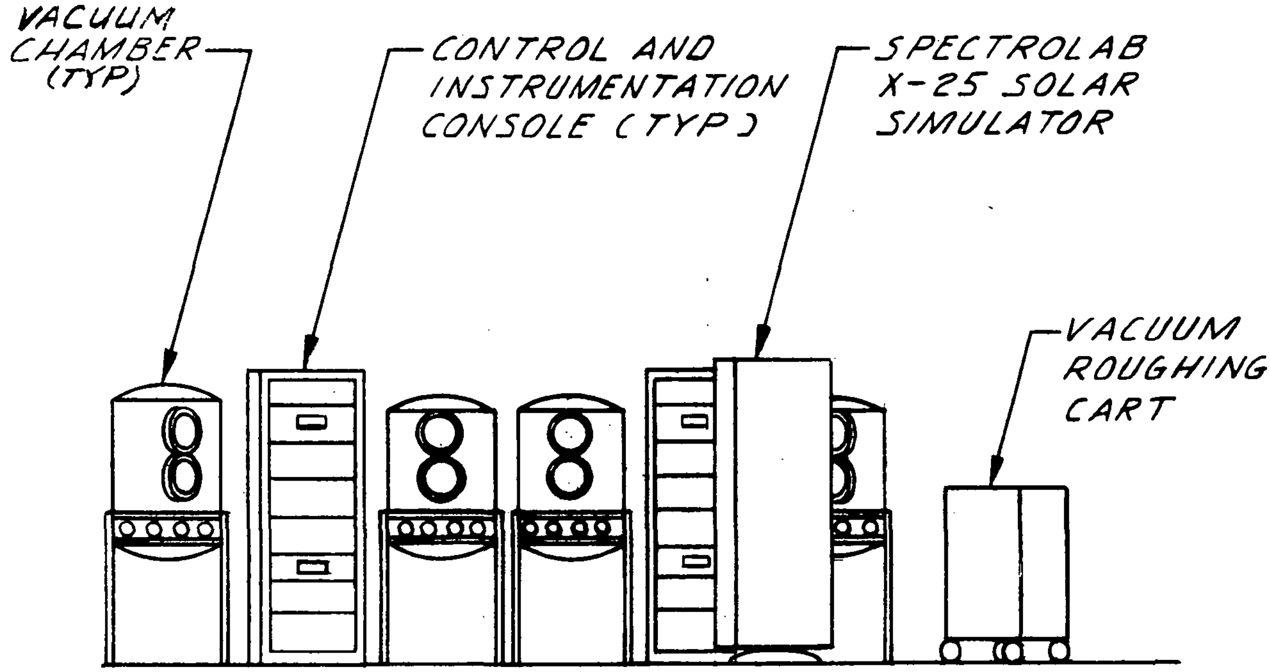
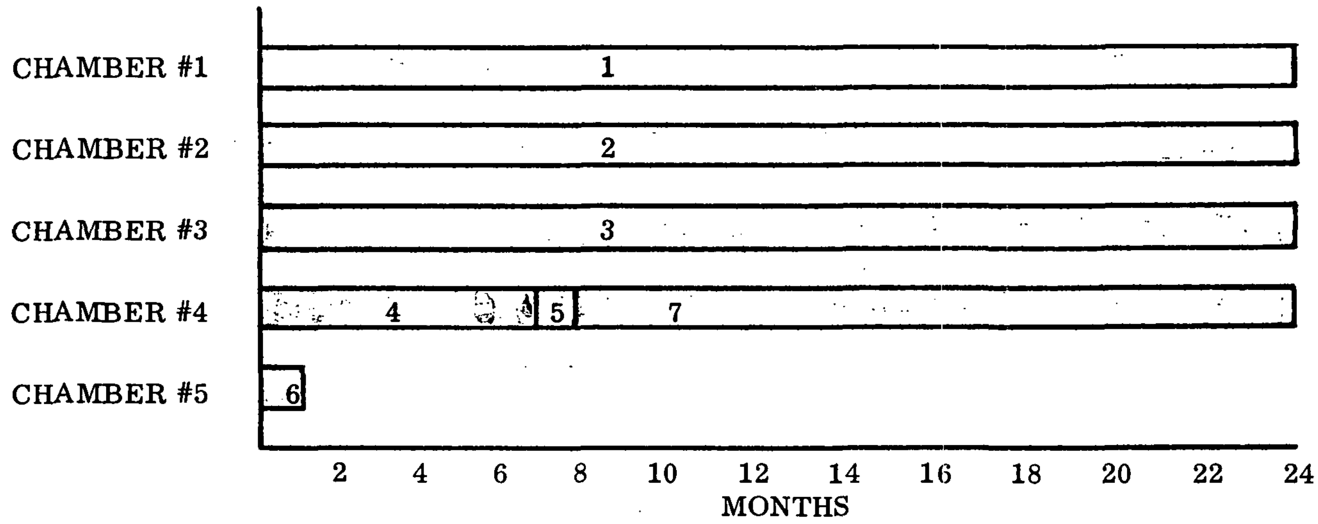


Figure 4.2-3 LMSC Thermal Vacuum Test Facility - Side View



NOTE: Numbers shown in bars are test numbers as listed in Table 3.7-5

Figure 4.2-4 Thermal Vacuum Test Schedule

TABLE 4.2-1

TEST DESCRIPTION SUMMARY

Test Number	Chamber	Test Environment	Sample Group (Note 4)	Orbit Simulation	Cycle Time	No. of Cycles
1	1	Vac	B	low	real time	10K cycles (2 year equivalent life, 2 year testing time), then special exam
2	2	Vac	B	mid range	real time	10K cycles (2 year equivalent life, 2 year testing time), then special exam
3	3	Vac	B	synch	real time rates, but high temp soak truncated to 9 hours	1K cycles (10 year equivalent life, 2 years testing time), then standard exam
4	4	Vac	A	low	accelerated	10K cycles (2 year equivalent life, 7 mos. testing time), then special exam
5	4	Vac	A	synch	accelerated	1K cycles (10 year equivalent life, 1 mo. testing time) then standard exam
6	5	Conv.	B	synch	accelerated	1K cycles (10 year equivalent life, 1 mo. testing time) then standard exam
7	4	Vac	--	Develop ways to accelerate temp cycles as itemized in Section 6.2		

Notes:

1. Chambers 1-4 are vacuum chambers; Chamber 5 is cold gas convection-cooled chamber
2. "Special" examination includes metallurgical exam and contact peel tests to detect incipient and future failures. "Standard" examination is non-destructive visual inspection under 30-power magnification.
3. Typical temp ranges, tracking array:
   - low earth orbit flex, $-118^{o}C$ to $+71^{o}C$
   - low earth orbit rigid, $-52^{o}C$ to $+68^{o}C$
   - synch orbit flex, $-193^{o}C$ to $+49^{o}C$
   - synch orbit rigid, $-157^{o}C$ to $+49^{o}C$
   - mid range, $-125^{o}C$ to $+65^{o}C$
4. Tables 9.2 and 9.3 describe the sample groups.

4.3 Alternate Applications Study

The third special project had as objective to survey several possible future NASA missions using solar photovoltaic power for primary electrical power source. The missions reviewed represent a wide diversity of power levels, array sizes, packaging and deployment constraints and provide the requirements base for assessing modularity of the solar arrays. During the Space Station Solar Array Program, modularity was incorporated into the mechanical and electrical elements of the design so that as power requirements became better defined the solar array size could be scaled up or down by simple addition or deletion of elements.

This study evaluated primarily incorporation of the Space Station solar array into Shuttle, RAM and a Space to Earth Power (STEP) experiment. Some consideration was also given to an Orbiting Lunar Station (OLS), advanced Skylab and Space Base applications. Pertinent characteristics of these missions are listed in Table 4.3-1.

It is noted that all of the missions with the exception of Skylab are directly related to the Shuttle for transfer to orbit. The presence of the shuttle affords assistance via manipulators or possible EVA for deployment and buildup of the arrays. However, in the study a primary integration constraint was that the solar array should be capable of automatic unassisted deployment. The primary study constraints were as follows:

- Maximum use of developed Space Station Solar Array designs
- Minimal or no EVA for solar array erection
- Direct substitution at the module or panel level wherein it is cost effective to retain existing packaging and deployment structure

Figure 4.3-1 shows an artist's sketches of four (4) of the current NASA missions (named above) and their solar array configurations. Modular sizing, power analysis, and system weight determinations were made for each of these configurations.

TABLE 4.3-1
CANDIDATE MISSIONS - COMPARATIVE DATA

MISSION	POWER REQUIREMENTS	ORBIT PARAMETERS INC.	ORBIT PARAMETERS ALTITUDE	VEHICLE ORIENTATION	CREW STATUS	SHUTTLE DEPENDENT
SHUTTLE	11 KW PLATFORM 1-10 KW PAYLOAD	28.5 - 100.7 0-100.7 PAYLOADS	100 - 270 CIRCULAR 100 - 700 ELIPTICAL	LONGITUDINAL AXIS COINCIDENT WITH LINE OF FLIGHT	MANNED	--
RAM RESEARCH AND APPLICATIONS MODULE	2.8 - 7.3 KW	0-55°	200-400 NM ACCEPTABLE 300-500 NM PREFERRED	MISSION DEPENDENT - STELLAR, EXCEPT SOLAR ON ASAO	FREE FLYING MODULE CATEGORY - MANNED ASSIST FOR START UP AND SERVICING ONLY	YES
STEP SPACE TO EARTH POWER	EXPERIMENT 10 KW	0-40°	250 NM & 19,323 NM	ARRAY INERTIALLY LOCKED ON SUN, TRANSMITTING ANTENNA LOCKED ON EARTH	UNMANNED	YES
	ULTIMATE SSPS SYSTEM 10^7 KW	0°	19,323 NM (SYNCH.)		POSSIBLY MANNED ON INTERMITTENT BASIS FOR SERVICING	YES (WITH TRANSFER STAGE)
ADVANCED SKYLAB	12.5 KW (BASED ON EQUIV. AREA)		235 NM	SOLAR INERTIAL, LOCAL VERTICAL	MANNED WITH QUIESCENT PHASES	NOT REQUIRED FOR BUILDUP
OLS ORBITING LUNAR STATION	20 KW DURING MANNED OPERATIONS NAS9-10924 VOL III, 1-4	0-90°	45-80	LUNAR VERTICAL	MANNED, WITH QUIESCENT PHASES	YES (WITH TRANSFER STAGE)
SPACE BASE	100 KW			SPIN STABILIZED	MANNED, WITH QUIESCENT PHASES	YES

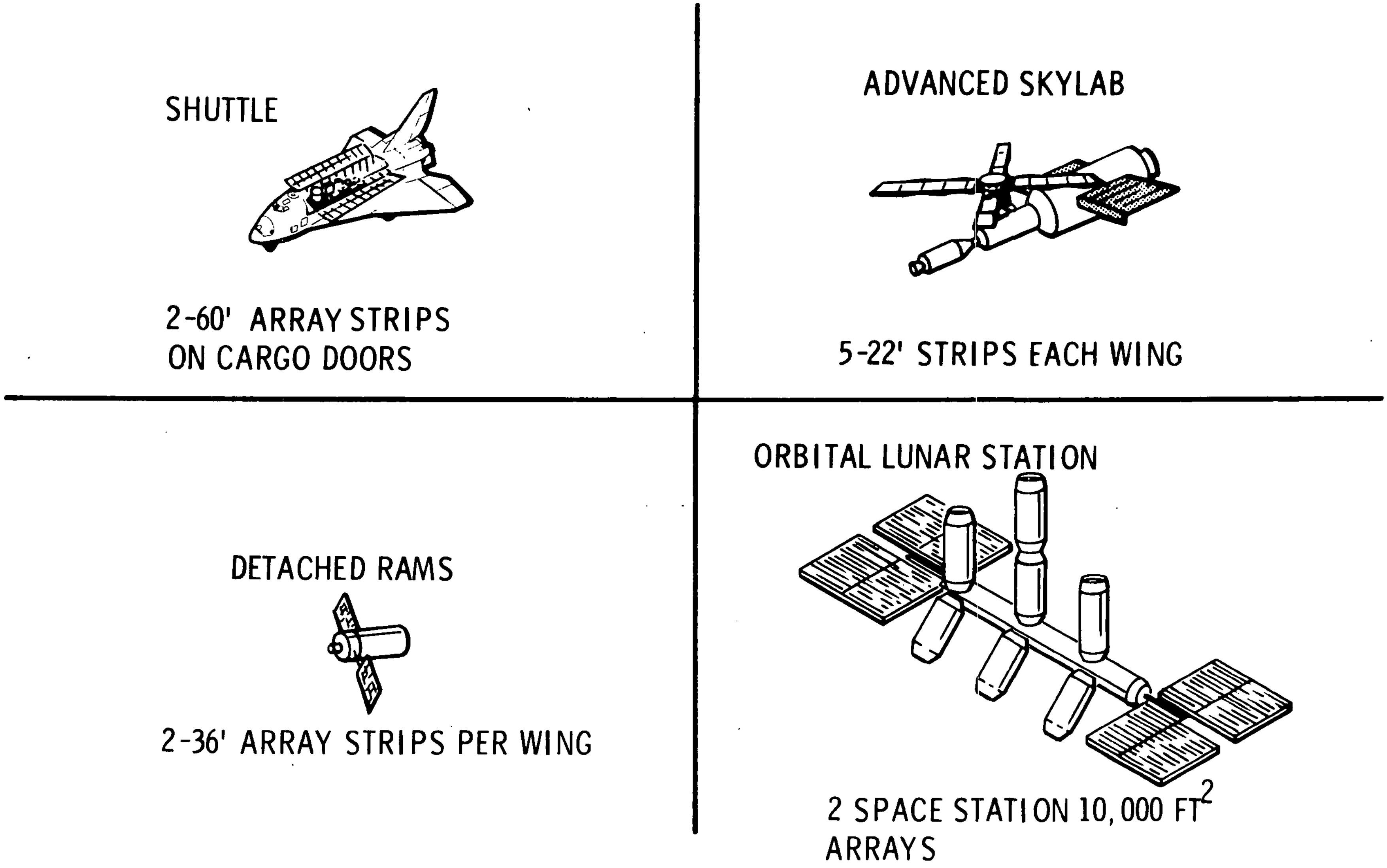


Figure 4.3-1 NASA Current Missions - Solar Arrays

Some general space allocation and geometric factors must be considered in selecting an array design compatible with the shuttle. Three major orientation modes are anticipated with the sortie missions, solar, stellar and earth. For the solar orientation 6 KW of supplementary power could be provided by deploying two array strips, one on each cargo door as shown in Figure 4.3-2. For earth orientation a similar installation may be feasible, but with the array strips displaced further outboard from the cargo doors to minimize shading and facing the opposite direction in that the shuttle would be flying upside down with experiments and sensors oriented with an earth view factor. This installation could be manipulated for better solar orientation in the roll axis, dependent upon the orbit inclination angle, by pivoting the strips about their root attachment to the edge of the cargo door as shown in Figure 4.3-3.

Much interest exists in Dr. Peter Glaser's concept (STEP - Space to Earth Power) which would collect solar energy in space and convert it to a microwave beam for transmission to earth.

A strong incentive exists to demonstrate techniques for low cost mass production of solar arrays to make such a system economically tenable. To demonstrate complete system technological feasibility a STEP experiment has been proposed by Lockheed which would utilize the Space Station Solar Array. Using published conversion efficiencies it is anticipated that 19.5 KW should be available on earth from the 100 KW on orbit input, which is graphically depicted in Figure 4.3-4. In addition to using the SSS array in a demonstration and evaluation experiment, for additional cost effectiveness it has been recommended that the ATS 30' diameter antenna reflector be used.

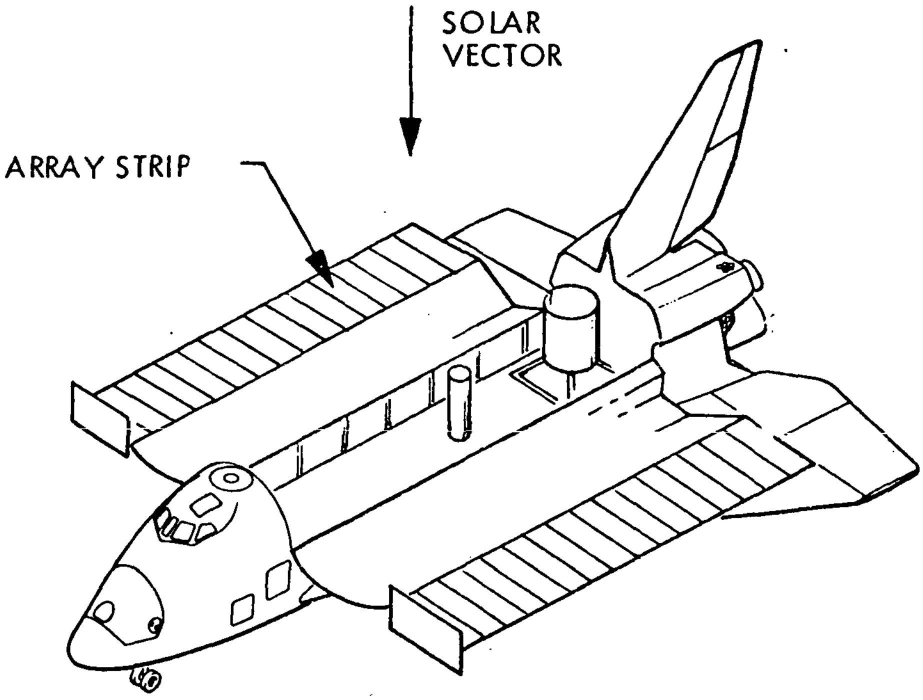


Figure 4.3-2 Array Strips on Shuttle Orbiter

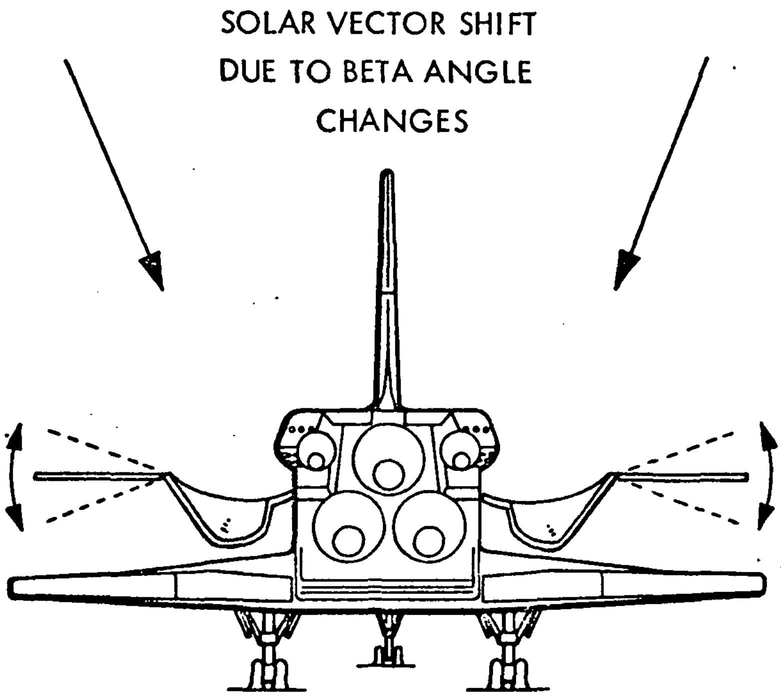


Figure 4.3-3 Adjustment of Array Strips for Shifting Beta Angles

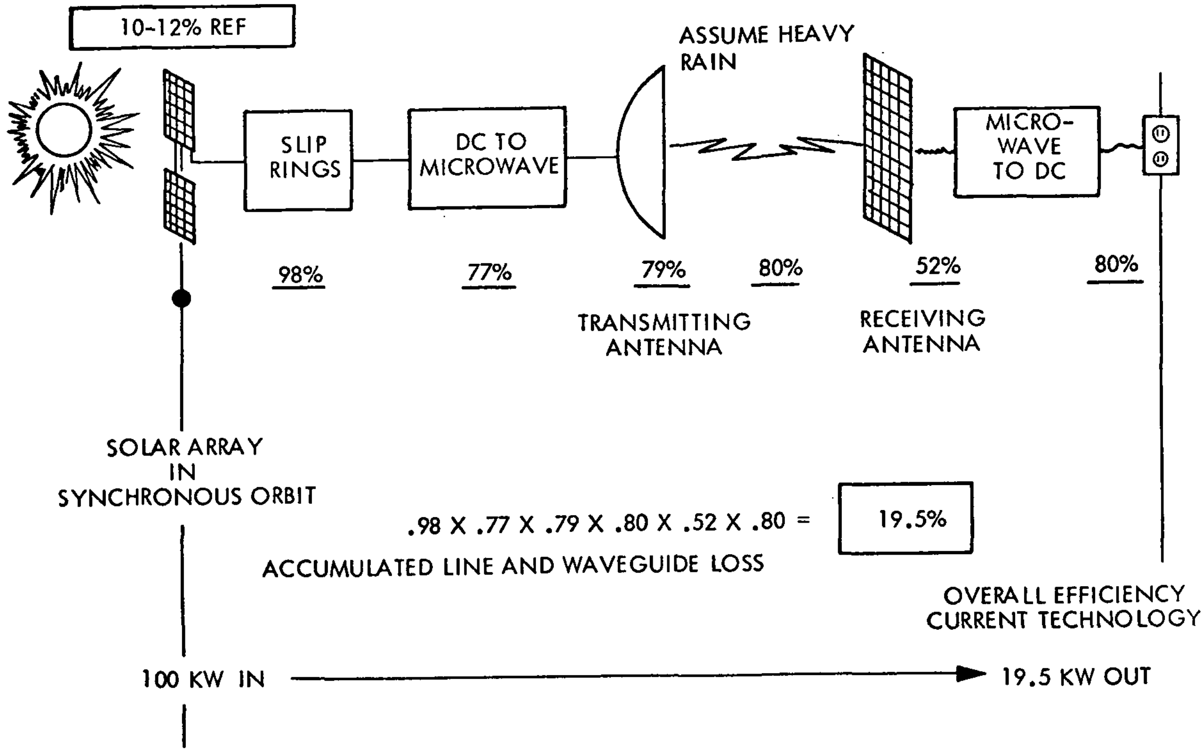


Figure 4.3-4 STEP System Conversion Efficiencies

The purpose of the evaluation was (1) to assess the adequacy of the shuttle cargo hold to accommodate various experiment combinations, (2) to determine experiment deployment and staging feasibility and (3) to identify areas of limiting technology.

Figure 4.3-5 depicts a stowed and a deployed view of the seven configurations.

Based on recently demonstrated large area flexible array technology and the survey reported herein, the following conclusions can be listed. a) Where there are large arrays anticipated and limitations in weight, stowed volume and multiple deployment-retraction requirements exist, lightweight flexible arrays will come into predominance. b) Conservative weight projections, including a 15% weight contingency and use of conventional 12 mil cells and covers, indicate that flexible array systems in the 9.2

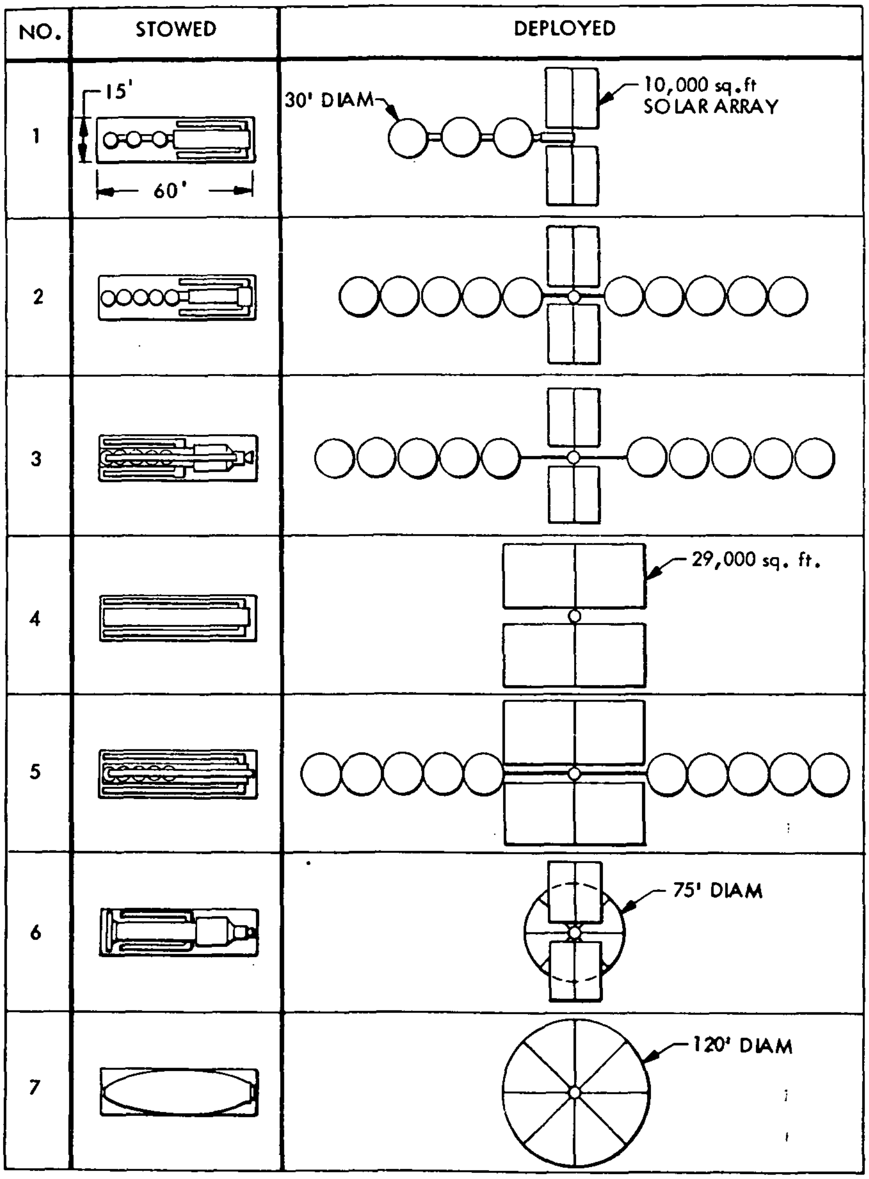


Figure 4.3-5 STEP Experiment Space Allocation Configurations

to 17.5 watt/lb range can be built now for the missions reviewed. c) The technology and designs evolved in the Space Station Array provide candidate components and systems that can find use in future selected applications in the weight and power ranges listed in Table 4.3-2.

TABLE 4.3-2

CONFIGURATION SURVEY SUMMARY

Future Mission	Basic Power	Flexible Array System Wt. - Pwr at Normal Incidence		Space Station Array Compatibility
SHUTTLE	Fuel Cell and Battery	340 lb (154 Kg)	6 KW	Designed for Shuttle Cargo Hold - Mission Extension Achievable with Solar Array
RAM	Solar Array	787 lb (357 Kg) 1232 lb (559 Kg)	7.2 KW 15.8 KW	4 Array Strips with 18-35 Modules Each
STEP	Solar Array	6140 lb (2785 Kg) 18,400 lb (8346 Kg)	100 KW 300 KW	Available Design for Feasibility Experiment - Substrate can be Mass Produced for Ultimate System
ADVANCED SKYLAB	Solar Array	1034 lb (4690 Kg)	17.4 KW	Modules in Peripheral Frame or 5 Array Strips in each Array Beam Fairing
OLS	Solar, Chemical and Nuclear Sources Studied	15,280 lb (6931 Kg)	200 KW	Two Space Station Solar Arrays Plus Regenerative Fuel Cells Could be Considered for Station Power
SPACE BASE	Nuclear and/or Solar	6140 lb (2785 Kg)	100 KW	Space Station Solar Array Compatible with Artificial G Spin-Up Mode

5.0 SIGNIFICANT PROGRAM RESULTS

The primary goal and the most significant result of the Space Station Solar Array Program was demonstrating the feasibility of fabricating and testing extremely large area solar arrays. The more general achievements of the program can be divided into the following three (3) categories which are discussed below:

- o Basic design data generated
- o Component technology developed
- o System technology demonstrated

5.1 Basic Design Data Generated

During the technology evaluation phase, areas were identified where design data needed to develop large area lightweight arrays was not available and as a part of the Technology Evaluation report the tasks necessary to obtain this data were defined and many of them were performed on this program.

The design support tests conducted in early stages of the program yielded needed design data, such as tensile and creep properties of substrate materials and assemblies, structural characteristics of large extendible/retractable truss booms, lubricant properties and slip ring materials characteristics.

The program also produced a comprehensive State-of-the-Art Handbook (LMSC/D159618) for Flexible Solar Array Designers. Specifically, it outlines, for each of the array components and functions such as packaging, deployment substrates, solar cell, solar cell interconnects, cell coverglasses, feeder harness, drives, bearings, motors, brushes and slip rings, the methodology and data required to make appropriate selections and design decisions.

5.2 Component Technology Developed

Based on satisfying the most difficult set of Space Station requirements, as defined by NASA and the two Space Station prime contractors this design study advanced the state of the art and/or produced new technology in the following areas:

- Adhesiveless Flexible Circuit Solar Cell Assembly
  - First application of wraparound solar cells
  - Simple repair at cell level
  - Major improvements in inspectability - first system inspectable after assembly
  - Superior mass producible assembly technique (low cost)

- Controllable, Variable Level Array Tensioning System
  - Provides built in solution to dynamic interaction
  - Negates need to rely on analysis

- First Modular Panel Flexible Solar Array
  - Removable hinge makes simple repair or replacement possible
  - Large savings in spares requirements
  - Easily applied to multiple missions--RAM, BMS, Shuttle, Skylab

- Astro-Boom - Completely Versatile Structure
  - Extendible, retractable truss
  - Load carrying capability during deployment

- 2 x 4 cm Wraparound Solar Cell
  - Now being proposed for DOD hardened applications
  - Advanced development of assemblies by MSFC/Lockheed

5.3 System Technology Demonstrated

- Demonstrated Virtually Unlimited Applicability of Flexible Arrays to Any Projected DOD or NASA Space Mission
  - o Power output of 100 KW (over 10 times any present requirements)
  - o Designed to withstand orbital "g" loads up to 1.0, non-symetric (over 10 times greater than any known orbital load requirement)
  - o Combination of above required many orders of magnitude more rigid structure design than previous system - demonstrated at over 10 watts/lab including all hardware

- Demonstrated 5 ft Dia 2 Axis Tracker
  - o Capability of transferring 100 KW of power
  - o First man-rated, maintainable design
  - o Capable of withstanding 10,000 ft^2 array inertial and docking loads

- Lowest Packaging Volume Ever Developed
  - o 1/2 lowest flexible array
  - o 1/10 lowest rigid array

- Established Unique Techniques for 1 "g" Testing of Large Structures
  - o Structural test techniques
  - o Servo technique for deployment testing

The program successfully accomplished all of the major goals and has provided an updated account of the extensive work accomplished throughout the industry in solar array design and development. All reports published during the contract are listed in Table 5.3-1.

TABLE 5.3-1
SPACE STATION SOLAR ARRAY TECHNOLOGY PROGRAM
DOCUMENTATION LIST - NAS9-11039

NO.	DATE	DOCUMENT NO.	TITLE
1	25 Sep 1970	LMSC A976081	PROGRAM PLAN
2	22 Sep 1970	LMSC A976179 (Rev A)	SUBSTRATE EVALUATION TEST PLAN
3	6 Oct 1970	HANDOUT	PROGRAM REVIEW
4	Dec 1970	LMSC A981486	1ST TOPICAL REPORT - TECHNOLOGY EVALUATION
5	1 Feb 1971	LMSC A984133	TEST EVALUATION PROGRAM PLAN
6	9 Mar 1971	LMSC A976279 (Rev A)	GUIDELINES & REQUIREMENTS DOCUMENT (7 BULLETINS)
7	23 Mar 1971	LMSC A984556	REVISED BASELINE DESCRIPTION (VOL I)
8	6-7 Apr 1971	HANDOUT	BASELINE DESIGN REVIEW
9	23 Apr 1971	LMSC A984556	REVISED BASELINE SELECTION JUSTIFICATION (VOL II)
10	5 May 1971	HANDOUT	MID-TERM REVIEW HANDOUT
11	10 Nov 1971	LMSC/A995719	2ND TOPICAL REPORT - DESIGN/ANALYSIS
12	12 Aug 1971	MSC-04747	PROGRAM STATUS REVIEW
13	Dec 1971	LMSC/A999457	TENSILE AND TEAR TESTS
14	30 Mar 1972	HANDOUT	FINAL REVIEW
15	17 May 1972	HANDOUT	NASA HEADQUARTERS REVIEW
16	Apr 1972	LMSC/D152986	CREEP TESTING
17	Jul 1972	LMSC/D159124	FIRST TOPICAL - UPDATE
18	Sep 1972	LMSC/D159198	TEMPERATURE CYCLING PLAN
19	Oct 1972	LMSC/D153526 MSC 07160	DESIGN SUPPORT & MAJOR HARDWARE TESTS
20	Nov 1972	LMSC/D153469	ALTERNATE APPLICATIONS STUDY
21	Feb 1973	LMSC/D159618 MSC 07161	DESIGN HANDBOOK
22	Jan 1973	LMSC/A995763 MSC 07162	DEVELOPMENT PROGRAM PLAN
23	Jan 1973	LMSC/D159744 MSC-07163	FINAL REPORT - SUMMARY

APPENDIX

PROGRAM DRAWING LIST

DRAWING NO. (SK - - - - - -)	TITLE	SIZE
720005	SOLAR ARRAY SYSTEM/POWER BOOM ASSY.	E
39205 (Ball Bros. Res. Corp.)	MODEL ASSY. ODAPT	J
39206	CYLINDER ASSY, FWD	E
39209	RING	C
39210	RING, ADAPTOR	C
39211	STRINGER	C
39207	CYLINDER ASSY, AFT	E
39208	CYLINDER ASSY, CTR	E
39212	CYLINDER ASSY, OUTER	J
39224	ANGLE BULKHEAD	D
39225	ANGLE BULKHEAD-OUTER	D
39226	ANGLE CORNER	D
39213	HOUSING ASSY-OUTER	J
39214	ATTACH FLANGE	C
39215	OUTER TUBE	C
39216	UPPER RING	C
39217	LOWER RING	C
39218	MOUNTING FLANGE	C
39219	HOUSING ASSY-INNER	J
39220	ADAPTER, TOP	E
39221	ADAPTER, BOTTOM	D
39222	TUBE, INNER	C
39223	SUPPORT ASSY-GEAR	D
39227	FITTING, ATTACH	E
39228	RETAINING, BEARING	C
39229	CLAMP, BEARING	C
39230	FITTING, ADAPTER-FWD	D
39231	SUPPORT ASSY, INNER-FWD	E
39232	SUPPORT ASSY, OUTER-FWD	E
39233	RETAINER BEARING	C
39234	CLAMP, RETAINER	C
39235	RETAINER, BEARING	C
39236	FITTING ASSY, ADAPTER-AFT	E
39237	SUPPORT ASSY, BEARING-AFT	E
39238	SLIP RING ASSY-DUMMY	E
39242	RING, INNER	C
39243	RING, OUTER	C
39239	BALLAST INSTL.	E
39777	BALLAST-POWER SLIP RING	D
40243	BALLAST-DUMMY SLIP RING	C

DRAWING NO. (SK - - - - - -)	TITLE	SIZE
39240	SLIP RING ASSY-POWER	E
39241	SUPPORT ASSY. SLIP RING	E
39244	RING, CONDUCTOR	C
39393	PLATE, INSULATOR	B
39394	SPACER ASSY.	D
39395	BRUSH BLOCK ASSY	E
39396	BRUSH ASSY	C
39397	HOUSING	D
39398	BRACKET, BRUSH BLOCK	E
39399	ANGLE ASSY, TERMINAL	C
39400	BOARD ASSY, INSULATOR	C
38957	SLEEVE	D
39774	ADJUSTMENT SCREW	C
39401	BRACKET ASSY, INNER DRIVE	E
39402	BRACKET ASSY, OUTER DRIVE	E
39767	SHIM, SUPPORT RING	B
39768	SHIM, BRUSH BLOCK	C
39769	STIFFENER	C
39770	COVER	C
39771	BRACKET ASSY, INNER DRIVE	E
39772	PLUG	C
39773	SHIELD	C
38060	BEARING ASSY	B
38165	BEARING, RADIAL	B
38854	GEAR, RING-OUTER GIMBAL	C
38871	GEAR, RING-INNER GIMBAL	C
38518	DRIVE ASSY, INNER GIMBAL	J
38507	HOUSING, OUTER	J
38508	HOUSING, INNER	E
38509	COVER, MOTOR	D
38510	SHAFT, MOTOR	D
38511	GEAR, SPUR	D
38512	RETAINER, INNER	D
38513	RETAINER, CENTER	D
38514	RETAINER, BEARING	D
38515	RETAINER, BEARING	D
38516	RETAINER	D
38517	RETAINER	D
38535	RETAINER, BEARING	D
38575	RETAINER, OUTER	D
38656	RETAINER	D
38937	RESERVOIR, RADIAL BEARING	C
38939	RESERVOIR, MOTOR	C
38940	RESERVOIR, SPLINE	C
39160	COVER, OUTPUT SHAFT	C
39161	RESERVOIR, HARMONIC DRIVE	C
39162	MOUNTING RING, RESERVOIR	D
38938	RESERVOIR, MOTOR BEARING	C

DRAWING NO. (SK - - - - - -)	TITLE	SIZE
39269	DRIVE ASSY, OUTER GIMBAL	E
39260	SHAFT, IDLER	C
39261	RETAINER, IDLER	C
39262	SHAFT, MOTOR	D
39263	COVER, BOTTOM	D
39264	GEAR, INPUT	C
39265	RETAINER, BEARING	C
39266	GEAR, IDLER	C
39267	COVER, MOTOR	D
39268	HOUSING, OUTER	E
38960	TEST FIXTURE, ODAPT SLIP RING	J
38951	RETAINER, IDLER SHAFT	C
38952	RETAINER, OUTPUT SHAFT	C
38953	PLATE, COVER	D
38954	BRACKET, MOUNTING	D
38955	SCREW, ADJUSTMENT	D
38956	BRUSH ASSY	D
38957	SLEEVE	D
38958	HOUSING ASSY, BRUSH	E
38959	HOUSING, ODAPT SLIP RING	E
38963	SLIP RING, POWER	D
38966	PLATE, SLIP RING	D
38973	SHAFT, IDLER	C
38974	SHAFT, INPUT	C
38975	SHAFT, OUTPUT	D
38976	GEAR, OUTPUT SHAFT	C
38977	GEAR, IDLER SHAFT	C
38978	GEAR, IDLER SHAFT	C
38987	SLIP RING, SIGNAL	D
38988	SLIP RING, SIGNAL	D
38991	CLAMP, SLIP RING	C
38996	LOCKNUT, MOTOR	C
38997	LOCKNUT, BEARING	C
38998	CLAMP, SLIP RING	C
38999	BUSHING, INSULATOR	C
39000	PLATE, BRUSH	D
39001	PLATE, LOWER BEARING	D
39005	RETAINER, INPUT SHAFT	C
39008	RESERVOIR, ODAPT SLIP RING	C
39009	INSULATOR, BRACKET	C
39010	INSULATOR, SLIP RING	D
39145	BRUSH ASSY, SIGNAL	C
39146	BRUSH BLOCK SIGNAL	D
39150	BRACKET, BRUSH BLOCK	C
39151	WASHER INSULATING	C
39152	SCREW, ADJUSTMENT	C
39680	SPACER, SLIP RING	B
39681	RETAINER, SLIP RING	B

DRAWING NO. (SK - - - - - -)	TITLE	SIZE
Additional Drawings		
39141	TEST FIXTURE, ODAPT	J
36932	TEST PLAN EVALUATING POWER - TYPE BRUSH/SLIP RING COMBINATIONS, ODAPT	A
38024	TEST PLAN FOR EVALUATING THE EFFECT OF VACUUM STORAGE ON TORQUE OF OILED BALL BEARINGS	A
38042	TEST PROGRAM PLAN MAJOR HARDWARE DEPT	A
38127	TEST PLAN FOR EVALUATING WEIGHT LOSS RATES OF OILS	A
38609	TEST PLAN FOR EVALUATING THE LUBRICATION PROPERTIES OF OILS AND GREASES FOR ODAPT (FOUR BALL TEST)	A
39203	LUBRICATION SPEC. PRELIM: ODAPT GIMBAL BEARINGS	A
SMS1808	GREASE FOR PINION AND RING DRIVE GEARS ODAPT	A
40353	LUBRICATION PROCEDURE GIMBAL BEARINGS - ODAPT MODEL	A
720006	WING ASSY	E
2600 (Astro Res. Corp.)	OUTLINE DRAWING, EXTENDIBLE MAST - LMSC MAST	D
2600-001	SECTIONAL VIEW OF LMSC CANISTER	D
2600-002	LMSC MAST ASSY	D
2600-003	BATTEN ASSY	D
2600-004	BATTEN	C
2600-005	BATTEN FITTING	C
2600-006	LONGERON	C
2600-007	DIAGONAL, LATCH	B
2600-008	FORK-END, MODIFIED	B
2600-009	DIAGONAL, PLAIN	B
2600-010	FORK-END, MODIFIED	B
2600-011	ROLLER, STEEL	B
2600-012	ROLLER, ALUMINUM	B
2600-013	CORNER FITTING	C
2600-014	ADAPTER	B
2600-015	SPACER	B
2600-016	ROD, LATCHING	B
2600-017	CORNER FITTING LATCHING ASSY	C
2600-018	CORNER FITTING	C
2600-019	LATCH PLATE, OUTSIDE	B
2600-020	LATCH PLATE, INSIDE	B
2600-021	RIVET	B
2600-022	SPACER	B
2600-023	RIVET, HINGE	B
2600-024	RIVET, SHOULDER	B

DRAWING NO. (SK - - - - - -)	TITLE	SIZE
2600-025	WASHER, ROLLER	B
2600-026	WASHER	B
2600-027	ROD	C
2600-028	SPACER	B
2600-029	NUT, PIVOT	B
2600-030	RETAINER	B
2600-031	MAST BASE	D
2600-032	ROLLER, BOTTOM	B
2600-033	LATCHING MECHANISM	D
2600-034	CANISTER	E
2600-035	RING, LOWER BEARING	D
2600-036	FLANGE, MOUNTING	D
2600-037	RING, MOUNTING	C
2600-044	TRANSITION GUIDE	D
2600-038	MAST GUIDE ASSY	E
2600-039	BACKING STRIP	B
2600-040	GUIDE RAIL	B
2600-041	BRACKET, LATCHING MECH.	C
2600-052	INSERT SECTION	B
2600-053	VERTICAL GUIDE	B
2600-042	BALL SCREW	B
2600-043	BALL NUT	C
2600-044	TRANSITION GUIDE	D
2600-045	CAM DRUM	C
2600-050	INSERT, INSIDE	B
2600-051	INSERT, TOP	B
2600-046	RING, UPPER BEARING	D
2600-047	BRACKET, MOTOR MOUNTING	D
2600-048	DRIVE GEAR	B
2600-049	CONTROL CIRCUIT	B

Other Documents

	DESIGN CHANGE RECOMMENDATIONS FOR LMSC/ASTRO EXTENDIBLE BOOM SYSTEM	A
	DESIGN/ANALYSIS DATA AND TEST SUMMARY SHEETS FOR SIX-BAY SEGMENT OF LMSC EBS	A
720000	I. S. A./O. S. A. ASSY	E
720001	ART "G" TENSION MECH.	E
730019	CYLINDER ASSY	D
730020	BASE	D
730021	CAP	D
730022	SHAFT	D
730023	ADAPTER	D
730024	BASE	D
730025	SPACER	D
730027	PULLEY	C
730038	SHAFT	C
730106	CLAMP	C
730118	HANGER	C

DRAWING NO. (SK - - - - - -)	TITLE	SIZE
720002	GUY TAPE MECH.	E
730018	OUTPUT DRUM	C
730039	OUTPUT FASTENER	C
730041	PULLEY ASSY	C
730119	PULLEY	C
730120	HUB	C
730043	PULLEY GUIDE	C
730044	BRACKET	C
730045	BASE PULLEY BRACKET	C
730046	SHAFT CONTAINER	C
730047	SHAFT	C
730048	BRACKET	C
730049	WASHER	C
730050	WASHER	C
730051	CHANNEL BASE	C
730052	ANCHOR FITTING	C
730053	UNIVERSAL BRACKET	C
730054	UNIVERSAL SPACER	C
730055	UNIVERSAL FITTING	C
730056	TAPE LOOP	C
730073	TAPE LOWER ANCHOR	C
720003	ZERO "G" TENSION MECH	E
730061	MOTOR REEL ASSY	E
730072	BRACKET	C
730075	DRAG BRAKE ASSY	C
730071	BRAKE HOUSING	C
730086	BRACKET	C
730083	PULLEY	C
730084	SHAFT	C
730085	MOUNT	C
730062	IDLER PULLEY ASSY	E
730076	PULLEY	C
730077	PULLEY BRACKET	C
730078	PULLEY SHAFT	C
730063	BRACKET	C
730064	BRACKET	C
730065	PLUG	C
730066	TAPE CONNECTOR	C
730067	TAPE FITTING	C
730070	NEGATOR ASSY	D
730079	NEGATOR SUPPORT	D
730080	NYLON PULLEY	C
730081	BRACKET	C
730082	SHAFT	C

DRAWING NO. (SK - - - - - -)	TITLE	SIZE
720009	STRIP PACKAGING ASSY	E
720010	GUIDE WIRE MECH ASSY	E
730009	WHEEL SUPPORT BRKT	C
730010	WHEEL	C
730011	GUIDE WIRE RETAINER	C
730012	MOUNTING FITTING	C
730013	END PLATE	C
730014	WIRE/TAKE-UP WHEEL	C
730015	NEGATOR BASE	C
730029	OUTPUT DRUM FASTENER	C
730003	COVER ASSY	E
730006	BASEPLATE ASSY	E
730032	STRIP INBD CROSSMEMBER	E
730033	CROSSMEMBER	C
730034	CABLE ATTACH FITTING	C
730035	CABLE ATTACH NUT	C
730036	CABLE ASSY	C
730092	CABLE END	C
730091	PLUG	C
730060	STRIP ASSY CONTAINER	E
730059	FITTING	D
730124	GUIDE	C
730087	SPRING	C
730088	SPRING	C
730089	FITTING	C
730090	FITTING	C
730093	LOWER PIVOT SUPPORT	C
730094	LOCKING PIN	C
730095	LOCKING PIN EXTENSION	C
730096	BELL CRANK	C
730097	BELL CRANK BRACKET	C
730098	PULL ROD ASSY	C
730099	PULL ROD RETAINER	C
730100	LOCKING PIN RETAINER	C
730102	LOWER PIVOT ASSY	E
730101	LOWER PIVOT FITTING	C
730103	RETRACTOR BAR	D
730104	PROTECTIVE PAD ASSY	E
810002	STRIP ASSY & TYP HARN. INSTALLATION	J
810000	112V FLEX. CIRC. ELEMENT	E
810001	JOINT	D
810005	OUTBOARD LEADER	E
730031	STRIP LEADING EDGE MEM.	D
810006	INBOARD LEADER	E
810007	LOCKING BAR	C
810008	MODULE ASSY	E

DRAWING NO. (SK - - - - - -)	TITLE	SIZE
7106000	SOLAR CELL	D
7106003	FILTER COVER	B
810003	FILTER COVER	C
810009	HARNESS ASSY	E
730004	CAP ASSY	E
730000	CAP RING	C
730002	TOP CAP RING	C
730005	ADAPTER ASSY	E
730001	ADAPTER BASE RING	C
730007	CAP/O. S. A. LOCK ASSY	D
730008	LOCKING NUT	C
730016	PIVOT PIN - O. S. A.	C
730017	PIVOT PIN - I. S. A.	C
730028	SPACER	C
730030	WASHER	C
730037	COMPENSATOR BAR	C
730057	COVER RELEASE	E
730058	FITTING	C
730109	ART "G" PULLEY CABLE ASSY	D
730108	PULLEY RETAINER ASSY	C
730110	PULLEY COVER	C
730111	PULLEY HOUSING	C
730112	SPACER	C
730113	ART. "G" CABLE GUIDE ASSY	C
730107	PULLEY BRACKET	C
730114	HOLD DOWN PLATE	C
730115	MOUNTING BRACKET	C
730116	ANCHOR FITTING	C
730117	ART. "G" TENSION CABLE	C
730123	GUY TAPE UPPER DEFLECTOR	C
730125	GUIDE FITTING - O. S. A.	C
730126	GUIDE FITTING - I. S. A.	C
810010	WIRING SCHEMATIC	D