

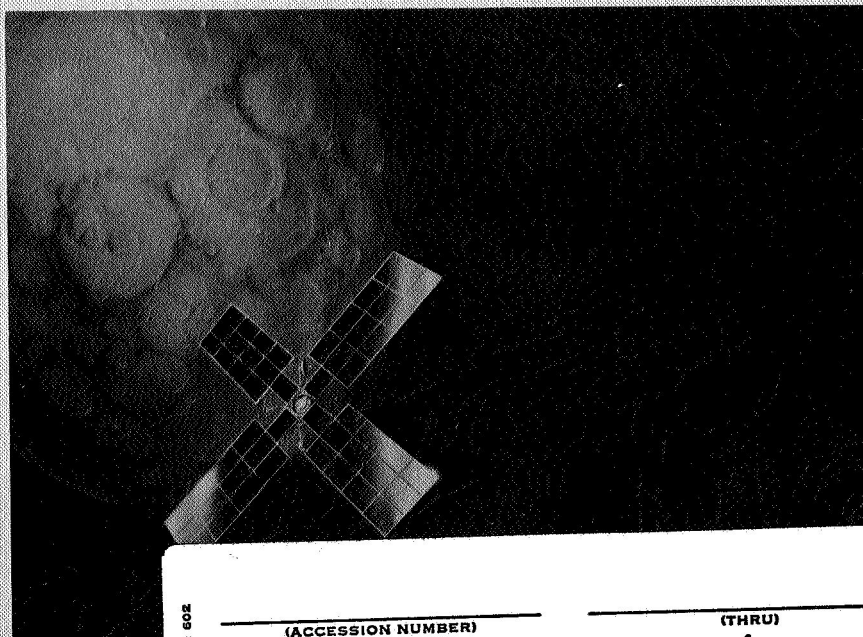
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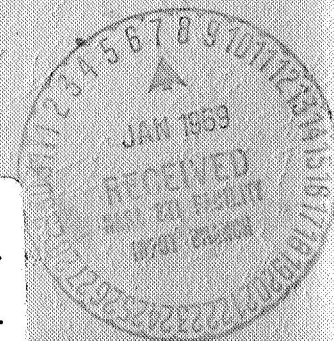
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THE **BOEING** COMPANY - SPACE DIVISION

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FINAL REPORT—PHASE II

Prepared for
Jet Propulsion Laboratory
Under Contract
JPL 951934

**CASE F
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LARGE AREA SOLAR ARRAY

October 1968

Prepared by
THE BOEING COMPANY
Aerospace Group
Space Division
Seattle, Washington

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ABSTRACT

This is the Large Area Solar Array, Phase II, Final Report required by Jet Propulsion Laboratory Contract 951934. Phase II is the second step in the Large Area Solar Array program and further develops fabrication procedures and design of a 1,250-square-foot section of the Large Area Solar Array. The results of this effort include: (1) the activity and results of the development testing of mechanism components for tiedown, release, and deployment of the array; (2) the fabrication methods and test activity and results for the 8- by 13-foot No. 1 Main Subpanel; (3) the development, test and analyses of the simulated zero g deployment fixture, and the deployment testing with full-scale dummy subpanel and prototype deployment mechanisms; (4) the preliminary design for Phase III GSE; (5) the production and usage of transportation handling and test equipment; (6) the development of deployment demonstration equipment; and (7) the electrical degradation and vibration tests and analyses of a 4-foot-square sample panel. The period covered by this report is from June 30, 1967, through October 31, 1968.

CONTENTS

	<u>Page</u>
1.0 INTRODUCTION	1-1
1.1 Purpose and Objectives	1-1
1.2 Background and Description	1-2
1.3 Organization of Document	1-3
2.0 SUMMARY	2-1
2.1 Task Objectives	2-1
2.2 Array Design and Analysis	2-1
2.2.1 Description of the Array	2-2
2.2.2 Phase II Design Development	2-17
2.2.3 Design Analysis of MTA-4	2-18
2.3 GSE Design Summary	2-26
2.4 Manufacturing Summary	2-27
2.5 Systems Test Summary	2-29
2.5.1 Tests of Small Component Mechanisms	2-30
2.5.2 GSE Test Summary	2-30
2.5.3 MTA-3 Mechanism and Aerodynamic Tests	2-32
2.5.4 Vibration Test of Subpanel No. 1	2-34
2.5.5 SCS-43 Subpanel Vibration Tests	2-34
2.6 Quality Assurance Summary	2-36
2.7 Product Assurance Summary	2-39
2.8 Cost and Schedules	2-40
2.9 Materiel Summary	2-41
2.10 Program Conclusions and Recommendations	2-42
3.0 SMALL COMPONENT MECHANISMS TESTING	3-1
3.1 Development Tests of Deployment Mechanisms	3-1
3.1.1 Description of SCM-1 and SCM-2 Test Articles	3-4
3.1.2 Description of SCM-1 and SCM-2 Test Articles	3-4
3.1.3 Outside Procurement and Production	3-6
3.1.4 Quality Assurance	3-7
3.1.5 Conduct of SCM-1 and SCM-2 Test	3-8
3.1.6 Results and Analysis of Tests	3-12
3.1.7 Conclusions	3-22
3.2 Development of Boost Tiedown Panel Release	3-24
3.2.1 Description of SCM-30 Test Articles	3-25
3.2.2 Design of the SCM 30-2 Test Fixtures	3-25
3.2.3 Conduct of Panel Tiedown Test (SCM 30-2)	3-27
3.2.4 Results and Analysis of Tests (SCM 30-2)	3-31
3.2.5 Explosive Actuated Cable Cutters	3-33
3.2.6 Conclusions and Recommendations (SCM 30-2)	3-36
3.2.7 Design of the SCM 30-3 Test Hardware	3-37
3.2.8 Conduct of Center Tie Release (SCM 30-3)	3-40
3.2.9 Results and Analysis of SCM-30-3 Tests	3-41
3.2.10 Conclusions and Recommendations (SCM 30-3)	3-45

CONTENTS (CONT)

	<u>Page</u>
4.0 MAJOR TEST ARTICLE 1 (MTA-1) DEVELOPMENT (TASK 3)	4-1
4.1 MTA-1 Design and Analysis	4-1
4.1.1 Description of MTA-1 Test Article	4-1
4.1.2 Analysis of MTA-1 Vibration Characteristics	4-4
4.1.3 Design of Simulated Power System	4-5
4.2 Fabrication of MTA-1	4-7
4.2.1 MTA-1 Manufacturing Plan	4-9
4.2.2 Fabrication of Details	4-12
4.2.3 Structure Subassembly Bonding	4-16
4.2.4 Fabrication of Simulated Power System	4-21
4.2.5 Structure Final Assembly Bonding	4-24
4.2.6 Repair of MTA-1	4-26
4.2.7 MTA-1 Final Assembly	4-38
4.3 Quality Assurance	4-41
4.4 Vibration Testing of MTA-1	4-44
4.4.1 Design and Fabrication of the Vibration Test Fixture	4-44
4.4.2 Test Conduct and Description	4-45
4.4.3 Test Results and Analysis	4-46
4.5 Solar Cell Deterioration	4-51
4.6 Radiation Damage Analysis	4-53
5.0 GROUND DEPLOYMENT TESTING (TASK 4)	5-1
5.1 Test Objectives	5-1
5.2 MTA-3 Test Article	5-2
5.2.1 MTA-3 Structure	5-2
5.2.2 MTA-3 Mechanisms	5-3
5.3 Simulated Zero-Gravity Deployment System Description	5-7
5.3.1 Mechanical Systems Design	5-7
5.3.2 Instrumentation Systems Design	5-7
5.3.3 Fabrication and Assembly	5-10
5.3.4 Installation and Systems Integration	5-11
5.3.5 Calibration	5-14
5.4 Test Conduct and Observation	5-17
5.4.1 Test Setup and Conduct	5-17
5.4.2 Value of MTA-3 as an Operational Mockup	5-19
5.4.3 Behavior of Mechanisms	5-20
5.4.4 Behavior of the Deployment Motor and Cable Drive System	5-23
5.5 Test Analysis and Results	5-25
5.5.1 Analytical Approach	5-25
5.5.2 Results of Deployment Calculations with Nominal Parameters	5-26
5.5.3 Approach to Resolution of Measured and Calculated Deployment Shape Discrepancies	5-26
5.5.4 Analysis of Aerodynamic Test Data	5-31
5.5.5 Initial Velocity Measurement	5-31

CONTENTS (CONT)

	<u>Page</u>
5.5.6 Diagonal Strut Loading	5-32
5.5.7 Application of Phase II Results to Phase III Testing and Analyses	5-33
5.6 Conclusions	5-33
5.6.1 Conclusions Concerning Mechanisms	5-33
5.6.2 Conclusions Concerning Deployment Verification Tests and Analysis of Test Data	5-37
5.6.3 Conclusions Concerning Deployment Test Equipment	5-37
6.0 HANDLING EQUIPMENT AND PHASE III PRELIMINARY DESIGN (TASK 5)	6-1
6.1 Equipment Deletions	6-1
6.2 Phase II Handling Equipment	6-2
6.3 Phase III Preliminary Designs	6-5
6.3.1 Deployment Equipment---Overhead Suspension	6-5
6.3.2 Deployment Equipment---Subpanel Unit Deployment Fixture	6-7
6.3.3 Deployment Demonstration Equipment	6-8
6.3.4 Environmental Test Equipment	6-14
6.3.5 Handling and Transportation Equipment	6-17
6.4 Failure Mode and Effect Analysis (FMEA) of GSE	6-17
7.0 SCS-43 SUBPANEL VIBRATION TEST PROGRAM	7-1
7.1 Description of Test System	7-2
7.2 Vibration Test	7-4
7.2.1 Results and Analysis of Tests	7-8
7.2.2 Conclusions	7-9
7.3 Power Degradation Tests	7-12
7.4 Quality Assurance	7-15
8.0 MATERIEL PROGRAM SUPPORT	8-1
8.1 Procurement Activities	8-1
8.2 Supplier and Program Managements	8-4
8.3 Materials Management	8-8
9.0 PROGRAM EVALUATION SUMMARY	9-1
9.1 Conclusions	9-1
9.1.1 Twenty-Watt-Per-Pound Goal	9-1
9.1.2 Fabrication and Assembly Results	9-1
9.1.3 Test Results	9-2
9.1.4 MTA-3 Configuration	9-2
9.1.5 Analysis	9-3
9.1.6 New Technology Disclosures	9-3
9.2 Recommendations	9-4
9.2.1 Design of Solar Array	9-4
9.2.2 Fabrication and Assembly Recommendations for Phase III	9-5

CONTENTS (CONT)

	<u>Page</u>
9.2.3 Test Recommendations	9-6
9.3 Phase III Program Evaluation Summary	9-7
10.0 BIBLIOGRAPHY AND DEFINITIONS	10-1
10.1 Major Program Documents	10-1
10.2 Additional Program Documents	10-1
10.3 Miscellaneous References	10-2
10.4 Symbols, Terms, and Definitions	10-3

ILLUSTRATIONS

<u>Figure</u>		<u>Page</u>
1-1	Deployed Array	1-1
1-2	LASA Three-Phase Program	1-5
2-1	LASA Phase II Program Tasks	2-3
2-2	Large Area Solar Array Configuration (MTA-4)	2-5
2-3	Boost Tiedown and Release System	2-10
2-4	Deployment Mechanisms	2-10
2-5	Deployment Sequence	2-12
2-6	Shutoff Switch Installation	2-12
2-7	Array Bus System	2-15
2-8	Dynamic Analysis Modes	2-19
2-9	Space Deployment Loads	2-24
2-10	Deployment Verification Equipment	2-26
2-11	MTA-1 Fabrication	2-28
2-12	Ground Support Equipment Tests	2-31
2-13	Effect of Substrate on MTA-3 Deployment Test	2-33
2-14	SCS-43 Subpanel	2-35
2-15	Total LASA Program Rejections Versus Functions	2-38
2-16	LASA Phase II Cost Performance	2-41
2-17	LASA Phase II Schedule Performance	2-43
2-18	Materiel Procurement	2-45
3-1	Test Summary	3-2
3-2	SCM Deployment Components	3-3
3-3	Test Setup SCM-1 & -2	3-3
3-4	SCM-1 Test Article Fabrication	3-5
3-5	Drive Motor	3-5
3-6	SCM-2 Test Article Fabrication	3-5
3-7	SCM-1 and SCM-2 Test Fixture	3-6
3-8	Array Deployment Drive	3-7
3-9	Rotary Damper	3-9
3-10	Undeployed Hinge and Four-Bar Linkage Latch	3-11
3-11	Deployed and Latched Auxiliary Hinge	3-11
3-12	Vacuum Time-History in Vacuum	3-13
3-13	Cable Slippage Test	3-15
3-14	SCM-1 Cable Slippage Measurement	3-15
3-15	Total System Friction Test	3-16
3-16	Midpoint Deflection of Subpanel No. 3	3-16
3-17	SCM-1 Test Results---Hinge Friction on Subpanel No. 2	3-17
3-18	Friction Time-History---SCM-1	3-17
3-19	Overcenter Latch Misalignment	3-19
3-20	Deflected Auxiliary Subpanel---SCM-2	3-19
3-21	Composite of SCM-2 Hinge Frictions	3-20
3-22	Damper Temperature	3-20
3-23	Vacuum Test Time-History---SCM-2	3-22
3-24	Deployment Time-History---SCM-1	3-23
3-25	Motor Temperature Time-History---SCM-1	3-23
3-26	SCM-30 Tiedown and Release	3-26
3-27	SCM 30-2 Test Elements	3-27
3-28	SCM 30-3 Test Elements	3-27

ILLUSTRATIONS (CONT)

<u>Figure</u>		<u>Page</u>
3-29	SCM 30-2 Retest Test Setup	3-28
3-30	SCM 30-2 Tiedown Release Test Details	3-28
3-31	SCM 30-2 Retest Visicorder Setup	3-29
3-32	SCM 30-2 Retest Camera Setup	3-30
3-33	Instrumentation Block Diagram	3-32
3-34	Cable Position 0.04 Seconds After Release---Run 1	3-34
3-35	Sequential Cable Positions---Run 1	3-34
3-36	Cable Position 0.09 Seconds After Release---Run 2	3-35
3-37	Sequential Cable Positions---Run 2	3-35
3-38	SCM 30-3 Test Setup	3-37
3-39	Center Spar Clamp Assembly---SCM 30-3 Test Article	3-39
3-40	Modified Cable Grip Mechanism	3-39
3-41	Modified Clamp Assembly Test Setup	3-41
3-42	Nomex Cable Load Decay Characteristics	3-43
3-43	Long-Term Load Decay	3-44
3-44	Nomex Cable Clamp Friction Coefficient	3-45
3-45	Wedge Angle Versus Clamping Load	3-46
3-46	Cable Friction Coefficient Versus Clamping Load	3-47
4-1	Subpanel No. 1 Assembly (MTA-1)	4-3
4-2	Module to Bus Connection	4-4
4-3	MTA-1 Live Cell Locations (Darkside Shown)	4-6
4-4	MTA-1 Vibration Test Setup	4-8
4-5	Assumed Volumes of Air Mass	4-8
4-6	Major Test Article No. 1 (MTA-1)	4-9
4-7	Exploded View (MTA-1)	4-10
4-8	Manufacturing Plan Schematic	4-11
4-9	Hot-Form Beryllium Channel	4-13
4-10	Hot-Press Loading of Channel Die	4-15
4-11	Channel Trim Fixture	4-15
4-12	Welded Titanium Fitting Fabrication Process	4-17
4-13	Stick Assembly	4-18
4-14	Frame Assembly Process	4-19
4-15	Substrate Assembly	4-20
4-16	Solar Cell Module	4-22
4-17	Solar Cell Bonding Roller	4-22
4-18	Seven Cell Soldering Fixture	4-23
4-19	Module Soldering	4-23
4-20	Final Structure Assembly Setup	4-25
4-21	Structure Final Assembly Bond	4-25
4-22	MTA-1 Intercostal Bond Failure	4-27
4-23	MTA-1 Conditions Repaired	4-31
4-24	MTA-1 Repair Bond Disassembly	4-30
4-25	Repair Bond Setup	4-34
4-26	MTA-1 Repair	4-36
4-27	Repaired MTA-1	4-37
4-28	Module Bonding	4-39
4-29	Solar Cell Module and Bus Bar Installations	4-40
4-30	MTA-1 Rejections vs Functions	4-42

ILLUSTRATIONS (CONT)

<u>Figure</u>		<u>Page</u>
4-31	MTA-1 Rejections vs Material In Work	4-43
4-32	MTA-1 Installed in Vibration Test Fixture	4-45
4-33	MTA-1 Test Setup	4-46
4-34	Effect of Pressure on Dynamic Magnification Factor Q	4-47
4-35	Maximum Acceleration and Stress Measurements	4-48
4-36	Solar Cell Deterioration	4-52
5-1	Weight/Balance of MTA-3	5-3
5-2	Comparison of MTA-3 and MTA-4 Bending Frequency	5-5
5-3	Deployment Mechanisms	5-6
5-4	Deployment Fixture Mechanical Operation	5-8
5-5	Instrumentation System	5-9
5-6	Erection of Deployment Fixture Structure	5-12
5-7	Hinge Misalignments	5-13
5-8	Fairing and Waxing Air Bearing Track Joints	5-15
5-9	Deployment Angle Calibration	5-16
5-10	Closing Velocity Calibration Sequence	5-18
5-11	Assembly of MTA-3	5-20
5-12	Diagonal Strut	5-22
5-13	Deployment Cable Deformations	5-24
5-14	Comparison of Panel II, Run 4, Type 1, with Nominal and Sigma Resistance Calculated Deployment	5-28
5-15	Comparison of Panel III, Run 5, Type 2, with Nominal and Sigma Resistance Calculated Deployment	5-28
5-16	Characteristic Deployment Regions	5-30
5-17	Sequential Deployment Demonstration	5-32
5-18	Main Hinge Deployment Interference	5-35
5-19	Panel 5A and 5B Release Bracket Modification	5-35
5-20	Interference Problems	5-36
6-1	Handling Equipment	6-3
6-2	Handling Equipment Tests	6-4
6-3	Phase III Test Article Flow Diagrams	6-6
6-4	Deployment Equipment---Overhead Air Bearings	6-7
6-5	Auxiliary Subpanel Deployment Equipment	6-8
6-6	Deployment Demonstration Equipment	6-9
6-7	SCG-4 Test Setup	6-11
6-8	Auxiliary Subpanel---MTA-3	6-12
6-9	Auxiliary Subpanel Deployment Demonstration Test	6-13
6-10	MTA-4 Thermal-Vacuum Test Fixture	6-15
6-11	Dummy Solar Panel Assembly	6-16
6-12	Vibration Test Fixture and Dummy Panels	6-16
6-13	Phase III Handling Equipment	6-18
6-14	Typical Usage Diagram	6-19
6-15	Typical FMEA Data Sheet	6-20
7-1	Test Setup---Clamped Configuration	7-2
7-2	Strain Gage Locations---SCS-43	7-3
7-3	Electrical Schematic---SCS-43	7-3
7-4	Test Setup---Pinned Support	7-5
7-5	Flexural Support	7-5

ILLUSTRATIONS (CONT)

<u>Figure</u>		<u>Page</u>
7-6	SCS-43 Instrumentation	7-6
7-7	SCS-43 Accelerometer Locations	7-6
7-8	X-200 Solar Simulator with Open Shutter	7-8
7-9	Measured Accelerometer Response in Air	7-10
7-10	Measured Accelerometer Response in Vacuum	7-10
7-11	Effect of Pressure on Dynamic Coefficient Q	7-11
7-12	Isosolar Plot of the X-200 Solar Simulator	7-13
7-13	Temperature Profile of SCS-43 Test Panel Power Output Tests A and B	7-13
7-14	Damaged Cells, Live Cell Area---SCS-43 Test Panel	7-14
7-15	Power Degradation Test---SCS-43 (Test A)	7-16
7-16	Power Degradation Test---SCS-43 (Test B)	7-17
7-17	Intercostal Cap Crack	7-18
7-18	Silver Mesh Interconnector Failure	7-19
8-1	Materiel Procurement by Percent of Total Dollars Spent	8-1
8-2	Beryllium Processing Operations	8-3
8-3	Erection of Deployment Structure	8-5
8-4	Motor/Cable Drum Assembly	8-6
9-1	LASA Summary Test Flow---Phase III	9-9

TABLES

<u>Table</u>		<u>Page</u>
II-1	MTA-1 Weight Status	2-8
II-2	MTA-4 Weight Summary	2-9
II-3	Power Generation Summary	2-14
II-4	LASA to Spacecraft Net Clearances	2-20
II-5	Summary of Dynamic Loads	2-21
II-6	MTA-1 Stress Analysis Summary---Beryllium	2-22
II-7	Stress Summary for Subpanels---2 through 5 Members	2-23
II-8	Minimum Margin of Safety---Hinges, Latches, and Diagonal Strut	2-25
II-9	Vibration Test Results	2-36
IV-1	Summary of Analysis of MTA-1 Vibration Characteristics	4-7
IV-2	MTA-1 Failure Committee Recommendations	4-33
IV-3	MTA-1 Repair Techniques	4-34
IV-4	Measured and Calculated Frequencies in Air and Vacuum	4-48
IV-5	Summary of Radiation Damage	4-55
V-1	Panel Mass Moments of Inertia	5-4
V-2	Parameter Values for Nominal and Sigma Resistance Deployment Calculations	5-27
VII-1	Accelerometer Type and Weight---Clamped and Pinned Configuration	7-7
VII-2	SCS-43 Test Panel---Electrical Characteristics	7-14
VII-3	Panel Performance Before and After Vibration Test	7-15
VIII-1	LASA Phase II Make-or-Buy Summary	8-7

1.0 INTRODUCTION

The information contained in this document is a record of the work accomplished in the Large Area Solar Array, Phase II, Development Contract 951934 with the Jet Propulsion Laboratories (JPL), Pasadena, California. The period covered is from June 30, 1967, to November 1, 1968.

This document is a complete report of the fourth quarter activity and summarizes the total Phase II contract.

1.1 PURPOSE AND OBJECTIVES

The Large Area Solar Array (LASA) program was initiated by JPL to advance the technology in lightweight electrical power systems capable

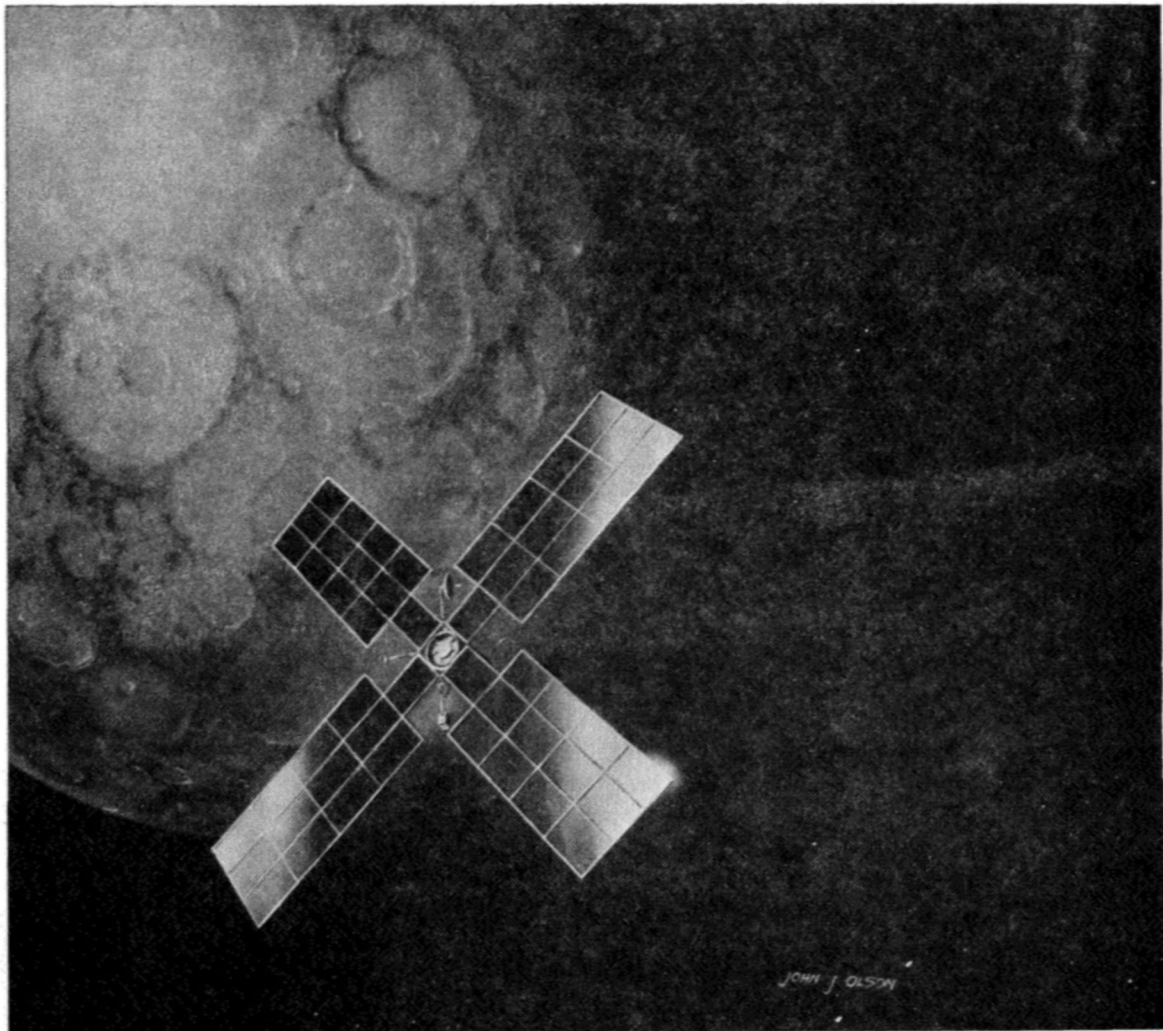


Figure 1-1: DEPLOYED ARRAY

of producing 50 kw of power for interplanetary spacecraft flights during the 1970's and 1980's.

Planning for unmanned electrically propelled spacecraft for Mars flyby and other interplanetary travel has shown that a requirement exists for a power system with lightweight, high-output capability.

The objective of the Large Area Solar Array program is to provide technology readiness for a 50-kw solar-cell array, with a minimum 20-watt/pound power-to-weight ratio at a Sun-probe distance of 1 AU (see Figure 1-1). The program was divided into three phases. The objective of each phase is as follows:

- 1) Phase I Objective: To analyze the Large Area Solar Array, complete the detail design, and provide partial experimental test verification.
- 2) Phase II Objective: Develop manufacturing technology by fabrication of one subpanel, obtain test data to evaluate the analysis, develop mechanisms, develop simulated zero-g deployment test equipment and procedures for verification of panel assembly deployment, and extend the preliminary design and development of deployment demonstration equipment.
- 3) Phase III Objective: To verify the design by fabrication, test, and analysis of a suitable simulation of a stowed panel assembly test article.

All Phase II objectives have been met. The technology developed in Phase II permits the fabrication of reliable, lightweight solar panels of up to 1-kw output. Although the LASA design is based on an electrically propelled spacecraft for Mars flyby missions, a major part of the technology developed will have direct application to numerous other space missions such as manned Earth-orbital and Mars-Venus spacecraft, and unmanned deep-space probes.

1.2 BACKGROUND AND DESCRIPTION

In December 1965, a Large Area Solar Array Fabrication Feasibility Study contract was completed by The Boeing Company for the Jet Propulsion Laboratory of Pasadena, California. As a result of this study, it was determined that the technology for a 50-kw, 20-watt-per-pound beryllium structured solar-cell array could be developed. The study also produced preliminary designs for both a 10-kw and a 50-kw array, and a program plan for this state-of-the-art advancement. Since that time various planning interruptions and policy changes have delayed the 1969 target by approximately 1 year.

The Phase I design contract was started in September 1966, and completed November 1967. During this first step of the three-phase program, the design concept produced by the feasibility study was thoroughly analyzed, and detailed design was accomplished. Allowables for the basic beryllium structure and other components comprising the array were established

by numerous tests and analyses. Also design of transportation and handling equipment was initiated and preliminary design of ground deployment equipment and a full-scale aluminum panel assembly deployment model was completed.

The Phase II development contract was initiated June 30, 1967, and completed October 31, 1968. This six-task development program, with Task 7 and Task 8 added by a contract change notice in March 1968, completes the intermediate step in the Large Area Solar Array technology advancement.

Mechanisms components were developed and tested with abbreviated models under ambient and vacuum chamber environments. They were then tested on an aluminum full-scale panel assembly deployment model using the simulated zero g deployment equipment that was developed and built during this phase. The No. 1 Main Subpanel (designated MTA-1, Major Test Article No. 1) was fabricated and tested. This 8- by 13-foot subpanel was made of two beryllium box section frames (Sunside and darkside) that sandwich a fiberglass webbing (substrate) to which the electrically connected solar cells are bonded. Insulated flat copper and aluminum buses were bonded to the frame members to complete the electrical circuit. A contract change replaced 90% of the live cells with dummy glass cells and transferred the electrical tests from MTA-1 to the 4-foot-square sample panel (SCS-43) that was built in Phase I. Dynamic response data also was developed for SCS-43 by vibration testing in air and partial vacuum.

During the final bonding cycle on the MTA-1 subpanel assembly, a number of adhesive bonds on the intercostal frame members and some of the gussets and clips were forced out of position. Excessive heat and side pressure were responsible for the failure. Repair techniques were developed and the assembly was restored to its basic configuration. Dynamic response of the subpanel was determined through vibration testing in air and in the vacuum space chamber. Manufacturing of the necessary ground support and test equipment, preliminary design for Phase III GSE, and thorough documentation, including program reviews covering all of the program activities, completed the Phase II effort.

The third and final phase was proposed and was scheduled to start July 1, 1968, with completion in April 1970. The delay in starting Phase III has contributed to the 1-year slide in our technology readiness goal. To reduce the cost for this phase, we proposed manufacturing a 7-subpanel beryllium structured array (rather than 13) using a dummy load for the other 6 subpanels. It is felt that this simulated stack will satisfactorily represent a specimen that can be tested for the launch vibration and thermal-vacuum modes. Figure 1-2 illustrates the sequential development for the three-phase program.

1.3 ORGANIZATION OF DOCUMENT

The format of this document has been especially designed to permit the reader to select the level of detail he desires.

This document can be read in three distinct levels of detail:

- 1) Section 2 provides a summary of the document, but necessarily contains the least amount of detail.
- 2) More detail is found in the proposition statements that appear in script at the beginning of each topic.
- 3) Further detail is available in the body of the individual subsections.

All data pertaining to each of the different tasks is reported in a separate section. The complete report consists of:

Section 1: Introduction

Section 2: Summary---A brief summary of each of the functional areas' activities during the Phase II contract.

Section 3: Small Component Mechanisms (SCM) Testing (Task 2)--- A detailed description of all work involving the Task 2 development of mechanisms by fabrication, testing, and analyses, and the results obtained by these efforts.

Section 4: Major Test Article No. 1 (MTA-1) Development (Task 3 and Task 7)---A detailed report on the fabrication, test, and analysis of the No. 1 Main Subpanel. Included is the CCN deletion of live cells, the assembly and bonding problems, the repair of bonds, and Quality Assurance program.

Section 5: Major Test Article No. 3 (MTA-3) Deployment Testing (Task 4)---A detailed report of the simulated zero g deployment equipment, its development, construction, checkout, the mechanisms and aerodynamic testing, analysis, and conclusions.

Section 6: Handling Equipment and Phase III GSE Preliminary Design (Task 5)---A detailed report on the transportation and handling equipment, the Phase III preliminary design, including preliminary design of deployment demonstration equipment, the extended deployment equipment work in response to the JPL Technical Direction Memorandum 951934-5, and the environmental test equipment.

Section 7: SCS-43 Subpanel Vibration Test Program (Task 8)---This section reports the additional test and analysis effort performed with the SCS-43 4-foot-square test specimen that was built during Phase I.

Section 8: Materiel---A report on the raw stock purchase plan, and the outside vendor and subcontract activity.

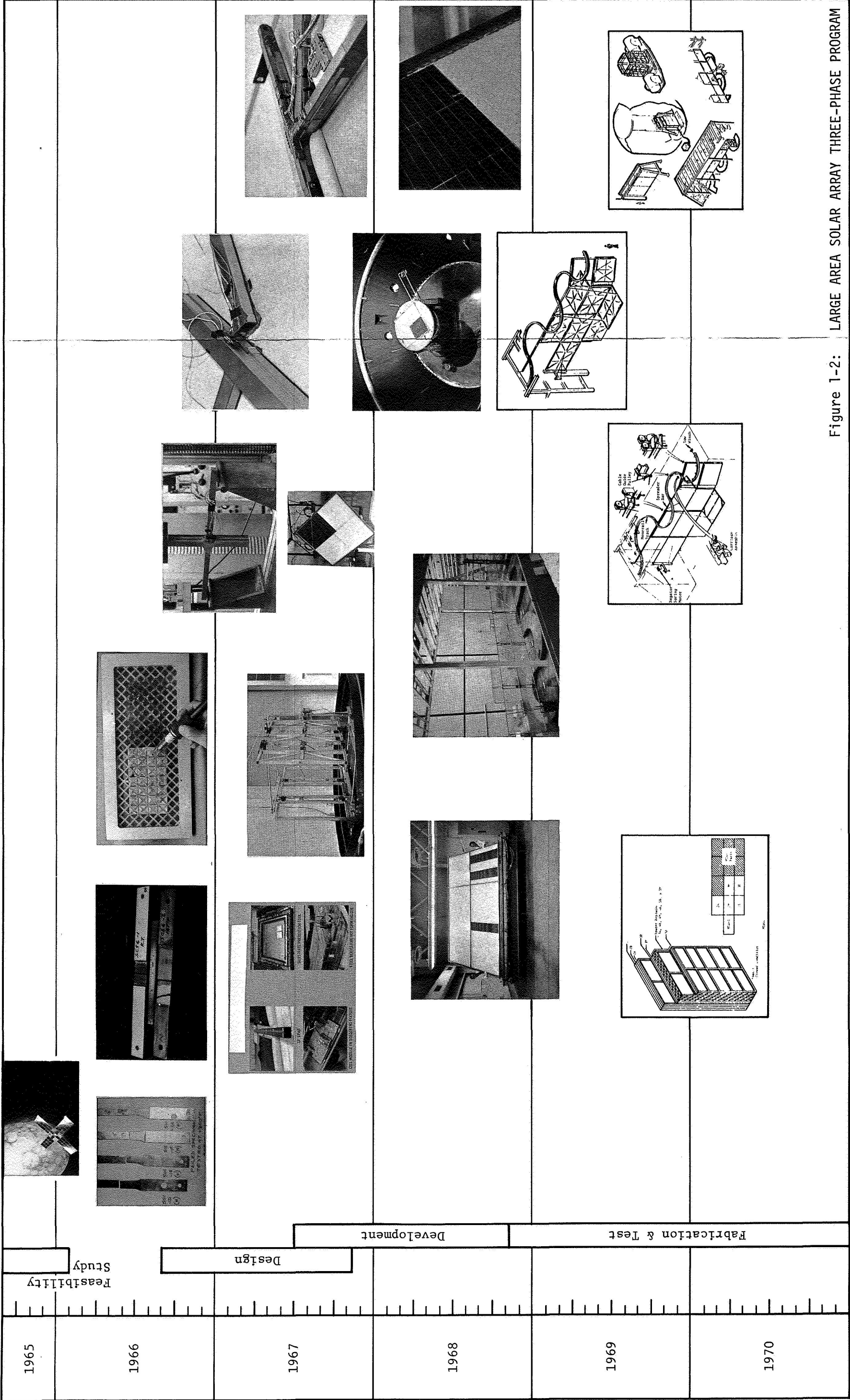


Figure 1-2: LARGE AREA SOLAR ARRAY THREE-PHASE PROGRAM

- Section 9: Major Conclusions and Recommendations---Conclusions concerning program status at the end of Phase II. Also, recommendations for revisions and additional work to be accomplished during Phase III.
- Section 10: Bibliographs and Definitions---A list of program documents, miscellaneous references, symbols, terms, and definitions.

2.0 SUMMARY

This section of the LASA Phase II Final Report provides a summary description of the work performed by the various functional organizations. These separate responsibilities were coordinated to accomplish the program objectives.

2.1 TASK OBJECTIVES

Task 1 objective was to prepare a workable plan and schedule for all events or milestones necessary to complete the program within the time specified by the contract.

Task 2 objectives were to test the boost tiedown release and deployment mechanisms to provide data for design application.

The objective of Task 3 was to fabricate, test, and analyze the No. 1 Main Subpanel. This included the development of manufacturing cost-effective techniques for producing this large beryllium, titanium, fiber-glass, and solar-cell unit. The manufacturing operation is documented in the Manufacturing Plan and Specification document, D2-113353-1.

The objective of Task 4 was to develop and build a panel assembly deployment model, with prototype deployment mechanism, and develop and build equipment that could be used to provide a test bed for zero g deployment testing of the panel assembly.

The Task 5 objective was to provide the necessary ground handling and transportation equipment to support the MTA-1 and to complete the preliminary design for Phase III GSE, including deployment demonstration equipment. The handling equipment was built and used and the preliminary design was completed.

Tasks 7 and 8 were authorized by a contract change notice. This change replaced the live solar cells with dummy glass cells on MTA-1 and provided for power degradation tests and vibration tests on the SCS-43 4-foot-square sample panel built during Phase I.

Figure 2-1 is a flow chart of Phase II which illustrates the relationship between the various tasks resulting in a complete program.

2.2 ARRAY DESIGN AND ANALYSIS

The Phase II program was primarily focused on the continued development of the detail design and refinement of the analyses established in Phase I. This section describes the Large Area Solar Array, defines the principal developments in design during the Phase II contract, and reports the results of analyses accomplished.

The description of the array is divided into structure, mechanisms, and electrical power system considerations.

Design analyses on the array were revised, refined, and developed for the following conditions:

- 1) MTA-4 structure loads during boost;
- 2) Vibration modes and resonances for MTA-1 during boost;
- 3) Velocities and member loads in MTA-4 during deployment in the Earth gravitational field and in space.

A more detailed description of areas affected by the Phase II work is given in sections covering that particular subject.

2.2.1 DESCRIPTION OF THE ARRAY

There were no major design changes resulting from Phase II testing; however, many detail changes were made.

A description of the Large Area Solar Array is included here to define the structure in the stowed and deployed conditions. The mechanisms used in the release and tiedown system and to deploy and lock the subpanels, and the electrical power generation and transmission system are also discussed.

2.2.1.1 Description of Array Structure

The arrangement of the solar array configuration stowed within the Saturn IB/Centaur shroud is illustrated in Figure 2-2. The four solar panels each are supported by the spacecraft at Station 10.5 by three hinges on Subpanel No. 1. These hinges allow for thermal expansion between the spacecraft and the solar panel assemblies. Subpanel No. 1 of each of the four solar panels is connected at Stations 124 and 170, forming a box. The remaining 12 subpanels of each solar panel are folded and stowed on the outside of the box. Stiffness is added to the stowed stack by providing shear transfer fittings between subpanels. A preloaded tiedown system prevents disengagement of the shear fittings under launch loads.

When deployed, the solar array configuration consists of four solar panel assemblies placed symmetrically about the spacecraft as shown in Figure 2-2. Each solar panel assembly has 13 subpanels connected by hinges and locked in a flat plane. The fundamental bending frequency of the array in this condition is calculated to be 0.068 Hz. This value is a function of the axial stiffness of the diagonal strut (which provides support for the solar panel from the spacecraft), the bending stiffness of the subpanel beryllium spars, and the stiffness of the subpanel hinges.

Each subpanel is made of a darkside frame, a Sunside frame, and a pre-tensioned substrate made of 0.003-inch-thick by 0.14-inch-wide fiberglass tape sandwiched between them.

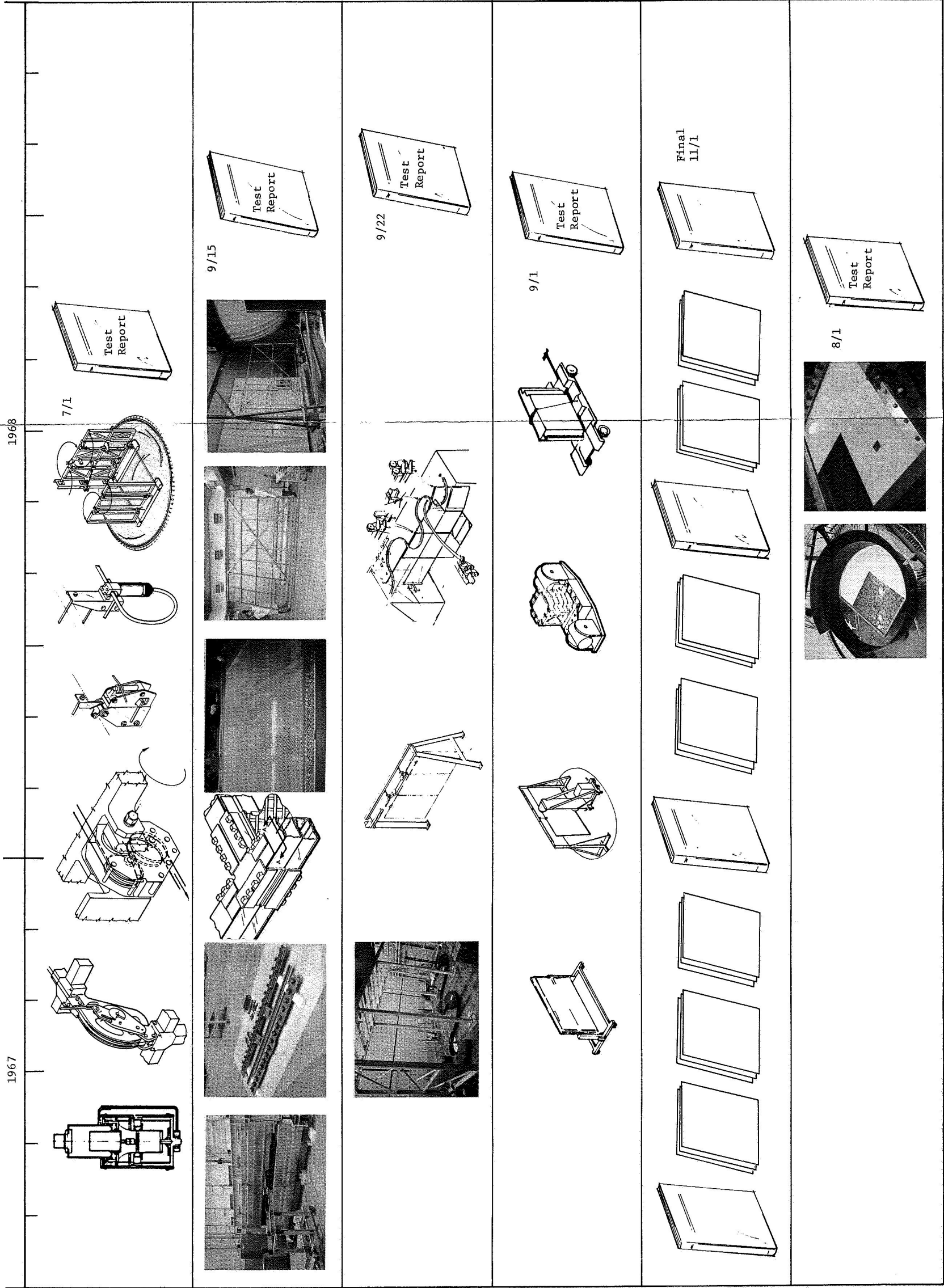


Figure 2-1: LASA PHASE II PROGRAM TASKS

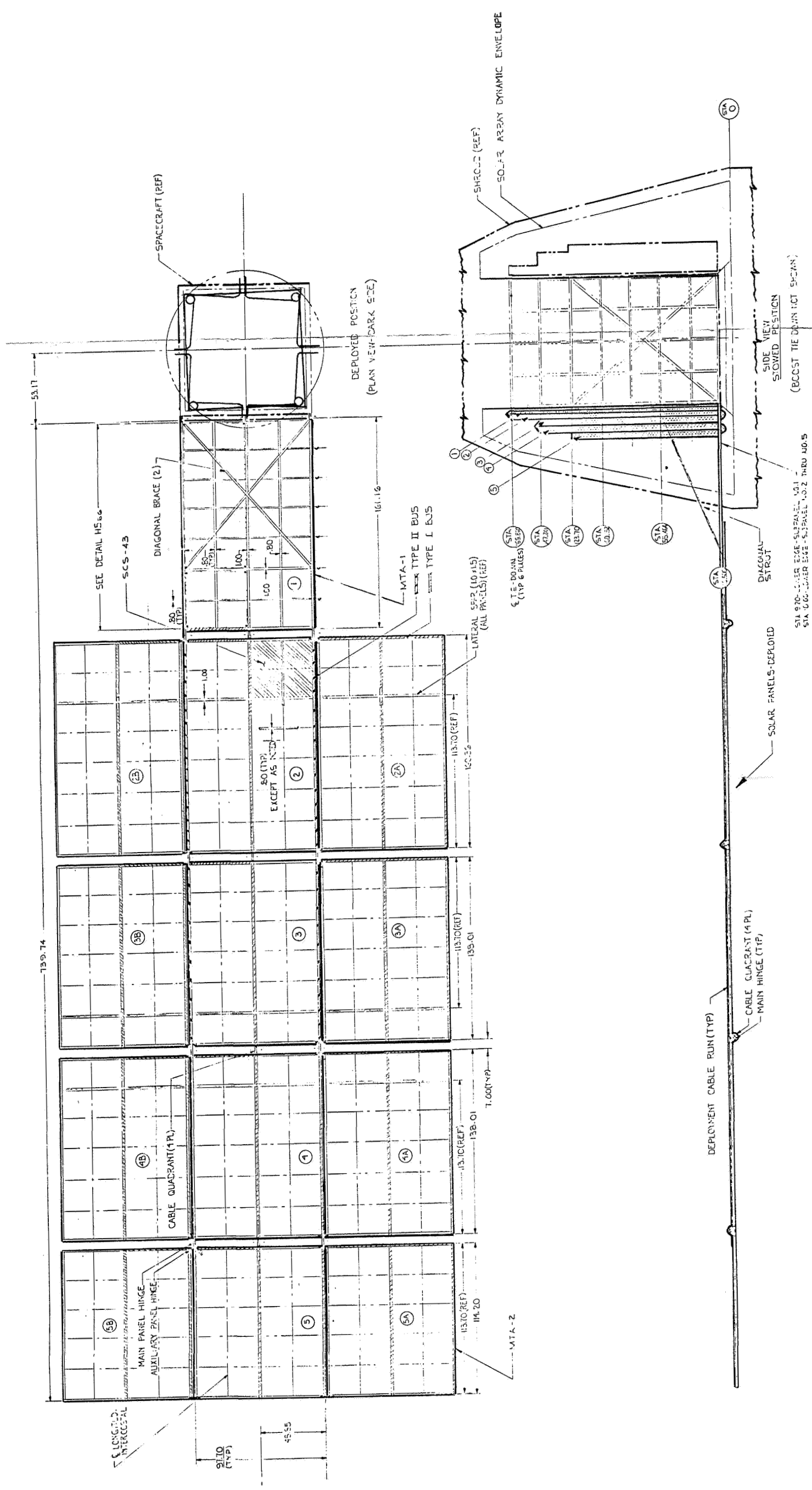


Figure 2-2: LARGE AREA SOLAR ARRAY CONFIGURATION (MTA-4)

The frames were made of built-up beryllium-bonded rectangular sections connected by titanium fittings where loads are concentrated.

A summary weight statement of MTA-1 is shown in Table II-1 and a revised weight statement of the baseline design is shown in Table II-2. The MTA-1 statement was prepared by using actual weights obtained during fabrication of the MTA-1. The baseline weight statement was prepared using the actual weights of MTA-1 and calculating the remainder from drawings and adjustments made as a result of actual weights on Subpanel No. 1.

The actual weights of the six solar-cell modules installed on MTA-1 were used to calculate the total weight. The weight variation between calculated and actual weight of the "stick" assemblies was less than 1%. The titanium fitting actual weights were higher than calculated because the manufacturing plan produced wall gages larger than nominal. The total weight of the solar array configuration is calculated to be 2,182 pounds. This results in a 21 watts/pound array. This includes 392.1 pounds of solar cells, which is 30 pounds over the 362.1-pound maximum specified cell weight. These figures are based on actual weights of sample lots of solar cells.

2.2.1.2 Description of Solar Array Tiedown-Release and Deployment System

Deployment of the solar panel is powered by redundant motors and cable drums that drive the subpanels and sequentially operate mechanical devices during deployment.

The solar panel release and deployment mechanisms must, upon command and in proper sequence, extend and lock the solar panels in their deployed positions. The release and deployment mechanisms designed to satisfy this requirement have three separate functions: release of the tiedown system, deployment of the main subpanels, and deployment of the auxiliary subpanels.

The release of the structural tiedown system for the four solar-array assemblies is accomplished simultaneously by the firing of an ordnance pin-release device located at solar panel Station 123.7. This device releases all tiedowns except one on the center spar at Station 123.7.

Figure 2-3 shows the release mechanisms at Stations 123.7 and 169.86. Each pin-release device releases a mechanical cable train at each of the four corners and one circumferential cable. Each cable train releases the remaining five circumferential tiedown cables.

Five of the six center-spar tie cables are released mechanically by its circumferential cable, as shown in Figure 2-3. The individual center spar cables are pretensioned as the solar panel is fitted to the spacecraft on the pad. A clamping device keeps the cables properly loaded until released by the circumferential cables.

Table II-1: MTA-1 WEIGHT STATUS

Item	Actual Weight	Calculated Weight	$WT_A - WT_C$	$\frac{WT_A - WT_C}{WT_C} \times 100\%$	Effect on Estimated Array Weight
Cells, Coverglasses, CG Adhesive, Connectors, Solder (cell quantity = 2,436)	1.51	1.50	+0.01	+0.7%	None
Simulated Cells (cell quantity = 20,300)	13.41	12.47	+0.94	+7.5%	None; however, simulated cells do not meet specifications of $\pm 5\%$ weight variation.
Substrate Adhesive, RTV-40 Thermal Control on Cells and Tape (Structure TCC Weight not Included)	1.90	2.18	-0.28	-12.8%	None; application process needs refinement to ensure uniform application and thickness control
Simulated Electrical Installation (Buses, Terminals, Diodes, Terminal Fasteners)	5.97	6.57	-0.60	-9.1%	None; simulated buses lack some insulation and adhesive
Total Frame Assembly with Substrate	(58.48)	(55.11)	(+3.37)	(+6.1%)	Breakdown shown below
Beryllium Members Weighted (Spar Halves, Intercostals, Diagonal Brace)---84% by Weight of Total Beryllium Members	26.65	26.40	+0.25	+0.9%	None; however, attachment 3 of Reference (1) shows that all members that exceed 10% variation from calculated weight are overweight
Major Fittings (Primarily Weldments)	15.15	13.63	+1.52	+11.2%	Weldments are not made per drawing; effect if all weldments are fabricated in the manner of the MTA-1 weldments is 8 pounds per array (75% by weight of weldments are on Subpanel No. 1)
Compression Pads and Shear Tooth Fittings	2.17	2.29	-0.12	-5.2%	None
Structure Repairs (Includes Discrete Parts and Adhesive to Apply Parts---Does not Include Adhesive to Full Voids)	0.79*	0	+0.79	--	Damage to MTA-1 was incurred in the development of a fabrication process. The process has been revised such that the probability of damage due to that source is reduced. Other repairs are probable, however, and an allowance for repairs should be included. Five pounds should be added to the array for repairs.
Beryllium Sticks, Clips, Splice Plates, Integration Adhesive, Substrate, Adhesive in Void Repairs, Drawing Errors, Calculation Errors	13.72	12.79	+0.93	+7.3%	Part of this difference can be rationalized away to adhesive on some parts, and possibly vacuum bag sealant. The variation as shown is within the range of variation on the parts and materials included in this weight. An allowance of 0.005 pound per square foot should be added to the structure to account for the unknowns.
Totals	81.27	77.83	3.44	4.4%	

*Based on a calculated weight per Reference (2). See Attachment

Table II-2: MTA-4 WEIGHT SUMMARY

Cell Stack, Substrate, and Thermal Coating	Weight-Pounds (838.5)
Cover Glasses	160.1
Cover Glass Adhesive	18.4
Cells	392.1
	(362.1 Spec Max.)
Connectors	56.4
Solder	3.7
Substrate Adhesive	63.2
Substrate	67.1
Thermal Coating	77.5
	(977.4)
Structure	
Panel 1 (without substrate)	214.5
Panel 2B	80.3
Panel 2M	75.0
Panel 2A	66.6
Panel 3B	59.0
Panel 3M	63.3
Panel 3A	59.3
Panel 4B	59.1
Panel 4M	59.8
Panel 4A	60.3
Panel 5B	46.5
Panel 5M	48.7
Panel 5A	57.0
Repairs	5.0
Contingency	23.0
Mechanisms	(242.0)
Main Hinges and Latches	22.9
Auxiliary Hinges, Latches, and Dampers	50.9
Quadrants and Sequencers	30.3
Strut Assemblies	14.5
Solar Curtains	13.2
Boost Tiedown System	71.0
Cable, Drive, and Miscellaneous	39.1
Electrical Connections	(124.0)
Deploy Termination Switch and Wiring	1.0
Buses and Diodes	81.6
Pigtails and Connectors	12.8
Squib Wiring	3.7
Terminals and Crossover Buses	25.2
	2,182.0
	(2,152 based on maximum spec weight for cells)

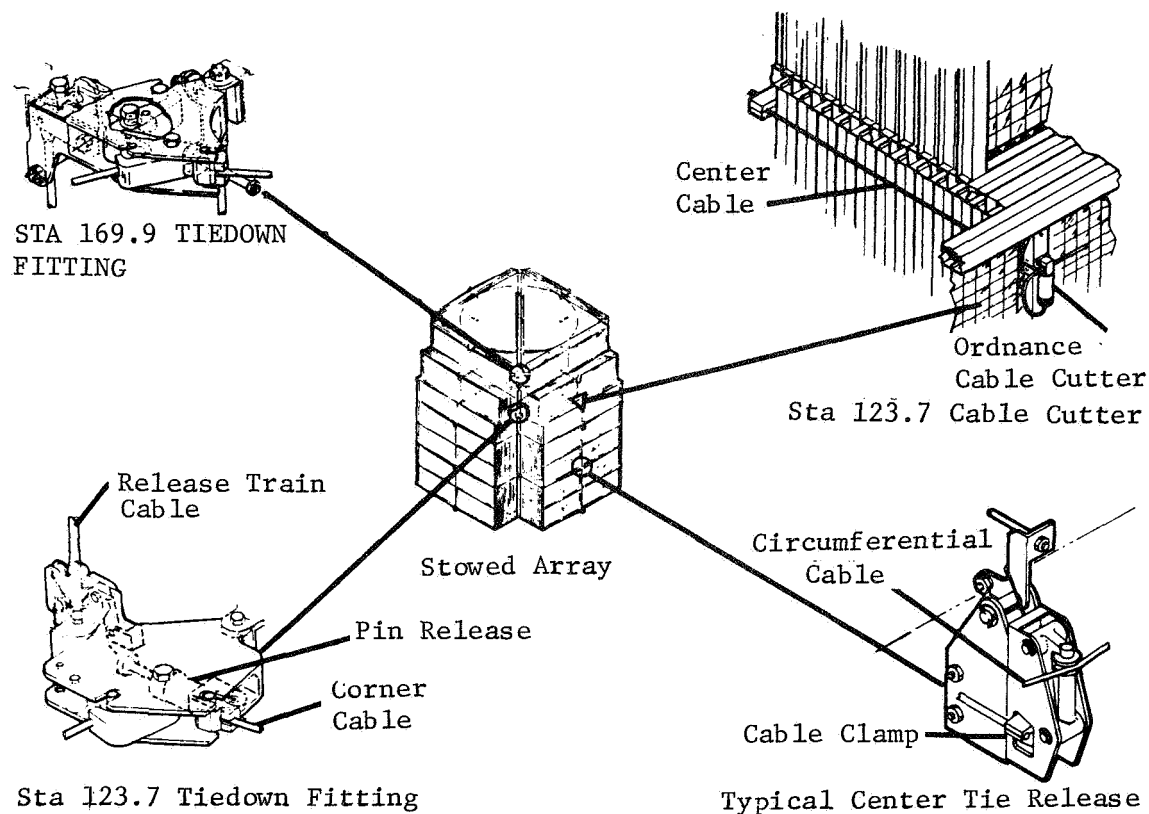


Figure 2-3: BOOST TIEDOWN AND RELEASE SYSTEM

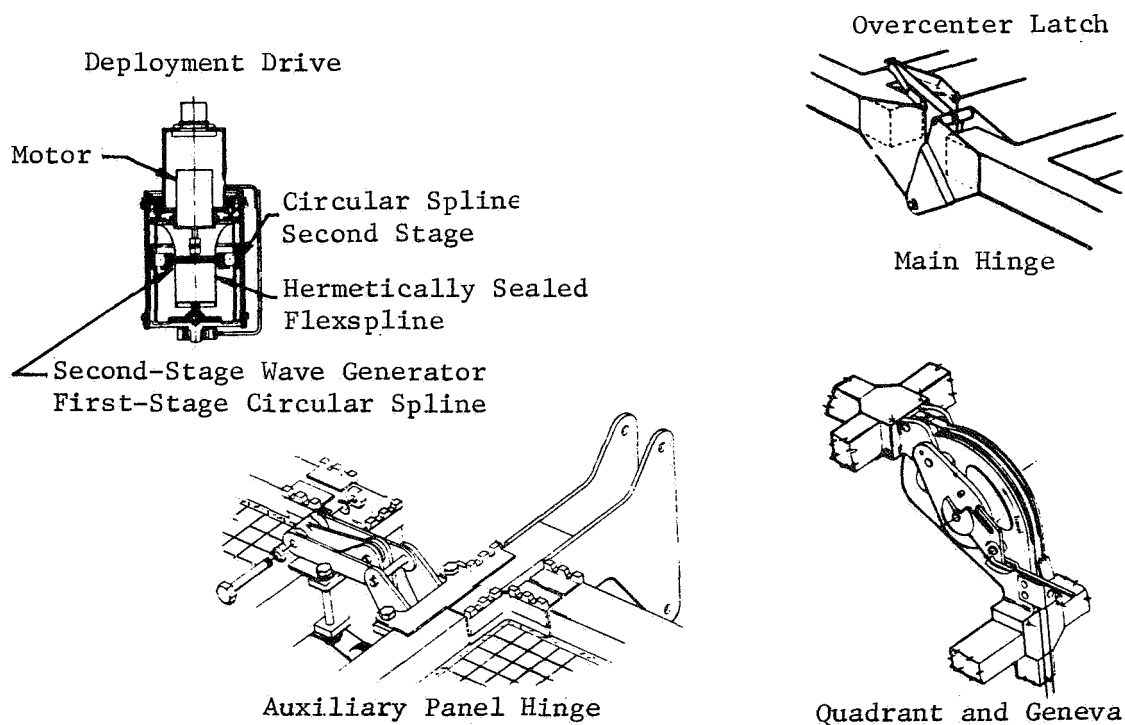


Figure 2-4: DEPLOYMENT MECHANISMS

The sixth center-spar tie, also shown in Figure 2-3 and located at Station 123.7, is pretensioned before panel installation. This tie is released by firing an ordnance-actuated cable cutter simultaneously with actuation of the deployment motor after all other ties have been released and circumferential cables have been ejected. The sixth tie prevents inadvertent partial deployment of the solar subpanel stack during the coast period and acts as a clamp during boost. The firing of the cable cutter ordnance and actuation of the deployment motor initiates deployment.

The main subpanel deployment system consists of four deployment motors located within the spacecraft envelope, as shown in Figure 2-4. The deployment cables are rigged so that diametrically opposite panels are driven by two motors operating in parallel. Each motor operates as the primary drive for the panel assembly mounted adjacent to the motor and, in addition, each motor serves as the backup drive for the panel assembly mounted opposite.

The deployment motor assembly (Figure 2-4) includes a 1/250-hp Globe-type SS d.c. motor rotating under load at 97 rpm. The motor drives a 2.75:1 smooth ball bearing planetary reduction. The ring forms the wave generator for a middle-stage harmonic drive reducer with a ratio of 72:1. The output of the middle reducer is the wave generator of the third-stage harmonic drive with a reduction of 140:1 for a total reduction of about 37,000:1. The third-stage output directly drives the cable drum that is the outside housing for the motor and reducing train. The effective pitch diameter of the cable on the drum is 2.46 inches; the drum revolves at 0.259 rpm, providing a cable speed of 0.033 inch per second.

As outboard main subpanels are deployed, the inboard auxiliary subpanels (hinged on the side of the main subpanels) are sequentially unlatched and deployed by torsion springs. The rate of deployment at the latching position is controlled by a small rotary damper. When the auxiliary subpanel reaches the deployed position, an overcenter latch snaps into place, locking the panels in a flat plane.

The direction of deployment of all 13 subpanels is indicated in Figure 2-5.

When the last subpanel unit (No. 5) is deployed, the drive motor is shut off by a switch activated by the cable tension.

The design of the deployment motor shutoff was postponed until motor current data were available from the MTA-3 deployment tests. There was a possibility that the motor current would rise abruptly before the deployment cable tension exceeded the 60-pound design shutoff point. This current surge would provide a shutoff signal. However, test data show very little change in motor current; therefore, this idea cannot be used.

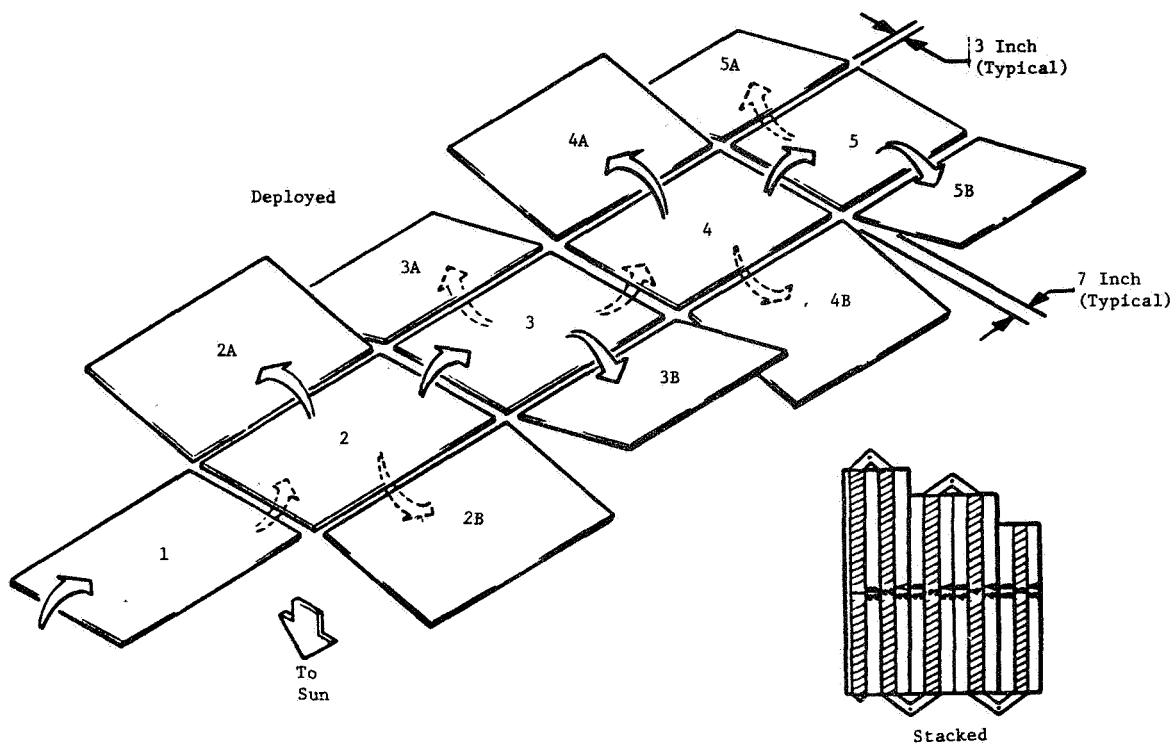


Figure 2-5: DEPLOYMENT SEQUENCE

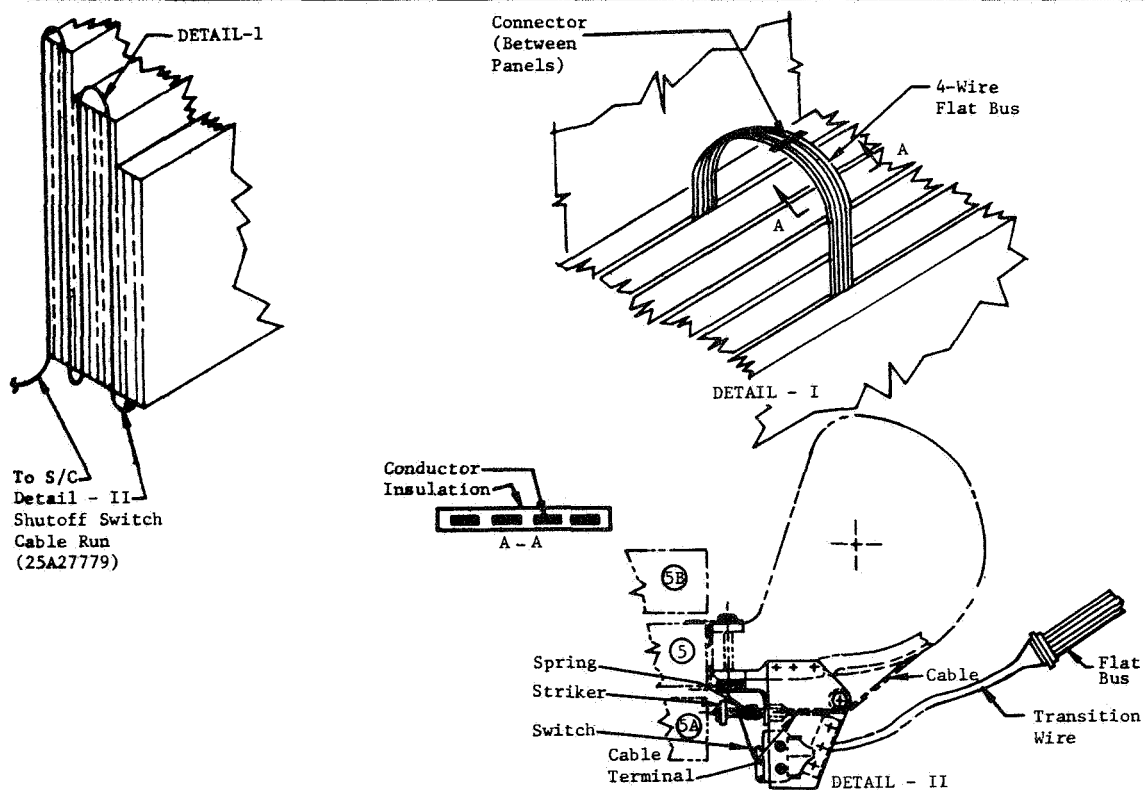


Figure 2-6: SHUTOFF SWITCH INSTALLATION

To satisfy this requirement a switch and striker device was designed for installation at the outboard hinge. The switch is normally closed and opens when the cable tension is approximately 40 pounds. The outboard end of the deployment cable is spring-loaded and as the cable tension rises, the striker at the cable end operates the switch. See Figure 2-6 for details.

Switch wiring consists of four-conductor, flat wire cables attached to the intercostals of each subpanel. An electrical connector is required at each hinge line to facilitate assembly and maintenance. The cable assemblies are required to meet NASA specifications.

2.2.1.3 Description---Electrical Power System

The design obtains the maximum coverage of the substrate with solar cells. The gross deployed area is 4,924 square feet and the net module area is 4,590 square feet. The ratio of these areas gives a solar-cell packing factor of 0.93. See Table II-3 for the power generation summary.

The power output for a complete array, based on 10 watts per square foot, is 45,900 watts. This value is given for air mass zero, and 1 AU at 55°C.

The basic power generation unit contains 406 solar cells and is a complete module for the 28-volt system and is a quarter module for the 100-volt system. Seven cells are connected in parallel and 58 groups of seven cells are connected in series. Four units are connected in series to make a 100-volt module.

All of the electrical terminations for the modules are adjacent to a main spar. The negative terminals are connected to the negative bus with expanded silver foil. The positive terminals are connected to a diode and the diode is connected to the positive bus. One diode is installed for each module. Diodes are used for the electrical isolation of each module for test purposes only. The type of diodes that were installed on Small Components Structures Test No. 43 (SCS-43) performed quite satisfactorily when exposed to a simulated space environment at maximum load.

Electrical buses are flat strips of aluminum or copper and are bonded to the main spars. The thickness of the conductors varies in accordance to the electrical current density. The array bus system is shown in Figure 2-7. The generated power is conducted across the gaps in the subpanels by means of crossover buses. The design of the crossover buses provides a very flexible twisted pair of straight-laid soft copper that has very low spring forces. Subpanel deployment forces are very low and the design of crossover buses was considered critical.

Weight estimates for the array are based on the use of 8-mil N/P solderless silicon solar cells and 3-mil cover glasses.

Table II-3: POWER GENERATION SUMMARY

	Subpanel Unit Number										Remarks
	Per Cell	Per Module 28v	Per Module 100v	No. 1	No. 2	No. 3	No. 4	No. 5	Per Panel Array	Per	
Number of Cells	-	406	1,624	22,736	22,736	19,488	19,488	16,240	256,592	1,026,368	
Number of 28v Modules	-	-	-	48	0	0	0	0	48	192	
Number of 100v Modules	-	-	-	2	14	12	12	10	146	584	
Voltage at 1.0 AU and 50°C	0.43v	25v	99v	-	-	-	-	-	-	-	
Voltage at 1.67 AU and 50°C	0.55v	32v	127v	-	-	-	-	-	-	-	
Power at 1.0 AU and 50°C	0.045w	18w	75w	1.1kw	1.1kw	870w	870w	730w	11.5kw	46kw	Bus loss not considered
Power at 1.67 AU and 50°C	0.018w	7w	29w	410w	410w	350w	350w	290w	5w	20kw	Bus loss not considered
Gross Deployed Area	-	-	-	-	-	-	-	-	-	4,924 Ft ²	
Net Module Area	-	-	-	-	-	-	-	-	-	4,590 Ft ²	

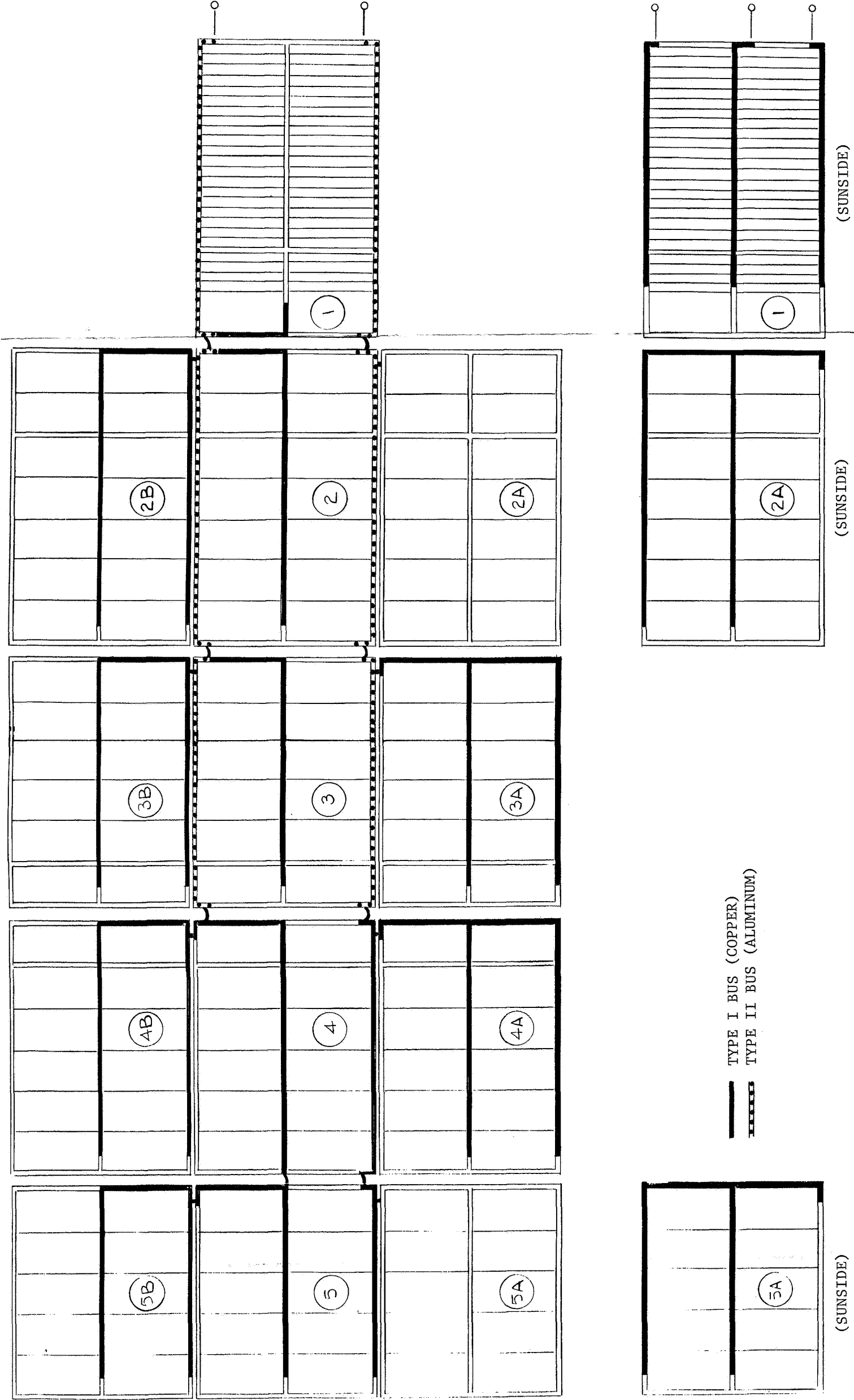


Figure 2-7: ARRAY BUS SYSTEM---DARKSIDE

2.2.2 PHASE II DESIGN DEVELOPMENT

Detail design development has greatly enhanced the technology readiness of the Large Area Solar Array.

Design developments made during Phase II included efforts on the mechanisms components tested in Task 2 and the deployment mechanism system tested in Task 4. The mass-simulated, full-size aluminum frame solar panel was used to check the deployment mechanisms.

Repair techniques were developed for MTA-1 to satisfy the needs to disassemble, clean, and rebond some beryllium members.

The following developments have increased the technical knowledge of the mechanisms components of the Large Area Solar Array.

Tiedown Cable----The 5/32-inch Nomex tiedown cable used to clamp the solar array subpanels together at the center spar was found to be adequate in five of the six tiedown locations. The cable strength and load relaxation rate made it inadequate for use at Station 10.5. An increase in diameter to 3/16 inch is recommended for all stations and is required for Station 10.5. The friction coefficient between the Nomex cable and the clamping wedge was experimentally determined to be 0.293 at a wedge angle of 5 degrees.

Center Spar Clamp Assembly----The center spar clamp assembly was inadequate as originally designed. The clamp was too weak to maintain the cable tension load. A redesign was made using a wedge action capable of loads beyond the breaking strength of the Nomex cable.

Auxiliary Subpanel Rotary Dampers----The off-the-shelf dampers satisfactorily controlled the opening rate of the simulated subpanels at temperatures below -100°F in a vacuum. Adjustment of the damping ratio was very sensitive and design changes are recommended for simplification of adjustment.

Motor/Cable Drum Assembly----The deployment system drive is provided by an electrical motor-driven harmonic drive reduction with a line take-up rate of 2 inches per minute at a line load of 60 pounds. This motor-cable drum assembly operated satisfactorily in a vacuum pressure of 10^{-8} torr and a temperature of -63°F. It also repeatedly powered the aluminum deployment model during simulated zero g deployment tests. The rate of the line takeup increased with use and was 5% over tolerance after qualification tests were completed.

Repair Techniques

Damage to the MTA-1 subpanel during the final bonding operation required the development of the following techniques and processes to perform successful repair.

Inspection---Bond damage was identified by three separate inspection techniques: visual with feeler gages; coin tapping; and the use of an experimental harmonic bond tester.

Removal of Bonded Parts---Some bonds had to be separated in repairing the beryllium structure to allow removal for recleaning and reassembly. A small high-temperature hydrogen torch was successfully used to heat adhesives rapidly enough to fail the bond in peel without appreciably increasing the temperature of adjacent bonds or parts.

Rebonding of Broken Bonds---Two methods were used to reinforce bonded surfaces damaged or not completely bonded where removal of the part was not warranted. These methods were:

- 1) Injection of BMS 5-29 and liquid AF-126 was used to reinforce areas where voids occurred and joint strength was marginal. This method provided good strength bonds at room temperature and required no heat cycle for curing.
- 2) In areas where broken BMS 5-51 bonds were found and removal of the part was not warranted and the bond strength required was low, a repair adhesive XA-2327 was used. This adhesive produces bond strengths above 2,000 psi with low contact pressure and at 167°F.

These bonding processes were developed to satisfy conditions that occurred in the fabrication of the beryllium solar panel assemblies. The development will reduce the fabrication costs in the future by producing suitable repair processes.

Electrical Design Development

The LASA electrical design development produced data on materials and processes for the design of flat power buses, solar-cell/coverglass assemblies and installation, and solar-cell interconnectors. Company-funded studies and experiments to develop solar-cell/coverglass installation processes have been in progress. New tools and new techniques have reduced coverglass assembly rejection rates.

Resistance-type pulse soldering was used on the MTA-1 modules; however, this method is considered slow. A company-funded study developed tools and techniques for multiple joint soldering of solar cells with infrared equipment. All of the positive or negative contacts of seven-cell groups can be soldered in one operation with this method. Evaluation and testing, by Manufacturing Development, of the infrared process indicates that the tools and processes are satisfactory.

2.2.3 DESIGN ANALYSIS OF MTA-4

Technical analyses performed in Phases I and II predict the Large Area Solar Array will yield 21 watts per pound and that the probability of successful boost and deployment in space is acceptable.

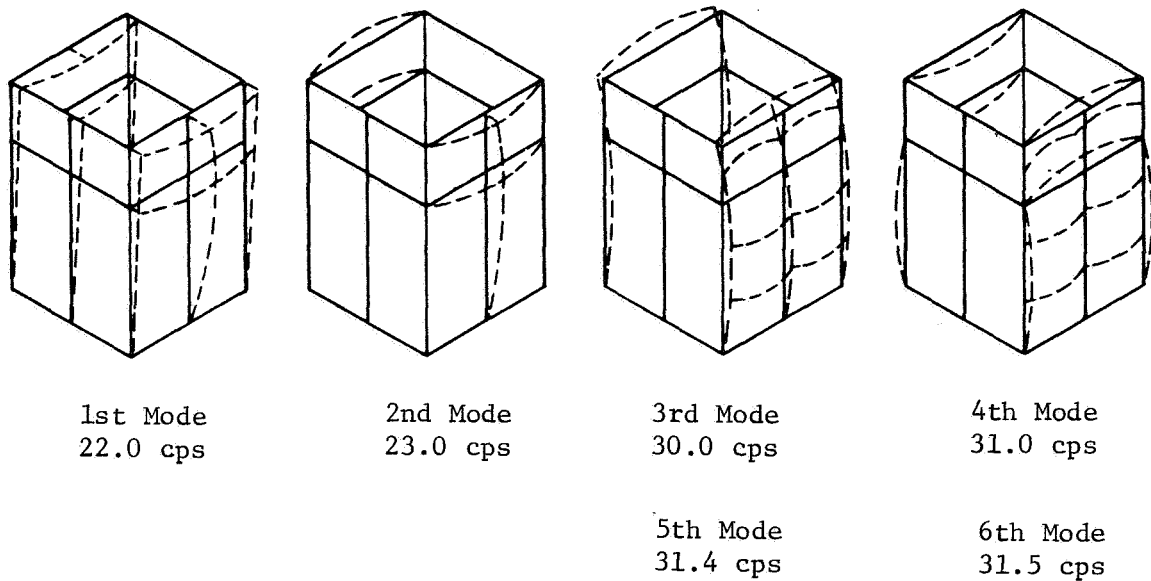


Figure 2-8: DYNAMIC ANALYSIS MODES ;

The primary changes to the MTA-4 required by analytical iterations were to increase the gages of the center spars of Subpanels 2 through 5. These changes were identified and described in detail in Section 2.5.3 of the Phase II 3rd Quarter Report, D2-113355-6. All other structural members of the LASA design have adequate margins of safety and conform with "minimum gage" requirements.

The technical analyses at the end of Phase II indicate that the MTA-4 design can survive the critical 1.5-g peak lateral sinusoidal boost environment with safety and tolerable degradation and/or damage, that there is an acceptably high probability of successful deployment in space, and that the electrical performance verifies the electrical design. Also, analyses indicate that appropriate ground deployment tests of the solar panel can be performed on the ground with the deployment test equipment.

The dynamic conditions create the critical loads and deformations in the stowed configuration. The results of the dynamic analyses are shown in Figure 2-8, and Tables II-4 and II-5. The margins of safety on strength of primary structural elements are summarized in Tables II-6 and II-7. The strength of fittings and mechanisms, as well as more details of the primary structure and its analyses, are to be found in the 3rd Quarterly Report, Part II, D2-113355-6. The loads analyses are based on an assumption of Q (load magnification factor) = 15. Tests to date indicate this is achievable, but Phase III vibration testing is essential to verify that the existing design does provide sufficient damping.

Table II-4: LASA TO SPACECRAFT NET CLEARANCES

Location of Clearance	Clearance	LASA Static Clearance (inches)*	Dynamic Deflection (inches)**	Net Clearance or Interference (inches)
<u>Stations 10 to 40</u>				
• Tiedown fitting at station 10.5 to spacecraft	Local	0.1	0.06	0.04
• Diagonal strut fitting at station 27.3 to spacecraft	Local	0.40	0.30	0.10
• Diagonal strut to spacecraft	General	0.84	0.55	0.29
• LASA frame to spacecraft	General	1.85	0.55	1.30
<u>Stations 40 to 87</u>				
• Diagonal strut to spacecraft	General	0.84	0.62	0.22
• LASA frame to spacecraft	General	1.85	0.62	1.23
<u>Stations 87 to 125</u>				
• Tiedown fitting at station 100.2 to spacecraft	Local	0.1	0.17	(0.07)
• Diagonal strut fitting at station 119.20 to spacecraft	Local	0.40	0.62	(0.22)
• Diagonal strut to spacecraft	General	0.84	0.62	0.22
• LASA frame to spacecraft	General	1.85	0.62	1.23
• Tiedown fitting at station 123.7 to spacecraft	Local	0.10	0.16	(0.06)
<u>Stations 125 to 151</u>				
• LASA frame to spacecraft	General	1.85	0.60	1.25
• Tiedown fitting at station 147.5 to spacecraft	Local	0.10	0.15	(0.05)
• Overcenter lock at station 150.2 to spacecraft	Local	1.33	0.15	1.18
<u>Stations 151 to 171</u>				
• Tiedown fitting at station 169.9 to spacecraft	Local	0.10	0.15	(0.05)
• LASA frame to spacecraft	General	1.85	0.35	1.50

*Includes stack tolerance buildup

**Based on lateral sine vibration input of 1.5 g

Table II-5: SUMMARY OF DYNAMIC LOADS

	Critical Frequency (Hz)	Maximum Structural Deflection (inches)	Total Shear at S/C Hinge (pounds)	Total B.M. at S/C Hinge (inch-pounds)	Load in Outer Spar (inch-pounds)	Load in Center Spar (inch-pounds)	Load in Lateral Spar (inch-pounds)	Cable Load at Sta 123.7 (pounds)	Cable Load at Sta 169.9 (pounds)	Mode Description
Case 1	24.7	0.534	17,200	1.845×10^6	15,023	57,125	38,400	1140	270	Fundamental bending
Case 2	23.8	0.510	15,600	1.675×10^6	13,800	51,066	34,320	1050	208	Fundamental bending
Case 3	23.7	0.33	17,500	1.875×10^6	12,160	35,456	23,336	1195	263	Fundamental bending (excited at -5°)
Case 2	42.0	0.003	70	1160						Torsional mode

Table II-6: MTA-1 STRESS ANALYSIS SUMMARY---BERYLLIUM

<u>Member</u>	<u>Critical Condition*</u>	<u>Margin of Safety</u>
Outboard Spar	In-plane panel vibration; channel crippling due to combined axial and bending loads.	+ 0.00
Lateral Spar (Station 123.7)	In-plane panel vibration; spar cap tension due to later spar bending.	+ 0.04
Lower Edge Member (Station 10.5)	In-plane panel vibration; Sun side spar cap crippling due to combined axial and bending loads.	+ 0.06
Diagonal Brace	In-plane panel vibration; channel crippling due to combined axial and bending loads.	+ 0.00

* Using ultimate safety factor of 1.25, applied to limit loads obtained from 1.5-g sine vibration input

Time-history and transient load studies are presented in the referenced document. The space deployment loads are summarized in Figure 2-9. The deployment of auxiliary panels together with the analysis of the deployment hinges, quadrants, latches, and release mechanisms are discussed in detail in Sections 3 and 5. Analyses and tests to date indicate that no major modification is necessary to the design of the release-deployment mechanisms or to the ground deployment test equipment.

This does not preclude modification of MTA-4A as a result of ground tests of MTA-4A in Phase III. A summary of margins of safety on hinges, latches, and diagonal strut is shown in Table II-8.

The electrical power system selected provides a lightweight system with an acceptable power loss. The baseline solar-cell stack assembly consisting of a 3-mil coverglass and an 8-mil cell will have acceptable radiation degradation characteristics for the 350-day reference trajectory. Section 4.6 of this document presents the radiation analysis. Mechanical stress on the cell-to-interconnector cell joints and the electrical resistance will not adversely affect array performance. Magnetic intensity caused by currents in the solar cells and interconnectors was reduced from 200 gamma to less than 1 gamma by using counter flow of currents.

Table II-7: STRESS SUMMARY FOR SUBPANELS 2 THROUGH 5 MEMBERS

Subpanel No.	Outboard Spar			Center Spar			Bottom Edge (Sta 10.5)			Lateral Spar (Sta 123.7)			Upper Edge (Sta 169.9)		
	t _{cap}	Margin of Safety	Critical Condition	t _{cap}	Margin of Safety	Critical Condition	t _{cap}	Margin of Safety	Critical Condition	t _{cap}	Margin of Safety	Critical Condition	t _{cap}	Margin of Safety	Critical Condition
2B	0.080	0.008	Sta 10.5-55.5	I1	0.025	0.05	0.035	0.003	I1	0.085	0.009	12	0.02	0.04	OP1
	0.045	0.003	Sta 55.5-123.7												
	0.025	0.062	Sta 123.7-169.9												
2M	0.02	0.09	I1	0.025	0.20	OP2	0.03	0.014	I1	0.055	0.04	12	0.02	0.12	OP1
2A	0.02	0.28	OP1	0.02	0.15	OP2	0.03	0.070	I1	0.04	0.06	12	0.02	0.17	OP3
3B	0.02	0.37	OP1	0.02	0.04	OP2	0.025	0.030	I1	0.03	0.05	OP1			
3M	0.02	0.42	OP1	0.02	0.11	OP2	0.02	0.013	I1	0.03	0.05	OP1			
3A	0.02	0.45	OP1	0.02	0.14	OP2	0.02	0.09	I1	0.03	0.05	OP1			
-B	0.02	0.46	OP1	0.02	0.21	OP2	0.02	0.18	I1	0.03	0.05	OP1			
4M	0.02	0.47	OP1	0.02	0.23	OP2	0.02	0.32	I1	0.03	0.05	OP1			
4A	0.02	0.48	OP1	0.02	0.25	OP2	0.02	0.47	I1	0.03	0.04	OP1			
5B	0.02	0.37	OP1	0.02	0.33	OP2	0.02	0.66	I1	0.03	0.02	OP1			
5M	0.02	0.43	OP1	0.02	0.45	OP2	0.02	0.87	I1	0.04	0.06	OP1			
5A	0.02	0.62	OP1	0.02	0.87	OP2	0.02	0.76	I1	0.055	0.05	OP1			

NOTES: 1. The above summary represents the strength requirements of the basic members.
 2. All values are based on an ultimate factor of safety of 1.25. Limit load is 1.5-g sine vibration. 0.02-inch channel gage used with indicated cap gages (t_{cap}).

NOTATION:

I ---In-plane panel vibration
 OP---Out-of-plane panel vibration
 1 ---Cap tensile stress
 2 ---Cap compression yield or crippling stress
 3 ---Channel crippling

Sequence	Shear Force		Diagonal Strut Load F_S	Hinge Moment	
	Locking Hinge F_1 (lb)	Rotating Hinge F_2 (lb)		Locking Hinge (in. - lb)	Panel 1 - 2 Hinge (in. - lb)
1 a	-	10.2	177	-	-
1 b		11.2	193		
2 a	16.4	9.2	96	1680	1680
2 b	17.9	10.1	105	1840	1840
3 a	5.7	4.6	81	585	1710
3 b	6.2	5.1	88.5	640	1870
4 a	3.37	3.3	22.8	715	1240
4 b	3.68	3.61	24.9	782	1360

a. Based on $\dot{\theta}_1 = 0.032$ rad/sec (D2-113355-3)

b. Based on $\dot{\theta}_1 = 0.035$ rad/sec (final design limit load)

Sketch is for Sequence 3

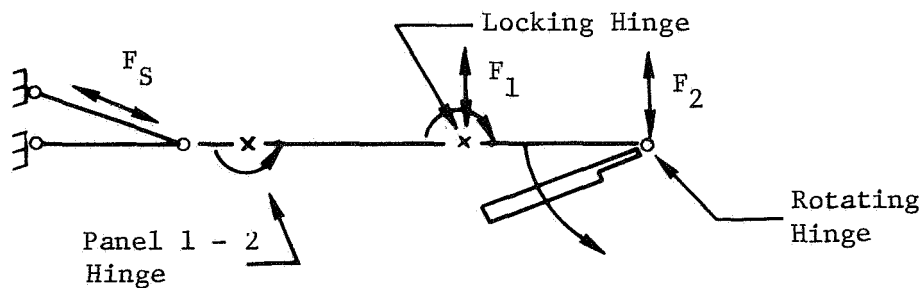


Figure 2-9: SPACE DEPLOYMENT LOADS

Table II-8: MINIMUM MARGINS OF SAFETY HINGES, LATCHES, AND DIAGONAL STRUT

<u>Component</u>	<u>Margins of Safety</u>	<u>Critical Item</u>	<u>Predicted Failure Mode</u>	<u>Remarks</u>
Main hinge (Panel 1-2)	0.17	Latch link	Compression buckling	
Main hinge (Panels 2, 3, and 4)	0.17	Latch link	Compression buckling	Same links are used on all main hinges; loads are reduced on outboard hinges.
Auxiliary hinge (Panel 2)	0.33	Latch link	Compression buckling	
Auxiliary hinge (Panels 3, 4, and 5)	0.33	Latch link	Compression buckling	Same hinges are used on all panels; loads are reduced on outboard hinges.
Diagonal strut	0.08	Main strut member	Compression buckling	

All components are critical for deployment loads

Quality assurance problems have been resolved. The development and refinement of specific inspection techniques expedited the detection of defects. The use of systematic hardware records are a principal part of the configuration control package.

The failure mode and effects analysis of the array is suitable as an initial baseline for future flight-article analyses. The complete failure mode and effects analyses are described in detail in Boeing Document D2-114081-1. Reliability prediction calculations indicate that an adequate design concept has been achieved.

The final weight breakdown is summarized in Table II-2.

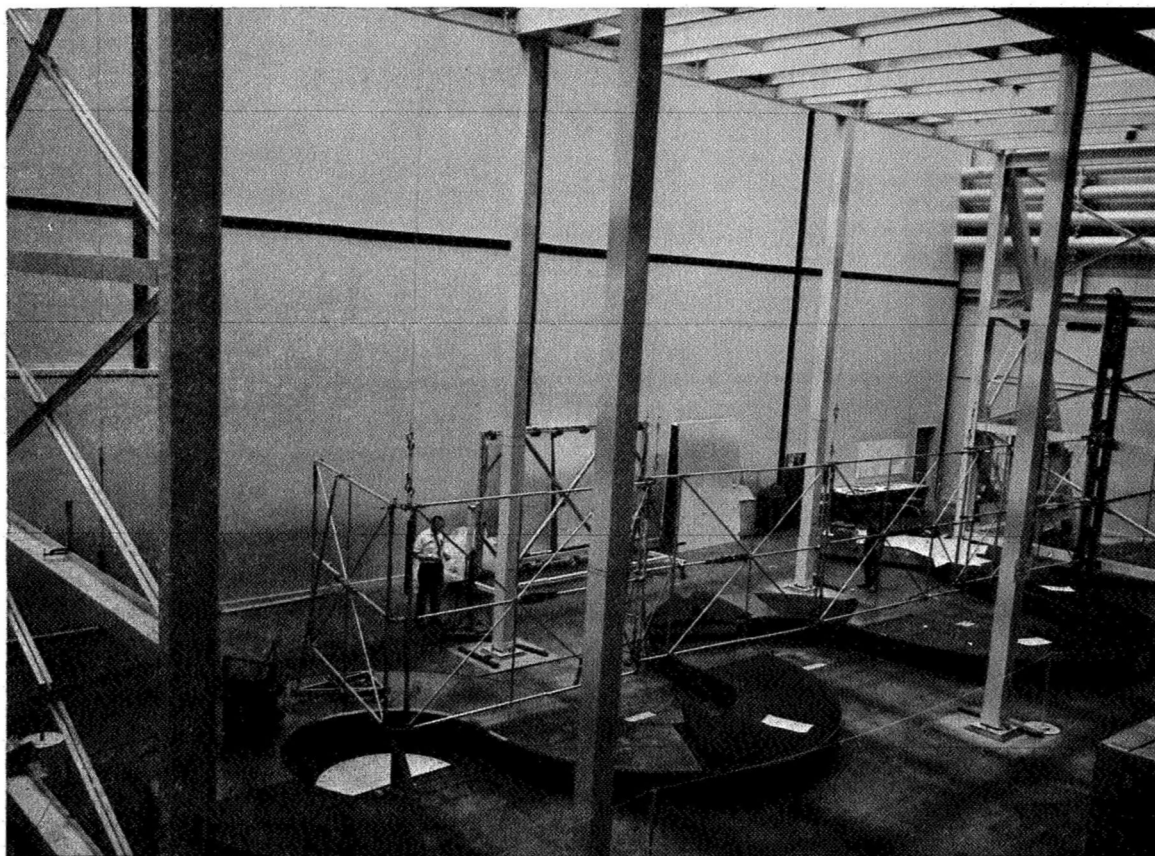


Figure 2-10: DEPLOYMENT VERIFICATION EQUIPMENT

2.3 GSE DESIGN SUMMARY

The simulated zero-g ground equipment concept has been proved successful.

The ground equipment designed and fabricated for Phase II deployment testing (see Figure 2-10) permitted the deploying subpanels extreme freedom of motion in the horizontal plane. The minimum-friction air bearing undercarriage supported the weight of the undeployed subpanels, yet was sensitive to very small changes in forces that would influence the deployment of a similar array in space. Planned Phase III modifications, such as the addition of overhead air bearings to support the deployed panels, will further enhance the ability to obtain test data under simulated zero-g conditions.

Handling and transportation equipment was provided in Phase II to transport MTA-1 between the manufacturing and test areas. Tests to establish fit and function, and to determine equipment performance under typical operating conditions were conducted prior to use with the beryllium test article.

Preliminary designs produced in Phase I for equipment required to support Phase III operations were updated and completed to a greater level of detail. The designs are based on a 13 beryllium subpanel MTA-4 configuration.

2.4 MANUFACTURING SUMMARY

MTA-1 fabrication and assembly verifies that manufacturing capabilities are developed to the main subpanel level.

Successful completion of the fabrication and assembly of the largest and most complex subpanel of the solar panel, shown in Figure 2-11, completed the manufacturing contract requirements and program objectives for Phase II.

Full-scale tooling has been proved and will be usable on Phase III by making revisions required by the different configuration of the remaining panels and minor tooling improvements.

The manufacturing techniques developed in Phase I have been proved under full-scale conditions, and, except for the final structural bond operation and some process refinement, will be applicable to Phase III.

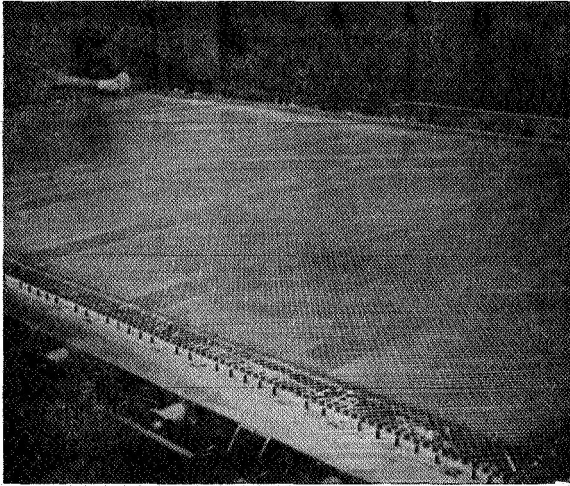
Forming of 8-foot-long beryllium channels to straightness tolerances of 0.002 in 10 inches and waviness or springback amounting to less than 0.005 was accomplished on approximately 80 parts. The forming rate, however, with present tools and equipment, is too slow for production quantities. Tooling capable of forming or hot-sizing several parts during each 4-hour forming cycle will be required for quantity production. Procurement of stock-length beryllium material to net widths for caps and shear webs and cutting to length after chem-milling has proved economical and practical.

The titanium fitting fabrication sequence developed for Phase II minimized machining time by utilizing full mill-cutter capability without distortion of thin-walled parts.

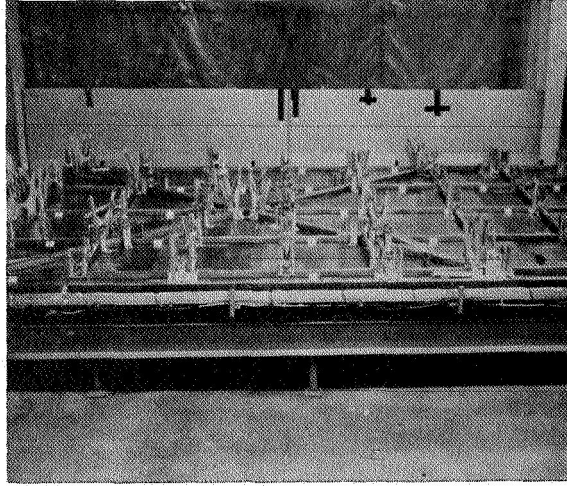
Bonding of 58 tubular beryllium bonded stick assemblies to the required straightness and cross-section tolerances was accomplished with no scrap rejections.

Bonding of these stick assemblies into Sunside and darkside frames with gussets, splice plates, and corner clips proved both the pneumatic pressure and spring pressure bonding methods. The darkside splice plate and clip bonding fixture is shown in Figure 2-11. Pneumatic pressure was applied by a silicone rubber pressure diaphragm and spring pressure was applied through metal plates and silicone rubber pads.

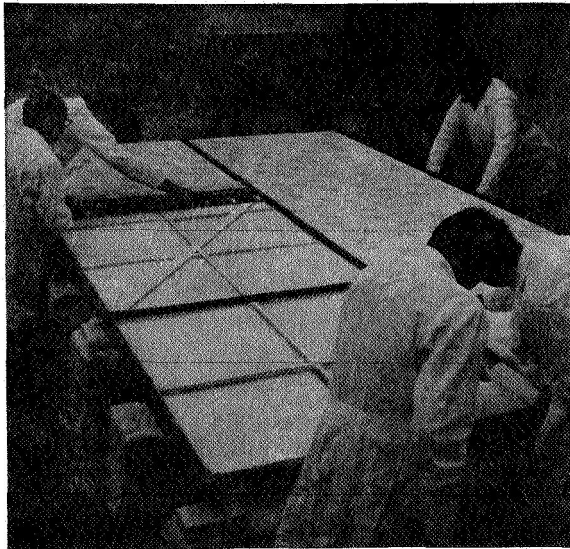
Full-scale assembly and bonding of fiberglass tapes to make the solar-cell substrate demonstrated the need for additional locators to position the long tapes within drawing tolerances. Expansion of the steel plate during the 350°F vacuum bag cure cycle pretensions the tape on curing. The substrate assembly tooling is shown in Figure 2-11.



Fiberglass Tape Layup---
Substrate Assembly



Splice Plate/Clip Bond---
Darkside Frame Assembly



Solar-Cell Module
Installation---MTA-1



MTA-1

Figure 2-11: MTA-I FABRICATION

Final structural bonding of the two subframes to the substrate using atmospheric bonding pressure resulted in a good bond. During the cure cycle, however, the channel-to-cap bonds on 19 of 36 intercostals failed due to unplanned side loads and excessive temperatures. Tests made to determine the allowable loading of previously cured bonds during subsequent cure cycles indicated a minimum safety factor of four. Tests made subsequent to the failure showed that bond strengths for sustained loading at bonding temperatures are approximately one fourth the standard pull-test strengths at these temperatures.

The bond failure precipitated a program for development of structural repair techniques. The results of this effort are perhaps the most valuable manufacturing information obtained during Phase II. The application of local rapid heating to disassemble bonded structure without damage to parts or adjacent bonds, plus the subsequent preparation and rebonding techniques, has advanced the practical aspects of this type of structural bonding significantly.

Assembly of solar cells into full-size submodules using the pulse-soldering techniques from Phase I verified the Phase I development work. Installation of these submodules was accomplished with a much-simplified method over that used in Phase I. The cells were applied to the substrate Sunside up, as shown in Figure 2-11, rather than the panel being applied to cells Sunside down. This permitted simple tooling constructed of urethane foam with Styrofoam backing and thin masonite hard surfacing in contact with the cell faces. One psi of atmospheric pressure applied by a vacuum bag produced a good bond.

Installation of mass-simulated bus bars and soldering of interconnections and diodes completed the panel assembly ready for testing.

Detail manufacturing cost data has been obtained throughout Phase II, and the cost of tooling and specific categories of parts has been established. The high-cost areas, as expected, are in the forming tooling, the bonding tooling, the beryllium fabrication, and the structural bonding.

2.5 SYSTEMS TEST SUMMARY

The activities in Phase II centered around the experimental fabrication and testing of test articles representing components of systems or system segments. The purpose in performing these tests was to collect data and determine the characteristics of the selected segments of the solar array design. This information is used to improve the performance of the array during Phase III testing.

The tests are divided into five groups or system segments as follows:

- 1) Mechanisms small component tests;
- 2) Ground support equipment tests;
- 3) Deployment mechanisms tests, including the determination of the effect of atmosphere on deployment;
- 4) Vibration testing of Subpanel No. 1;
- 5) Vibration testing of the Subpanel Sample SCS-43 fabricated in Phase I.

Each group is reported separately. The purpose of the test, its conduct, results, conclusions, and recommendations are summarized.

2.5.1 TESTS OF SMALL COMPONENT MECHANISMS

Detail design data obtained during small component mechanisms tests was necessary to develop the mechanism design.

The purpose of the small component test series was to provide development test data and determine characteristics of mechanisms systems and their components. Test data were obtained in both Earth-ambient and thermal-vacuum environments for deployment mechanisms. These tests were accomplished by installing prototype developmental mechanisms on small-size deployment subpanels. Hinge frictions were measured for both normal and deflected panels. Deployment cable and quadrant frictions were measured. Test results included the values for friction forces required to open the solar panels, determination of friction forces between the deployment cable and quadrants, and operating characteristics of the deployment system in a thermal-vacuum environment. (Station 10.5 boost tiedown components were selected and mounted in a suitable fixture for test.) Boost tiedown circumferential cable release system elements operated flawlessly, and the circumferential cable exited from the simulated spacecraft at velocities of 400 inches per second. There was no tendency to hang up or whiplash the adjacent panels even with release delays of 0.10 second between the two cable ends. The center spar release clamp required redesign and an increase in Nomex cable diameter to meet all installation requirements.

It is concluded that presently designed mechanisms will perform satisfactorily throughout a deployment sequence. Station 10.5 boost tiedown system delivered adequate performance but a total six-station system must be tested to determine any system degradation due to launch vibration and thermal-vacuum environments.

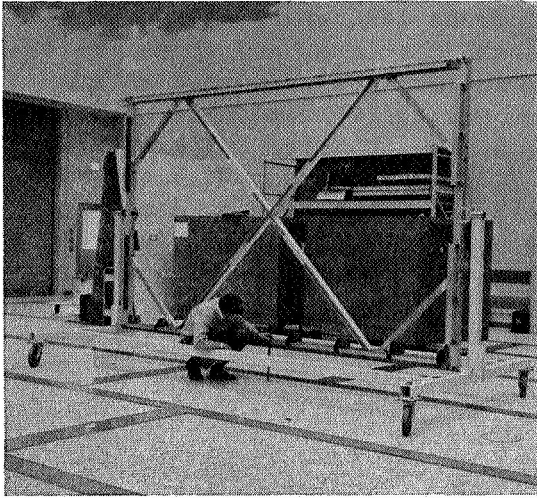
It is recommended that the tooling plan for Phase III be reviewed and improvements made to meet the required alignment tolerances for hinges and latches. It also is recommended that a full-boost tiedown system be constructed, subjected to a launch vibration environment, and release tested as a total system.

2.5.2 GSE TEST SUMMARY

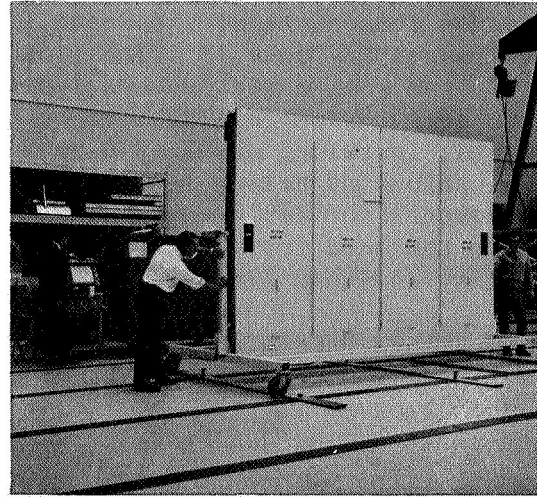
The GSE test program ensured satisfactory equipment functional performance prior to first use.

The subpanel handling frame and dolly were checked for fit and function with MTA-3 Subpanel No. 1. Handling frame deflection under load and shock loading experienced during dolly operation were determined by test before use with MTA-1 (see Figure 2-12).

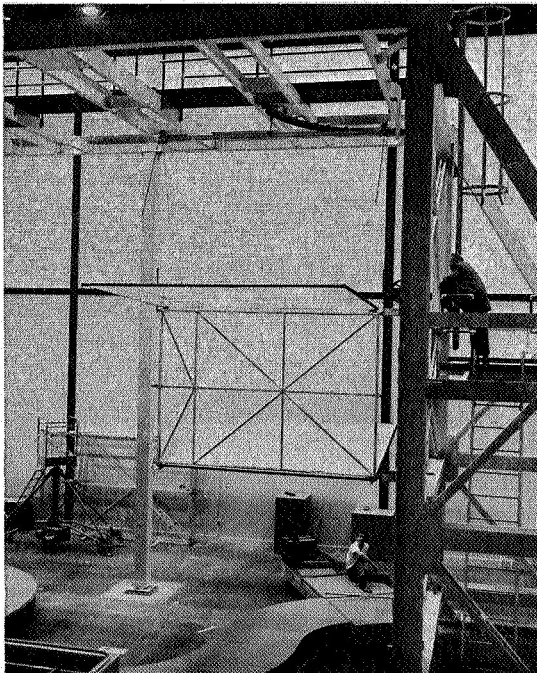
Comprehensive tests were conducted on the main subpanel deployment verification fixture to calibrate system instrumentation, set equipment



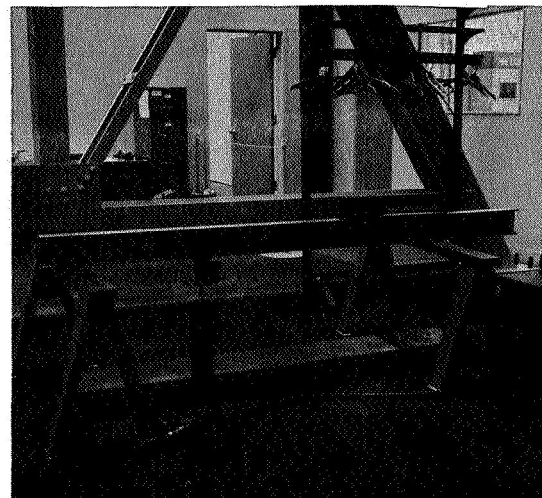
Handling Frame
Deflection Test



Handling Dolly Shock
Loading Test



Demonstration Equipment
Auxiliary Subpanel
Deployment Test



Demonstration Equipment
Carriage Test

Figure 2-12: GROUND SUPPORT EQUIPMENT TESTS

operating limits, and demonstrate proper equipment performance before the start of formal MTA-3 deployment testing.

Selected components of the Phase III deployment demonstration system were tested in Phase II. The overhead track, carriage, and spreader-bar system was fabricated and a series of deployments performed with a test panel. A set of lightweight aluminum auxiliary subpanels was fabricated, installed on MTA-3 Main Subpanel 4, and deployed using several combinations of helper springs, counterweights, and dampers to supply and control the deployment force. The results of these tests demonstrate the need for additional deployment demonstration design and test activity in Phase III.

2.5.3 MTA-3 MECHANISM AND AERODYNAMIC TESTS

Experience during deployment tests on the mass simulated solar panel (MTA-3) provided valuable knowledge of the mechanisms design characteristics.

The purpose of the mechanism and aerodynamic parts of the MTA-3 test program was to:

- 1) Obtain developmental operating data on array mechanisms for design application;
- 2) Develop the simulated zero-g deployment test equipment and procedures and provide data to compare to the deployment analysis of the Phase II aluminum deployment model;
- 3) Determine the significance of aerodynamic effects on deployment test results for application to the future ground deployment testing of MTA-4.

The tests were performed with and without plastic membranes on the panel frames. Each test was conducted for two conditions. One condition, to obtain time-history data, was started with zero cable tension and zero panel velocity. The second condition imposed a controlled closing moment on one panel to duplicate continuous deployment velocities. Tests were conducted to obtain operating characteristics with a deflected end spar simulating deflection due to thermal gradients.

The test program provided operating data and deployment dynamic response parameter information valuable for comparison to analytical values used previously. The aerodynamic test results shown in Figure 2-13 were in very close agreement with the bare-frame results. Unexpectedly high friction in MTA-3 masked test data that should have shown small differences on outboard deployment sequences. Performance of many detail hardware items could be observed and evaluated and resulted in minor design improvements. Interferences were discovered and corrected. Alignment of hinged elements required improvement.

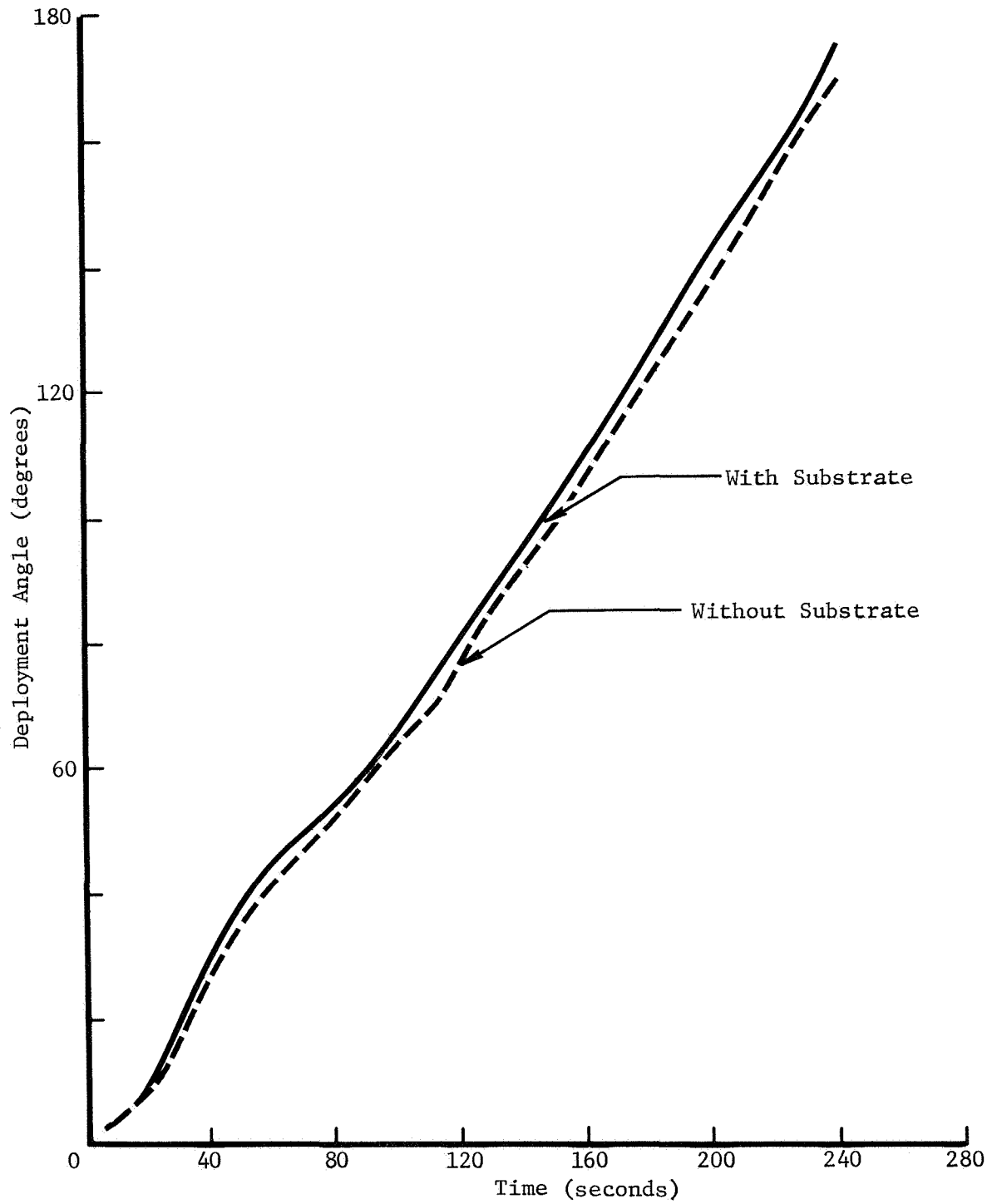


Figure 2-13: EFFECT OF SUBSTRATE ON MTA-3 DEPLOYMENT TEST
(Typical Run)

It was concluded that the mechanisms as revised are suitable for space deployment. Improved tooling must be constructed to ensure hinge center-line alignment on assembly.

2.5.4 VIBRATION TEST OF SUBPANEL NO. 1

Vibration testing of Subpanel No. 1 provides information on the dynamic characteristics of beryllium subpanels.

The purpose in running the vibration tests on MTA-1 was to evaluate the damping characteristics in air and partial vacuum, and determine the resonant frequencies in air and a vacuum for use in comparison with predicted calculated values.

The tests were conducted in the Boeing 50,000-cubic-foot vacuum chamber in air and at 1/20 atmosphere.

Diagnostic sweeps of input and response accelerations and strains were obtained for bending and torsional modes in air and vacuum up to 50 Hz.

Decaying oscillation records for two bending and one torsion mode were obtained. One bending mode was selected and the excitation level raised to obtain substrate accelerations comparable to the maximum expected for the stowed array. Input and response data were obtained through the selected resonant band.

The results of the vibration tests show acceptable correlation between the measured and calculated resonant frequencies for the first bending and torsion modes in vacuum. The measured frequencies in air did not agree with the calculated frequencies. The proximity of the vacuum chamber walls is considered a possible cause for the disagreement.

Energy dissipation capability of the subpanel is very encouraging. Measured load magnification in a vacuum averages a Q of 18 at the lowest bending frequency and a Q of 29 at the second bending mode. The substrate sustained a peak value of 37 g's and the structure sustained a maximum stress level of 9,500 psi with no visible damage.

2.5.5 SCS-43 SUBPANEL VIBRATION TESTS

Air and vacuum environments made significant differences in the SCS-43 subpanel dynamic response characteristics.

The 4-foot-square small component structure (SCS-43)(see Figure 2-14) subpanel with two support configurations (clamped and pinned) was subjected to vibration tests in air and vacuum (40 torr). Appreciable reduction of damping and an increase in the substrate frequencies was observed with the air pressure reduced from ambient to 40 torr for both

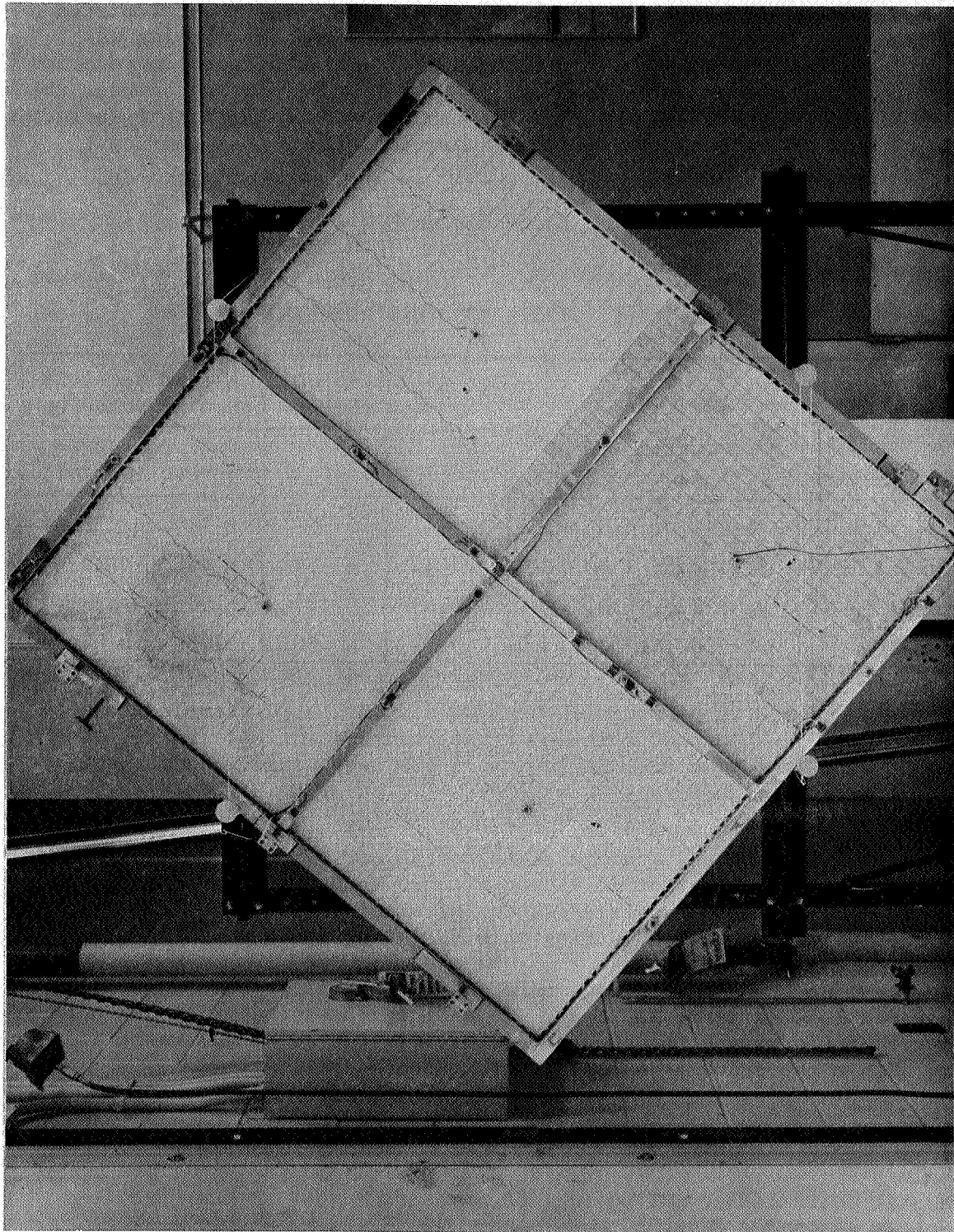


Figure 2-14: SCS-43 SUBPANEL

support configurations. Corrections for the accelerometer weights and an assumption of the air inertial effect were made for the calculated frequencies. Comparison of calculated and measured frequencies was acceptable for the low-frequency band but did not show good correlation for the high frequencies. The storage time (8 to 9 months) between the last test in Phase I and the Phase II test did not significantly affect the dynamic characteristics of the panel, indicating no significant creep in substrate.

Results of the four vibration tests run on SCS-43 are as shown in Table II-9.

Table II-9: VIBRATION TEST RESULTS

Description	Normal Input Test Acceleration(g)	1st Mode Resonance Frequency (Hertz)		Maximum Acceleration Response (g)*	
		Sub- strate	Inter- costal	Sub- strate	Inter- costal
Clamped in Air VA-1	1.0, 3.0, 4.25	30-40	30	38	13
Clamped in Vac VV-1	0.5, 1.0, 1.2	42-51	42-51	65	20
Pinned in Air VA-2	1.0, 3.0, 4.25	30-41	31	90	32
Pinned in Vac VV-2	0.55, 0.75, 1.1	40-50	40-50	105	59

*Resonances were in a band with maximum response at different frequencies, dependent on the position of the instrumentation. In air, the peaks were less discrete; in vacuum, there were usually three prominent peaks in the band.

Failure of a previously damaged intercostal cap occurred before the pinned configuration test in vacuum with no apparent major change in the panel dynamic characteristics during subsequent tests. Damage to seven live cells occurred during this series of tests.

Power output measurements made before and after all vibration tests showed a 1.6% decrease in power. It was concluded that this did not significantly affect the electrical characteristics of the panel and was within the accuracy of the instrumentation.

2.6 QUALITY ASSURANCE SUMMARY

Program surveillance was accomplished as stipulated in the "Detailed Quality Assurance Plan."

The "Detailed Quality Assurance Plan" was released in February 1967. Revision A was published on May 13, 1967. This plan has required only

minor revision with contractual changes affecting the document the most. Briefly, this plan explains four basic subjects: (1) Boeing quality assurance capabilities; (2) the quality philosophy to be used on the LASA program; (3) requirements to be satisfied in complying with LASA quality philosophy; and (4) those selected LASA program areas where varying degrees of quality assurance would be applied. A second revision to the plan was released in October 1968.

Procurement controls adequately supported the program. One of the most important and definitely the largest receipt was raw beryllium stock. At one time the acceptance of 120 pieces of beryllium was in doubt until a questionable surface condition was corrected. One order of beryllium with surface defects was returned to the supplier. The next order was rejected in his plant before shipment and held until surface defects were corrected. All beryllium material corrections were made within schedule requirements.

The beryllium surface conditioning supplier represented the other main procurement. Almost every beryllium part was shipped between the supplier and Boeing at least twice for surface chemical milling and/or cleaning and priming. A formal survey of this supplier had been performed early in the program. A Boeing quality source representative also was assigned to provide surveillance at this supplier's plant. Formal documentation including supplier and Boeing historical and discrepancy records were used for processing parts.

Informal surveys were conducted at the titanium "shear tooth" and motor cable drum suppliers' plants prior to start of LASA work at these locations. Hardware received from these suppliers represented excellent workmanship and met required specifications.

New and revised line inspection techniques and procedures developed and proved in Phase I were readily used during Phase II operations. These included solar-cell and cover glass receiving and processing sampling plans, combination penetrant and eddy current inspection of beryllium parts, and radiographing of titanium weldments. Aerospace material and process personnel utilized experimental harmonic adhesive bond testing equipment on MTA-1 to help locate broken bond joints. More information on this subject is found in Section 4.6.

A total of 630 manufacturing shop instructions were reviewed and approved by Quality Control. Required tests, delivery of test reports, and certifications also were conducted by the quality control service laboratories on and before use of materials and supplies used in LASA MTA-1 fabrication.

Numerous quality control trend data reports have been submitted in the past. This information reported conditions in the Task 3 MTA-1 operations or concerned particular materials and parts (i.e., solar cells, beryllium, etc.). Figure 2-15 summarizes rejection reports written on all Phase II hardware and compares this with the particular process of function taking place when the original rejection was written. This figure also identifies final report dispositions summarizing them into "use-as-

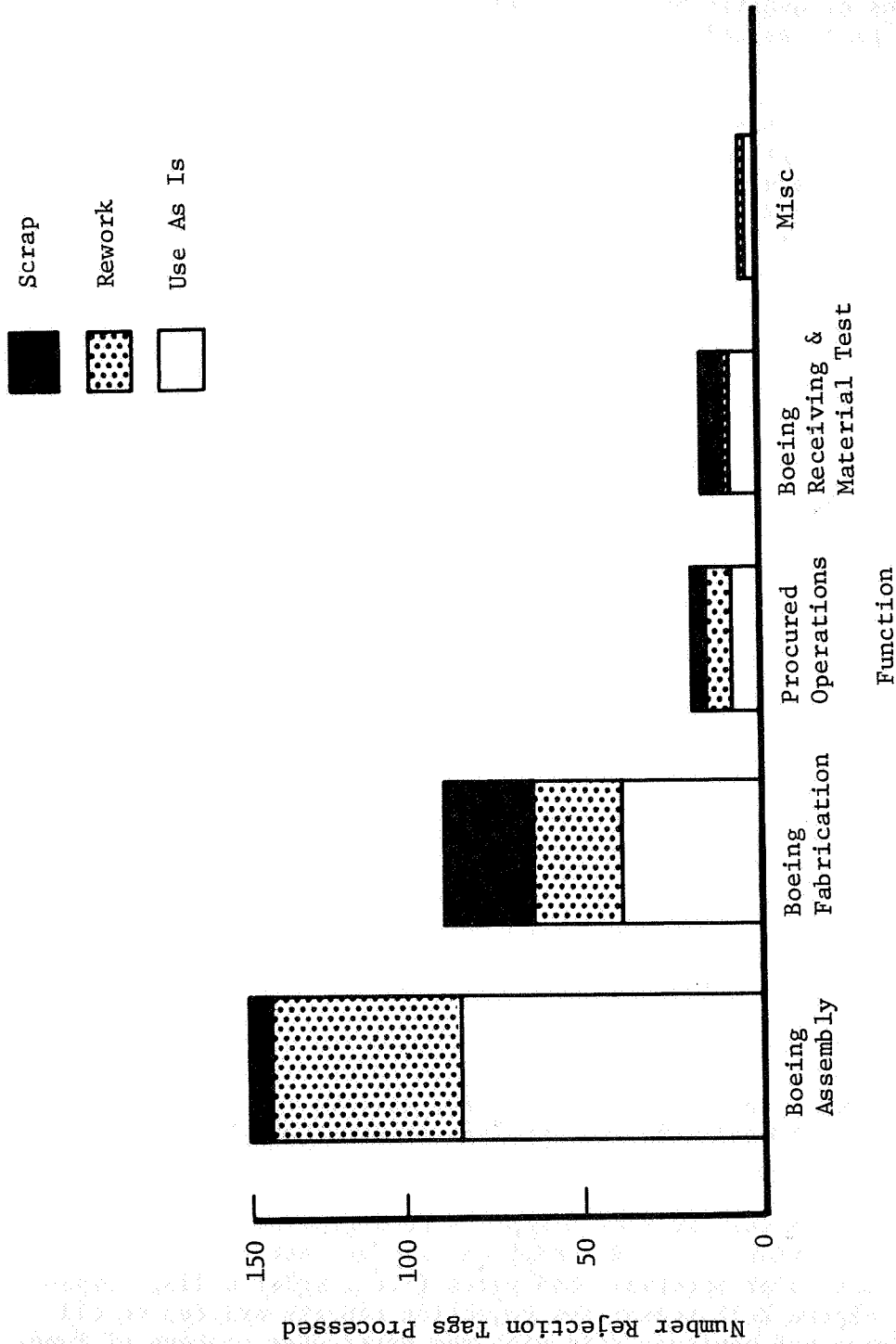


Figure 2-15: TOTAL * LASA PROGRAM REJECTIONS VERSUS FUNCTIONS (* Comprises Tasks 2 through 8)

is," "rework," or "scrap." One third of all rejection reports were dispositioned "rework." This resulted in successfully correcting defects, allowing parts to be accepted. Thirteen percent of all reports, including solar cells, were dispositioned as scrap. A total of 15 beryllium and 4 aluminum parts were scrapped.

2.7 PRODUCT ASSURANCE SUMMARY

Significant progress has been made in Phase II toward assurance of a satisfactory LASA design.

Phase II experience has continued to back up the predictions and analyses of Phase I. Accordingly, revision of reliability prediction and failure mode and effect analysis documents has not been considered necessary. A brief summary of specific items is as follows:

- 1) The discovery of blistered and peeled ohmic contacts on the customer-furnished cells caused the use of these cells on MTA-1 to be suspended pending resolution of the problem with the cell supplier.
- 2) Explosive-actuated cable cutters proposed for the MTA-1 final tie-down release were used as test devices on SCM 30-2. It became apparent that this particular supplier's products should not be used on flight articles. This is discussed in Section 3.2.5.
- 3) A failure mode and effect analysis (FMEA) was prepared for handling and deployment equipment. The analysis identifies those failure modes that could cause damage to test articles or cause invalid test data. It includes consideration of human error effects and externally caused conditions. See Section 6.4.
- 4) The Phase II testing showed that the viscous dampers would function in a vacuum of at least 10^{-8} torr at temperatures below -100°F . Qualification tests were completed on the motor/cable drum assembly drive unit with satisfactory results. The motor/cable drum assembly also operated satisfactorily in the SCM-1 and MTA-3 deployment tests and in a vacuum of at least 10^{-8} torr and a temperature of -63°F .

Continued attention to details of design and installation involving dimensional controls, spring forces, and friction effects will be necessary in Phase III. The Phase II tests have been extremely useful in identifying areas where reliability needs improvement.

- 5) The practical experience gained with structural repairs in Phase II has shown that disassembly and repair of segments of a full-size bonded beryllium subpanel is practical. A detailed discussion of structure repair is found in Section 4.2.6.
- 6) Some insight also was gained into the load-sharing behavior of the structure when the 0.010 gage intercostal cap fractured during

SCS-43 vibration testing. The structure was able to tolerate this fracture without damage to other members, thus increasing the confidence in the design.

2.8 COST AND SCHEDULES

All Phase II objectives were accomplished and some additional development efforts were initiated, within the Phase II budget.

The Phase II program was completed 12% under the planned cost (see Figure 2-16). This was made possible by careful planning and monitoring of each milestone event, and its allocated budget. By completing the Task 4 effort ahead of schedule, we were able to, with the direction of JPL, achieve additional deployment equipment development.

The task milestones of the eight tasks are shown in Figure 2-17.

Task 1, the Program Plan, was completed on schedule.

Task 2, Small Component Mechanisms, Fabrication, Test, and Analysis---Some retesting of the tiedown release system was conducted after the original tests proved that some design changes were required.

Task 3, MTA-1 Fabrication, Test and Analysis---Manufacturing problems in meeting the close design tolerances and bond failure during the final assembly bonding operation forced a 6-week slide to the MTA-1 fabrication completion date. However, this schedule slippage was recovered before the end of the program. The electrical tests on MTA-1 were deleted by the Contract Change Notification (CCN).

Task 4, Panel Assembly Deployment Model Design, Fabrication, Test, and Analysis---Much of the activity in this task was completed from 1 to 12 weeks ahead of schedule. The track and sled development tests proved the adequacy of the original design and eliminated the need for track redesign and modification. The accompanying chart shows the event deviations from the original schedule.

Task 5, GSE and Facilities---Most of these events also were finished ahead of schedule.

Task 6, Documentation---All documentation and reports were delivered as scheduled.

Tasks 7 and 8 were initiated by a Contract Change Notice. This change replaced the live solar cells on MTA-1 with dummy cells and deleted the power degradation tests on MTA-1. An additional test program on the SCS-43, a 4-foot-square test panel built in Phase II, was run to obtain vibration characteristics of the beryllium subpanel model in air and vacuum for pinned and clamped support conditions.

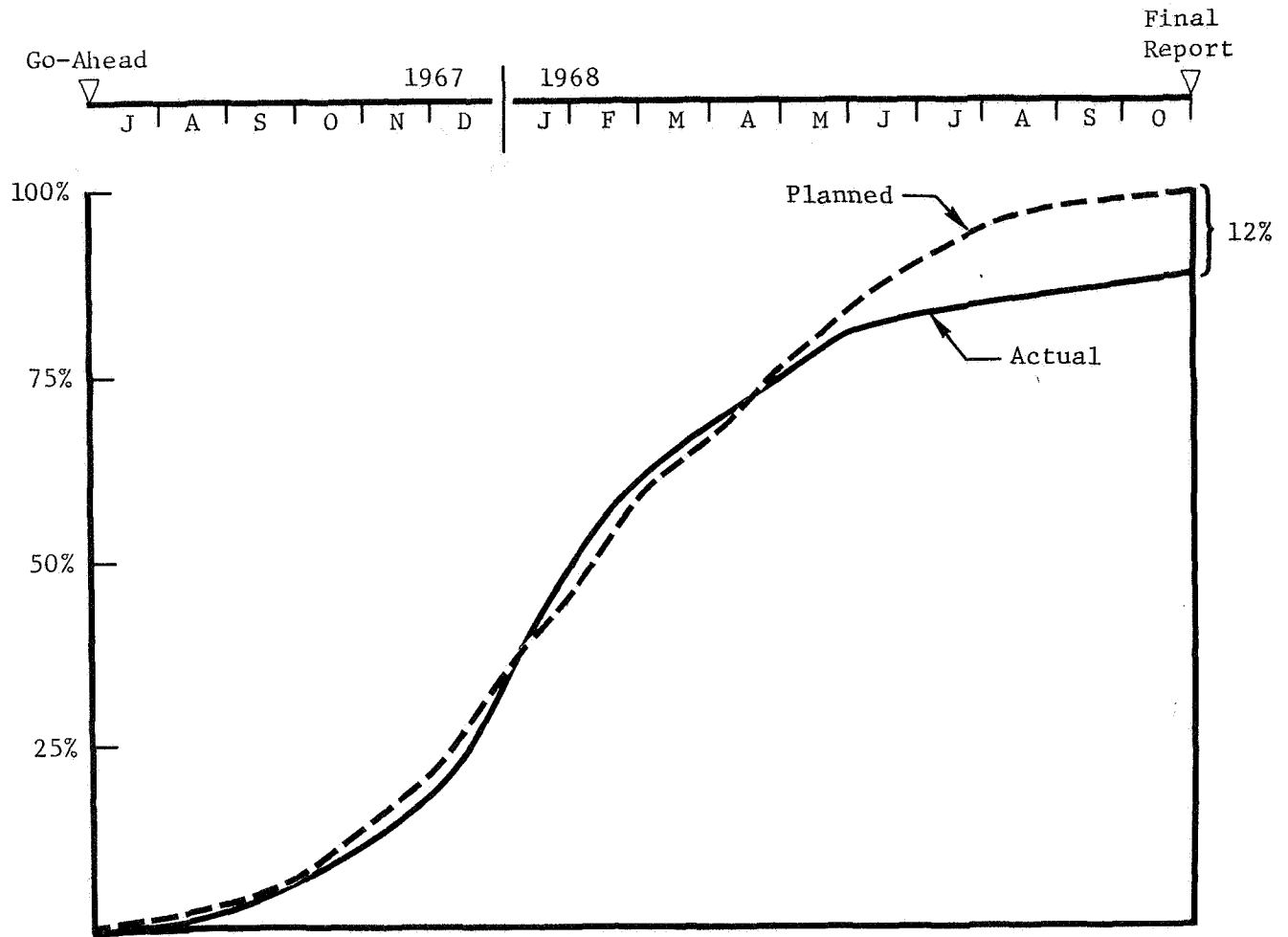


Figure 2-16: LASA PHASE II COST PERFORMANCE

2.9 MATERIEL SUMMARY

Procurement activities were accomplished as outlined in the procurement plan.

Program hardware, components, assemblies, and services were procured within the budget allocated. The Materiel budget was approximately 16% of the total program dollars. Cost savings in the beryllium procurement, deployment fixture detailing, fabricating and erection, and other outside production orders allowed procurement of additional requirements without increase to the program budget. A breakdown of total Materiel expenditures by hardware elements is shown in Figure 2-18.

All procurements except raw material and standards were made by a LASA program Materiel representative. Procurement of raw material and standards was monitored by the Materiel representative to ensure proper schedule and cost feedback to program management.

Control of inventory was accomplished by close coordination with Material Control, Traffic and Stores personnel. Representation in other areas such as change board, make-or-buy and program reviews was provided.

2.10 PROGRAM CONCLUSIONS AND RECOMMENDATIONS

Phase II experimental fabrication of test articles and test results provide adequate data to proceed with the fabrication and testing of MTA-4A in Phase III.

Major conclusions are briefly summarized in this section. More detailed recommendations are collated in Section 9.0 of this document.

It is concluded that:

- 1) The technology advancements made in Phase II provide the necessary engineering information required to proceed with the final phase of the program, with confidence.
- 2) Assuming a dynamic load magnification factor, Q , of 15 or less exists in the present design, the goal of 20 watts per pound can be achieved. Testing of Major Test Article No. 1 in Phase II indicates that a Q of 15 is achievable. However, MTA-4 vibration testing of a full simulated solar array, as proposed for Phase III, is required to verify the LASA array, Q .
- 3) Analysis and component level testing show that the design is satisfactory, meeting or bettering the requirements of the Large Area Solar Array specification. The results of the Phase III system and environmental tests remain to be completed to verify the program requirements.
- 4) The experimental fabrication of MTA-1 has shown that the technology for manufacturing a bonded beryllium and titanium frame is available. Based on Phase II problems experienced, Phase III will implement improved process reliability in final structural bonds.
- 5) Mechanism development has made satisfactory progress; however, beryllium subpanel deployment rigging procedures and system operations for the boost tiedown and release must be verified in Phase III.
- 6) Deployment tests identified the faults in the array deployment mechanisms and the fixture and established the criteria for solutions. The deployment fixture, associated handling equipment, and operational procedures are acceptable for Phase III testing with some minor modifications.
- 7) Analysis of the structure, under dynamic boost conditions, was iterated and resulted in a better balanced detail design.
- 8) More complete understanding of several test and analytical problems, to be resolved in Phase III, has been accomplished by the Phase II test and analysis programs.

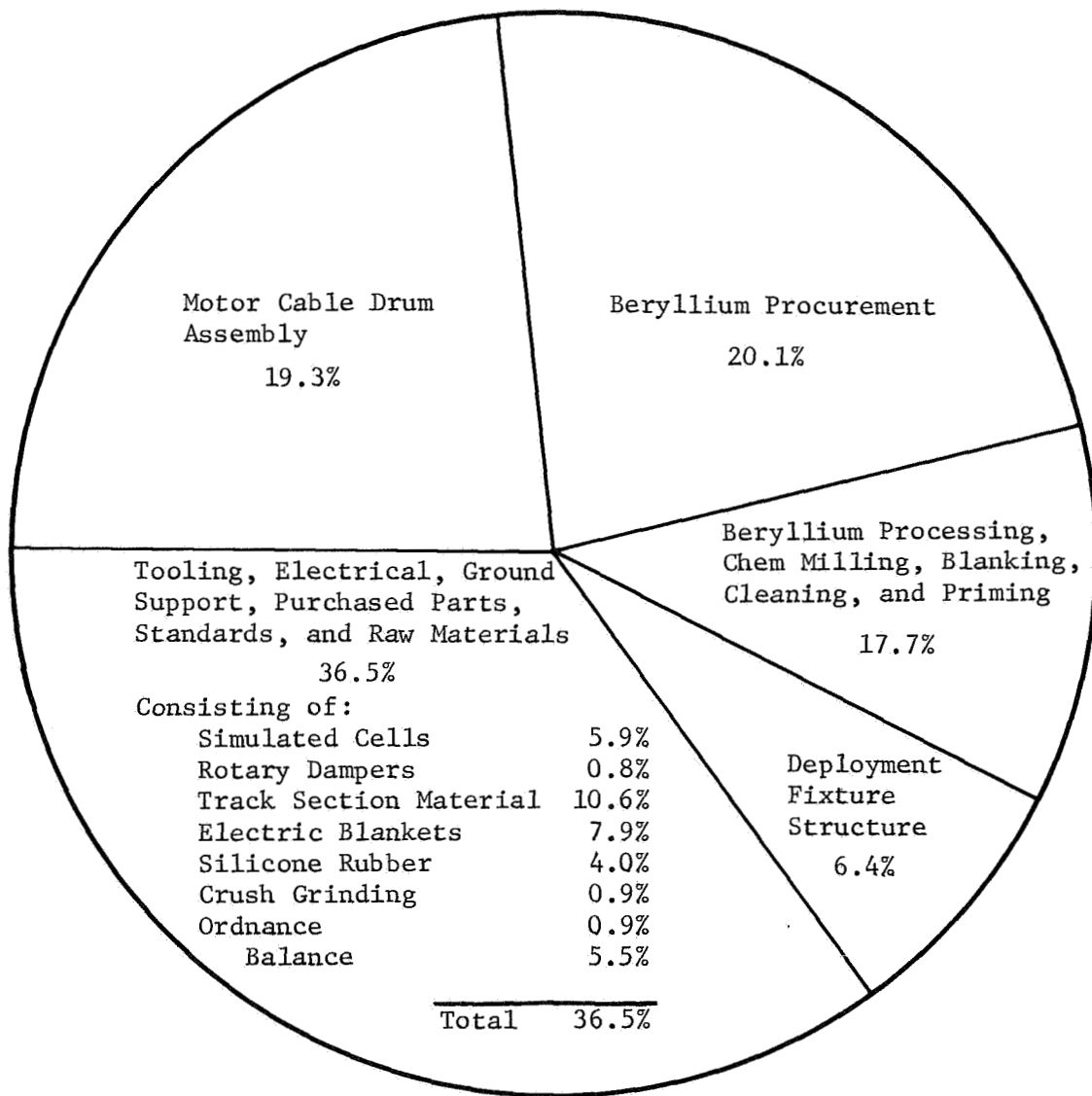


Figure 2-18: MATERIEL PROCUREMENT

Verification of the technology developed in Phases I and II remains to be accomplished. Therefore, it is recommended that fabrication of the Major Test Article required in Phase III proceed and that the tests required to meet the Phase III verification objectives be conducted. Specific recommendations include:

- 1) Test and analysis plans and procedures be developed in detail early in Phase III to ensure compatibility with specification requirements;
- 2) Tiedown and release system mechanism rigging and testing be developed early in Phase III to ensure early identification and solution of possible problems;
- 3) Vibration simulators and test fixtures be developed early in Phase III to provide sufficient schedule for necessary simulator modifications.

3.0 SMALL COMPONENT MECHANISMS TESTING

Nearly all phases of the mission expected for the array design can be duplicated in man-made environments using full-scale prototype hardware. The exception is the deployment sequence. Small Component Tests 1 and 2 (SCM-1 and SCM-2) were designed to provide tests for prototype panel deployment mechanisms and drive, within the limited space of existing vacuum chambers and without the need for simulating zero-g support. The primary objective of the small component tests was to provide design development data for mechanism components. Prior to testing, the design was based on material and process selection using supporting data developed for similar functions.

Small Component Tests SCM 30-2 and SCM 30-3 were developed to test the boost tiedown release elements. These latter tests are principally development type tests and did not include thermal-vacuum tests. Figure 3-1 is a summary of these tests.

3.1 DEVELOPMENT TESTS OF DEPLOYMENT MECHANISMS

Small-size panels that fit within existing vacuum chambers were the best means of testing deployment.

The small component test program was scheduled far enough in advance of the full-scale panel deployment tests to supply operating characteristics of the individual hardware items and thus avoid operational or fit-up problems. System tests were scheduled to provide an early test bed of the deployment system both at Earth-ambient and thermal-vacuum environments.

Deployment component testing in the SCM series refers to tests of hinge assemblies with and without spring bias and of cable/quadrant operating elements. These tests satisfied the mechanism design development objectives and increased confidence in the design of Phase III hardware. Figure 3-2 shows the location of the main components tested in the SCM-1 and SCM-2 tests.

Full-scale panels, being too large for existing vacuum equipment, make the SCM tests particularly valuable for obtaining vacuum operating information. These tests also confirm that subassembly and component operating characteristics are suitable for use on full-scale verification and demonstration tests planned for Phase III.

The test articles, test setup, and arrangement are depicted in Figure 3-3. Sizes of the aluminum panels were selected to use the medium-sized vacuum chamber for space deployment testing. Only three sample panels of the main panel deployment system were used because of space limitations and the decision that three panels will adequately represent the mechanisms of the total deployment.

Test	Results	Recommendations
SCM-1 ambient deployment tests (main subpanels)	Deployment satisfactorily performed in 10 min; motor operated at uniform rate within 570 to 600 sec	Ambient test data on main subpanel deployment accepted as satisfactory. Special attention must be given alignment of hinge lines in fab
SCM-2 ambient deployment tests (auxiliary subpanels)	Auxiliary subpanels deployed at 70°F in 79 to 100 sec; dampers maintained 12.5 in.-lb rad/sec damping coefficient at 70°F	Repeat SCM-1 recommendations for SCM-2 also
SCM-1 and -2 vacuum deployment tests (main and auxiliary subpanels)	Auxiliary subpanels deployed at -100°F and 1 x 10 ⁻⁸ torr; main subpanel deployed in 10 min, 34 sec; motor temperature was -63°F	Provide thermal coating or insulation on damper to control heating and/or cooling rate
SCM 30-2 tiedown and cable release test	Test confirmed that release method and stored energy were adequate to jettison tiedown cables; delay function demonstrated ejection without damage to adjacent panels	System must be rigged and tested as a total system for critical environmental conditions
SCM 30-3 center-spar cable tiedown and release test	Cable deflection with time becomes asymptotic at about 62% of rigged load; clamp, with wedge jaw design, is adequate to hold all stations	The new wedge design with larger (3/16) Nomex cable be used for the Phase III test program. A 90-day load decay test should be run on sample installations

Figure 3-1: TEST SUMMARY

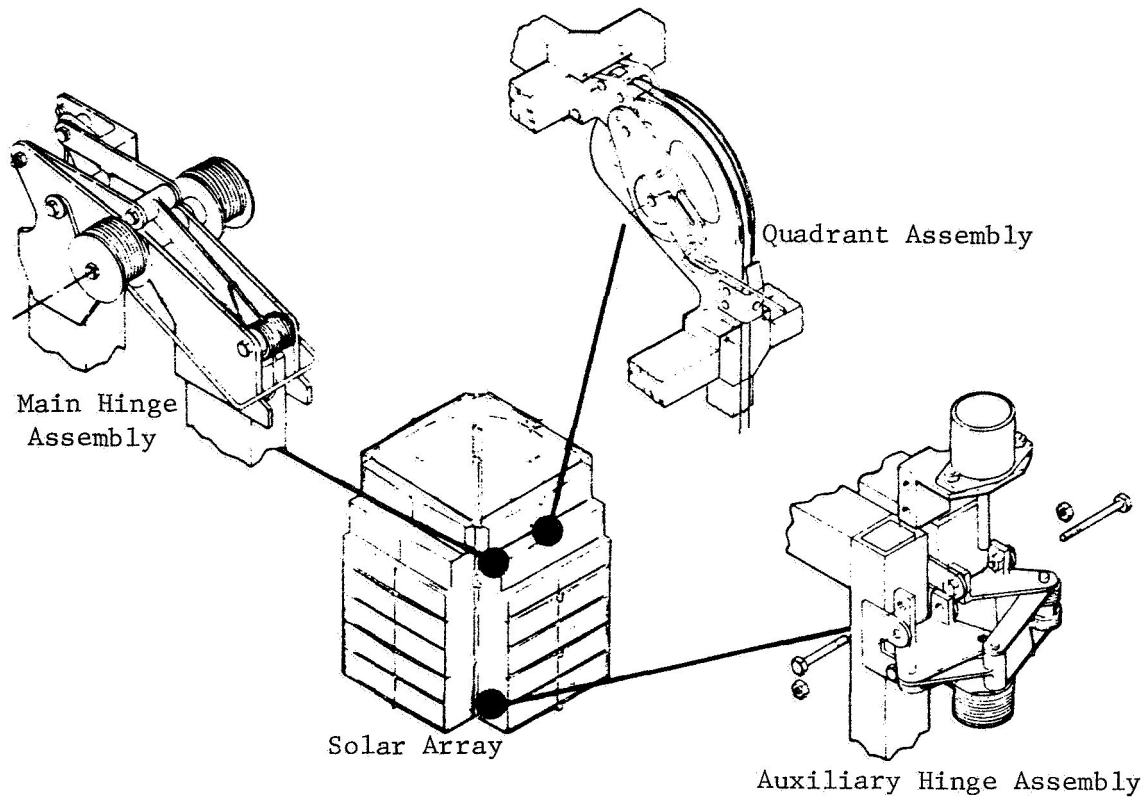


Figure 3-2: SCM DEPLOYMENT COMPONENTS

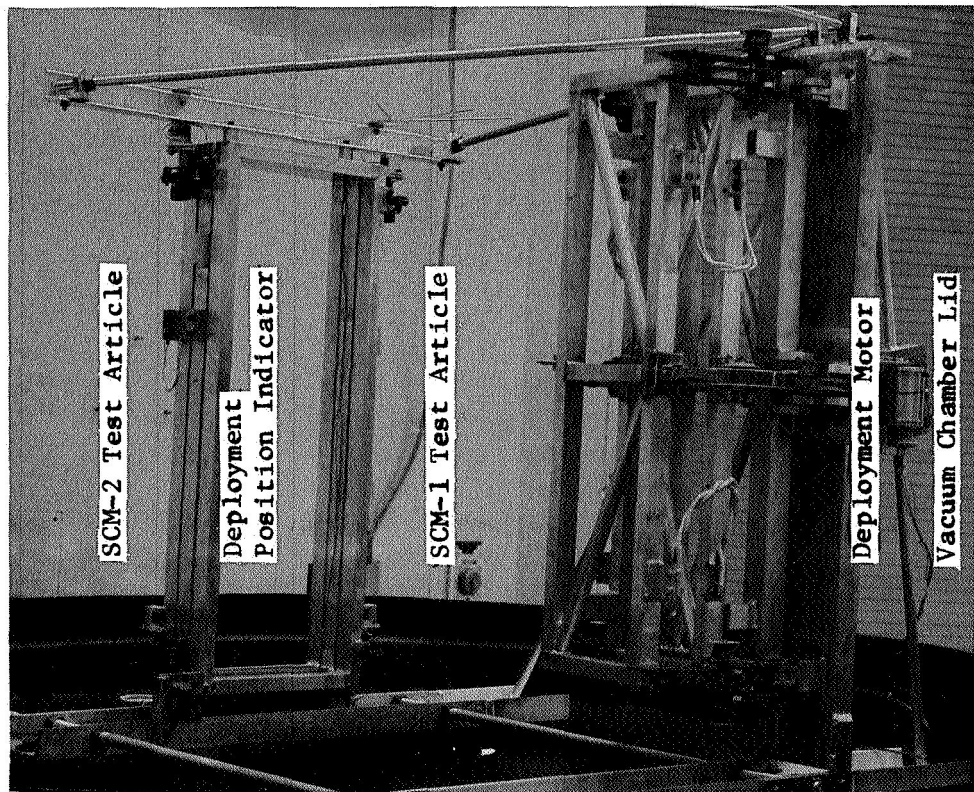


Figure 3-3: TEST SETUP, SCM-1 & -2

3.1.1 DESCRIPTION OF SCM-1 AND SCM-2 TEST ARTICLES

SCM-1 and SCM-2 test was a small-scale system using full-scale mechanisms providing a suitable test bed for ambient and thermal-vacuum testing.

The main panel SCM-1 test consists of three foreshortened panels (shown under construction in Figure 3-4), the drive motor (Figure 3-7) mounted to the test fixture, and a simulated diagonal strut to limit Panel 1 deployment travel and lock Panel 1 in position. The panels were fitted with hinges and quadrants complete with Geneva release as shown in Figure 3-2. Auxiliary panel release mechanisms were tested in the MTA-3 series tests reported in Section 5.0 rather than the small component mechanism test series reported here.

The auxiliary panel SCM-2 test consists of three foreshortened panels, four hinge sets, drive springs, latches, and dampers.

Mechanisms shown in Figure 3-3 are prototype, and the hinge assemblies are constructed to mount on plates welded on the panel frame which simulate prototype hardware mounting. Figure 3-6 is a prototype of the SCM-2 panels under construction.

All frames were made of extruded aluminum of the approximate cross section of the prototype beryllium panels. Where required, fillers are added at mounting points on mechanism/panel interface to ensure proper mechanical function.

3.1.2 DESCRIPTION OF SCM-1 AND SCM-2 TEST FIXTURES

The test fixtures simulated the spacecraft interface and mated with the vacuum-chamber lid.

The test fixtures for SCM-1 and SCM-2 were combined into one fixture as an economy measure to reduce the total time that the vacuum chamber was required. The fixture was designed to support both test articles and mate with the vacuum-chamber lid as shown in Figure 3-5.

The fixture and the test articles were mounted on the vacuum-chamber lid, and all power and instrumentation leads were passed through the vacuum feedthrough of this lid. Both Earth-ambient and thermal-vacuum tests were run using this arrangement to reduce the connection time.

The motor mount, diagonal strut, and spacecraft hinge halves were all part of the SCM-1 test fixturing. Material used was aluminum, and construction was completed using welding and bolting. The auxiliary panel release fixture for SCM-2 was made from two electrically actuated solenoids. No attempt was made to duplicate prototype release for this test. Flat-wire-mesh radiant-strip heaters, rated at 1,200 watts, were installed to supply heat if required to the auxiliary panel dampers and to the motor. Temperature sensors were mounted at strategic locations to supply a complete thermal history.

Figure 3-4: SCM-1 TEST ARTICLE FABRICATION

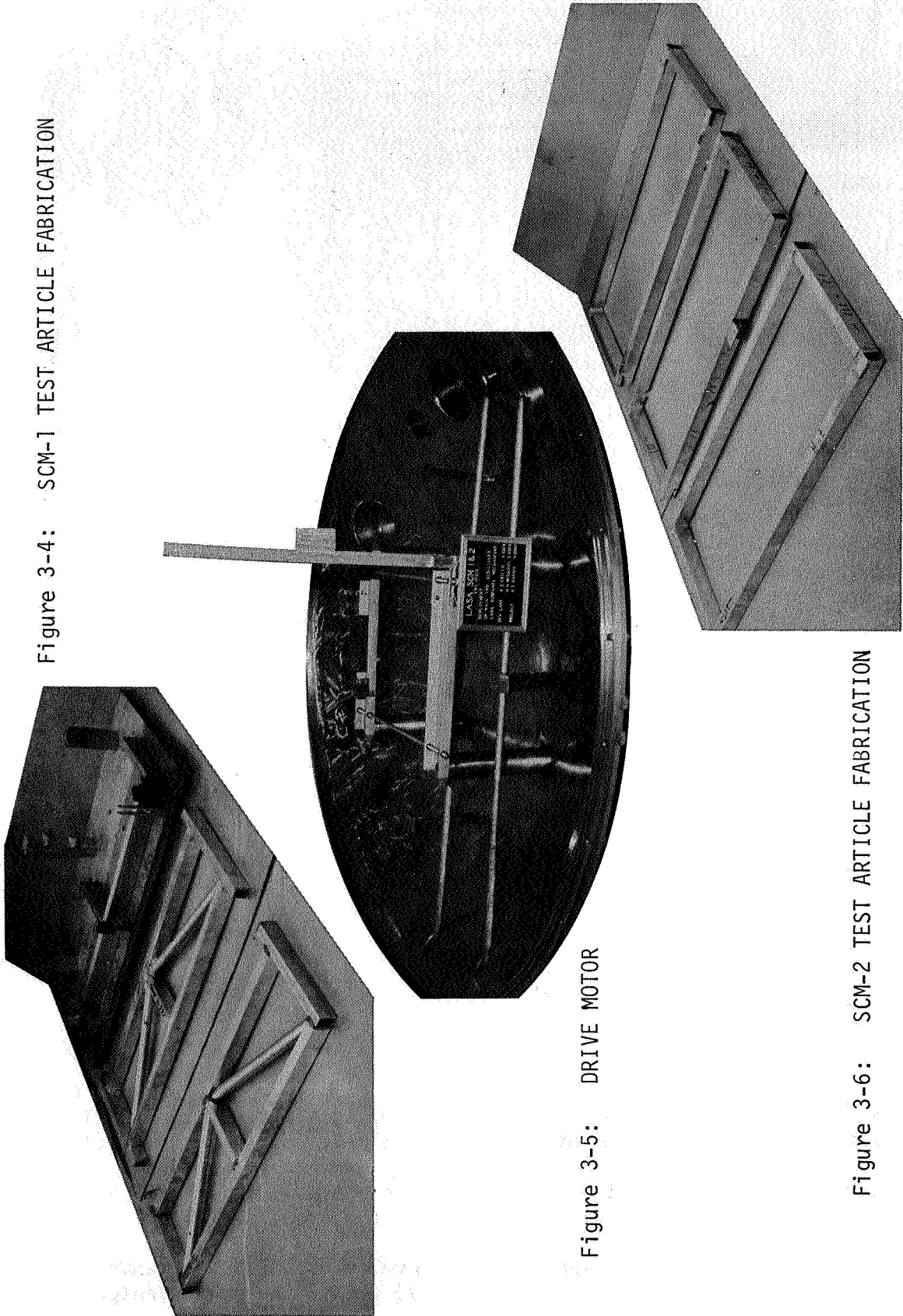


Figure 3-5: DRIVE MOTOR

Figure 3-6: SCM-2 TEST ARTICLE FABRICATION

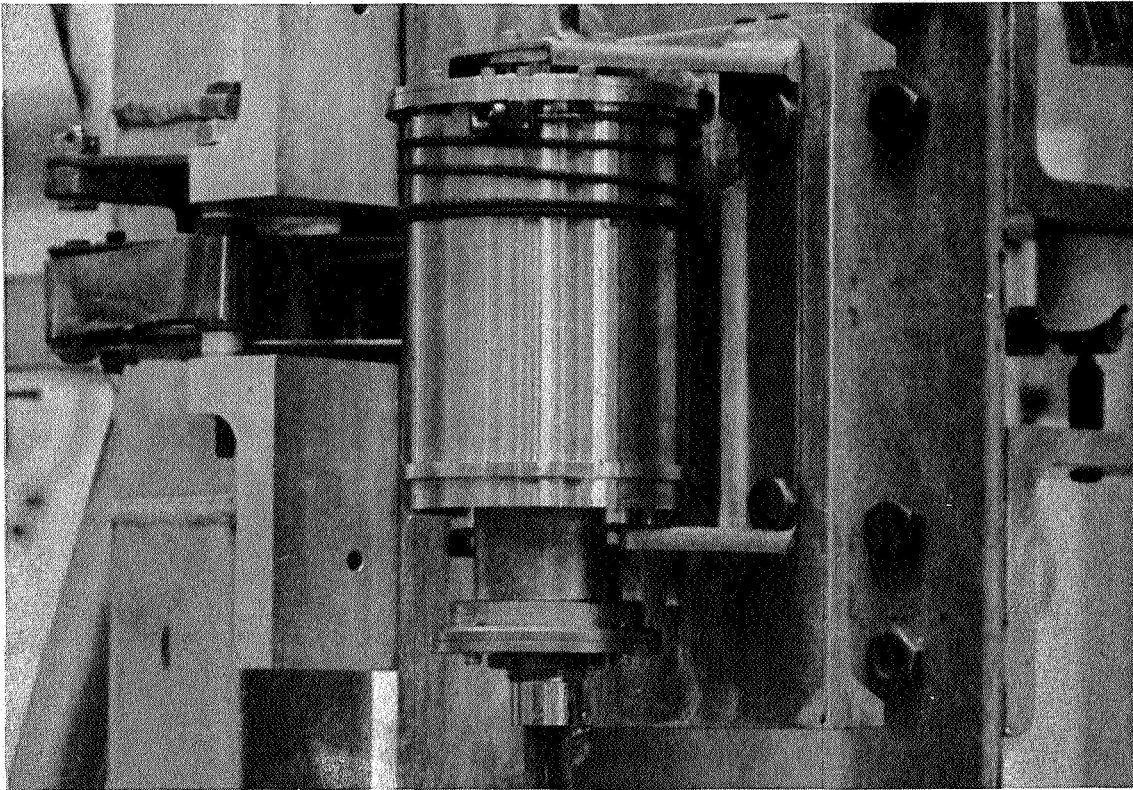


Figure 3-7: SCM-1 AND SCM-2 DRIVE MOTOR

Gravity forces are the most predominant hazard to the deployment tests. Panels for the SCM tests were mounted vertical with the hinge centerlines plumb to reduce gravity effects to a minimum. The use of short panels eliminates the need for auxiliary supports for the panel weight, although additional fixturing was required to prevent small deflections from distorting test results.

Indicators were supplied to visually gage panel opening position against time while viewed through small vacuum-chamber portholes.

3.1.3 OUTSIDE PROCUREMENT AND PRODUCTION

It was cost effective to purchase dampers and cable drums from outside sources.

The motor/cable drum drive assembly was purchased from the Harmonic Drive Division of the United Shoe Machinery Company to a design prepared in response to a Boeing procurement specification developed in Phase I of the program. The assembly is unique in that the drive motor and three stages of speed reduction are all housed within the cable drum itself. Figure 3-8 shows an outline drawing and cutaway of the array deployment drive.

The viscous rotary damper assembly is a development of Sesco Manufacturing Company and is essentially an off-the-shelf item. The basic design was

modified, however, to supply 190 degrees of rotational travel and to provide a long shaft. Figure 3-9 shows an outline drawing and cutaway section of the damper.

The dampers are filled with 1,000-centistoke silicone fluid and are velocity- and temperature-sensitive. The damper is rated at 6.25 inch-pound/radian/second at 160°F throughout the deployment travel.

Torsion springs used throughout the test program were specially made to Boeing detail design drawings by the Renton Coil Spring Company. All torsion springs were made of Inconel wire. Neg'ator springs used as a cable guard on all quadrants were made by Hunter Spring Company to modified requirements of their standard stock springs.

3.1.4 QUALITY ASSURANCE

Selected activities provided adequate control of Task 2 fabrication operations.

Only those controls required to ensure the quality of selected parts were implemented in Task 2. This included receiving inspection and test of selected procured items. In one instance some dry-lubed parts were rejected because of improper processing by the supplier which eventually caused scrapping of five small parts. Shipping damage to the tiedown cable anchor assembly also was disclosed during receiving inspection.

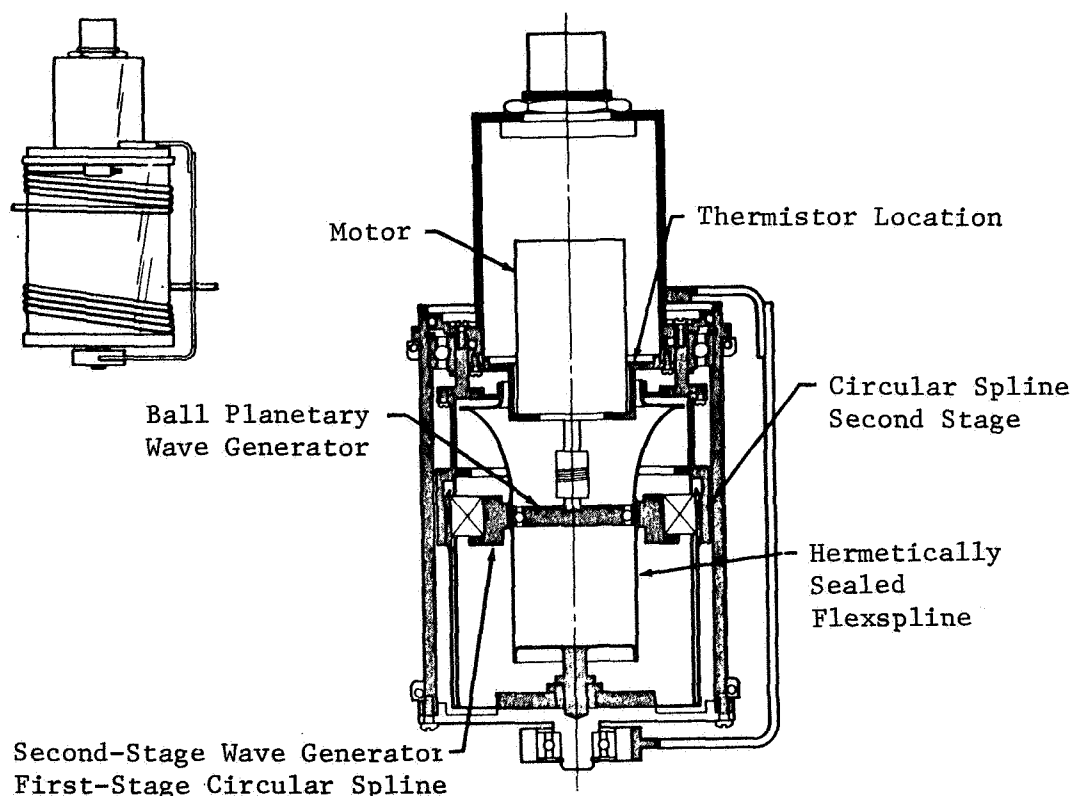


Figure 3-8: ARRAY DEPLOYMENT DRIVE

The following parts and operations were inspected to verify drawing requirements: Geneva release cam, hinge brackets, mechanical test fittings, and installation of the boost tiedown mechanism and set up of SCM-30 test fixture assembly details.

3.1.5 CONDUCT OF SCM-1 AND SCM-2 TESTS

Simulated deployment tests under laboratory conditions demonstrate both ambient and thermal-vacuum performance.

The SCM-1 main panel deployment drive consists of a motor/cable drum winching in a cable using quadrant segments to develop a hinge moment. Deployment dynamics are directly affected by the effective spring rate of the cable. The first deployment of a prototype panel involves a panel stack with a mass moment of inertia of 803 slug-feet squared and a very short cable acting as a spring. As deployment moves outboard, the mass moment of inertia reduces to 93 slug-feet squared for Subpanels 5, 5A, and 5B, and the longer cable will have an effective spring rate of minimum value. Spring rate measurements were deferred to the MTA-3 test to take advantage of a full-length cable.

The SCM-1 test article was installed on the vertical post of the test fixture as shown in Figure 3-3. A shake test was conducted on the test articles in the deployed position to verify that the overcenter latches would remain latched.

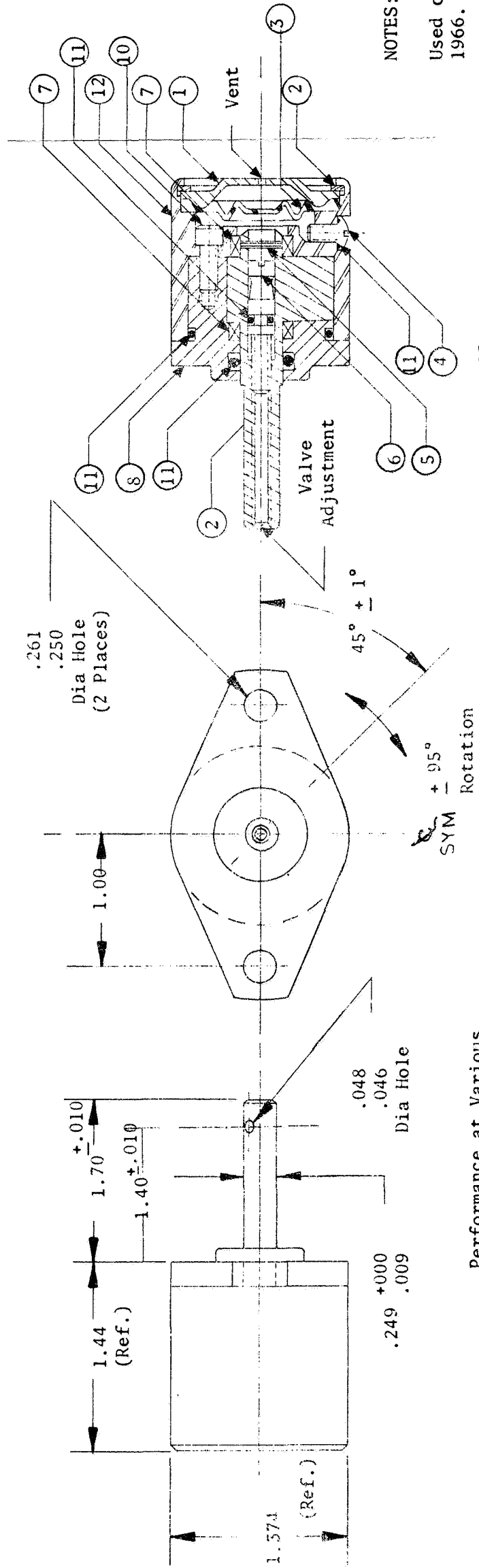
Hinge friction loads were then measured for each 45 degrees of deployment angle of Subpanels 1, 2, and 3. Subpanels 2 and 3 were then deflected 0.10 inch at their midpoint as measured between the upper and lower outboard spars. Hinge friction loads were again measured.

The deployment cable was installed in the test article and total system friction on the cable was measured with 60-pound force applied to the motor end. Cable slip force measurements also were taken as a function of contact angle for one cable quadrant.

The cable was then attached to the motor and the test article was manually stowed. The motor was activated and complete deployment was obtained. The deployment angle of each subpanel was measured as a function of time. The test article was then restowed for vacuum-deployment tests.

The SCM-2 auxiliary subpanel deployment drive consists of a dual-torsion spring system and uses a viscous rotary damper to control the deployment speed. Figure 3-10 shows the deployment elements before deployment and Figure 3-11 shows the deployment elements after the panels are opened and locked.

Three different tests were conducted on the SCM-2 test article as part of the SCM-2 test package. After the test article was assembled and installed on the test fixture, a two-part engineering shake test was conducted. Then static friction measurements were made on each auxiliary subpanel at



NOTES:

Used on OV3-2 launched October 29, 1966.

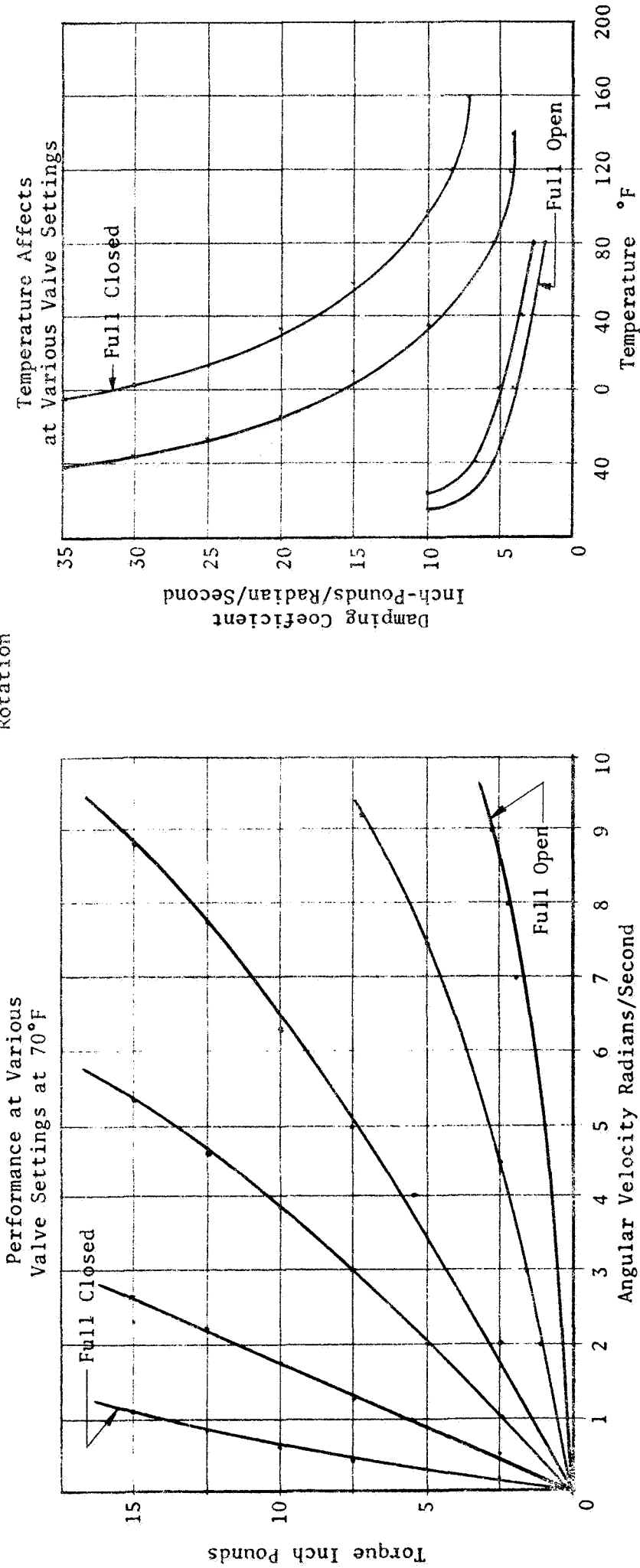
Ratings: Operating Temperature Range -65° to 165°F. Damping Range 0.25 to 6.25 in-lb/Rad/Sec

Minimum Shaft Rotation: 190°

Backlash of Shaft: +1/2° Maximum.

Acceptance Test: Following a dimensional check, each unit shall be functionally tested as follows:

1. Rotation - The shaft of each unit shall be rotated to determine the angular travel characteristics. Travel less than 180° and/or backlash in excess of 1/2° shall be cause for rejection.
2. Damping Rate - Damping rate of the unit shall be checked with the metering valve in the full open and full closed position. The damping rates shall be as follows:
Fully open; .25 in-lb/Rad/Sec. Max.
Fully closed; 6.25 in-lb/Rad/Sec. Min. at 75°F.
3. Following the above tests, there shall be no evidence of external leakage.



ITEM	
1	Cover
2	Retaining Ring
3	Diaphragm
4	Seal Screw
5	Roll Pin
6	Valve
7	Bearings
8	Abutment
9	Vane-Shaft
10	Cap-Screw
11	O-Rings
12	Case
	Damping Fluid

Figure 3-9: ROTARY DAMPER

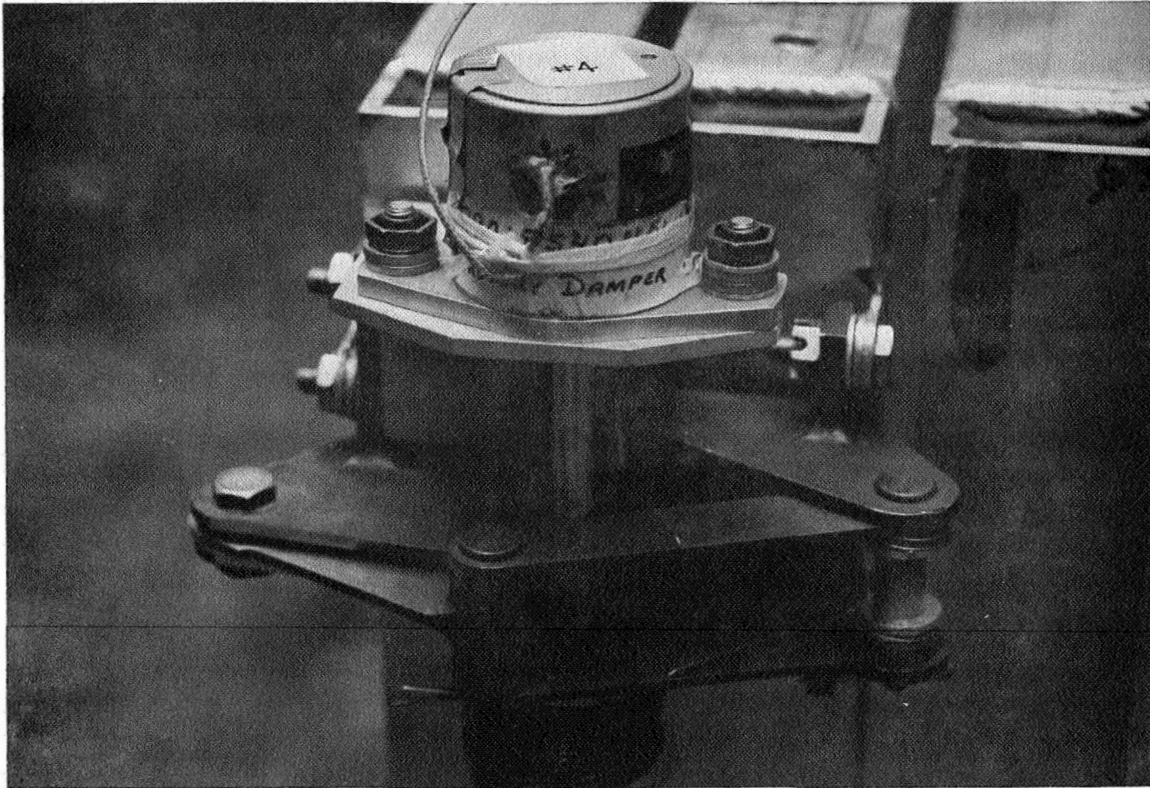


Figure 3-10: UNDEPLOYED HINGE AND FOUR-BAR LINKAGE LATCH

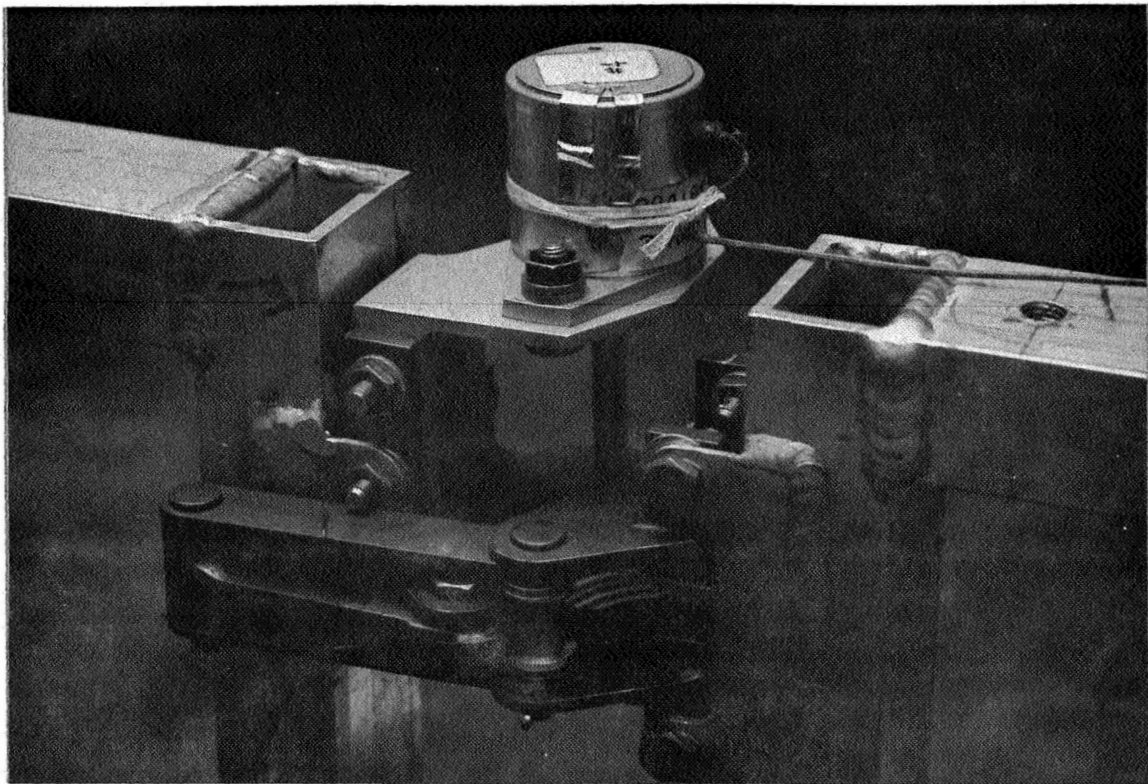


Figure 3-11: DEPLOYED AND LATCHED AUXILIARY HINGE

various angular positions. These friction measurements were made for both undeflected and deflected panels. Finally, the test article was stowed and allowed to deploy under its own power. The test setup was then prepared for thermal-vacuum testing.

The combined test of SCM-1 and SCM-2 was arranged to use a common vacuum chamber, test fixture, and time period.

The SCM-1 main subpanels and SCM-2 auxiliary subpanels test articles were mounted on the test fixture as shown in Figure 3-3. A temperature control system was installed for thermal-vacuum tests. Flat resistance heater blankets were located above each rotary damper for the SCM-2 test and around the deployment motor drum for the SCM-1 test. Heater power was supplied on demand by variable a.c. power supplies. The purpose of the heaters was to maintain the critical test parts at operating temperature while radiating to -320°F liquid-nitrogen cold walls. Thermocouples were installed and strip heaters were connected to the power supplies.

The entire test setup was bolted to the base lid of the vacuum chamber and a pretest deployment of both test articles was accomplished in the vacuum chamber at Earth-ambient temperature and pressure. Deployment times were measured as a function of angle for both test articles.

The lid was lowered and the test articles were manually restowed. The test articles were again inserted in the vacuum chamber, the lid was bolted to the chamber, and vacuum pumping was initiated. The chamber walls were cooled to a temperature of -320°F . The pumping time-history is shown in Figure 3-12.

The SCM-2 deployment test was conducted first. The dampers were heated only during the last 45 degrees of deployment in order to speed deployment time. The SCM-1 deployment test was then conducted. The motor heater was not used. Deployment times were measured as a function of angle for both test articles.

A posttest deployment in air duplicating the pretest run was conducted to determine degradation of the systems due to the thermal-vacuum environment.

3.1.6 RESULTS AND ANALYSIS OF TESTS

Friction, cable slippage, resistance to vibration, and operation were successfully tested and evaluated during the SCM-1 and SCM-2 series tests.

- 1) Shake Tests, SCM-1---The first tests to be conducted were the shake tests wherein the test hardware was mounted in the test fixture, the latch drive springs were removed, the panels were manually latched and the assemblies were individually subjected to a manual engineering shake test. Next the springs were reinstalled and the shake test was repeated. In both test sequences, the latches remained locked.

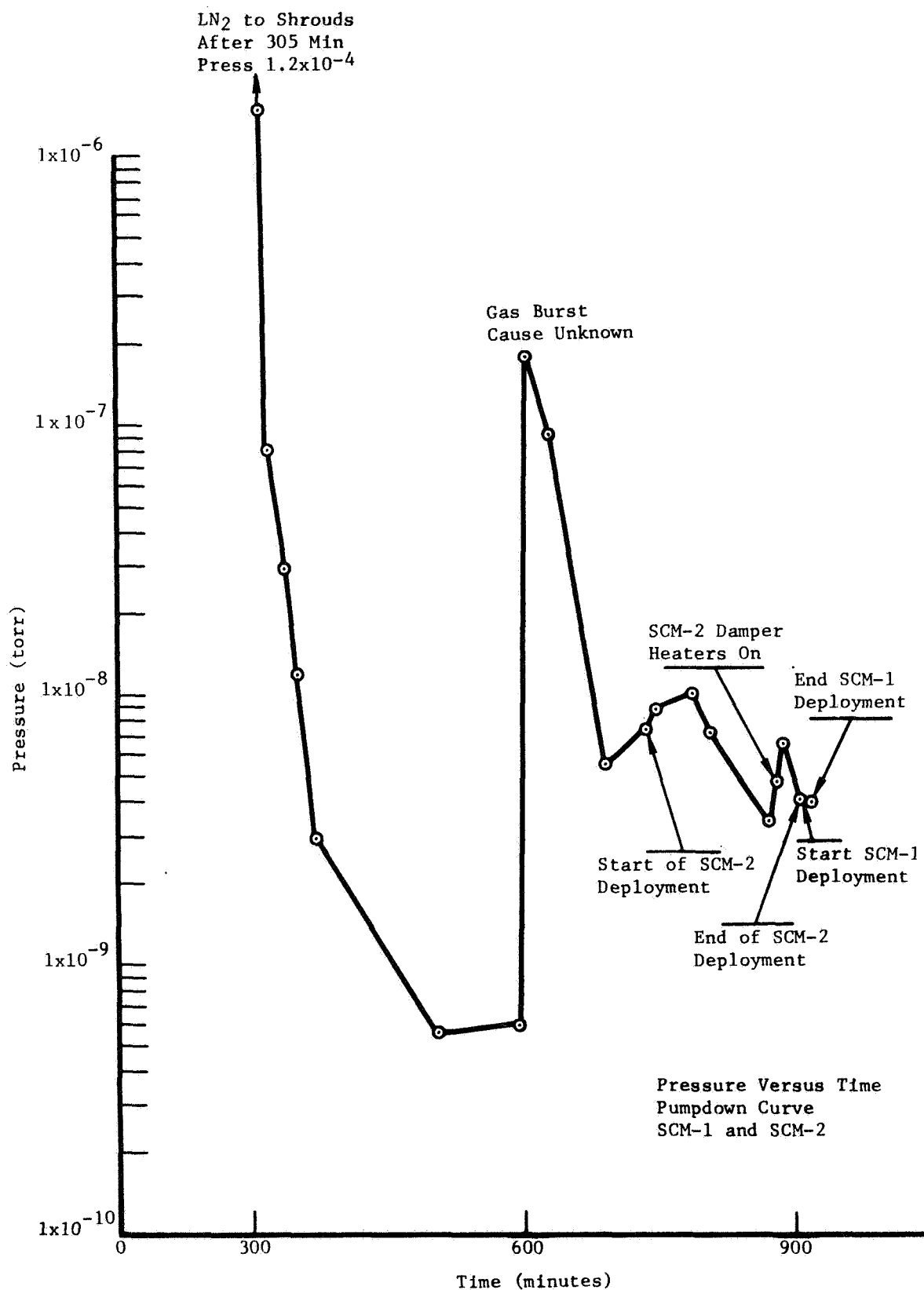


Figure 3-12: VACUUM TIME-HISTORY IN VACUUM

- 2) Cable Slippage Test, SCM-1---The cable slippage test setup is shown in Figure 3-13. The cable used was a 78-inch length of 1/16-diameter Monel 7 by 7 strand. Spring scales, as shown, were attached to each end. The outboard end was tied down while the inboard or spacecraft end was pulled. When the outboard spring scale indicated a change from zero tension, the tension in the other scale was recorded. Any indication of tension in the outboard scale meant slippage in the cable. The test data plotted in Figure 3-14 indicates the "pull" values at which slippage between the cable and quadrant occurred.
- 3) Total System Friction Test, SCM-1---The test setup used for measuring total cable-to-quadrant system friction is shown in Figure 3-15. The same cable was used for this test as for the cable slippage test. Spring scales were again attached to both ends. The outboard end was anchored as shown while the other end was pulled toward the motor with a scale force of 60 pounds. With this "motor" force the scale at Subpanel 3 read 22 pounds, indicating a cable-to-quadrant friction equivalent to 38 pounds of "pull" for two quadrants. This test was to determine the coefficient of friction between the quadrant surface and Monel cable. The contact angle of the cable was about 360 degrees.
- 4) Hinge Friction Test, SCM-1---A spring scale was attached to Subpanel 3 at the center of the outboard edge. Static hinge frictions were then measured for each 45 degrees of deployment angle. The test was then repeated for Subpanels 1 and 2.

Subpanel 3 was deflected 0.10 inch at its midpoint as shown in Figure 3-16. Static hinge frictions were again measured. The test was then repeated on Subpanel 2 and results are plotted in Figure 3-17.

Test results for Subpanel 3 are not consistent with the results for Subpanel 2. Two deflected-friction readings for Subpanel 3 were less than their corresponding undeflected frictions. In one case the undeflected friction was almost three times the deflected friction. The lack of precise alignment was the primary cause of hinge friction anomalies.

Closer tolerances must be established to ensure precise alignment. A repeat of the deflected panel test on one MTA-3 joint was accomplished to get a better grasp of the alignment problem. Results are reported in Section 5.0.

- 5) Ambient Deployment Test, SCM-1---It was evident during testing that the actual SCM-1 deployment time-history closely resembled the plot in Figure 3-18. In other words, hinge frictions for all swings were significant enough to reduce both slack and stretched cable regimes so that they were barely noticeable to the observers. A continuous time-history plot of an SCM-1 deployment, had instrumentation been installed, would have shown panel angular position closely tracking motor drum position with perturbations of the kind shown in Figure 3-18.

Although the system performed predicably, deployment testing did reveal some component problems.

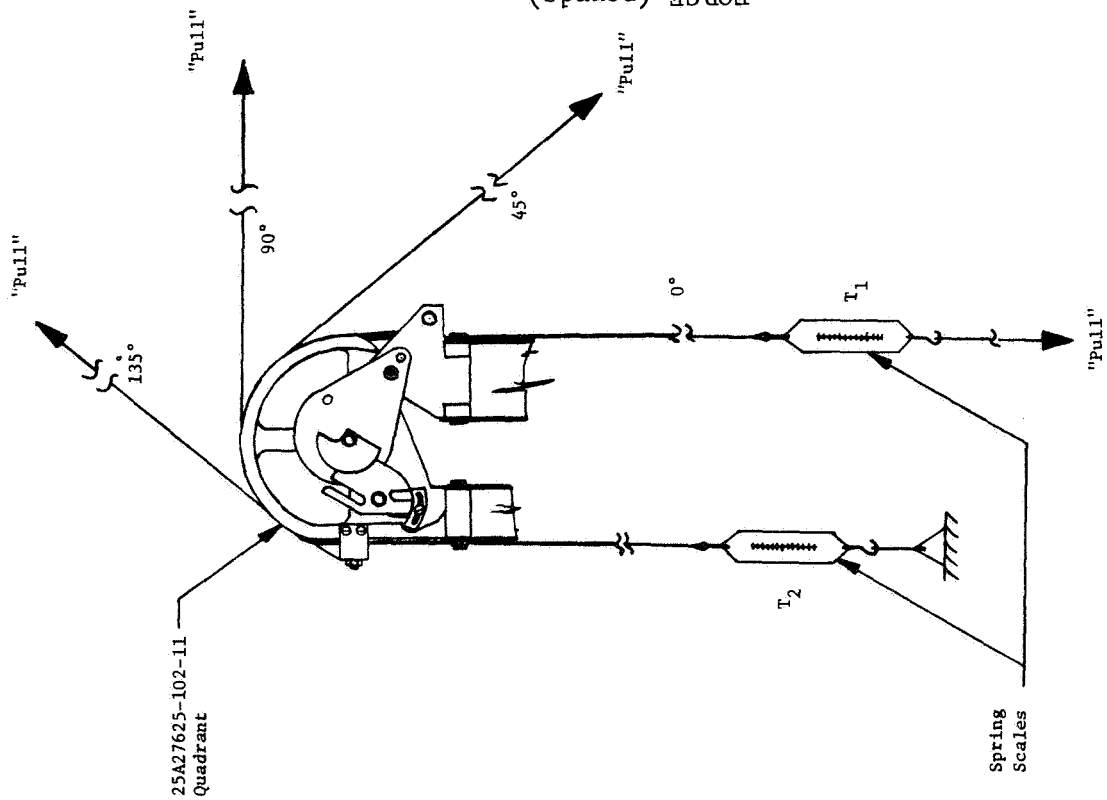


Figure 3-13: CABLE SLIPPAGE TEST

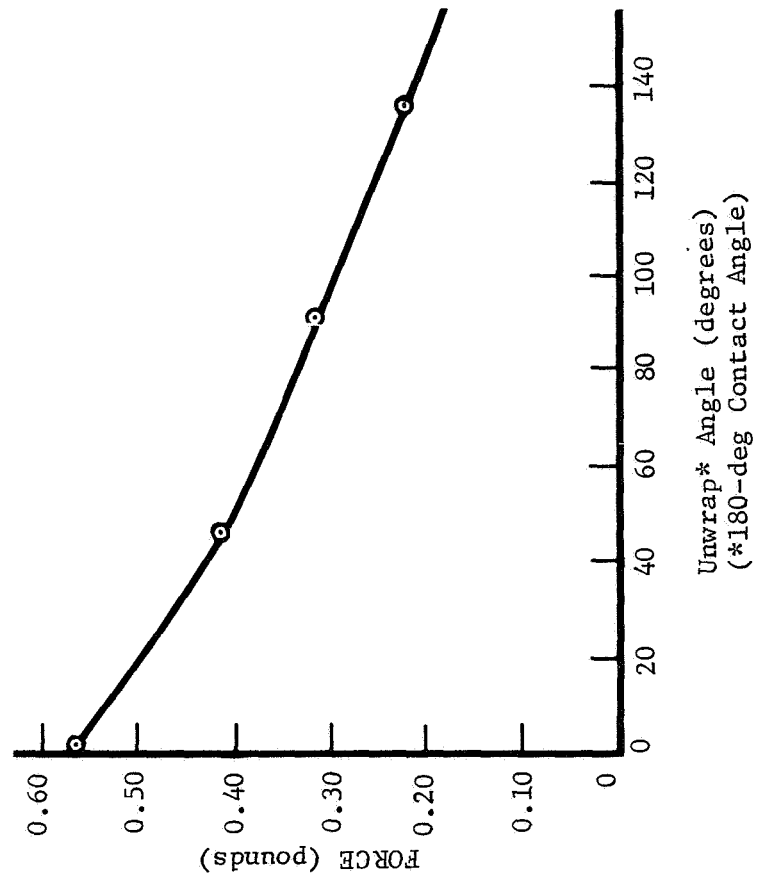


Figure 3-14: SCM-1 CABLE SLIPPAGE MEASUREMENTS

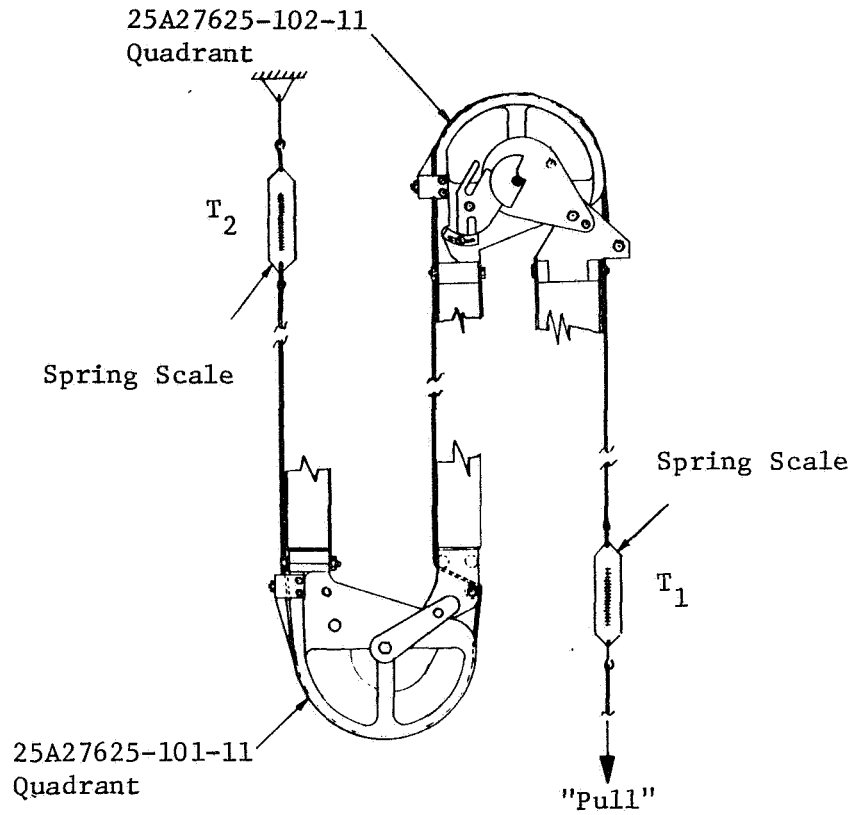


Figure 3-15: TOTAL SYSTEM FRICTION TEST

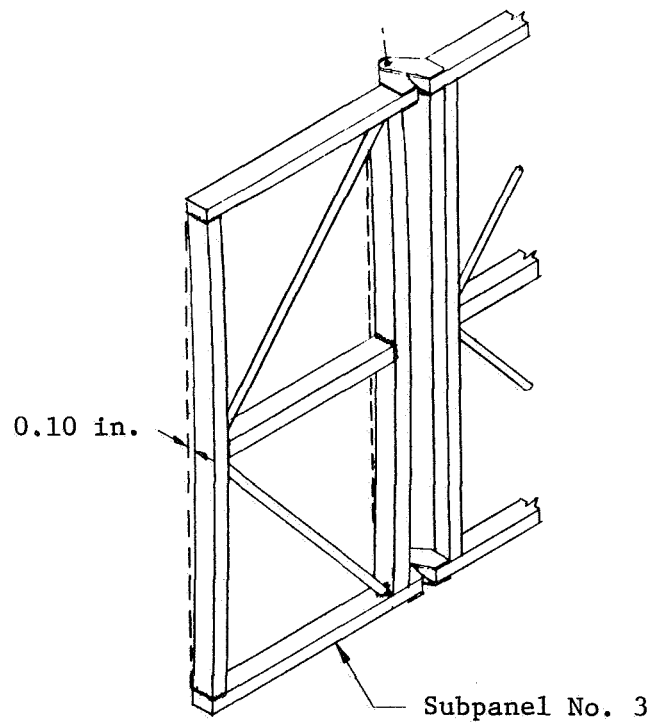


Figure 3-16: MIDPOINT DEFLECTION ON SUBPANEL NO. 3

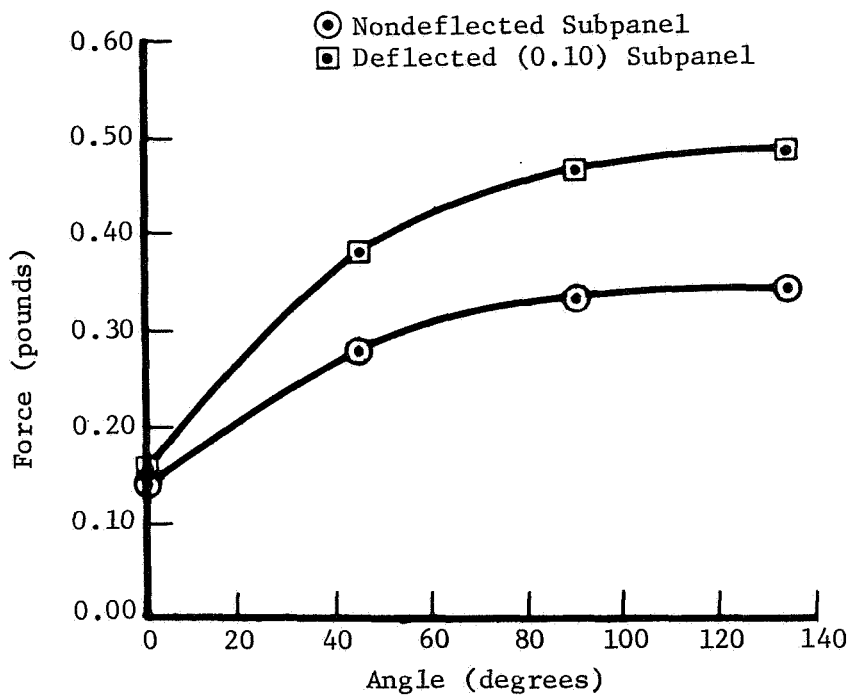


Figure 3-17: SCM-1 TEST RESULTS---HINGE FRICTION ON SUBPANEL NO. 2

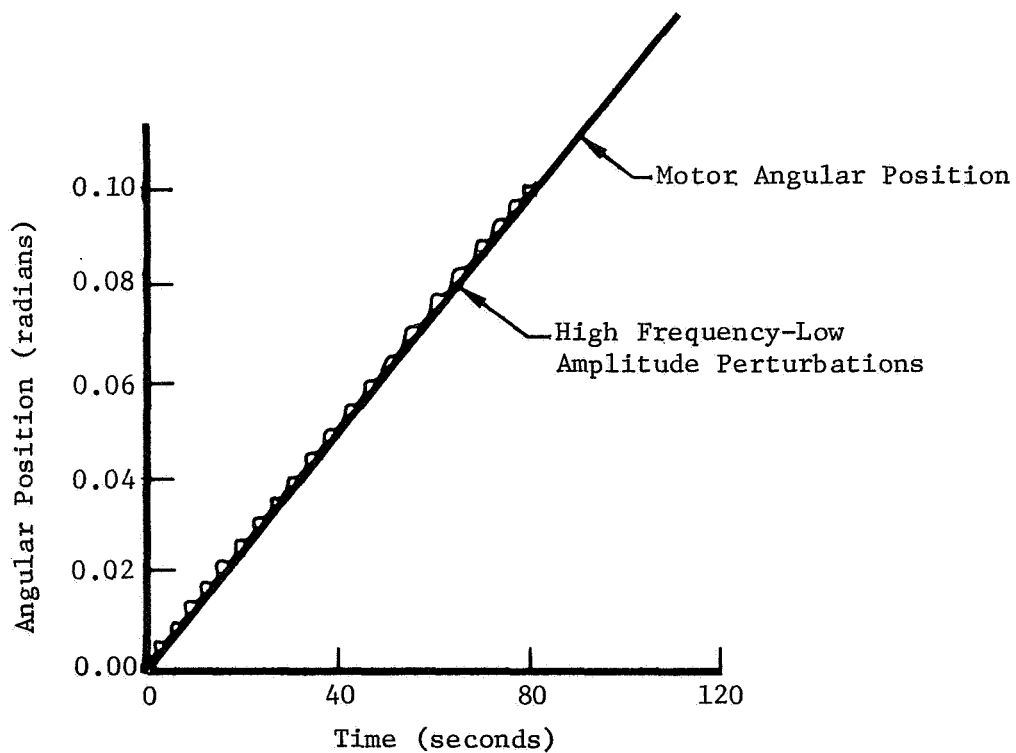


Figure 3-18: FRICTION TIME HISTORY---SCM-1

Hinge locating tolerances resulted in misalignment between the three pins in the latch assembly. Friction, binding, and gravity affected the latch movement from its stowed (and drilled) position to the latched position as exaggerated in Figure 3-19. The total resistance was easily overcome by the deployment motor system, but the light-duty springs were inadequate. As a result, the springs were paired on SCM-1 to double the spring latching force and the problem was overcome. New springs were procured for MTA-3 to alleviate this problem. Although improved tolerance would negate the need for stronger springs, the change was made to improve overall performance.

- 6) Hinge Friction Tests, SCM-2---Following the shake tests, the drive and latch springs and damper shear pins were removed. A spring scale was attached to simulated Subpanel 2A at the center of the outboard edge and the subpanel was then pulled through a 180-degree arc to the fully deployed position. Static friction was recorded at 45-degree intervals. The scale was then attached to Subpanel 2B and the test was repeated.

A panel clamp was attached to Subpanel 2A inducing a 0.10-inch deflection at the center. A sketch of the deflected panel is shown in Figure 3-20. The friction test was repeated. Subpanel 2B was then clamped and the friction test was again repeated.

Figure 3-21 is the plotted data for the friction tests. The magnitude and range of friction values represented by the plot are satisfactory.

The zero-degree deflected breakaway friction for Subpanel 2A was larger than other readings. This was caused by a fitting hole being elliptical after hardcoating. The minor diameter was 0.002 inch undersize. Readings in other pin positions were not affected by this tolerance problem. Deflected and undeflected readings were almost equal at the 90- and 135-degree positions of Subpanel 2A; but for Subpanel 2B, all deflected frictions were lower than undeflected frictions. The consistency of this decrease in friction for Subpanel 2B suggests that the hinge pins were misaligned and that the induced deflection actually improved the alignment.

- 7) Ambient Deployment Tests---After the hinge friction tests were completed, the drive springs, latch springs, and damper shear pins were installed. The auxiliary subpanels were stowed. The solenoid power leads to the 100-volt a.c. power supply were connected and deployment was initiated. After the subpanels had fully deployed, they were restowed for thermal-vacuum testing.

It was found that overhead reinforcing structure was required to prevent gravity forces from deflecting the center panel as the auxiliary subpanels deployed. During the test, all auxiliary springs and dampers functioned satisfactorily.

The friction forces introduced as a result of panel weight were not a major item and did not compromise the test results significantly.

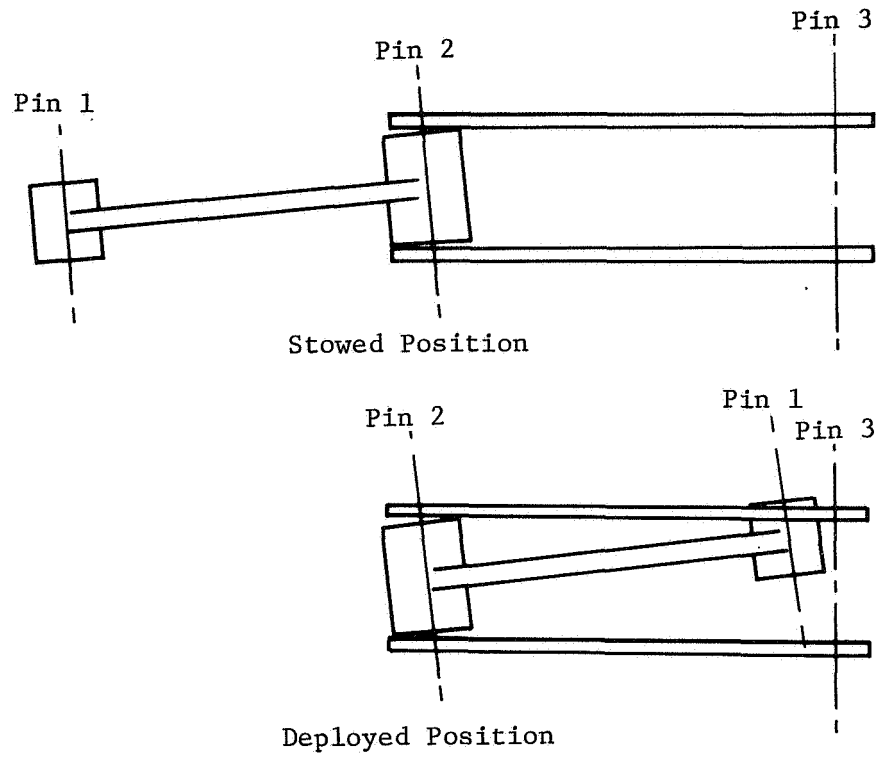


Figure 3-19: OVERCENTER LATCH MISALIGNMENT

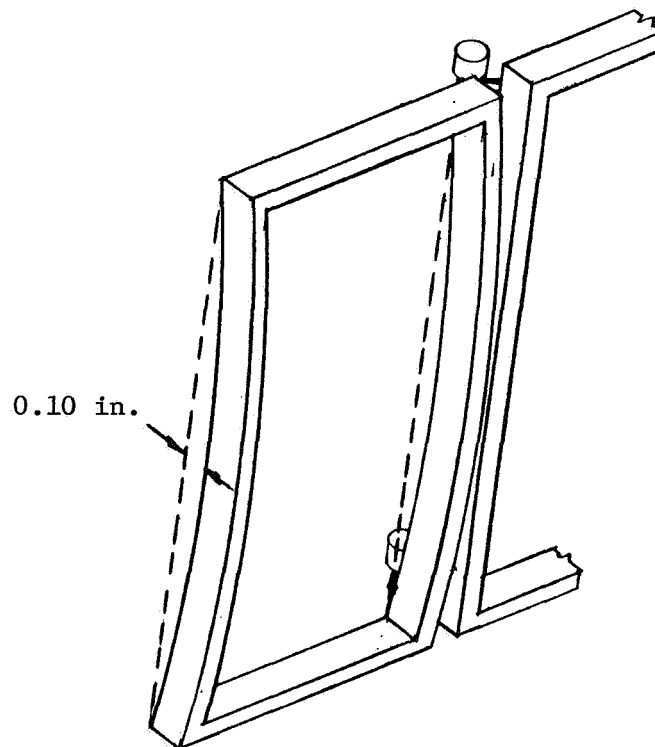


Figure 3-20: DEFLECTED AUXILIARY SUBPANEL---SCM-2

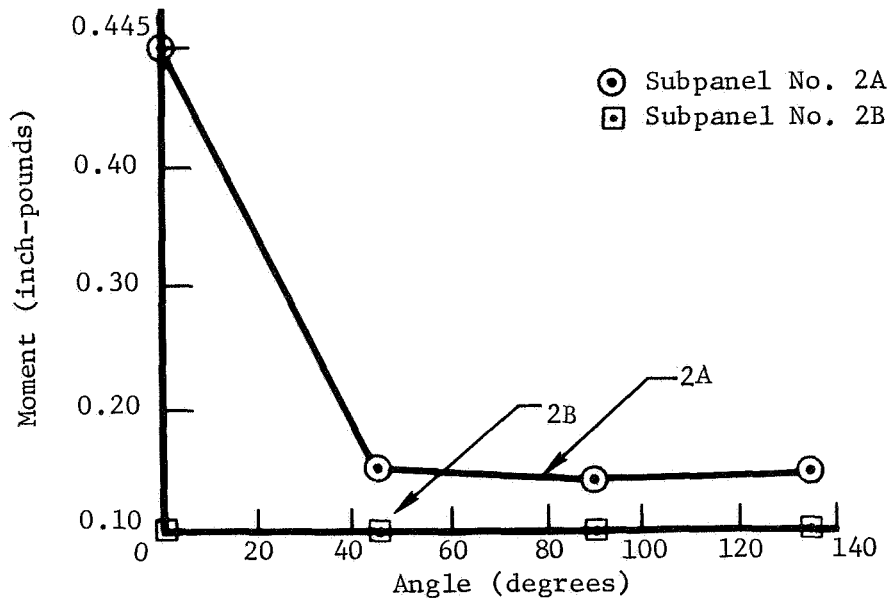


Figure 3-21: COMPOSITE OF SCM-2 HINGE FRICTIONS

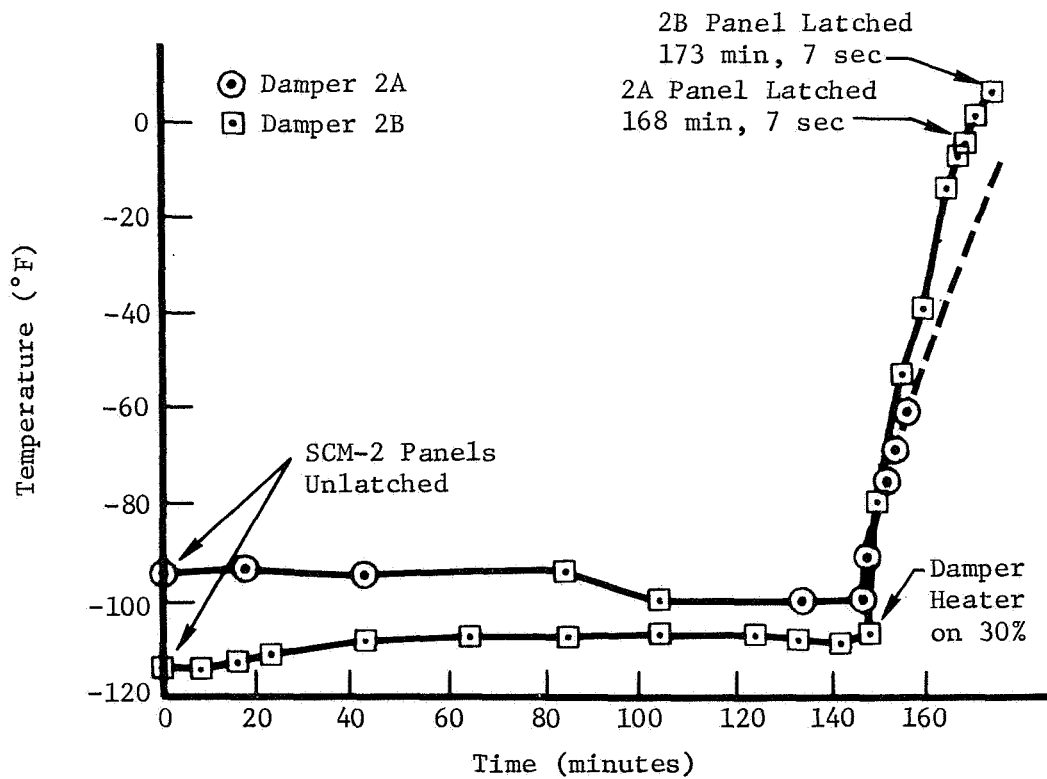


Figure 3-22: DAMPER TEMPERATURE

Without friction, the velocity time-history would have been determined by spring torque divided by the damper constant. In the actual deployment with friction, the effective torque near closing was appreciably reduced, with a consequent considerable increase in the time to deploy.

- 8) SCM-2 Thermal-Vacuum Deployment---During vacuum deployment of the SCM-2 test article, the dampers were at -95 and -110°F. Deployment time was very slow due to thickening of the damper oil. Test results indicated that all mechanisms, including the dampers, operated satisfactorily.

Figure 3-22 shows plots of temperature versus time for dampers on Subpanels 2A and 2B. All damper heaters were activated after about 145 minutes of deployment to speed deployment. The subpanels would very probably have completed their deployment without use of the heaters. In any case, the heaters did speed the last few degrees of deployment and the subpanels successfully latched. To ensure sufficient heat for the dampers used in space, it is recommended that a special thermal control coating be applied to the dampers so that heat may be absorbed from the Sun, or that an insulation blanket be added to reduce heat loss.

Deployment times were recorded for each 45 degrees of deployment angle using the deployment-position indicators located above each hingeline. Comparative time-histories are plotted in Figure 3-23. The damper oil was thicker during the thermal-vacuum deployment because of the low temperatures. As a result, vacuum deployments were much slower than ambient deployments. In a flight test, dampers are exposed to the Sun over 50% of their cylindrical body and will warm naturally after the main panels are deployed.

- 9) SCM-1 Thermal-Vacuum Deployment---The SCM-1 vacuum deployment was successful. One pair of overcenter latches remained at top, dead center, as a result of residual tension (which is a redundant lock mechanism) in the deployment cable. Removing cable tension allows the springs to drive the latch links over center. A change in latch link geometry has resolved the problem on the array design. Test results and observations indicate that no new design and/or operational problems arose as a result of the thermal-vacuum environment.

Time-history data for both ambient and thermal-vacuum SCM-1 deployments are recorded in Figure 3-24. A comparison of these two runs shows that the thermal-vacuum environment had very little effect on the deployment time-history.

Figure 3-25 is a plot of motor temperature versus time recorded during the thermal-vacuum deployment. Although strip heaters were added to the motor drum, they were not used either before or during the deployment. The motor maintained a steady deployment rate during the test, which corresponded closely to ambient rates (see Figure 3-24).

3.1.7 CONCLUSIONS

The deployment mechanisms are suitable for deploying and locking the solar array in a vacuum environment.

The results of the vacuum tests were satisfactory. The motor deploying the main panels operated at -63°F and at a cable rate above 2 inches/minute. Both dampers operated at temperatures of -100°F or lower. Figure 3-12 provides a history of the environment during tests. There was no evidence of cold welding. Pressures of 6×10^{-10} torr were obtained before test, and all testing was performed at pressures below 1×10^{-8} torr. The auxiliary subpanels deployed at temperatures below -100°F . The rate of deployment for the auxiliary subpanels was initially 45 degrees in 9 minutes for Panel 2A and 20 minutes for Panel 2B. The final deployment rate before the heaters were turned on was approximately 3 degrees in 20 minutes for Panel 2A and 15 degrees in 20 minutes for Panel 2B. The last 35 degrees of deployment after the heaters were turned on the panels took 18 to 23 minutes. All test objectives were met satisfactorily.

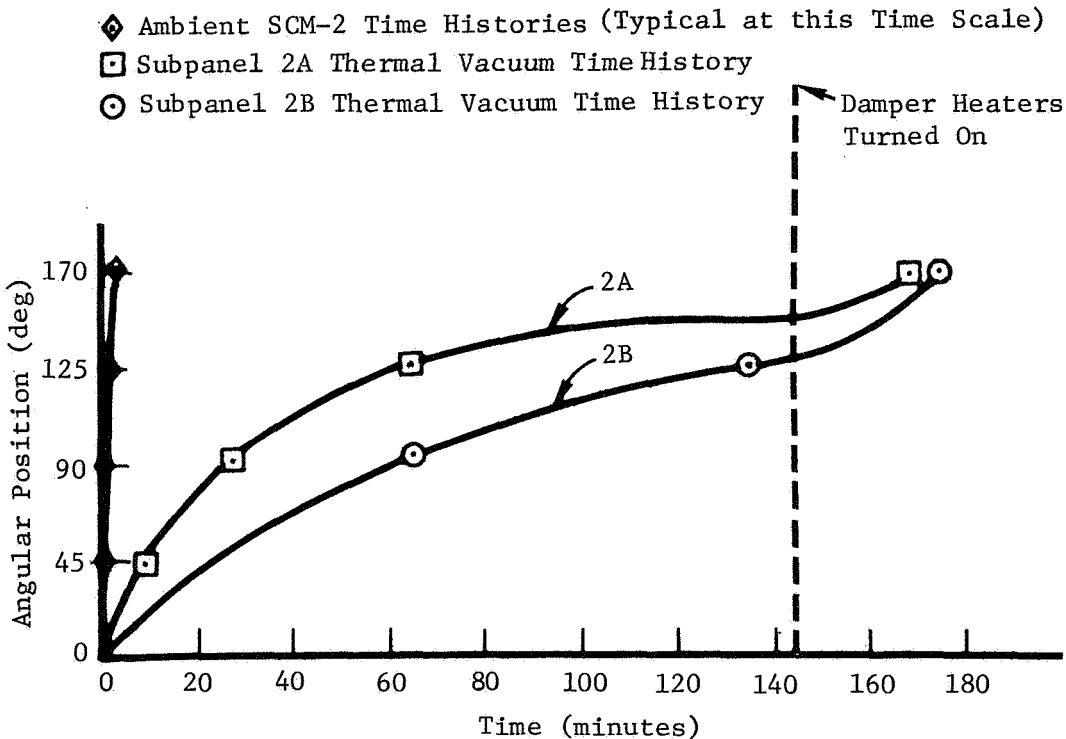


Figure 3-23: VACUUM TEST TIME-HISTORY---SCM-2

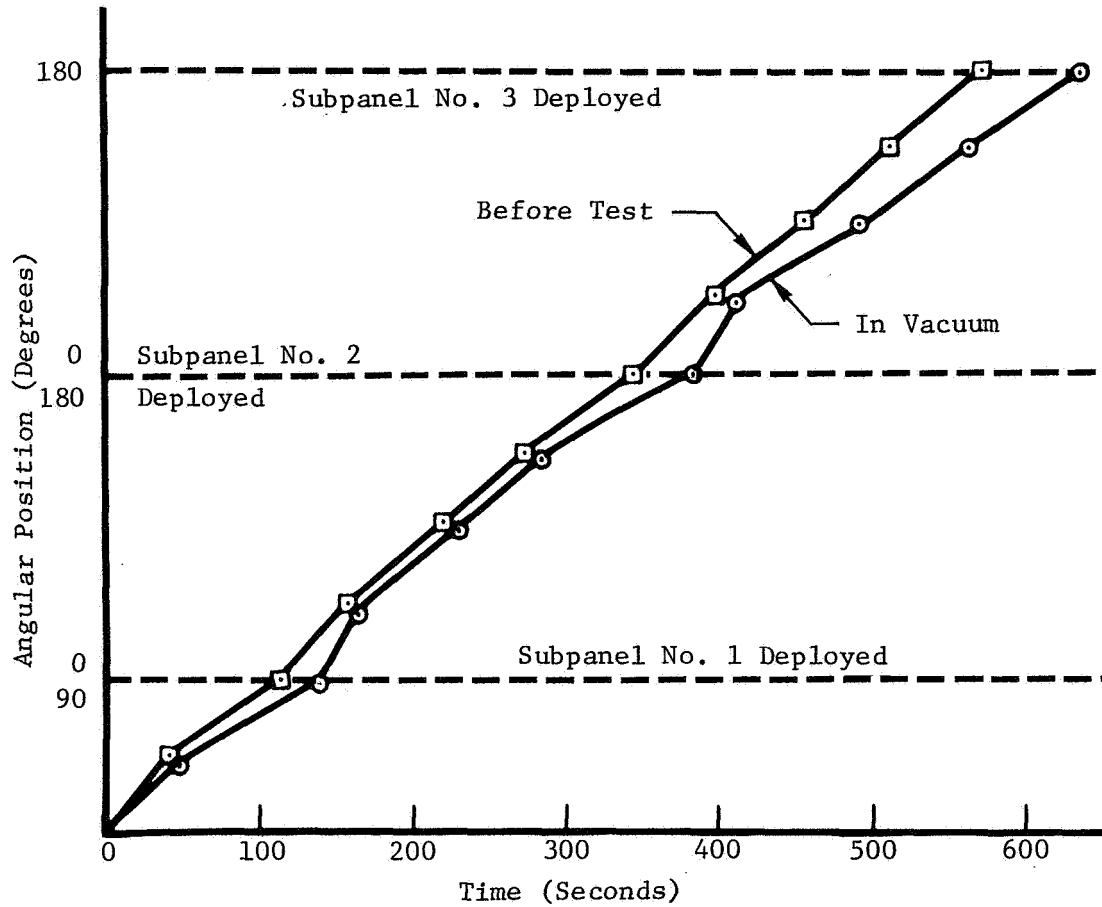


Figure 3-24: DEPLOYMENT TIME-HISTORY---SCM-1

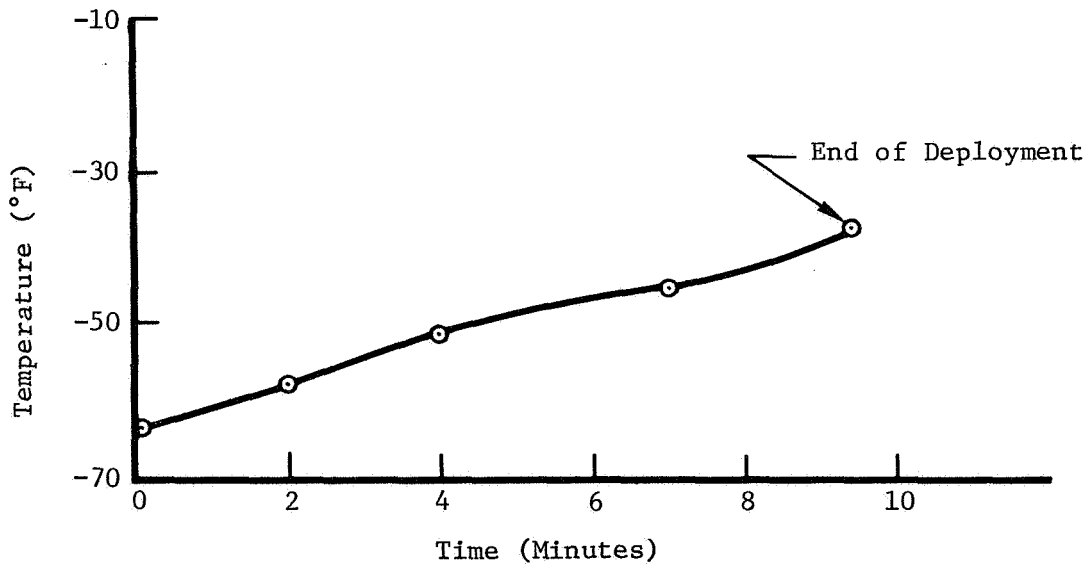


Figure 3-25: MOTOR TEMPERATURE TIME-HISTORY---SCM-1

Three salient facts are apparent:

- 1) The main and auxiliary subpanel deployment mechanisms operated satisfactorily after exposure of more than 4 hours in the thermal-vacuum environment below -100°F and 1×10^{-8} torr.
- 2) There was no evidence of cold welding or thermal-vacuum degradation in the test.
- 3) The auxiliary panels deployed even though the damping ratio was substantially increased due to the cold environment.

Problems with individual hardware items were discovered during the tests and are discussed below.

Hinge Alignment---The main and auxiliary panel hinges were both sensitive to hinge alignment. The austere tooling used was inadequate in alignment of hinges, and both tests required rework of the hinge brackets to obtain satisfactory alignment before testing. Better alignment methods, tooling, and additional instructions will be required for future parts.

In the initial studies, self-aligning bearings were discarded in favor of Journal bearings because of weight and the inability to obtain consistent frictions in aluminum bearings lined with glass cloth and plastic. However, there is a current investigation into a Monel race with Inconel ball using dry-film lubrication and constructed with a 0.001 to 0.002 diametrical clearance for possible inclusion into the array design to alleviate alignment problems.

Overcenter Latching Interference---The overcenter latch links interfered with the large bracket attach bolt heads. Three solutions were available for this problem: countersink the bolt head, change the link, or move the links to the outside of the bracket. Moving the links on the outside of the bracket was selected as the most economical and structurally stable method, and the latches were changed before testing.

Overcenter Latching Springs---The springs used to hold the main subpanel latching links on or beyond dead center were not strong enough to overcome the relatively high friction forces. This gave the appearance, in testing, of the links not latching because they did not move beyond dead center. Future latching mechanisms will be equipped with stronger springs to provide more positive motion in latching.

3.2 DEVELOPMENT OF BOOST TIEDOWN PANEL RELEASE

The boost tiedown cable release design was based on the concept of releasing a spring under high-tension load and allowing it to propel itself in a path in space. Its velocity is dependent on the spring energy at release. The satisfactory performance of this design at a single station was verified by test. The use of organic cable materials and clamping devices that must release instantly and completely was also tested.

SCM 30-2 and SCM 30-3 testing programs tested the boost tiedown release mechanisms. Selected assemblies of the mechanical release system were installed on a simulated solar panel mockup using prototype tiedown anchor and release fittings. The system was rigged to specification loads and released using an ordnance cable cutter to simulate the mechanical release train. Center-spar cable clamp holding and release capability was tested using the Nomex cable intended for flight and flight prototype mechanisms. Figure 3-26 shows the test items and fixture.

3.2.1 DESCRIPTION OF SCM-30 TEST ARTICLES

The test articles used as many release elements as possible and tested to the most rigorous requirements.

The boost tiedown loads are highest at Station 10.5 for both the circumferential and center-spar tiedowns resulting in Station 10.5 hardware being selected for test. The circumferential elements (SCM 30-2) consist of a cable assembly with special end fittings that mate with an anchor hook at each end. A turnbuckle is supplied to provide the rigging adjustment. The anchor assemblies are held closed by a taut cable simulating the prototype release train. An ordnance-actuated cable cutter (designed for use on the Station 123.7 center spar) was used to release each end of the tiedown cable so that proper release delay control could be applied, by use of a computer, and the extremes of cable behavior could be tested. Figure 3-27 shows the SCM 30-2 test elements.

SCM 30-3 was designed to evaluate the performance of the center-spar release clamp to grip the Nomex cable until after boost burnout and to release the cable once the circumferential cables have ejected. Physical properties of the Nomex cable (i.e., breaking strength, yield under extended load, and spring rate) in the torsionally stable braid were not available. The test was designed to measure these properties.

The clamping mechanisms tested was Station 10.5 prototype design and typical of the center-spar boost tiedown at five stations. The function of the assembly is to maintain a clamping grip on the through stack Nomex organic cables and, or release of the circumferential boost tiedown cable, release the center-spar cable, and allow the spring of the cable to begin retracting through the fiberglass substrate. In addition to the mechanism, the cable was tested for spring rate, breaking strength, and relaxation under load. Figure 3-28 shows the SCM 30-3 test elements.

3.2.2 DESIGN OF THE SCM 30-2 TEST FIXTURES

The fixtures were simple and did not compromise test results.

The entire SCM 30-2 test fixture was built to Boeing design and is shown in Figures 3-29 and 3-30. The fixture was made of welded and bolted aluminum structural shapes. The fixture is approximately 2 by 10 feet

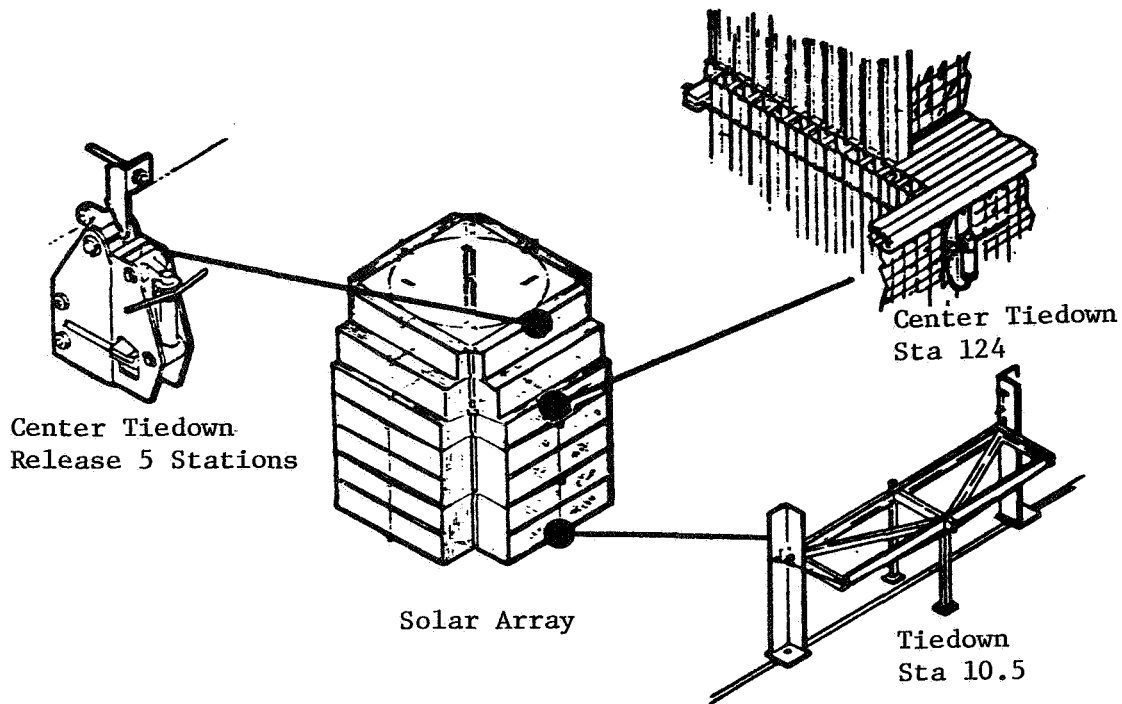


Figure 3-26: SCM-30 TIEDOWN AND RELEASE

and is mounted above floor level. The size and shape simulates the cross section of the stowed array at Station 10.5. There was no attempt to simulate the stacked beryllium panel spring rate because results could not be verified by any logical means, and actual release tests of a beryllium stack will be made in Phase III.

The test cable and anchor fittings used prototype details and standards. The prototype center-spar cable clamp and release fitting was mounted and release demonstrated. Center-spar cable tests are covered as test SCM 30-3. A skid board with grid tapes on the surface provides the back-drop for high-speed photographs used to record the cable ejection characteristics.

Test equipment used for this program includes the following:

- 1) 25A27758-101-11 Test Fixture Assembly;
- 2) Grid Marked Skid Board (10 by 12 ft);
- 3) SDS-910 Computer (Program Control);
- 4) Visicorder and Ordnance Control Console (Figure 3-31);
- 5) High-Speed Camera (Milliken) (Figure 3-32).



Figure 3-27: SCM 30-2 TEST ELEMENTS

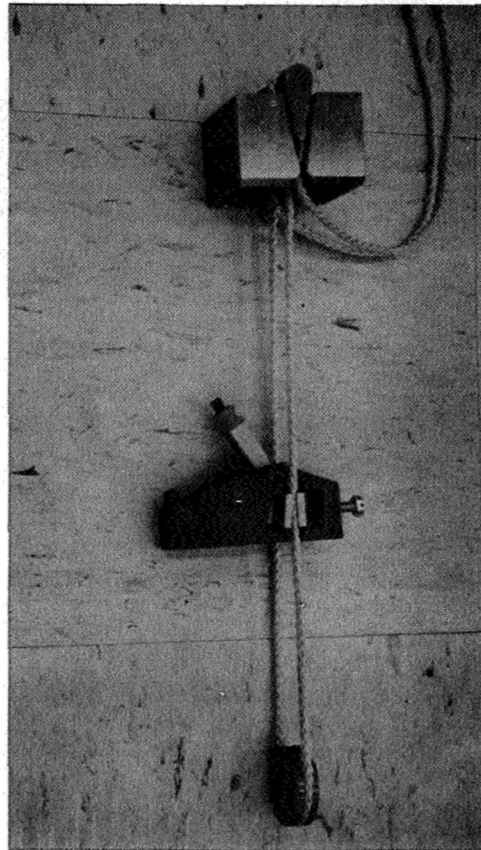


Figure 3-28: SCM 30-3 TEST ELEMENTS

3.2.3 CONDUCT OF PANEL TIEDOWN TEST (SCM 30-2)

The test employed hardware and rigging loads that duplicate the highest load cable installation.

In preparing the SCM 30-2 test, the circumferential cable was inserted in the anchor fitting and the release train threaded through the cable cutters and pretensioned to limits established in the design. The cable was located around the actuator arm of the center-spar clamp and release fitting, and rigged to drawing preload requirements.

When all hookups were made, the ordnance was inserted in the cable cutters and the area cleared except for the test operator. The time delay for cutter firing was adjusted to 0.1 second, the high-speed cameras started, and the test switch closed. The test required less than 10 seconds for each run. The resultant cable motion was filmed by a high-speed camera mounted directly above the test setup. The ordnance devices were monitored (squib voltage) by a visicorder, the firing voltage delays were recorded, and cable cutter action was monitored by photographic flashbulbs (retest only).

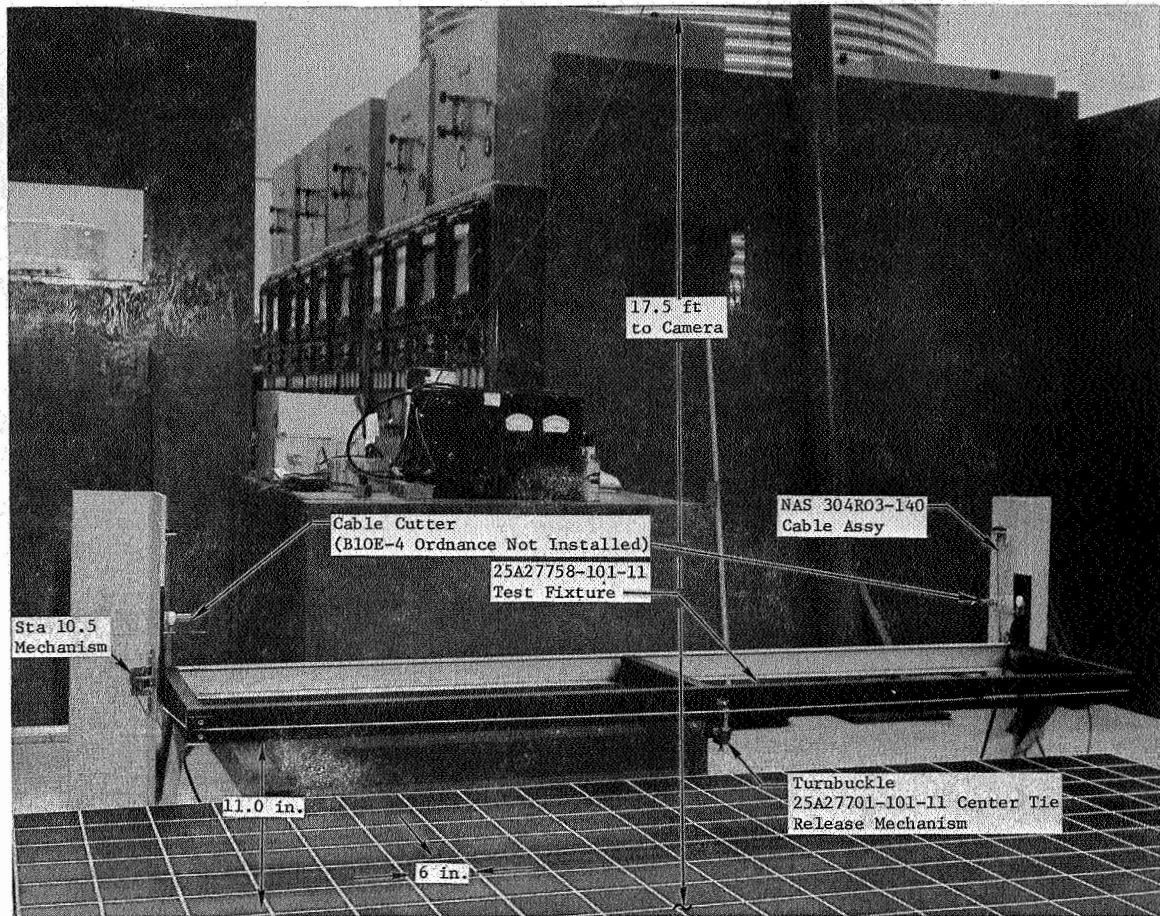


Figure 3-29: SCM 30-2 RETEST TEST SETUP

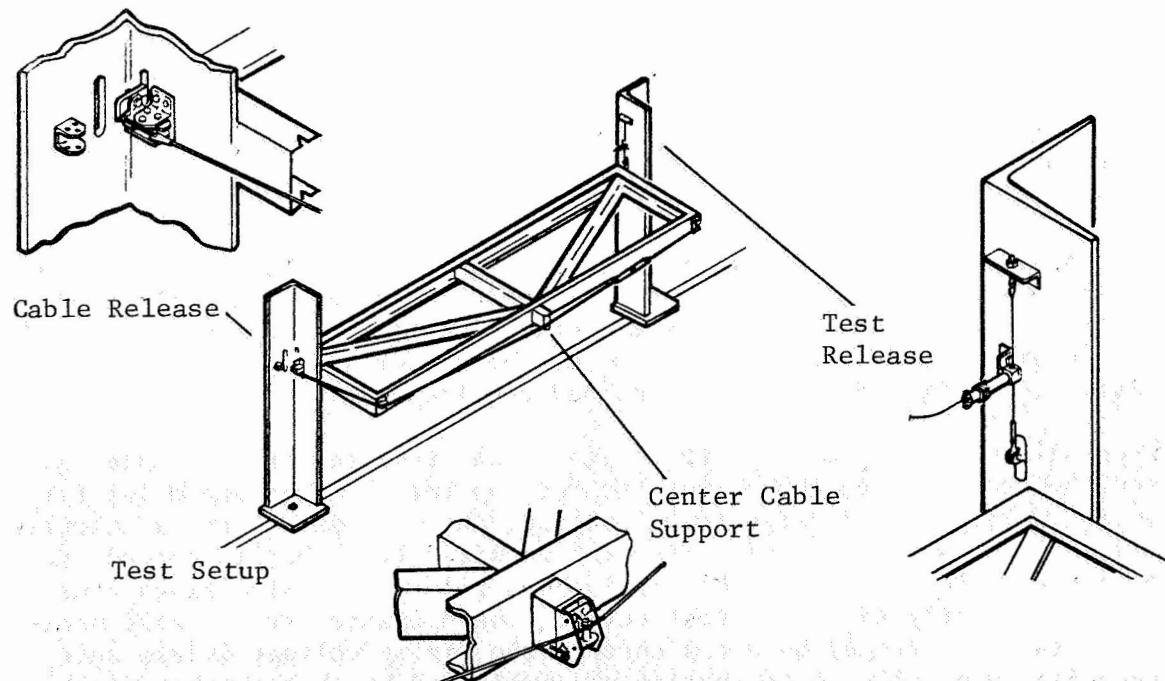


Figure 3-30: SCM 30-2 TIEDOWN RELEASE TEST DETAILS

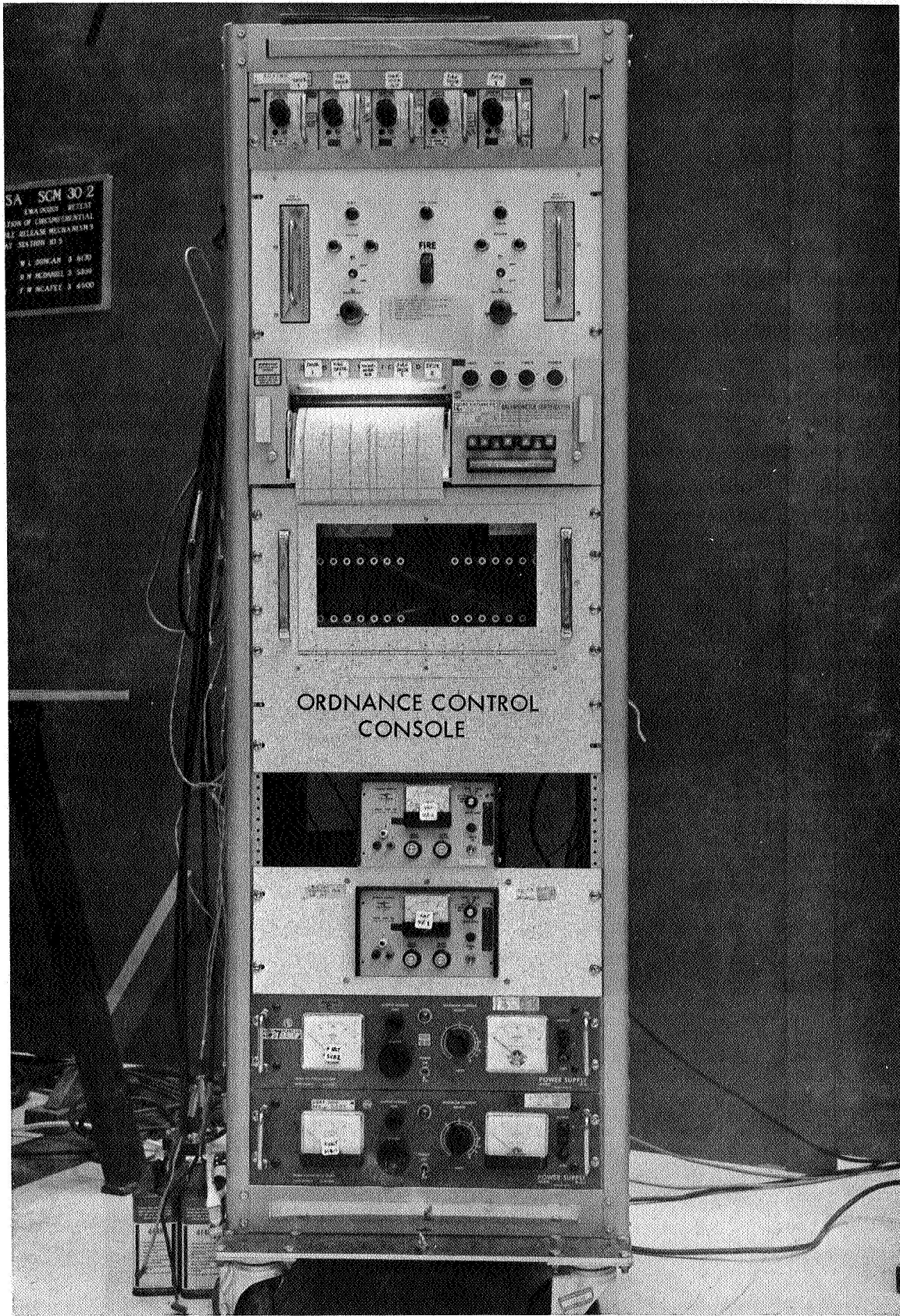


Figure 3-31: SCM 30-2 RETEST VISICORDER SETUP



Figure 3-32: SCM 30-2 RETEST CAMERA SETUP

Cable anchors depend exclusively on the spring rate of the cable and panel stack for release. Although the spring rate of the cable is many times more than the panel, both must be considered.

Two test series were run as a result of electrical anomalies in the first three tests.

The principal difference between the first series of tests and the retest was the firing circuitry, the camera speed, and the flashbulb cable cutter monitor.

The test block diagram is shown in Figure 3-33. The ordnance firing circuit was designed to initiate firing of the two ordnance devices (B-10B4) with a nominal 100-millisecond time differential.

The timing differential was accomplished by the SDS-910 computer, which controlled and fired a separate circuit to each cable cutter squib. The circuits were functionally checked using fuses that electrically simulated the actual test circuits. No discrepancies were noted during the pretest circuitry checkouts.

3.2.4 RESULTS AND ANALYSIS OF TESTS SCM 30-2

The boost tiedown release mechanism met all test objectives and provided valuable engineering design information.

A. Test Run 1

The first test run was intended to incorporate a 0.1-second delay between the release of the two cable end fittings. The visicorder record showed that simultaneous firing signals were imposed on both squibs with no apparent time delay in the cable and fitting release. The high-speed motion pictures showed that the cable ejected from the test fixture in an outward direction approximately 20 degrees to the right of center, with little tendency to rotate about either end of the cable. The frame rate of the motion picture was not sufficiently high to allow exact evaluation of the cable position; however, the direction of travel was established.

B. Test Run 2

The second test run was made with no firing circuit delay time programmed to better evaluate the test results from the first run. The results of the second run were essentially the same as those of the first run. It was, therefore, concluded that the first run was made with no time delay.

C. Test Run 3

The third run was made with a programmed 0.1-second time delay, but some question exists as to whether a delay did occur. The photographic

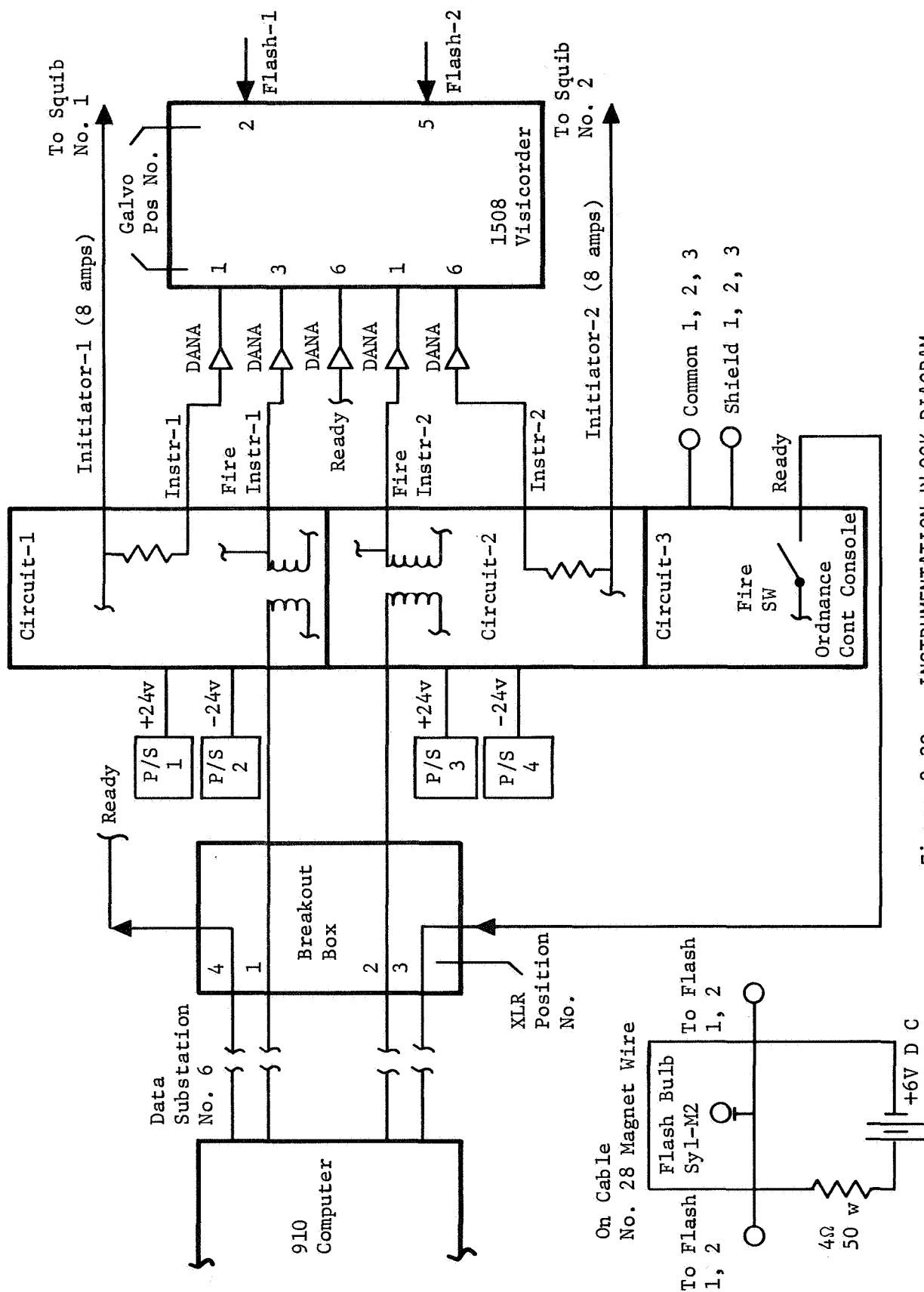


Figure 3-33: INSTRUMENTATION BLOCK DIAGRAM

data indicated that a delay did occur because the cable ejected in a sideward direction (about 45 degrees to the right of center) instead of as observed in Runs 1 and 2. The visicorder record did show a voltage spike $t = 0$ for the delay squib. After Test Run 3, modifications were made to the ordnance circuitry and delay tests were repeated and identified as Rerun Tests 1 and 2.

D. Retest Run 1

Retest Run 1 was programmed to provide an 0.1-second time delay for release of the cable end opposite to the turnbuckles. The exact delay was achieved as evidenced by both the visicorder record and the high-speed photographs. Figure 3-34 is a typical photo of cable position approximately 0.04 second after initial release.

The interval between each high-speed photograph taken is approximately 0.0032 second. Approximate time after firing is noted on the picture. Figure 3-35 is an overlay of photographs taken during Run 1, showing cable positions at 0.013-second intervals.

E. Test Evaluation

Both Retest Runs 1 and 2 had marked evidence of a time delay. The cable direction followed the analysis prediction closely although the turnbuckle speed was somewhat slower than predicted: 400 inches per second as compared to an estimated 1,000 inches per second. The release of the cable ends in Retest Runs 1 and 2 was clean with no evidence of cable hangup on the delayed end, indicating that cables will clear a spacecraft with four panel assemblies where no more than 0.1-second delay is experienced. Improved release under even more extreme conditions could probably be assumed if the cable anchor fittings were spring-loaded open.

F. Evaluation of Photographic Data

Color motion pictures of the retest runs were made at 310 frames per second against a dull-black background marked with white 6-inch grid. The cable ends were color coded as was the adjusting turnbuckle. Behavior of the cable was observed to be essentially as predicted. Operation of the delayed action cable cutters was easily detected by observing the photo flashbulbs. The 12th frame, after firing of the high-speed film, was selected for this report with the turnbuckle accented with a circle on the left. Figure 3-36 illustrates cable and turnbuckle travel in 0.09 second. Figure 3-37 is an overlay of the photographs at an interval of 0.013 second for Run 2.

3.2.5 EXPLOSIVE ACTUATED CABLE CUTTERS

Phase II experience proved that the originally proposed supplier's configuration control of the cable cutters and ordnance is inadequate and should not be used on MTA-4.

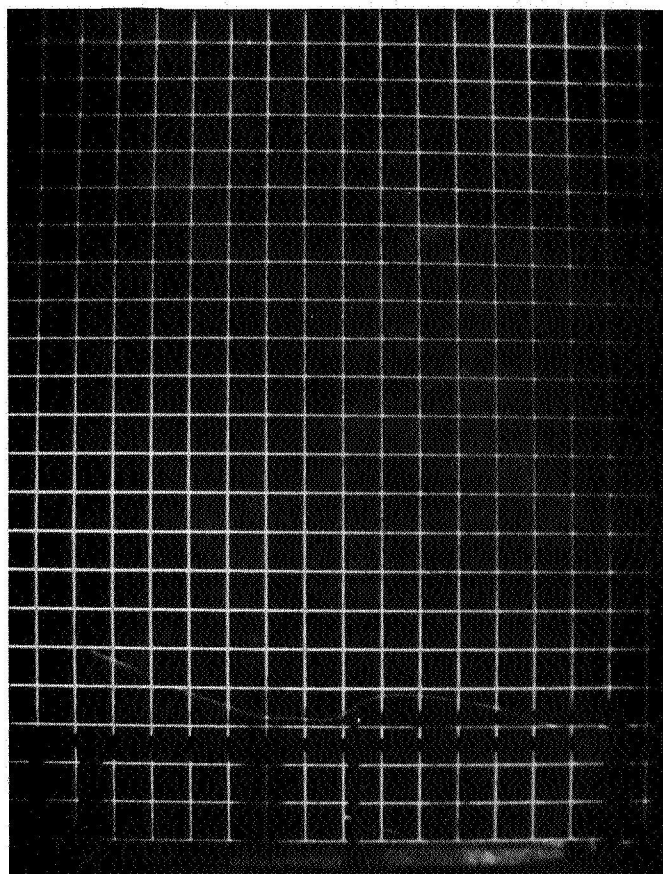


Figure 3-34: CABLE POSITION 0.04 SECOND AFTER RELEASE---RUN NO. 1

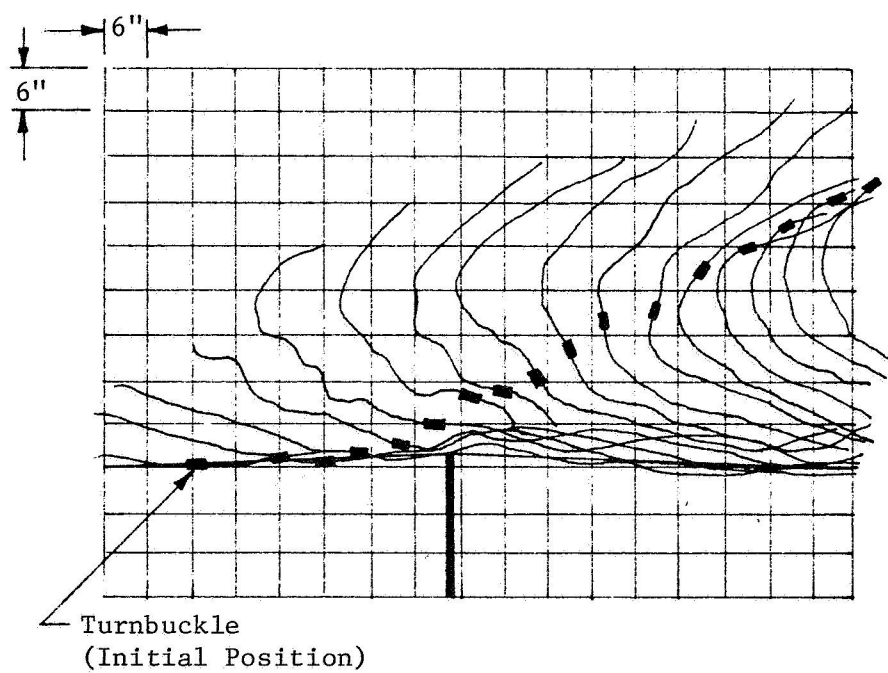


Figure 3-35: SEQUENTIAL CABLE POSITIONS ---RUN NO. 1

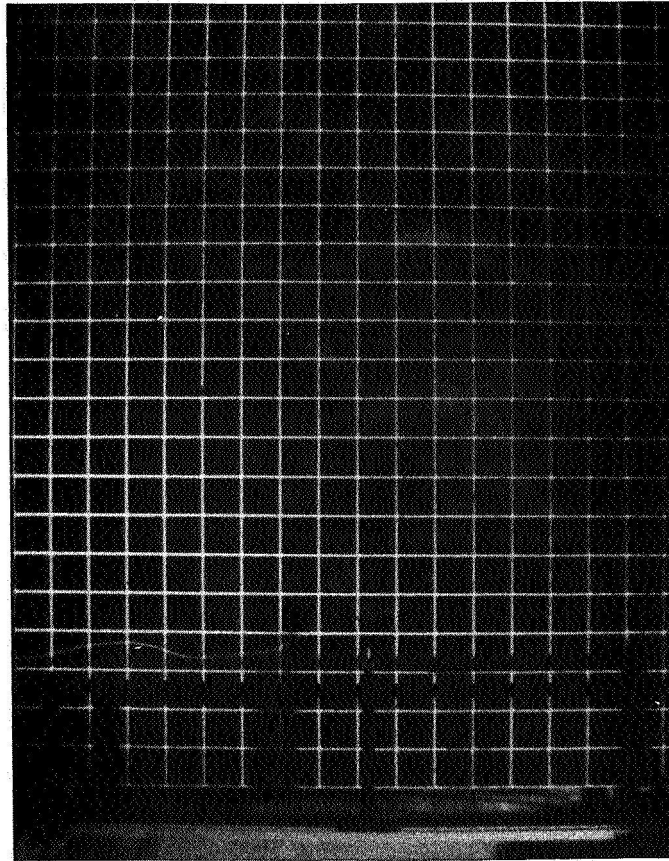


Figure 3-36: CABLE POSITION 0.09 SECOND AFTER RELEASE---RUN NO. 2

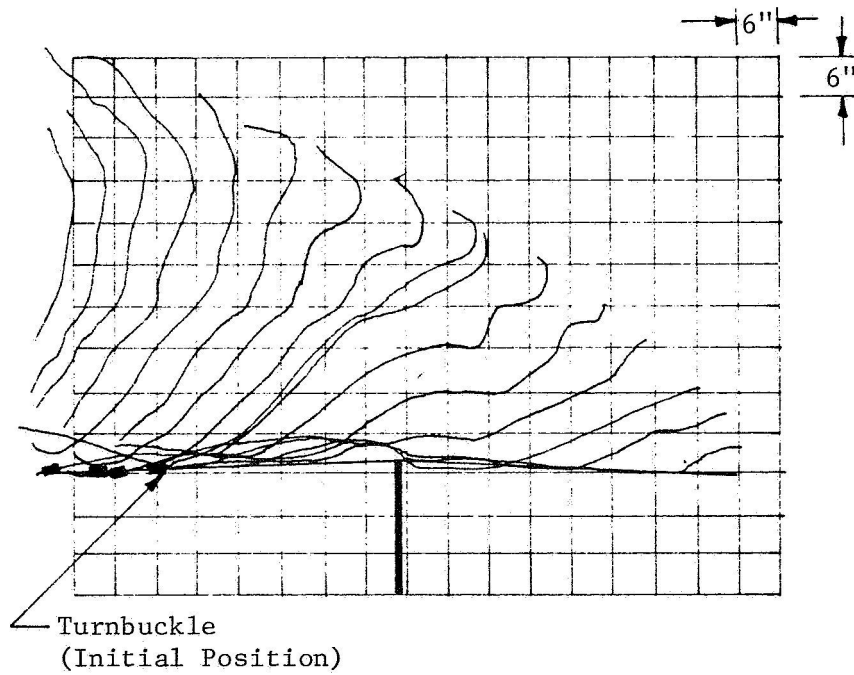


Figure 3-37: SEQUENTIAL CABLE POSITIONS---RUN 2

An explosive-actuated cable cutter at Station 123.7 will be used for the final tiedown release of each panel assembly and will be tested on MTA-4. The particular cutter originally proposed for this application will not be used because of supplier deficiencies noted in Phase II that also may have contributed to possible test anomalies on SCM 30-2.

The supplier failed to respond to repeated requests for evidence that the cutter would qualify as Category B devices under Air Force Eastern Test Range Manual (AFETRM) 127-1 safety requirements. When a response was received, it was negative and also noted that the devices used on the SCM 30-2 tests had a "no fire" current rating of 0.2 ampere. Further investigation disclosed that Boeing Quality Control had "disapproved" this supplier on another program. In addition, enough inconsistencies in part description and identification were noted to suggest a lack of adequate configuration control.

It has, therefore, been recommended that this supplier not be used for flight articles; however, their equipment was suitable for test in Phase II.

3.2.6 CONCLUSIONS AND RECOMMENDATIONS (SCM 30-2)

The boost tiedown release mechanisms will reliably release and eject the circumferential cables, but must be tested as a complete entity to evaluate performance of the total system.

It is concluded that:

- 1) The intermediate tiedown cables will jettison from the spacecraft without causing array damage when both ends of the cable are released simultaneously;
- 2) The intermediate tiedown cables will jettison from the spacecraft without causing array damage when both ends of the cable are released with a delay of 0.1 second between ends;
- 3) Friction values and bearing allowable stresses are within acceptable limits for the test operating conditions, and dry lube coatings performed as expected;
- 4) The ordnance cable cutters used were satisfactory as test equipment. However, the supplier's configuration control is inadequate for flight articles. These cutters are similar to the cutter to be used on the array at the Station 123.7 center tie.

It is recommended that:

- 1) The boost tiedown cable system be rigged with the turnbuckles on the early side of the release train;

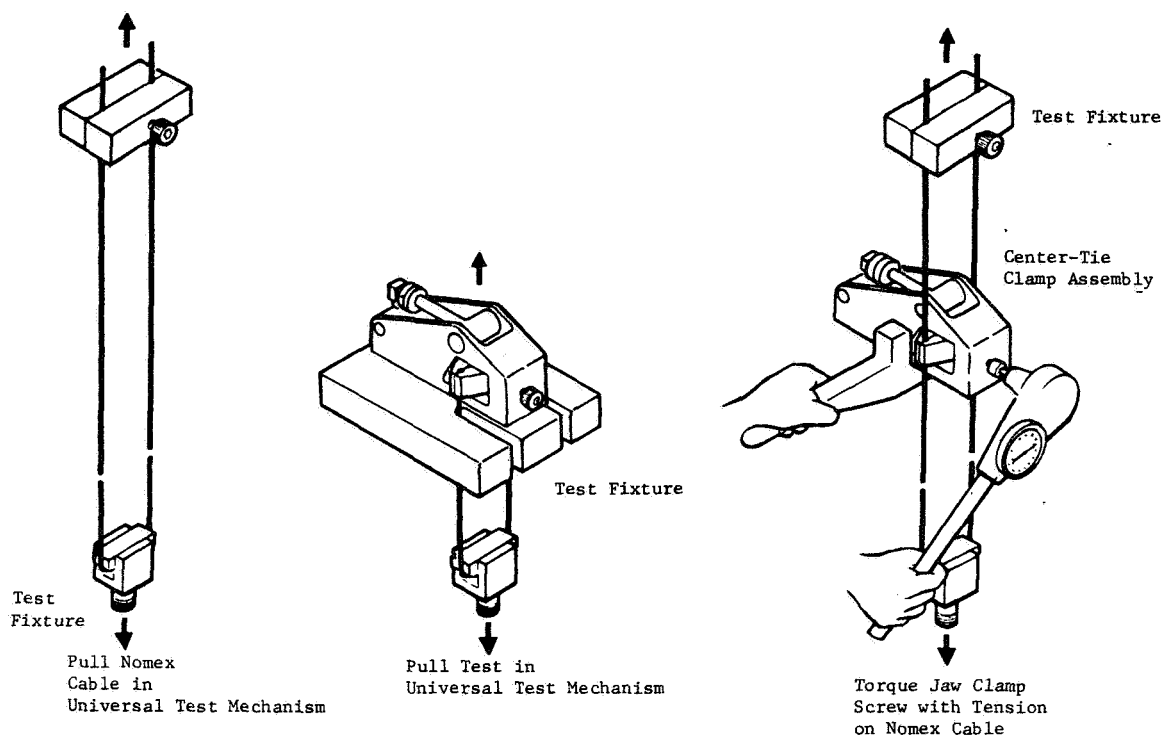


Figure 3-38: SCM 30-3 TEST SETUP

- 2) The boost tiedown cable system be revised to include release springs installed on the cable anchors;
- 3) A test be run to ascertain if hard-coat anodizing can be deleted from the boost tiedown pivot points.

3.2.7 DESIGN OF THE SCM 30-3 TEST HARDWARE

All test hardware was fabricated to prototype drawings and standards.

The fixtures for the SCM 30-3 test were designed to simulate the center-spar boost tiedown configuration and to mate with the universal tension test machine. Simulation of the 13 layers of substrate was not planned because it will be tested during Phase III.

Special tools required to adjust and set the clamps were tested to confirm adequate support of the system. Figure 3-38 shows the test fixtures for the three test conditions.

Test equipment used for this program includes the following.

Clamp Mechanisms Tests

- 1) Universal tensile test machine of 5,000-pound capacity
- 2) Baldwin-Tate-Emery testing machine (5-kip cap.) (Runs 22 through 25)
- 3) 11-0418207-2 cable guide pulley (1.45-in. diameter) with 3/16-in.-diameter pin hole added for Runs 10 through 28
- 4) 11-0418207-1 cable tension grip (wedge type)
- 5) Torque wrench (0- to 160-lb range)
- 6) Holding clip and release lanyard
- 7) PDI-M deflectometer
- 8) X-Y plotter (load versus deflection)
- 9) Strip chart recorder (load and deflection versus time)

Friction Coefficient Tests

- 1) Tinius Olsen 20,000-pound tensile test machine
- 2) Special hydraulic ram test fixture
- 3) X-Y plotter (load versus deflection)

The test article for SCM 30-3 was built to Boeing Drawing 25A27701 for the first series of tests and then modified per Drawing 29A18552 for the retest series. Figure 3-39 is the old clamp configuration and Figure 3-40 is the new clamp configuration. Incorporation of the wedge principle in the new design was based on the premise that the original configuration, having a fixed jaw displacement after initial clamping, lost its grip on the cable as the cable's cross section reduced under high tensile loading. Wedges, free to displace in the direction of the applied tensile load, would tend to continue closing on the cable, thus maintaining their clamping grip.

The design of the wedge inserts was based on the wedge formula:

$$P = 2Q \tan (\alpha + \phi)$$

where P = force to drive wedge

Q = resistance to movement (normal forces)

α = wedge angle

$\tan \phi = \mu$ = friction coefficient

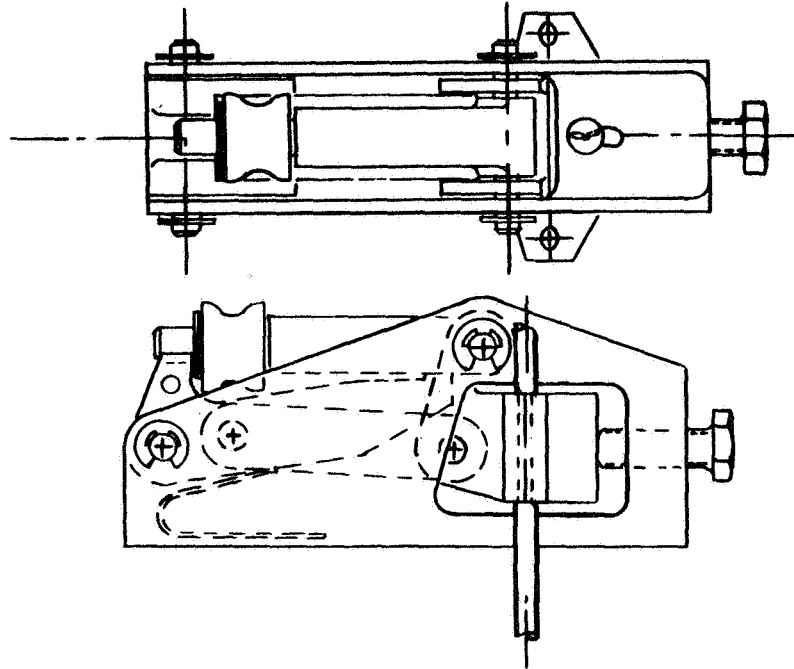


Figure 3-39: CENTER SPAR CLAMP ASSEMBLY---SCM 30-3 TEST ARTICLE

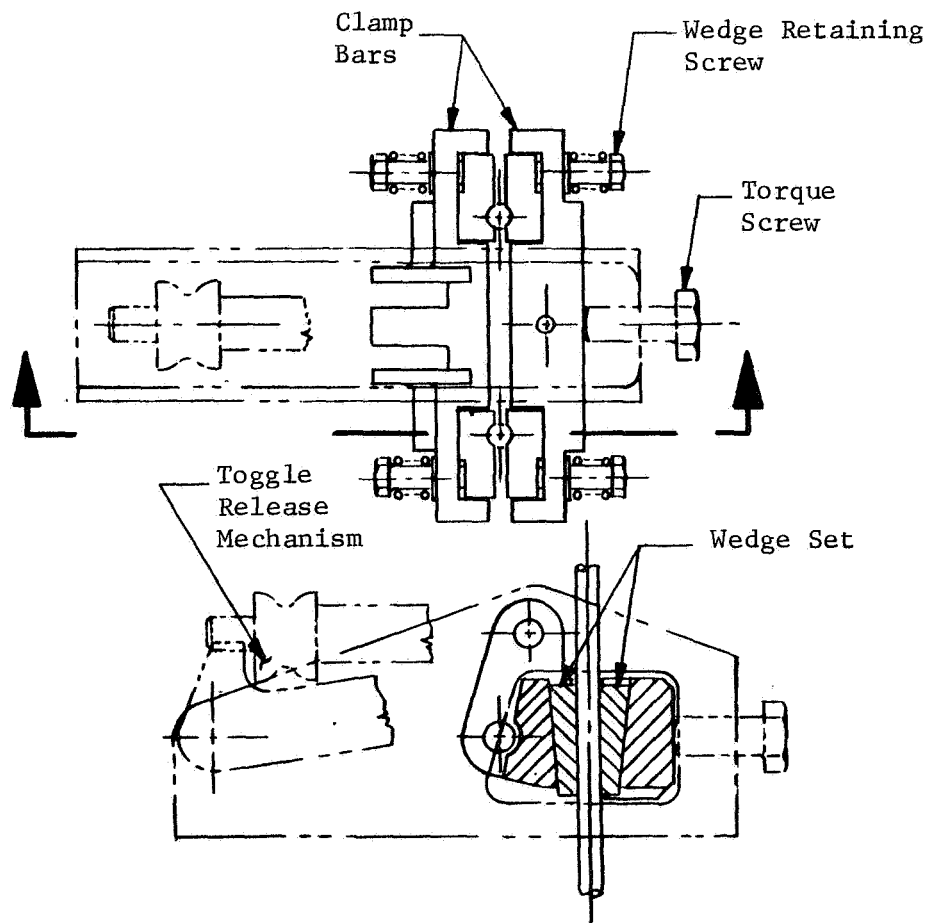


Figure 3-40: MODIFIED CABLE GRIP MECHANISM

3.2.8 CONDUCT OF CENTER-TIE RELEASE TEST (SCM 30-3)

The center-tie cable clamp redesign and test was required to improve clamping performance.

Test of Clamp Mechanism

Test procedures consisted of cutting the Nomex cable to length, clamping the free ends with the cable tension grip, installing assembly in the tensile test machine using the cable guide pulley grip, applying 1,090 pounds preload, installing the clamp assembly approximately 12 inches above the pulley, installing the safety clip with the release lanyard, and torquing the clamp torque screw to a predetermined value. Preload was then reduced to zero, the assembly was removed from the machine, and the cable ends were cut off flush with the tops of the wedge inserts. The clamp assembly, with cable, was then installed directly in the test machine, and tensile load was applied until cable slippage or breakage occurred. Load versus deflection was recorded by an X-Y plotter and strip-chart recorder.

Tests were run in three series. The first series was made using the smooth grip wedge sets, no cable inserts, and with clamp screw torques to 50, 76, and 100 inch-pounds. The second series was made using the same wedge sets and test conditions as the first series. The difference was that the pulley was fixed to prevent rotation and polyurethane inserts used for some of the test sequences. A design deficiency was discovered wherein the clamp jaws that house the taper wedges deflected under the wedge loads. The obvious solution would be to rebuild the parts designed to reduce deflections, but it was decided to use machinist C-clamps on each end and complete the wedge evaluation test. Accordingly, the third and final series was completed using only the smooth-grip wedge sets without polyurethane cable inserts. Because of this lack of structural stiffness, parallel clamps were applied to both ends of the clamp bars in line with the wedge sets, after the initial preloading operations, as shown in Figure 3-41. These clamps were installed by hand, with no data being obtained on clamping force.

The first valid run in this series, Run 23, produced a tensile load of 1,140 pounds, with one wedge set bottomed, and the other near the end of its stroke. Increased clamping pressure with the parallel clamps for Run 24 produced a tensile load of 1,400 pounds before a wedge set bottomed and released its cable end. A further increase was made in the parallel clamp pressure for Run 25 and tensile loading was applied until one cable end broke at 1,705-pound loading. This value actually corresponds to approximately 852 pounds per cable, ignoring any possible differential in cable tensions between the two sides resulting from fixing the cable guide pulley.

Run 25 exceeded the required Station 10.5 preload value despite the clamps' structural deficiency and Runs 26 and 27 were made to evaluate

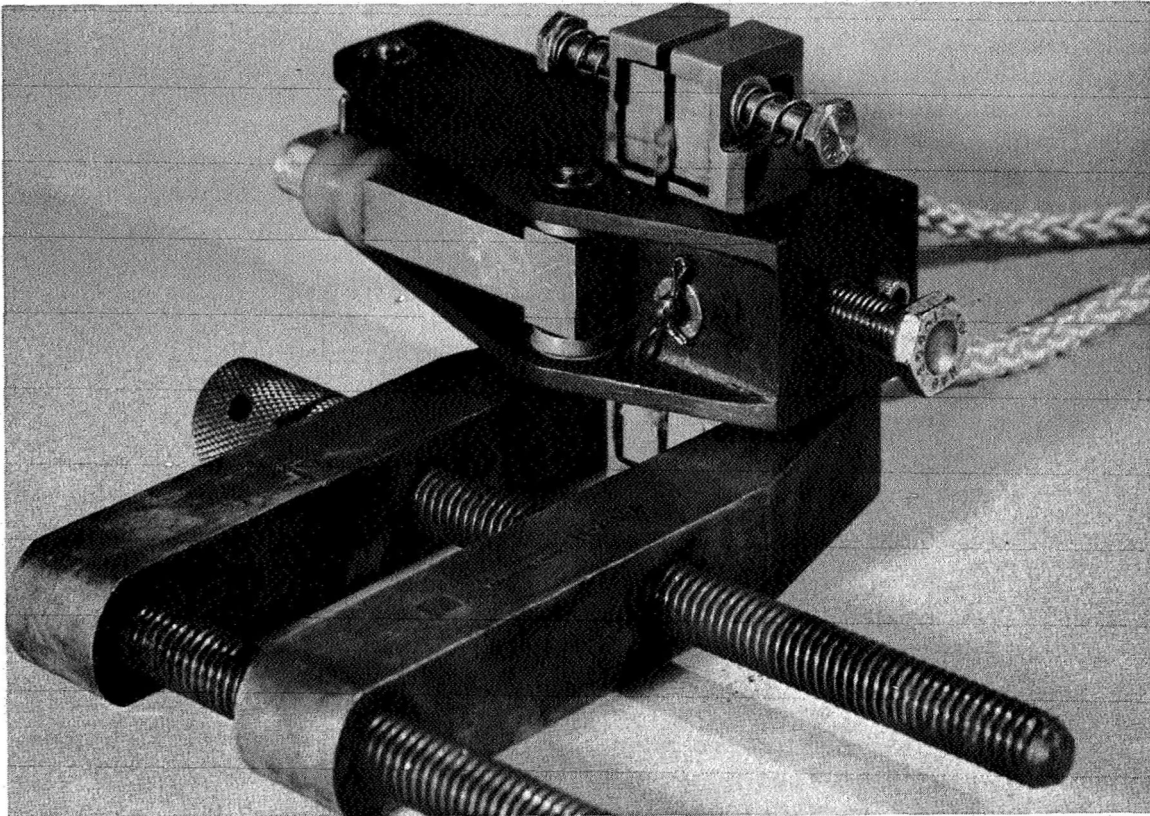


Figure 3-41: MODIFIED CLAMP ASSEMBLY TEST SETUP

the clamps' long duration (7-day) cable grip capability and release characteristics, as well as the cable load relaxation characteristics with one preload cycle.

3.2.9 RESULTS AND ANALYSES OF SCM 30-3 TESTS

The center-tie release mechanism met all test objectives and provided valuable engineering design information.

- A. Development of the tensile preload required to achieve the final Station 10.5 load of 1,090 pounds

This test objective was met in Test Run 25, when the cable was broken at 1,705 pounds without release, and again in Run 27, when the planned 1,560-pound preload was applied at the beginning of the 7-day test.

These runs demonstrated the effectiveness of the wedge principle of this application despite the structural deficiency of the clamp bars. Use was made of parallel clamps to react the higher than expected clamping loads as a matter of time and cost expediency.

The tests also indicated that the principal reason for cable slippage during the original clamp mechanisms tests was the reduction

of cable cross section at high-tensile loading, rather than a too-low friction coefficient. This was evident from the fact that in every valid run during the retest, with exception of Run 20, the cable slipped out only after, and if, one or both wedge sets had bottomed as a result of deflections of the clamp bars. This action effectively prevented further closing of the wedge sets on the cable ends and slippage resulted.

B. Maintenance of grip on cables for 7 days

This test objective was met at the end of Run 27, although the final load at that time had decayed 120 pounds below the required Station 10.5 final load of 1,090 pounds. Neither wedge set had displaced more than approximately one half its available travel, and most of this travel was noted within the first minutes after applying pre-load tension.

Based on the approximate cable load relaxation percentage experienced during this run, an initial preload tension of 1,760 pounds would have had to be applied. This load could be developed by a clamp mechanism with stronger clamp bars, but the breaking strength of the 5/32-inch Nomex cable used in the tests would have been exceeded. The alternative solutions are to increase cable size, revise the preloading operation to include several successive preload cycles to a lower level, as will be done in the preparation for a launch, or a combination of both.

C. Release of both cable ends simultaneously

The test objective was met at the end of Run 27 when the lanyard release was actuated. Although the tensile load had to be reduced to 500 pounds to preclude wedge bottoming and cable slippage when the parallel clamps were removed, this test was considered valid since, as it is reasonable to assume, cable release would be at least as effective at the higher required Station 10.5 load of 1,090 pounds.

D. Nomex cable load relaxation characteristics versus time

Plots produced during Run 27 provided data on cable load relaxation characteristics. At the end of the 7-day period, the initial tensile load of 1,560 pounds had reduced 590 pounds, resulting in a load relaxation value of 37.8%. This value was higher than that resulting from the original test. Load decayed rapidly during the first few minutes, as shown in Figure 3-42, and then leveled off to a fairly even slope. Figure 3-43 shows that the long-duration load decay is essentially exponential.

E. Friction coefficients effects on testing

The design of the wedge sets was based on a conservative estimate of 0.20 for friction coefficient (μ_1). Data in Figure 3-44 shows the actual value to be approximately 0.07 to 0.08 for the range of bearing pressures experienced during the test.

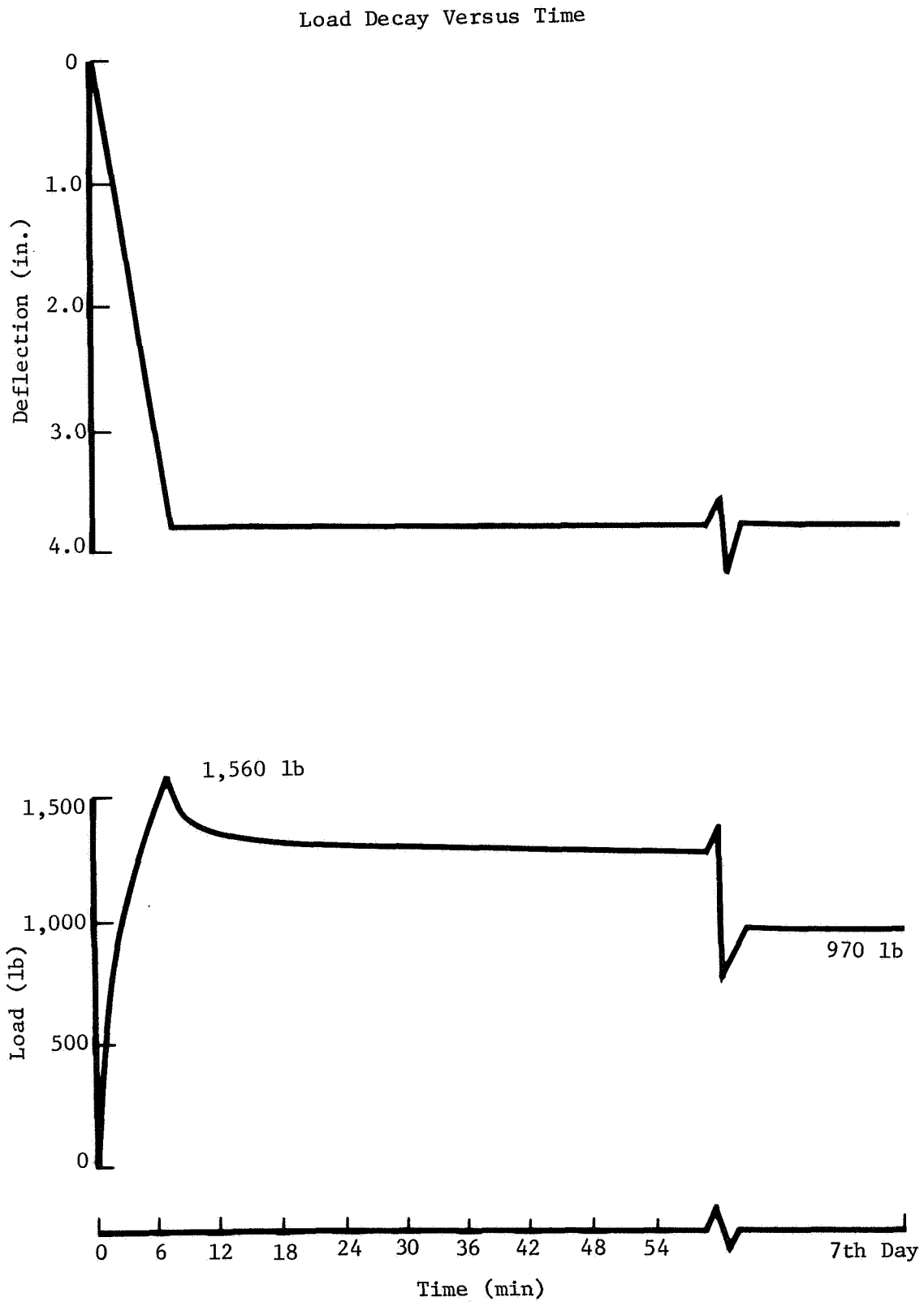


Figure 3-42: NOMEX CABLE LOAD DECAY CHARACTERISTICS

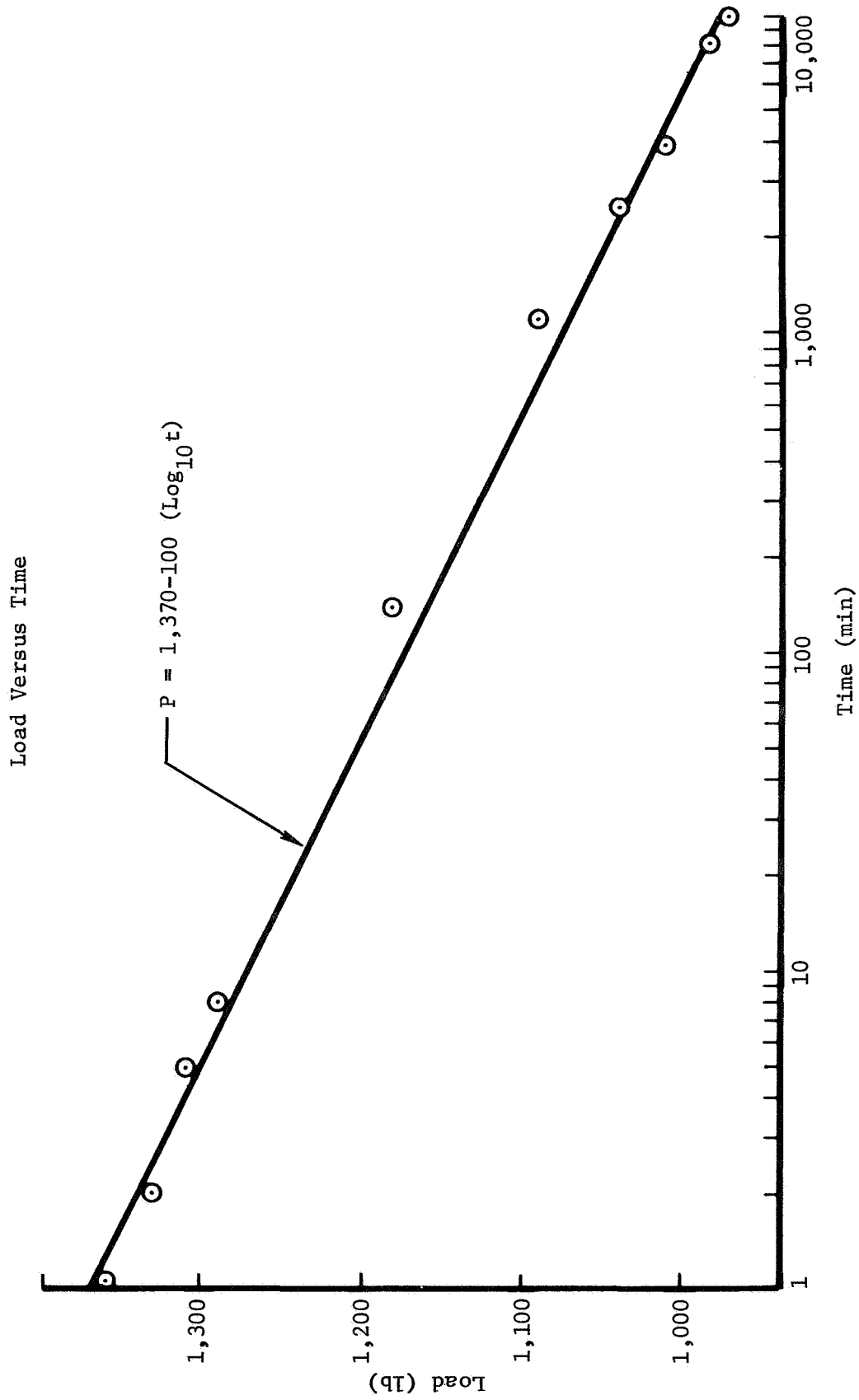


Figure 3-43: LONG-TERM LOAD DECAY

Coefficient of Friction Versus Clamping Load

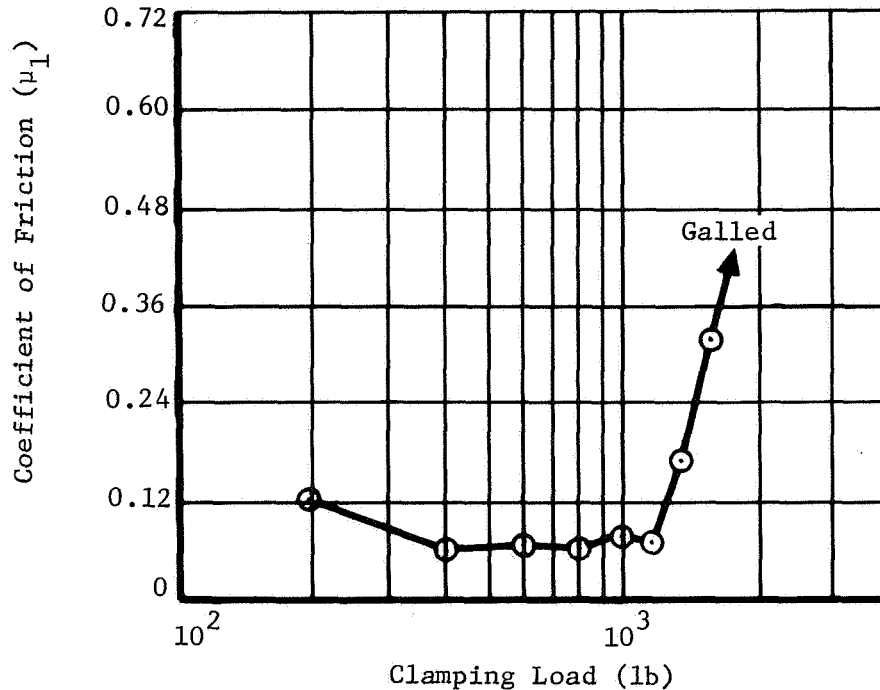


Figure 3-44: NOMEX CABLE CLAMP FRICTION COEFFICIENT

Since clamping load (Q) is a function of wedge angle (α) and (μ_1), the actual loads experienced by the clamp bars were significantly greater than those predicted.

The resulting total clamp load (Q) for the Station 10.5 final cable load (P) of 1,090 pounds could approach 3,450 pounds instead of the calculated 1,860 pounds, as shown in Figure 3-45.

The total clamp load when subjected to the preload (P) of 1,560 pounds could have approached approximately 4,875 pounds. The calculated load for this condition was 2,660 pounds.

The minimum cable friction coefficient (μ_2) necessary to prevent slippage is calculated to be 0.158.

Friction coefficient data shown in Figure 3-46 indicate that actual (μ_2) values using hydroblasted surfaces exceed the minimum required value of 0.158 by a small margin at the higher clamping load.

3.2.10 CONCLUSIONS AND RECOMMENDATIONS (SCM 30-3)

The boost tiedown center-spar release mechanisms will successfully clamp and release the center tiedown cables but must be tested as a complete entity to evaluate performance of the total system.

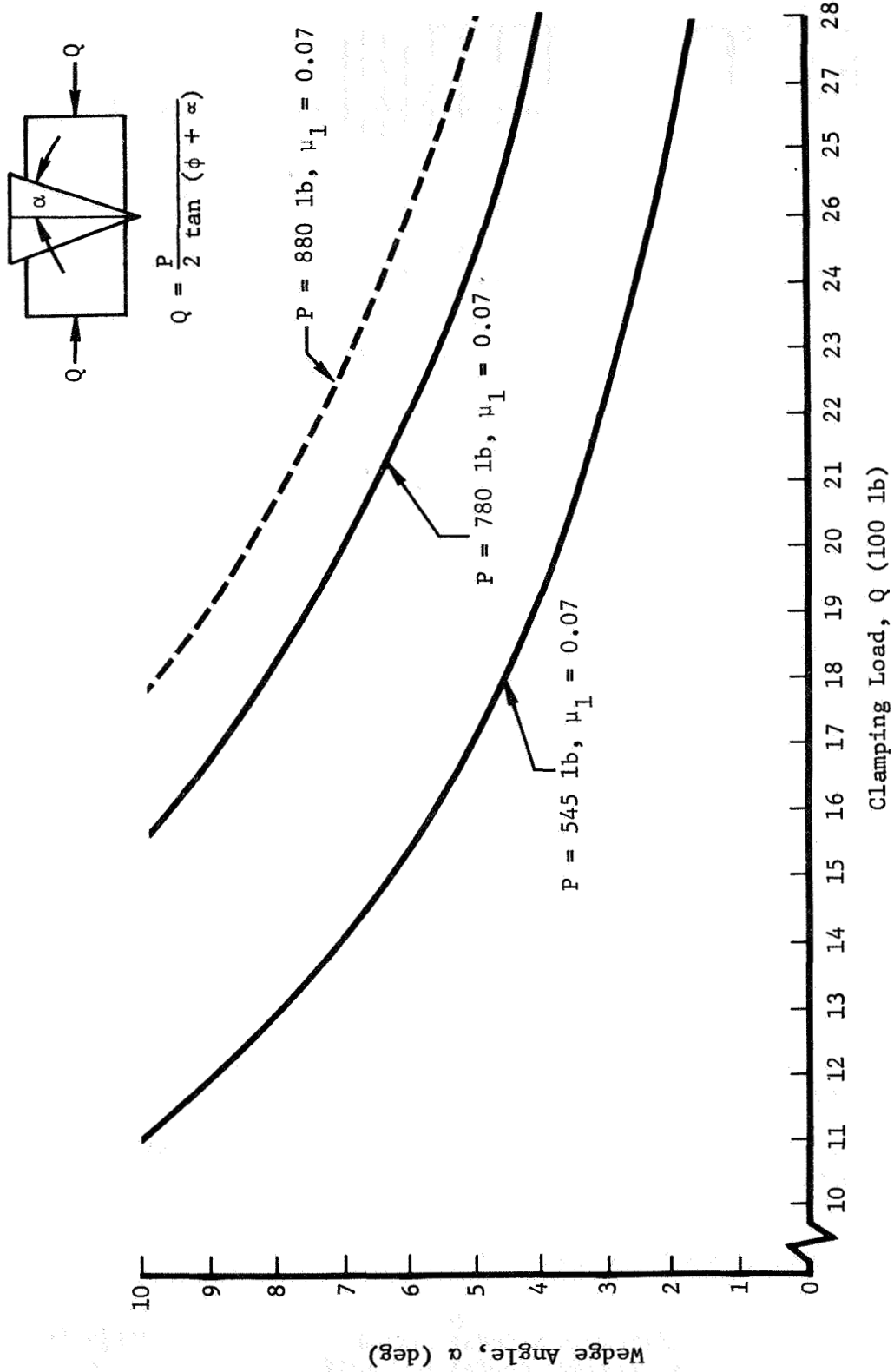


Figure 3-45: WEDGE ANGLE VERSUS CLAMPING LOAD

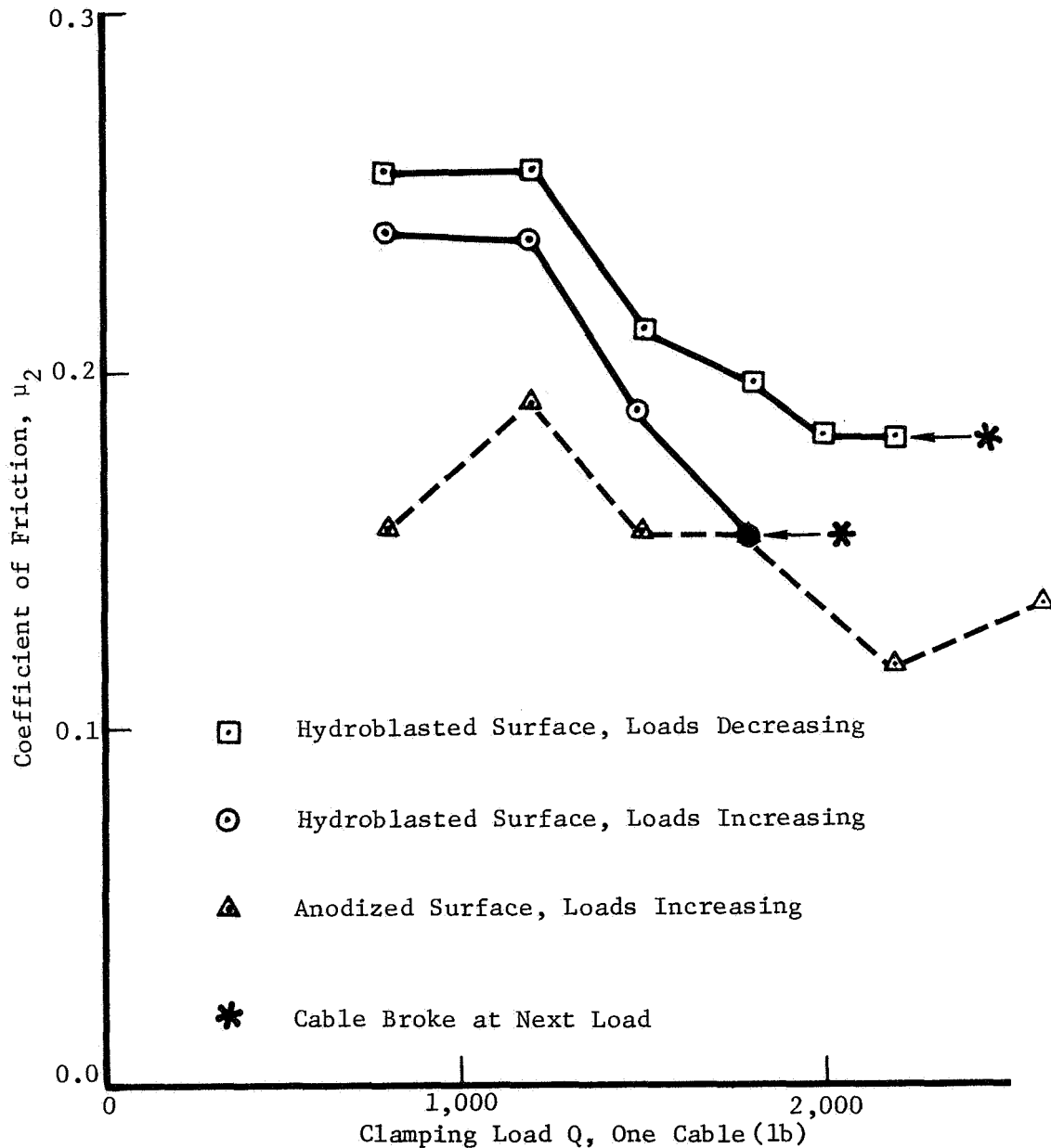


Figure 3-46: CABLE FRICTION COEFFICIENT VERSUS CLAMPING LOAD

Conclusions:

- 1) The clamp mechanism, incorporating the wedge principle, is capable of developing and maintaining the required Station 10.5 tensile load conditions.
- 2) Cable coefficient of friction is adequate, when using smooth-grip wedges, for the Station 10.5 loads. The use of threaded-grip wedge sets and cable inserts provides no clamping advantages.
- 3) The clamp mechanism, as designed and tested, does not have sufficient strength and rigidity to react the clamping loads generated by the Station 10.5 tension loads.

- 4) The load relaxation characteristics of the 5/32-inch-diameter Nomex cable used in testing are such that its breaking strength will be exceeded if a single preload cycle is used to attain the final required Station 10.5 load of 1,090 pounds.
- 5) The clamp assembly as modified will maintain the cable under load for more than 7 days.
- 6) The clamp assembly will reliably release both ends of the Nomex cable.

Recommendations:

- 1) That the clamp mechanism design using the wedge clamping principle be modified to correct structural deficiencies and minimize clamping loads and assembly weight;
- 2) That a larger diameter cable be provided for the Station 10.5 application;
- 3) That rigging procedures be tested for the total system using MTA-4 hardware;
- 4) That further storage tests under load be conducted as a Phase III activity to determine if rigging loads require checking after a specified time.

4.0 MAJOR TEST ARTICLE 1 (MTA-1) DEVELOPMENT (TASK 3)

The Large Area Solar Array Main Subpanel 1 (MTA-1) was selected to verify fabrication capability and evaluate its structural characteristics. This subpanel encompasses all of the problems in the array that are found in a single subpanel.

This section provides a comprehensive report of the activities associated with the fabrication and vibration testing of MTA-1. There are five sections dealing with MTA-1 and one section dealing with a radiation damage analysis of a LASA-type subpanel during a 210-day mission to Mars.

The five sections are as follows:

- 1) Design and Analyses of MTA-1;
- 2) Fabrication of MTA-1;
- 3) Quality Assurance;
- 4) Vibration Testing of MTA-1;
- 5) Conclusions.

4.1 MTA-1 DESIGN AND ANALYSES

The Major Test Article 1 (MTA-1) was designed in Phase I. The structural and dynamic analyses for the solar panel assemblies also were completed during Phase I. Some minor additional design and analyses were necessary in Phase II to satisfy changing requirements in live solar-cell installation and tests.

The purpose of this section is to provide a general description of the MTA-1 design analyses necessary to evaluate the test results, and changes in the test article design arising from the reduced requirements for live cells on the subpanel test article.

4.1.1 DESCRIPTION OF MTA-1 TEST ARTICLE

The MTA-1 test article is a prototype structural main subpanel.

Major Test Article 1 is the primary structural subpanel of the Large Area Solar Array. The four No. 1 subpanels, when tied together by the tiedown system, make a box attached to the spacecraft by 12 hinges, three locations on each side. All other subpanel units, 2 through 5, are supported by this subpanel.

This subpanel was selected for the MTA-1 because it contained all of the design and fabrication problems of the other subpanel units plus its own unique characteristics.

The subpanel is identical to prototype design except for the simulation of live solar cells and power buses. The cells were replaced with glass, which simulates the weight of the cells, coverglass, and adhesive. The buses simulate mass and strength.

The MTA-1 structure is a rectangular, semirigid, beryllium frame supporting an open-matrix pretensioned, fiberglass substrate, which in turn supports the solar-cell modules. An illustration of the subpanel is shown in Figure 4-1. The structure is bonded with AF-126 adhesive. The frames are assembled from 0.010- to 0.085-inch-thick powdered beryllium sheet. The major structural joints use electron beam (EB) welded titanium fittings bonded into the frame to distribute concentrated loads into the beryllium. The substrate is made of 0.003-inch-thick fiberglass tape 0.14-inch wide pretensioned diagonally across the panel to form a flat membrane support for the solar-cell modules.

The substrate tape is positioned to provide a node or cross at the center and each corner of all solar cells.

The basic member of the frame is a rectangular section made of two beryllium channels and two cap strips. The spars and edge members consist of two of these sections, one on each side of the substrate. Titanium shear clips and splice plates are used at the intercostal joints where beryllium is not necessary for stiffness, and the minor weight increase is acceptable.

MTA-1 consists of a darkside frame that contains all the intercostals, and a Sunside frame with a pretensional substrate bonded between them.

The power generation and transmission systems on MTA-1 were revised by Contract Change Notice 622. The subpanel originally contained all live solar cells and two transmission systems, a 28- and a 100-volt. It was changed to a subpanel having only six live modules, 50 modules of simulated solar cells, and power buses that physically simulate the live system but are without electrical continuity. With deletion of live solar cells the prototype bus system was not required; therefore, it was simulated to reduce cost.

The power generation system on MTA-1 consists of a net area for modules of 101.57 square feet. The net solar-cell module area is defined as the sum of all the areas enclosed within the perimeter of a module. This area includes the cells and spaces between cells within the module, but does not include the spaces between modules or the spaces between the modules and the spars. The power output for this subpanel, calculated in accordance with Paragraph 3.4.1 of JPL Specification GMP-50505-FNC-B, is 1,015.7 watts at air mass 0, 1 AU, and at 55°C.

The electrical design of MTA-1 obtained the highest possible solar-cell packing factor. During Phase II all panels were redimensioned to fit the 406-cell module. Spacings of 0.01 inch on noncontact sides and 0.02 inch on the contact sides were selected. Each group of seven cells also provides the 0.03-inch spacing between modules so that spacing on the non-contact sides is closer to 0.005 inch.

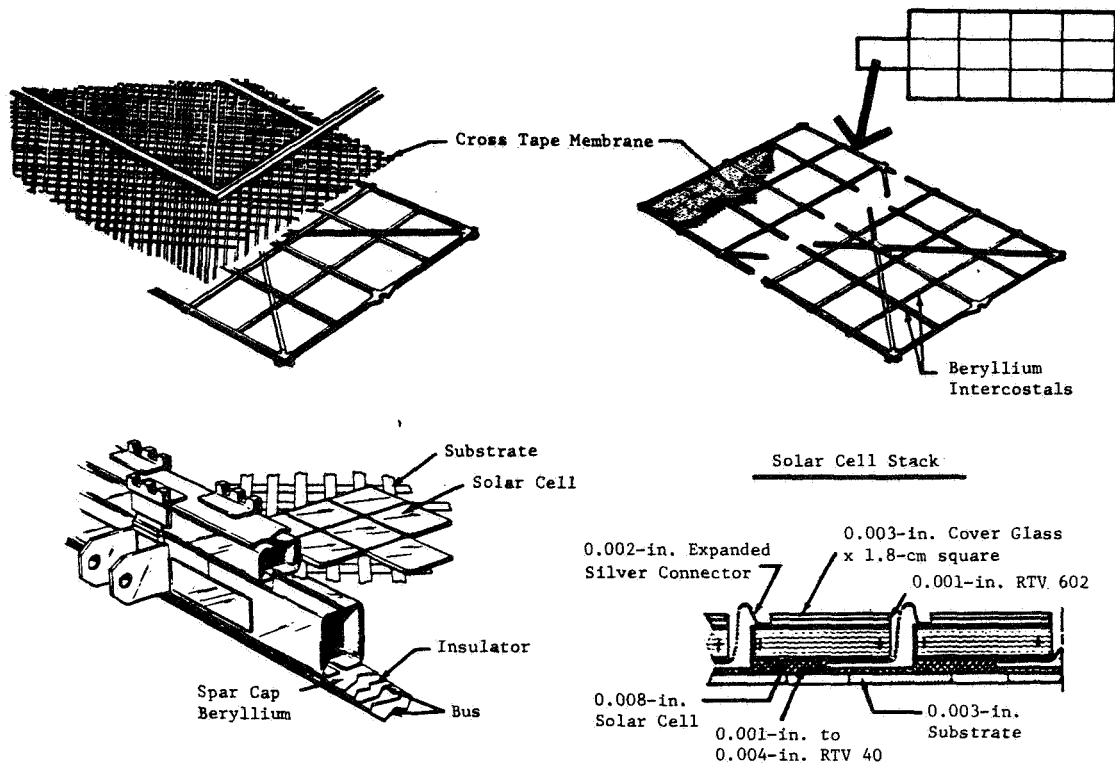


Figure 4-1: SUBPANEL NO. 1 ASSEMBLY (MTA-1)

The spacings that were selected gave a solar-cell density of 223 cells per square foot of net module area. The MTA-1 net module area is large enough to contain 56 modules. Six of the modules contain live solar-cell assemblies. Each module contains 406 8-mil N on P solar cells arranged in a 7- by 58-cell rectangle. Seven cells are parallel connected into a group. Fifty-eight of these 7-cell groups are then series connected to form the basic 406-cell module with a 28-volt power output. Four 28-volt modules connected in series form one 100-volt module. The remaining area contains glass cells that mass-simulate the solar-cell assemblies. Prototype connections between the six modules and the bus bar were made to provide a test article for checking the structural capabilities. This connection is shown in Figure 4-2. A pulse soldering process utilizing a Raytheon Weldpower Heat Programmer and weighted soldering electrode is used for solar-cell interconnection joints.

The design of the solar-cell silver mesh interconnectors was revised to allow the use of one tool for three parts, thereby reducing the tooling costs.

It was assumed that the average width of the cells would be at the mid-point of the solar-cell tolerance, 0.786 inch. However, when solar cells were received and measured, a great proportion were at the upper limit of tolerance, 0.791 inch. Very few cells were as low as 0.788 inch, which was the assumed average.

The power transmission system on MTA-1, shown in Figure 2-7, is made of flat aluminum and copper sheet bonded to the structure with the specified adhesives (RTV-630). Both materials are used on MTA-1 to simulate the prototype (28- and 100-volt) systems. The buses on MTA-1 were designed to simulate the structural and not the electrical characteristics. The material used was equivalent to the prototype design. However, the insulation was eliminated and the wires were made discontinuous to facilitate fabrication.

4.1.2 DESIGN OF SIMULATED POWER SYSTEM

Simulated electrical parts match all physical characteristics of electrically operative parts.

During October 1967, the rejection rate for solar cells rose from approximately 1 to 10% due to peeling of the contact surface. The rejection rate continued to rise and the assembly of solar-cell modules was discontinued. At this time, JPL issued a Contract Change Notice 662, which authorized the deletion of all live cells and replacement with simulated solar cells and buses.

Simulated Solar Cells---All but six 28-volt modules were replaced with glass. Cell characteristics simulated were size, weight, and modulus of elasticity.

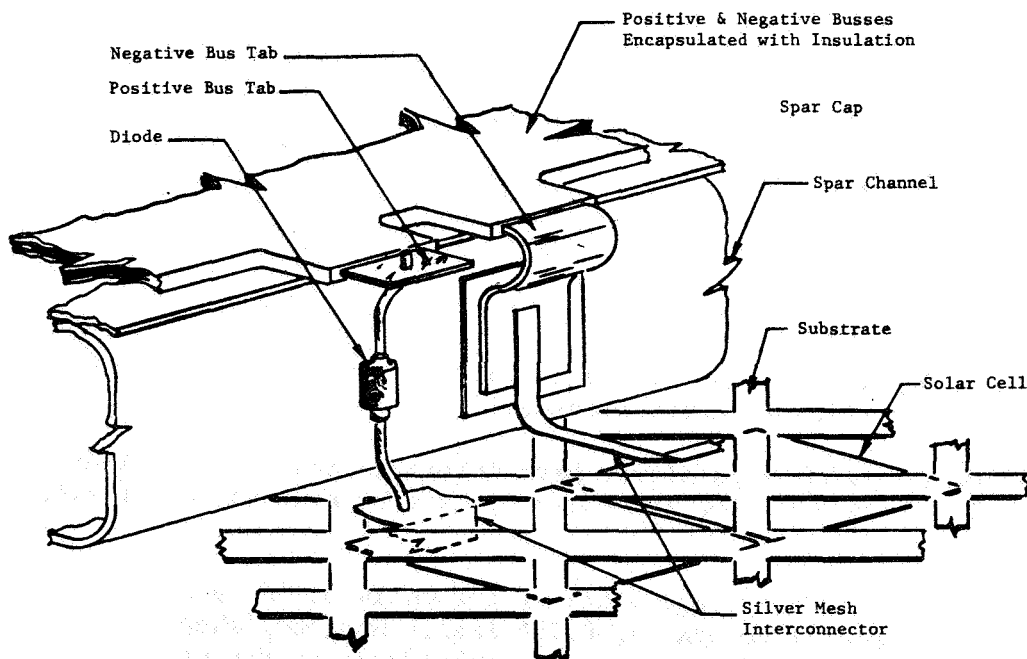


Figure 4-2: MODULE TO BUS CONNECTION

The six solar-cell modules are not electrically operative, but all electrical connections and diodes were installed to obtain a physical evaluation of the electrical system. The location of the live cells is shown in Figure 4-3.

Simulated Power Buses---The simulated power buses are very similar to electrically operative buses. Aluminum and copper were used but the buses are not made in one piece. This had a negligible effect on the MTA-1 vibration test but substantial manufacturing cost savings were realized. No new processes were required to make this installation. Bondline thickness and tolerances are the same as those that were specified for electrically operative buses.

4.1.3 ANALYSIS OF MTA-1 VIBRATION CHARACTERISTICS

Pretest analysis of the vibration characteristics of the MTA-1 was required to predict the response levels and to establish the correct requirements and procedures.

MTA-1 was subjected to a vibration test that had as its basic objectives:

- 1) The determination of the structural capability of electrical components when subjected to large-amplitude substrate motion;
- 2) The comparison of measured with calculated frequencies in air and in vacuum;
- 3) The evaluation of damping characteristics in air and in vacuum.

To accomplish these objectives it was necessary to duplicate the predicted boost vibration environment at the substrate supporting the cells.

The normal modes of the pinned-free MTA-1 panel have been computed to predict the response levels and select the most appropriate shaker locations. The following conditions were considered:

- 1) Panel alone in vacuum;
- 2) Panel in vacuum with 3.5-pound shaker mass at Station 169.9 on center spar;
- 3) Panel in air with 3.5-pound shaker mass at Station 169.9 on center spar;
- 4) Panel in air with 3.5-pound shaker mass at Station 123.7 on the center of a lateral spar;
- 5) Two 25-pound shakers at Station 169.9 on outer spar in vacuum.

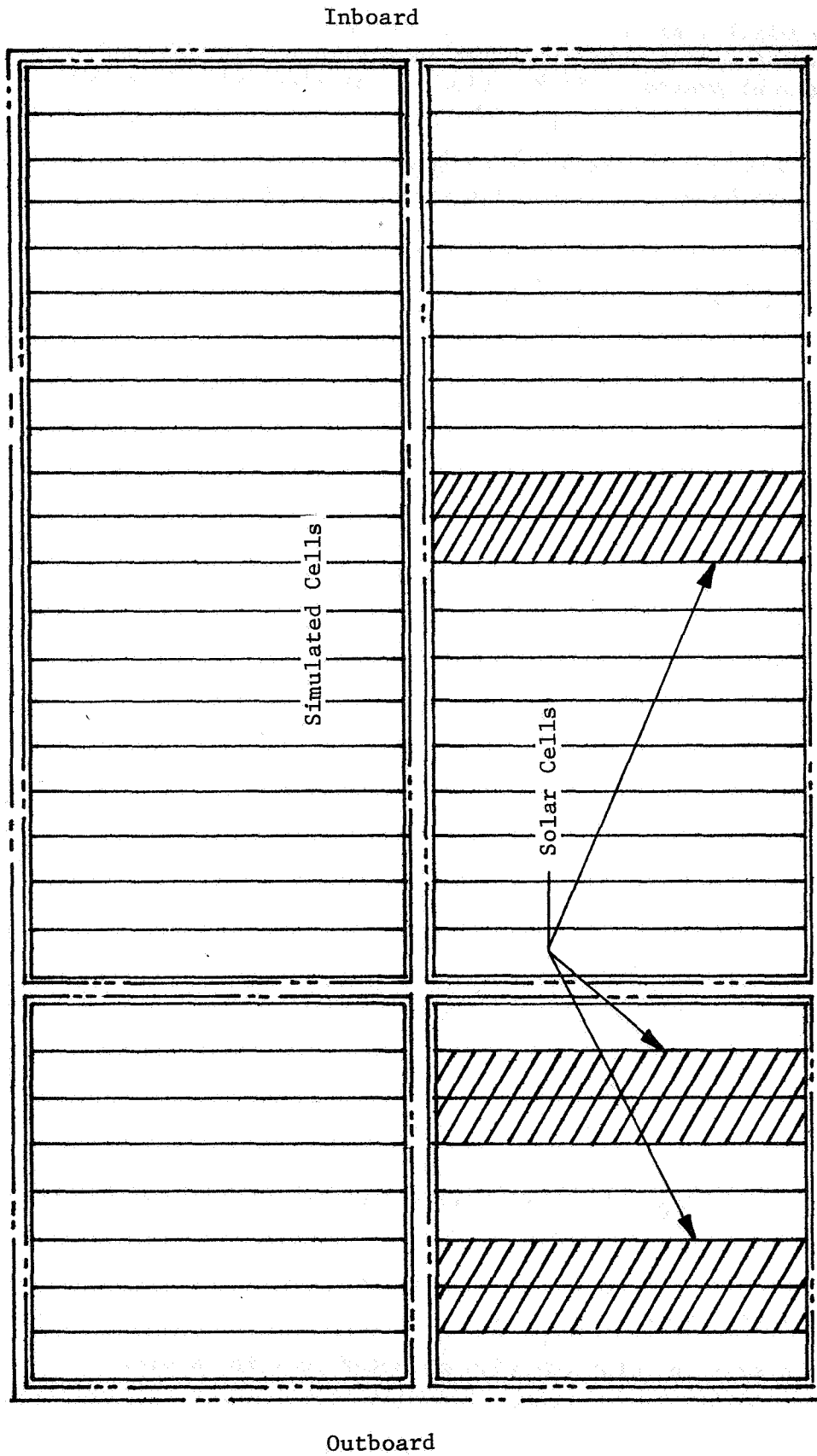


Figure 4-3: MTA-1 LIVE CELL LOCATIONS (DARKSIDE SHOWN)

Condition 5 was selected because it provides bending and torsion modes with adequate force with low voice coil mass. The test setup showing the location of the shakers is shown in Figure 4-4.

In the analysis the mass of air applied to the panel for the ambient atmospheric test was taken as the mass in the volume shown in Figure 4-5. The mass of air in a 2-foot-diameter sphere was applied to each substrate mass, and the remainder was distributed over the panel structure. A summary of the analysis of the nominal case (Condition 5 above) is given in Table IV-1.

Table IV-1: SUMMARY OF ANALYSIS OF MTA-1 VIBRATION CHARACTERISTICS

<u>Mode</u>	<u>Frequency (Hz)</u>	<u>Generalized Mass</u>	<u>50-lb Input Substrate Response</u>	<u>50-lb Input Structure Response</u>
1	6.50	0.00330	--	--
2	15.4	0.00230	33.4	32.9
3	24.3	0.00187	--	--
4	31.6	0.000736	37.6	6.15
5	32.6	0.000471	42.6	3.42

Condition 5: Vacuum, two 25-lb shakers at Station 169.9 outer spars; Shaker Mass = 1.25-lb total.

4.2 FABRICATION OF MTA-1

The fabrication and assembly of MTA-1 was completed on August 2, 1968. A photo of the completed MTA-1 is shown in Figure 4-6. The fabrication effort proved manufacturing techniques experimentally developed in Phase I and developed manufacturing cost data based on actuals for Phase III costing.

MTA-1 was selected for Phase II fabrication because it is the most complex and was, therefore, the best test of manufacturing capability. MTA-1 is a lightweight bonded beryllium and titanium 8- by 13-foot structure consisting of Sunside and darkside frames bonded about a pretensioned fiberglass tape substrate matrix. Live and dummy solar-cell modules are bonded to the substrate. An MTA-1 exploded view identifying its main elements is shown in Figure 4-7.

The text that follows is devoted to the LASA Phase II fabrication and assembly operations developed to manufacture MTA-1. It is organized in a manner to familiarize the reader with the overall manufacturing plan and followed by reviews in greater detail of the fabrication stages within the plan. Those wishing additional detail may refer to D2-113353-1, *Manufacturing Plan and Specification*.

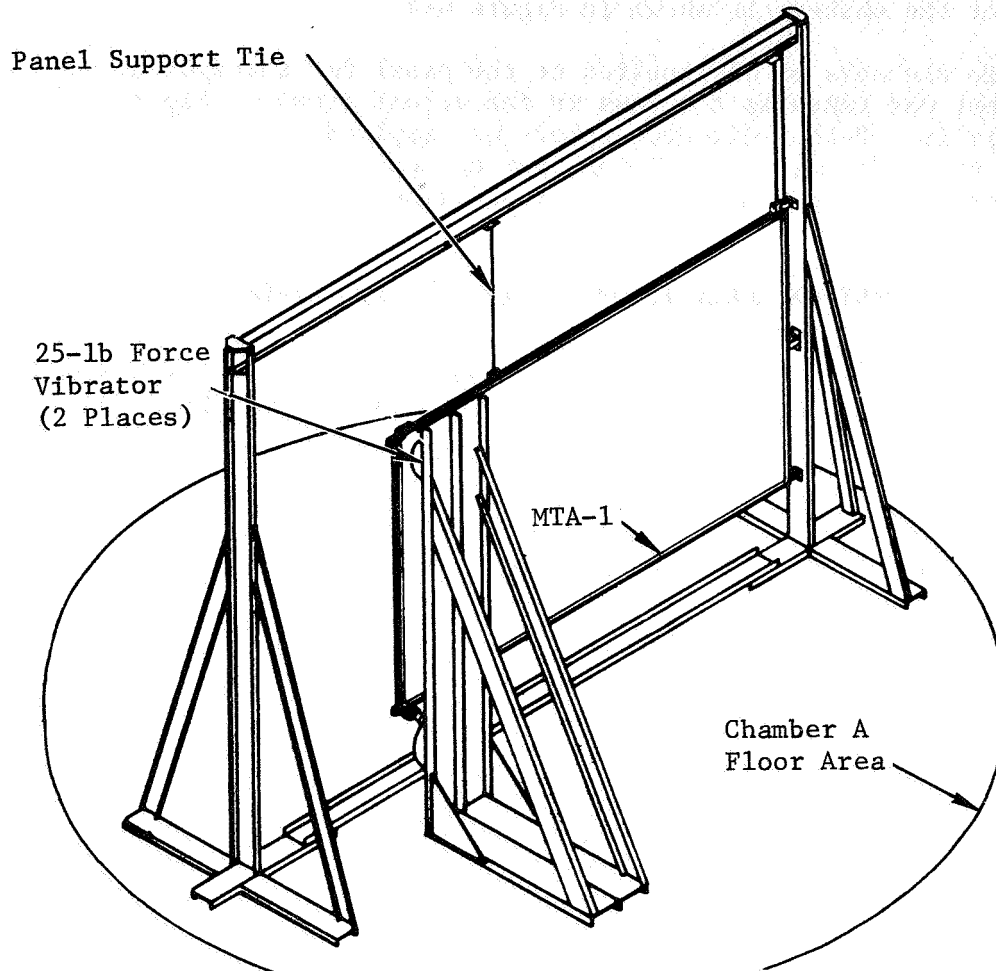


Figure 4-4: MTA-1 VIBRATION TEST SETUP

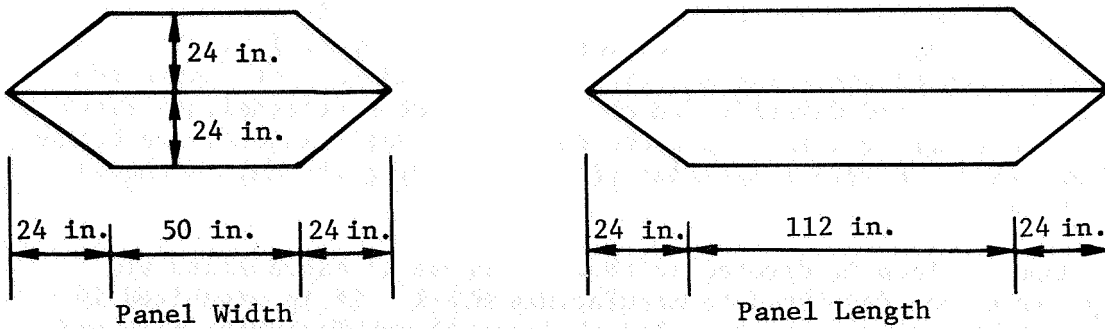


Figure 4-5: ASSUMED VOLUMES OF AIR MASS

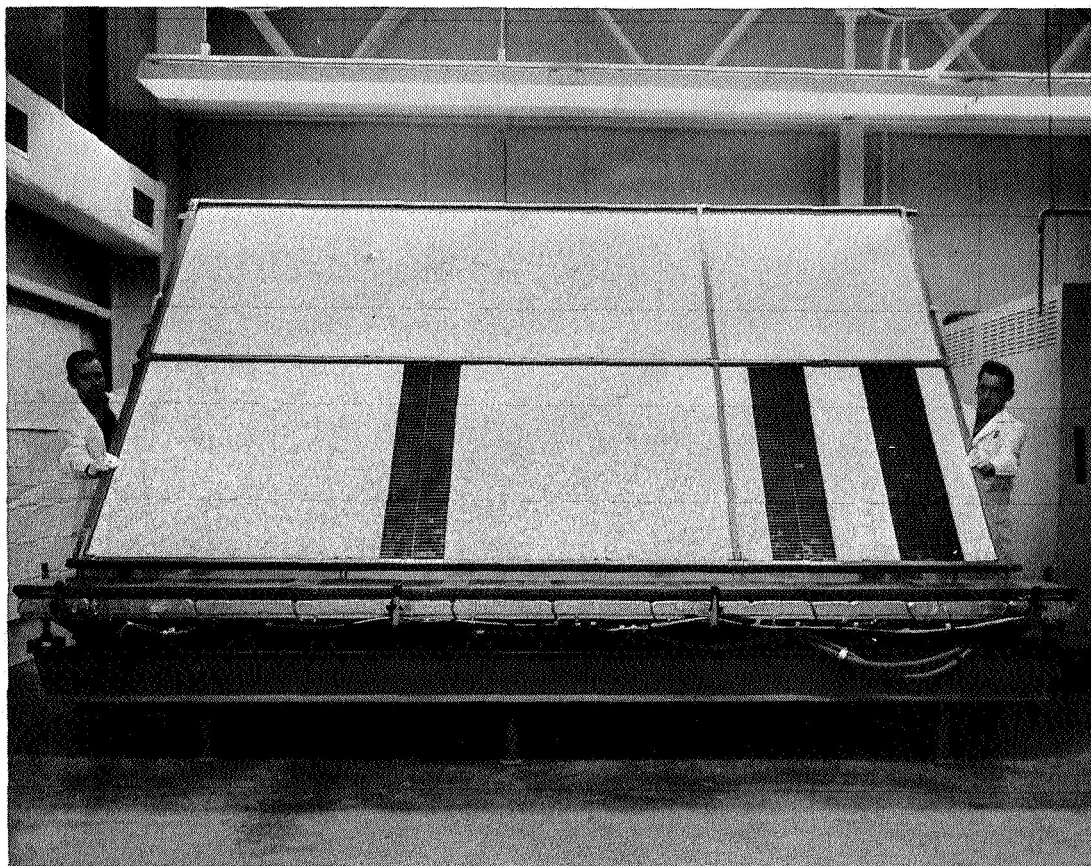


Figure 4-6: MAJOR TEST ARTICLE NO. 1, MTA-1

4.2.1 MTA-1 MANUFACTURING PLAN

Phase II fabrication effort demonstrated that manufacturing plans developed in Phase I were sound.

Manufacturing effort implemented in Phase II followed the plan developed at the conclusion of Phase I with revisions to accommodate the change from a live to a simulated power system. The plan is schematically shown in Figure 4-8.

Beryllium cross-rolled sheet material is procured in strips to the width required for forming channels or to the net width for flat detail parts. The channel stock is formed at 1,375°F in stainless steel dies. Pneumatic pressure is applied after initial forming to size the channel cross section to the punch and to bottom the punch in the die. Parts are cooled to 800°F before removal from the die. Channel sides are trimmed to the finished dimension in a trim fixture, using an abrasive saw. Channel

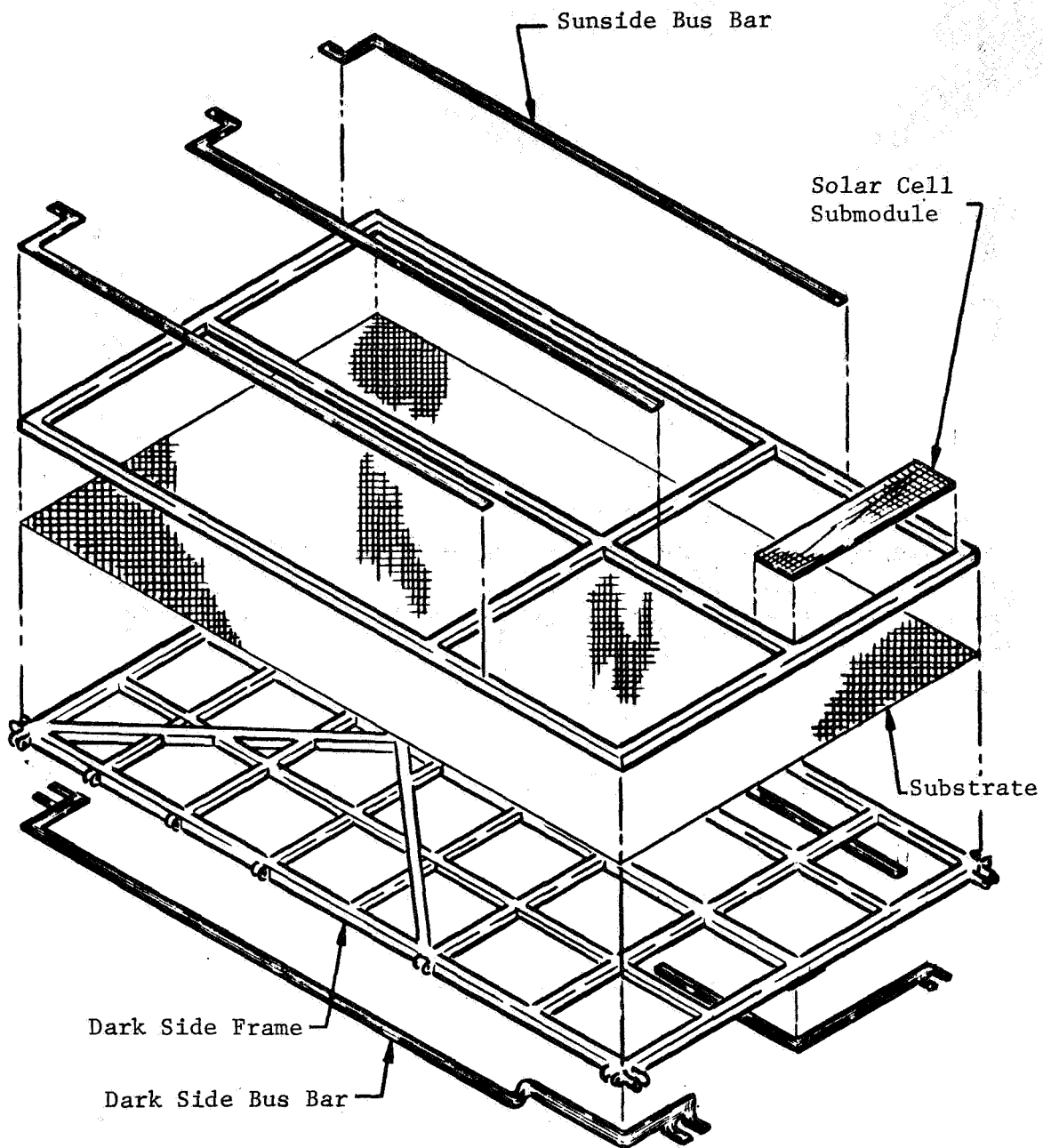
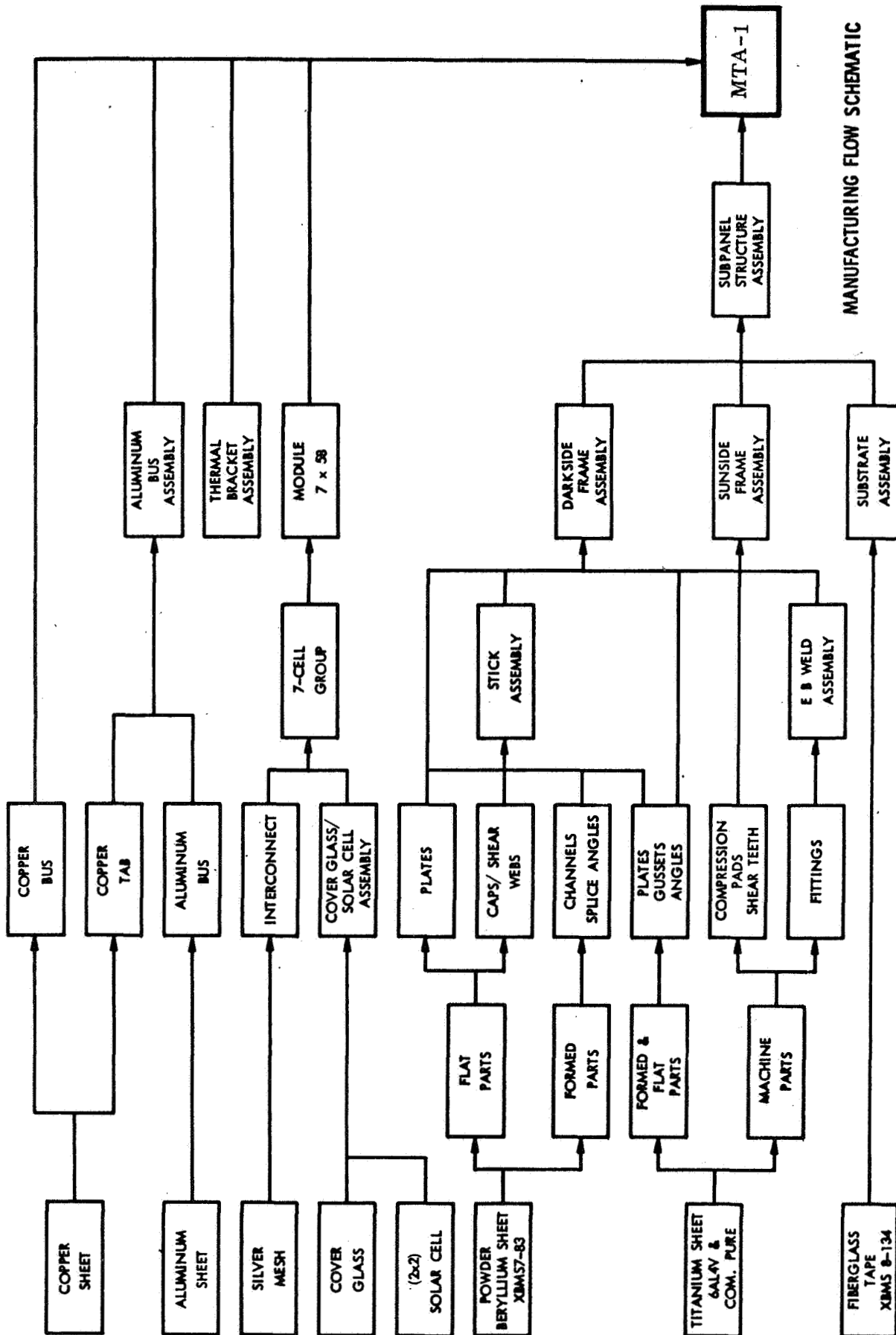


Figure 4-7: EXPLODED VIEW, MTA-1

stock and flat-strip stock are chem-milled to remove surface stains and to detect flaws. After chem-milling, beryllium components are cut to length, identified to drawing number, penetrant-inspected, and then cleaned and primed ready for bonding into tubular sections commonly referred to as "sticks."



MANUFACTURING FLOW SCHEMATIC

Figure 4-8: MANUFACTURING PLAN SCHEMATIC

The stick assemblies are made by bonding cap strips and shear webs to channels over teflon mandrels with AF-126 adhesive using pneumatic pressure at 17 to 20 psi and 250°F electric blanket heat.

The sticks are bonded in two bond cycles into a Sunside frame and a dark-side frame. Titanium gussets, clips, and EB welded fittings make the joints between sticks. The Sunside frame has shear teeth added in a third bond cycle. The frame assemblies are bonded at 225 to 250°F to a previously fabricated substrate assembly that has been manually laid up from fiberglass tape and cured at 350°F for 4 hours while held under 10 inches of mercury vacuum. During substrate-to-frame bond, the substrate is held under sufficient tension to produce 12-lb/in. tension in the tape at ambient temperature.

Solar-cell submodules are assembled by bonding coverglasses to cells, pulse-soldering cells to silver-mesh interconnectors in groups of seven, and then pulse-soldering 58 groups into a 7 by 58 submodule ready for bonding to substrate.

MTA-1 final assembly was accomplished by positioning the structural frame assembly Sunside up and room temperature bonding the solar-cell modules to the substrate under 1.0-psi vacuum. Final assembly is concluded by fitting and room temperature bonding bus bars to the frame under contact pressure followed by interconnect of the solar-cell submodules.

4.2.2 FABRICATION OF DETAILS

Achieving detail fabrication to LASA design requirements was instrumental in successful lightweight structures bonding.

The MTA-1 detail fabrication required manufacturing techniques to produce: formed beryllium channels; flat beryllium caps, shear webs and splice plates; flat titanium gussets and splice plates, and formed titanium angles; titanium weldment fittings; and machined titanium compression pads and shear teeth. Due to design requirements inherent with lightweight bonded structures, unusually tight dimensional tolerances had to be met to control adhesive and beryllium prestressing resulting from parts fitup. For example, beryllium channels were formed to a flatness tolerance of 0.002 bow in 10 inches maximum with channel leg perpendicularity to be within 0.005. In addition, tolerances of ± 0.010 were required to control weight buildup on titanium components normally considered to be sheet metal parts built to ± 0.03 .

Parts fitup for bonded assembly and assembly of MTA-1 at the designed weight were dependent upon meeting detail design tolerances. The manufacturing techniques employed to fabricate MTA-1 details within the requirements are described below.

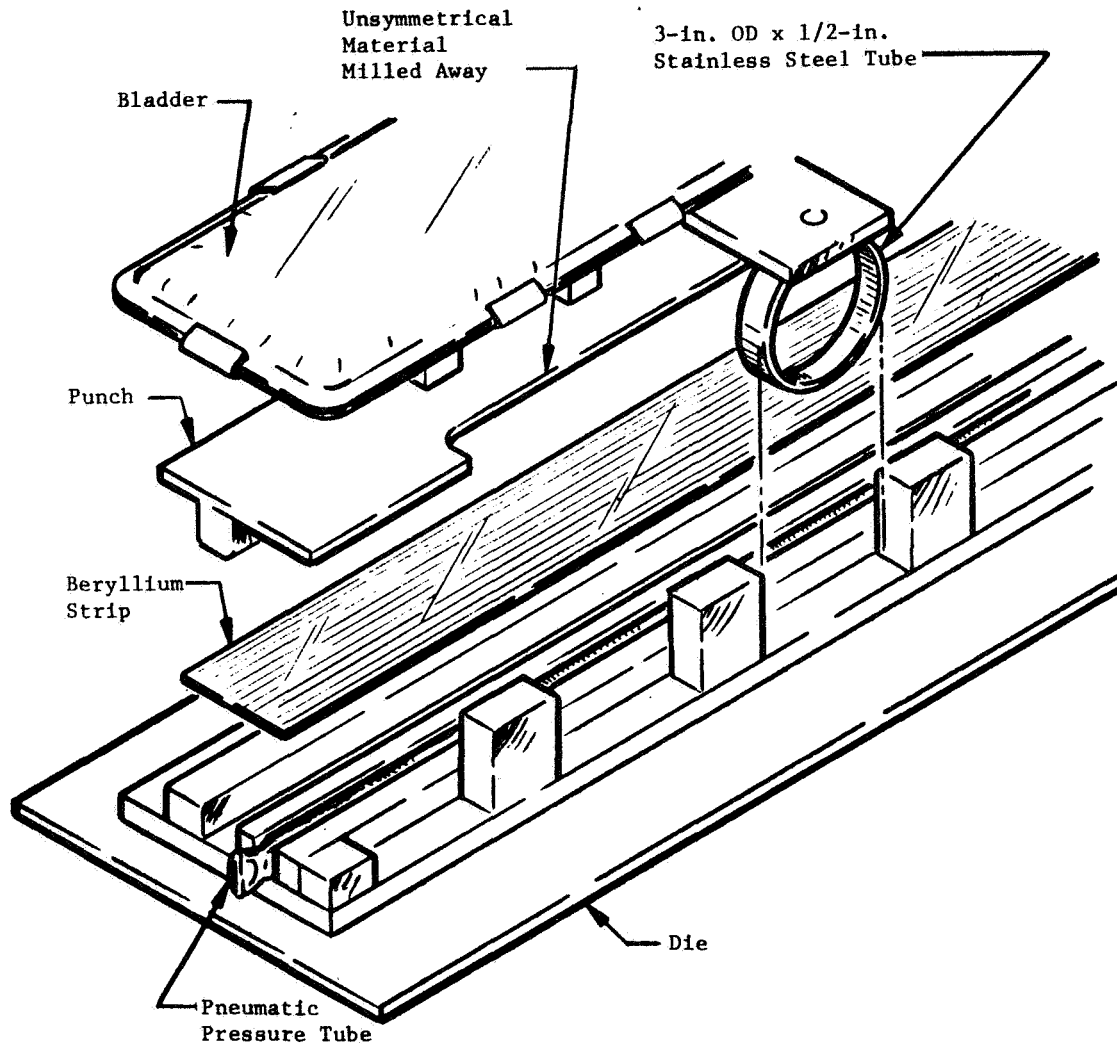


Figure 4-9: HOT-FORM BERYLLIUM CHANNEL

4.2.2.1 Beryllium Channel Forming

Successful beryllium channel forming verifies that flatness tolerances required for bonding can be achieved.

Formed beryllium channels were fabricated within the flatness, perpendicularity, and linear tolerance requirements. The creep form method was used. Cross-rolled beryllium blanks were hot formed at $1,375 \pm 25^\circ\text{F}$ in stainless dies using pneumatic pressure to bottom the die and size channel cross section. The features of the die are shown in Figure 4-9. Die loading was accomplished on a bench followed by hot-press loading as shown in Figure 4-10. Upon satisfying visual and dimensional inspection for the

surface condition and straightness, channels were fixtured, as shown in Figure 4-11, and edge trimmed. Channels were shipped in stock lengths to the Chemical Energy Co. where chem-milling was accomplished. Following chem-mill, channels were net length trimmed and penetrant-inspected at Boeing. A second shipment was made to the Chemical Energy Co. for etch-clean and EC2320 priming to complete channel processing before bonding.

Problems encountered in channel fabrication were in punch support, punch symmetry, and forming lubricant. The punch support blocks used in Phase I, which carried the punch weight before reaching forming temperature, were not satisfactory because of punch and blank shifting on part removal. Four 1-inch sections of 3-inch-diameter steel tubing, shown in Figure 4-9, were ultimately used to support the punch weight; the rings deformed uniformly during forming, holding the punch level as it entered the die.

Unsymmetrical punch design, which allowed a common die to be used with several punches, was found to influence part straightness. By milling the unsymmetrical material away, a symmetrical die was produced as shown in Figure 4-9. The condition was alleviated except in local areas required for guides. Boron nitride was found to be superior to talc as a forming lubricant. Upon resolving these problems, parts were formed consistently within the 0.002 bow in 10 inches required by engineering drawings.

Phase II channel forming experience, averaging over 4 hours per part, indicated that forming rate problems should be anticipated when using present methods in large quantity production. In addition, the two separate shipments required to accomplish chem-mill and etch-clean and prime created logistic problems affecting assembly shop performance.

MTA-1 channel detail requirements were satisfied with following excess factors: raw material procurement excess, 26%; fabrication excess covering scrapage, trimming residue, and backup, 14%. Both excess factors are expressed in terms of blanks purchased and formed against net forming requirements.

4.2.2.2 Flat Beryllium Fabrication

Beryllium sheet stock procurement to net widths minimized manufacturing costs by eliminating edge trim.

Phase II beryllium sheet stock for flat parts fabrication (caps, shear webs, and splice plates) was purchased to net width with an 0.006 allowance for chem-milling. As stock, flat strips were chem-milled to the net finish thickness and width dimension. Chem-milling and blanking irregularly shaped flat parts were accomplished at the Chemical Energy Co. Upon return to Boeing, shear webs, caps, and splice plates were abrasive-sawed to length. Dimensional and penetrant inspection followed. Both channels and flat parts again were shipped to the Chemical Energy Co. for etch-clean and EC2320 prime followed by protective bagging.

Savings in flat parts cutting were realized in Phase II by purchasing to net width. MTA-1 flat parts fabrication was accomplished in 498 flat parts

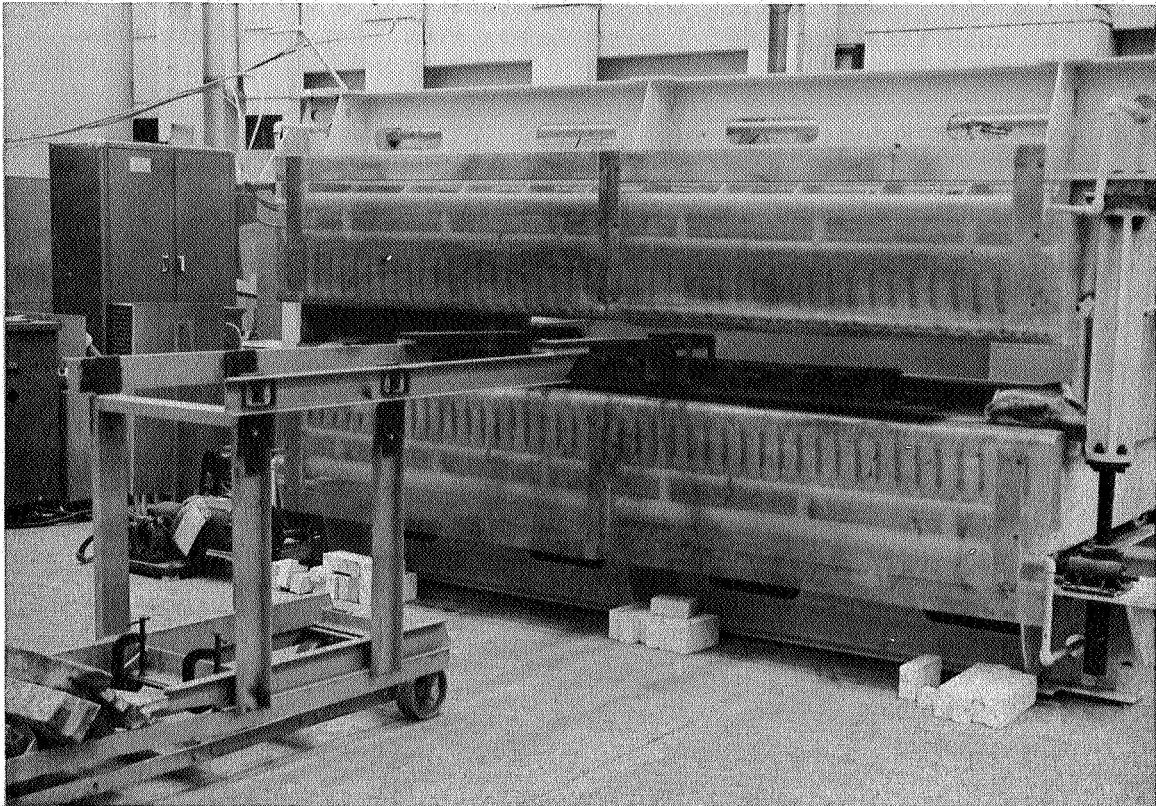


Figure 4-10: HOT-PRESS LOADING OF CHANNEL DIE

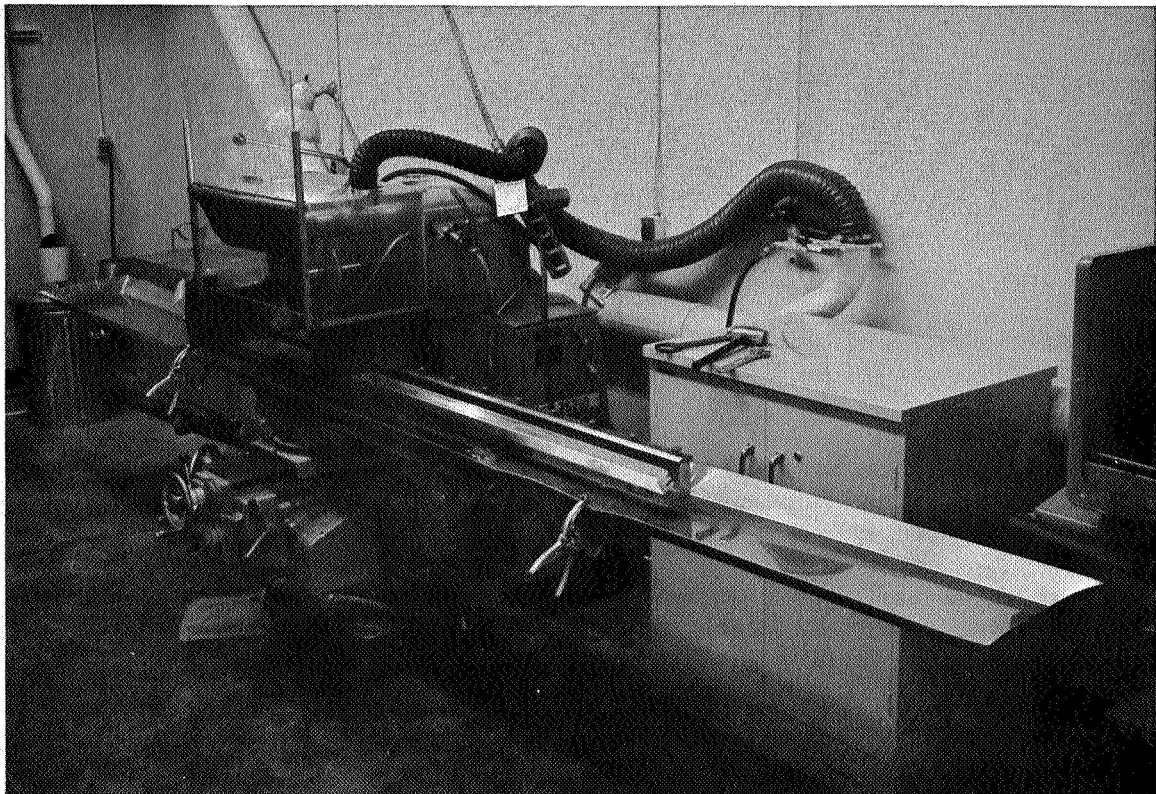


Figure 4-11: CHANNEL TRIM FIXTURE

cuts totaling 447 linear cutting inches as opposed to a minimum requirement for 747 cuts totaling 5,785 linear inches when flat parts are edge trimmed.

4.2.2.3 Titanium Fabrication

Manufacturing techniques developed for thin-walled weldment fabrication allowed full-power cutting, resulting in reduced machine costs.

MTA-1 titanium fabrication required manufacturing techniques for flat sheet metal gussets and splice plates, formed sheet metal angles, machined compression pads and shear teeth, and EB weld of machined titanium fittings. Gussets, splice plates, compression pads, and shear teeth blanks were machined using standard techniques. Normal sheet metal fabrication practices were not employed on gussets and splice plates due to the ± 0.010 dimensional tolerances specified to control weight buildup and to maintain part flatness. Shear teeth were crush-ground by the Roselle Tool and Die Co.

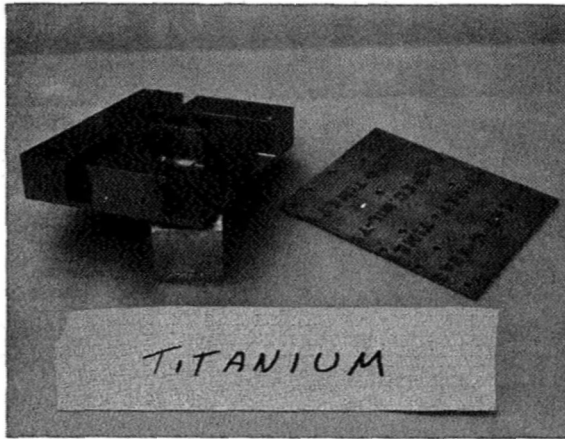
Titanium EB-welded fitting fabrication used a special machining process designed to avoid the Phase I experience of thin-wall distortion during machining. Rather than finish the fitting and cover plate completely before EB welding, the raw stock for both fitting and cover plate was jig-bored at intersections of the outside trim lines to provide indexing points for welding and machining. The internal cuts were then made as shown in Figure 4-12 (upper left). The EB welding and stress relieving were then accomplished and the outside finish machined. This sequence allowed full-power cutting of internal hog-outs and gave maximum support for finish cuts on the outside. While fitting fabrication costs were improved, tolerance buildups in the approach resulted in wall thicknesses at the high end of the tolerance range resulting in a weight penalty.

4.2.3 STRUCTURE SUBASSEMBLY BONDING

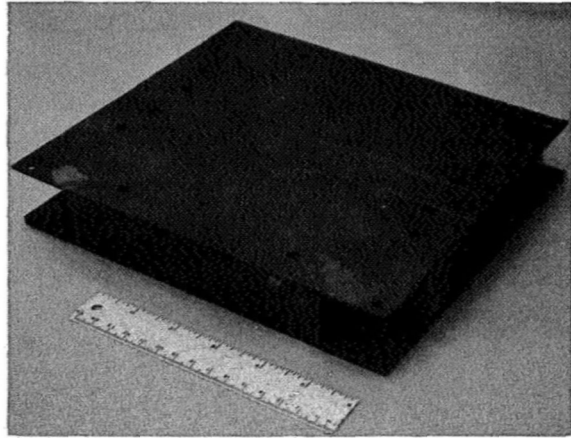
Manufacturing processes and tooling for substrate, stick, and frame bonding were successfully demonstrated in full-size production.

Structures subassembly bonding for MTA-1 consisted of assembly and bonding of sticks, assembly and bonding of Sunside and darkside frames, and fiberglass tape layup for substrate bond.

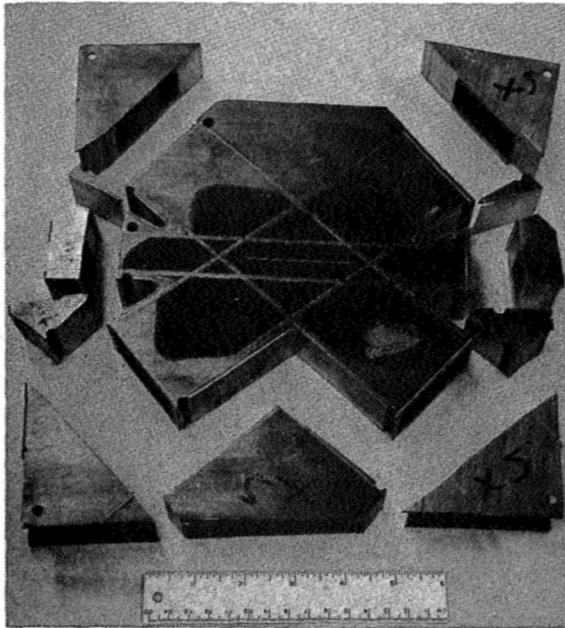
Fifty-eight stick assemblies were made by bonding cap strips and shear webs to channels over teflon mandrels (Figure 4-13). The 17- to 100-psi bonding pressure was applied pneumatically. The 5-mil AF-126 film adhesive was cured at 225 to 250°F for 1 hour using electric blanket heat. Part fitup problems were not encountered except in the case of one bowed channel that was shimmed to prevent channel straightening in the stick tool and resultant prestressing of the part.



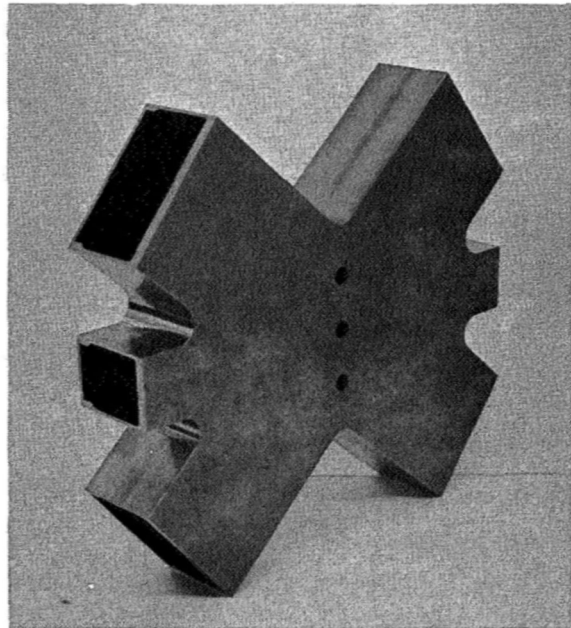
Jig-Bore
Hog-Out



EB Weld
Stress Relieve



Surface Grind
Rough Saw



Finish Machine

Figure 4-12: WELDED TITANIUM FITTING FABRICATION PROCESS

Sunside and darkside frames were assembled by joining sticks and EB fittings with gussets, splice plates, clip angles, and compression pads in two bond cycles as shown in Figure 4-14. On the Sunside a third cycle was used to bond shear teeth. Both pneumatic bladders and mechanical spring pressures were used to apply the 17- to 100-psi bonding pressure through metal plates and silicone rubber pads. AF-126 film adhesive was cured for 1 hour at 225 to 250°F. Electric blanket heat was used.

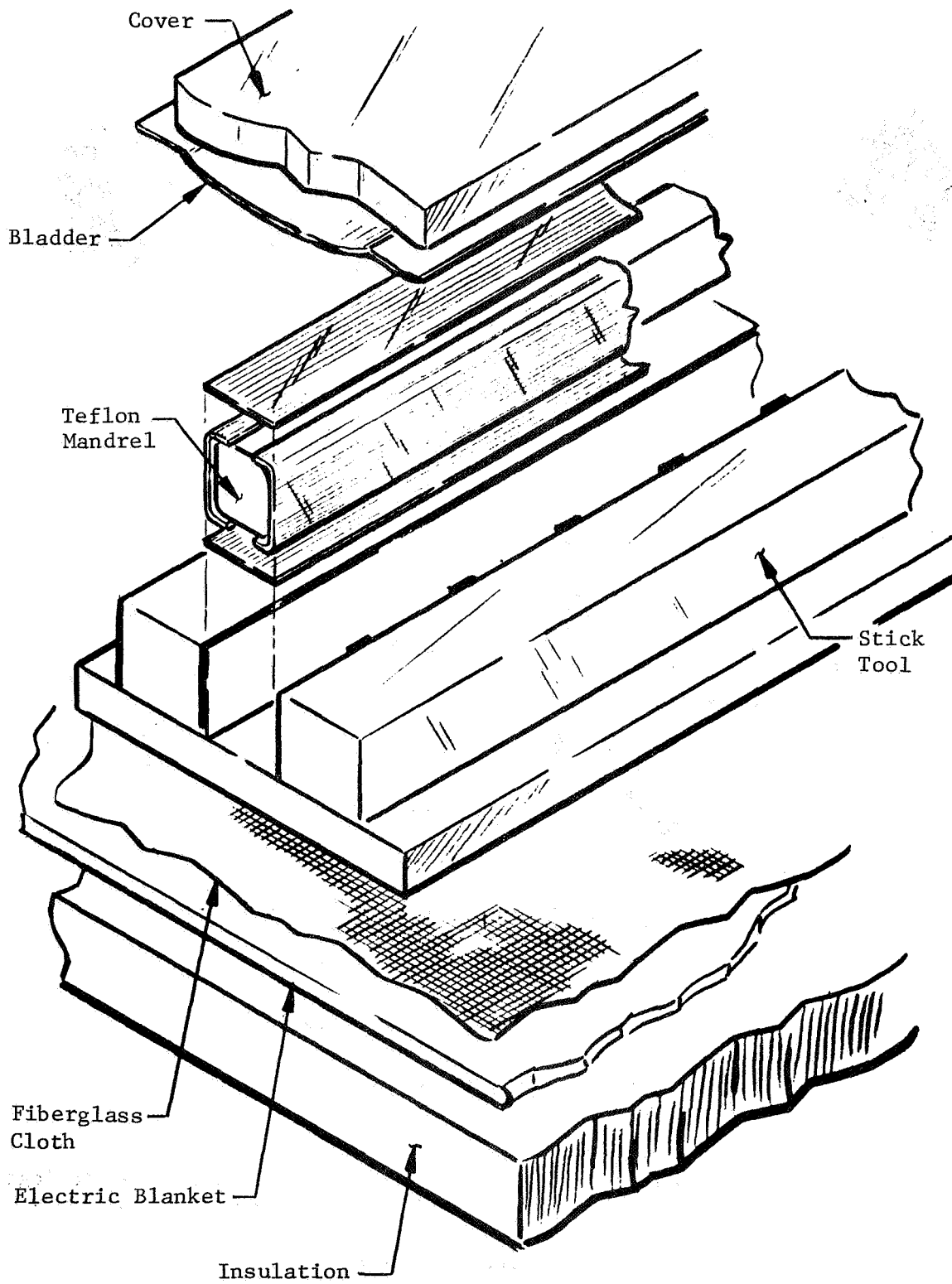
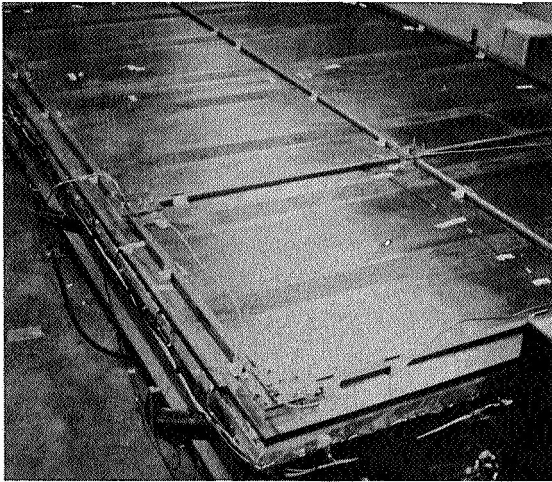
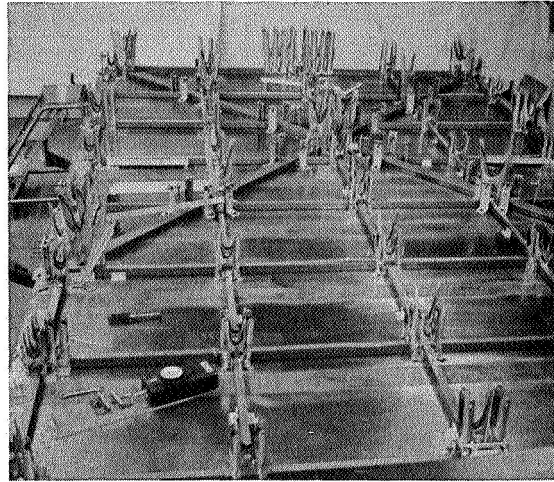
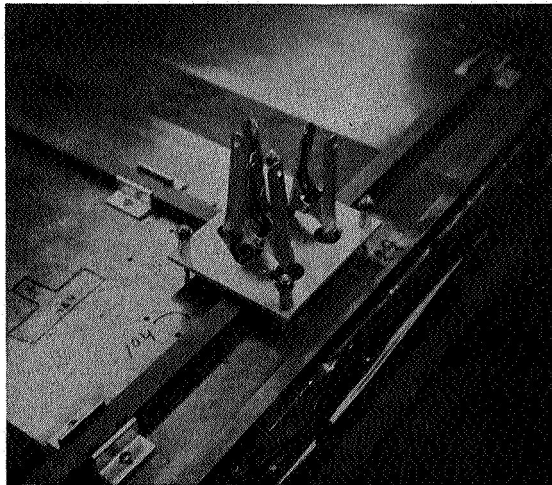


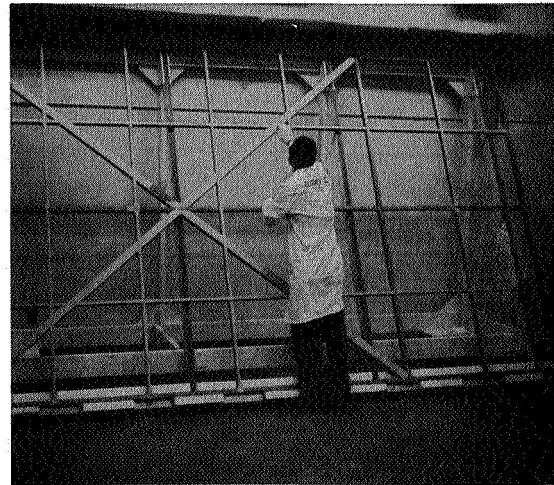
Figure 4-13: STICK ASSEMBLY



Sunside Gusset Bond

Darkside Splice Plate/
Compression Pad Bond

Sunside Shear Teeth

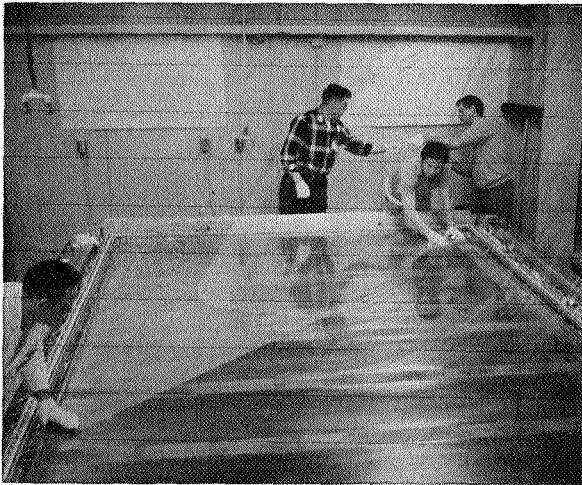


Darkside Frame Assembly

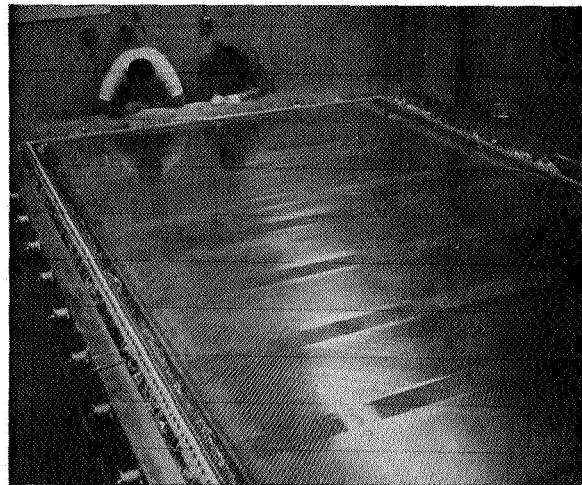
Figure 4-14: FRAME ASSEMBLY PROCESS

Substrate layup, shown in Figure 4-15, was made with shrink scale tooling to compensate for tape growth during pretensioning. Layup was over a teflon parting film on a steel platen. The final substrate bond cure was accomplished at 350°F for 4 hours. A vacuum pressure of 10 inches of mercury was maintained throughout.

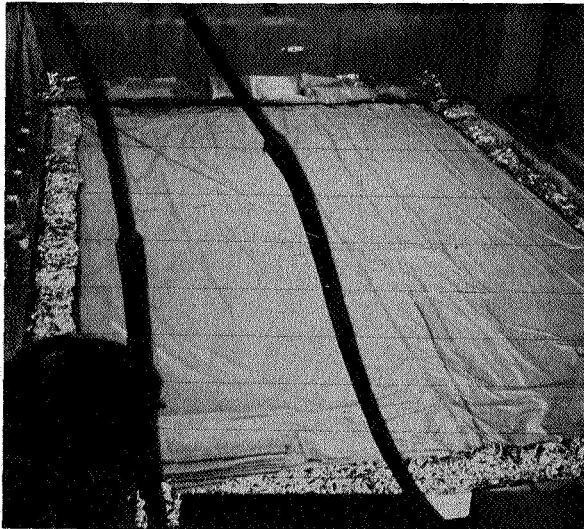
During the MTA-1 cure cycle, the prescribed 0.75-lb/in. draw-bar tension was not maintained because the friction between the steel platen and the teflon parting film, between the tape and the platen, was sufficient to load the long tapes to a higher tension as the platen expanded with heat. This caused the long tapes to be slack at the tension bars. The spacing



Lay Tape



Check--Straighten Tape



Lay In Second Layer



Cure

Figure 4-15: SUBSTRATE ASSEMBLY

established during layup was maintained, however, and after cool-down the long tapes were again under tension. The test of the assembly at 12 lb/in. showed that the tape growth has been correctly computed. The tape intersections were within 0.06 on the 96-inch dimension. Diagonal dimensions were within ± 0.12 . The diagonal dimensions were improved by adjusting the fixed tension bars at the time of final frame bonding.

Variation in spacing between tapes exceeds drawing tolerance of ± 0.01 . Variations up to ± 0.10 occurred. Currently envisioned modifications to existing tooling will improve spacing variation; however, drawing tolerances also may have to be relaxed.

4.2.4 FABRICATION OF SIMULATED POWER SYSTEM

Simulated power system fabrication included demonstrations of manufacturing processes required for live power system fabrication.

The simulated power system fabrication consisted of three independent processes as follows:

- 1) The assembly of live solar cells into modules of 7 cells in parallel and 58 cells in series. A total of six modules was assembled. A complete module is shown in Figure 4-16.
- 2) The assembly of microsheet glass, with dimensions equal to the size and weight of an actual solar-cell stack, into simulated solar-cell modules. The unit of assembly was a submodule of 7 by 29 glasses. A total of 100 of the submodule units was required.
- 3) The fabrication of simulated electrical buses consisting of two layers of aluminum without the dielectric and single-layer copper bus bars equivalent to the actual double-layer laminated construction. Two aluminum and four copper bus bars were fabricated.

The live solar-cell modules were assembled in three steps. First the coverglasses were bonded to the cells and inspected, then the cell assemblies were soldered to interconnectors to make 7-cell parallel groups, and in the third step the groups were assembled into the series string module configuration and inspected. The assemblies at this stage may be installed into 28- or 100-volt modules.

The coverglass bonding was accomplished by prelocating the glasses and the cells in a tool, applying low-tack plastic tape and transferring the cells 10 at a time to the bonding fixture. The required quantity of RTV-602 adhesive was applied to the cells with a hypodermic syringe and the tape containing the glasses was positioned on the cells. The assembly was then passed between mechanized rollers, shown in Figure 4-17, to squeeze out excess adhesive and establish the specified 1-mil bondline thickness. Mislocation and void problems were encountered and were reduced to a minimum by allowing the adhesive to flow to the bondline edges before forcing the bondline thickness and by operator inspection and correction before the roller operation. The assemblies were thoroughly cleaned with MEK and alcohol after full adhesive cure.

Assembly of the 7-cell parallel groups was accomplished by positioning cell assemblies face down in a soldering fixture that locates the N contact bar end of the cell. Cell spacing was established with 5-mil spacers between cells. Paste solder was applied to the P side of all seven cells in the required spot configuration with a stencil. The silver-mesh interconnector was placed on the cells and held in location. The solder joints were completed by pulse soldering each spot. The setup is shown in Figure 4-18. The assemblies were immediately cleaned with MEK and alcohol and transferred to individual plastic boxes.

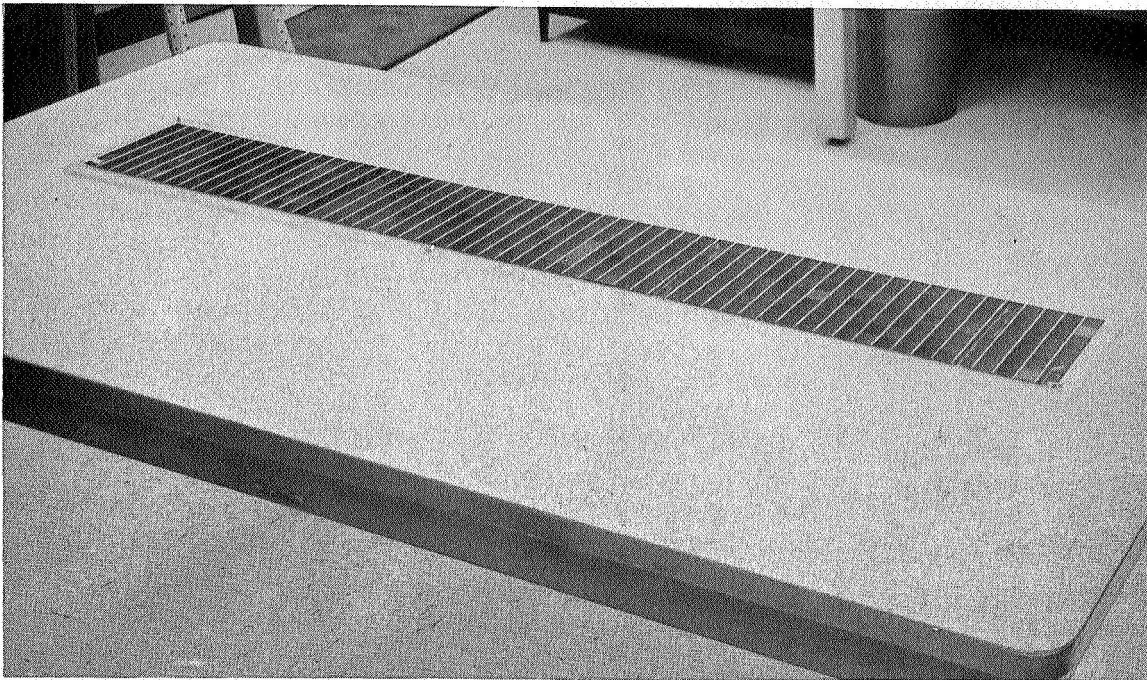


Figure 4-16: SOLAR CELL MODULE

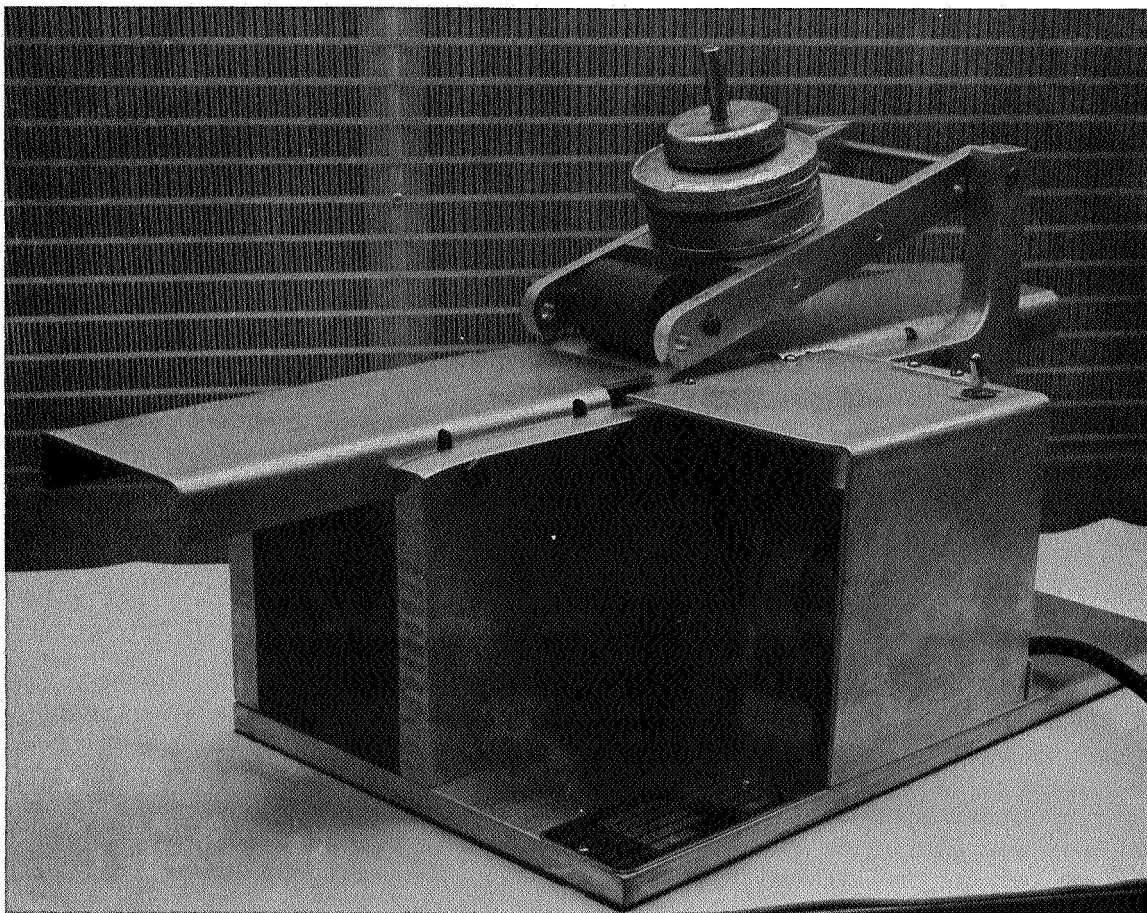


Figure 4-17: SOLAR CELL BONDING ROLLER

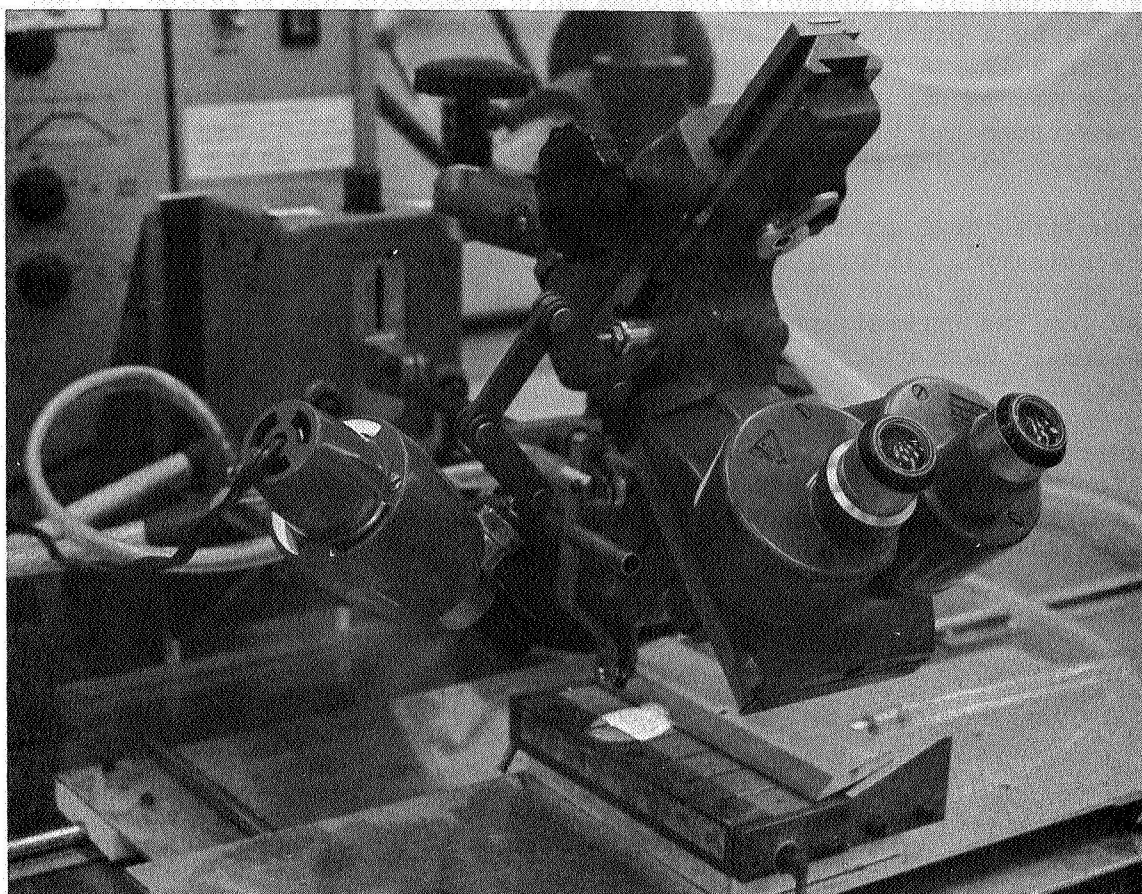


Figure 4-18: SEVEN-CELL SOLDERING FIXTURE



Figure 4-19: MODULE SOLDERING

Series strings of 58 solar-cell groups were assembled and soldered (N contact bar) by a process identical to the group soldering. The setup is shown in Figure 4-19. A problem encountered at this stage was the difficulty in removing the module from the fixture even though one side rail was removable. This was corrected by lifting each row of cells from the surface of the tool after soldering and then reseating on the locators.

The simulated cell modules were assembled by locating the glasses in the module soldering fixture, shown in Figure 4-19, and applying adhesive-faced protecting paper to maintain position, facilitate handling, provide a mask for the thermal control coat application, and maintain outside dimensions. Half modules of 7 by 29 were assembled to simplify handling and installation.

Simulated electrical buses were fabricated by conventional sheet metal fabrication methods, then trimmed and fitted to conform to the actual structural configuration. Copper interconnect tabs to the live cell modules were ultrasonically bonded to the aluminum bars with a silver-foil interface, and soldered to the copper bars by conventional hand-soldering methods.

4.2.5 STRUCTURE FINAL ASSEMBLY BONDING

Temperature and pressure controls are critical considerations in tool designs for hot bonding of lightweight beryllium structures.

Structure final assembly bond is made by joining the Sunside and darkside frames with the pretensioned fiberglass substrate on which the solar-cell modules are later installed. Due to intercostal bond failures in final assembly bonding, two phases in MTA-1 structures final assembly bonding were required. This discussion will report the initial phase leading up to failure. The repair phase is reported separately in Section 4.2.6.

The structure final assembly bond setups for MTA-1 were similar to the vacuum-bag concept successfully demonstrated on the 4- by 4-foot panel (SCS-43) fabrication of Phase I. The Sunside frame was indexed to the bonding platen with the Sunside down and the frame shimmed to clear the shear teeth. Aluminum filler plates supported with aluminum spacer blocks were set within the frame for substrate and intercostal support. The typical intercostal setup is shown in Figure 4-20. A steel filler plate was sealed about the Sunside frame for a vacuum seal. Following AF-126 film adhesive application to the Sunside frame, the previously abraded substrate and its clamping bars were positioned and hooked up. The darkside frame with adhesive applied was positioned to align and lay with the Sunside frame and substrate. A vacuum bag, Figure 4-21, tucked in to minimize bridging, was placed and sealed and vacuum-pulled to 26 inches of mercury. Electric blanket heat was used for the 1-hour, 225 to 250°F cure cycle while the substrate was tensioned to produce the required 12 lb/in. at ambient.

During MTA-1 final structure assembly, cure-cycle temperatures on the bonding platen were not controlled within the 225 to 250°F requirements. Temperatures as high as 269°F were reached along the center spar cap in

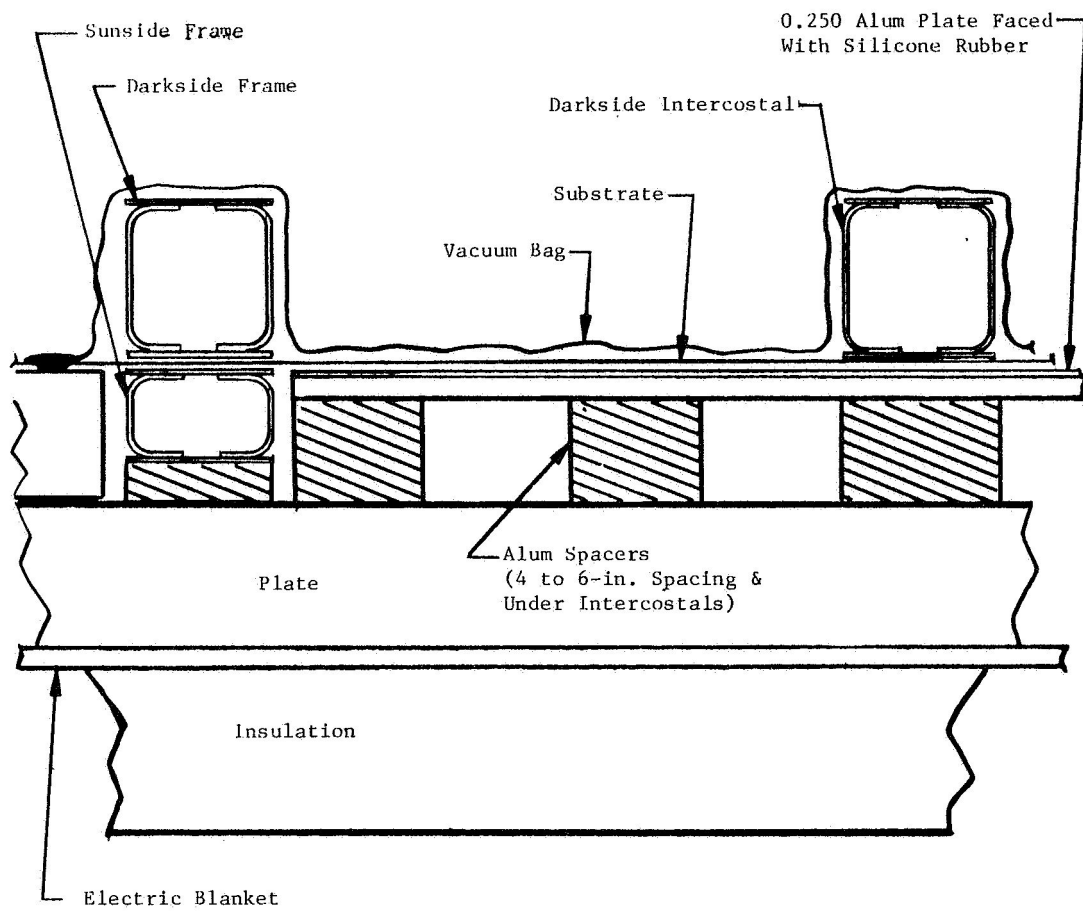


Figure 4-20: FINAL STRUCTURE ASSEMBLY SETUP

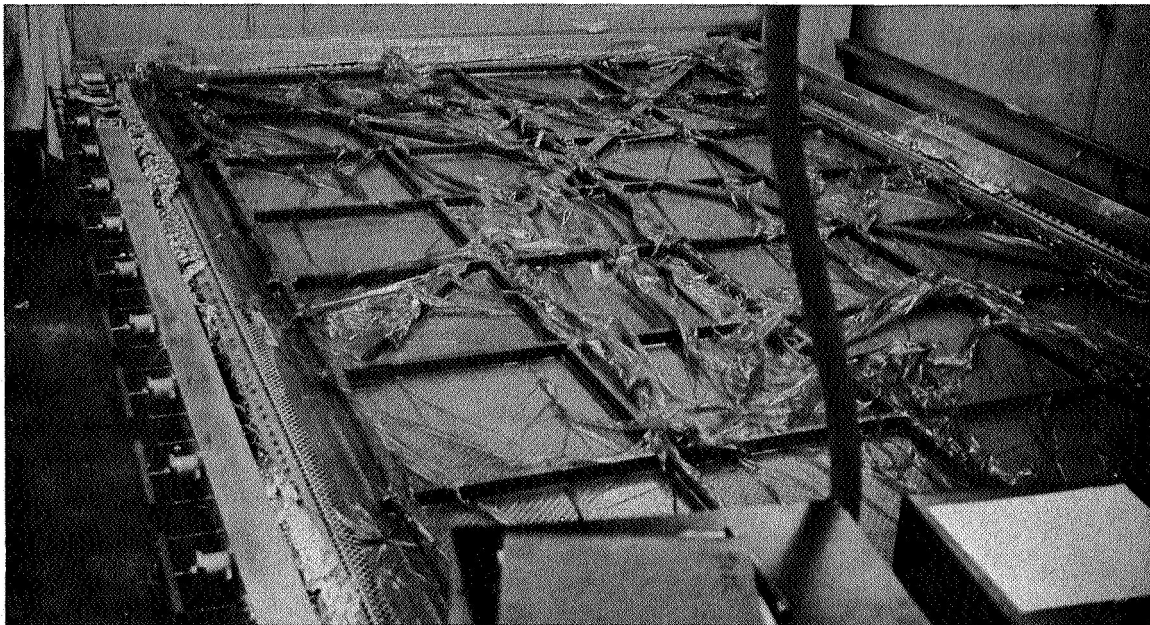


Figure 4-21: STRUCTURE FINAL ASSEMBLY BOND

maintaining the remaining structure above 225°F. Prolonged vacuum-bag side loads, in combination with high temperatures reached, induced channel displacement and bond failures at some intercostals. Initial coupon pull tests had indicated bond strengths above 230 psi at 250°F. This is about four times the vacuum-bag load expected. The actions taken to assess the damage and repair MTA-1 follow in Section 4.2.6.

4.2.6 REPAIR OF MTA-1

Unique repair techniques were developed to successfully repair the MTA-1 structures.

Repair of bond failure damage incurred in MTA-1 final structure assembly bonding required the following actions:

- 1) Complete structural inspection to identify and record the damage incurred;
- 2) Determine the cause(s) of failure;
- 3) Corrective actions to control the cause(s) of failure;
- 4) Repair techniques to accomplish repair;
- 5) Repair.

An MTA-1 Repair Committee was established to oversee the above and coordinate the MTA-1 repair plan development.

Inspection to assess MTA-1 damage identified 19 failure areas. Damaged areas were located by three methods: (1) feeler gage, (2) coin-tap tests, and (3) harmonic testing. Harmonic tests of bondlines (an experimental process under development by the Materials and Processes Laboratory) showed correlation with voids previously identified by coin tapping and feelers. In addition, suspect areas were identified that were not picked up by coin-tap tests and feelers. Collectively, the damage identified ranged from intermittent adhesive failure without apparent parts displacement to parts displacement and bond failure accompanied by intercostal cap fracture. Figure 4-22 identifies and describes the failure areas by location.

Based on findings of environmental testing simulating the MTA-1 conditions at failure, the MTA-1 Repair Committee identified seven causes as contributing to the failure.

- 1) The temperature of the bonds exceeded 250°F.
- 2) The creep strength of the adhesive in peel and shear is low and unpredictable at 250°F (80 psi was obtained).
- 3) Stress concentrations in the adhesive caused by deflection of the channel contributed to the stress in the bond causing the failures.

- 4) Static load-carrying capability is reduced in the adhesive under load and elevated temperatures.
- 5) Thermal gradients between adjacent parts were a factor in the failure. Since thermocouples were not installed on intercostals, full evaluation of gradients was not possible.
- 6) Variations in temperature caused by excessive leaks in the vacuum bag contributed to the failure.
- 7) Fit of parts causing prestressing of the adhesive also was a contributing factor to the failure.

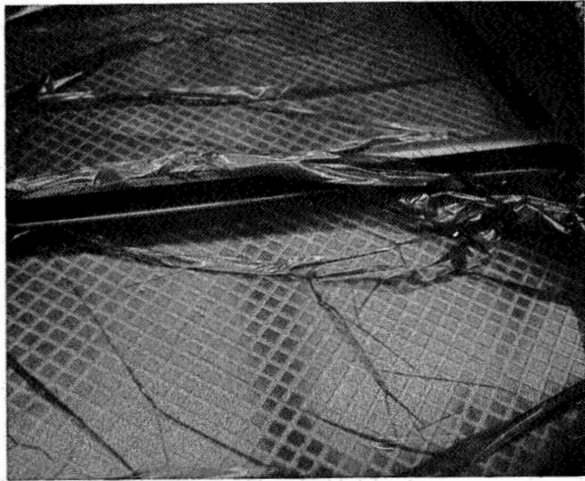
In addition, the Repair Committee recommendations based upon failure causes were made relative to MTA-1 repair:

- 1) Control the temperature of the bonded structure to keep it below 250°F.
- 2) Maintain even temperature of the bonded structure throughout the bonding cycle.
- 3) All bonds shall be loaded normal to the bonding surface with proper support, during cure, so that side and peel loads are eliminated.
- 4) Precaution should be taken in preparing the vacuum bag to reduce leakage.
- 5) Determine and use the minimum temperature range that is satisfactory during the final bonding cycle.
- 6) Check the effect of extreme thermal gradients on the strength of adhesive bonds.

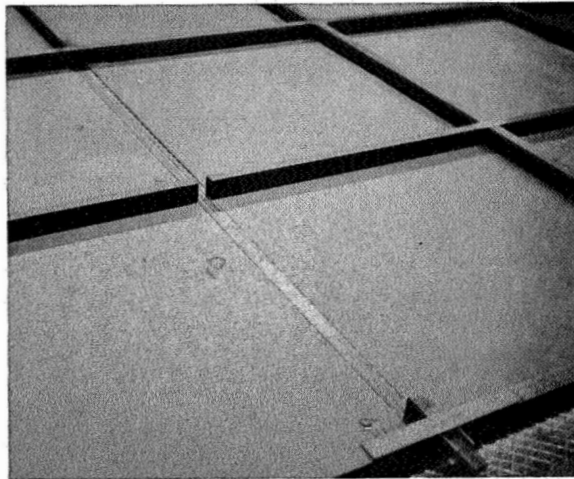
The actions taken to fulfill the above recommendations are tabulated in Table IV-2.

Nine differing conditions required repair on MTA-1. Five of the conditions were common to the failure in the final structures bonding, Section 4.2.5, with four additional conditions developing during the repair bond cycle. The conditions repaired are shown in Figure 4-23.

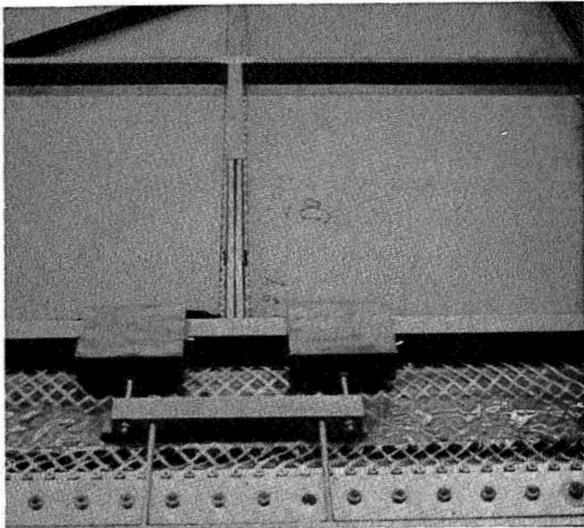
Accomplishing MTA-1 intercostal disassembly and removal required a means to support the substrate loads normally carried through the intercostals. Removal of displaced intercostal channels and caps (Figure 4-24) was initiated following transfer of substrate loads to the substrate tensioning frame through retractor clamps bearing upon the substrate cap or interfacing with the auxiliary hinge fittings, as shown by Figure 4-24. Components were selectively removed, using a high-intensity oxy-hydrogen flame to thermally shock the adhesive while gently prying the removal component. Components were removed without loss due to fracture. As shown in Figure 4-24, component removal and disassembly varied from removal of both channels and cap or removal of single channel and cap, and cutting and removal of fractured cap.



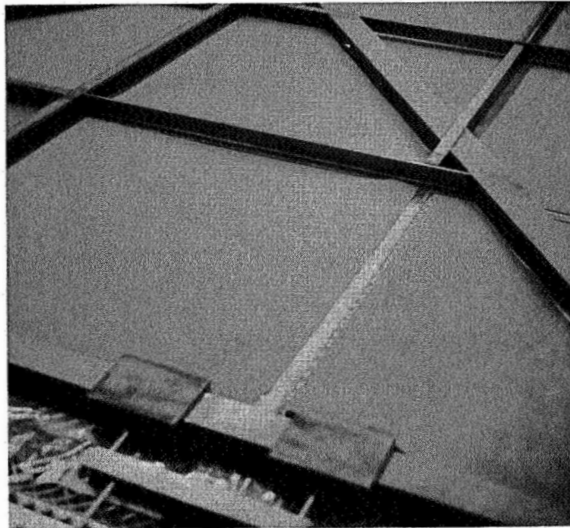
Displaced Intercostal Channel
(Vacuum Bag in Place)



Two Channels and Cap
Removed



Cap Trim and Removal

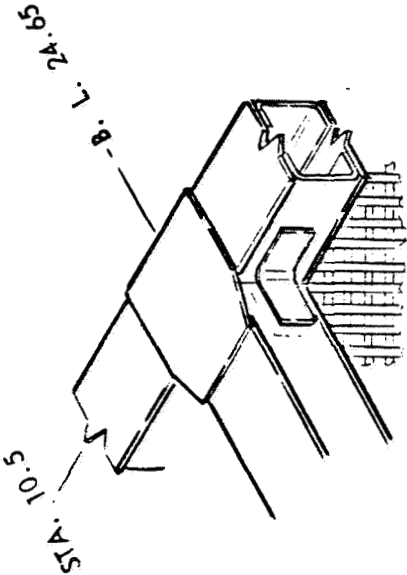


Selective Cap and Channel
Removal

Figure 4-24: REPAIR BOND DISASSEMBLY, MTA-1

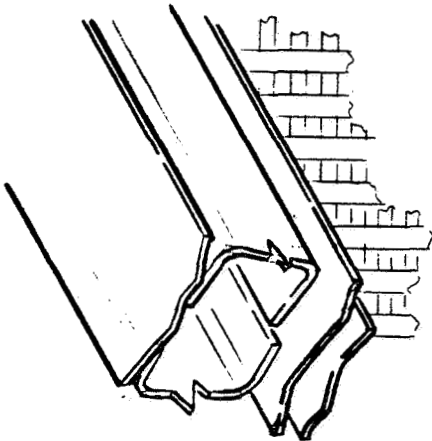
In accomplishing repair of MTA-1, special tests were conducted to evaluate repair techniques. The repair techniques developed and selected for repair of MTA-1 are tabulated by condition in Table IV-3.

During preparation for the repair bond cycle, a bowed condition was identified in the MTA-1 structure. Consequently, provisions were made to straighten the subpanel. Bow removal was a precaution to preclude undue steel protector plate loads upon the substrate.



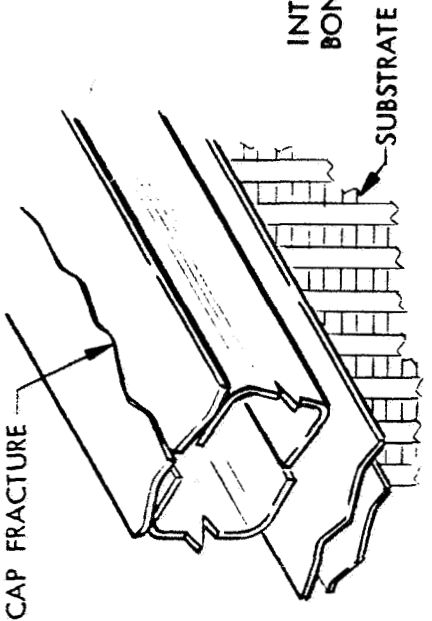
CONDITION 1

CLIP/SPLICE PLATE BOND FAILURE



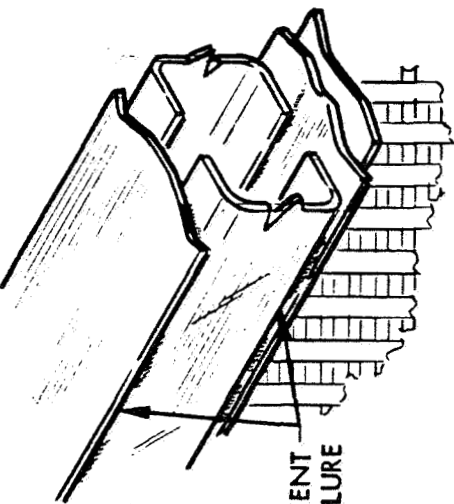
CONDITION 2

CAP BOND FAILURE & CHANNEL DISPLACEMENT
ONE SIDE ONLY



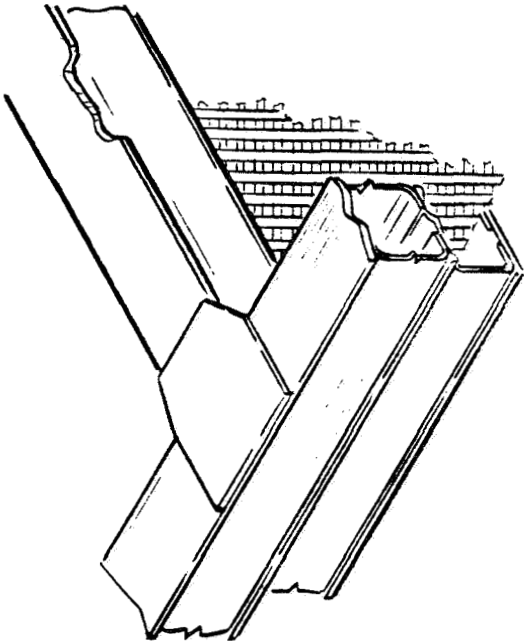
CONDITION 3

CHANNEL BOND FAILURE &
CHANNEL DISPLACEMENT
BOTH SIDES



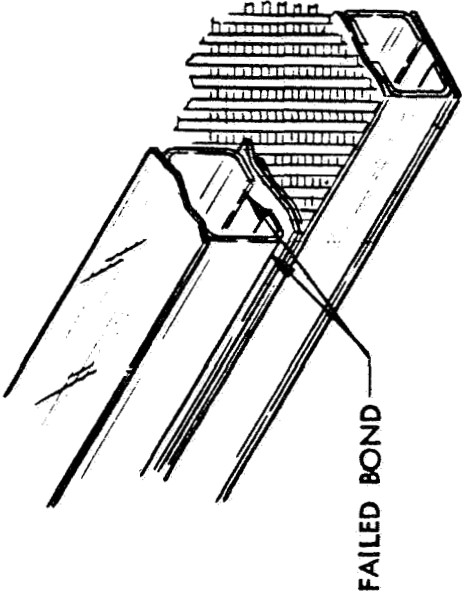
CONDITION 4

INTERMITTENT BOND FAILURE -
NO VISIBLE CHANNEL DISPLACEMENT



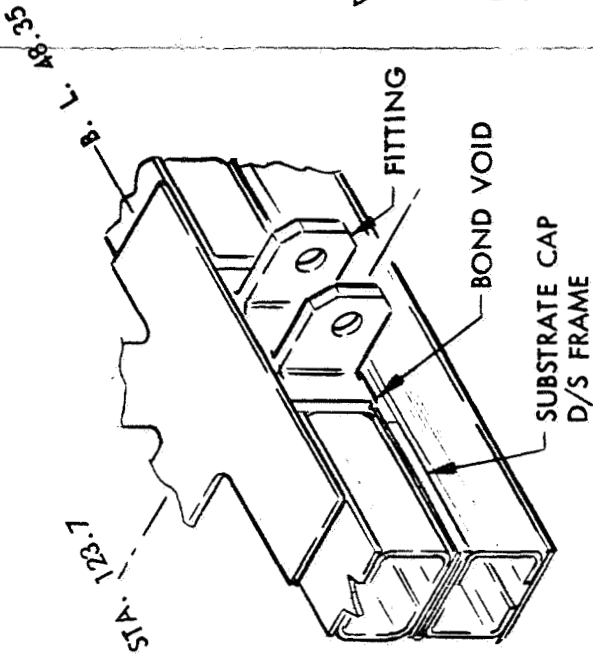
CONDITION 5

CAP FRACTURE



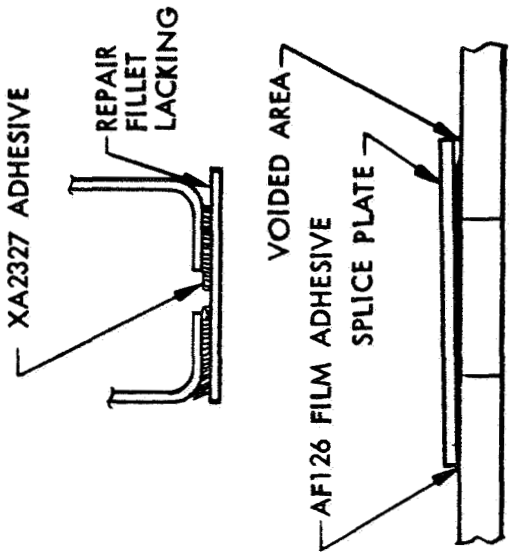
CONDITION 6

BOND FAILURE - UPPER EDGE MEMBER



CONDITION 7

VOID, FITTING TO
SUBSTRATE DARKSIDE



CONDITION 8 & 9

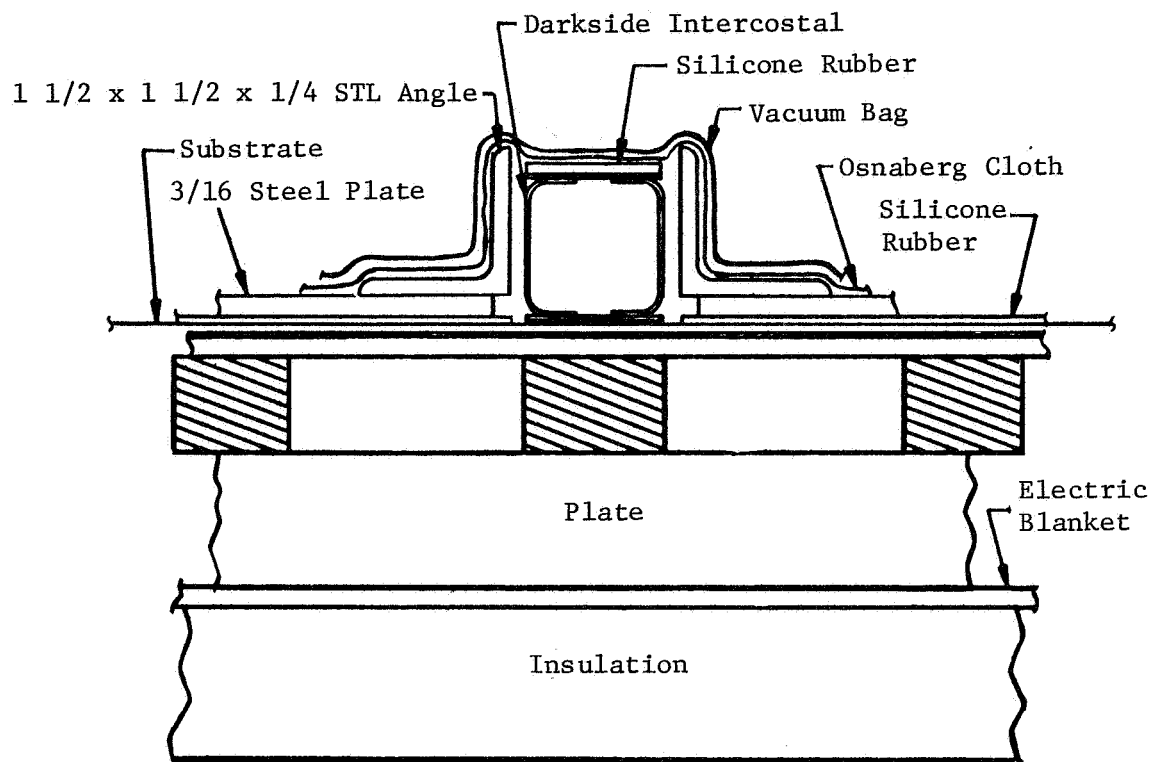
SPLICE PLATE LIFTING/LACK OF FILLET

Figure 4-23: MTA-1 CONDITIONS REPAIRED

Table IV-2: MTA-1 FAILURE COMMITTEE RECOMMENDATIONS

<u>Failure Committee Recommendations</u>	<u>Resultant Actions</u>
All bonds shall be loaded normal to the bonding surface during cure.	Added protector plates (Figure 4-25) with steel angles to carry vacuum-bag side load.
Control the temperature of bonded structure to keep it within 250°F.	a. Added eight auxiliary heating blankets about structure perimeter.
Maintain even temperature of bonded structure throughout the bonding cycle.	b. Reduced heat-up rate in coming up to the 225 to 250°F cure temperature to facilitate finer heat adjustments.
Use the minimum temperature range that is satisfactory during the final bond cycle.	c. Added 11 additional thermocouples, making a total of 27 thermocouples sensing structure and bond platen temperature.
Precaution should be taken in preparing vacuum bag to reduce leakage.	a. Added Osnaberg cloth under vacuum bag. b. Revised vacuum inlet port to be independent of vacuum-bag seal. c. Padded sharp protrusions and faired intersections with wadded aluminum foil.
Check the effects of extreme thermal gradients on strength of adhesive bonds.	(No fabrication impact.)

The repair bond cycle was accomplished on July 12, 1968, following a 6-week preparatory period. The fitted steel protector plates and Osnaberg cloth (Figure 4-26) improved vacuum-bag performance. Leakage was not a problem. Molded rubber pressure pads were used to transfer bonding pressures to clip angles (Figure 4-26) and splice plates. The auxiliary heating blankets (Figure 4-26), additional thermocouples, and the slower heat-up rate were instrumental in maintaining the structure within the specified 225 to 250°F cure temperature requirements. With an exception along the upper edge member, the steel protector plates worked well in



Typical intercoastal cross-section

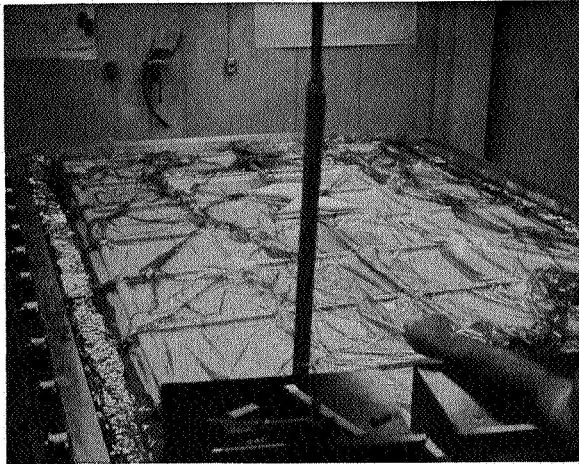
Figure 4-25: REPAIR BOND SETUP

Table IV-3: MTA-1 REPAIR TECHNIQUES

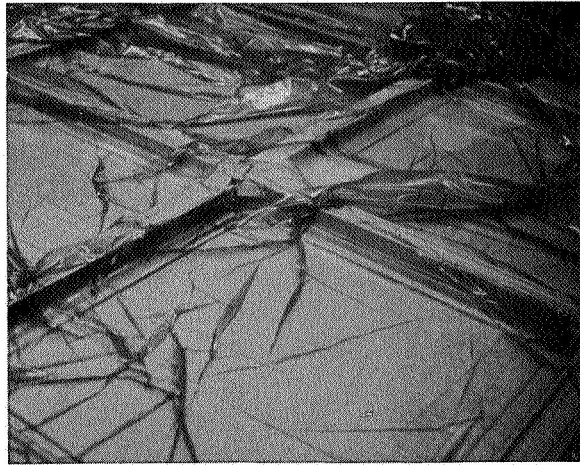
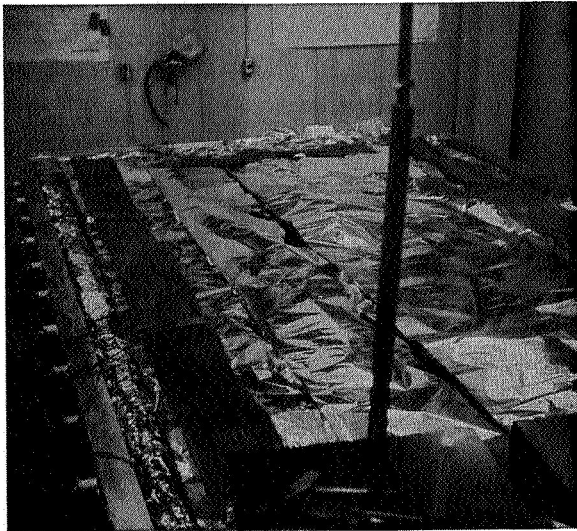
<u>Condition</u>	<u>Repair Technique</u>
1. Splice Plate and Clip Bond Failure	● Warm bondline area and remove clip or splice plate. ● Scrape adhesive off removed component. ● Abrade off primer using Scotch Brite. ● Etch-clean and prime per D2-113354-1. ● Position and bond per D2-113354-1.
2. Cap Bond Failure and Channel Displacement--- One Side Only	● Warm bondline area and remove displaced components. ● Scrape adhesive off removed components. ● Abrade off primer using Scotch Brite. ● Penetrant-inspect components. ● Etch-clean and prime for bond per D2-113354-1. ● Position and bond per D2-113354-1 using channel spacer.

Table IV-3 (cont)

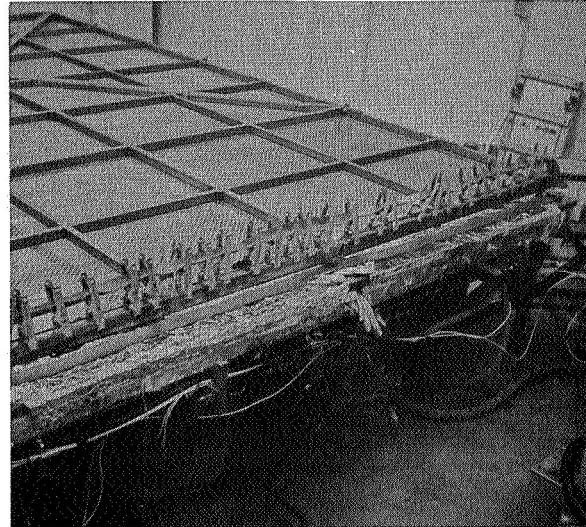
<u>Condition</u>	<u>Repair Technique</u>
3. Cap Bond Failure and Channel Displacement--- Both Sides	● Warm bondline area and remove displaced components. ● Remake fractured caps. ● Scrape adhesive off removed components. ● Abrade off primer using Scotch Brite. ● Penetrant-inspect components. ● Etch-clean and prime for bond per D2-113354-1. ● Subassemble and bond cap and channels per D2-113354-1. ● Position and bond partial stick per D2-113354-1 using channel spacer.
4. Intermittent Bond Failure--- Visible Channel Displacement	● Locate and identify area using nondestructive methods. ● Seal area about open-bond area with Mystic 7503 teflon tape. ● Apply AF-126 liquid adhesive by injecting through teflon tape with No. 18-20 gage hypodermic needle on syringe. ● Assist capillary action by slightly working joint parts. ● Remove tape. ● Bond and cure per D2-113354-1.
5. Cap Fracture	● Scribe trim cap. ● Warm bondline and remove fracture area. ● Fabricate repair cap and splice plate per D2-113354-1. ● Position and bond per D2-113354-1.
6. Bond Failure--- UEM	● Locate and identify failure areas. ● Seal about open-bond areas with Mystic 7503 teflon tape. ● Apply XA 2327 adhesive into voided areas by injecting through teflon tape with No. 18-20 gage hypodermic needle on syringe. ● Remove tape. ● Cure for 5 hours at 200°F under contact pressure.
7. Void Fitting to Substrate Cap--- Darkside	● Locate and identify failed area. ● Fill voided area with BMS 5-29, Type II. ● Cure for 24 hours at room temperature under contact pressure.
8. Splice Plate Lifting and Lack of Fillet	● Fill voided areas with BMS 5-25 adhesive. ● Locally heat and cure for 6 hours at 200°F maximum.



Repair Bond Setup

Protector Plate and
Pressure Pad Detail

Auxiliary Heating Blankets



Edge Member Repair

Figure 4-26: MTA-1 REPAIR

controlling vacuum-bag side loads. Along the upper edge member, an error in load path analysis allowed deflection that induced loadings sufficient to fail the bonds. In addition, vacuum-bag bridging at the dead-weight support plate caused partial voids in splice plate bonds at two locations across the diagonal brace.

The initial repair bond cycle was followed by localized repairs of the upper edge member. Two repairs of the upper edge member were made using Minnesota Mining and Manufacturing XA 2327 adhesive cured for 5 hours at 200°F. Use of XA 2327 was approved for repair following lap shear pull tests that showed minimum ultimate tensile strength of 1,685 psi at 167°F in specimens simulating repairs of failed AF-126 film adhesive. The adhesive was selected to minimize the heating of the remaining structure. Repair of the bond between channels and cap strips was accomplished by

injecting XA 2327 into the failed joint and by locally heating with a strip blanket (Figure 4-26) to 200°F for 5 hours while holding the balance of the structure at about 125°F. The first repair bonded the caps successfully but did not restore tension in the substrate at the upper right-hand intercostal. Subsequently, one gusset and two clips were removed and one gusset loosened at the substrate level. Substrate tension was applied and the clips and gussets were successfully rebonded with XA 2327. The repaired MTA-1 is shown in Figure 4-27. .

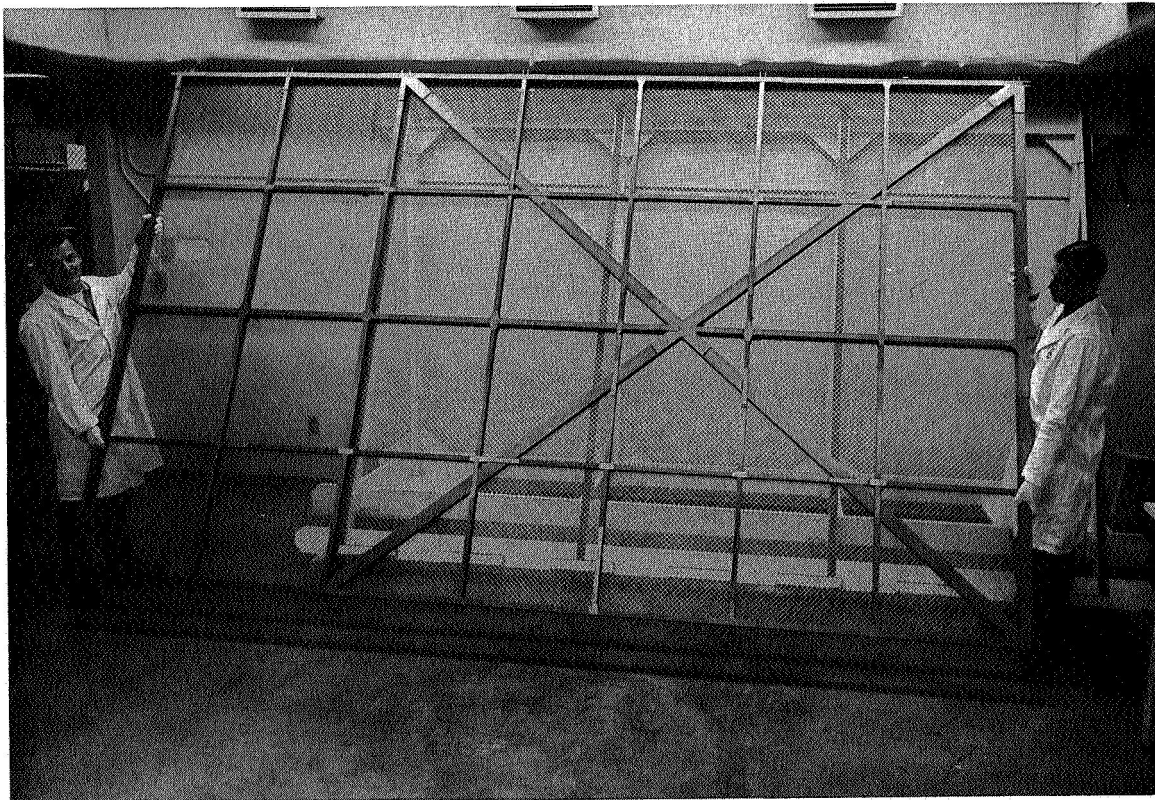


Figure 4-27: REPAIRED MTA-1

The one bond failure later identified between the EB-welded fitting and substrate cap at Station 123.70 was filled with BMS 5-29, Type II, adhesive and room-temperature cured for 24 hours.

Repair of degraded splices was accomplished by injecting BMS 5-25 into the voided areas. Additional BMS 5-25 repairs were made to fill localized areas, primarily along the upper edge member, which had failed to fillet. BMS 5-25 was cured at 120 to 135°F for 6 hours. The BMS 5-25 repairs that were accomplished concurrent with MTA-1 final assembly concluded repair of MTA-1.

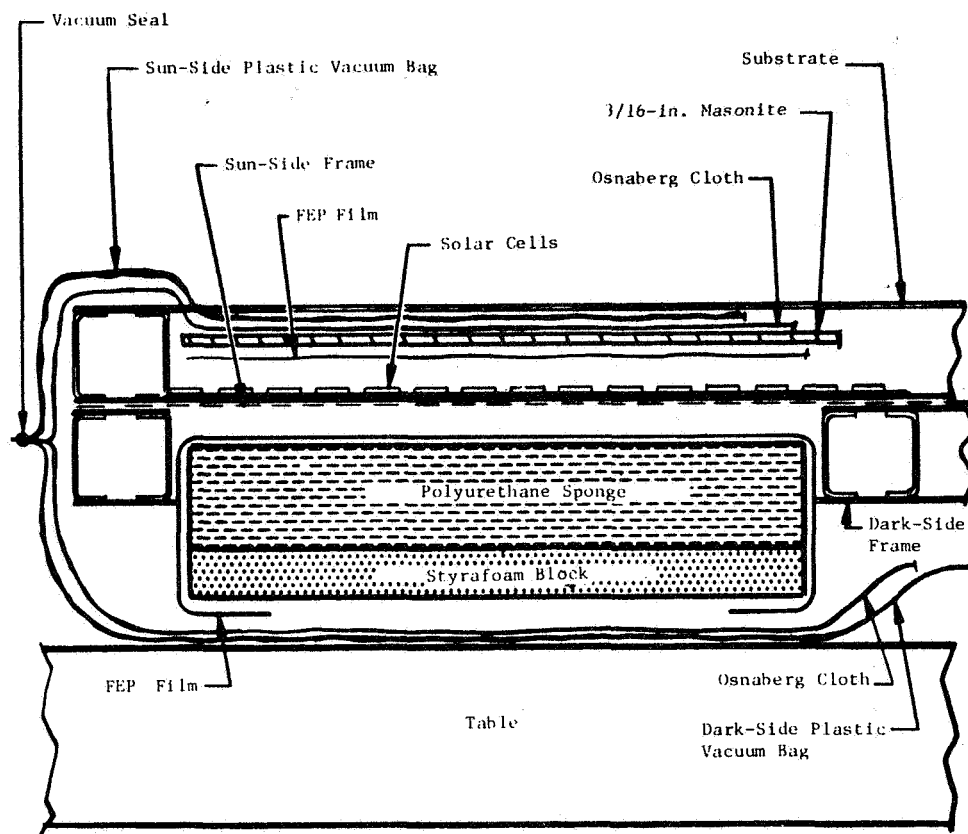
4.2.7 MTA-1 FINAL ASSEMBLY

Installation of the simulated power system was accomplished with simplified tooling.

The final assembly for MTA-1 proceeded after all structure assembly was complete. The assembly was accomplished in three bond cycles. The bonding was cured at room temperature. The first cycle was for bonding solar-cell modules to substrate; the second for Sunside bus bars to spars; and the third for darkside bus bars to spars. After bonding, all of the electrical interconnections were made from buses to live cell modules by soldering.

In preparation for live cell module and simulated module installation, a thermal control coating of RTV-40 was sprayed on the backside of each module to obtain a cured thickness of 2 mils. Cleaning and spray priming with SS-4044 silicone primer preceded the RTV-40 application. The coating was applied in two coats with the modules in a horizontal position to eliminate sags and runs. Cure was at room temperature for a maximum of 24 hours. Low-tack plastic tape had been applied to the Sun surface of the live cell modules to mask the spray, protect cells from damage, and facilitate handling.

Solar-cell modules and simulated modules were manually located and then bonded, using a simplified tooling concept illustrated in Figure 4-28. This concept eliminated the costly filler blocks and module spacers originally proposed. The FEP-film-covered styrofoam and polyurethane-sponge filler blocks were assembled and prefitted into the spaces formed by spars and intercostals on the darkside of the panel, as shown in Figure 4-28. The required thickness of sponge was predetermined by testing to provide an approximately level plane with the outer surface of the structure members after compression by the vacuum bag for bonding pressure. The substrate was cleaned by wiping with MEK, then primed by spraying with SS-4044 primer. The panel was placed on the bonding table, darkside down, and the vacuum bag, Osnaberg cloth, and filler blocks installed. The panel was supported so that a space was maintained between the substrate and the FEP covering on the filler blocks. Catalyzed and deaerated RTV-40 was applied to the substrate with a polyurethane-sponge-covered paint roller. The technique was predetermined so as to apply a coating that would not exceed a weight of 0.15 gram on a 10-cell area. Immediately after completing a suitable working area, the cell modules were positioned on the substrate. Spacers clamped to the inside surface of spars provided structure reference location and shim tools positioned to the edge of previous modules determined the module spacing. Adhesive application and module positioning is shown in Figure 4-29. The module laydown was completed within 2 hours. FEP parting film was laid on the module surface as the work progressed across the panel. Immediately after the last module was positioned, the 3/16-inch-thick masonite sheets were placed on the modules to provide a smooth and rigid pressure surface. Breather cloth and plastic vacuum-bag material was applied and seated.



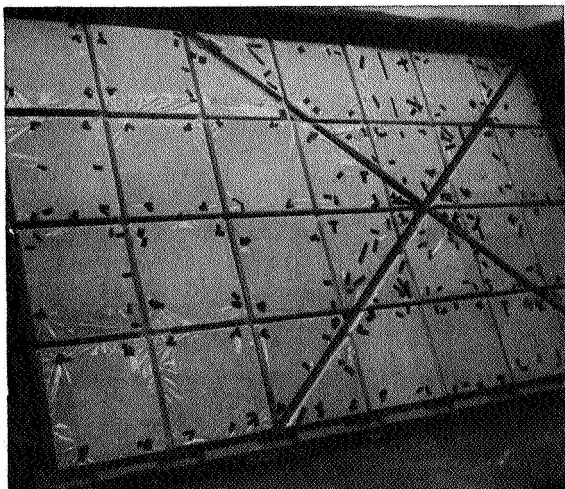
Cross Section of Vacuum-Bagged Assembly

Figure 4-28: MODULE BONDING

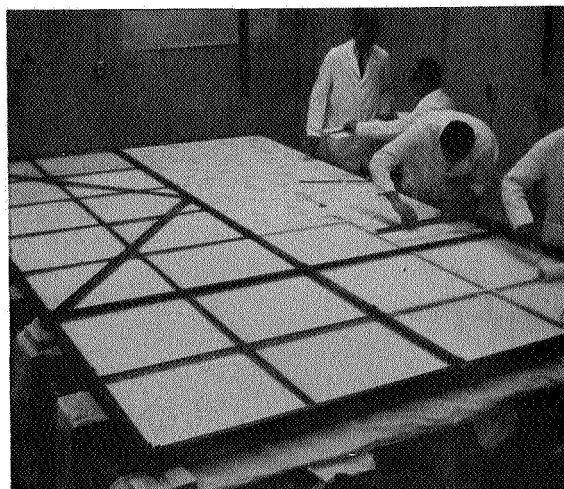
A vacuum was pulled to apply a 1-pound-per-square-inch atmospheric pressure on both sides of the assembly. The applied load was held and monitored for 6 hours, the vacuum released, and the assembly allowed to cure for a total of 48 hours before work proceeded. The vacuum bag, pressure plates, and protective tape were then removed.

While the panel was still in the Sunside-up position on the bonding table, the copper bus bars and brackets for interpanel bus connection were bonded in place. The operation consisted of cleaning the spar surface, brush priming with SS-4155 primer, applying a bead of RTV-630 with a syringe, positioning the bus bars on the adhesive, and applying rigid pressure bars. Pressure was applied by hand until excess adhesive was squeezed from the bondline; then the assembly was taped or clamped to the structure, as shown in Figure 4-29, until the adhesive was cured. The panel was turned over and the operation repeated for the aluminum bus bars and their brackets on the darkside of the panel.

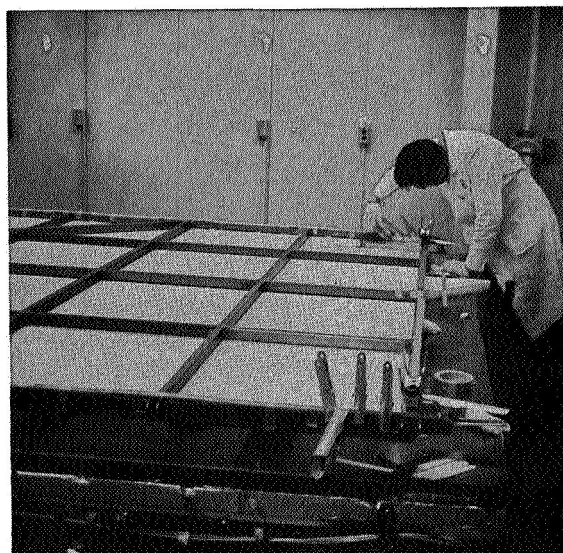
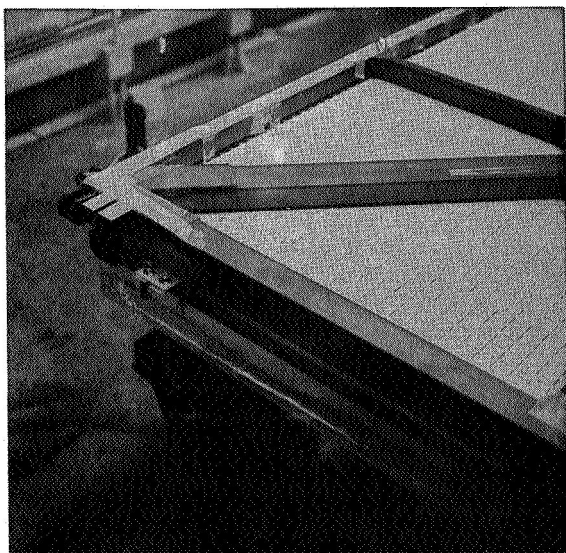
After the minimum cure period of 48 hours for bus bar bonding, the final operations of cleanup, diode installation, module interconnection, and



Prefit Filler Blocks



Apply RTV 40 Adhesive
Position Solar-Cell Module



Bond Bus Bars to the Beryllium Structure

Figure 4-29: SOLAR-CELL MODULE AND BUS BAR INSTALLATIONS

module to bus bar and diode soldering were performed. Because of the large size of the panel, it was possible to perform these operations simultaneously with several operators without interference. Excess adhesive was removed by trimming and pencil-type solder irons were used for making solder joints with conventional tin lead solder with 2% silver. A final inspection of the complete installation concluded the MTA-1 fabrication.

4.3 QUALITY ASSURANCE

MTA-1 fabrication was the largest Phase II Quality Assurance effort and resulted in significant process data.

The *Detailed Quality Assurance Plan* states that design, development, fabrication, assembly, and test of MTA-1 will be subjected to "complete Quality Assurance" and explains just what this entails. Past reports have treated design, development, and software coverage of MTA-1; a summary of fabrication, assembly, and test are discussed in this report.

Solar-cell and coverglass fabrication progressed normally until mid-January 1968, when cell rejections increased due to contact peel. As a result of this cell defect, fabrication and control of solar-cell/coverglass assemblies were suspended by program change order. The Task 7 effort, added to the program by Contract Change Notice 622, replaced the original solar-cell plan. This resulted in adding simulated cells and six modules of live cells to MTA-1 in place of all live cells. Rejections in Task 7 consisted of 70 live and 45 simulated cells that were all dispositioned "use-as-is." The live and simulated cell defects referred to consisted of cell and/or coverglass breaks and cracks.

Beryllium processing consisted of varying degrees of inspection dictated by process sequence and the nature of suspected defect conditions. The Quality Control plan, early in Phase II, identified that penetrant-inspection of all beryllium parts at specifically designated times during processing was necessary. Eddy current techniques were used as backup to provide detailed data on apparent defects, so that the most cost-conscious and best engineering decisions could be provided for each part defect. The next major quality control effort took place as beryllium channels and strips were bonded into sticks. The last process stage consisted of locating and bonding the beryllium-stick assemblies to make up the MTA-1 test panel. Historical and discrepancy data on all of the above work was recorded and is being retained, and is summarized in Figures 4-30 and 4-31.

Details concerning the repair to MTA-1 subsequent to bond failures that occurred in bonding the Sunside and darksides together are contained in the failure mode and analysis report. Records were kept on this operation as explained above for the preplanned MTA-1 operations. In addition, an *MTA-1 Configuration Summary* was released; this document contains final configuration data and a brief explanation of discrepancies that occurred in all MTA-1 work.

Figure 4-30 compares the number of rejection reports written during particular MTA-1 manufacturing or procurement operations. The majority of reports were written during fabrication of solar cells and unbonded beryllium. The assembly operation rejections shown include the MTA-1 final bond failure condition and repair. Procurement, receiving, and miscellaneous operations resulted in relatively fewer rejections. Rework

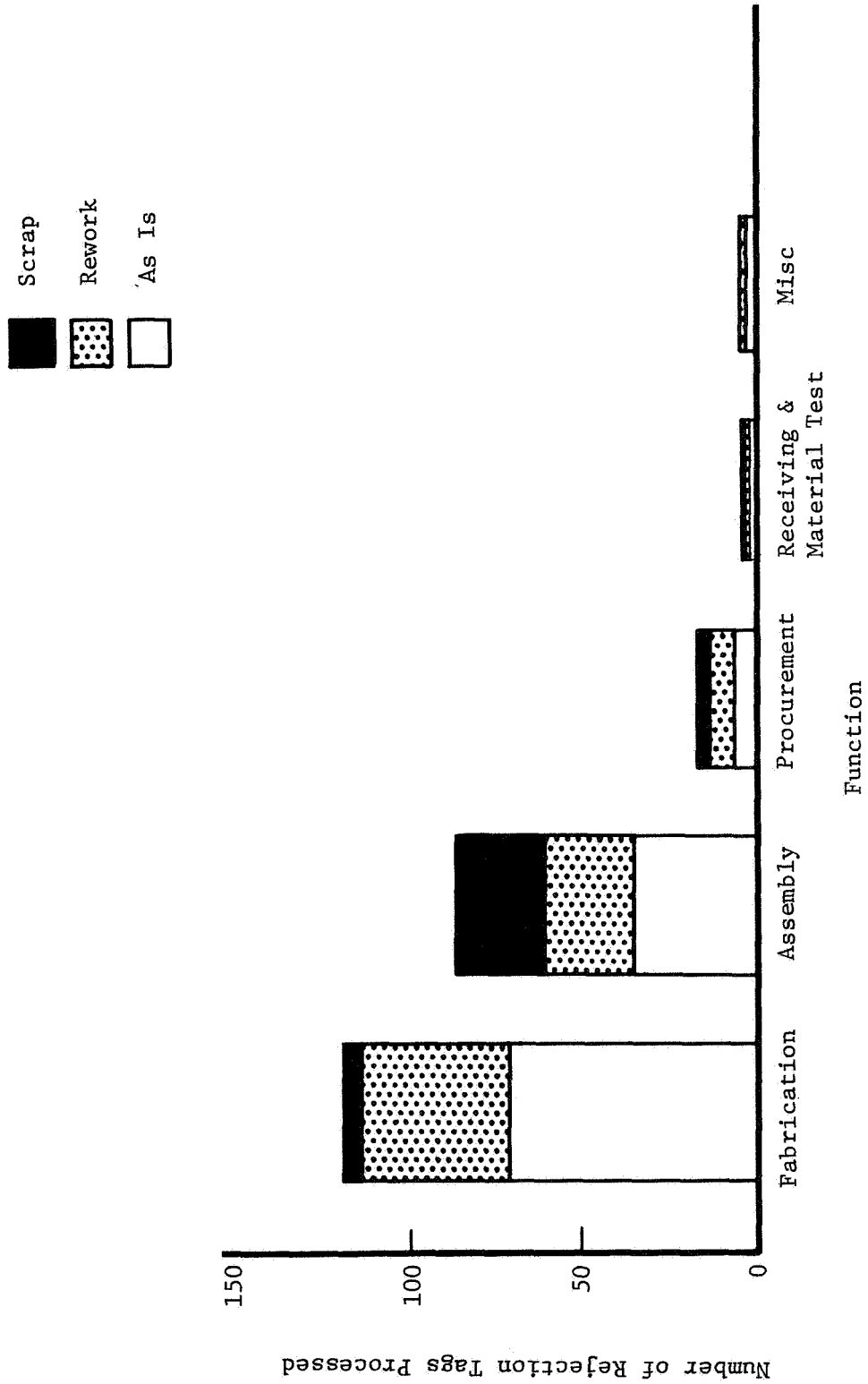


Figure 4-30: MTA-1 REJECTIONS VERSUS FUNCTIONS (TASKS 3 AND 7 ONLY)

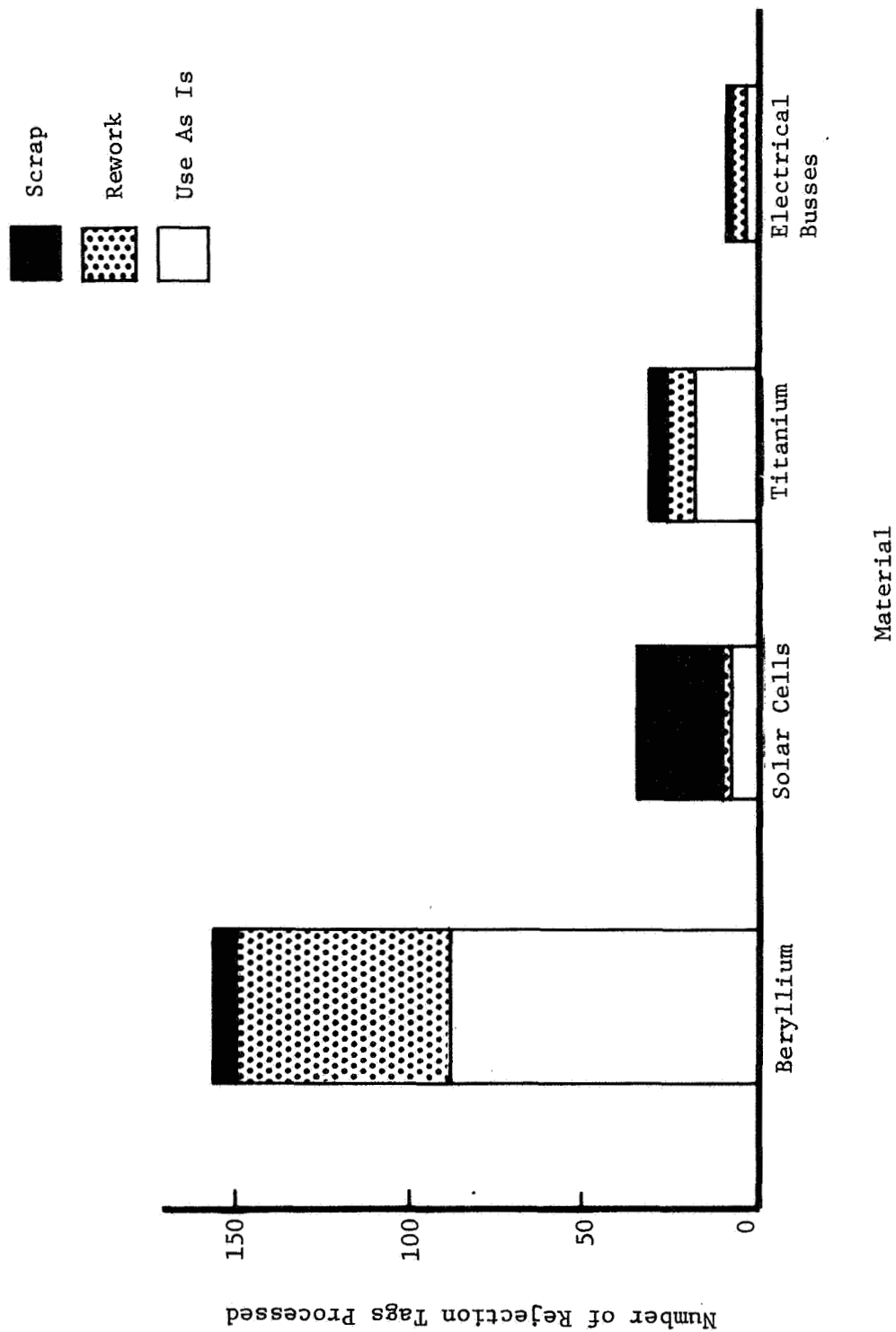


Figure 4-31: MTA-1 REJECTIONS VERSUS MATERIAL IN WORK (TASKS 3 AND 7 ONLY)

of defective parts allowed approximately one third of MTA-1 rejections to be accepted for use during each of the functions shown in Figure 4-30. Approximately 15% of the total rejection reports written in all functions of Figure 4-30 resulted in a "scrap" disposition.

Figure 4-31 further defines MTA-1 rejection reports by comparing rejections, dispositions, and material being processed. This indicates that the majority of scrap consisted of solar cells, the basic cause being the contact peel condition previously mentioned. Rejection reports (4%) of beryllium resulted in scrapping 15 parts.

Figures 4-30 and 4-31 reveal that approximately one half of all rejection tag dispositions were "use-as-is." This last fact made it apparent that some engineering requirements required revision as a result of processing experience gained. The revisions made to engineering requirements allowed fewer acceptable parts to be rejected during the latter part of Phase II. This experience will benefit further LASA work, allowing finer definition of defective part conditions.

Final inspections of MTA-1, performed before and after vibration tests, indicate that testing contributed no structural degradation to the test article.

4.4 VIBRATION TESTING OF MTA-1

The MTA-1 vibration testing was conducted in the large vacuum chamber of the Boeing Space Division laboratory in August 1968. This test was run to obtain the dynamic characteristics of a single subpanel of the Large Area Solar Array in both air and 1/20 atmosphere.

This section describes the test fixtures, test conduct, and test results and analysis.

4.4.1 DESIGN AND FABRICATION OF THE VIBRATION TEST FIXTURE

Hinge-free mounting of MTA-1 in the test fixture facilitated modal survey testing.

The hinge-free mounting of MTA-1 minimizes damping from the test fixture attachments and provides the best test data with the least complicated test setup. The fixture consists of two primary structures. The MTA-1 holding fixture (Figure 4-32) is a steel framework of modified A-frame construction. Three simulated spacecraft hinges attach to the MTA-1 hinges. The weight of the subpanel is removed from the hinges through cable ties at the hingeline and at Station 123, the approximate first-node point.

The vibrator holding fixture is an L-shaped steel structure that locates two 25-pound shakers with 24-ounce voice coils at the free end (Station 169.9) of the subpanel. The armatures of the vibrators attach to the

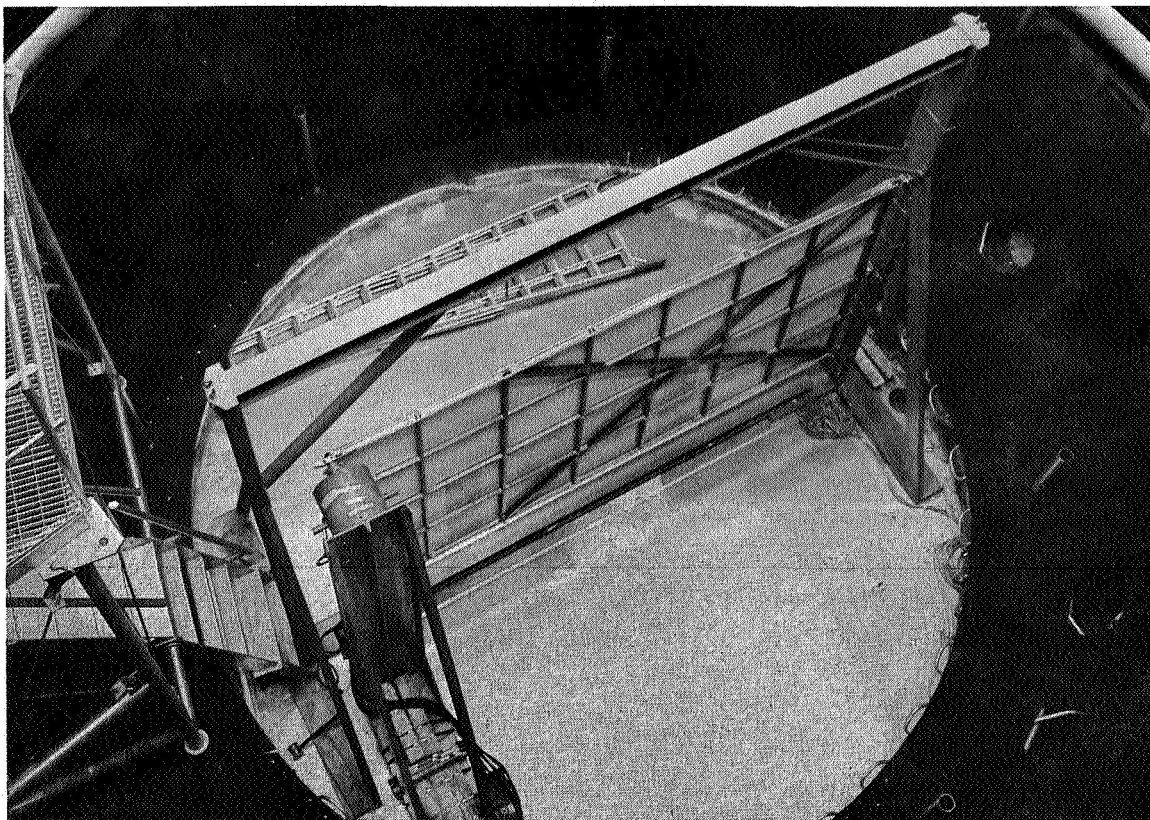


Figure 4-32: MTA-1 INSTALLED IN VIBRATION TEST FIXTURE

subpanel outboard spars through flex rods attached to spar clamps. The flex rods allow movement of the subpanel.

4.4.2 TEST CONDUCT AND DESCRIPTION

The conduct of the MTA-1 vibration test met the test objectives defined in Phase II.

The MTA-1 subpanel was installed in the fixture in a vertical position suspended by two steel wires and tied to the fixture by the three hinge bolts. The strain gages and accelerometers were installed at the locations specified in the test procedure.

The test fixture, with the MTA-1 installed, was mounted on the floor of the space chamber and the vibrator positioned as shown in Figure 4-33.

Tests were run in air first. A low-level sweep from 7 to 40 Hz was used for the bending modes and 3 to 30 Hz for the torsion modes. The sweep rate was 2 octaves per minute.

Two bending modes, 15.3 and 37.5 Hz, and one torsional mode, 5.9 Hz, were selected to obtain a record of decaying oscillations from a stabilized

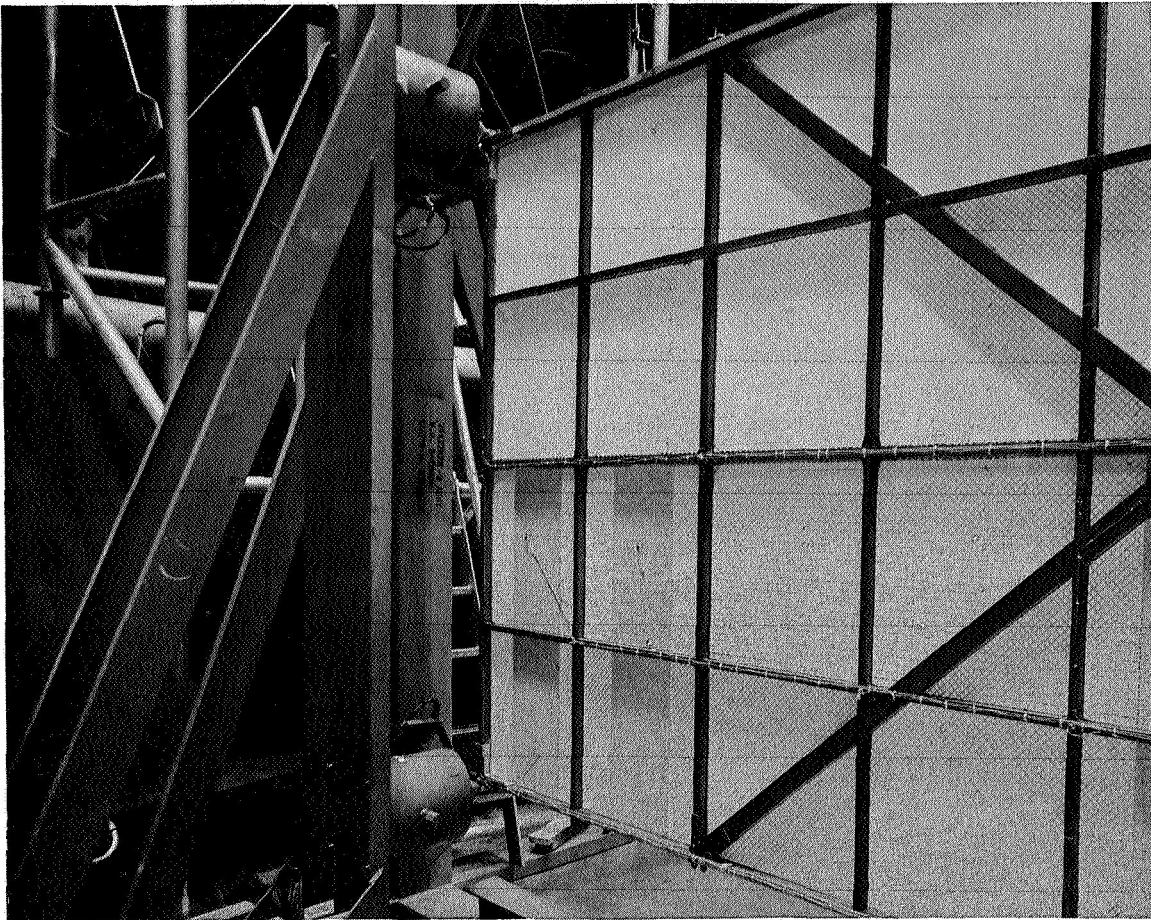


Figure 4-33: MTA-1 TEST SETUP

level. A sweep between 25 and 45 Hz was selected for testing at an excitation level that would cause substrate displacement comparable to the maximum expected for the stowed array. This procedure was then repeated in a vacuum equivalent to 1/20 atmosphere.

The selected bending modes in the vacuum, for decay measurements, were at 15.5 and 37.7 Hz. The torsional mode selected was 6.1 Hz.

Runs were made at the maximum permitted voice coil current (electrical) level setting for the first and second modes in air and vacuum and for the first torsional mode in air and vacuum.

4.4.3 TEST RESULTS AND ANALYSIS

A satisfactory approximation of damping for a single subpanel has been made from MTA-1 test results.

The damping values measured from prototype Subpanel 1 tests, in terms of the magnification factor Q , are shown in Figure 4-34. Approximate values for the Phase II SCS-43 tests are superimposed.

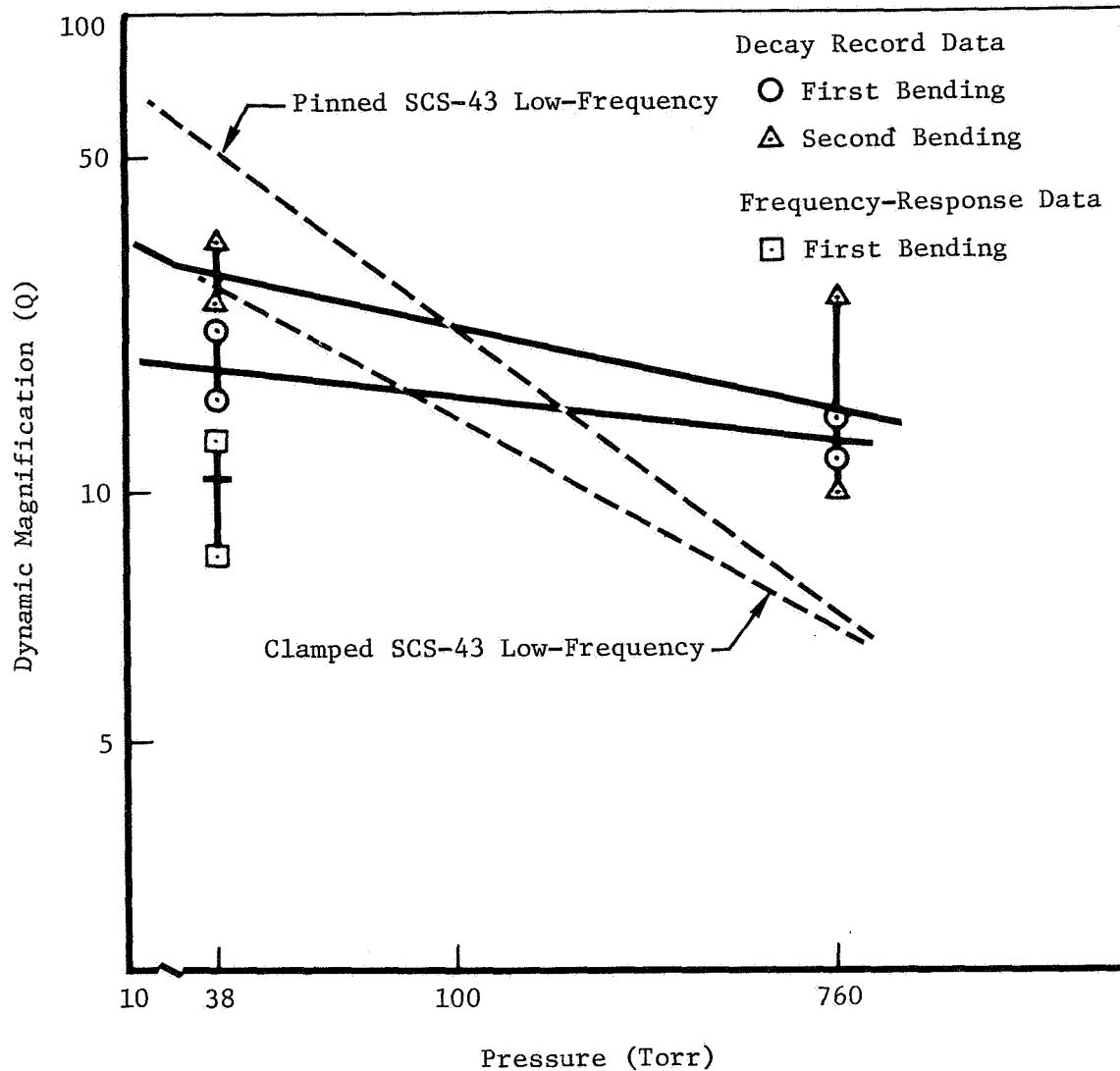


Figure 4-34: EFFECT OF PRESSURE ON DYNAMIC MAGNIFICATION FACTOR Q

As shown in Figure 4-35, a peak stress of 9,500 lb/sq in. was attained during a maximum amplitude run in air at the first-mode bending frequency. A maximum acceleration of 37 g was reached on a substrate at the free end of the panel during a subsequent bending sweep. No damage to the structure or to live or simulated solar cells was observed due to the test environment.

Comparison of the measured resonant frequencies in vacuum, with an analytical calculation, shows acceptable agreement for the first torsion and bending modes. The measured and calculated frequencies in air and vacuum are shown in Table IV-4.

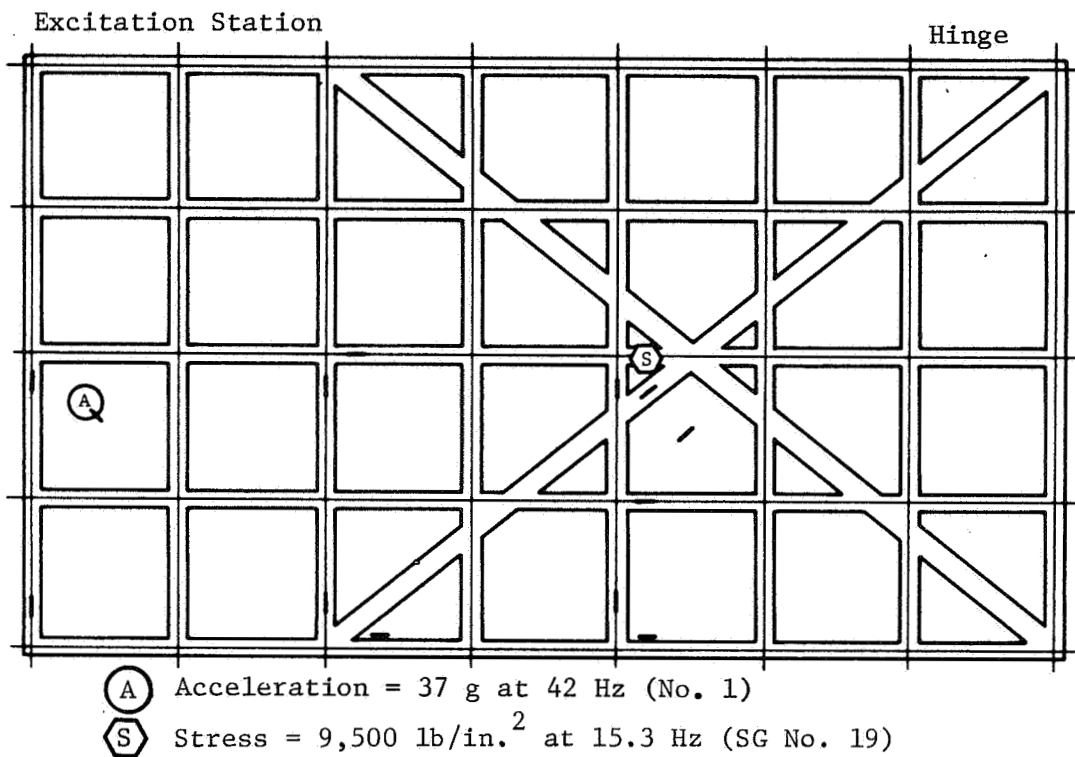


Figure 4-35: MAXIMUM ACCELERATION AND STRESS MEASUREMENTS

Table IV-4: MEASURED AND CALCULATED FREQUENCIES IN AIR AND VACUUM

Mode	Air		Vacuum	
	Calculated	Measured	Calculated	Measured
1st Torsion	5.47	5.9	6.07	6.1
1st Bending	12.54	15.3	14.9	15.6
2nd Torsion	18.7	23.7	22.9	
2nd Bending (Plate?)	21.3		31.6	
	21.7		32.4	26
	21.8		33.1	
	21.9		33.2	29
			33.4	
			33.4+	
		37.5		37.7

The second-bending mode exhibits small motions of the structural elements similar to plate modes, with large motions measured on some of the substrates. The resonant frequency is thus strongly dependent on the basic frequency of the substrate bays. The actual basic frequencies for the substrates of MTA-1 may have been higher or lower than the value used in the analysis for several reasons.

The pretension load in the tapes before bonding is based on a value that will result in the desired substrate frequency for a single frame where all four edges are distorted at the equilibrium conditions. For a system of frames, such as the MTA-1 panel, the distortion of interior members can be less, because only a differential tension between adjacent bays loads the member. The full-tension load must be taken at the edge members, which for MTA-1 are stiffer than the typical panel. Since repairs were made to MTA-1, some tensions may have been reduced or increased. The overall effect could be a variation of tension in the substrate bays, with resultant increased or decreased frequencies for the individual bays.

The measured response values for the second-bending mode at 37 Hz show peak values on three of four substrates, with negligible peaking near 28 Hz. On one substrate and on adjacent structural members, low-level peaks appear at 28 Hz. Low-level response of modes near 31 Hz are predicted by the analysis, which shows a node close to the excitation stations; the consequent low effective driving force would be expected to result in low-level response peaks. Since the structural representation of the beams and intercostals was adequate, as evidenced by the lower-frequency modes, it is concluded that variation in substrate tension contributed to the differences between the measured and calculated frequencies, and that adequate effective separation of the substrate resonances with the critical array mode frequency will be achieved.

Comparison of the measured resonant frequencies in air to the frequencies calculated with assumed incremental mass of air do not show close correspondence. For both the low- and high-frequency modes, the presence of adjacent out-of-phase motion of the panel could contribute to a lower effective air mass, but not to the degree indicated by the frequency differences. Further, comparison of the measured in-air and in-vacuum frequencies shows only small reduction due to air. Examination of the installation configuration within the vacuum chamber and consideration of the chamber dimensions indicate that an effective air stiffness, caused by reflections from the walls, is a probable reason for the almost identical resonant frequencies measured in air and vacuum.

Damping values were obtained from the measured data with two techniques, one of which is biased toward higher damping values and the other toward lower values.

Measurements of decay records showed considerable high-frequency noise, and, in some cases, indications of beating. This technique tends to result in underestimation of the damping.

Measurements of the bandwidth of the response curve at the half-power point give correct values of damping if no adjacent modes distort the response shape. For actual cases, this technique tends to overestimate the damping.

The presence of free play in the hinges is evidenced by noise on the oscillograph traces and by high-frequency energy in power spectral density plots. This phenomena can draw energy from the basic mode and cause imperfect estimations of damping.

The measured values of damping from the decay technique were limited to two decay records in vacuum and four in air on which the undesirable effects were minimum.

Measured values for damping by the response curve technique were obtained by a multiplicity of cases (15-Hz mode only) and the average and maximum values presented.

The power spectral density curves for an in-vacuum test show a distribution of energy in high-frequency modes, most of which may be present because of free play. A rough evaluation shows about twice as much energy in the excitation band as in all the higher frequencies.

Results of the measurements in air and vacuum are shown by Figure 4-34, with data from the Phase II SCS-43 test for comparison. These data are plotted in terms of the magnification factor Q , which is the inverse of the damping coefficient.

The MTA-1 panel experienced lower damping (higher Q) in air in both the 15- and 37-Hz modes than did the comparable SCS-43 pinned or clamped configuration (30 to 40 Hz). This result is consistent with the expectation that the air damping increment is related to the airflow around the panel edges. The length of edge relative to the moving area and the presence of a node on the panel surface both tend to reduce the energy loss due to air for the 15-Hz mode. At the higher frequency (37 Hz), the flow of air between oppositely phased substrate displacements would be expected to have less energy dissipation than those SCS-43 substrate modes where the motion was all in phase.

In vacuum, the damping is dependent on the energy dissipated in the structure only. The results show more damping at 15.6 Hz than was measured in either the pinned or clamped SCS-43 configuration. At 37.7 Hz, the damping was comparable to the SCS-43 clamped condition and more highly damped (lower Q) than the pinned. The results of the two techniques for calculating damping at the 15-Hz mode exhibit the expected bias, with an average Q for the bandwidth techniques of about 11, and for the decay technique of about 18. Even with a correction for damping appropriate to an estimate of the energy loss in high frequencies, it is apparent that the assembled structure contributes a significant energy dissipation. The primary source must be from the many load transfers through bonded splice plates and doublers connecting the assembly of beryllium parts.

The results of the tests show:

- 1) The measured frequencies in vacuum agree acceptably with calculated values except for the second-bending mode.
- 2) The measured frequencies in air did not agree with calculated values. An effective stiffness of air caused by the proximity of vacuum chamber walls is a possible explanation.
- 3) The bonded connections of splice plates and doublers between the beryllium parts contribute a significant energy dissipation. Measured damping (using the most critical technique) in vacuum average $Q = 18$ at the lowest bending frequency and $Q = 29$ at the second-bending mode.
- 4) The measured damping in air is about 30% greater than in vacuum.
- 5) A peak value of 37 g was sustained on a substrate without damage to the cells. A maximum stress level of 9,500 lb/sq in. was measured on the structure.

An estimation of the MTA-4 configuration damping, based on available data, can be little better than an opinion at this time. However, neglecting the energy dissipation in the tiedown structure, it is believed that a worse value of $Q = 25$ to 30 might be realized, and a best value of $Q = 16$ to 20 is possible.

4.5 SOLAR-CELL DETERIORATION

Observed solar-cell deterioration has invalidated the Phase II reliability and power loss analysis reported in D2-114070-1.

A serious problem appeared with increased production quantities of cells for MTA-1. Blistered and peeled ohmic contacts were found on a significant portion of the cells, some of which had previously been inspected without any evidence of deterioration. Figure 4-36 shows some examples of severe cases of deteriorated contact surfaces. The obviously defective cells were rejected and put aside for possible future study.

The use of live solar cells on MTA-1 was suspended by a contract change notice pending resolution of the deterioration problem by JPL and the cell supplier. The unused customer-furnished cells have been retained in storage at Boeing and are available for further investigation.

Based on the evidence available in Phase II, it was found that:

- 1) The cause of the deterioration has not been conclusively identified. The appearance of the cells was similar to that of laboratory specimens exposed to high relative humidity for extended periods. Some Boeing laboratory test specimens have shown similar deterioration

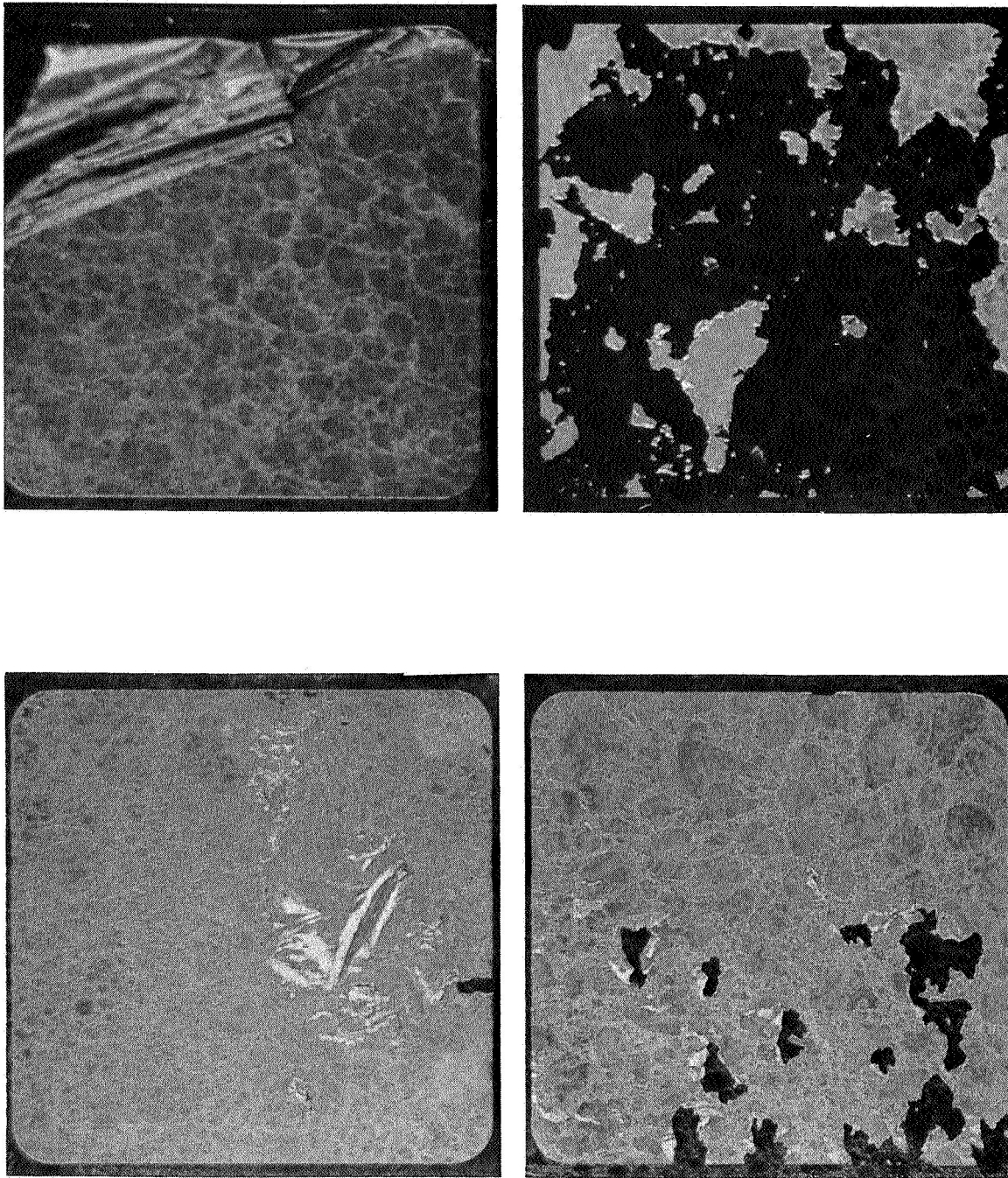


Figure 4-36: SOLAR CELL DETERIORATION

after exposure to 75% relative humidity at near-room temperature for 34 days. However, there has been no evidence to show that comparable humidity exposure occurred during normal indoor storage.

- 2) There was no evidence that the deterioration could be arrested. If such a method were available, it would have been possible to consider procedures to salvage those cells with acceptable electrical characteristics.
- 3) The occurrence of deterioration may be random; i.e., have equal probability for any given cell. However, the deterioration process appears to be a function of time and environment. The Phase I reliability and power-loss calculations were based on a low random failure rate of 1×10^{-7} failures per cell hour. The observed cell-deterioration phenomena simply does not fit such an analysis model. Therefore, the results of the Phase I analyses were invalid for these particular cells.

4.6 RADIATION DAMAGE ANALYSIS

Radiation degradation of the LASA will be less than 3.7% during 210-day missions to Mars in the 1971 to 1973 period.

A previous analysis, where backside radiation was not considered, showed radiation degradation from 9 to 16%. The earlier analysis was conducted for a 350-day mission and for Sunspot numbers of 112 for 1970. Using Webber's predictions for Sunspot Cycle 20 and extrapolating through 1973, the average Sunspot number was found to be 58 for 1971, 38 for 1972, and 25 for 1973. Accordingly, the latest analysis was based on less than 58 Sunspots per year.

An assumption of a $1/r^2$ dependence of particulate radiation for cosmic ray events can lead to erroneous results. Instead, a model based upon a diffusion process has been proposed. This model is valid in a region extending to 1.7 AU from the Sun, which is beyond the orbit of Mars. A $1/r^2$ comparison with the diffusion model indicates that the radiation level may be as much as 25% higher, using the diffusion model that was used in this analysis.

There are at present two methods of calculating radiation damage: (1) correlation of power with changes in diffusion length, which in turn can be related to a damage coefficient, at least for conventional solar cells; and (2) conversion of the incident radiation into a damage-equivalent, normally incident, 1-Mev electron flux.

The second method was used for the present analysis for several reasons. For example, the degradation of the diffusion length has not been accurately and consistently correlated with the creation of defect centers by radiation. Also, for LASA cells; nominally 0.00667-inch thick, no data are available for the relation between power output and diffusion length. For

such thin cells, an effective diffusion length, much shorter than the bulk-material diffusion length, further complicates the use of a diffusion length calculation. Also, considering the reduced thickness of the cells and the radiation incident on both surfaces of the cell, the radiation-induced defects should be more homogeneous than in the conventional solar cell irradiated from one side. Thus, the conversion to equivalent 1-Mev electrons is considered adequate for damage evaluation.

The energy distribution of particles incident on the solar-cell stack is not the same as that incident on the solar cell itself. The particles lose energy passing through matter. The lower-energy protons are stopped by the nominally 0.0031-inch microsheet coverglass on the front surface, protons in the range of approximately 3 to 30 Mev are modified, and protons with energies greater than 30 Mev are essentially unaffected. Likewise, at the back of the solar cell, the incident radiation is modified by the 0.003-inch fiberglass tape substrate and 0.002-inch layer of RTV-40 used as adhesive and thermal-emittance coating.

The modified proton-energy distributions were calculated using the proton stopping-power data (Reference 16). No specific stopping-power data exists for microsheet, fiberglass, or RTV-40. However, each of these is a Si-O compound, thus justifying use of quartz data adjusted for equivalent thickness in units of g/cm^2 . The back surface of the solar cell is a 3-micron-thick silver-titanium (mostly silver) contact, which also modifies the radiation.

Damage equivalent curves are based on uniform omnidirectional radiation on a solar cell that is assumed to have effectively infinite shielding on one side. Thus, the degradation can be separated into components due to (1) irradiation on the solar-cell front surface, and (2) irradiation on the solar-cell back surface.

The fiberglass tape substrate covering 40% of the back surface of the solar cell modifies the incident proton spectrum differently than the RTV-40 on the remaining open 60%. The analysis accounted for this difference by determining the proton spectrum through each of the applicable layers, and then converting each resultant spectrum incident on the back surface of the silicon of the solar cell to damage-equivalent normally incident 1-Mev electrons. Degradation of the solar cells as a function of 1-Mev electron fluence for 210-day missions in the best and worst conditions of the 1971 to 1973 period is summarized in Table IV-5.

Passage through the Van Allen belts could result in degradation of the solar array but the launch trajectory for a typical Large Area Solar Array application shows that the array remains within a sunshade until 3 hours after launch. During the short time in the Van Allen belts, the protection provided by the sunshade, and the fact that the array is folded, providing additional protection for each subsequent panel in the stacked array, should result in negligible Van Allen radiation effects to the solar array. Were a parking orbit occupied before an Earth escape, then additional analysis would be necessary, taking into account the orbit parameters of inclination and altitude as well as the time spent in orbit.

Table IV-5: SUMMARY OF RADIATION DAMAGE

<u>Incident Radiation</u>	<u>Degradation* (percent for a 210-day mission)</u>	
	<u>1971</u>	<u>1973</u>
Front Surface	1.2	0
Back Surface	2.5	0

*Based upon the constraints described in the text.

Other regions of the solar cell that can be damaged are the surface and the pn junction. Experimental data show that surface damage is possible with 27.5-kev protons. However, the LASA fluence of 27.5-kev protons is below any levels where degradation can be observed. No significant damage in the junction region of the solar cell is expected. The energy of a proton from the front surface stopping in the junction region is approximately 70 kev. Higher-energy protons, approximately 4.7 Mev, penetrate to the junction region from the backside. Calculation shows that less than 10 protons/cm³ would be deposited in the vicinity of the junction. This is several orders of magnitude less than the majority carrier concentrations in either the shallow diffused layer or the base region of solar cells. Otherwise, there is insignificant data available for evaluating junction damage to the cell.

The present 0.002-inch adhesive thermal-control coating on the backside of the solar cells is also adequate as a low-energy particulate radiation shield. This analysis shows that there is no need for increasing the thickness of this material to provide additional radiation shielding.

5.0 GROUND-DEPLOYMENT TESTING (TASK 4)

The overall objective of Task 4 was to develop equipment and procedures for LASA ground-deployment testing. A secondary objective was to develop array mechanisms test data for design applications. This section contains a discussion of the Phase II ground-deployment equipment design, fabrication, and tests conducted to meet these objectives.

Ground deployment of the array is necessary to confirm the deployment system design and the folding modular packaging concept. Because the array is designed for space deployment only, ground deployment presents many difficult GSE design problems. After considering many possible design concepts, it was concluded that the best approach to ground-deployment testing was to perform separate tests--verification and demonstration. Verification tests provide experimental data to support the design and analysis of the array; demonstration tests prove the complete deployment sequence and geometry. This concept is the least-costly approach commensurate with meeting the basic deployment testing objectives.

Because of the long-lead time involved in the design and fabrication of the main subpanel deployment verification equipment, this equipment was built and tested in Phase II to permit early hardware development and gain equipment operating experience. The equipment permits intermittent deployment of a panel assembly. A lightweight air-bearing sled running on a precision track, and a set of overhead suspension lines provide a minimum friction and inertia support system for the panel assembly during deployment.

A full-size aluminum panel assembly deployment model (MTA-3) was designed and the main subpanels only fabricated for preliminary deployment tests in Phase II. Flight prototype mechanisms were installed on the model. Later, in Phase III, the beryllium panel assembly (MTA-4) will be deployed using the equipment and techniques developed in Phase II.

A GSE failure mode and effect analysis (FMEA) was made that included the MTA-3 deployment equipment, handling equipment, and preliminary analysis of MTA-4 equipment. This FMEA effort is discussed in Section 6.4, under Task 5.

5.1 TEST OBJECTIVES

A valid reason exists for each of the specific test objectives.

Three primary test objectives were established for the Phase II deployment tests:

- 1) Obtain developmental operating data on array mechanisms for design applications and, where practical, determine that the mechanisms designs are adequate to perform the main subpanel deployment functions;

- 2) Develop the simulated zero-g deployment test equipment and procedures and provide data to compare to the MTA-3 deployment analysis;
- 3) Determine aerodynamic effects on deployment test results.

Operational data on mechanisms was required to permit revision of mechanisms design, if necessary, before Phase III fabrication. The mechanisms system contains several long-lead procurement items. In addition, changes in mechanisms installation would be much more difficult to accomplish on the Phase III beryllium MTA-4.

By comparing test data with that predicted by the MTA-3 deployment analysis, differences were identified and explained. In Phase III, this technique will be used to confirm the MTA-4 ground-deployment analysis and, by analogy, the analysis of panel assembly deployment in space.

Deployment of the MTA-4 panel assembly on Earth will be modified by aerodynamic drag. However, the drag of MTA-3 with bare frames is negligible. The frames of MTA-3 were covered with a plastic membrane and a series of deployment runs conducted. Results of these runs were compared with bare-frame deployment to determine the aerodynamic drag caused by the membrane and to provide data for adjusting Phase III MTA-4 test results to eliminate drag effects. Small increments of drag may have been masked by high frictions in outboard subpanel deployments sequences.

5.2 MTA-3 TEST ARTICLE

5.2.1 MTA-3 STRUCTURE

The aluminum structure of MTA-3 provided a suitable representation of the MTA-4 beryllium structure for Phase II test purposes.

MTA-3 consists of an assembly of five aluminum rectangular panels of the same basic dimensions as MTA-4. The primary structural members are rectangular extruded tubes and the diagonal brace members are round tubes. All connections are made by welding. The geometry is duplicated in the areas of mechanisms and GSE interfaces. Spacers simulate the envelope of the auxiliary subpanels.

Lead weights added to each MTA-3 subpanel simulated the mass, center of gravity, and mass moment of inertia of the flight prototype subpanel units (see Figure 5-1). The effect of the GSE contributions to the moment of inertia for the mechanisms and aerodynamic deployment tests is shown by Table V-1.

Duplication of the stiffness of the beryllium structure was not a requirement for deployment testing in Phase II. Actual values of bending for the MTA-3 aluminum panels were approximately 30% of that for beryllium for the first panel and 85% for the second through the fifth panel. The effect of the overhead cable support was to increase the effective stiffness of the deployed panels. Figure 5-2 shows comparison of the first three bending

frequencies of the aluminum MTA-3 in the deployment fixture, and of the MTA-4 structure in space.

As a consequence of the overhead cable effect and the elastic stiffness of the cable, transient conditions were not modeled, and only time-history data were obtained.

5.2.2 MTA-3 MECHANISMS

Prototype mechanisms were necessary to provide an adequate simulation of characteristics of a flight configuration array.

Flight prototype mechanisms were installed on the aluminum deployment model. The use of prototype parts permitted the accumulation of actual operating data, life expectancy information, and alignment experience, and revealed several design discrepancies.

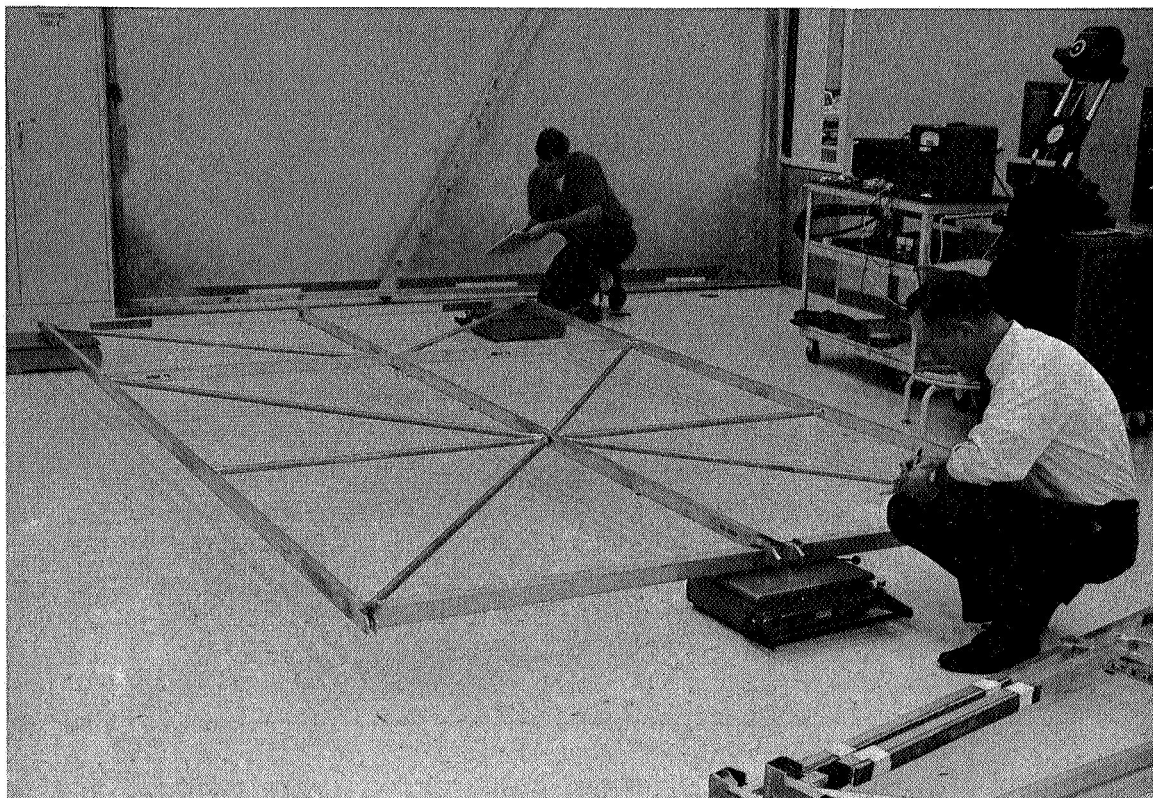


Figure 5-1: WEIGHT/BALANCE OF MTA-3

Table V-1: PANEL MASS MOMENTS OF INERTIA

MTA-4		MTA-3 Plus GSE		
Panel Sequence	Present	(Bare Frame) Present	Present Plus Air Inertia	Total With Visqueen*
1	9,640	10,195	11,090	11,297
2	11,680	12,694	13,600	13,853
3	4,900	5,371	5,945	6,039
4	3,630	4,197	4,770	4,908
5	1,120	1,539	1,800	1,846
* Added membrane to bare frame				

MTA-3 (Hz)	MTA-4 (Hz)
6.5	12.3
12.0	17.5
29.0	42.0
0.55	0.72
2.83	3.8
7.97	10.3
0.27	0.25
1.10	1.43
2.97	3.79
0.22	0.12
0.61	0.73
1.57	1.98
0.21	0.08
0.44	0.48
1.05	1.31

- MTA-3 With Overhead Cable Support
- MTA-4 With Overhead Air Bearing

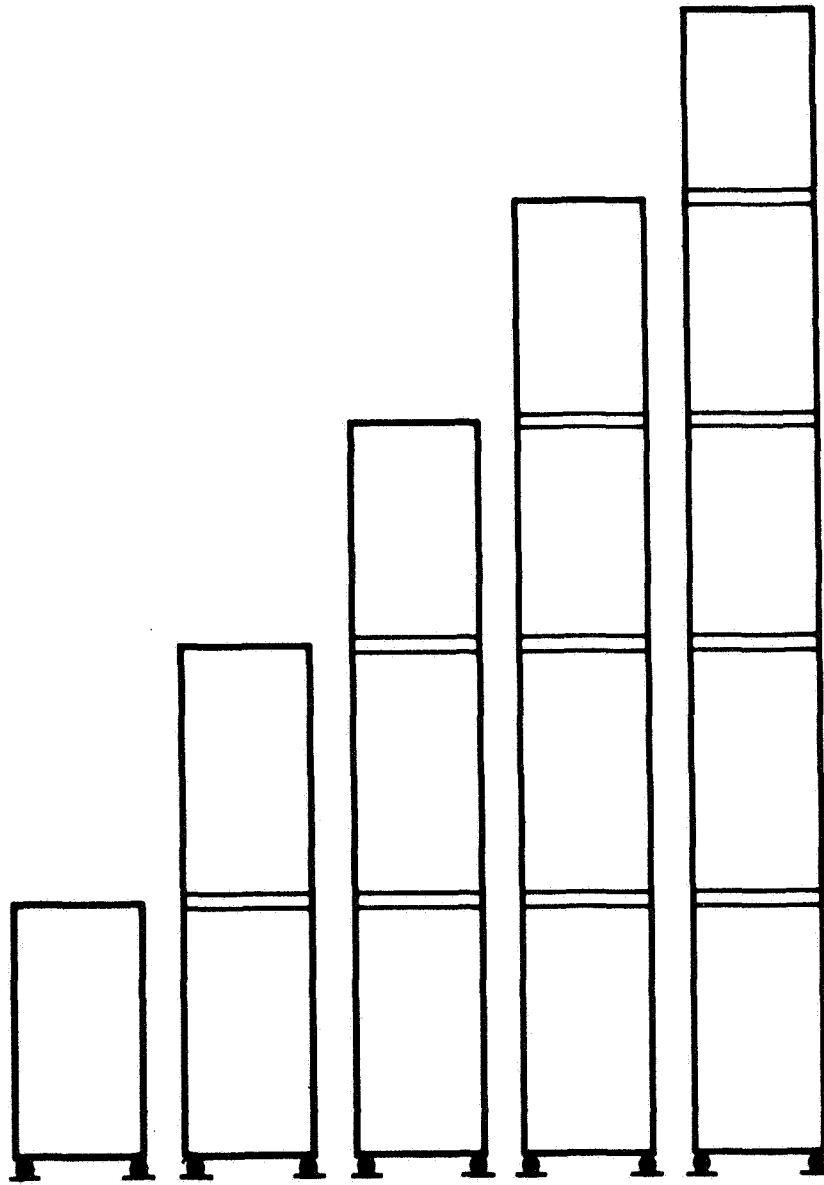


Figure 5-2: COMPARISON OF MTA-3 AND MTA-4 BENDING FREQUENCY

The main subpanel deployment drive consisted of a motor/cable drum that retracted a cable over quadrants (see Figure 5-3) located at the center of the end spar of each main subpanel. The cable was anchored to the outermost subpanel.

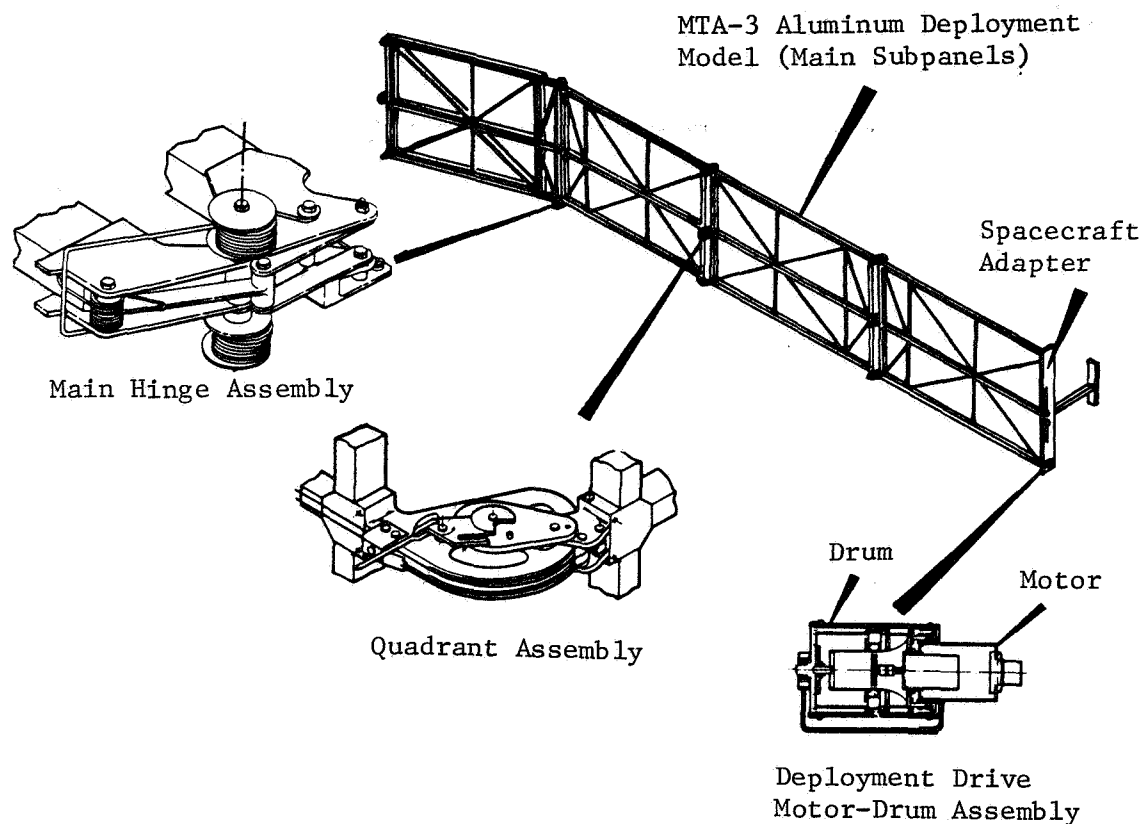


Figure 5-3: DEPLOYMENT MECHANISMS

As the cable was winched in, it applied a force tangent to the quadrant rims, which are offset from the hinge centerlines. This force converted to an opening moment about the hinge centerlines.

The quadrants were machined from aluminum plate and mounted to the panel and spars by bolts.

The main panel hinges, shown in Figure 5-3, contained resistance springs that opposed the cable opening force to prevent runaway deployment. Deployed panels were locked in place by overcenter latches.

5.3 SIMULATED ZERO-GRAVITY DEPLOYMENT SYSTEM DESCRIPTION

5.3.1 MECHANICAL SYSTEMS DESIGN

The design of the mechanical systems of the deployment fixture allowed self-deployment of MTA-3 with minimum gravity effects.

Deploying sets of subpanels were supported on a virtually friction-free air-bearing sled as shown in Figure 5-4. This sled, traveling on a precisely leveled air-bearing track, allowed free X-Y motion of each deploying mass. Moments due to the slope of the track were minimum. For example, the slope of the track for the first deployment motion (or swing) was leveled within 0.003-inch elevation in 40 inches of track length. This resulted in a maximum moment of 3.1 inch-pounds on the deploying set of panels. The slope could be overcome by 1-pound tension in the deployment cable. Such small moments are felt to be the best practical capability for any ground-deployment system of such large size.

The deployed subpanels were supported by low spring-rate suspension lines to minimize gravity loading on the hinges. These suspension lines were designed as lightweight units so that their inertia would have a minimum effect on the deployment behavior of the array.

The adapter simulated a spacecraft at the mechanical interface with the first subpanel and the diagonal strut, and provided a mounting for the deployment motor and winch unit. The adapter was suspended by a counter-weighted overhead line and restrained by ball- and slider-type fittings that allowed limited rotation and translation except that rotation in the direction of deployment was prevented.

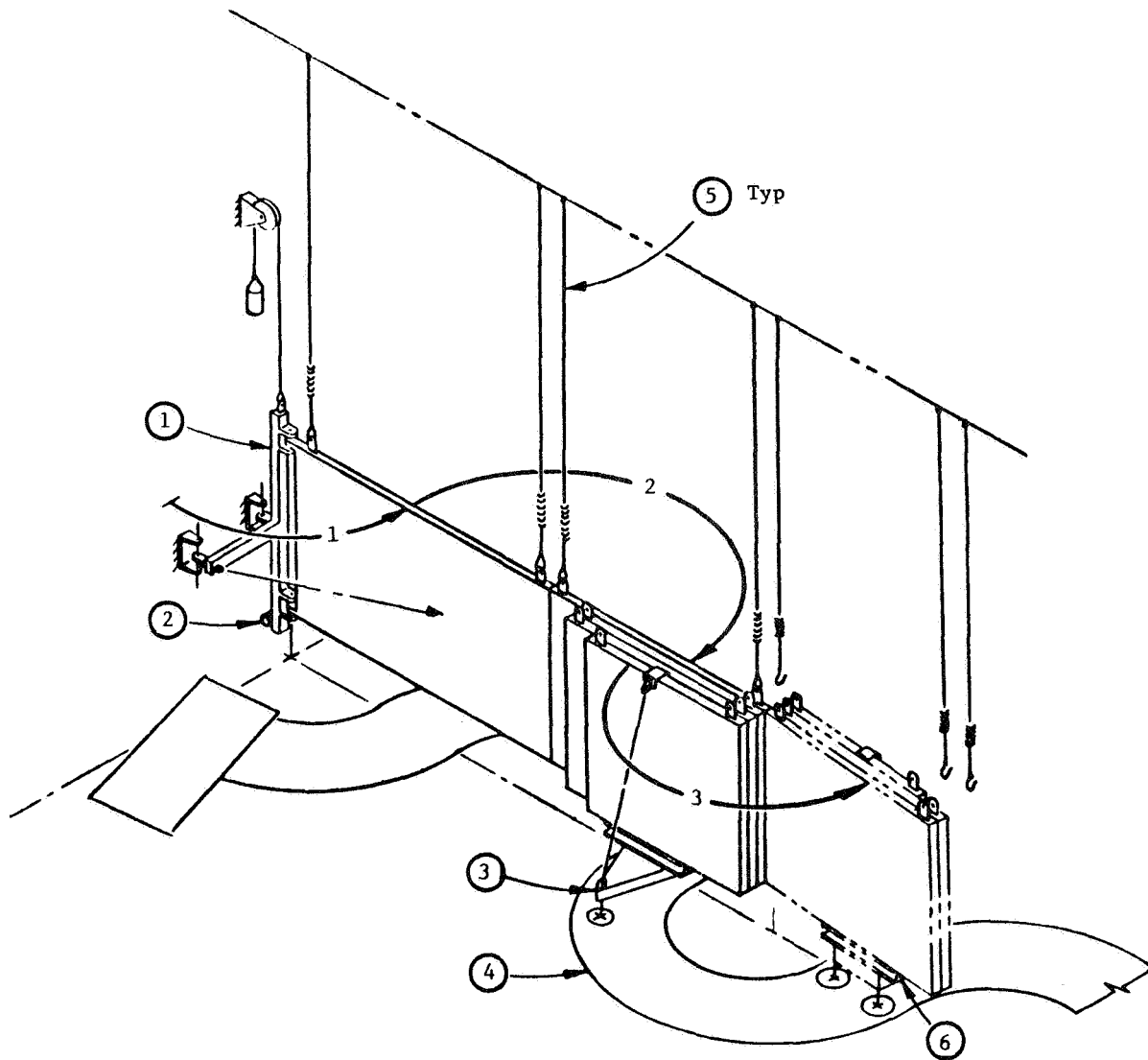
The mechanical operation of main subpanel deployment was intermittent in nature. At the conclusion of each deployment swing the deployed panel was transferred from the air sled to the overhead suspension lines by the simple rerigging process described in Figure 5-4.

5.3.2 INSTRUMENTATION SYSTEMS DESIGN

The design of the instrumentation systems of the deployment fixture provided for the required recording, monitoring, and control of the various parameters.

Angular displacement of each of the five deploying main subpanels was measured by a small, infinite-resolution potentiometer mounted in line with the hinge bolts. The body and shaft of the potentiometer were held to adjacent hinge plates by flexible brackets. The shaft was visually aligned with the adjacent hinge bolt.

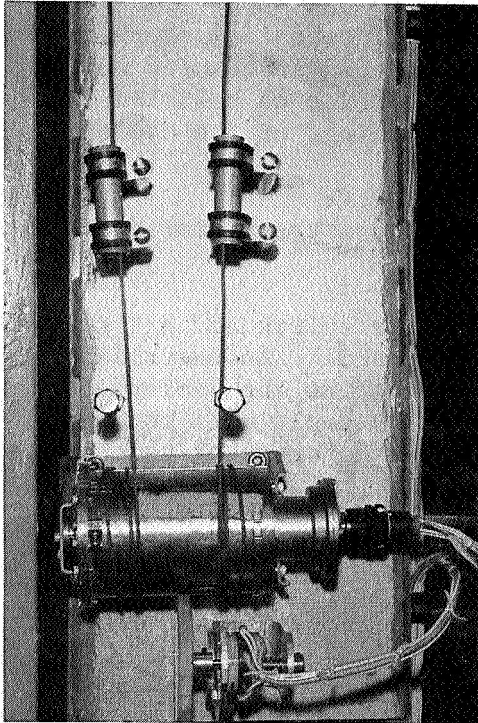
Tension in each deployment cable was measured by means of a spring-loaded linkage that actuates a linear variable differential transformer (LVDT) (see Figure 5-5).



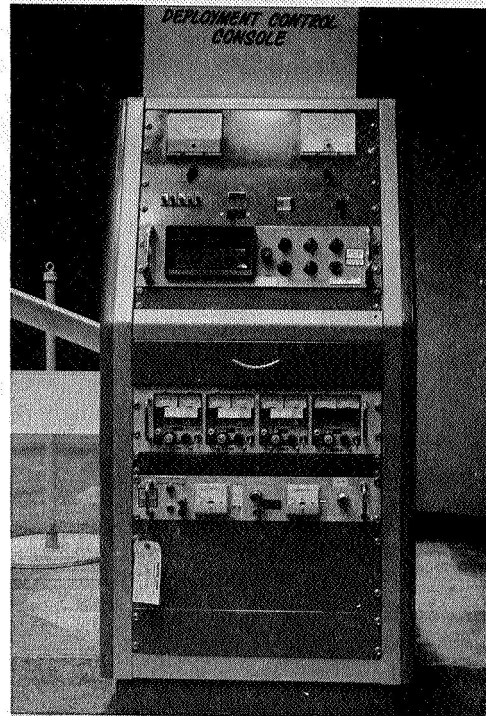
The Adapter, (1), simulates a spacecraft at the mechanical interface with the first subpanel. A Deployment Motor Winch, (2), working through a system of cables and quadrants, provides all the energy for deployment. The Air Sled, (3), supports each deploying set of panels and provides a zero friction carriage. The Air Bearing Track (4), is a precision, level surface on which the Air Sled travels.

At the conclusion of each deployment swing, the deployed panel is attached to a pair of soft spring-rate Overhead Suspension Lines, (5), and a Cross Beam, (6), is removed from the Air Sled to allow deployment of the succeeding set of panels.

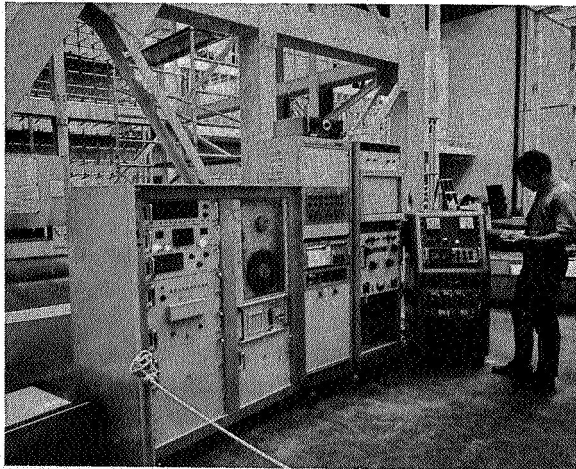
Figure 5-4: DEPLOYMENT FIXTURE MECHANICAL OPERATION



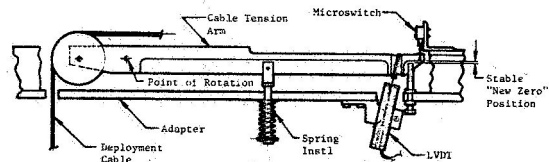
Motor Windup Rate
Instrumentation



Control Console



Data Acquisition System



LVDT Installation

Figure 5-5: INSTRUMENTATION SYSTEM

Motor windup rate was measured by means of a sine-cosine potentiometer driven by the motor drum. A rubber-covered wheel was mounted on the potentiometer shaft and this unit was spring-loaded to hold the wheel against the center of the drum between the windings of the deployment cables.

Monitoring of motor temperature was provided by a thermistor installed on the motor.

The deployment motor was controlled by a master switch and a three-way "deploy-off-reverse" switch on the control console. An overriding remote control switch also was provided. Input voltage to the deployment motor was adjusted and monitored by means of a power supply unit mounted in the console. Meters on the console were used to monitor motor temperature and cable tension.

The data acquisition system is shown in Figure 5-5. The deployment control console, X-Y plotter, and visicorder provided quick-look capability for the real-time monitoring of critical test parameters. The digital recording system was used to store all test measurements on magnetic tape for later reduction by an SDS-910 computer.

5.3.3 FABRICATION AND ASSEMBLY

Fabrication and assembly of deployment fixture components utilized imaginative applications of modern materials, techniques, and processes.

A cost-effective method of producing the large, precision surface air-bearing track sections was developed. Each of the 60-degree track sections was cast of 356 aluminum and annealed to prevent warpage. This method uses reclaimed material and is superior to welded steel as a cost-effective means of producing this type of large, stable base structure. The track sections were coated with 0.5 inch of epoxy (2L2 Epocast). The epoxy surfaces were then machined two at a time on a large vertical Hypro lathe. Flatness as measured on the lathe after machining was 0.0005 inch TIR. Local irregularities in the surfaces, after machining, were repaired by routing out the area affected and filling with epoxy. The patches were then reduced to 0.001 to 0.003 height with a rivet shaver and carefully hand-sanded with fine wet paper on a sanding block. These local irregularities, caused by bubbles or incomplete mixing of the epoxy, can be minimized if a power mixer is used rather than mixing by hand.

A series of overhead suspension cables is required to eliminate gravity loads at the subpanel hinges during deployment on the fixture. Music wire was used to make the lightweight, low spring-rate, minimum-creep suspension lines. This configuration proved to be lighter and more creep-free than similar lines constructed earlier of organic materials.

The air bearings for the air sled were made of 7075 aluminum for light weight and stiffness. The bearing surface was faced and the six circular recess areas were masked with microshield compound. The remaining bearing

surface was aluminum hard-coated to a total thickness of 0.002 inch (0.001-inch penetration and 0.001-inch buildup). The masking was removed leaving 0.001-inch-deep recesses. The aluminum hard-coating process produces an aluminum oxide skin on the parent aluminum. This skin prevents scratches that could reduce the lifting capacity of the air bearing. It also provides a low coefficient of friction with the track surface, which is useful in braking the array to a stop by turning off the air supply to the bearings.

Conventional aluminum welding and heat-treating techniques were used to make the rigid, lightweight adapter and air sled. The air sled frame was made of welded 6061 aluminum tubing. It was heat-treated after welding to strengthen the material softened by welding.

5.3.4 INSTALLATION AND SYSTEMS INTEGRATION

Numerous minor design and procedural discrepancies were discovered and corrected during installation.

The structure was erected to normal structural steel tolerances (generally ± 0.25 inch). The detailing, fabrication, and erection of this structure was subcontracted to a local steel fabricator who, working to these tolerances, was able to complete the structure ahead of schedule and at a minimum cost to the program. The structure being erected is shown in Figure 5-6. The track sections were positioned to arcs scribed on the floor. The sections were then leveled by means of jackscrews to a tolerance of 0.003-inch total for each section and 0.006-inch total overall. Levelness was measured with a Wild optical level reading a scale placed at six different points on each track section. Initial track levelness was accepted by Quality Control. With this exception, the fabrication, installation, and calibration of all equipment were accepted by Engineering inspection.

The track surface was used as a basic horizontal reference plane. The adapter and the overhead suspension lines were set at specified heights above this plane and in the proper position over the centerline scribed on the floor.

The MTA-3 was installed in the fixture and a series of checkouts was performed to determine the resistances to deployment built into the combined GSE/MTA-3 system. These checkouts consisted of pushing each deploying set of subpanels with a measured starting impulse (a plunger-type spring scale was used), and observing the angular distance traveled and the consistency of rate. The subpanel sets were pushed from various starting positions and in both directions.

These checkouts indicated an appreciable increase in hinge-bolt friction after the first 60 degrees of deployment due to misalignments. The misalignments found and the method of correction are diagrammed in Figure 5-7.

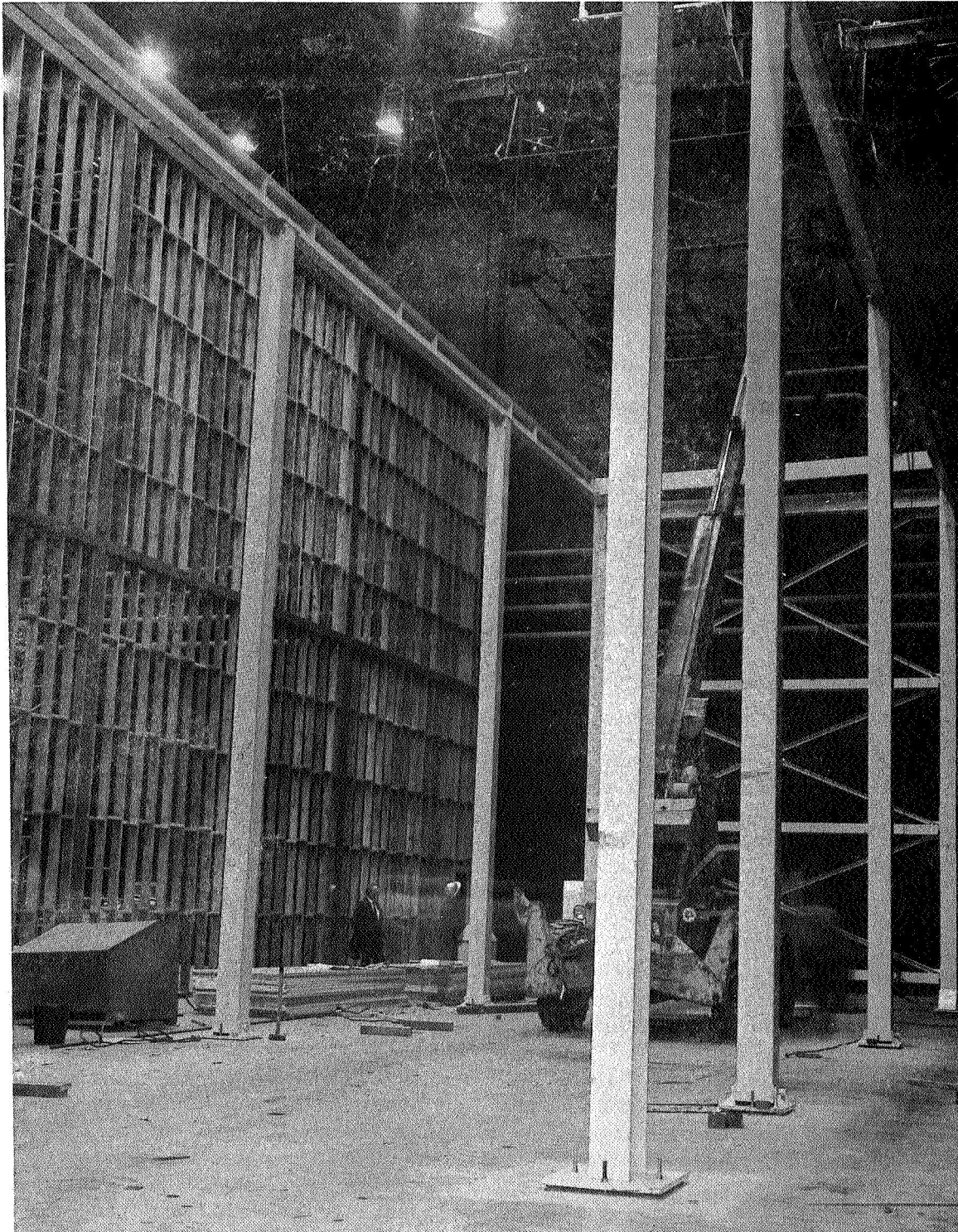
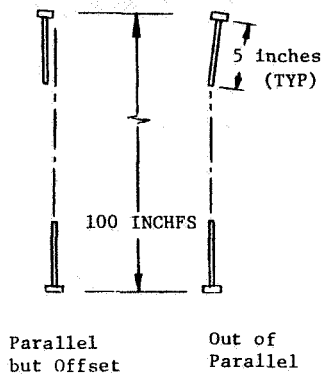


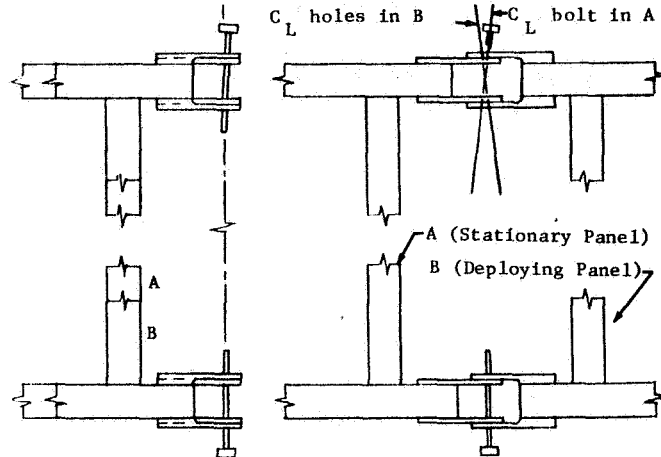
Figure 5-6: ERECTION OF DEPLOYMENT FIXTURE STRUCTURE

1. Initial Misalignment of Hinge Bolts (Typ at hinge lines 3, 4, and 5)



The maximum misalignment was approx 0.07 in 5 in.

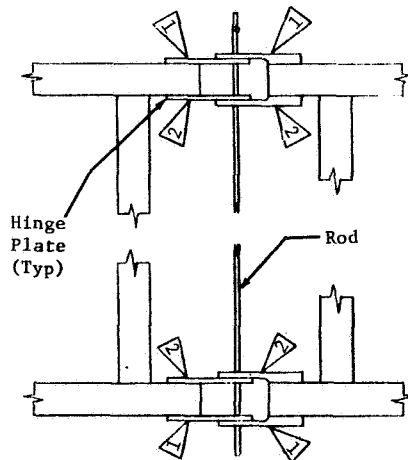
2. Resulting Friction During Deployment



The hinge bolts were free in the stowed position - but -

As the fully deployed position was approached the bolts became tight, the hinge plates were deflected, and higher frictions were observed

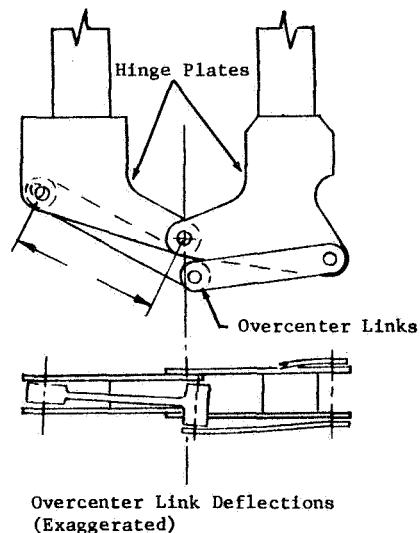
3. Correction of Misalignment



- Panels hung from suspension lines in deployed attitude
- Bolts removed and long rod inserted thru hinge plates
- Hinge plates noted by 1 were left in place as a reference
- Hinge plates noted by 2 were loosened and moved so that the rod appeared straight and turned freely

The flexibility of the rod limited the accuracy of this method, but the final alignment was felt to be within 0.03 in 5 in.

Other Misalignments



Tolerance variations in dimension d above, in otherwise identical hinge plates, caused misalignment of the overcenter links. The resulting deflections and loading contributed to hinge friction

Figure 5-7: HINGE MISALIGNMENTS

After correction of these misalignments, the various sets of subpanels again were pushed through their swings and generally satisfactory results were obtained, although hinge friction remained relatively high.

The first complete deployment of MTA-3 under its own power was used as a functional checkout of the integrated ground-deployment system.

The air-bearing track was measured and adjusted to the specified levelness tolerance of 0.003 total per 60-degree block. The fairing at the joints was then adjusted in accordance with the procedure that specifies that the outboard end of each block remain fixed and the height of the inboard end of each block be adjusted to obtain fairing. This caused the levelness of some of the blocks to be outside of tolerance. The worst condition resulted in an average slope, as seen by the air sled, of 0.0015 inch in 10 inches. This possibility had been anticipated and was resolved by measuring the track, plotting the track profile for each air bearing, and including this profile in the test data package so that track slope effects could be included in the analysis. Operating procedures were revised to include track adjusting and profile recording procedures and sequences.

During the first series of test runs, the air bearings sometimes bottomed slightly when passing over the track joints. These hangups were sometimes too slight to notice by observation but were evident on the X-Y plotter graphs. The problem was primarily caused by thermal expansion affecting the beeswax at the track joints and allowing excessive air leakage. A procedure was adopted in which the track joints were checked before each test run and new beeswax was added and scraped flush where required. Figure 5-8 shows the checking of fairing with a dial indicator and the melting of beeswax into the joint.

This hanging-up tendency was aggravated in cases where one of the three air-bearing feet bore a disproportionate share of the weight of the test article. This condition was relieved by moving the air sled to the most favorable position under the test article relative to the center of gravity of the deploying set of panels. The aluminum test article was lifted manually to allow repositioning of the air sled.

During functional checkouts the deployment cables tended to ride over the fitting that attaches the cables to the motor drum. The cables sometimes, after a slack period, hopped off the drum entirely. Guide screws were added to the adapter adjacent to the motor drum to prevent this condition.

5.3.5 CALIBRATION

Calibration of system instrumentation was necessary to set equipment operating limits and to provide calibration information necessary for the reduction of raw test data.

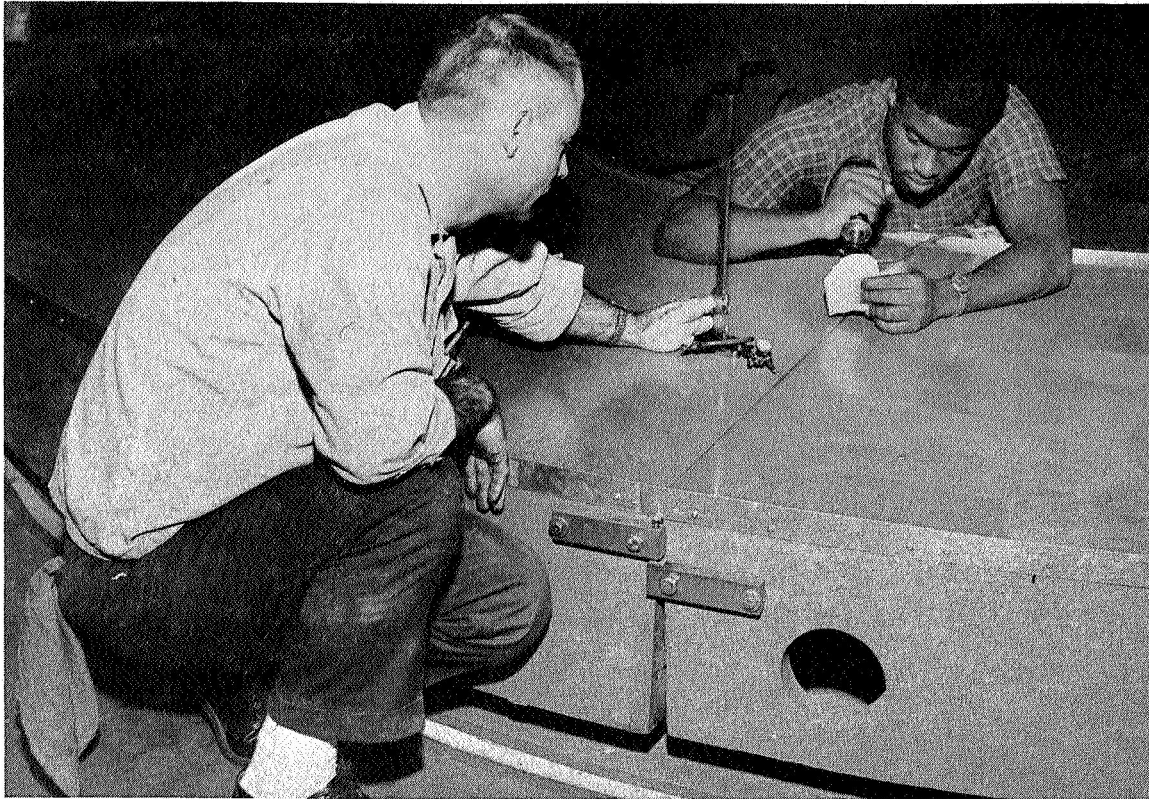


Figure 5-8: FAIRING AND WAXING AIR BEARING TRACK JOINTS

Except for cable tension, calibration of instrumentation met design requirements. Microswitches in the deployment motor circuit to prevent overloading were set to trip at a cable tension of 48 to 50 pounds. The deployment motor windup rate potentiometer was calibrated by plotting voltage output versus time data for three motor revolutions. The lengths of the all-metallic suspension lines that replaced the original dacron lines were set without difficulty.

The design requirements established for the cable tension measuring equipment were based on the assumption that cable tensions would reach 40 pounds during deployment, rather than the 6 to 8 pounds actually encountered. As a result of this and misalignment of the linear variable differential transformer (LVDT) sensing device, the accuracy of the cable tension measuring equipment was not within the originally established design limits in the low cable tension range. Redesign of this equipment is recommended before use in Phase III.

Hinge-angle calibration was accomplished by positioning a large protractor under each hingeline (see Figure 5-9). The voltage output of the hingeline potentiometer then was recorded at 10-degree intervals. Although the potentiometers demonstrated excellent sensitivity, two potentiometers failed mechanically and were replaced during test. Potentiometers of more rugged construction will be provided for future testing.

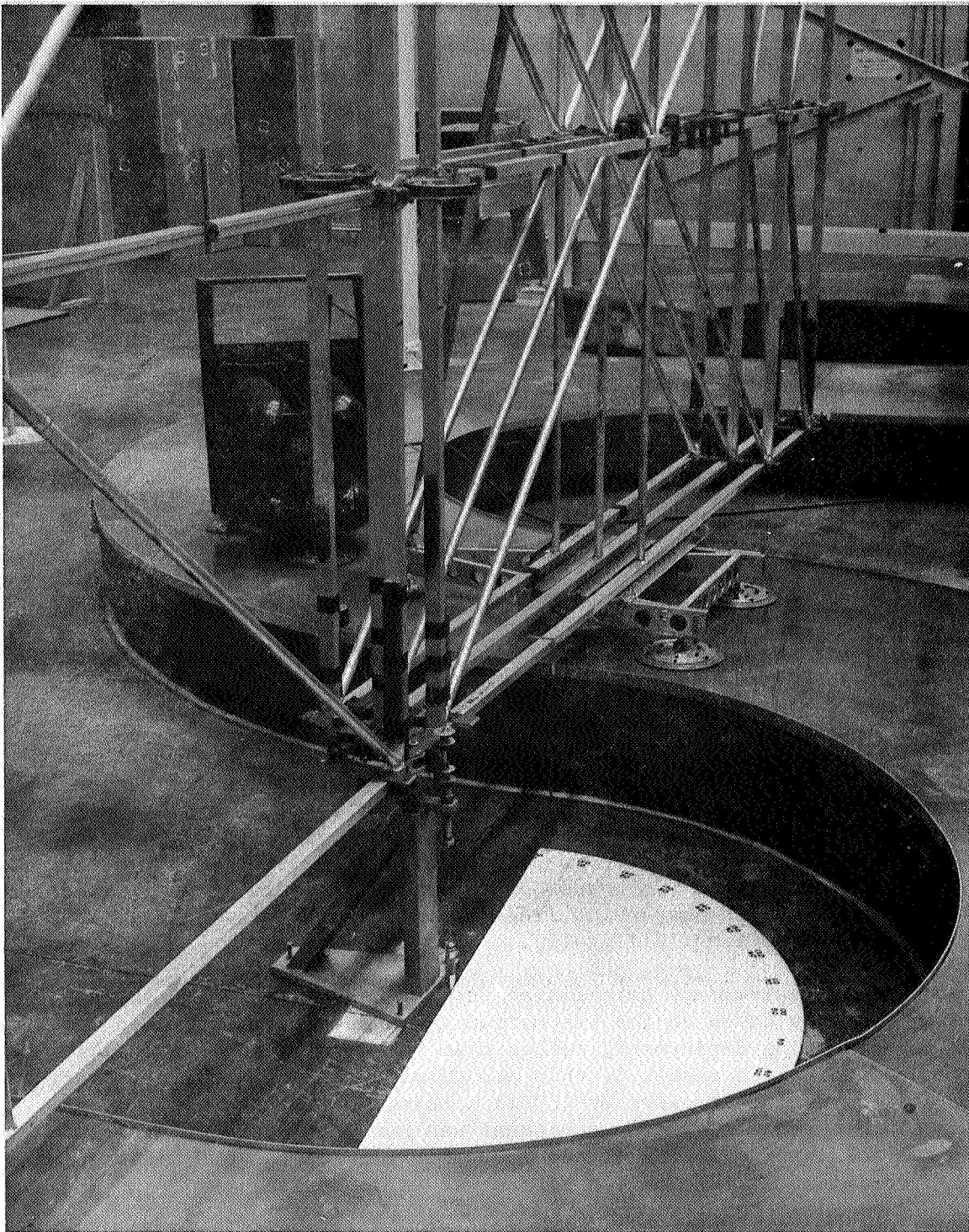


Figure 5-9: DEPLOYMENT ANGLE CALIBRATION

A series of closing-velocity calibrations was conducted to obtain a curve for each deployment swing that related the offset or starting position of the deploying stack to its latching or closing velocity. Using these curves, specific closing velocities could be estimated as a function of the distance from latching.

The deployment moment for the stack at the offset position was provided by the spring force of the suspension lines (see Figure 5-10). The closing velocity values were repeatable within $\pm 10\%$.

5.4 TEST CONDUCT AND OBSERVATION

5.4.1 TEST SETUP AND CONDUCT

All test objectives were satisfied by an intermittent test sequence in which the swing of each deploying set of subpanels was essentially tested separately.

Five types of tests were conducted during the test program. Although these tests involved intermittent rather than continuous deployment, all data required for the deployment analysis can be obtained in this manner. The five test types are described below.

- 1) Zero Initial Velocity Tests---The object of these tests was to obtain time-history and other data for a given deployment swing starting with zero velocity and zero cable tension. With the panels to be deployed at rest, deployment was initiated by turning on the deployment motor and allowing the panels to proceed through approximately 177 degrees. While not typical of flight behavior (except Swing 1), this test provided useful data for purposes of comparison and definition of parameters.
- 2) Controlled Starting Condition Tests---The object of these tests was to obtain time-history and other data for a given deployment swing as influenced by the closing velocity of the previous swing. The closing panels were tied by cord a given distance from closing so as to provide a given closing velocity. The cord was cut simultaneously with the start of the deployment motor and the panels allowed to close and continue on through 177 to 180 degrees of the following swing. This type of test simulates continuous deployment velocities.
- 3) Aerodynamic Tests---Aerodynamic tests were conducted identical to the zero initial velocity and controlled starting condition tests described above except that a plastic membrane was taped over the MTA-3 frame to simulate the aerodynamic drag of the Phase III test article. Comparison of the time-histories obtained for the test runs showed no consistent differences that definitely could be attributed to the aerodynamic membrane.

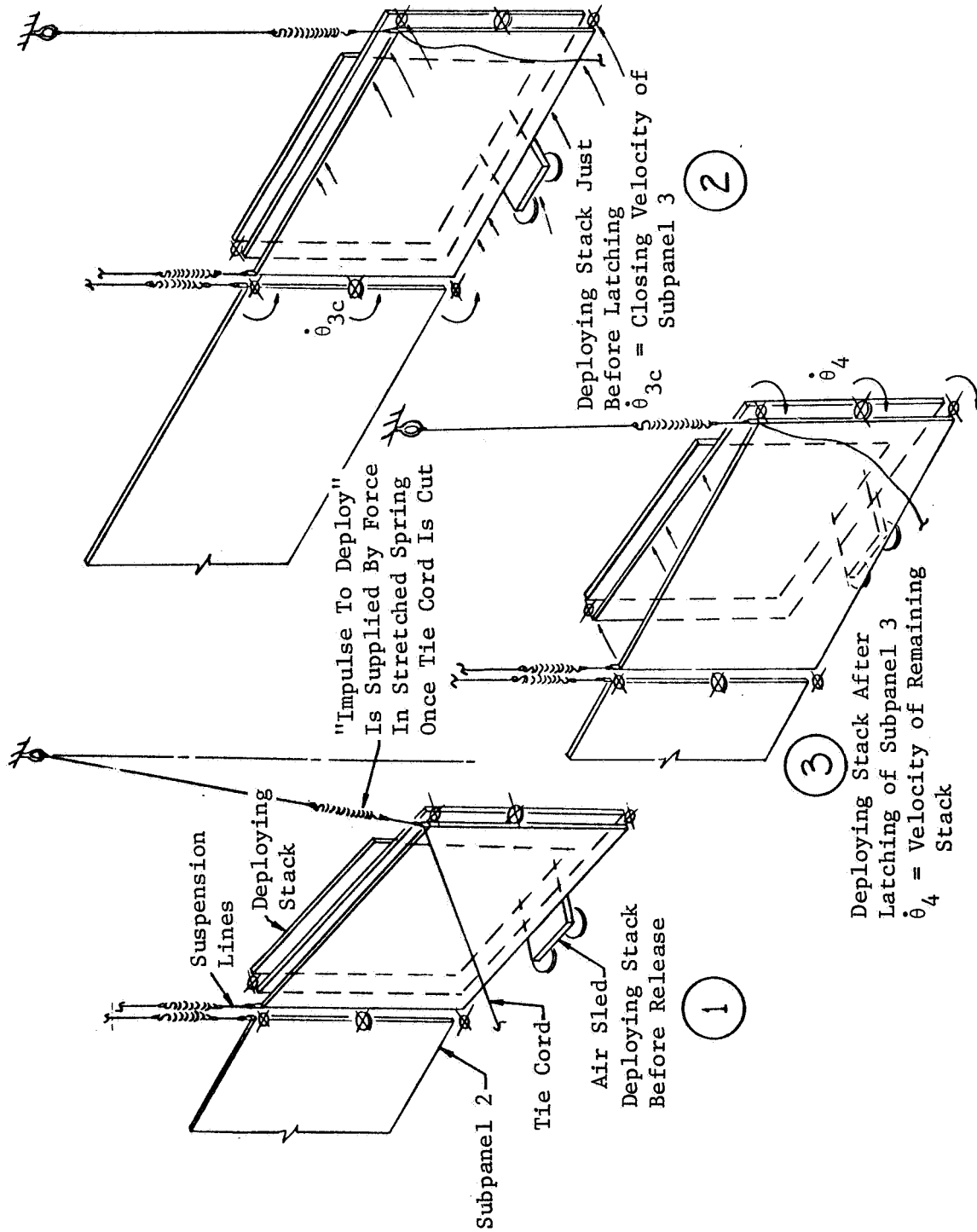


Figure 5-10: CLOSING VELOCITY CALIBRATION SEQUENCE

- 4) Sigma Resistance Tests---The object of these tests was to evaluate the sum of all resistances on each deployment swing. The resistances arose from such sources as hinge, latch link and GSE instrumentation frictions, effects of slight hills and valleys in the air-bearing track, and ambient air effects. The test was conducted by restraining the panels, introducing a specified tension in the deployment cable, releasing the panels, and measuring the resulting time-history. This data, and the known forces that act to produce it, can be used to calculate the unknown resistance forces.
- 5) Deflected Panel Tests---These tests were conducted to determine the effect on deployment of a presumed deflection of a panel due to differential heating of the Sunside and the darkside. A 0.28 deflection corresponding to a 100°F (a 50°F steady-state differential was actually shown by analysis) temperature differential was introduced into Subpanel 2 and zero-initial-velocity-type test run.

5.4.2 VALUE OF MTA-3 AS AN OPERATIONAL MOCKUP

The building and testing of MTA-3 provided valuable data concerning mechanism interferences not apparent in drawings and layouts.

The fabrication and assembly of MTA-3 revealed specific areas requiring closer tolerances and more refined tooling. The location of each of the four hinge plates at the ends of a main subpanel must be controlled relative to the adjacent hinge plates, the quadrant axis, and the subpanel structure. Misalignment of these hinge plates can cause the friction effects described in Section 5.3. Misalignment of the holes in the hinge plates cause interference with other mechanical parts during deployment. Misalignment of the hinge plates relative to the quadrant can induce deflection into the subpanel structure. The holes in the overcenter latch parts must be closely controlled to ensure that the latch will cross over center and lock the deployed subpanel at the 180-degree position.

The assembly of MTA-3 is shown in Figure 5-11. The subpanels were stacked on a surface plate. The lower edges then were aligned to provide even seating on the air sled, and the quadrants installed. The hinge plates were located dimensionally by means of positioning blocks using the quadrant axis as a reference. While suitable for MTA-3 assembly, this minimum type of tooling will not provide the alignment accuracy required on MTA-4, and a similar tool with permanent locating blocks is planned for Phase III.

Several areas of mechanical interference were discovered during the assembly of MTA-3. These were generally of a minor nature and were corrected by providing reliefs or adjusting the location of the interfering parts. These items were noted and corrections were incorporated into the MTA-4 design.

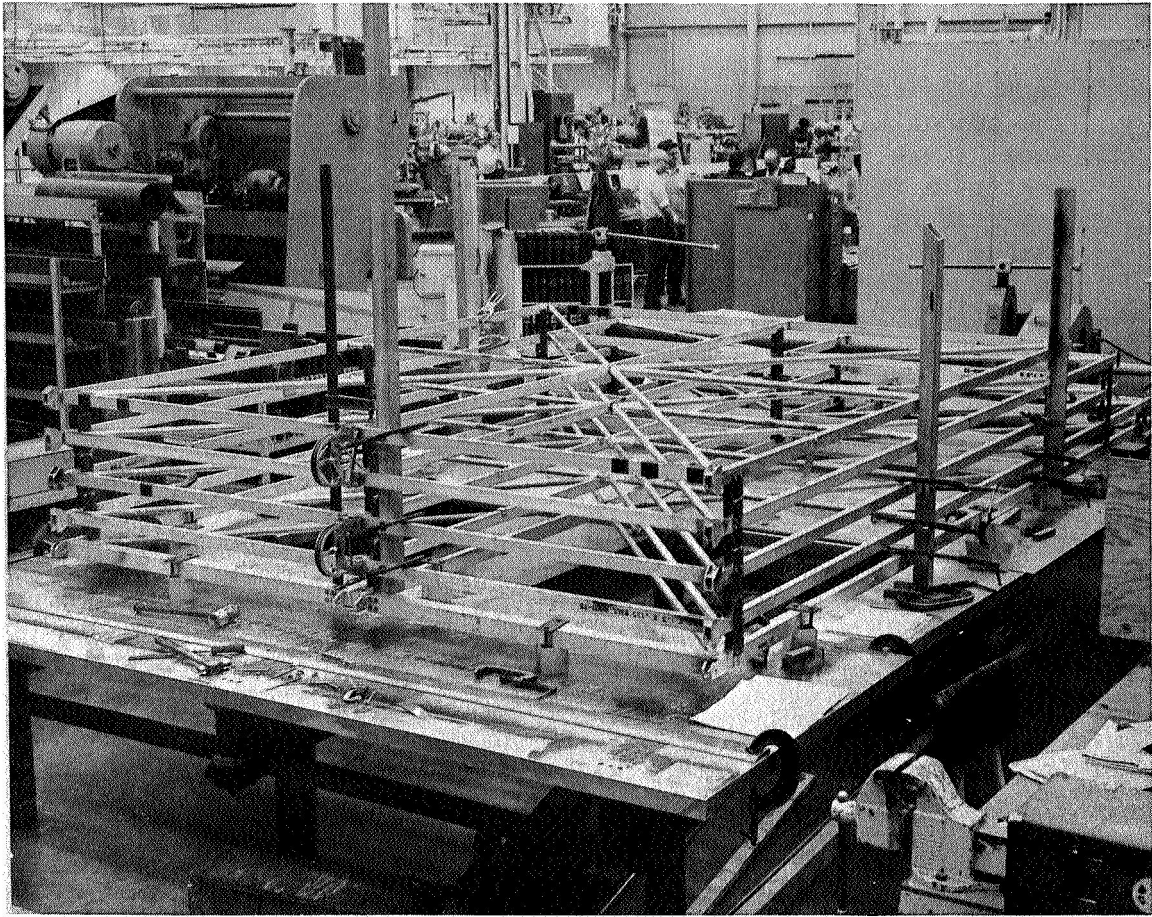


Figure 5-11: ASSEMBLY OF MTA-3

Less obvious areas of interference and mechanical irregularities were discovered during the checkout and testing of MTA-3. For example, the ends of the main hinge bolts interfered with the disks retaining the retarding springs as certain angles of deployment were reached. This was corrected by using shorter bolts and omitting the cotter pins. The release rod holding Subpanel 3 in the stowed position against Subpanel 2 did not withdraw properly because of friction resulting from inherent warpage of the aluminum subpanel structure. The MTA-4 design of this mechanism is being revised to provide more reliable operation.

5.4.3 BEHAVIOR OF MECHANISMS

Closing momenta in excess of design limits were incurred without damage.

The main subpanel hinge and latch mechanisms were designed to withstand closing velocities as high as 0.035 rad/sec. The loads in the mechanisms

from such a closing velocity would be highest at the Subpanel 1 to 2 , hinge joint because the highest rotational inertias and the shortest latch couplings are experienced at the close of Swing 2.

During pretest calibrations, velocities higher than 0.035 rad/sec were inadvertently induced at the closing of Swings 3 and 4. Swing 3, which included the inertias of Subpanels 3, 4, and 5, and the air sled, closed once at 0.088 rad/sec.

The angular momentum generated by this closing was calculated and found to be about 7% higher than the momentum that would be generated if Swing 2 closed in space flight at 0.035 rad/sec.

Because the coupling distance between the hinge bolt and the latch is 3.7 inches at the Subpanel 2 to 3 joint, but only 3.2 inches at the Subpanel 1 to 2 joint, the design limit load for the hinge-latch coupling was probably not quite reached. It should be noted further that EI characteristics for the aluminum MTA-3 subpanels are different than for MTA-4 beryllium subpanels.

While there was no test requirement to close each deployment swing at the limit velocity, the fact that higher closings were experienced is felt to have qualified the three outboard hinge joints for closings in excess of 0.035 rad/sec.

At closing velocities below 0.025 rad/sec, the diagonal strut between the adapter and Subpanel 1 did not lock in the fully deployed position. The main strut assembly straightened sufficiently to hold Subpanel 1 rigidly in the fully deployed position. However, the bracing linkage that locks the strut in place stopped about 5 degrees short of its intended 180-degree position, as shown in Figure 5-12. A second spring was added to the bracing linkage, which then locked properly at lower closing velocities.

Variations from the designed lengths of the various diagonal strut members also were noticed. These variations caused the diagonal strut to lock the Subpanel 1 at a deployed position 91.5 degrees from the face of the adapter and to hold the Subpanel 1 1.5 degrees from the face of the adapter in the stowed position. Since the face of the adapter represents a basic spacecraft plane, this condition must be corrected in the final MTA-4 design. However, for Phase II test purposes the adapter was adjusted 1.5 degrees off from the basic spacecraft plane.

The overcenter latches on the upper spars only, at Hinge Lines 2 through 5, sometimes did not cross over center at lower closing velocities. The latches on the lower spars crossed over properly under all conditions.

Five possible causes are suggested:

- 1) Warpage of the aluminum panels could cause an angular lag of the upper spar behind the lower spar.

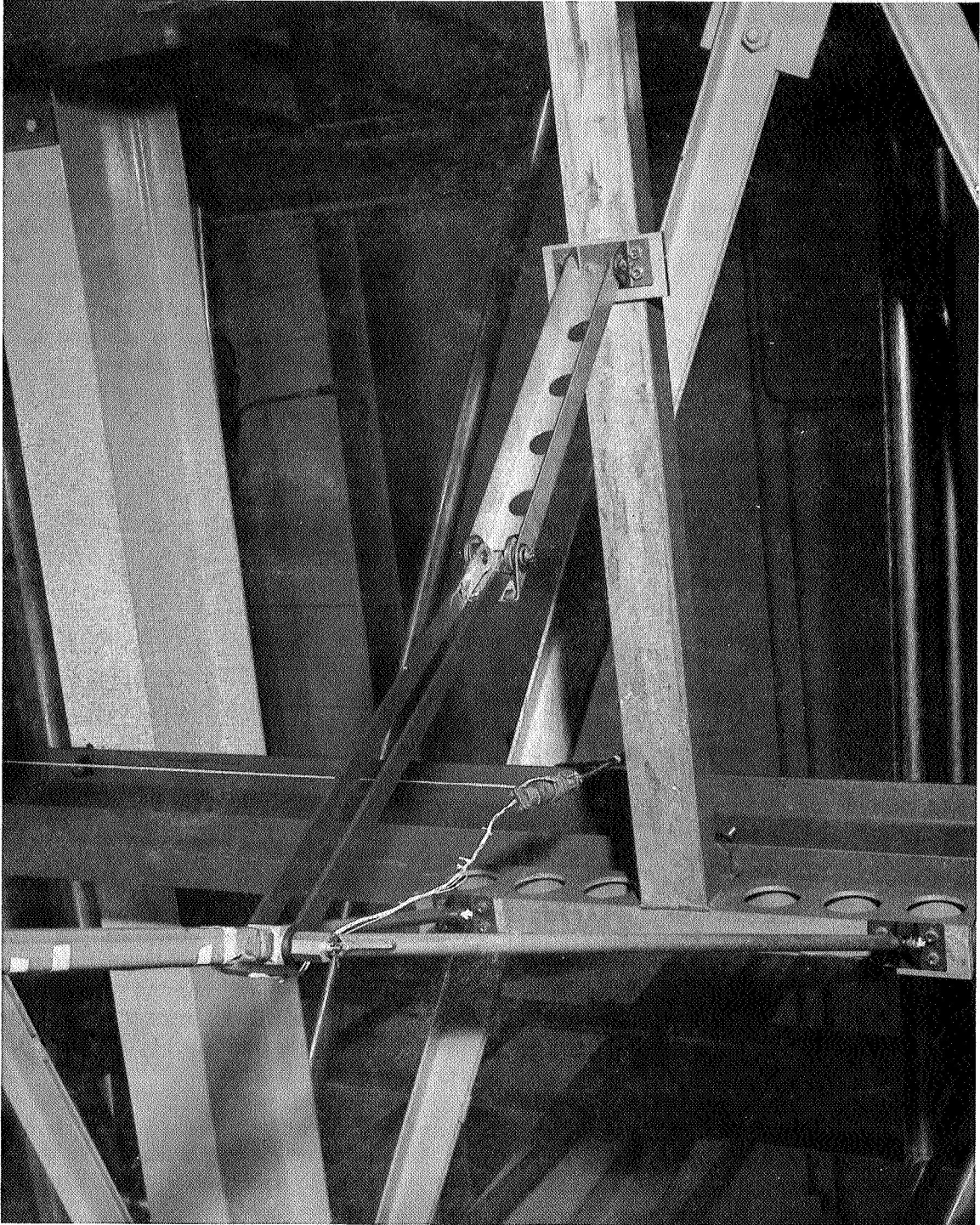


Figure 5-12: DIAGONAL STRUT

- 2) The inertia effect of the air sled is greater at the lower spar, thus providing a higher closing momentum.
- 3) Corresponding upper and lower overcenter links may not have been drilled in identical positions relative to deployment of the panel.
- 4) The latch spring force may be insufficient.
- 5) Friction due to misalignment of the pivot bolts could inhibit latching.

This condition did not appear to affect angular time-history results so no fix was made during testing. However, the force of the closing spring has been increased and will be tested in Phase III.

5.4.4 BEHAVIOR OF THE DEPLOYMENT MOTOR AND CABLE DRIVE SYSTEM

The deployment motor and cable drive system provided suitable power for deployment with a wide safety margin.

The test article repeatedly deployed itself through each swing using the deployment motor as the only significant energy source. Counting only significant test runs, where no mechanical irregularities were observed, the equivalent of six complete deployments was performed.

The deployment motor operated satisfactorily throughout all testing. A total of approximately 7.5 hours of motor-on time (including reverse) was logged.

Tensions in the No. 1, or primary, deployment cable, for all significant test runs, did not exceed 6.5 pounds. Within this range there was no tension in the No. 2 (or secondary) cable as indicated at the tension readout linkage on the adapter. Some tension, less than 1 pound, could have been present at the quadrant. The lack of tension in the No. 2 cable is the result of the intentional adjustments of the cable lengths so that the No. 1 cable was about 7 pounds more taut.

Closing rates for each deployment swing generally were close to the theoretical stiff-cable rate determined by the constant windup of the motor drum. The maximum and minimum closure rates were 0.025 and 0.007 rad/sec, respectively. Most closings were in the range of 0.010 to 0.020 rad/sec. No closings approaching the limit 0.035 rad/sec were observed.

Unexpected deformations of the deployment cables incurred during testing affected deployment behavior. As testing proceeded, the deployment cables were repeatedly pulled over the small-diameter rollers adjacent to the quadrants. This caused a series of wave-shaped deformations as shown in Figure 5-13. This condition influenced deployment behavior in two ways:

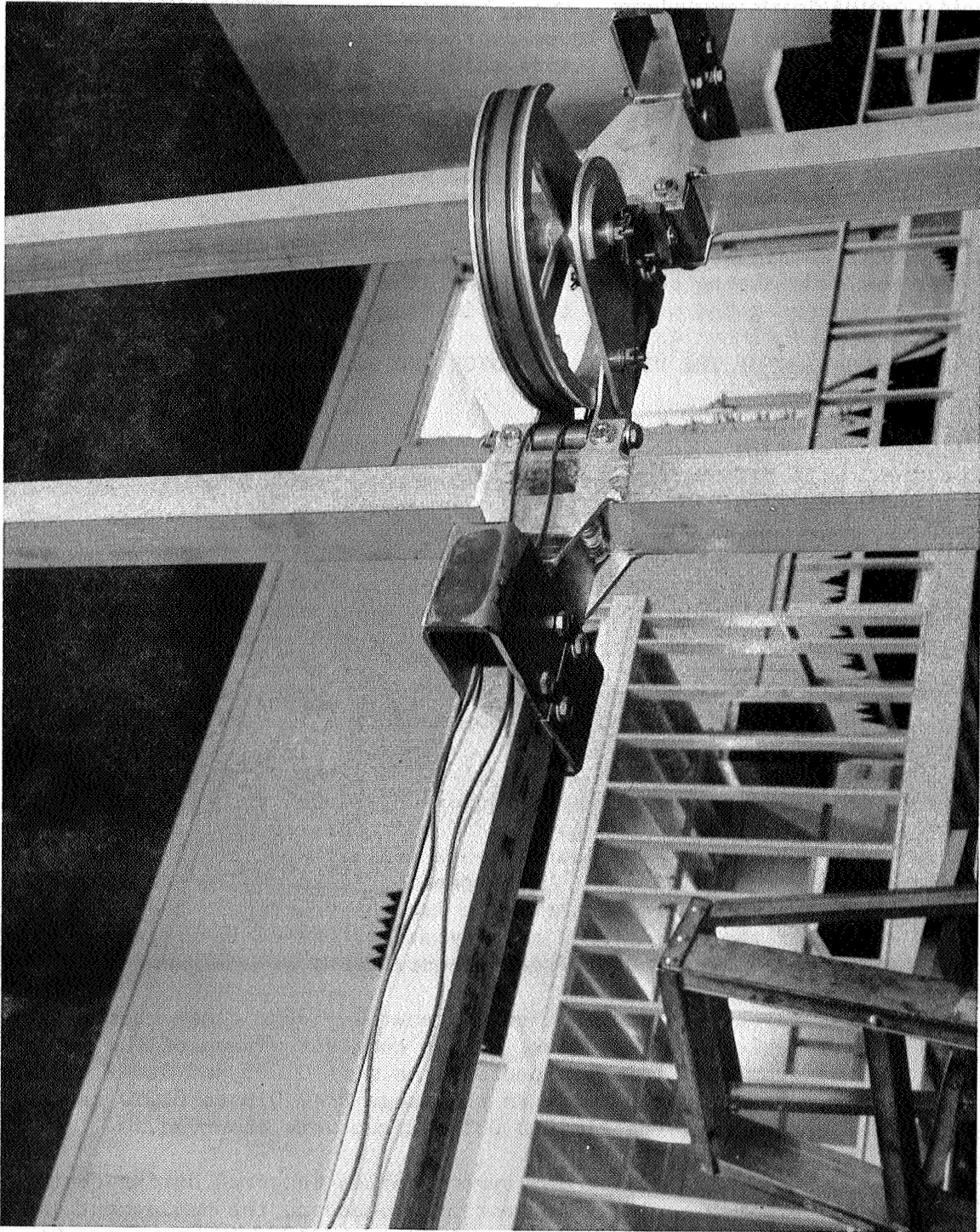


Figure 5-13: DEPLOYMENT CABLE DEFORMATIONS

- 1) The spring constant of the cables was changed.
- 2) Additional friction was induced between the cables and the cable guides. Starting from a slack cable condition, the deployment motor had to pull sufficient tension into the cables to straighten the deformations and allow passage of the cables through the guides. As a result, the tension indicated at the sensing linkage was not the same as the tension experienced out at the deploying quadrant.

5.5 TEST ANALYSIS AND RESULTS

In the following paragraphs, the approach to the analysis of the MTA-3 zero-g simulated deployments is described, and typical comparisons between initial calculations and measured deployments are shown. The approach to resolution of shape discrepancies is described and improvement necessary in Phase III discussed. Details of these analyses are not included. Results of specific tests not amenable to analytic comparisons are presented.

5.5.1 ANALYTICAL APPROACH

Time-history analyses have established the proper combinations of parameters required for successful deployment.

The time-history motions for the main panels are calculated with the assumption that there is no coupling with the transient oscillation motion. The folded panel sets are considered rigid. The differential equation used for analysis is given and discussed in detail in D2-113355-6, Page 3-15.

Initially, analyses were conducted for space and ground deployment using the best available data, and with variations between expected limits of selected parameters. These analyses were used to establish conservative requirements for design levels of mechanisms and structures and to define the characteristic response due to the parameters.

With assumption of minimized effect from GSE, the time-history of the MTA-3 with Visqueen membrane attached should be closely equivalent to MTA-4 in air, and the motion of MTA-3 with bare frames should be equivalent to MTA-4 in space.

The GSE fixture design for the MTA-3 test precluded an evaluation of transient loads and motions. Deletion of the overhead bearings required support with overhead cables fixed to the fixture that introduced spring forces not present in the MTA-4 test configuration. With stiffnesses of the aluminum MTA-3 structure differing in magnitude and distribution, the dynamic response could be expected to differ from that for the MTA-4 configuration.

MTA-3 tests were selected to verify the parameter values used for design, or to define actual values where a range of values was used in analysis.

5.5.2 RESULTS OF DEPLOYMENT CALCULATIONS WITH NOMINAL PARAMETERS

Significant differences exist between measured and calculated deployment shapes.

Calculations were made for all tested deployments using the measured values of parameters (nominal) with zero friction, and were repeated with the friction values obtained by evaluation of the sigma resistance tests. The values used are tabulated in Table V-2. Comparison with the measured curves plotted from the tabulated data output of the computer was made. Examples of the measured and calculated curves are shown by Figures 5-14 and 5-15. In most cases, the initial portion of the curve was reasonably predicted in general characteristics by the "nominal" analysis without friction for both the "initial values" condition and for zero initial values condition. During approximately the first 60 degrees of deployment, the general characteristics of the curves agreed. During the latter part of the deployment, the calculated curves oscillated with periods of cable tension and slack cable, whereas the measured deployments always indicated tension and had only small oscillations about a nearly straight line.

Analysis with additions of friction torques from the sigma resistance tests showed an initial slack tension interval of much shorter duration than the zero friction calculations, and somewhat shorter than measured deployments. During the latter portion of deployment, the frequency of oscillation was greater than for the no-friction calculations and the amplitude of oscillation was nearly the same. As could be expected, the latter part of the curves tended toward a condition where the cable was always in tension.

Discussion of differences between calculated results and test results follows in Sections 5.5.3 through 5.5.7.

5.5.3 APPROACH TO RESOLUTION OF MEASURED AND CALCULATED DEPLOYMENT SHAPE DISCREPANCIES

The differences between the predicted and measured nominal deployed shapes can be resolved by adjustment of the friction parameters.

Differences in the measured and calculated time-histories are an indication that more parameters are involved in the actual deployment than was considered analytically or that the magnitudes and distribution of the forces considered were not correct. To evaluate and arrive at conclusions for the MTA-4 tests, the causes of the differences must be determined and the probable source, MTA-3 or GSE, be defined. The indicated areas of investigation from comparison of the test and calculated curves and from the test log were: variations in the initial tensions and/or

Table V-2: PARAMETER VALUES FOR NOMINAL AND SIGMA-RESISTANCE DEPLOYMENT CALCULATIONS

Panel	Mass Moment of Inertia, I (Slinch-sq in.)	Retarding Spring Rate, k (in.-lb/rad)	Opening Moment (in.-lb)	Cable Spring Rate			Sigma Resistance	
				K_1 (lb/in.)	K_2 (lb/in.)	Point of Rate Change (in.)	Deployment Angle (deg)	Resistance Moment (in.-lb)
1	10195 11297 (Aero)	2.60	0.62	3.3	69.7	0.24	0-11.3 11.3-47.2	6.4 3.1
2	12694 13853 (Aero)	2.30	0.46	3.3	69.7	0.24	0-13.1 13.1-64.0 90.0-103.0 103.0-134.7	0.9 7.7 -3.9 12.0
3	5371 6039 (Aero)	2.82	-0.06	15.0	53.9	0.22	0-14.2 14.2-75.9 75.9-90.1 90.1-136.4	-1.0 8.1 0.1 9.5
4	4197 4908 (Aero)	2.87	-0.11	4.3	41.7	0.86	0-29.3 29.3-108.0 62.5-91.8 91.8-153.0	6.2 6.7 7.4 8.5
5	1539 1846 (Aero)	2.55	0.21	6.3	33.5	1.00	0-27.9 27.9-49.0 90.0-117.9 117.9-135.0	13.9 9.5 8.5 31.0

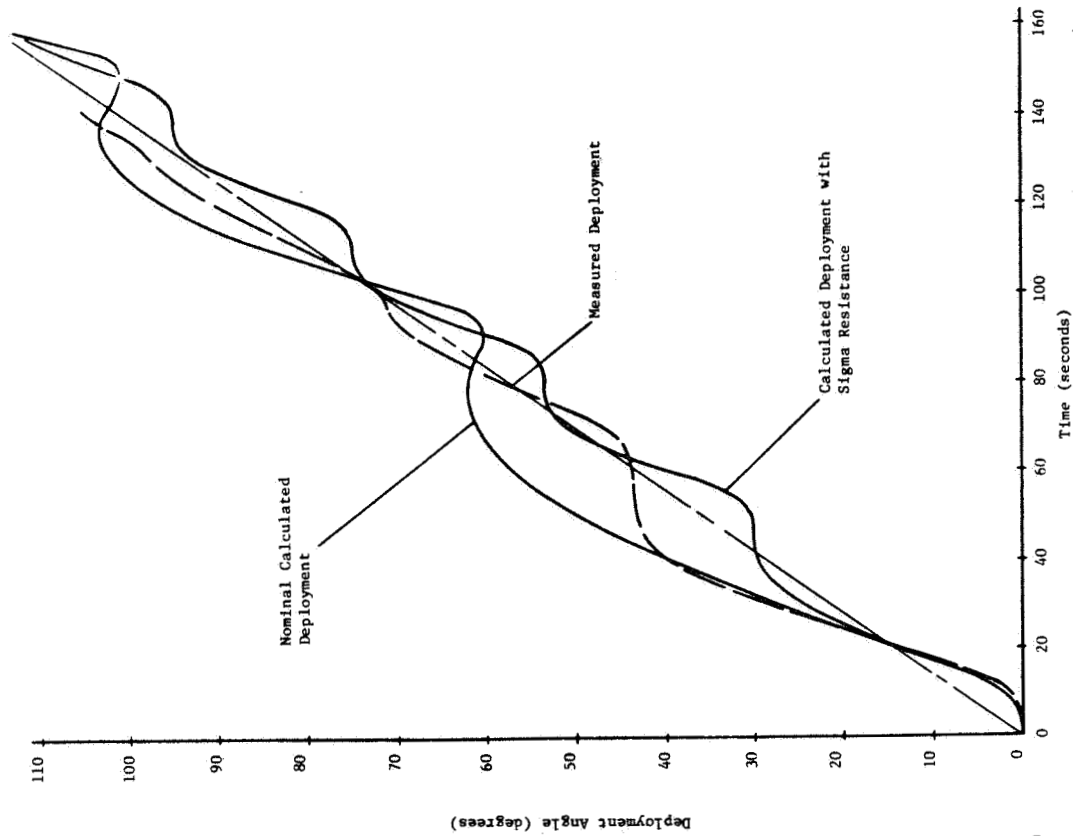


Figure 5-14: COMPARISON OF PANEL II, RUN 4, TYPE 1, WITH NOMINAL AND SIGMA RESISTANCE CALCULATED DEPLOYMENT

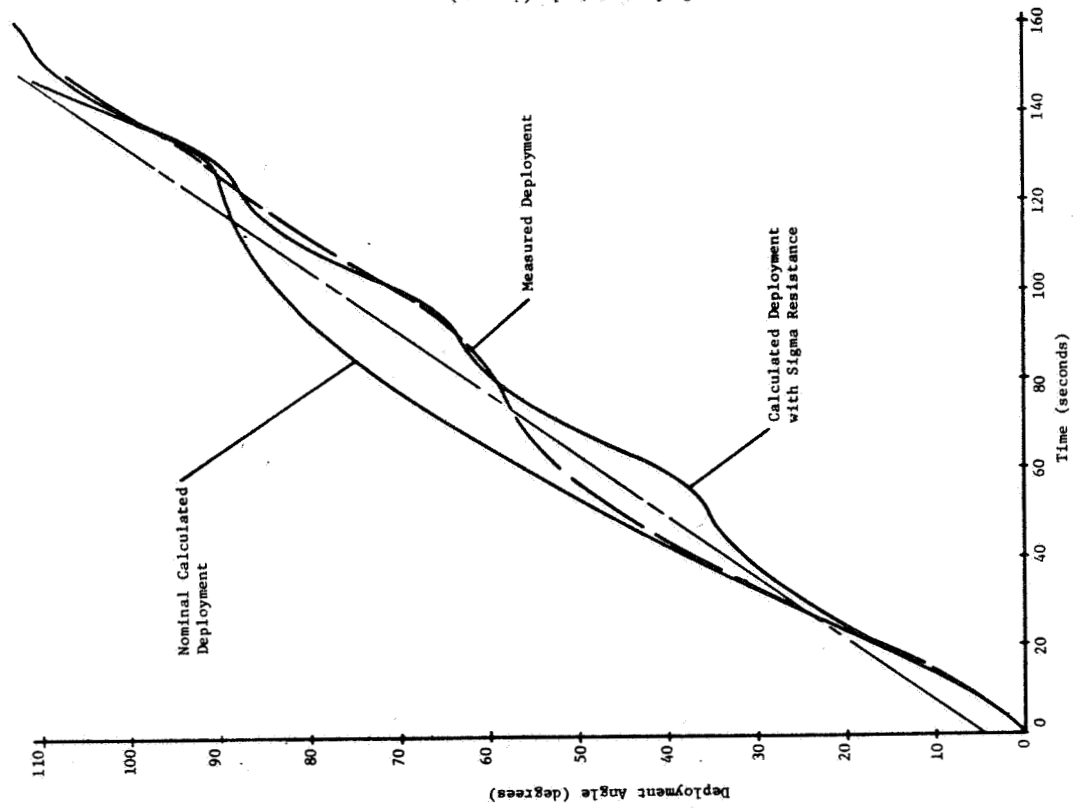


Figure 5-15: COMPARISON OF PANEL III, RUN 5 TYPE 2, WITH NOMINAL AND SIGMA RESISTANCE CALCULATED DEPLOYMENT.

velocity for the Type 1 (initial velocity) curves, possible incomplete elimination of the hangups attendant to the fairing problem, and inclusion of friction values different from those indicated from the static and sigma resistance tests.

The possibility existed that the nonlinear cable stiffness values measured after the deployment tests were not constant during the recorded tests. The cable deformations (evidently responsible for the double-slope, load-deflection curve) could have changed from repetitive loadings. Comparison of experimental and calculated results during the last 60 degrees of some deployments seems to indicate that a higher motor speed could have been used. These, however, were likely to be small changes and would not explain the major discrepancies.

Four regions of the deployment became apparent as the calculated and measured shapes were compared, as shown by Figure 5-16. During the first period, reasonable agreement can be shown with some nominal calculation curves, indicated as Region I. During the next period, the measured curve indicates that the coasting system has lost energy not accounted for by the analysis. Region III is a transition region to a deployment time-history, characteristically almost a straight line and showing nearly uniform cable tension.

For Type 1 tests, the match in Region I could be improved by making small adjustments to the initial cable tension and/or initial velocity. The adjustments, where made, were small and within the measurement error.

For Type 2 (zero initial velocity) tests, the match in Region I was more difficult to adjust since the position and time of initial motion were not well defined.

The importance of matching the first loop of the deployment is self-evident since it establishes that the cable stiffness, retarding and negator spring rates, and panel inertias are known with acceptable accuracy. This means that deviation from the calculated deployment has to be accounted for from array frictions and GSE inputs. Further analysis can establish the magnitude and position of frictional forces that, when applied to the nominal deployment, result in acceptable matching of the actual time deployment history.

In the second region, two methods were used to obtain initial estimates of the magnitude and distribution of the change in energy apparent in the actual deployments. The magnitude of the deploying moments was obtained from the actual deployment time-history by taking graphic second derivatives of the actual time-history. Comparison of these accelerations and the calculated values defines the level and distribution of moments to be added to the analytical calculation. The second method used was to relate the difference in the change in energy of the actual deployment and the analytical deployment over a given displacement to an incremental constant torque over the given displacement. The deployment shape in the second region thus was matched using these techniques, together with small arbitrary adjustments in the friction levels and/or distribution as necessary.

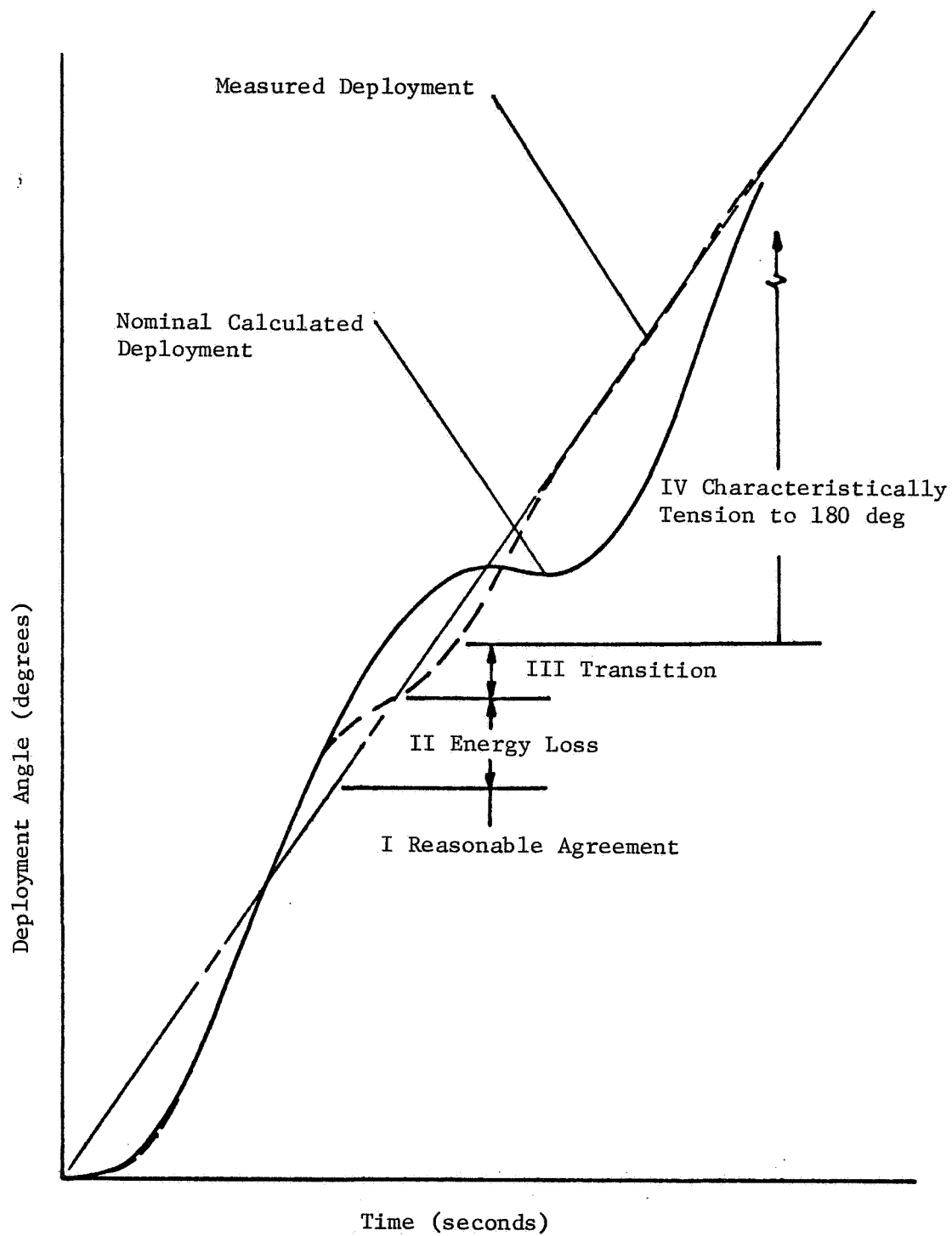


Figure 5-16: CHARACTERISTIC DEPLOYMENT REGIONS

The fourth region had an obvious, but statistically unlikely, possible solution. Previous analyses, where the initial velocity of the deployment was nearly the same as the nominal (rigid cable) velocity, had shown a time-history with very small oscillations essentially parallel to the nominal line. If the transition region ended with the same velocity as the nominal, and if the cable tension torque was balanced by the retarding spring and by a friction torque, the motion would be initially parallel to the nominal curve and only small oscillations would be expected. The coincidence of balanced forces at the equal-velocity condition for three of the five deployment swings was the deterrent to this approach. However, if the motion were adequately similar with less than a complete balance of torques, an acceptable indication of the reason for the characteristic motion would be defined.

5.5.4 ANALYSIS OF AERODYNAMIC TEST DATA

The effects of ambient air on deployment behavior were of limited significance and were obscured by other variables.

The analysis for the effect of air conducted early in the program showed an appreciable change in the deployment characteristic because of estimated drag forces. During Phase II, the effect of air inertia was included, using as a basis for the analysis the data from a group of experimental evaluations from a 1941 NACA technical report. The results, including both inertia and damping for the Swing 1 test condition, tended to alleviate the differences as it did for Swings 2 and 3. The relative increase in inertia for Swings 4 and 5 due to air is much greater than for 1, 2, and 3. The result is that during the latter portion of deployment the inertia effect exceeds the air-drag effect and a considerable difference appears in the opposite direction. These effects were not seen in measured deployments since the indicated effect is large only during that portion where perturbation forces were large.

5.5.5 INITIAL VELOCITY MEASUREMENT

Qualitative verification of the ratio of initial velocity to the closing velocity of the previous swing was determined from a sequential deployment demonstration.

For the initial velocity of the second and fourth swings, analysis predicts an initial velocity between 40 and 50% of the final velocity of Swings 1 and 3. For Swings 3 and 5, the ratio is near 90%.

A sequential deployment demonstration was set up for Swings 1 and 2 with some adjustment of the sled position under the panels, and with the added weights on Panel 2 shifted to new positions. The object of the demonstration was to deploy continuously through closing of Panel 1 and initial motion of Panel 2, which required that the outboard end of Panel 1 should not be supported by the overhead cable. Figure 5-17 is a copy of that record and provides a qualitative verification of the analysis.

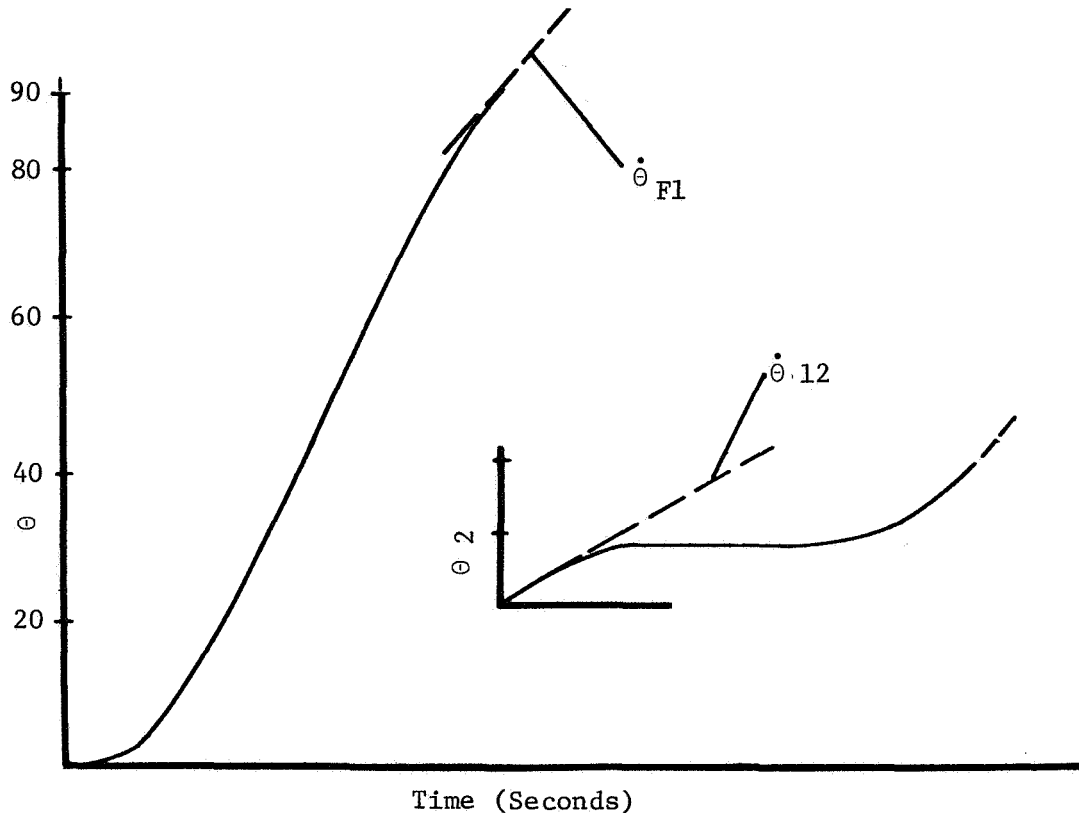


Figure 5-17: SEQUENTIAL DEPLOYMENT DEMONSTRATION

5.5.6 DIAGONAL STRUT LOADING

Deployment loads were within the strength capability of the diagonal strut.

Strain gage measurements of the transient loads in the diagonal strut show maximum load occurring at the conclusion of the first swing as predicted by analysis (Reference D2-113355-4, Pages 6 to 82). The peak initial value (tension) was 130 pounds and the peak compression value was 110 pounds.

The critical Euler load for the strut is 262 pounds, but is calculated for a closing velocity about twice the value used experimentally. An increase in moment of inertia for the first panel is responsible for some increase in the measured load as compared to the predicted value.

The actual margin for the dynamic case being considered is much greater than is indicated by the Euler load capability. After the Euler load is reached, the strut will absorb a large amount of kinetic energy before failure stresses are reached. The measured stress in this test was only 1,700 lb/sq in.

5.5.7 APPLICATION OF PHASE II RESULTS TO PHASE III TESTING AND ANALYSIS

Further Phase III ground-deployment testing, using MTA-4, is necessary to observe performance degradation after environmental tests and to investigate the effects of transients.

The MTA-3 tests utilized overhead cables for support of the deployed panels and was not designed to the stiffness characteristics of the beryllium MTA-4 array. Consequently, the transient loads that occur when each panel locks at the end of its deployment is not modeled and motions and stresses are not those of the beryllium structure.

To ensure that the array will deploy successfully in space, it must be exposed to the environments it will experience before deployment and to the actual loads it will experience during deployment. Phase III provides for each environmental test to be followed by deployment with overhead air bearings.

The results of analysis of the MTA-3 test indicated several areas in the GSE deployment fixture and in the mechanism design and construction where improvements are needed to perform an adequate test with MTA-4. Some additional test procedures are needed that will better define and separate the effects of track slope and array friction. As a result of the test a program of calculated solutions for the continuous deployment of all five swings, with primary attention to the variation in level and distribution of friction forces, is indicated as necessary.

5.6 CONCLUSIONS

5.6.1 CONCLUSIONS CONCERNING MECHANISMS

The fabrication, installation, and testing of MTA-3 provided valuable data for refinement of mechanisms design.

The construction of the MTA-3 test article provided the engineering project with a fully operational mockup. The use of flight prototype mechanisms provided a measure of their operating characteristics under a simulated zero-g environment.

The deployment drive concept has proved to be a sound method of developing main panel opening hinge moments. The motor at no time approached the upper output limit (60-pound cable pull). The Monel cable proved to be too stiff to operate around the guide rollers. Permanent deformations resulted and, although the cable strength was not affected, the negator spring was incapable of containing the deformed cable in the quadrant grooves under zero cable load.

The deformations also proved troublesome in the dynamic analysis because the spring rate changed significantly during deployment. It is concluded that a cable change is necessary.

A change to a high-strength flat tape (perhaps Elgiloy) would result in about a 4-pound weight saving and eliminate the variable spring rate problem. The Monel cable still would be used from the motor drum through the spacecraft and to the darkside of Main Subpanel 1. The transition to flat tape would be located 50 inches outboard of the Panel 1 quadrant, and would provide a connection of both the primary and backup cable to a single flat tape. The tape method provides the unique opportunity of fabricating the exact spring required by reducing the tape width over a selected length. A 0.003- by 0.50-inch tape will not take a permanent set moving around the rollers.

Although the diagonal strut met the program functional objectives, two design problem areas were uncovered. The latch strut deployment spring proved to be too weak to perform its design function. The locking washers that secure the latch strut in the extended position were bonded to the strut links with RTV-630 adhesive. During the test program, one washer broke its bond and was rebonded with BMS 5-29, Type II, adhesive. The drawing is being revised to change the bond material.

Diagonal strut loads were larger than expected, but were within the design capability of the strut.

Main panel hinge sets for the 2-3 and 4-5 panels had an interference as the 2-3 latch reached its maximum travel (see Figure 5-18). The hinge assemblies were reversed on MTA-3 so that testing could continue, and the drawing has been revised to reflect the change. A compatible auxiliary shear pin retraction method has been designed.

The Subpanels 5A and 5B release brackets would not operate due to a physical interference with both the quadrant and the Geneva operator bolt head (see Figure 5-19). Drawings have been revised to reflect the change.

The outboard panel stack release rods at the 2-3, 3-4, and 4-5 joints had a tendency to hang up on bolt ends and nuts of the negator mounting brackets. Bolts were reversed, as shown in Figure 5-20, to eliminate the interference.

The Geneva guard interfered with the internal radius of the quadrant at the faying surface and required rework.

The Subpanel 1-to-spacecraft quadrant has a cam between the hinge fitting and quadrant, which was installed with round head rivets. Rivet heads interfered with the hinge fitting and were removed and countersunk.

The milled head bolts of the diagonal strut lock brace interfered with the main strut tube in the stowed position. The heads were ground down on MTA-3. The drawing has been changed to use countersunk bolts to ensure adequate clearance.

It was found during testing that the bell crank on the Panels 3 and 5 center-spar releases, the Panel 5 auxiliary panel releases, and the Geneva operators did not have an adequate aspect ratio (length to diameter) at the mounting shaft. This resulted in excessive wobble during operation. The design has been changed to overcome this design deficiency.

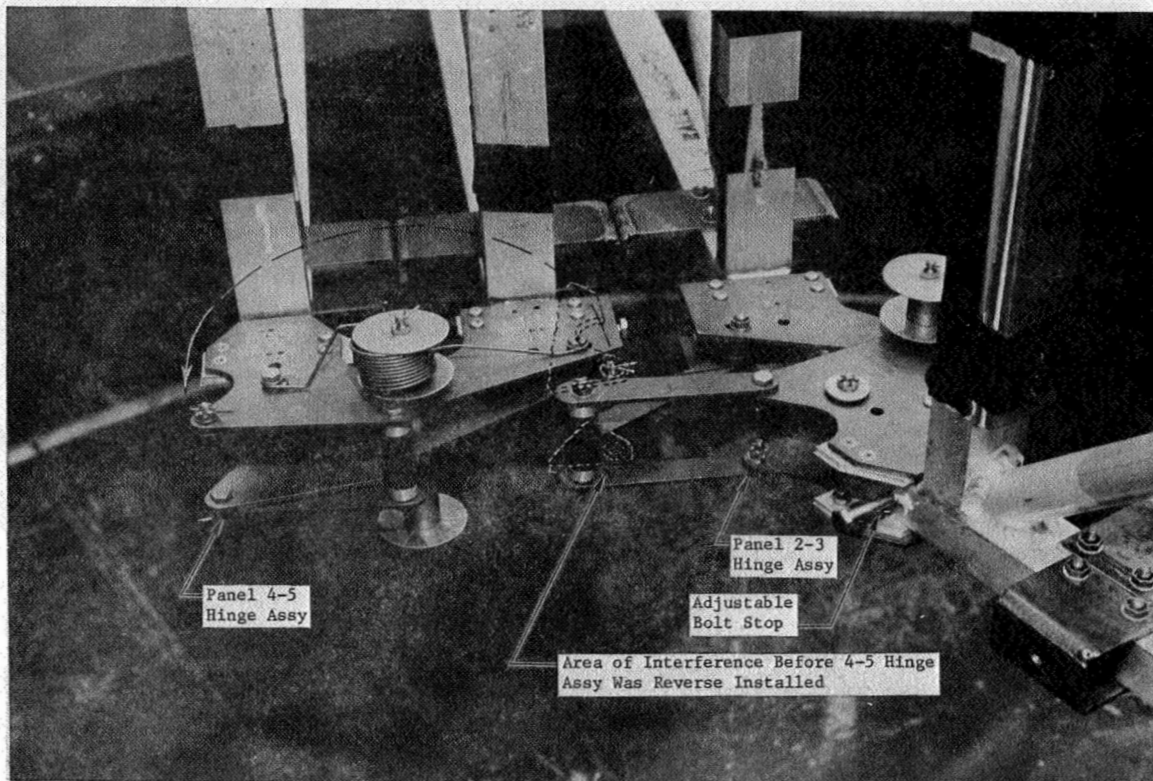


Figure 5-18: MAIN HINGE DEPLOYMENT INTERFERENCE

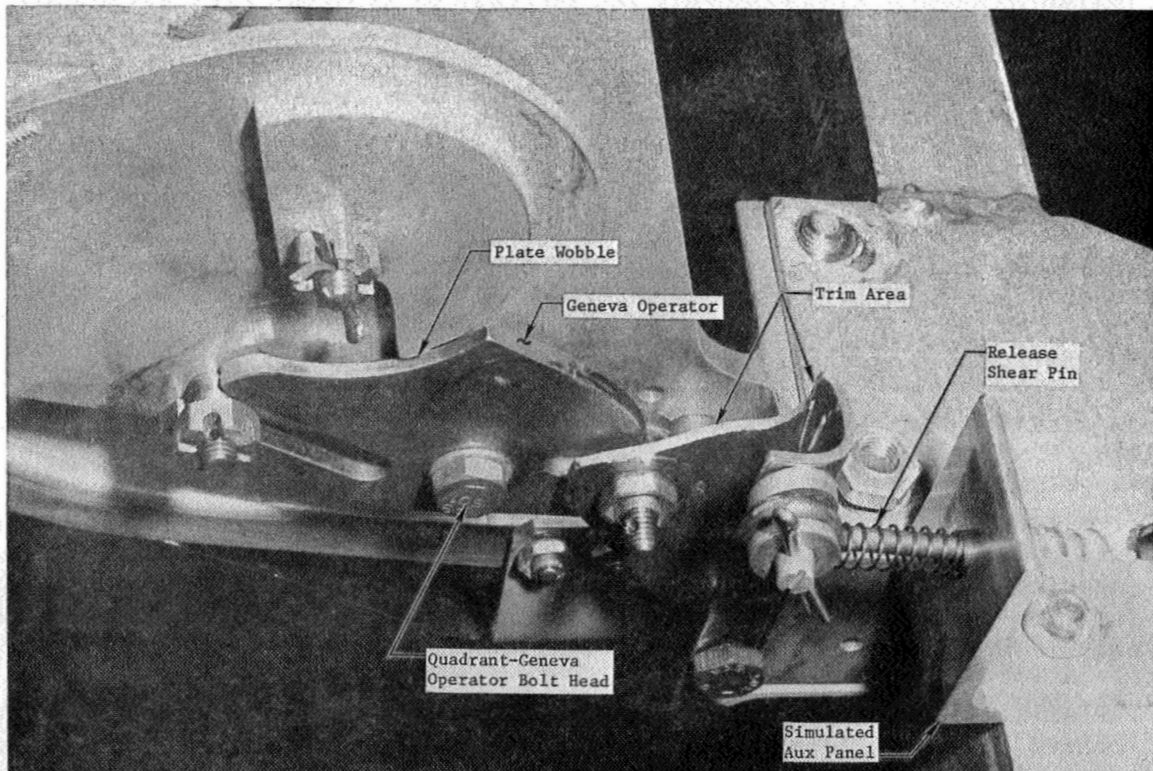


Figure 5-19: PANEL 5A & 5B RELEASE BRACKET MODIFICATION

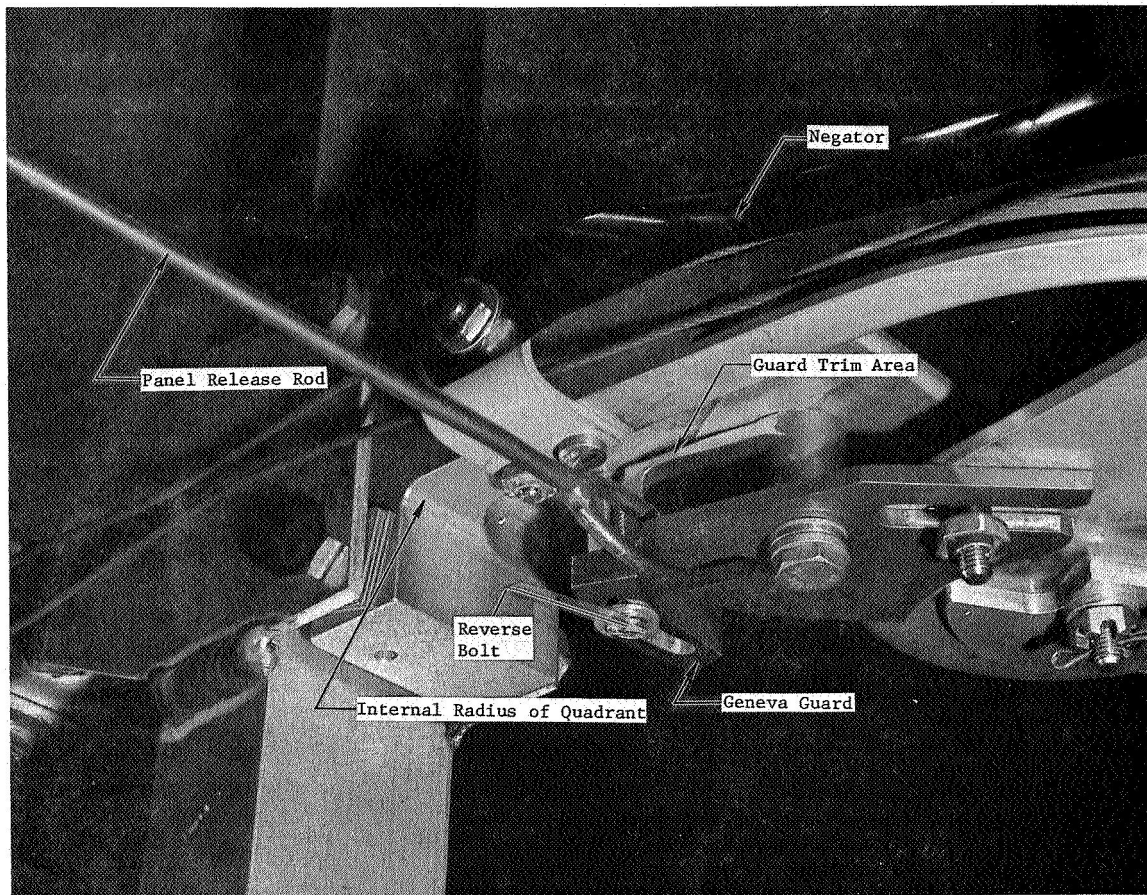


Figure 5-20: INTERFERENCE PROBLEMS

Link lengths and hole locations on the main panel latches allowed over-travel on latching. Under certain maneuver loads, panel joints could work excessively. A new geometry and linkage length were established that alleviate this problem. Adjustable bolt stops were installed on MTA-3 under each latch to prevent panel reversal during locking.

It was found that shear pinhole-to-release-rod diametral clearance for both main and auxiliary subpanels was not adequate for all operating conditions. Larger holes will be drilled in shear pin brackets in Phase III.

Each rotary joint in the mechanical system used bolts secured with castle nuts and cotter pins as the pivot. Installation control using thickness gages was required to make certain that the bolts would not cause rubbing friction in the faying surfaces due to bolt tension. A design change has been made to use simpler pins and locking rings wherever practical.

The original design required that hinges and quadrants be predrilled and located before drilling titanium mounting clips to match. A design change has been made to allow predrilled titanium clips. The aluminum hinge plates and quadrants will be drilled on assembly to match in Phase III.

5.6.2 CONCLUSIONS CONCERNING DEPLOYMENT VERIFICATION TESTS AND ANALYSIS OF TEST DATA

The analysis of MTA-3 deployment tests shows that changes are required in the mechanism and GSE design, and in the test procedures to achieve a successful performance of MTA-4A tests and to ensure the capability of the array to deploy in space.

Evaluation of the perturbation forces necessary to show acceptable correlation between measured and calculated deployment motions show that some residual hangup forces existed between the air-sled feet and the track, and that hinge frictions were large over a portion of the deployments.

Well-developed cable deformations observed during the deployment measurements demonstrate a lack of repeatable characteristics for the cable stiffness in sequential deployments.

Improvement in the data measuring system to obtain valid cable tension readings in the low-load-level region is required.

Measured motor speeds are generally higher than the manufacturer's nominal during deployment by 8% and in some deployments appear to exceed the maximum speed tolerance. This condition evidently results from a natural wearing-in of the harmonic drive reducers and yields an efficiency increase. Future procurement must provide for a wear-in period during acceptance tests to reduce this variation in performance.

Procedure changes that will include provisions to measure and separate the track slope and the panel friction effects are required for Phase III. A rerun of some panels with a plastic membrane is needed in Phase III to obtain aerodynamic data.

Recommendations are made in Section 5.6.1 for the cable changes, Section 3.1.7 for hinge alignments, and Section 5.6.2 for the air-sled clearance, improvements in the data system, and procedure changes.

The recommended changes and additions will allow a successful performance of Phase III deployment tests and provide test data to sufficient accuracy to analytically establish the capability of the array to deploy in space.

5.6.3 CONCLUSIONS CONCERNING DEPLOYMENT TEST EQUIPMENT

The deployment equipment and procedures, with planned modifications, will be suitable for Phase III deployment testing.

The test equipment performed the required functions throughout Phase II. The equipment allowed the deployment of each set of main subpanel units of MTA-3 with the flight prototype deployment motor as the only source of energy. The extreme freedom of motion allowed by the air-bearing

undercarriage system proved to be extremely valuable in determining the effects of the minute changes in forces that would influence the deployment of a similar array in space. These tests are felt to have confirmed the superiority of the passive, minimum friction, ground-deployment concept in obtaining useful test data for this or any similar large, lightweight, space-deployable structure. In this respect, the deployment equipment used in these tests can be considered preprototype for future flight acceptance test equipment.

With the addition of planned modifications and a few minor improvements, the test equipment and procedures will be suitable for the future testing of the beryllium array. Planned modifications include:

- 1) The Addition of Overhead Air Bearings----These will allow close control of the closing velocities and cable tensions for the last 20 degrees of a given deployment swing, so that the transient as well as time-history effects on the succeeding set of subpanels can be determined.
- 2) Provisions for Indexing the Deployment Angle Instrumentation to the Array---This will eliminate the need to recalibrate for the deployment degradation tests run after each environmental test.
- 3) The Addition of Special Fittings for the Attachment of the Overhead Suspension Lines to the Beryllium Subpanels---This will minimize GSE loads on the beryllium structure.
- 4) The Addition of Operating Check Lists and Handling Precautions to the Operating Procedures---These additions, which were developed during Phase II testing, will reduce any chance of damage to the beryllium array during deployment testing.

In addition to these previously planned modifications, certain equipment and procedural changes are recommended for Phase III. These changes, described below, will result in more accurate test data and minimize the chance of damage to MTA-4.

- 1) Revise the cable tension-sensing linkage to increase sensitivity in the 0- to 10-pound range.
- 2) Revise the air-sled air bearings to increase the operating gap between the bearings and the track, and to reduce deflection of the bearings to minimize the change of hangups.
- 3) Revise the operating procedures to provide for moving the air sled under MTA-4 from an initial position for Swings 1 and 2 to a second position for Swings 3, 4, and 5. This will more evenly balance the load on the two most heavily loaded bearings. This change and Item 2, above, combined with the track joint conditioning procedure developed during these tests will ensure the completely free operation of the air sled in future testing.

- 4) Add a pair of jackscrew beams to the Phase III deployment equipment for use in supporting MTA-4 while the air sled is shifted as described above. These beams should be padded at their interface with the lower spars of MTA-4, and should provide for careful vertical adjustment in the range of travel necessary to lift MTA-4 free of the air sled.
- 5) The use of the heavier, less-sensitive potentiometers at each hinge joint is recommended for Phase III testing.
- 6) A clamp-type device is recommended to lock the deployment angle potentiometer to the No. 1 subpanel.
- 7) The deployment equipment checkout, using MTA-3, should include a complete deployment to verify the freedom of operation of the overhead air bearings and air sled. MTA-3 should be used to perform a complete checkout of the procedures for loading, handling, and removing MTA-4 from the deployment fixture. All ancillary equipment, such as the handling sling, should be used.
- 8) The initial testing of MTA-4 should include the calibration, and doweling in place for later removal and replacement, of the deployment angle instrumentation. An accurate calibration should be provided for the last 20 degrees of each deployment Swing so that closing velocities and closing cable tensions can be duplicated later within the required ranges. Variations in distance from closing and initial cable tension should be used to achieve closing conditions within the required ranges. Two X-Y plotters should be employed to monitor both hingelines during controlled starting-condition testing.
- 9) The deployment degradation tests involve complete, unassisted deployment immediately after each environmental test. No manual walk-through can be allowed or evidence of degradation would be obscured. To eliminate the possibility of air-bearing hangups, the tender sled, at low air pressure, should be used to check the air-bearing track before each deployment swing.

In addition to testing functions, the deployment equipment proved to be very useful in checking out the installation of the mechanisms on the subpanel structures. The suspension of the five main subpanels on the soft overhead lines allowed the separation of any pair of subpanels with a minimum of handling. Subpanels could be held apart simply by tying them in the desired position with string. This proved to be a safe and convenient method of gaining access for minor reliefs and adjustments. In stowing the completely assembled array, the free motion allowed by the deployment equipment allowed the detection of small frictions and mechanical irregularities.

6.0 HANDLING EQUIPMENT AND PHASE III PRELIMINARY DESIGN (TASK 5),

Task 5 consisted of the design and fabrication of necessary Phase II handling equipment and the preliminary design of identified Phase III ground equipment.

Phase II handling equipment was designed, fabricated, and successfully employed for the transport of MTA-1. Tests to establish fit and function and to determine equipment performance under typical operating conditions were conducted prior to use with the beryllium test article.

Preliminary designs of Phase III equipment were updated and completed to a greater level of detail than reported at the end of Phase I. Design drawings were released in Boeing Document D2-113352-5, *Preliminary Design of Phase III Ground-Support Equipment*.

Tests of selected elements of the Phase III deployment demonstration system were conducted in Phase II. Portions of the overhead track and trolley systems were fabricated and tested. Test results indicated the need for further design study of these systems in Phase III.

6.1 EQUIPMENT DELETIONS

System-design improvements have resulted in significant cost savings through reduction in ground-equipment requirements.

Several ground-equipment items described in the Phase I Final Report were not required to support the Phase II test program as finally implemented. Knowledge and experience gained in Phase I and program redefinition during Phase II permitted simplification of the test program and elimination of certain equipment tests.

The SCS-43 acoustic and thermal-vacuum tests conducted in Phase I provided sufficient information to permit deletion of these tests in Phase II. As a result, the MTA-1 acoustic and thermal-vacuum test fixtures were eliminated.

The successful tests of SCS-43 with the X-200 solar simulator and the operational difficulties of conducting natural sunlight tests at a remote site with so large a test article produced the decision to perform power degradation tests of MTA-1 with the X-200. This decision eliminated the need for the MTA-1 shipping container and modification of the existing equatorial mount. A further decision to delete all live solar-cell testing on MTA-1 eliminated the need for the X-200 dolly and the instrumentation rack.

6.2 PHASE II HANDLING EQUIPMENT

The handling equipment provided in Phase II performed intended functions during handling and transport, and facilitated manufacturing and test operations.

The Phase II handling equipment consists of the handling dolly, handling frame, protective cover, and sling (see Figure 6-1).

The dolly and frame are used to transport MTA-1 between the fabrication and test area, and as a holding fixture during installation of test instrumentation.

The handling frame is constructed of 3-inch-square aluminum tube. The subpanel is held in place by angles bolted to the frame. Soft rubber pads bonded to the angles and frame protect the subpanel structure.

A plywood protective cover is installed over the frame to prevent damage to the subpanel during transport or temporary storage. The cover is divided into four separate sections for ease of handling.

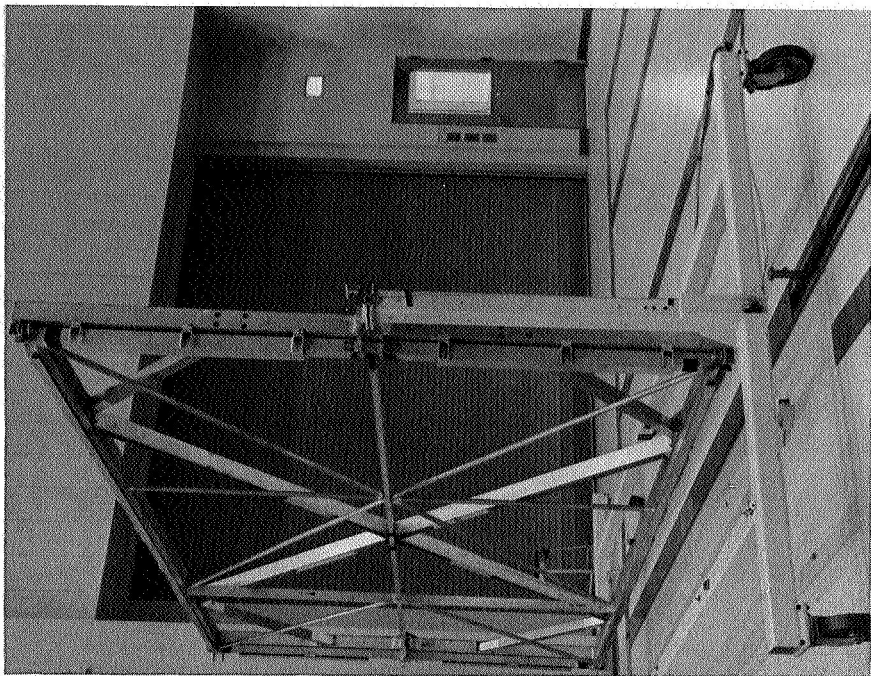
The sling is used to lift MTA-1 in its handling frame. It consists of a simple spreader bar with short wire-rope cables at either end. A commercial self-adjusting eye is used to center the lift point above the center of gravity.

The handling equipment is designed with a safety factor of 5 times the design-limit load, based on the ultimate strength of the material. The sling and handling frame are periodically proof-loaded to twice the working load in accordance with Washington State law.

To ensure the safety of MTA-1, the handling equipment was checked for fit and function with MTA-3 Subpanel 1 before use with MTA-1. The frame and dolly have undergone additional tests to determine frame deflections and accelerations experienced during typical dolly operations (see Figure 6-2).

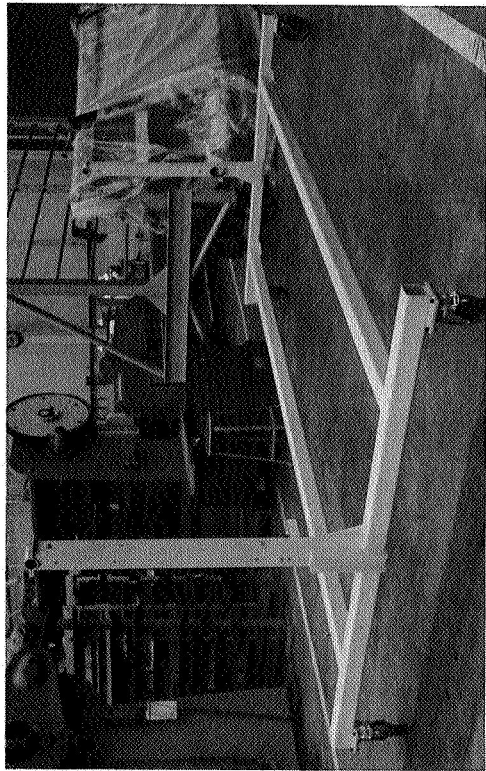
Handling-frame deflections were determined with the frame mounted in the handling dolly. Lead weights twice the weight of MTA-1 were placed around the frame periphery. Deflections were measured both in-plane and out-of-plane of the frame by means of a transit and precision scale. Corner deflection simulating pickup of the frame at one corner only also was measured.

Acceleration tests were performed with MTA-3 Subpanel 1 in the handling frame, with protective covers installed. The dolly was manually pushed over a course of 1- by 6-inch boards bolted to the floor. Accelerations were measured at five points on the frame and dolly when one set of casters crossed the boards and when all casters crossed simultaneously.

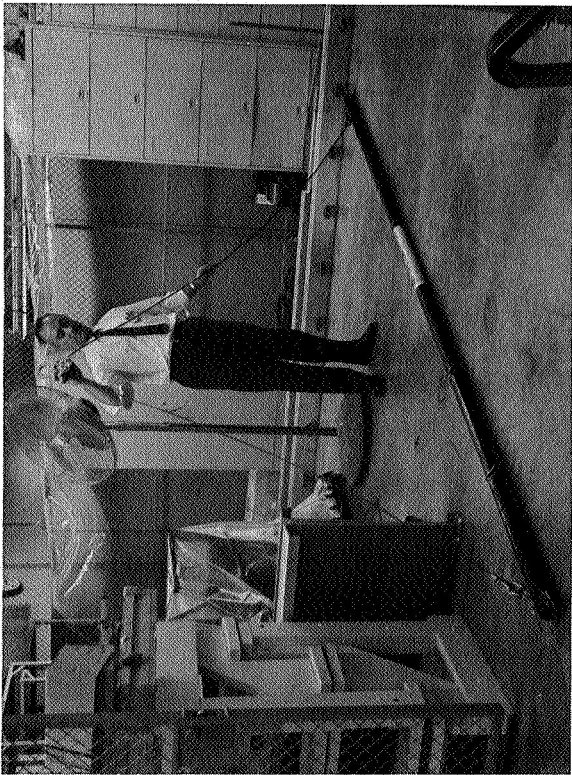


Handling Dolly and Frame
with MTA-3 Subpanel No. 1 Installed

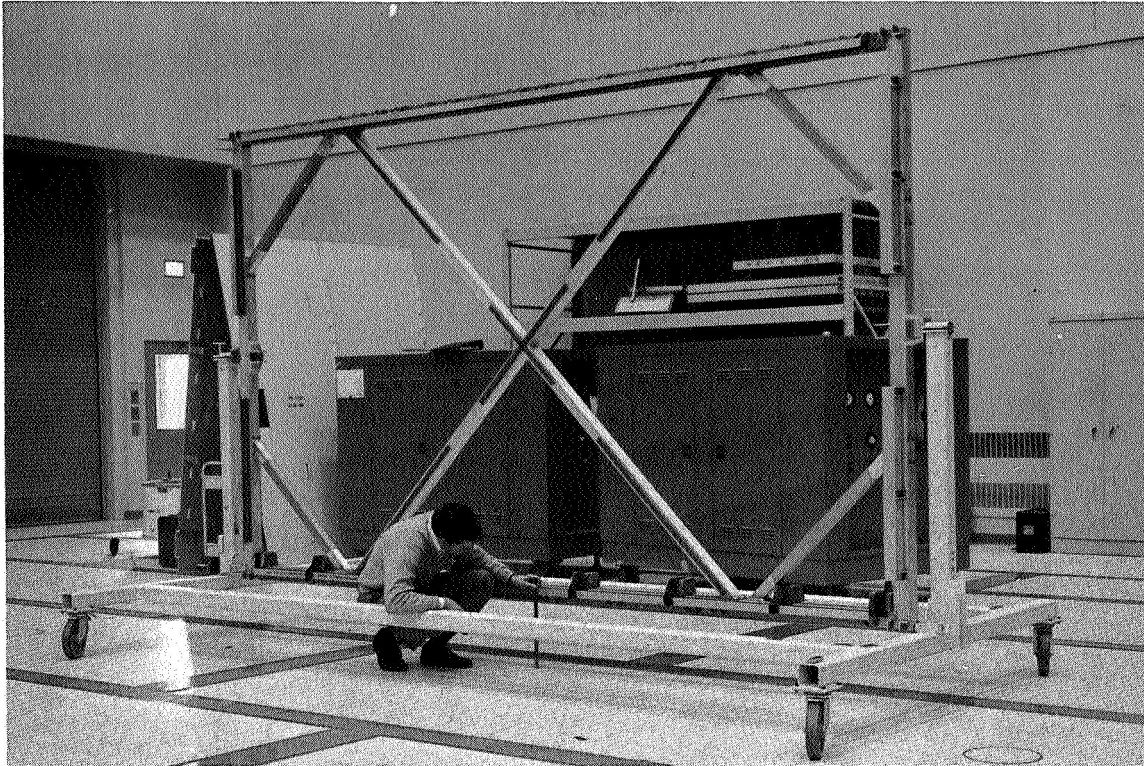
Figure 6-1: HANDLING EQUIPMENT



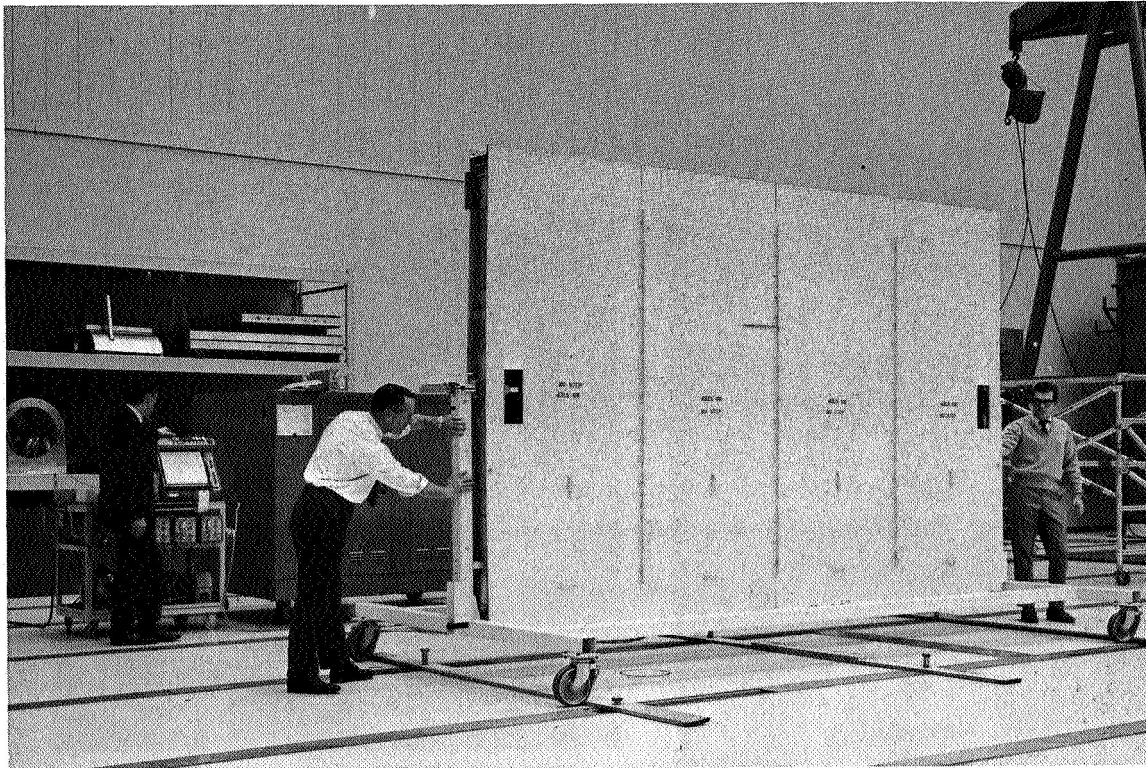
Handling Dolly



Handling Sling



Handling Frame Deflection Test



Handling Dolly Shock Loading Test

Figure 6-2: HANDLING EQUIPMENT TESTS

Analysis of the test data indicated that frame deflections are within acceptable limits and that the MTA-1 would not be damaged during inplant transport on the handling dolly when normal handling precautions are observed.

6.3 PHASE III PRELIMINARY DESIGN

The preliminary designs produced in Phase I for Phase III equipment were updated to include greater design detail and to reflect the Phase III manufacturing and test plans as they existed during the winter of 1967-68. These plans are shown in Figure 6-3.

MTA-4 is the complete panel assembly. Subpanel Unit No. 2 will first be assembled and then deployed on the subpanel unit deployment fixture. The remaining subpanels will then be assembled into MTA-4 and deployed on the deployment verification fixture. After deployment, MTA-4 will be stowed and subjected to a series of environmental tests. Deployment and power degradation tests will be performed after each environmental test.

MTA-3, a full-size deployment model, will be used for checkout of all Phase III GSE and then will undergo a series of deployments on the demonstration fixture.

The preliminary designs discussed below are based on a 13 beryllium subpanel MTA-4 configuration.

6.3.1 DEPLOYMENT EQUIPMENT---OVERHEAD SUSPENSION SYSTEM

Overhead air bearings must be added in Phase III main subpanel deployment verification testing to obtain accurate data on transient oscillations during latching.

One of the objectives of panel assembly deployment verification testing in Phase III is to observe the transient dynamic behavior of the deploying array, particularly during latching. The latching of each main subpanel will set up oscillations in the deployed subpanels that will extend back throughout the panel assembly. These oscillations will be masked unless the panel assembly is supported by a lightweight, minimum-friction system.

Such a system using overhead air bearings is shown in Figure 6-4. The main subpanels are suspended from the same lines used in Phase II, but are connected to air bearings running on short tracks.

The tracks are mounted between pairs of 12-inch structural channels. The channels are sized to minimize deflections and maintain track levelness during operation. The tracks are steel weldments faced with epoxy and machined to the required flatness.

The air supply is piped along the catwalk and is delivered, by means of soft tygon hoses, from points 6 feet above the tracks to the air bearings.

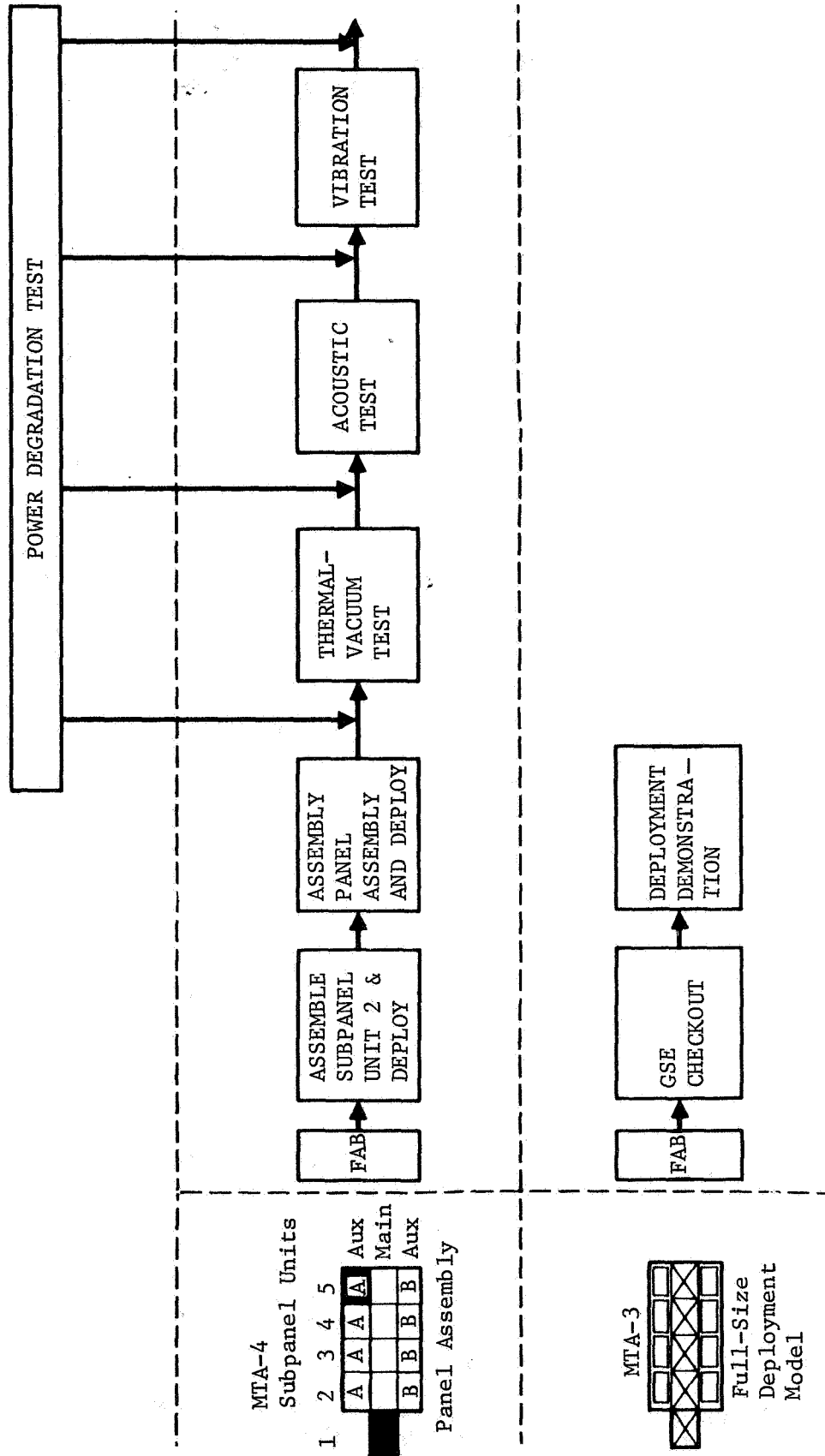


Figure 6-3: PHASE III TEST-ARTICLE FLOW DIAGRAMS (USED AS BASELINE FOR PHASE III GSE PRELIMINARY DESIGNS)

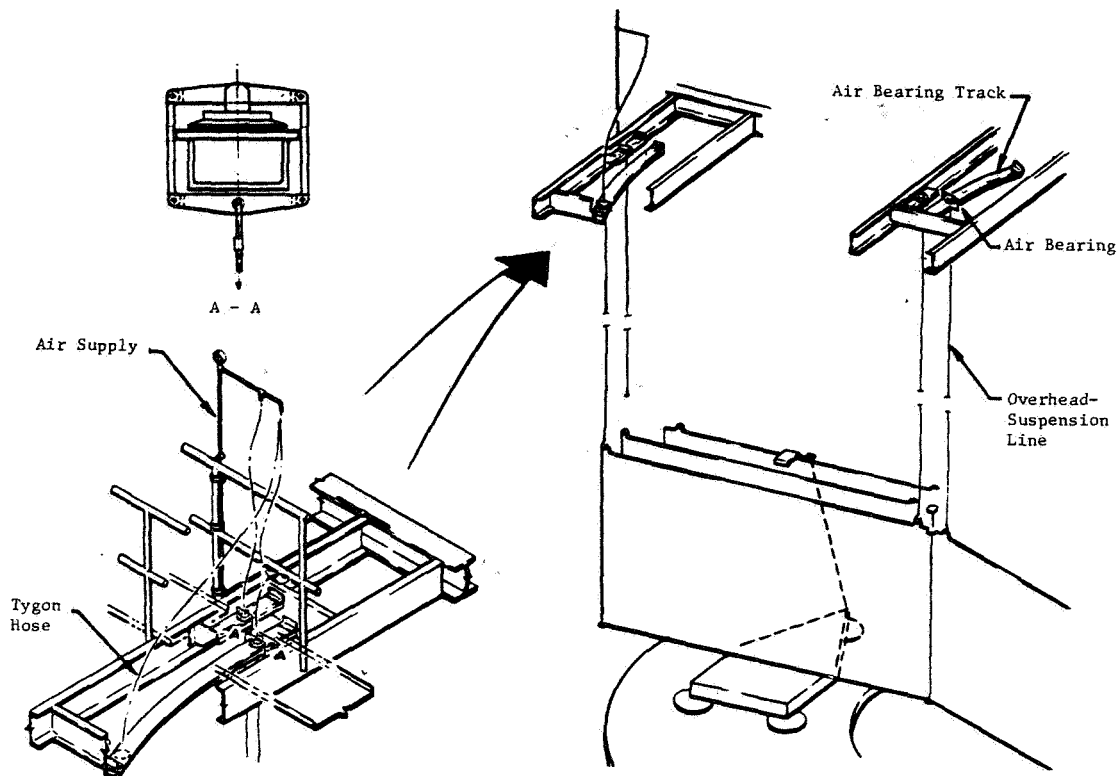


Figure 6-4: DEPLOYMENT EQUIPMENT---OVERHEAD AIR BEARING

This system was originally proposed for Phase II, but was eliminated because the transient behavior of the aluminum MTA-3 was not representative of the transient behavior of the beryllium MTA-4.

6.3.2 DEPLOYMENT EQUIPMENT---SUBPANEL DEPLOYMENT FIXTURE

Larger values of GSE inertia and friction can be tolerated in the subpanel deployment fixture because of the simplicity of the auxiliary deployment system.

Subpanel deployment tests will be conducted to verify that pairs of auxiliary subpanels will deploy under their own power without damage. The factors affecting auxiliary subpanel deployment are relatively straightforward. For example, the angular velocity depends on the deployment spring torque, viscous damper drag, GSE and subpanel inertia that are known, and GSE friction that can be measured. Thus, in a ground deployment test the GSE effects can be analytically subtracted from the test results to obtain data applicable to space deployment.

The fixture will consist of a simple A-frame support structure attached to the base tower of the deployment fixture structure (see Figure 6-5). The auxiliary subpanels will be supported by counterweighted overhead arms, pivoting on low-friction bearings.

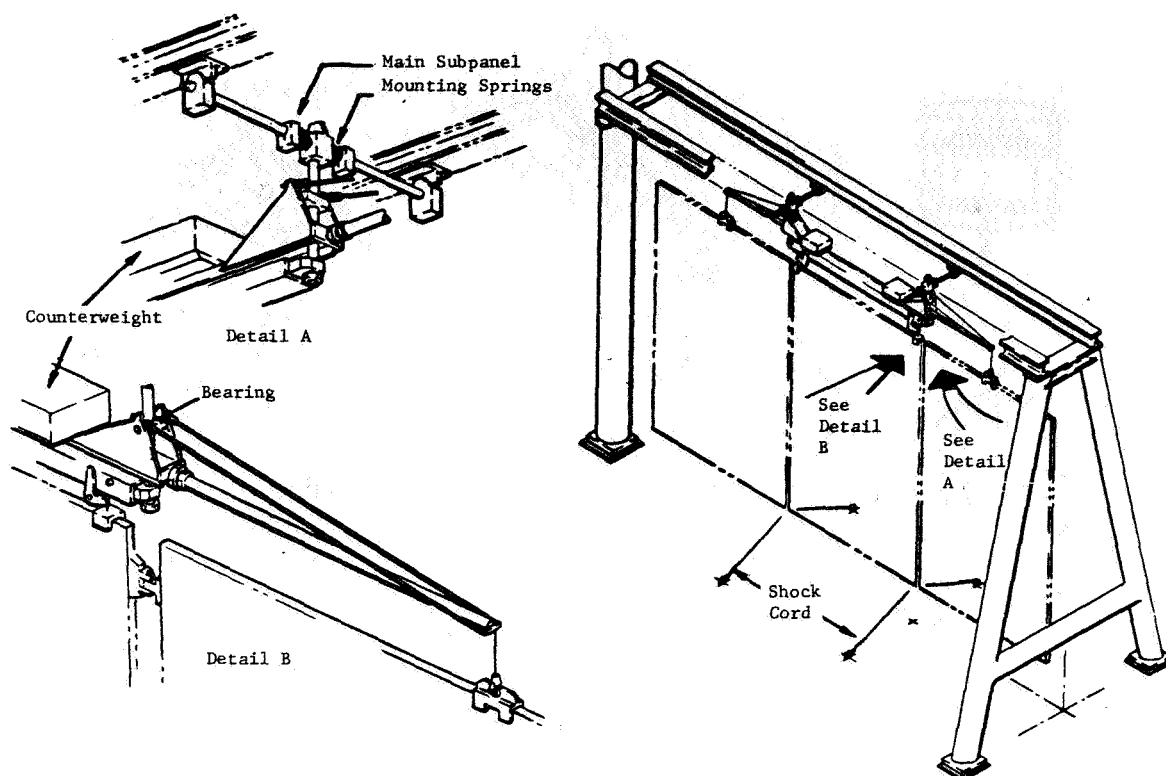


Figure 6-5: AUXILIARY SUBPANEL DEPLOYMENT EQUIPMENT

The opening and latching of the auxiliary subpanels will produce some bending and twisting of the parent main subpanel. To simulate this effect, the main subpanel is mounted through a set of horizontally acting springs and two shock-cord tiedowns, thus duplicating the stiffness of the main subpanel's position in the array.

6.3.3 DEPLOYMENT DEMONSTRATION EQUIPMENT

The deployment demonstration equipment will be provided in Phase III to demonstrate, by means of a sequential deployment of a full-scale aluminum deployment model, the operational geometry of the folded modular panel assembly. Evaluation of this demonstration will be visual; there is no requirement to gather test data such as accelerations, transient oscillations, or deployment time-history. The objective of achieving realistic deployment at reasonable equipment cost guided the preliminary design effort.

The following sections describe the deployment demonstration equipment, development testing of certain system components, and recommendations for further Phase III design effort.

6.3.3.1 Baseline Configuration

The deployment demonstration equipment is designed on a cost-effective basis.

The planned Phase III deployment demonstration equipment has been designed to permit full, sequential deployment of MTA-3, including auxiliary subpanels, to prove the geometry of the folding modular-packaging concept. Systems trades conducted in the early stages of demonstration equipment design indicated that a low-cost system would satisfy the Phase III demonstration requirements. Equipment of greater sophistication would not significantly improve system performance, and would result in disproportionate increases in cost.

The deployment demonstration equipment is shown in Figure 6-6. MTA-3 will be supported during deployment by an overhead monorail track carriage and spreader bar system. Primary power is provided by a winch-and-tow cable arrangement. The cable is guided by pulleys located along the track. A secondary source of power is provided by a deployment cable routed over the quadrants and reeled in by a negator spring motor. This will help smooth out oscillations during deployment.

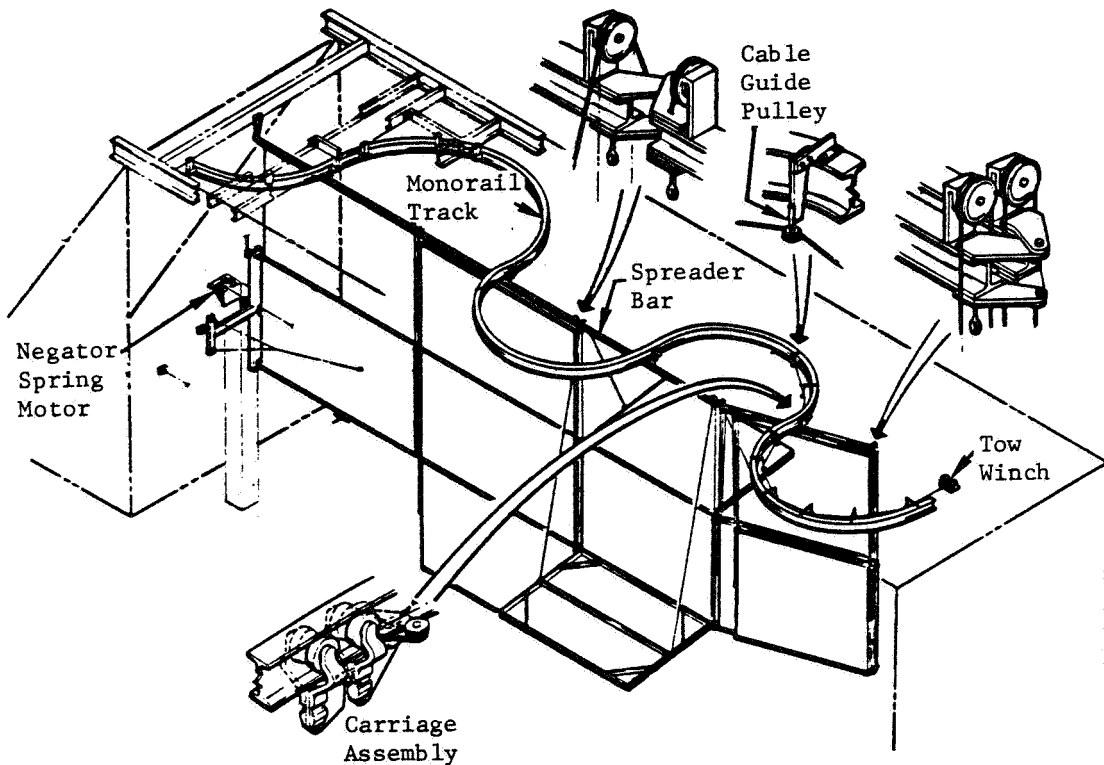


Figure 6-6: DEPLOYMENT DEMONSTRATION EQUIPMENT

The spreader bars consist of aluminum I-beams hinged directly above the MTA-3 main hinges. Pairs of auxiliary subpanels are interconnected by cables that drape over pulleys on the spreader bars. A light counterweight is added to the downswinging subpanel to provide deployment impetus. Helper springs at the auxiliary hinges augment the counterweight force and sustain subpanel motion through dead-center points.

6.3.3.2 SCG-4 Development Test of Main Subpanel Deployment

Main subpanel deployment development testing (SCG-4) revealed areas of necessary design improvements.

Main subpanel deployment testing provided experimental data concerning the operation of the overhead monorail carriage system. The test setup consisted of a prototype track section, carriage, and spreader bar (see Figure 6-7). Power was provided by a hand winch driven by an air motor and a counterweight to simulate the negator spring motor. A plywood test panel was used because MTA-3 was not available. The track was suspended from an overhead framework located between the sides of the base tower of the deployment fixture structure.

The test consisted of deploying the panel through a typical 90-degree swing at various speeds using the winch and observing the results. Photographic coverage also was provided.

The basic design philosophy was proved satisfactory by these tests. However, detailed design deficiencies were noted in some areas. Manual operation of the winch was too slow and caused considerable operator strain. An air motor was adapted to the winch to solve this problem.

The deployment cable is guided along the curved track by means of pulleys. Tension is maintained in the cable by a fishing-reel drag. During deployment, the cable often rode above the guide pulleys. It was concluded that the present pulleys should be replaced by a guide of greater length. In addition, the fishing reel did not have the capacity to store the required cable length and should be replaced with a larger reel or drag winch.

The carriage assembly moved along the track with a surging motion that imparted undesirable oscillations to the suspended panel. Track surface unevenness and stretch in the cable appeared to be the main causes. This problem was further investigated during the auxiliary subpanel deployment tests described in Section 6.3.3.3.

6.3.3.3 Additional Development Tests of Main and Auxiliary Subpanel Deployment

Accelerated Phase II development of demonstration test equipment revealed the need for additional design improvements.

JPL Technical Direction Memorandum (TDM) 951934-4 authorized fabrication and testing of deployment demonstration equipment components originally scheduled for Phase III. This activity included the design and fabrication of the lightweight MTA-3 auxiliary subpanels, the modification of MTA-3 Main Subpanel 4 to the deployment demonstration configuration by the addition of auxiliary deployment mechanisms, and fabrication of the overhead spreader bar and suspension system.

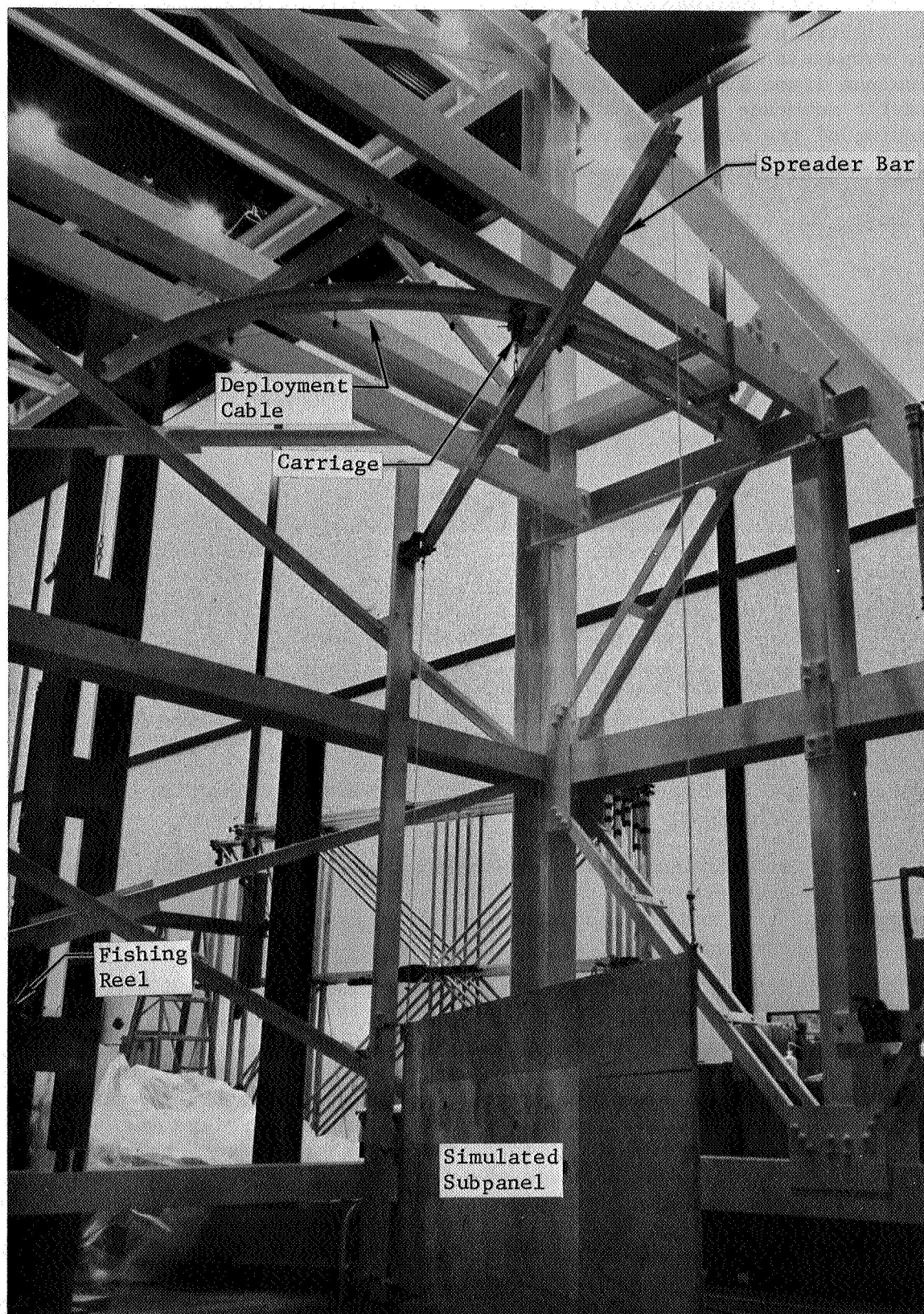


Figure 6-7: SCG-4 TEST SETUP

The lightweight auxiliary subpanels simulated subpanel geometry only. The subpanels (see Figure 6-8) consisted of an all-welded, thin-gage aluminum frame similar in configuration to the MTA-3 main subpanels. A plastic membrane was stretched over the frame during test to provide air damping of the deployment motion.

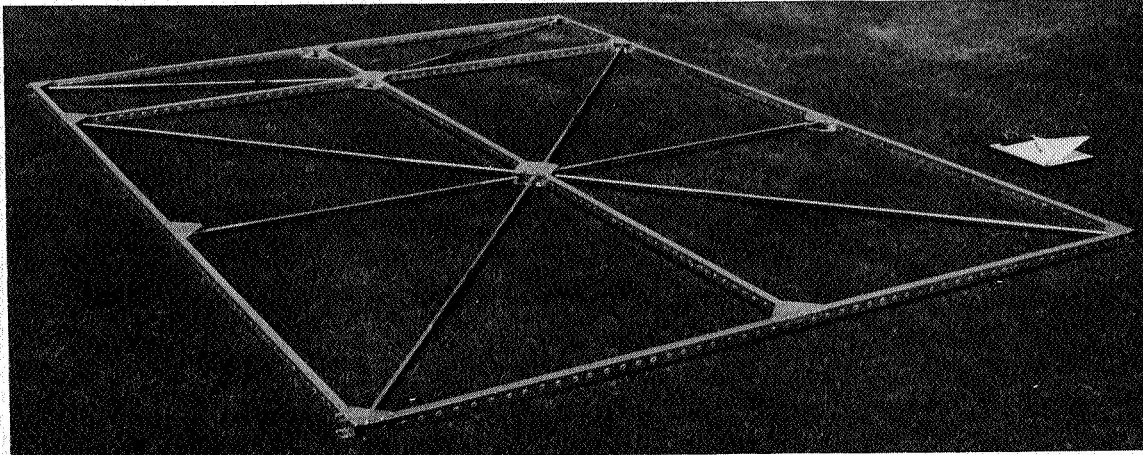


Figure 6-8: AUXILIARY SUBPANEL---MTA-3

The test setup is shown in Figure 6-9. The overhead track and carriage from the SCG-4 test were used (see Section 6.3.3.2).

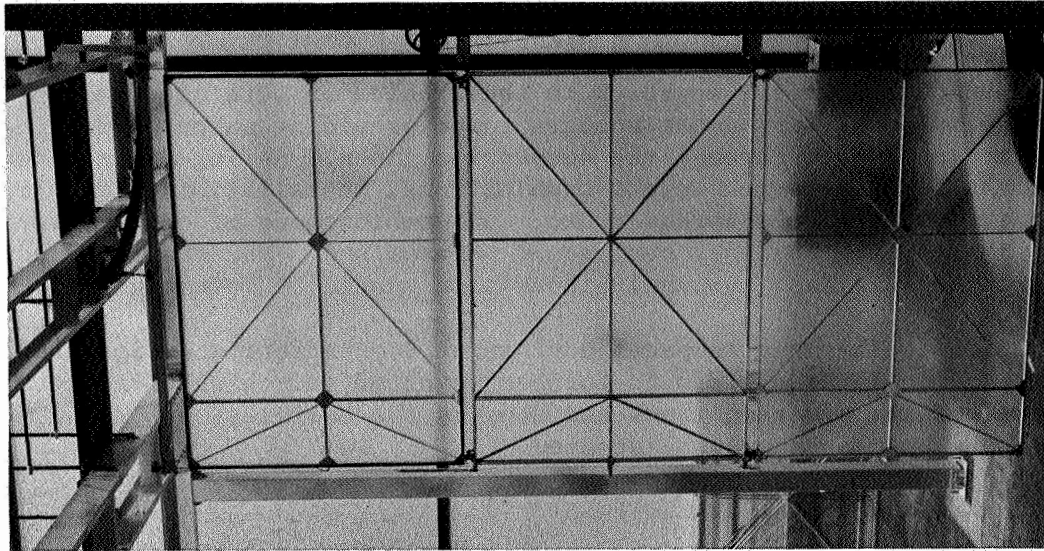
Trial deployments indicated the need to dampen the subpanel to smooth the deployment motions and prevent latching at excessive speeds. A rotary dashpot was installed on the spreader bar. Positive interconnection between the dashpot and subpanel was accomplished by replacing the music wire and smooth pulleys that support the subpanels with a chain and sprocket.

Deployment with the dampened system was not satisfactory. The increased size of the counterweights required to overcome damper resistance and proceed past the initial dead spots resulted in excessive velocities at closing.

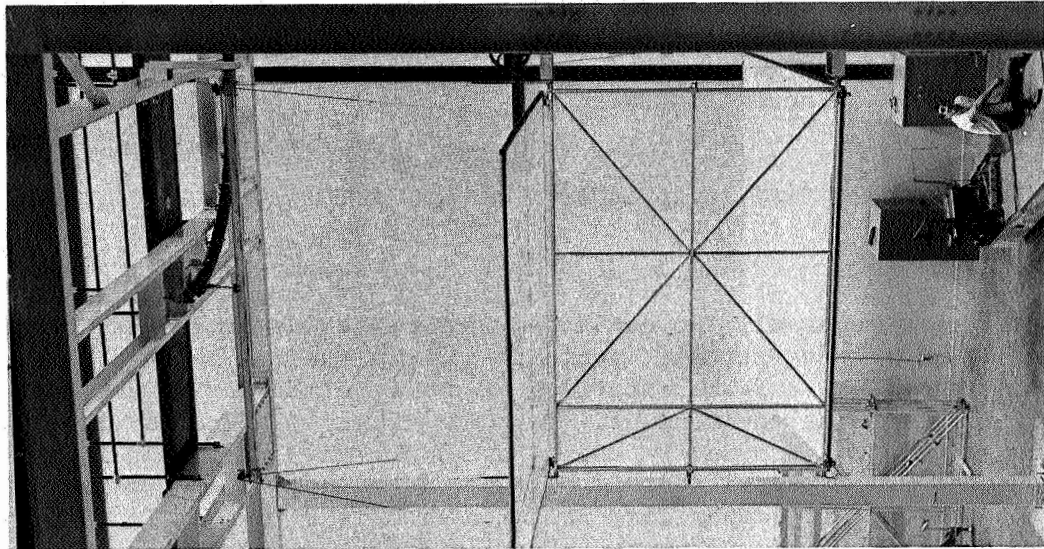
The test setup also permitted swinging of the subpanels through 90 degrees using the overhead track and carriage. Experiments to improve carriage motion were first conducted at ground level on a straight-track section with different types of carriage wheels. Rubber wheels were found to produce the smoothest carriage motion. Rubber wheels were then installed on the overhead track. The motion of the carriage was noticeably smoother than with the original steel wheels; however, the oscillation of the suspended subpanels observed during the SCG-4 test was still present.

6.3.3.4 Recommendations for Further Design and Test

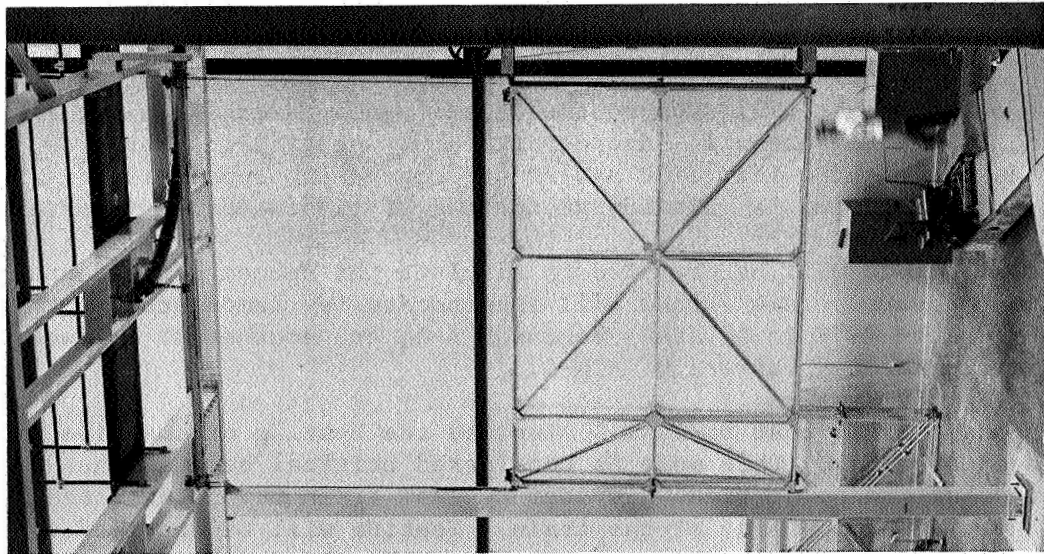
Design and equipment modifications are required prior to Phase III development tests.



Fully Deployed



Partially Deployed



Stowed

Figure 6-9: AUXILIARY SUBPANEL DEPLOYMENT DEMONSTRATION TEST

The results of the main and auxiliary subpanel deployment demonstration test discussed above indicate the need for further design study in Phase III. The most practical approach to the problem of obtaining smooth auxiliary subpanel deployment appears to be providing subpanel driving and retarding forces through the suspension lines. Spring-dashpot combinations, dashpot-counterweight combinations, installation of an electric drive motor on the spreader bar, and combinations of opening-retarding springs on the spreader bar should be examined. Mounting an electric motor at the auxiliary subpanel hinges also should be investigated.

In addition to the main subpanel deployment demonstration design changes recommended in Section 6.3.3.2, further means of smoothing carriage motion along the overhead track and damping the oscillations of the suspended subpanels should be studied.

6.3.4 ENVIRONMENTAL TEST EQUIPMENT

Environmental test equipment is required to test the panel assembly in the environments experienced during boost, orbit, and flight.

6.3.4.1 Thermal-Vacuum Test Fixture

The thermal-vacuum test setup closely simulates the critical thermal environments.

Thermal-vacuum tests will be conducted on MTA-4 to determine structure and cell temperature extremes, evaluate induced thermal strains, and observe the behavior of the materials of construction in a simulated thermal-vacuum boost environment. Two critical thermal modes will be tested. In the first mode the stowed panel assembly will be held in the launch (upright) position and will be illuminated by the solar simulator in the vacuum chamber lid. Illumination in this position is expected to produce the greatest thermal gradients in the structure because of the long path between the heated and cold surfaces.

The thermal-vacuum test fixture (see Figure 6-10) will consist of an aluminum frame that will support the stowed panel assembly in the launch position. The fixture will incorporate the boost tiedown system for one panel assembly and a restraint system to support the subpanels after the boost tiedown system is released at the end of the thermal-vacuum tests.

A thermal shroud will be provided to simulate the spacecraft external skin temperature. This shroud will incorporate two temperature control zones---one at 75°F immediately behind MTA-4, and another at a lower temperature located forward of MTA-4.

The second thermal test mode will simulate the heating of one side of the panel assembly. This mode is considered critical because the operation of the boost tiedown system release train may be affected by solar heating of one side of the train. Heating will be provided by an auxiliary infrared lamp stand.

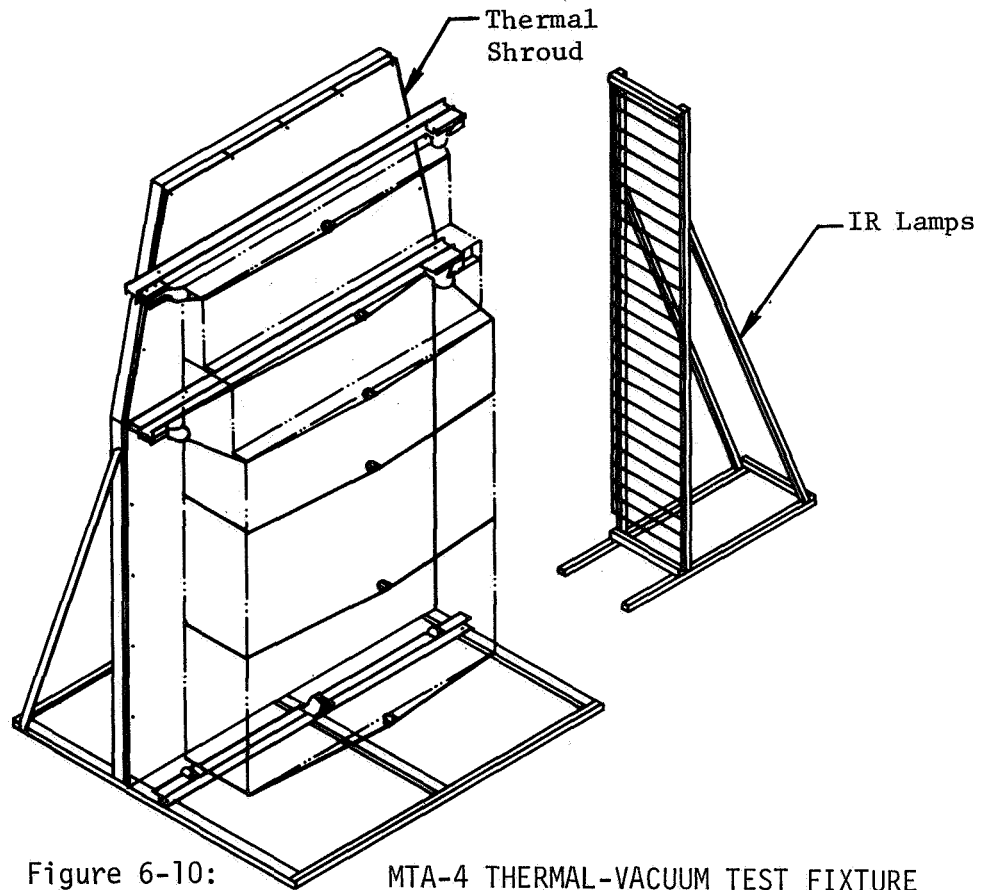


Figure 6-10: MTA-4 THERMAL-VACUUM TEST FIXTURE

6.3.4.2 Vibration Test Fixture

An aluminum model of sophisticated design and construction is required to adequately simulate the dynamic characteristics of a stowed panel assembly.

MTA-4 will be vibration-tested in both the shear and normal directions. For each test setup, three dummy panels will be provided to simulate the structural response of the adjacent stowed panels on the spacecraft.

During Phase I, an idealized mathematical model of the panel assembly was developed for the dynamic analysis of the stowed array. This model formed the basis for dummy panel design.

A physical representation of the model is shown in Figure 6-11. The beryllium section properties for the structure were calculated and aluminum beams designed that duplicate, as closely as possible, the stiffness of the model.

The aluminum dummy panel with section properties adjusted to match the beryllium structure stiffness will weigh just under 400 pounds. This will allow about 100 pounds to aid in matching the center of gravity and dynamic characteristics of the stowed panel assembly.

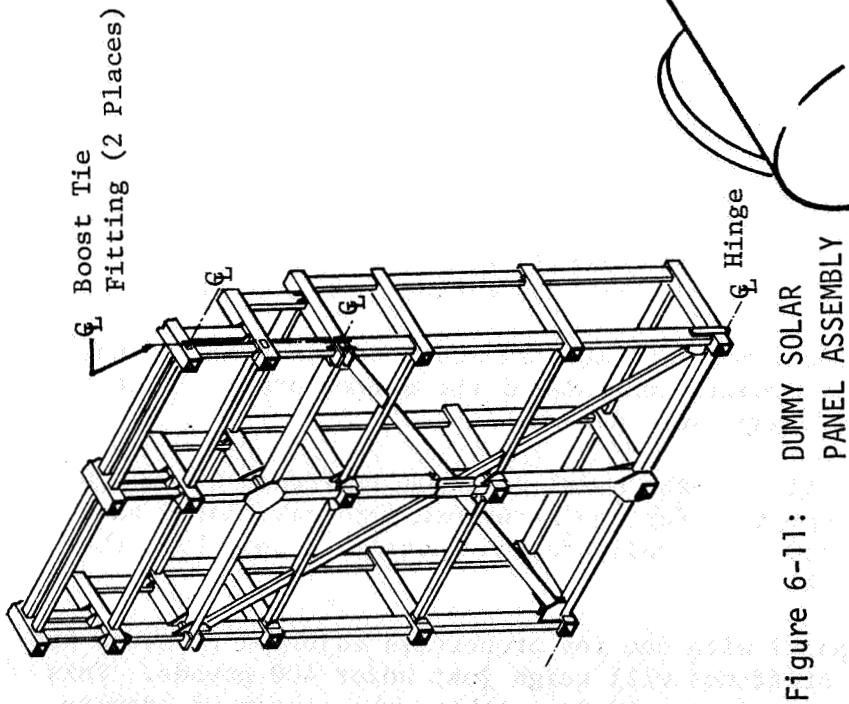


Figure 6-11: DUMMY SOLAR
PANEL ASSEMBLY

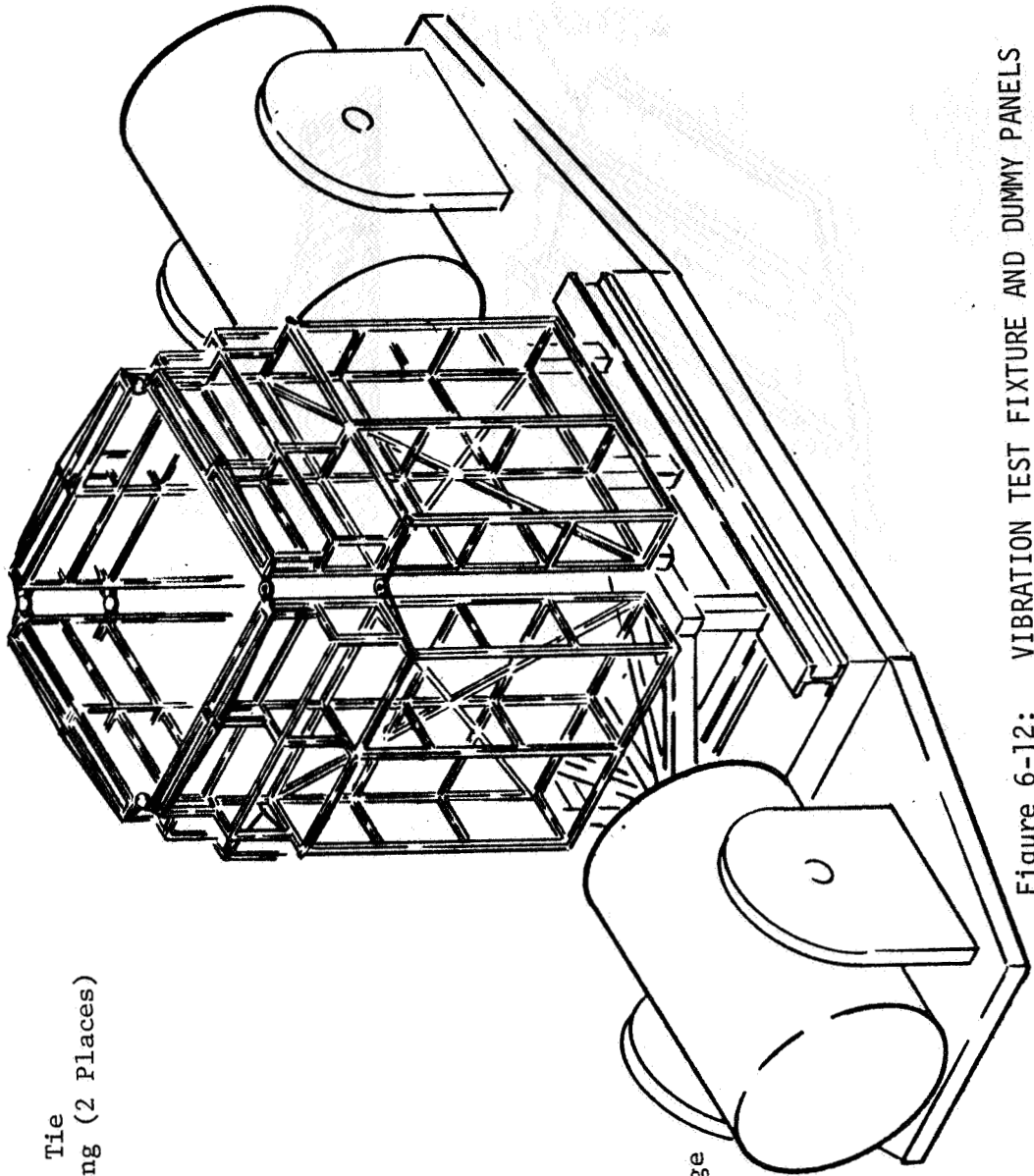


Figure 6-12: VIBRATION TEST FIXTURE AND DUMMY PANELS

The combined weight of MTA-4, the dummy panels, and the vibration fixture exceeds the force capabilities of a single vibrator. MTA-4 vibration testing will, therefore, be conducted with two Ling 249 vibrators synchronized with control circuitry in a manner similar to that developed by Boeing for Saturn LOX valve testing (see Figure 6-12). The basic vibration fixture will consist of a table on which simulated spacecraft hinges are mounted. The table will be supported by flexure members that permit table motion. Force adapters will connect the table with the two vibrators.

6.3.5 HANDLING AND TRANSPORTATION EQUIPMENT

A specially designed sling and transporter are required in Phase III to facilitate handling of the panel assembly and movement between test areas.

During Phase III, MTA-4 must be transported between the final assembly and test areas and between various test locations. This will be accomplished with the stowed panel assembly transporter (SPAT).

The SPAT will consist of a tubular welded frame and a vertical column to which the MTA-4 spacecraft hinges attach (see Figure 6-13). Two strongbacks attached to MTA-4 will provide anchor points for simulated boost tiedown cables. These strongbacks will remain with the panel assembly during all transport and hoisting operations to prevent inadvertent opening of the array and to keep the shear teeth engaged.

For long-distance shipment, the SPAT will be placed in an air-ride van. Protective covers will enclose the SPAT during shipment.

The panel assembly sling will be capable of handling MTA-4 in either the horizontal or vertical position. It will consist of a spreader bar and cradle assembly. The panel assembly is stabilized by brackets on the strongbacks that engage sling cables. Rubber bumpers prevent contact between the cables and the panel assembly.

A universal handling frame similar to the Phase II MTA-1 frame will be provided in Phase III to handle any one of the main or auxiliary subpanels. Crossbeams with removable clamps will be added to the frame to accommodate the various subpanel lengths.

6.4 FAILURE MODE AND EFFECT ANALYSIS (FMEA) OF GSE

A documented analysis was desirable to identify possible risks of damage to test articles or of invalid test data.

A failure mode and effect analysis (FMEA) for deployment and handling equipment was prepared. The FMEA identified known potential causes of

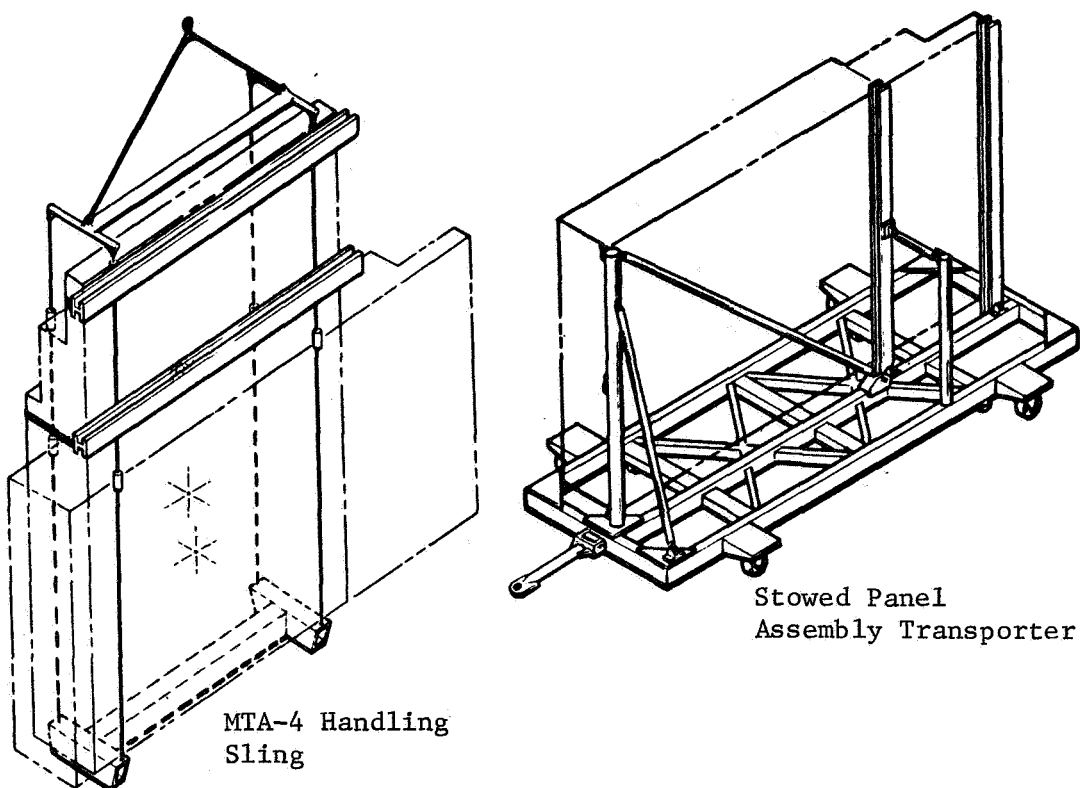


Figure 6-13: PHASE III HANDLING EQUIPMENT

damage to the test articles or of invalid test data. The analysis technique was similar to that previously used for array design in Phase I.

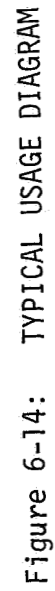
Flow charts or usage diagrams based on test article flows were prepared to show the use of GSE and the interface with test articles and operations. A typical usage diagram is shown in Figure 6-14. The diagrams are based on test article flow, because the objective of the FMEA is to identify risks to the test article or test article data.

The FMEA data sheet format has been modified for GSE; a typical example is shown in Figure 6-15. Known human-error risks and externally caused conditions have been included on these data sheets. The purpose has been to indentify these risk areas, in addition to the customary hardware failure risks, so that test personnel may take adequate precautions.

Only a preliminary FMEA for the MTA-4 deployment and handling equipment has been included in this Phase II effort.

Significant risks noted in the deployment equipment FMEA include:

- 1) Air gusts---It will be necessary to enforce strict supervision of entrances to the high-bay test area during deployment test operations;



Equipment Category MTA-4 (Preliminary)										
Equipment Identification (Drawing Number, Nomenclature, Etc.)	Usage (List Specific Tests or Solar Array Operations)	Functions (Including Specification Limits if Applicable)	Failure Mode and Causes	How Can Failure Be Observed?	Effects of Failure Mode (Including Likelihood Rating) ^a		Possible Alternatives or Corrective Actions	Notes		
					On Equipment	On Array or Test Article				
MTA-4	MTA-4 Deployment Tests	Simulate one panel assembly of array during ground deployment and TASC tests	Damage caused by air gusts	Visual if damage is severe, undetected damage possible		Damage array	P	Strict supervision at en-trances to high bay area to keep them closed during deployment will be	These are the three failure modes MTA-4 which are peculiar to the test procedure and independent of test equipment	
UNDERCARRIAGE SUPPORT SYSTEM	MTA-3 & MTA-4 Deployment Tests	To provide as close to a no drag support as possible for array deployment	Damage caused by stowing array at too fast a rate	Visual if damage is severe, undetected damage possible		Damage array latches	C	Proper closure rate is given in operating instructions		
			Damage caused by physical stopping of array by any means at too fast a rate	Visual if damage is severe, undetected damage possible		Damage array	C			
			Tipping a) During loading b) During deployment (Human Error) Running into guard rail (Human Error)	Damage to sled and track. Damage to sled track.	P				Tests will be run before MTA-4 deployment to show that these three failure modes should not occur	Proper setup of equipment will negate possibility of tipping
			Stabilizing clamps losing grip on array Losing air supply Defective Hydra-Set drops array	Damage to sled and track (if tipping occurs) Failure	C C C					
Hydra-Set	MTA-4 Deployment Tests	Set array onto air sled	Losing air supply	Air sled stops		Damage array	C		Standard shop equipment is periodically tested as part of shop procedure	
Tender Sled	MTA-4 & MTA-3 Deployment Tests	To provide a source of air for the air sled	Losing air supply Failure to stay close enough to air sled (Human Error)	Tender stops Visually by handler of tender sled or test observer		Test stopped Invalidate test data	C P		The operating instructions tell how tender should be handled. It is possible for the tender to get out of the specified range without notice, thus providing bad data	
General Note										
Except as noted, spreaders are not available. A failure which results in damage to the equipment would either cause the test to be stopped for days or weeks until repair or replacement was accomplished, or cause the test procedure to be altered to do without the damaged piece of equipment										
^a Likelihood Rating Code										
C - Certain: i.e., greater than 90% chance of effect resulting if failure mode occurs.										
P - Possible: i.e., from 10% to 90% chance of effect resulting if failure mode occurs.										
R - Remote: i.e., less than 10% chance of effect resulting if failure mode occurs.										
SD 2-7720-0-3 11/67										

- 2) Defects in track;
- 3) Loosening of potentiometer attachments;
- 4) Tipping, collision, or improper support during loading of the test articles into the deployment fixture or during deployment;
- 5) Failure to follow air sled with tender;
- 6) Improper adapter installation to cause binding.

Human error and externally caused conditions were found to constitute the major risk areas for handling equipment. Some of the risks are:

- 1) Interference of protruding parts of the protective covers with the subpanel during installation or removal of the covers;
- 2) Dropping, upset, or collision caused by error in transporting the handling frame or dolly;
- 3) Collision of other vehicles with the unprotected subpanels;
- 4) Defects in the GSE, such as omission of support pads and out-of-tolerance conditions;
- 5) Failure to use clamps and other pieces of equipment properly.

Where practical, the FMEA results have been incorporated in equipment operating instructions and test procedures to minimize the possibility of damage to the test articles.

7.0 SCS-43 SUBPANEL VIBRATION TEST PROGRAM

The SCS-43 subpanel model was designed, fabricated, and tested in Phase I to determine the panel response when subjected to a vibration environment in air. Also in Phase I, analyses were verified by test for the lowest structural frequency, but the assumed damping was much lower than the high damping that resulted from the tests in air. This high damping was believed to have been caused by air effects on the substrate.

Therefore, during Phase II, supplemental vibration testing in vacuum was requested by JPL to determine the effect of vacuum on frequency and damping.

The SCS-43 vibration tests and objectives are summarized as follows.

VIBRATION TESTS

<u>Title</u>	<u>Edge Support</u>	<u>Overall Objectives</u>
Ambient Vibration Test 1 (SCS-43-VA1)	Clamped	1) Evaluate the effect of storage time on substrate frequency.
Vacuum Vibration Test 1 (SCS-43-VV1)	Clamped	2) Measure dynamic response characteristics of SCS-43 panel for comparison with analytical values.
Ambient Vibration Test 2 (SCS-43-VA2)	Simply Supported	3) Determine the effect of air and vacuum on panel damping.
Vacuum Vibration Test 2 (SCS-43-VV2)	Simply Supported	4) Determine the panel's capability to withstand high-level vibration input.

POWER DEGRADATION TESTS

<u>Title</u>	<u>Objective</u>
Power Degradation Test A	Measure power output prior to SCS-43-VA1 test.
Power Degradation Test B	Measure power output after SCS-43-VV2 test to determine any power degradation caused by the vibration tests.

7.1 DESCRIPTION OF TEST SYSTEM

An industry "first" was established when a Ling 249 vibrator was installed in the 39-foot-diameter Boeing vacuum chamber to vibration-test the SCS-43 subpanel.

The SCS-43 vibration test system primarily consisted of a 30,000-pound-force Ling 249 vibrator (15 ton) and the 50,000-cubic-foot (39-foot-diameter) Boeing vacuum chamber. It is believed that this was the first time in industry that a vibrator of this force range had been placed within a vacuum chamber (see Figure 7-1) to conduct vibration tests.

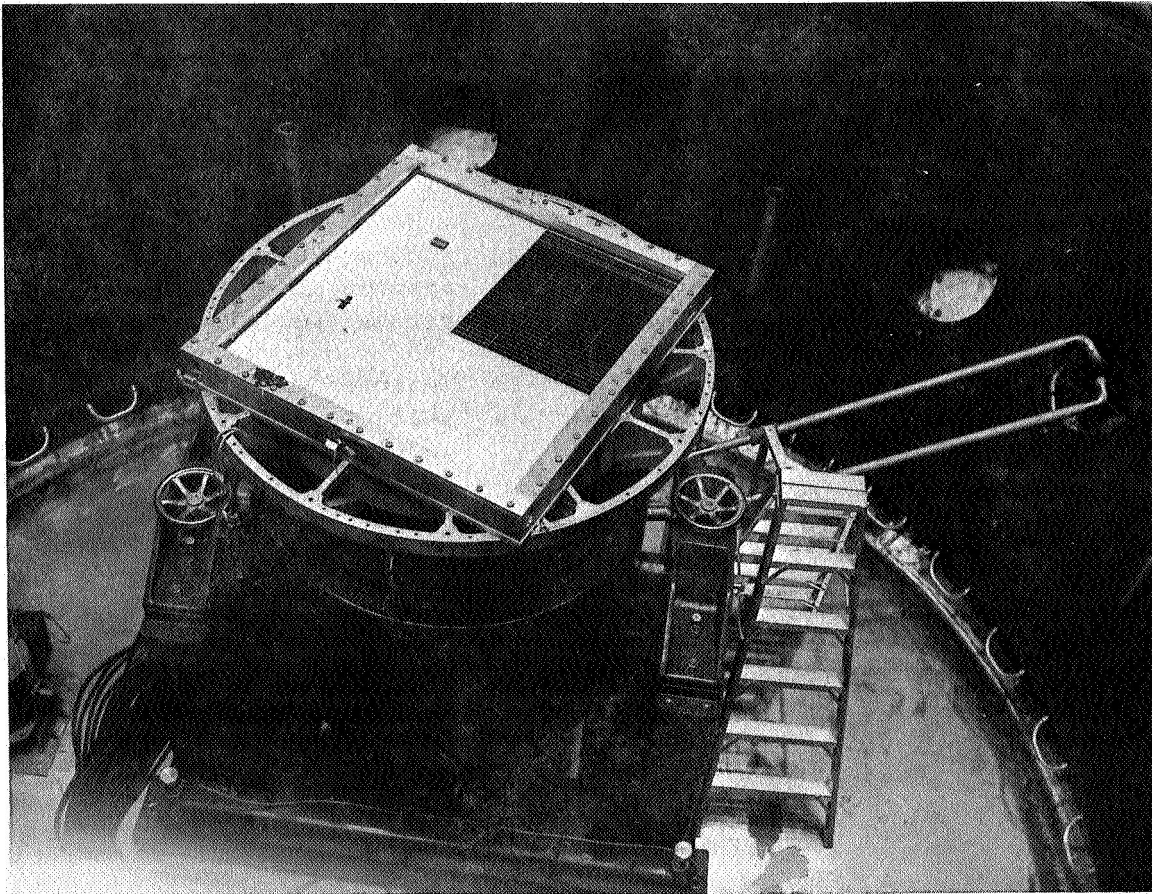
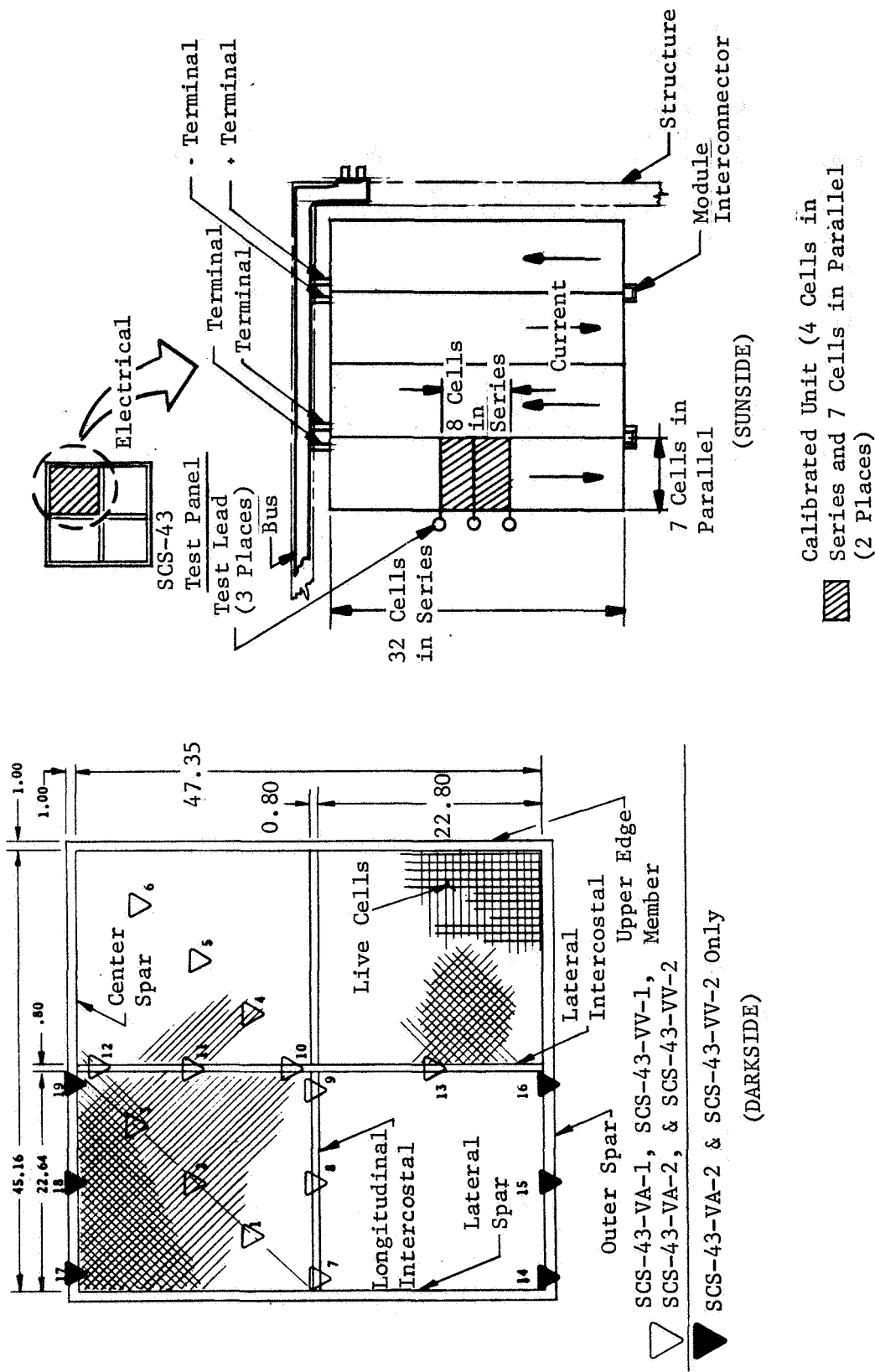


Figure 7-1: TEST SETUP---CLAMPED CONFIGURATION

The SCS-43 test article (see Figure 7-2) consists of hollow rectangular beryllium edge members and intercostals that support the pretensioned fiberglass tape substrate. The test article is approximately 4 feet square. One quarter of the panel area is covered with 896 live 8-mil cells (2 by 2 cm) and the remaining area has simulated microsheet glass cells (2 by 2 cm). The live-cell schematic is shown in Figure 7-3. Prototype buses and diodes were used.



The Ling 249 vibrator was used to excite vibration normal to the SCS-43 panel with two different edge conditions. The edges were fully clamped during VA1 and VV1 testing (see Figure 7-1) and then the panel was modified with two opposite edges simply supported during VA2 and VV2 testing (see Figure 7-4). The reason for this modification was to determine if there was a significant difference in the panel dynamic response characteristics between the pinned and clamped support. The simply supported configuration consists of three out-of-plane steel flexures on each of the two opposite edges and two in-plane flexures bonded along one edge to prevent lateral motion of the panel. The flexures were designed to provide minimum-bending resistance to the edge-member rotation. Figure 7-5 shows a typical in-plane and out-of-plane flexure installation.

The complete test panel with instrumentation and flexures is shown in Figure 7-6. Specific locations and applicable tests are shown for strain gages and accelerometers in Figures 7-2 and 7-7, respectively. Table VII-1 lists specific accelerometer types used and their respective weights. All vibration data were recorded on Sangana-type recorders. Two 14-channel recorders were required for the VA1 and VV1 tests and three recorders for the VA2 and VV2 tests. A 14-channel visicorder recorded selected channels of accelerometers and strain gages for quick-look evaluations, and a twin-beam oscilloscope and averaging meter were used to peak the selected resonances.

The SCS-43 test panel power output measurements were made using the X-200 solar simulator (Figure 7-8) and associated equipment.

7.2 VIBRATION TEST

Test equipment installation in the vacuum chamber required special care in development of procedures.

Because this was the first time that a 30,000-pound-force Ling 249 vibrator was installed in the 50,000-cubic-foot vacuum chamber (see Figure 7-1), special precautions were taken to check out the test system and procedures in air and partial vacuum (40 torr). The amount of cross-axis coupling of the test fixture during vibration testing was determined to be insignificant.

The SCS-43 clamped configuration (VA1 and VV1) and simply supported configuration (VA2 and VV2) were vibration-tested in air and vacuum. Sinusoidal sweeps were made at the prescribed g level. Following each level of excitation, a frequency in the lower band of frequencies was peaked and a decay record obtained. All tests were conducted in accordance with vibration test procedures provided in Boeing Document D2-113350-2.

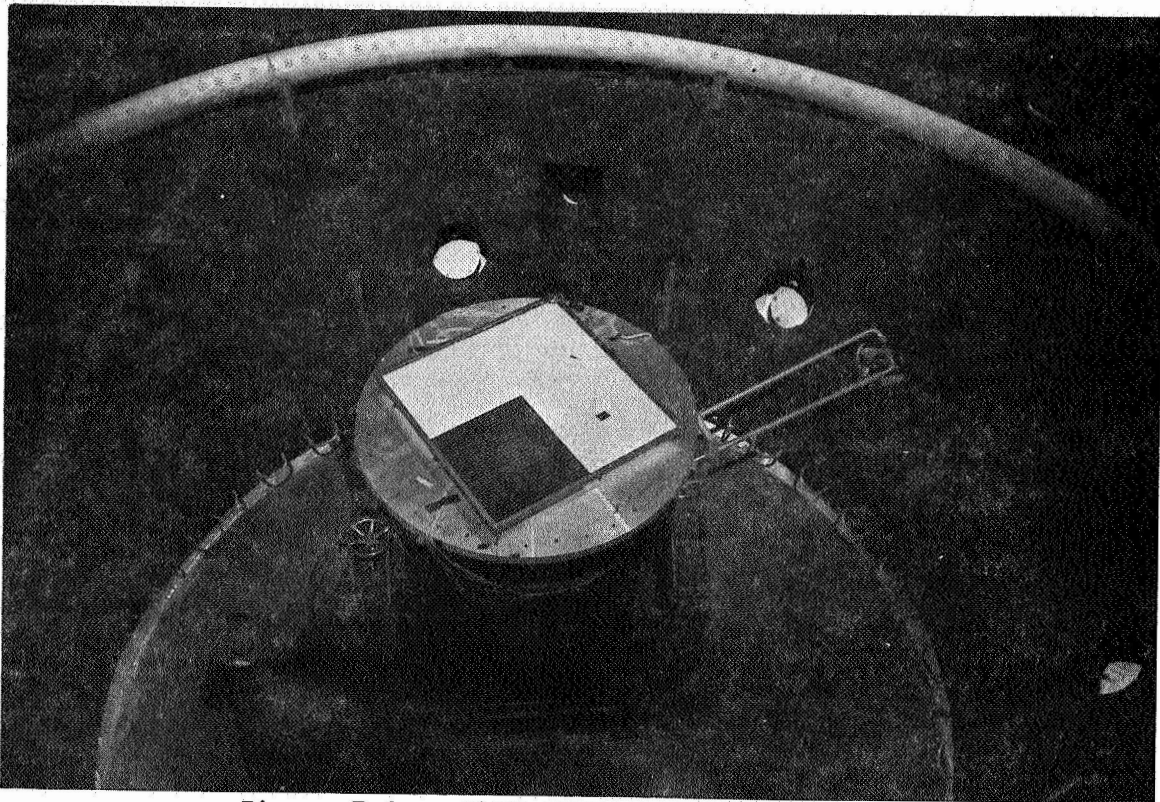


Figure 7-4: TEST SETUP---PINNED SUPPORT

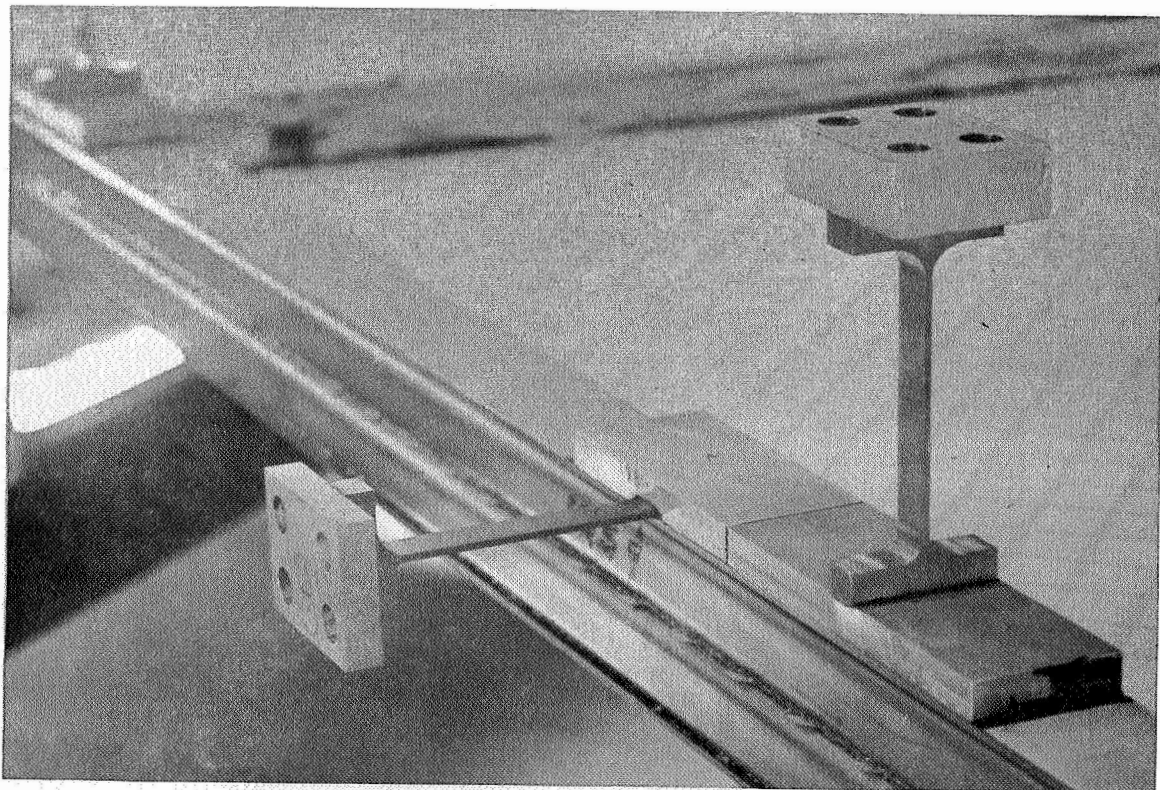


Figure 7-5: FLEXURAL SUPPORT

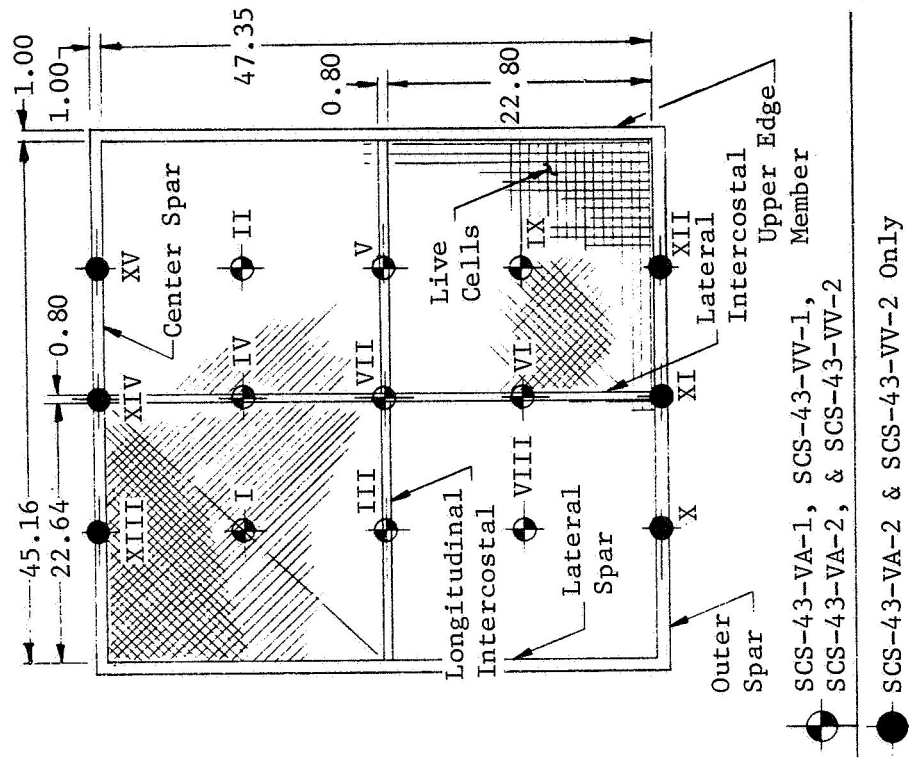


Figure 7-7: SCS-43 ACCELEROMETER LOCATIONS

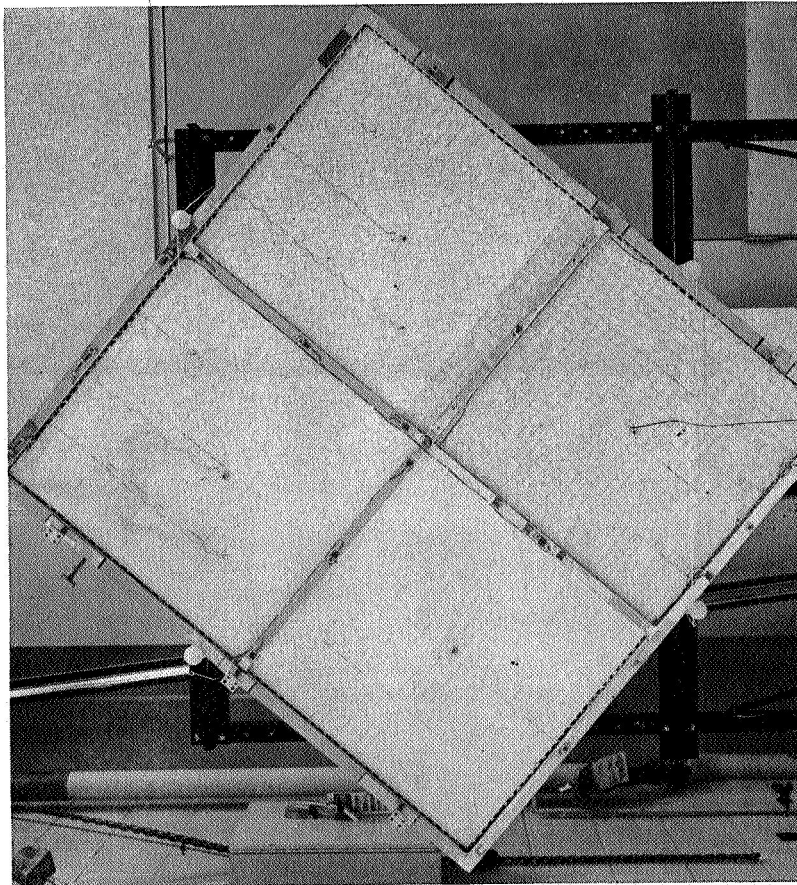


Figure 7-6: SCS-43 INSTRUMENTATION

Table VII-1: ACCELEROMETER TYPES AND WEIGHTS

CLAMPED CONFIGURATION

<u>Location</u>	<u>Accelerometer Type</u>	<u>Weight</u>
Accelerometer I	606-2	1.5 grams + 2.5-gram pigtail
Accelerometer II	606-2	Same as above
Accelerometer III	2226C	2.8 grams
Accelerometer IV	2226C	2.8 grams
Accelerometer V	2226C	2.8 grams
Accelerometer VI	2226C	2.8 grams
Accelerometer VII	2221C	12 grams + 0.5-in. 6-32 screw
Accelerometer VIII	606-2	1.5 grams + 2.5-gram pigtail
Accelerometer IX	606-2	Same as above

PINNED CONFIGURATION

<u>Location</u>	<u>Accelerometer Type</u>	<u>Weight</u>
Accelerometer I	606-2	1.5 grams + 2.5-gram pigtail
Accelerometer II	606-2	Same as above
Accelerometer III	2226C	2.8 grams
Accelerometer IV	2226C	2.8 grams
Accelerometer V	2226C	2.8 grams
Accelerometer VI	2226C	2.8 grams
Accelerometer VII	2221C	12 grams
Accelerometer VIII	606-2	1.5 grams + 2.5-gram pigtail
Accelerometer IX	606-2	Same as above
Accelerometer X	2226C	2.8 grams
Accelerometer XI	2221C	12 grams
Accelerometer XII	2226C	2.8 grams
Accelerometer XIII	2226C	2.8 grams
Accelerometer XIV	2221C	12 grams
Accelerometer XV	2221C	12 grams

- NOTES: 1) Standard accelerometer cable weighs approximately 2.7 gm/ft;
6 lb/1,000 ft = 0.006 lb/ft = 28 gm/oz (0.006 lb [16 oz/lb])
= 2.7 gm/ft.
- 2) Add 0.25 gram for dental cement and plastic block to each accelerometer.

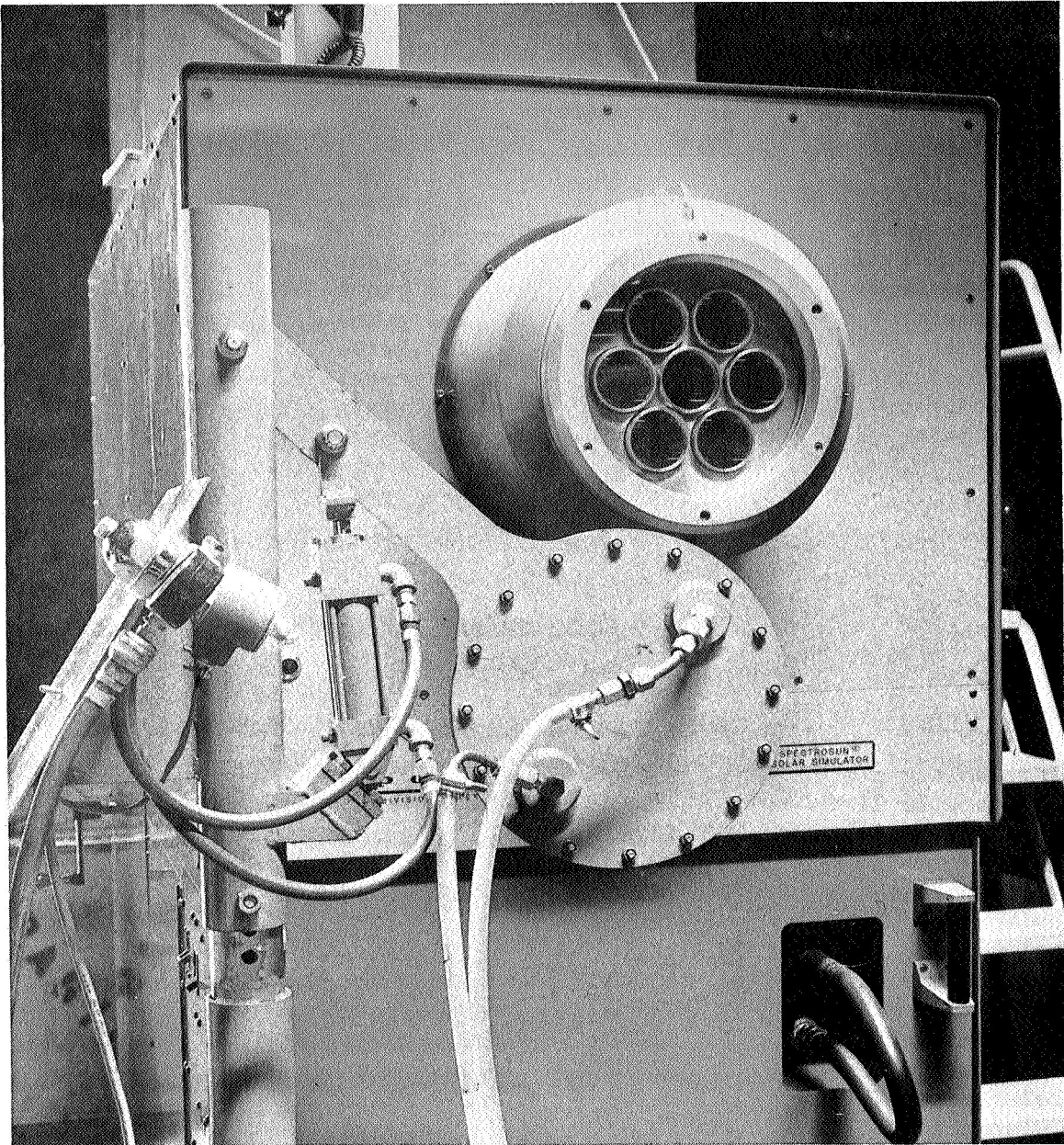


Figure 7-8: X-200 SOLAR SIMULATOR WITH OPEN SHUTTER

7.2.1 RESULTS AND ANALYSIS OF TESTS

The test results show small differences in dynamic characteristics due to panel support, but significant differences were noted between an air and vacuum environment.

Remeasurement of the Phase I SCS-43 test panel showed no significant difference in the panel dynamic characteristics due to the 8- to 9-month

interval between Phase I and Phase II testing. During this interval, the panel was packaged, transported to JPL, electrically tested by JPL, and returned to Boeing.

The predicted grouping of substrate frequencies in a band around 35 Hz (see Figure 7-9) in air and 45 Hz (see Figure 7-10) in vacuum is substantially validated. The existence of a multiplicity of closely spaced modes predicted to be in a band of frequencies between 70 and 100 Hz was found experimentally for the vacuum tests, but specific predicted high frequencies were not identifiable with corresponding frequencies in the tests. The spread of frequencies is greater in the tests than in the theory.

The effect of air on damping is a major result of this experiment. The storage of 8 to 9 months did not appreciably alter the damping characteristics of the panel. The reduction of air pressure from ambient to 40 torr caused changes in frequency and damping for both the clamped and pinned configurations (see Figure 7-11). The effect of reduced air pressure was to reduce the damping in the worst case to 1/10th of its value in air. The worst values approached, but did not become as small as, the values for the 13-foot beryllium spar (SCS-50) tested in Phase I.

A structural failure occurred in an intercostal cap strip (see Figure 7-17) that had suffered damage during fabrication. This work occurred after more than 100,000 cycles at stress levels above 9,000 psi. No major change in frequency or structural capability nor damage to the remainder of the intercostal was observed.

7.2.2 CONCLUSIONS

The SCS-43 test program successfully demonstrated its objectives.

- 1) The 8- to 9-month storage and transportation environment had no effect on the panel dynamic characteristics.
- 2) Comparison of calculated and measured frequencies was acceptable in general (see Section 7.2.1).
- 3) Appreciable reduction of damping was observed at 1/20th atmosphere. The lowest damping values were still greater than the values for the 13-foot beryllium spar (SCS-50).
- 4) The panel suffered no major structural damage and that which did occur did not significantly affect the structural or electrical characteristics of the panel.

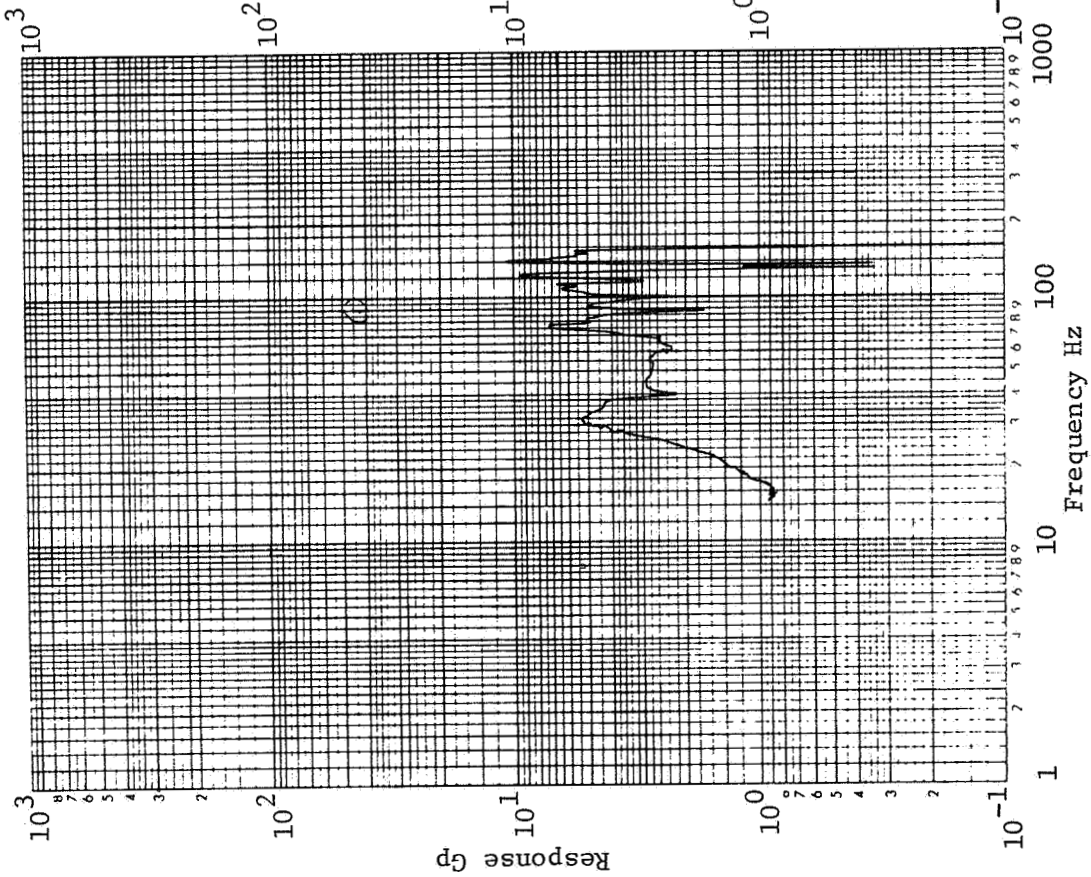


Figure 7-9: MEASURED ACCELEROMETER
RESPONSE IN AIR

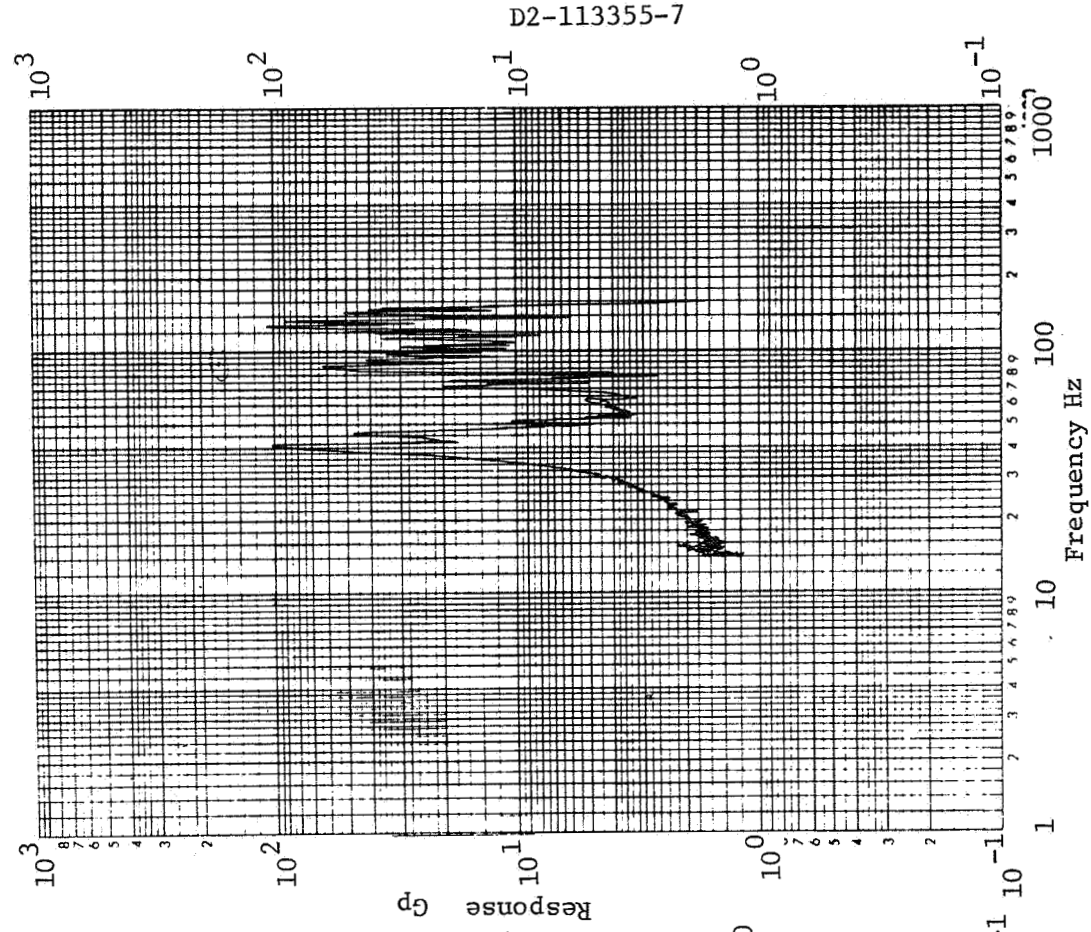


Figure 7-10: MEASURED ACCELEROMETER
RESPONSE IN VACUUM

D2-113355-7

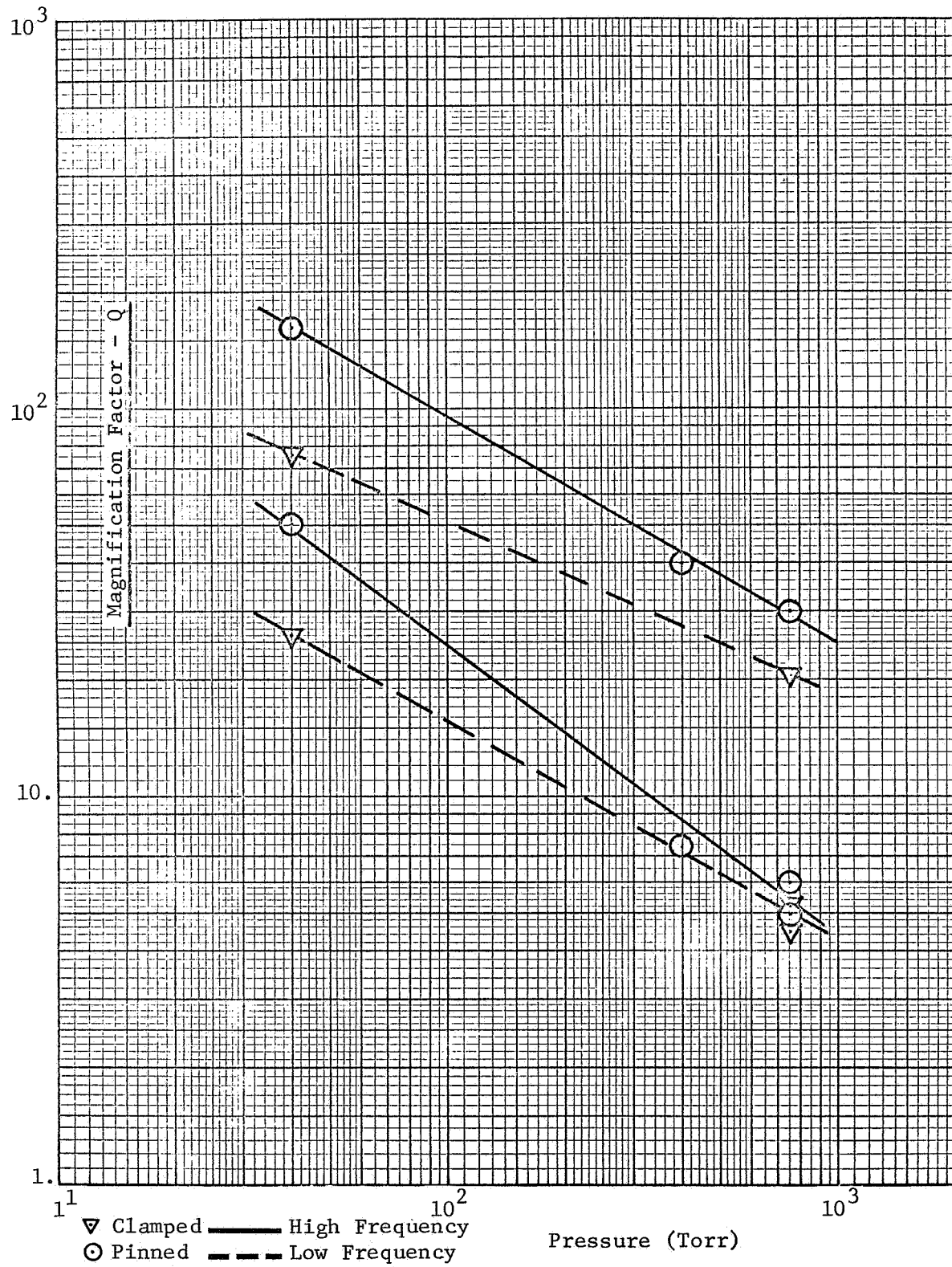


Figure 7-11: EFFECT OF PRESSURE ON DYNAMIC COEFFICIENT Q

7.3 POWER DEGRADATION TESTS

Power output test results and analysis provided sufficient information to evaluate the electrical characteristics of the SCS-43 panel due to vibration testing in air and vacuum.

Before and after vibration tests, the electrical output of the SCS-43 functional test panel was measured while illuminated with simulated space sunlight having an intensity of 140 milliwatts per square centimeter. The intensity and spectrum of this illumination was measured before each test. The solar-panel temperature was measured during this electrical performance test, and the open-circuit voltage, short-circuit current, and maximum-power readings were adjusted to correspond to 55°C values.

A Spectrolab X-200 solar simulator was used to illuminate the solar panel during the power degradation tests. The intensity in the illuminated area was mapped with a Boeing intensity scanner and was found to be uniform within $\pm 2\%$ (Figure 7-12). The intensity of illumination was measured with JPL balloon-flown Standard Cell Number 301, located as shown in Figure 7-12.

The spectrum of the simulated space sunlight was measured with a Beckman W139323 spectroradiometer. Within the solar-cell sensitivity range of 0.35 to 1.5 microns the spectrum of the simulated sunlight was within 17.7% of the natural space sunlight when compared in 0.1-micron bands. Natural sunlight in space was considered to be that defined in NASA SP-8005, *Solar Electromagnetic Radiation*, which is based on Johnson's data.

Panel temperatures during testing were measured with thermocouples. As shown in Figure 7-13, the thermocouple temperatures for Test A (before vibration) and Test B (after vibration) were nearly identical; however, there was a temperature difference of up to 25°C between thermocouple readings at the top and bottom of the panel. Therefore, the effective temperature of the panel was determined from open-circuit voltage. Room-temperature, open-circuit voltage was measured an instant after the illumination was turned on. The open-circuit voltage was again measured 15 minutes after the solar simulator was turned on, after panel temperatures had stabilized. Coefficients for correcting open-circuit voltage, as well as short-circuit current and maximum power, to a standard temperature of 55°C were supplied by J. Sandstrom of JPL.

Important characteristics of the SCS-43 test panel are summarized in Table VII-2.

Test Results and Analysis

The SCS-43 test panel was examined for damage before and after vibration tests. As shown in Figure 7-14, there was cell damage during Phase I testing that, of course, was not attributable to the Phase II vibration test.

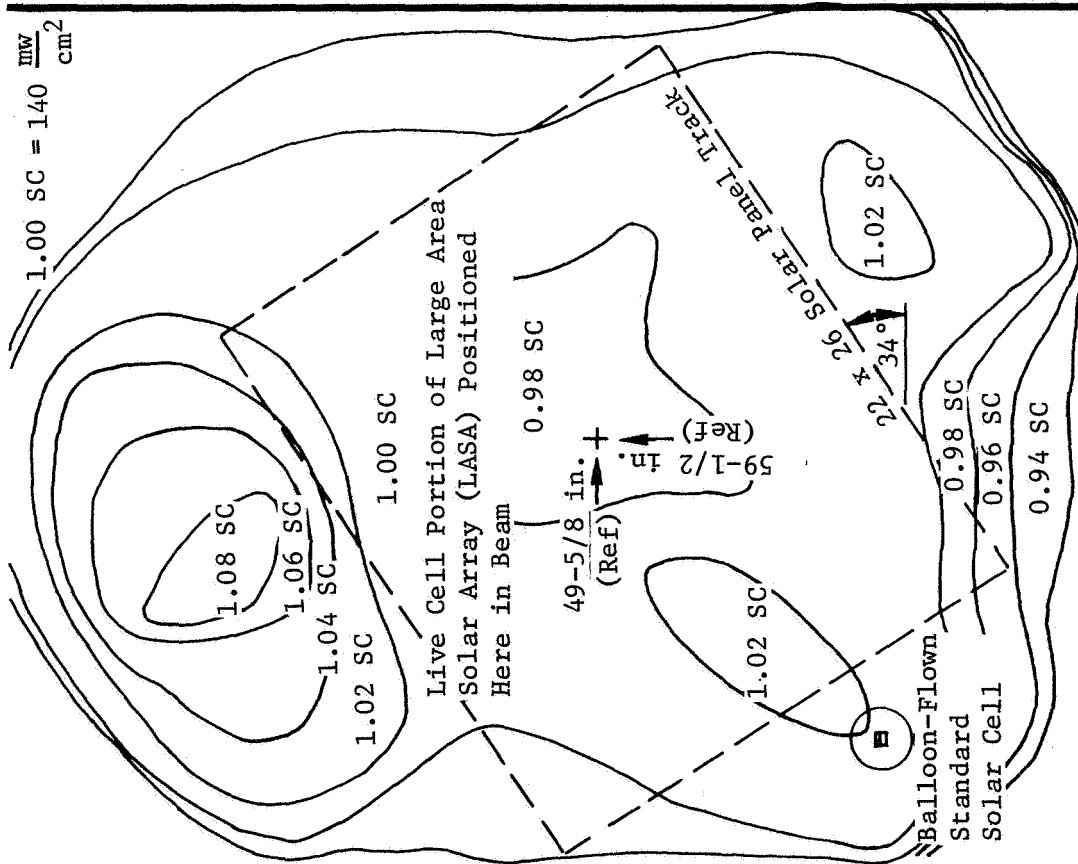


Figure 7-12: ISOSOLAR PLOT OF THE X-200 SOLAR SIMULATOR

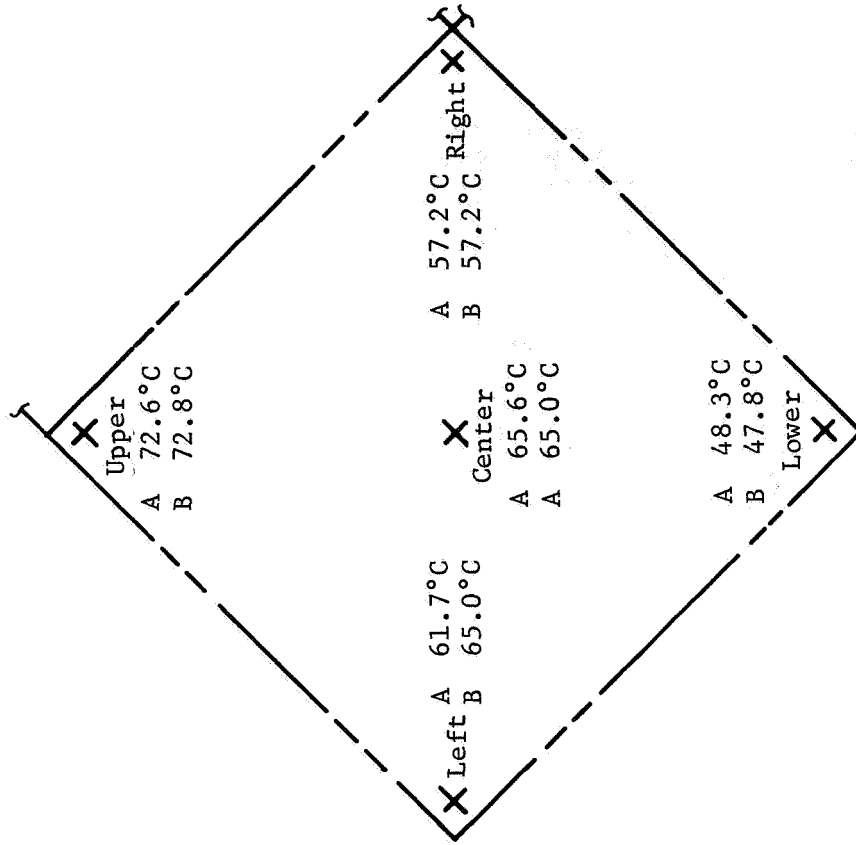


Figure 7-13: TEMPERATURE PROFILE OF SCS-43 TEST PANEL POWER OUTPUT TESTS A AND B

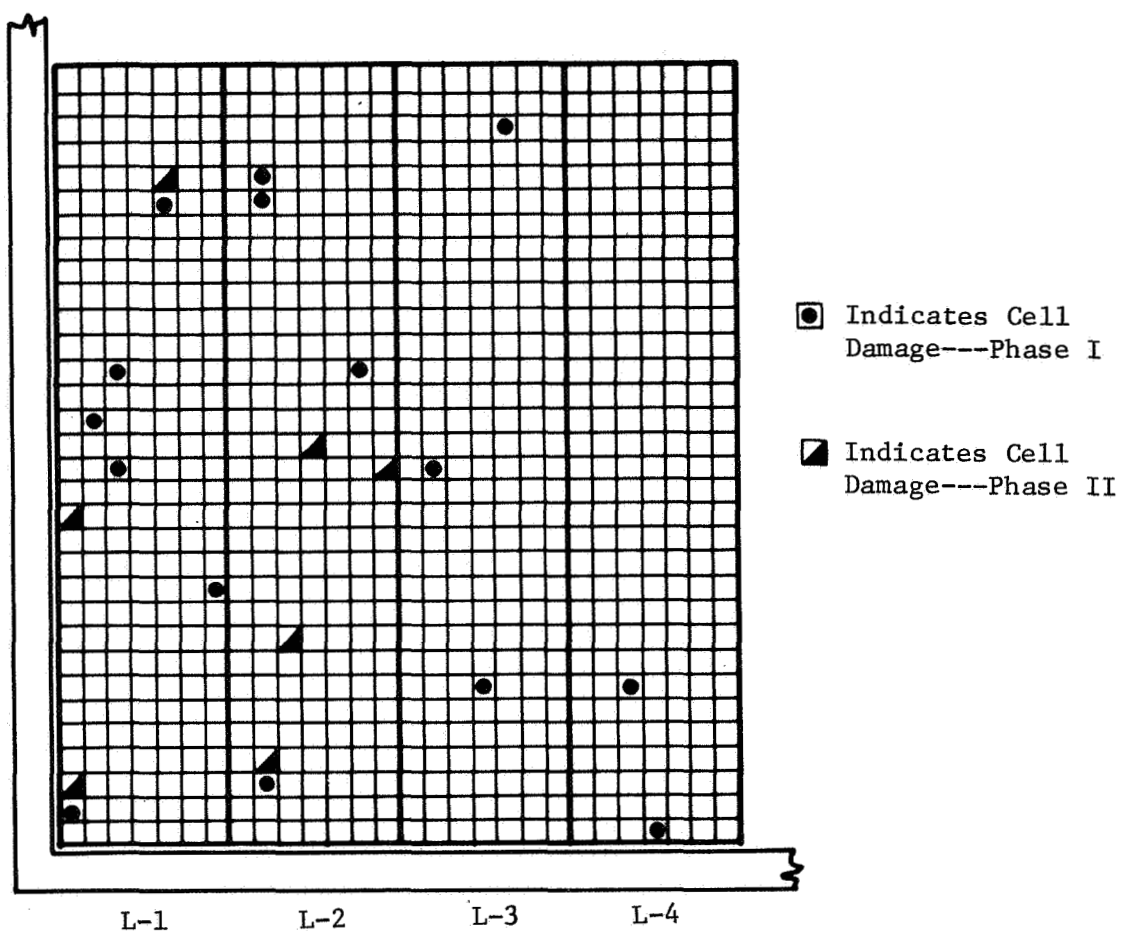


Figure 7-14: DAMAGED CELLS, LIVE CELL AREA---SCS-43 TEST PANEL

Table VII-2: SCS-43 TEST PANEL---ELECTRICAL CHARACTERISTICS

Size of Panel	4 by 4 feet
Size of Area with Solar Cells	2 by 2 feet
Solar-Cell Size	2 by 2 cm
Solar-Cell Thickness, Nominal	8 mil
Cells in String	32 series-connected 7-cell parallel groups
Number of Strings in Panel	4
Number of Cells in Panel	896

One cell was found to have been damaged during the vibration test. In addition, two interconnectors between strings that had not been properly bonded-down to the substrate were damaged in vibration testing. These connectors were repaired before the postvibration electrical performance test.

Panel performance before and after the vibration test is shown in Table VII-3. The maximum power values shown are from the volt-ampere curves shown in Figures 7-15 and 7-16, adjusted to correspond to 55°C, as previously described.

Table VII-3: PANEL PERFORMANCE BEFORE AND AFTER VIBRATION TEST

Performance Measured	Open- Circuit Voltage (volts)	Short- Circuit Current (amperes)	Maximum Power Output (watts)	Maximum Temperature Difference Across Panel (°C)	Panel Temperature (°C)	
					Avg	Max
Before Vibration (Test A)	31.4	1.62	36.6	24.3	58.8	72.6
After Vibration (Test B)	31.4	1.60	36.0	25	58.2	72.8

Test Conditions

Illumination intensity:	140 mw per sq cm
Illumination uniformity:	Within ±2%
Standard cell used:	JPL, balloon flow, No. 301

Conclusions

The maximum power point of the SCS-43 test panel after the vibration test was 1.6% below the pretest value. This difference in maximum power is within the experimental error of the test setup and cannot be considered significant.

Likewise, the change in short-circuit current caused by the vibration test was within experimental error. There was no change in the adjusted open-circuit voltage.

7.4 QUALITY ASSURANCE

Timely inspections of SCS-43 noted panel defects and their causes.

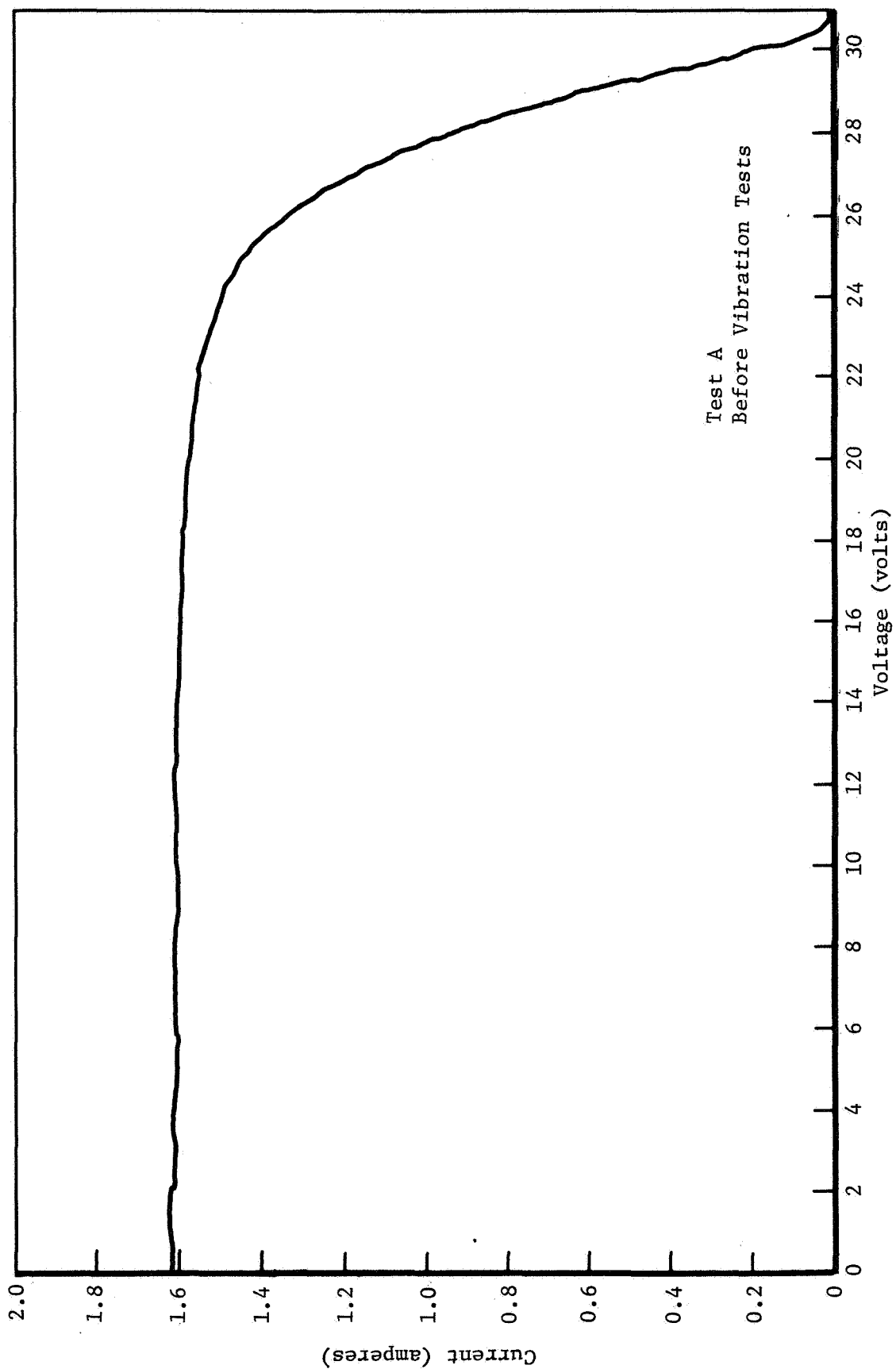


Figure 7-15: POWER DEGRADATION TEST---SCS-43

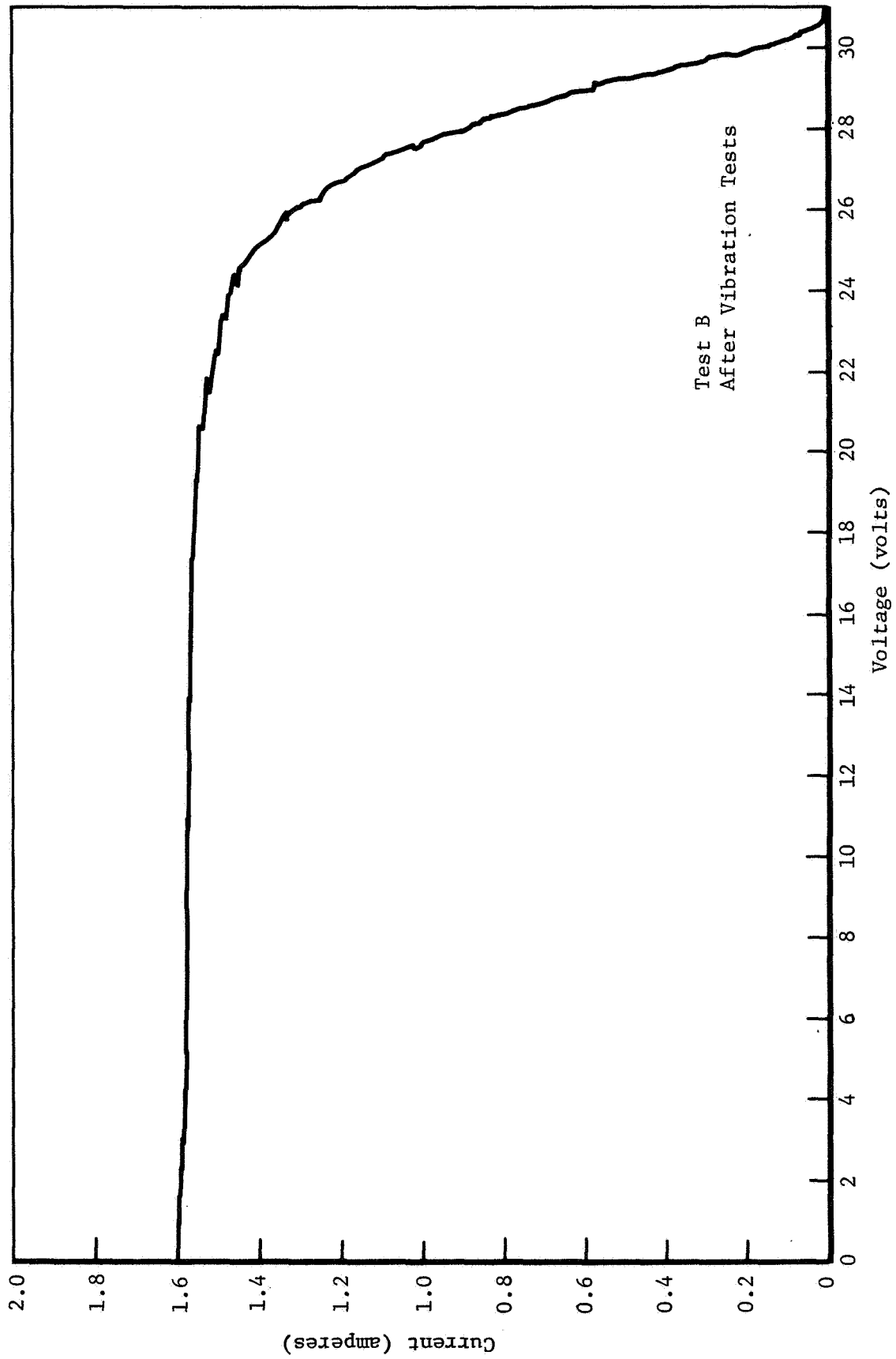


Figure 7-16: POWER DEGRADATION TEST---SCS-43

Test panel SCS-43 was processed through receiving-inspection immediately on return from JPL and in preparation for Task 8 testing. After each VV1 and VV2 environmental test, it was subjected to complete visual inspections, and any posttest defects were recorded.

Original receiving-inspection indicated a loss of tension in one substrate tape strip in the lower right-hand corner of the panel in addition to one cracked simulated cell. These were the only defects in addition to those that existed before shipment to JPL at the conclusion of Phase I. Phase II processing and handling of SCS-43 resulted in seven additional broken cells (see Figure 7-14). (Sixteen cells were broken at the conclusion of Phase I.) No additional defects were noted after the VV1 vibration test. Inspection after the VV2 test revealed cracked intercostal cap (see Figure 7-17) on the panel darkside. (Edges on this beryllium strip were broken in original bonding operations.) In addition, one cell solder joint and an interconnector were broken (see Figure 7-18).

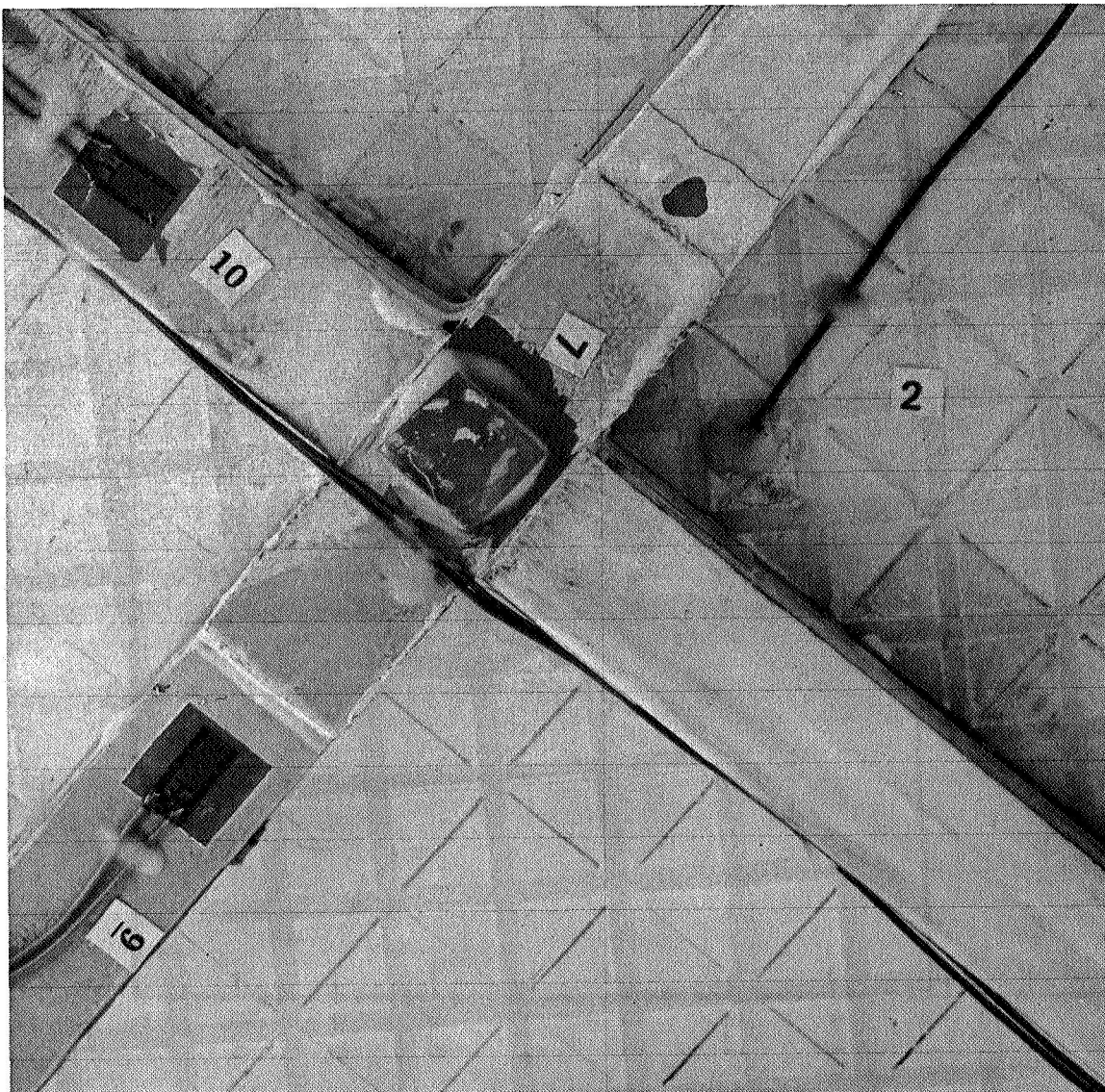


Figure 7-17: INTERCOSTAL CAP CRACK

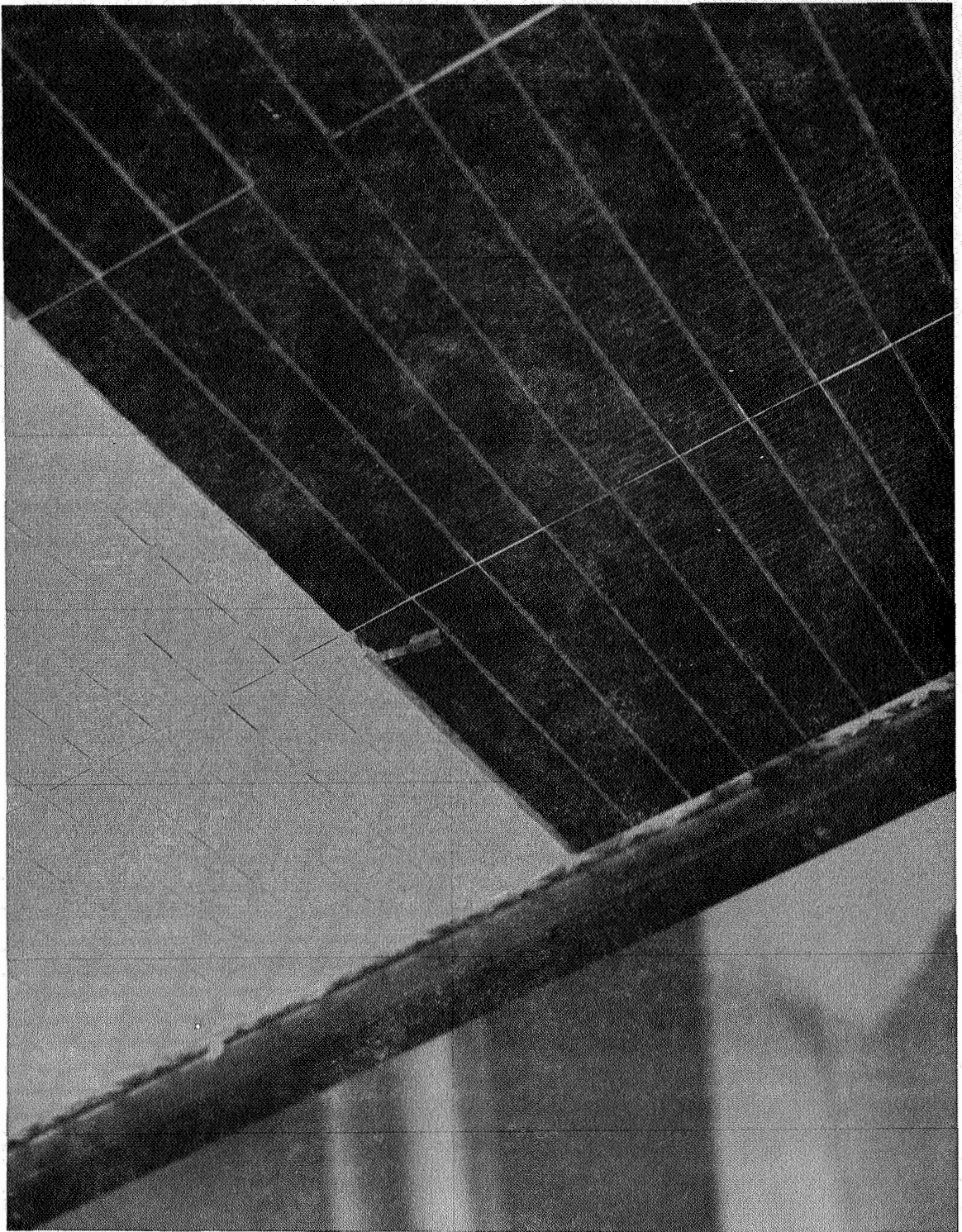


Figure 7-18: SILVER MESH INTERCONNECTOR FAILURE

8.0 MATERIEL PROGRAM SUPPORT

Continuous Materiel support contributed to on-schedule delivery of material, parts, and services, and resulted in a total program savings in excess of \$58,000.

8.1 PROCUREMENT ACTIVITIES

Purchased parts, components, outside production, and raw materials were obtained in support of program requirements and were completed within established program budget. A display of budget expenditure by type is shown in Figure 8-1.

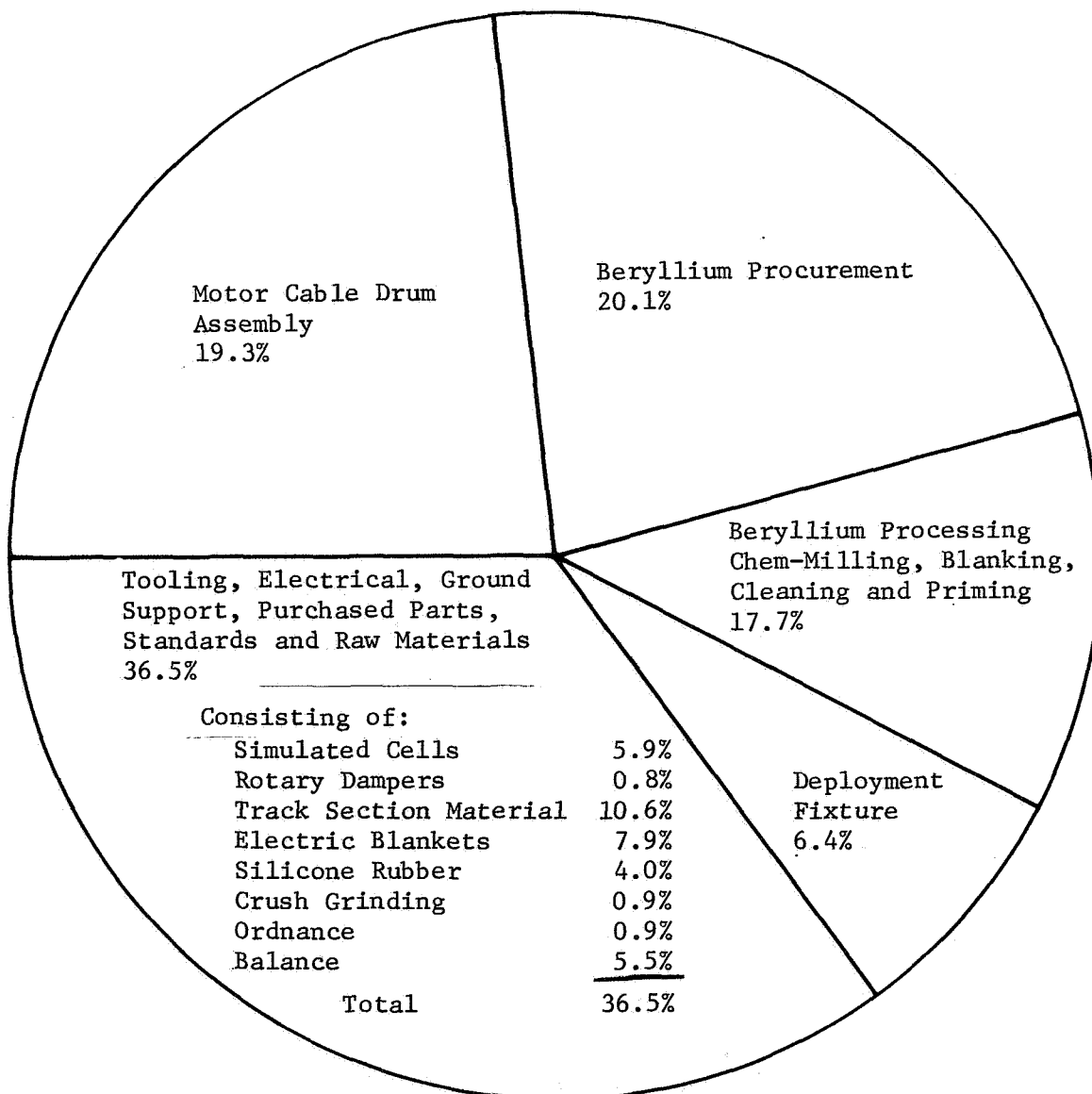


Figure 8-1: MATERIEL PROCUREMENT BY PERCENT OF TOTAL DOLLARS SPENT

Purchased parts, components, outside production, and subcontract items were procured by the program Materiel representative with assistance as required by Space Division purchasing specialists. Titanium, beryllium, aluminum, primers, miscellaneous raw materials, and standards were purchased by the Aerospace Group Central Procurement units. Costs and schedule performance for these items was monitored by the program Materiel representative located in the project area.

Purchase orders for beryllium sheet stock were placed with Berylco and Brush Beryllium at the start of Phase II. One shipment of beryllium stock was received from Brush Beryllium with apparent watermarks or some similar type discoloration. Because of a tight manufacturing schedule, the material was accepted and cleaned at Boeing. Brush was notified of the discrepancy and agreed to process material through their laboratory to simulate and identify the cause of the discoloration. This was accomplished at no cost to Boeing and results were forwarded to Boeing and JPL for review.

Chemical Energy Company in San Diego, California, was given purchase orders to process beryllium to D2-114048, *Requirements for LASA Purchased Beryllium Metal Surface Conditioning Services*. The Chemical Energy Company processed 900 parts in Phase II, which included blanked, chem-milled, and cleaned and primed parts. Based on original estimates and as a result of program refinements and supplier negotiations, there was a program saving of \$33,740 in this area. The cooperation of Chemical Energy Company personnel in processing beryllium requirements for the program was outstanding. Typical operations at the Chemical Energy Company are shown in Figure 8-2.

During the early part of the program, the validity of the manufacturing cost estimate to fabricate the boost tiedown shear teeth fittings was questioned and reviewed. During a meeting, it was suggested that a reduction in costs could be accomplished by utilizing a supplier crush grinding facility. Materiel surveyed the industry and located Roselle Tool and Die Company in Roselle, Illinois. A purchase order was placed with Roselle for two titanium fittings for evaluation. The completed fittings were evaluated and met the specifications, resulting in a buy decision. A production purchase order was placed with Roselle Tool and Die for a total program saving of \$6,900.

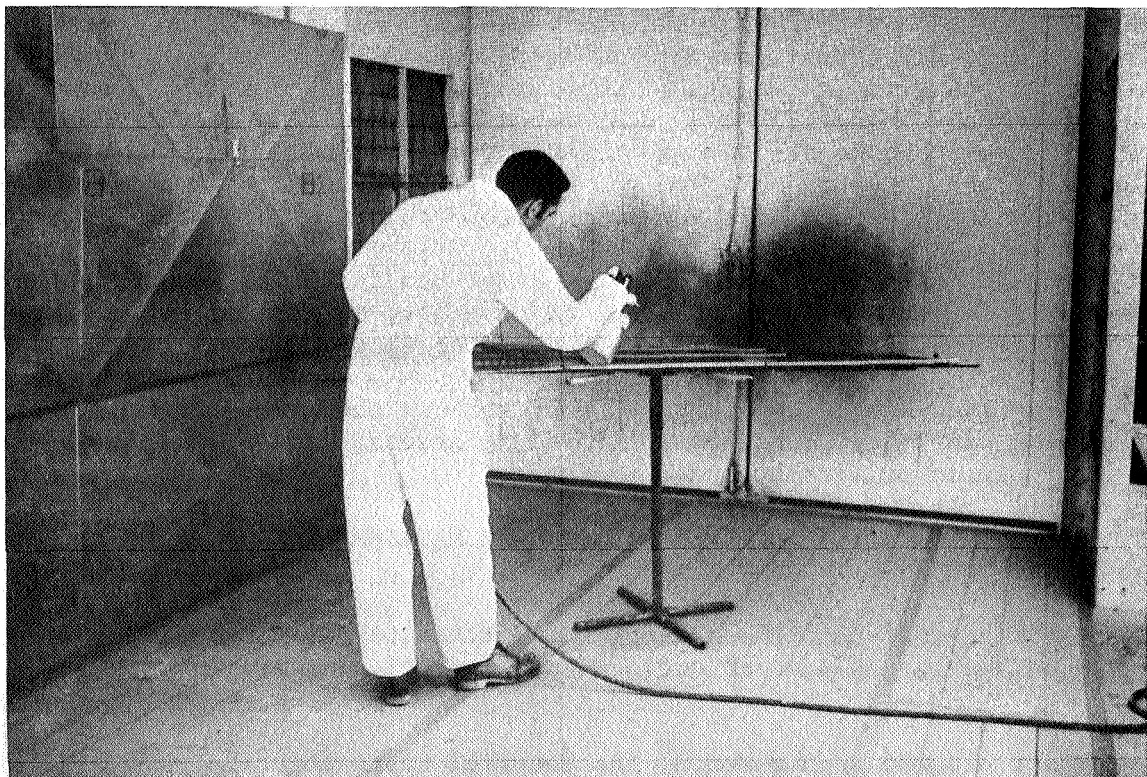
Parts requiring dry filming and lubrication were processed at the Industrial Plating Company in Seattle, Washington, one of the approved sources for this process. These parts were processed in 1 to 2 days as required to maintain the manufacturing schedule.

Electric blankets, required in the bonding cycle, were purchased from the Briscoe Company, Columbus, Ohio. Briscoe engineers designed and fabricated these blankets to meet Boeing requirements.

Based on a shortage of engineering manpower, shop loads, and inhouse estimates, a decision was made to buy a structure for panel deployment testing. Request for Proposals were forwarded to six qualified steel



Chem-Milling and Cleaning Facility



Spraying Primer on Cleaned Parts

Figure 8-2: BERYLLIUM PROCESSING OPERATIONS

fabricators. After the competitive bidding was reviewed, the Leckenby Company was given a contract to prepare shop details and fabricate the fixture to Boeing design. They not only met all schedule milestones, but accelerated the schedule by 2 months. The fixture was fabricated and installed for a total saving of \$10,239. The fixtures erection is shown in Figure 8-3.

The motor cable drum assembly, utilizing the harmonic drive principle, was designed, manufactured, and tested by the United Shoe Machinery Corporation. Other manufacturers were contacted, but did not respond to the Request for Proposal, because the harmonic drive principle is patented.

United Shoe Machinery proposed to design, fabricate, and test a qualification test unit and two production units for \$52,300. The specification was reviewed to eliminate or reduce certain requirements to lower the cost. As a result, the reliability requirements were reduced, which subsequently reduced the total cost to \$45,000 for a saving of \$7,300. A purchase order was issued to United Shoe Machinery, and forwarded to JPL for approval on August 11, 1967. Approval was received on October 5, 1967, and confirmed with United Shoe.

United Shoe encountered manufacturing problems in January and reported a potential 3- to 4-week slide. Materiel and engineering representatives immediately visited the facility to review and recommend a plan to accelerate this schedule. United Shoe implemented an all-out effort working overtime at no cost to Boeing and delivered one unqualified unit on March 2, 1968. This unit was available 2 weeks before testing. Qualification testing was subsequently completed with satisfactory results. The review of the qualification test report was completed July 2, 1968, and was acceptable in all but two areas. These areas required a revision to the test report only and did not affect the delivered units. Figure 8-4 contains photographs of the motor cable drum assembly.

Rotary dampers were purchased from the SESCO Manufacturing Company. The dampers were tested inhouse as SESCO did not have sufficient engineering or testing capabilities to produce to a Boeing specification.

Other miscellaneous parts were purchased through the central procurement group with costs and schedules monitored as necessary to ensure compliance to program requirements and goals.

8.2 SUPPLIER AND PROGRAM MANAGERMENTS

Supplier technical problems and potential schedule slippages were identified and worked in time to avoid program impact.

To support program requirements, the Materiel representative participated in all make-or-buy decisions, Change Board activities, program reviews, and ordered material or monitored other purchase order activity as necessary. Table VIII-1 summarizes all make-or-buy decisions accomplished during Phase II.

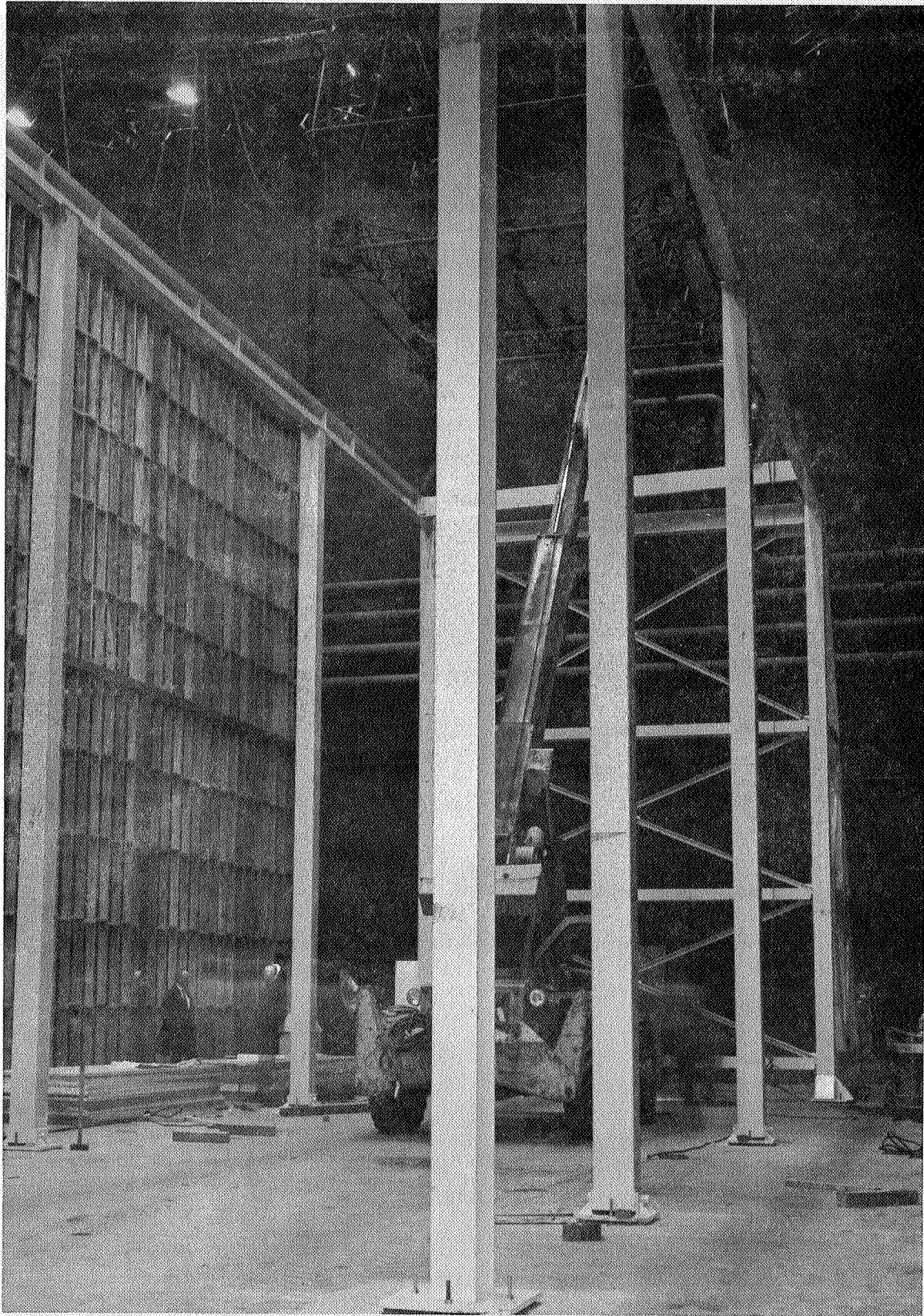
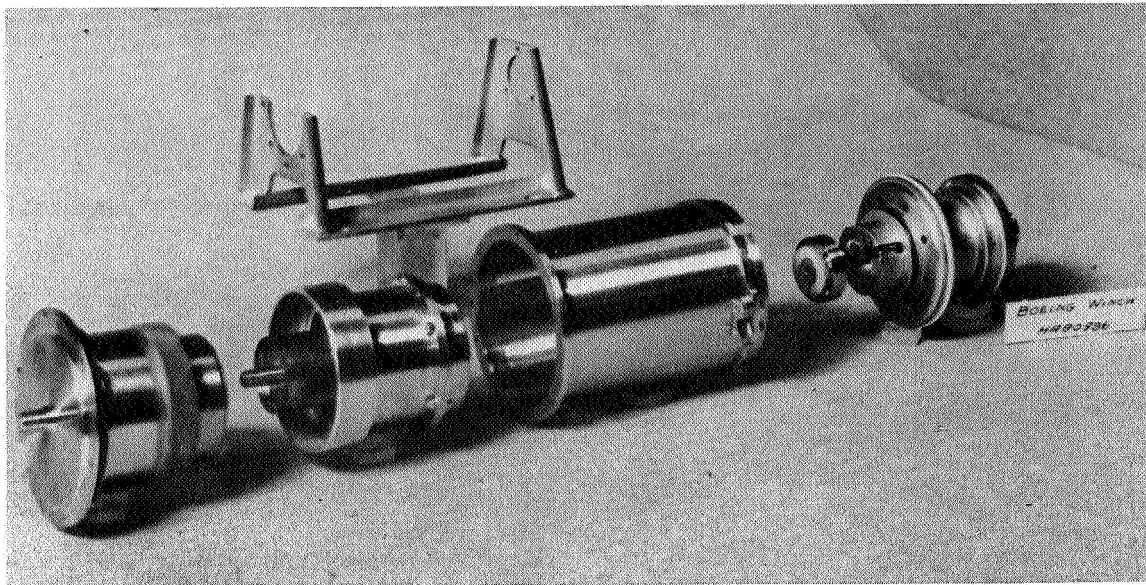
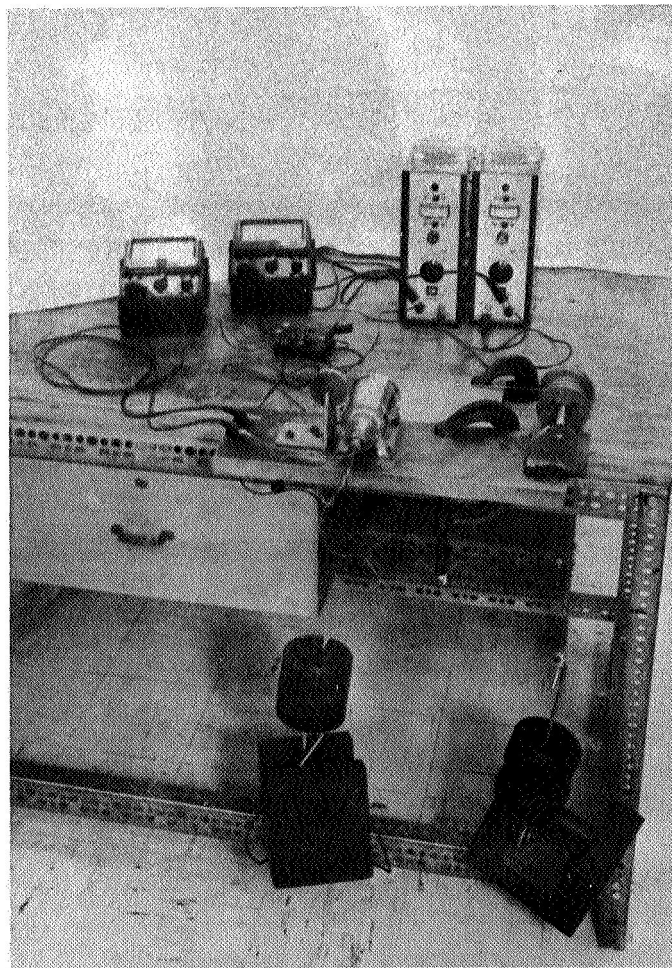


Figure 8-3: ERECTION OF DEPLOYMENT STRUCTURE



Exploded View



Test Setup

Figure 8-4: MOTOR/CABLE DRUM ASSEMBLY

Table VIII-1: LASA PHASE II MAKE-OR-BUY SUMMARY

Item	Make	Buy
Beryllium	Make	Buy to Boeing Specification
Beryllium Fabrication and Assembly		
Beryllium Chem-Milling, Etch-Clean, and Priming		Buy to Boeing Process Document
Fiberglass Tape		Buy to Boeing Specification
Microsheet Glass (Simulated Solar Cells)	Make	Buy
Solar Cells and Coverglasses		Customer-Furnished (CFE)
Solar-Cells Matching, Stock Installation and Test		
Panel Dampers		Buy
Machine Parts	Make	
Crush-Grinding of Boeing-Furnished Fittings	Make	Buy
Miscellaneous Raw Material		Buy
Miscellaneous Standards		Buy
Panel Deployment Structure		Buy
Other Ground-Support Equipment	Make	
Miscellaneous Purchased Parts and Components		Buy

Credit reviews were obtained on new suppliers or when large dollar procurement was anticipated. As a result, there was no subcontractor financial problems nor was assistance required.

Legal services were used when revisions to the terms and conditions of the purchase order were required. During Phase II, the only revisions negotiated were with United Shoe Machinery. These revisions were of a minor nature and a mutual agreement was reached between Boeing and United Shoe Machinery and coordinated with JPL.

Weekly activity reports were submitted to program and Materiel management personnel outlining program activity and potential problems. This enabled the management personnel to keep abreast of Materiel performance or activity and immediately assist when required.

Procurement activity reports were included in the program monthly progress reports to JPL.

8.3 MATERIALS MANAGEMENT

Material Control, Traffic, and Stores personnel supported Phase II by supplying services and materials as required.

Material requirement releases were screened by control personnel prior to release to the procurement group. If parts or materials were located in stores stock, requisitions were issued for the material. These requisitions were processed to the applicable stores area for issue of stock material.

Traffic personnel were responsible for routing and tracing all incoming materials.

9.0 PROGRAM EVALUATION SUMMARY

Phase II results have significantly increased confidence in the LASA concept for high-power, lightweight solar array development. A major part of the results includes better understanding of the several problems to be resolved in Phase III. This section contains a collation of the more important conclusions and specific recommendations to complete technology readiness by the end of 1970.

9.1 CONCLUSIONS

Phase II experimental fabrication and test results provide adequate data to proceed with fabrication and testing of MTA-4A in Phase III.

9.1.1 TWENTY-WATT-PER-POUND GOAL

MTA-1 actual weight data verified the Phase II weight calculations to an acceptable tolerance, and indicates a 21-watt-per-pound solar array.

The goal established for the array efficiency is 20 watts of power generated per pound of system weight for a solar array producing nominally 50 kw at 1 AU, air mass 0, and 55°C. The area of the array required to produce this power is approximately 5,000 square feet. Total expected power output for the array design is 45.9 kw, and total calculated system weight is 2,182 pounds. From this, it is concluded that technology developed for the LASA will result in the capability to produce arrays that meet this requirement by the end of Phase III.

9.1.2 FABRICATION AND ASSEMBLY RESULTS

Successful fabrication and assembly of Subpanel 1 (MTA-1) demonstrated manufacturing capability through the main subpanel level.

Assembly of a structurally sound, lightweight solar panel of bonded design has been demonstrated.

Fabrication of light-gage beryllium sheet material parts to the "machine" tolerances required for proper bonding and to avoid built-in stresses has been proved feasible.

Techniques developed for repair of lightweight bonded structure have improved the practical aspects of this assembly method.

Practical manufacturing methods for the assembly and installation of solar cells, to a pilot quantity level, on the LASA substrate have been demonstrated.

9.1.3 TEST RESULTS

Phase II testing has significantly improved assurance of satisfactory mechanisms systems.

The Phase II mechanism testing of the SCM series and MTA-3 provided valuable insight into specific items requiring continued attention. These may be grouped into three main categories:

- 1) Dimensional controls including hinge and latch alignment and clearances between moving parts;
- 2) Development of proper spring forces;
- 3) Control of friction effects.

The SCM-2 test of auxiliary panel mechanisms showed that the viscous dampers would not "freeze up" and prevent auxiliary panel deployment even if cooled to -100°F. Thus, risks of failure associated with excessive viscosity of the fluid used in these dampers appear to be negligible.

The motor/cable drum assembly used for main panel deployment appears to be a basically satisfactory unit. A qualification test was conducted and reviewed, and successful operation on SCM-1 and MTA-3 tests was accomplished.

Additional evaluation of the boost tiedown release linkage action, and the risk of a significant time delay, should be conducted in Phase III. The simulated testing of SCM 30-2 showed no impact in critical areas if the delay is 0.1 second or less.

The gripping action of the center-tie release mechanism will be adequate. The additional testing planned during Phase III is needed to provide adequate assurance that all elements will perform adequately as a system, after exposure to launch vibration and thermal-vacuum environments.

9.1.4 MTA-3 CONFIGURATION

MTA-3 mechanisms must be identical to MTA-4 for deployment test and check-out purposes.

For MTA-3 to provide the maximum usefulness during Phase III, the panel assembly model should be maintained in "mechanism configuration" identical to MTA-4 design. The MTA-3 drawings should be updated to reflect the mechanism changes involving form, fit, or function made in Phases II and III, and the MTA-3 hardware should be upgraded to the MTA-4 mechanism configuration. The hardware change work should be done on a minimum-cost noninterference schedule basis.

9.1.5 ANALYSIS

The analytical program has satisfied objectives by confirming the design of the primary elements of the array and its test equipment or by identifying the appropriate modifications to the design and/or operational procedures.

Test investigations performed in Phase II have confirmed the load paths, the static and dynamic analyses, and member sizes to a subpanel level. Careful evaluation of component and deployment tests has proved the basic validity of the deployment analysis and the ground test approach.

The first analysis of the primary structure and mechanical elements established the first iteration of the load paths and member sizes. The second analysis revealed a need for some modifications to the design of the primary structure and mechanisms. The most significant of these changes were as follows:

- 1) Center-spar cap gages were increased;
- 2) Tiedown mechanism was redesigned to provide more efficient load paths and reduce some internal loads due to tiedown;
- 3) Release mechanism was modified to get a much higher degree of reliability.

Serious problems still remain to be resolved in Phase III to complete the LASA technology readiness goal. The complex problem of confirming the dynamic response of a stowed solar array is further complicated by the physical limitation of the vacuum chamber to conduct large-sized vibration tests and by the economic need for extensive use of dynamic simulators in lieu of beryllium panel assemblies.

Also, the prediction of the dynamic behavior of the array during a space deployment is influenced by numerous factors of very small magnitude requiring sophisticated techniques for analysis.

Array design details should be carefully evaluated analytically, in light of present knowledge, to determine if minor design changes could significantly reduce the analytical problems or test requirements.

9.1.6 NEW TECHNOLOGY DISCLOSURES

Phase II resulted in important new technology disclosures.

Five new technology disclosures have been transmitted to the NASA new technology officer at JPL. They are:

- 1) Beryllium hot-forming lubricant---The use of boron nitride as a lubricant when forming beryllium sheet into channels;
- 2) Detecting subsurface defects in beryllium by eddy current---A method for using the eddy current technique to discover flaws that have been undetectable in the past;
- 3) Disassembly of structure bonded with AF-126 adhesive---A technique for disassembling bonded beryllium joints without damage to the parts.

- 4) Solid State Switch---A switch with no moving contacts.
- 5) Solid State Potentiometer---To vary potential and resistance uniformly and without noise or wear.

9.2 RECOMMENDATIONS

Phase II results clearly indicate that further optimization of the array design, fabrication, and test and analysis plans is possible. The following specific recommendations will result in weight, cost, and technology improvements.

9.2.1 DESIGN OF SOLAR ARRAY

Minor changes will further optimize the Large Area Solar Array design.

During the development of the LASA design, materials and fabrication processes have been improved. There are also changes in design that have been identified that could improve the array performance but require development and evaluation.

Three changes that show promise of materially affecting the cost of the solar array are (1) substitution of extruded beryllium sections for the bonded beryllium sections, (2) substitution of improved adhesives for the AF-126 adhesive, and (3) revision to substrate design.

It is recommended that an evaluation of these three design changes be made to further optimize the Large Area Solar Array design.

A brief description of each design change recommended for evaluation follows.

- 1) Extruded Beryllium Sections---The method of assembling beryllium sections from hot-formed channels and cap strips bonded under pressure and heat would be replaced with beryllium extrusions. This would save the fabrication time required to hot-creep form and edge-trim the channels, chem-mill, inspect and assemble the tubular sections.
- 2) Substitution of Improved Adhesives---During mission analysis of the Large Area Solar Array, temperature extremes were defined that require the structural adhesive to maintain its strength at 167°F. The present adhesive, AF-126, meets this requirement. Use of this adhesive requires curing at 225°F and a 17-psi pressure. Since the selection of AF-126 adhesive, several new adhesives have been developed and a reevaluation of the new adhesives could simplify fabrication procedures, reducing cost.
- 3) Revision to Substrate Design---The design of LASA substrate was completed in Phase I. With the additional experience gained in fabricating the substrate for MTA-1, it is believed that a revised design would reduce the cost and weight of fabrication. Substrate spacing and tape width would be evaluated.

In addition, approximately 50 pounds of array weight can be eliminated by further design optimization. Changes recommended for potential weight savings are as follows:

- 1) Shear web gage reduction;
- 2) Revision of titanium fittings to delete weld lands, increase heat treatment, and add lightening holes;
- 3) Reduction of gage of shear teeth and compression pads;
- 4) Removal of excess material from outboard quadrants;
- 5) Gage reduction of main-hinge assembly;
- 6) Combination of the auxiliary subpanel hinges and damper bracket and hinge gage reduction;
- 7) Replacement of the two round Monel deployment cables with one flat steel cable outboard of Buttock Line 100.

9.2.2 FABRICATION AND ASSEMBLY RECOMMENDATIONS FOR PHASE III

High-cost manufacturing operations have been identified and tentative cost reduction methods for these operations established.

- 1) Those items of beryllium sheet material that do not require hot-forming should be procured to net width and prepared for bonding as stock material. Flat stock to be formed should be procured to forming blank width, and all operations required prior to bonding should be completed on the stock material. Both flat and formed parts should be cut to length from stock material under controlled clean conditions, so that no subsequent cleaning is required before bonding.
- 2) New tooling for future production should provide for multiple-channel forming on each forming cycle.
- 3) Intercostal stick assemblies should be cut to length from stock-length bonded assemblies. The width of the intercostal caps should be reduced and their gage correspondingly increased to reduce danger of breakage in handling.
- 4) EB welding capability should be evaluated further to determine the advisability of minor design changes in the titanium fittings to reduce weight and improve machinability.
- 5) Substrate assembly tooling should be improved by additional tape locators.
- 6) The frame bonding operation should be simplified and tooling revised to accomplish gusset and clip bonds in a single-cure cycle applying bonding pressure with removable overhead structure. This will permit more uniform heating during the frame cure cycle.

- 7) Assembly of solar-cell submodules with infrared soldering should be reconsidered as a potential cost saving for future assemblies.

9.2.3 TEST RECOMMENDATIONS

Because dynamic analyses are the nucleus to a successful Phase II program, it is recommended that the first task in Phase III should be the reassessment of our analytical plan and goals and the development of a thoroughly integrated test plan to support the analyses.

9.2.3.1 Vibration Testing

An additional vacuum vibration test of MTA-4A may be required to determine damping characteristics.

MTA-1 vibration test results show that air damping contributes about 30% to the total structural damping of an individual solar panel. The tests also show that a dynamic magnification factor Q of a single solar panel, for first mode bending in vacuum, is between 14.5 and 21. Excluding the damping contribution of the various fittings in the panel assembly launch configuration, an estimate of $Q = 16$ to 30, for first mode bending, is possible at this time. Therefore, the shear-teeth and boost tiedown system may have to provide additional significant damping if the design Q of 15 for the stowed array is to be achieved. A necessary part of the Phase III test program is the determination of stowed panel assembly damping, so that the possible need for auxiliary damping can be evaluated.

The vibration test described in the Phase III proposal is a test of the stowed panel assembly to the levels specified in GMP-50505-FNC-B. The specification did not relate vibration environments to atmospheric pressure, but the levels specified are assumed to be the maximum that occur during the initial boost phase at 1 atmosphere. However, examination of the specified environmental levels suggests that these levels also may be produced by the Centaur upper stage. In this case, air damping would not be effective, and all damping must be derived within the array structure.

To confirm the array design under vibration at altitude, study of an additional vacuum chamber mode shape and damping test is recommended early in Phase III. MTA-4A with the vibration simulator installed would be mounted with the three dummy-panel assemblies on a vibration fixture. The test would be conducted in the 39-foot vacuum chamber. Small electrodynamic shakers would drive the stowed "array." Two setups would be required to drive the MTA-4A both in-plane and out-of-plane. The test would consist of a low-level modal survey and damping determination, and would not be carried out at the 1.5-g levels specified in GMP-50505-FNC-B.

9.2.3.2 Deployment Testing

Phase III deployment tests should compare deployment performance against established criteria and simple dynamic response evaluations to confirm the deployment analysis.

Analysis of the MTA-3 deployment tests has shown that some minor changes are required in the mechanisms, the design of the deployment fixture or equipment, and the deployment test (or operational) procedures. These changes are necessary to achieve a successful performance of MTA-4A tests in Phase III and to ensure the capability of the array to deploy in space. (Refer to Section 5 and Test Report D2-113350-3 for details.)

It is desirable to provide general data acceptance test criteria giving limiting values on closing velocities, cable tension, spring rates, and damper characteristics for test evaluation. It is recommended that the following changes be made in Phase III to generate these criteria:

- 1) Reevaluate the data-measuring devices and procedures to improve the accuracy of data and provide some additional data items;
- 2) Determine the magnitude of frictional moments in the array for each of the five specific deployment sequences and their sources;
- 3) Provide a deployment test procedure that separately identifies the track and GSE effects.

Detailed discussion of the design procedural changes is found in Section 5 and the Test Report.

It is concluded that the deployment tests confirmed that the basic design of the ground deployment equipment and the deployment procedures are sound and that, if the above recommendations are followed, successful performance of deployment tests in Phase III can be accomplished.

9.3 PHASE III PROGRAM EVALUATION SUMMARY

Phases I and II developed the technology to a high-confidence level in designing, fabricating, and testing single LASA-type solar panels for specific mission application. Technology is ready today to build 20-watt-per-pound solar panels using simple panel packaging for four panel arrays of power levels up to 4 kw. The Phase III objective is clearly one to verify the technology for high-power, multiple-panel arrays requiring complex packaging and deployment techniques. Technology verification falls into seven primary areas of concern:

- 1) Thermal design confirmation;
- 2) Mode shape and resonant frequency confirmation;
- 3) Damping measurements;
- 4) Deployment characteristics confirmation;

- 5) High-level vibration design verification;
- 6) Acoustic design verification;
- 7) Boost tiedown system release characteristics evaluations.

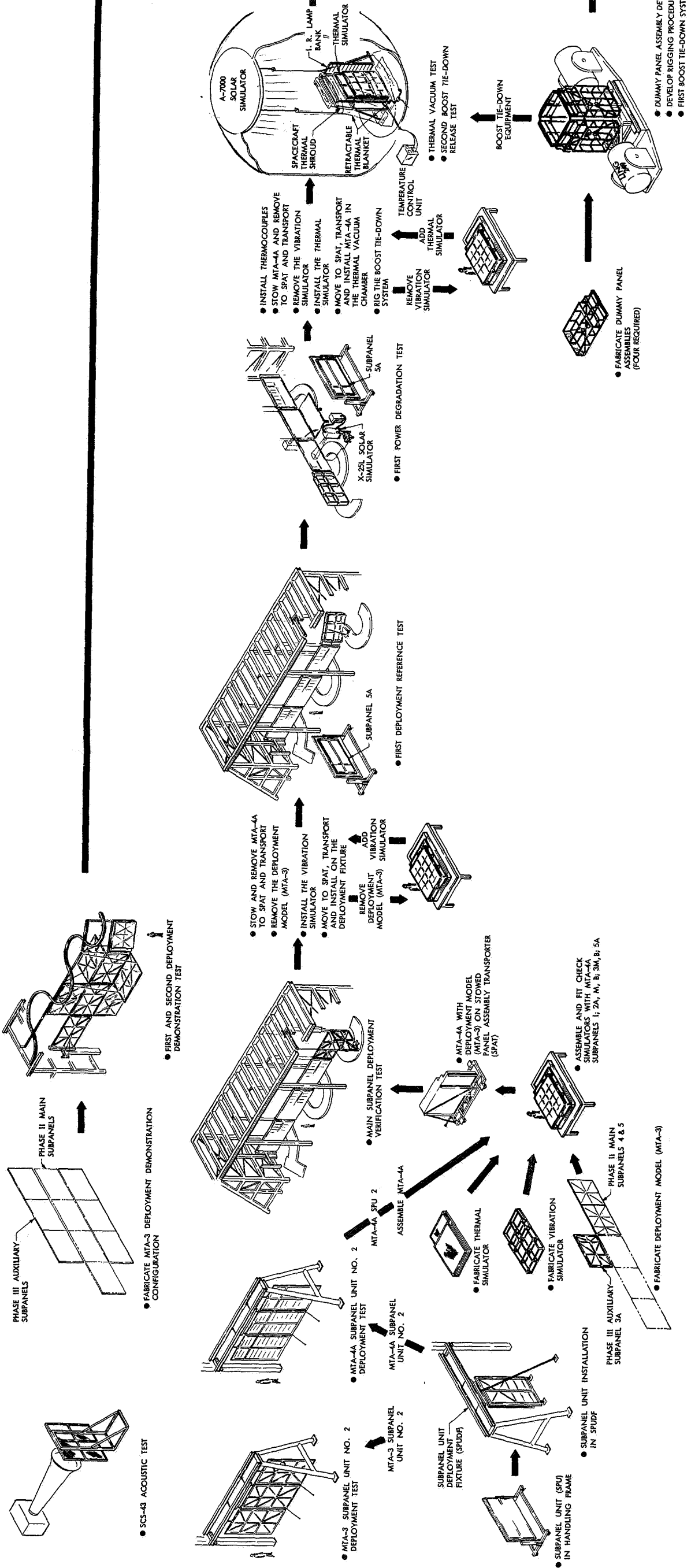
The plan for accomplishing the Phase III LASA Program is contained in Boeing Proposal, D2-114179. The essential elements of the test program are depicted by the test-flow illustration, Figure 9-1.

An evaluation of the present technology status and risks involved in carrying out the Phase III program is summarized below, with respect to the areas of concern listed above.

- 1) Low-Risk Analysis Area---The thermal design confirmation and mode shapes and resonant frequency confirmation fall into this category. The analysis techniques are quite well-understood and proved, and the consequence of a miscalculation could be readily corrected by minor hardware changes.
- 2) Low-Risk Design Areas---The acoustic design verification and the boost tiedown release characteristics areas are both relatively low-risk "design" areas that can be proved only by test. The consequences of failing a test for acoustic design would be severe, but adequate margin probably exists in the LASA design based on the previous lower level acoustic test results.

If the boost tiedown and release system fails tests, a number of alternate designs are feasible and could be developed to replace the present mechanism system with a probable attendant weight increase.

- 3) Medium-Risk Areas---Deployment characteristics confirmation is more an analysis verification problem than a hardware problem, since the magnitude of ground test forces overshadows that actually expected in space deployment. The exactness of the analysis verification can be lost in the "noise" of the test equipment and instrumentation. Alternately, the array can be shown to successfully deploy in space by analysis, and test results will probably confirm the analysis for a few representative deployments where initial conditions of latching velocity and cable tension are established. If the analysis and ground tests become totally incompatible, simple hardware changes can be incorporated in the MTA-4 mechanisms to overcome foreseeable deployment problems. Therefore, deployment is not seen as a critical technology problem, but could become a detail design problem.
- 4) High-Risk Areas---The high-risk areas are measurement of the array damping characteristics and testing the stowed panel assembly to the specification environments. The proposed Phase III test program uses vibration simulators both within the MTA-4 panel assembly "stack" and to substitute for the missing 3-panel assemblies. The techniques for designing these simulators, performing meaningful tests, and forming conclusions based on the test analysis are highly complex.



2

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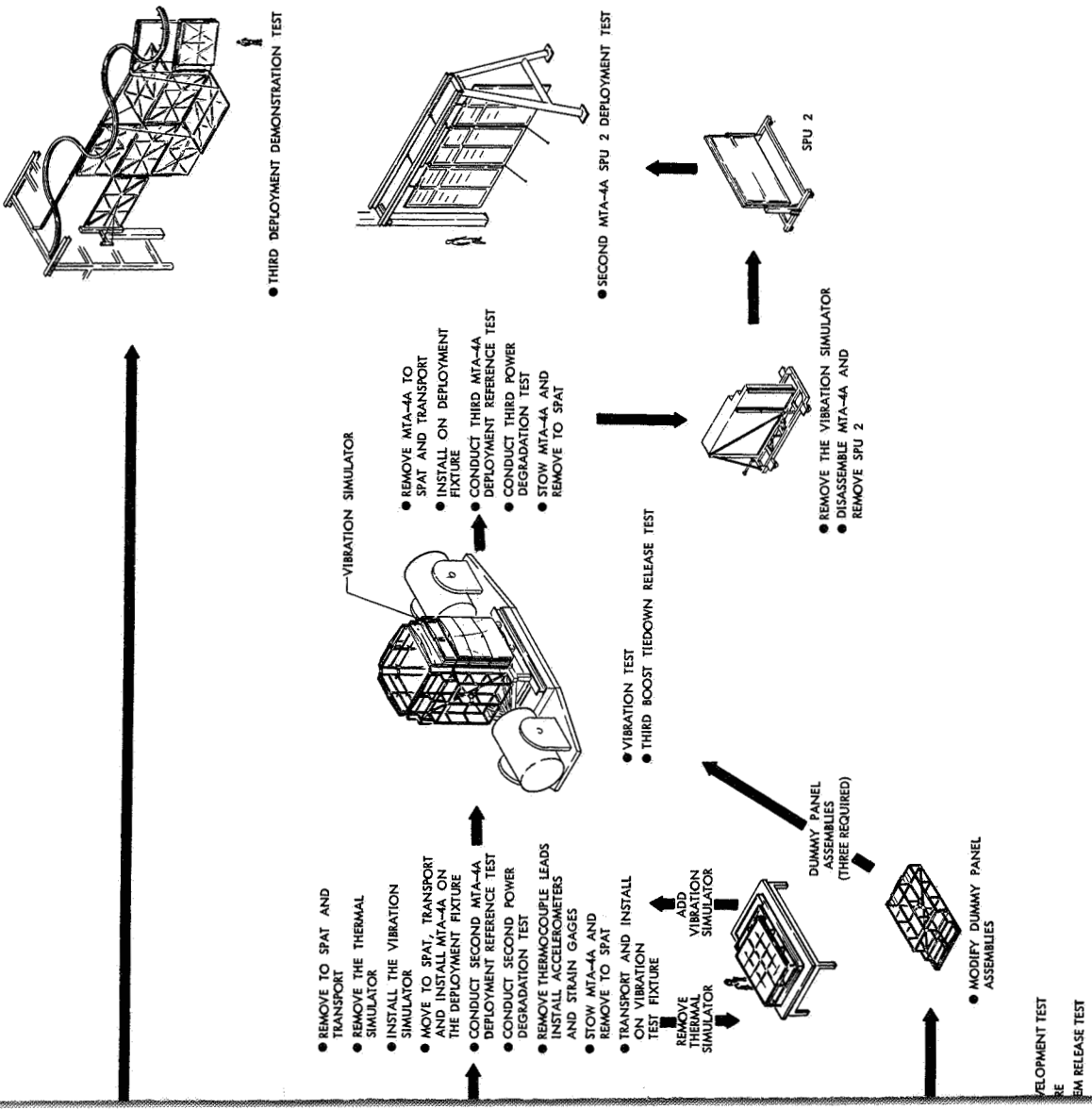


Figure 9-1: LARGE AREA SOLAR ARRAY SUMMARY TEST FLOW
PHASE III ALTERNATE PROPOSAL

The risks of not adequately determining Q for the array or duplicating the MTA-4 mass, stiffness, damping, and mechanical interface characteristics within the simulators are quite high. In view of the high cost of a Phase III development program to provide sufficient representative beryllium structure for vibration test purposes, the risk of not obtaining the required results calls for careful reevaluation of the hardware and vibration test program at the start of Phase III.

10.0 BIBLIOGRAPHY AND DEFINITIONS

Individuals or organizations desiring documented information referenced should contact Jet Propulsion Laboratory, 4800 Oak Grove Drive, Pasadena, California, 91103.

10.1 MAJOR PROGRAM DOCUMENTS

<u>Document Number</u>	<u>Title</u>
D2-113349-1	Large Area Solar Array Design Program Plan
D2-113349-2	Large Area Solar Array Design Program Plan, Phase II
D2-113350-1	Large Area Solar Array Design General Test Plan
D2-113350-2	Large Area Solar Array Design Test Procedures
D2-113350-3	Large Area Solar Array Design Test Reports
D2-113351-1	Large Area Solar Array Design Detailed Quality Assurance Plan
D2-113353-1	Large Area Solar Array Design Manufacturing Plan and Specifications
D2-113354-1	Large Area Solar Array Design Process Document
D2-113355-1	Large Area Solar Array Design First Quarterly Report
D2-113355-2	Large Area Solar Array Design Second Quarterly Report
D2-113355-3	Large Area Solar Array Design Third Quarterly Report
D2-113355-4	Large Area Solar Array Design Final Report---Phase I
D2-113355-5	Quarterly Report (Phase II), Large Area Solar Array
D2-113355-6	Large Area Solar Array Third Quarterly Report (Phase II)

10.2 ADDITIONAL PROGRAM DOCUMENTS

<u>Document Number</u>	<u>Title</u>
D2-113352-1	Large Area Solar Array Design GSE Requirements (Volume I, General Requirements)
D2-113352-2	Large Area Solar Array Design GSE Requirements (Volume II, Detail Requirements)
D2-113352-3	Large Area Solar Array Design GSE Requirements (Volume III, Master Equipment List)
D2-113352-4	Large Area Solar Array Design GSE Requirements (Volume IV, Phase I GSE Preliminary Design)
D2-113352-5	Large Area Solar Array Design---Preliminary Design of Phase III Ground Support Equipment
D2-113445-1	Large Area Solar Array Structural Design Criteria
D2-113565-1	The Influence of Strain Rate and Temperature Upon the Fracture Toughness of AMS 7902 Beryllium
D2-114070-1	Effects of Random Cell Failures on a Large Area Solar Array
D2-114080-1	Reliability Prediction Calculations on a Large Area Solar Array
D2-114081-1	Failure Mode and Effect Analysis, Large Area Solar Array
D2-114286-1	Failure Mode and Evaluation Report, MTA-1

10.3 MISCELLANEOUS REFERENCES

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2. Walter C. Hurty, "Dynamic Analysis of Structural Systems Using Component Modes," *AIAA Journal*, Vol. 3, No. 4, April 1965.
3. M. J. Turner, H. C. Martin, and R. C. Weikel, *Further Developments to the Direct Stiffness Method of Structural Analysis*, Paper presented to AGARD in Paris, France, 1962.
4. L. B. Kiersky, *COSMOS, A Computer Program for Structural Analysis*, Boeing Document D2-4513.
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11. *Fabrication Feasibility Study of a 20-Watt-Per-Pound Solar Cell Array*, Boeing Documents D2-23942-1 through -5.
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10.4 SYMBOLS, TERMS, AND DEFINITIONS

LASA	Large Area Solar Array
GSE	Ground Support Equipment
SPAT	Stowed Panel Assembly Transporter
PADM	Panel Assembly Deployment Model
SCS	Small Component Test Article---Structures
SCC	Small Component Test Article---Cells
SCE	Small Component Test Article---Electrical
SCM	Small Component Test Article---Mechanisms
MTA	Major Test Article
Electric Power	Spacecraft prime electrical power is provided by one or more two-wire ungrounded systems. Power is supplied by solar cells and is available as unregulated direct current at specified voltages.
Load, Electrical	Any component or subsystem that uses electric power and is normally operated as a unit is considered to be a load.
Voltage Range	This expresses the upper and lower voltage limits that will permit normal operation of load equipment in a mission environment.
Limit Loads	Limit loads represent the maximum loads the structure is expected to experience under specified conditions of operation, use, or test. All loads and load factors used in this document are limit loads unless otherwise noted.
Limit Design Loads	A limit design load is a limit load multiplied by the appropriate hazard factor. For the test loads specified herein, the hazard factor is unity. The yield design load is identical to the limit design load.
Ultimate Design Load	The ultimate design load is the limit design load multiplied by 1.25.
Allowable Load	The allowable load for a structural element is that load at which failure is imminent for the yield or ultimate condition, as appropriate.
Bus Voltage	The bus voltage is measured at the main electrical bus terminals. The measured value will vary with the load but will always be within the specific voltage range.
Transients, Electrical	An electrical transient is the changing condition of a characteristic that goes beyond the steady-state limits and returns to the steady-state limits within 10 milliseconds.

Electrical Load Profile	A chart or graph showing power demands (watts) at the main electrical bus as a function of time.
Source (Generation) Impedance	The source impedance is a variable that depends on solar intensity, temperature of the solar cells, and the amount of electrical load.
Reverse Current	A condition where current flow reverses direction due to a higher generated voltage from the electrical load. Reverse current is defined as having a duration of 10 or more milliseconds and is, therefore, not considered a transient occurrence.
Inrush Current	Current that is greater than normal operating current that may be caused by various conditions, but is always associated with switching on an electrical load.
Fault Current	The increment of current that appears at the generator terminals and is caused by a physical short circuit of the power leads.
Ripple	Ripple is the a.c. variation of voltage about a fixed d.c. voltage during steady-state d.c. electrical system operation.
Solar Array Configuration	Four solar panels---a complete assembly for one vehicle. The largest element of the solar array
Solar Panel (Assembly)	that attaches to the spacecraft.
Subpanel	The largest element of a solar panel.
Subpanel Unit	A main subpanel and two auxiliary subpanels.
Solar-Cell Module	A module is the minimum group of solar cells that will independently produce the electrical subsystem voltage at the output terminals of the cell group. Modules will assume various physical outlines as required by the configuration of the structural support. Electrical power output will vary with the number of solar cells used in the module.
Solar Cell	The smallest electrical producing element of the module. Single crystal silicon elements are utilized.
Substrate	Fiberglass tape matrix that supports the solar cells.
Electrical Bus	Metallic conductors that provide electrical continuity between solar-cell modules and between each solar panel assembly and the spacecraft.
Bus Crossovers	Flexible or hinged buses that provide electrical continuity across solar panel, subpanel, or solar array hinge points.