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DEPLOYABLE LARGE AREA

SOLAR CELL ARRAY TESTING

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RYAN REPORT NO. 20869-4

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JPL CONTRACT 951107

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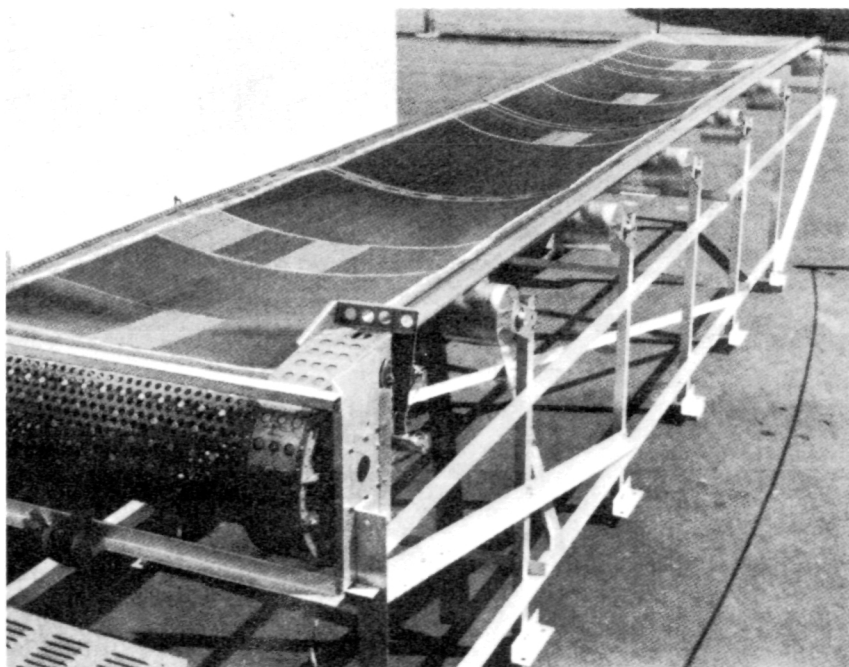
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ABSTRACT

Results of testing and evaluation of a prototype large area rollup solar array structure are presented. The 50-square foot solar panel was designed as one element of a 200-square foot four-panel array of silicon solar cells suitable for a hypothetical Mars or Venus mission using chemical systems and an Atlas/Centaur launch vehicle.

Environmental tests conducted include deployment cycling in a 1.0g field, launch vibration, deployment in vacuum in earth shadow, deployment and retraction sun-oriented near Earth and near Venus, cruise maneuver acceleration loads, and deployed dynamics characteristics.

Performance of the array in each environment is evaluated and conclusions with respect to detail design features are presented.



50-Square Foot Rollup Solar Array on Deployment Support Fixture

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IV.

LIST OF ILLUSTRATIONS

<u>FIGURE</u>	<u>TITLE</u>	<u>PAGE</u>
1	Periods of Electrical Power	15
2	Solar Irradiance Versus Minimum Solar Cell Temperature .	15
3	Solar Array Plan Schematic for the Hypothetical Spacecraft	19
4	Deployable Solar Panel Assembly	21
5	Solar Array Installation	23
6	Solar Array	26
7	Section Through Deployable Beam	27
8	Solar Cell Array Structure	28
9	Solar Cell Panel - Deployed	28
10	Storage Drum	31
11	Electrical Harness-Deployed and Retracted	31
12	Support Structure End Assembly	33
13	Guide Sleeve-Beam Support	33
14	Solar Cell and Wiring Layout for Test of Deployable Solar Panel	35
15	Deployment Support Fixture	46
16	Comparison of Ryan and Spectrolab Test Data On Solar Cell Matrices	48
17	Solar Panel Fully Deployed	50
18	Substrate Deflection Showing Tape Added to Simulate Mass	53
19	Beam Pull-Away at Guide Sleeve When Loaded (Cell Side Down)	56
20	Beam Cap Doupler Added at Guide for Loading (Cell Side Down)	57

LIST OF ILLUSTRATIONS

<u>FIGURE</u>	<u>TITLE</u>	<u>PAGE</u>
21	Deflected Panel Under 0.2g Loading (Cell Side Down) . . .	58
22	Solar Panel Partially Deployed	61
23	Deployment Motor Current Using Original Support Fixture .	62
24	Reworked Roller Support Fixture	64
25	Deployment Motor Current Using Reworked Support Fixtures	65
26	Location of Active Solar Cell Modules	69
27	Solar Cell Electrical Test Arrangement	73
28	Solar Cell Voltage Vs. Test Cycle	76
29	Solar Cell Power Vs. Test Cycle	77
30	Solar Cell Current Vs. Test Cycle	78
31	Deployed Dynamic Test Setup	80
32	Panel Setup in High Bay Building for Deployed Dynamic Test	81
33	Looseness of Guide Fit to Deployment Beam	83
34	Solar Cell Module Layout	86
35	Vibration Test Setup, Y and Z Axis	87
36	Vibration Test Setup, X Axis	88
37	Accelerometer Access Hole in Stowed Panel Wraps	90
38	Test Setup on Slip Table for Spacecraft Axis Vibration	90
39	Test Setup on Slip Table for Drum Axis Vibration	91
40	Test Setup on Shaker for "X" Axis Vibration	91
41	Fatigue Cracks in Beams at Attachment to Tip Intercostal	95

LIST OF ILLUSTRATIONS

<u>FIGURE</u>	<u>TITLE</u>	<u>PAGE</u>
42	Damage in Outboard Substrate Attachment Areas	95
43	Fatigue Cracks in Spacecraft Mount Attach Angles	96
44	Fatigue Cracks in Spacecraft Mount Side Plant Bend Radii	96
45	Fatigue Cracks in Substrate at Panel Edge	97
46	Substrate Module Hinge Pins Displaced Laterally	97
47	Fatigue Crack in Spacecraft Mount Internal Shear Web	98
48	Block Diagram of Random Vibration Control and Monitoring Equipment	101
49	Test Setup on Slip Table for Spacecraft Axis Vibration Test	102
50	Input Excitation, $0.1 \text{ g}^2/\text{cps}$	104
51	Response at Panel Outer Wrap, $0.1 \text{ g}^2/\text{cps}$	105
52	Input Excitation, $0.2 \text{ g}^2/\text{cps}$	106
53	Response at Panel Outer Wrap, $0.2 \text{ g}^2/\text{cps}$	107
54	Proposed Test Cycle	109
55	Deployed Panel in Thermal Vacuum Chamber	110
56	Relationship of Panel to Heat Source in Vacuum Chamber	112
57	Cold Wall of Thermal Vacuum Chamber	113
58	Lamp Installation	115
59	Thermocouple Locations	116
60	Damage to Aluminum Clip Attachment at Left Deployment Beam	118
61	Fiberglass Attachment on Left Deployment Beam	118
62	Reverse Loop Buckle in Deployable Beams	121

LIST OF ILLUSTRATIONS

<u>FIGURE</u>	<u>TITLE</u>	<u>PAGE</u>
63	Thermal Vacuum Cycle, First Test	124
64	Thermal Vacuum Cycle Second Test	125
65	Phase 1 Time-Temp. History of Solar Cells at Panel Centerline	126
66	Phase 1 Time-Temp. History of Solar Cells at Panel Edge	127
67	Phase 1 Time-Temp. History of Deployment Beam	128
68	Phase 2 Time-Temp. History of Solar Cells at Panel Centerline	129
69	Phase 2 Time-Temp. History of Solar Cells at Panel Edge	130
70	Phase 2 Time-Temp. History of Deployment Beam	131
71	Phase 3 Time-Temp. History of Solar Cells at Panel Centerline	132
72	Phase 3 Time-Temp. History of Solar Cells at Panel Edge	133
73	Phase 3 Time-Temp. History of Deployment Beam	134
74	Phase 4 Time-Temp. History of Solar Cells at Panel Centerline	135
75	Phase 4 Time-Temp. History of Solar Cells at Panel Edge	136
76	Phase 4 Temperature Locations-Deployment Beam	137
77	Phase 5 Time-Temp. History of Solar Cells at Panel Centerline	138
78	Phase 5 Time-Temp. History of Solar Cells at Panel Edge	139
79	Phase 5 Time-Temp. History of Deployment Beam	140
80	Phase 6 Time-Temp. History of Solar Cells at Panel Centerline	141

LIST OF ILLUSTRATIONS

<u>FIGURE</u>	<u>TITLE</u>	<u>PAGE</u>
81	Phase 6 Time-Temp. History of Solar Cells at Panel Edge .	142
82	Phase 6 Time-Temp. History of Deployment Beam	143
83	Deployment Motor Current, Ambient Room Temperature Conditions	146
84	Deployment Motor Current, Phase 2 of Test Cycle	147
85	Deployment Motor Current, Phase 4 of Test Cycle (After Repair)	148
86	Deployment Motor Current, Phase 6 of Test Cycle (After Repair)	149
87	Effect of Motor Temperature on Motor Current	154
88	Lateral Load Test Setup	155
89	Method for Simulating Thermal Contraction Induced Loads .	155
90	Deployment Motor Current, Evaluation Test	156
91	Restraint Area Showing Substrate, Deployment Beam, Guide Sleeve, and Crosstie	157
92	Areas Affected in Refinement of System	159
93	Outboard Substrate Attachment	160
94	Deployment Motor Current Summary Plot, Evaluation Test .	162
95	Checking for Load Versus Deflection at Point Contact . .	178
96	Checking for Load Versus Deflection Through a Load Bar .	179
97	Position of Load Designation	181
98	Solar Cell Impact - Test Setup on Vibration Exciter . . .	184
99	Effect of Excitation Acceleration on Response Acceleration at Resonance	185

LIST OF TABLES

<u>TABLE</u>	<u>TITLE</u>	<u>PAGE</u>
1	Summary of Physical Changes to Solar Panel	66
2	Summary of Solar Cells Damage	70
3	Incidence of Cracked Solar Cells	71
4	Summary of Electrical Tests	74
5	Thermocouple Locations	123
6	Sine Vibration Test Equipment	173
7	Random Vibration Test Equipment	174
8	Deployed Dynamics Characteristics Test Equipment	175
9	Thermal Vacuum Test Equipment	176
10	Equipment for Electrical Check of Cells	177
11	Position of Load Deflection	182
12	Summary of Deployment Cycles	187

1.0 INTRODUCTION

This report presents the results of the testing and evaluation of a prototype large area solar array structure. The program was conducted by the Ryan Aeronautical Company for the Jet Propulsion Laboratory under JPL Contract 951107. This report supplements Ryan Report No. 20869-3 "Final Report for Design and Fabrication of a Deployable Large Area Solar Cell Array Supporting Structure", 28 April 1967.

The purpose of the program has been to develop concepts and technology and to design, fabricate, and test a large area deployable solar cell array.

Previously, in the program a four-element 200-square foot rollup solar array structure was designed and a 50-square foot prototype element was fabricated. Concept development and detail design have been described in Ryan Reports 20869-1, "Preliminary Development Report" and 20869-2, "Final Design Report". The fabrication of the prototype structure is described in Report 20869-3, "Final Report."

During the test program, a solar cell installation design was completed and the solar panel was covered with active and simulated solar cells. Environmental tests were conducted to evaluate the conformance of the array structure to specification requirements and analytical predictions. Tests performed included deployment cycling, sinusoidal and random vibration, deployed dynamics, thermal vacuum cycling and 0.2g cruise maneuver.

The report contains a summary of the specification requirements, a design description and test plan summary. Test results are presented in detail along with an evaluation of the array.

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2.0 SUMMARY

The testing and evaluation of the large area solar array structure has been completed in accordance with the requirements of JPL Contract Number 951107. The contract established hypothetical structural and environmental requirements similar to a 1969 Mars launch and encounter using chemical systems and an Atlas/Centaur launch vehicle.

A test program was developed to simulate the critical environments to which the array is exposed before, during and after launch. The tests included the following:

1. Deployment cycling in a 1.0g field to determine effects of structure on solar cells.
2. Launch vibration consisting of sinusoidal and random vibration to measure effects on structure and solar cells.
3. Thermal vacuum test consisting of two deployment and retraction cycles simulating Mars and Venus missions to determine behavior of deployment system and solar cells in a space environment.
4. Deployed dynamics characteristics to determine the beam first bending mode, panel first torsional mode and substrate first mode.
5. Simulation of 0.2g cruise maneuver accelerations to determine effect on structure.

Special test equipment provided included a deployment support fixture, vibration test fixture and electrical test equipment to measure solar cell performance.

A detailed design was completed of an active and simulated solar cell installation which could be used to evaluate the environmental effects on solar cell performance. Approximately ten percent of the panel area was covered with seven groups of 8 mil 2 cm X 2 cm silicon solar cells, with aluminum

dummy chips used on the remainder of the panel to simulate the total mass of solar cells and wiring.

Preliminary tests were performed to determine effects of cold deployment on solar cell adhesion, stress levels on solar cells during deployment, and effect of compactness of stowed panel wraps on resonant vibration characteristics.

The 0.2g cruise maneuver test was performed in a 1.0g field before installation of solar cells on the substrate. Test results indicated that the 0.2g acceleration was met without permanent set or elastic buckling with a tip deflection of 24 inches in one direction and 20 inches in the opposite. The test revealed the need for an increase in beam thickness at the open side of the beam guide to compensate for lack of beam support.

After mass simulation and electrical assembly, cracks were noted in 4.7 percent of the active cells installed on the panel. These defects were caused by handling problems associated with the solar cell installation and assembly of the flexible panels on the deployable beams, indicating the need for improved methods and experience in handling this type of assembly.

There also was evidence of longitudinal crazing near the edges of the fiberglass substrate due to variations between substrate width when deployed and wrapped.

The deployment cycling test was used as the basic reference to measure changes in the solar panel following each environment. During each deployment cycle motor drive current was measured. Following visual inspection, measurements of solar cell voltage, current and power were made to determine solar cell performance.

Test results indicate that solar cell performance was not affected by deployment cycles. Incidence of cracked cells during the first five cycles was 1.8 percent. Motor current measured at 27 volts dc varied from 0.4 amperes for extension to 0.6 amperes for retraction. Deployment currents in excess of 1.0 amperes were required to overcome effects of friction in the solar

support fixture coupled with intermittent binding of the beam guide cam mechanism. It was necessary to change the roller support bearings and disengage the beam guide cam to obtain uniform deployment currents of less than 0.4 amperes.

Launch vibration tests consisted of a sinusoidal vibration sweep at 1.6g rms from 6 to 20 Hz and 4.0g rms from 20 to 200 Hz followed by random vibration at $0.1g^2/Hz$ and $0.2g^2/Hz$ from 200 to 2,000 Hz each for 180 seconds.

A modal scan was conducted at low level (0.5g peak) sinusoidal excitation from 5 to 2,000 Hz to determine dynamic transmissibilities in each of the three spacecraft axes. Results showed the Z axis (spacecraft thrust axis) to be most critical to the solar cells on the outer wrap with resonance of 43 Hz and a transmissibility of 20 to 1. The Y axis (wrap drum axis) was more critical to the drum and end plate structure with a critical resonance at 30 Hz and a transmissibility of 5 to 1.

At design level (5.66g peak) the Z axis outer wrap resonance was at 30 Hz with amplification of 14, and in the y axis the drum end plate resonance was 23 Hz with an amplification of 1.77.

The sinusoidal vibration had little effect on solar cells. There was no change in electrical performance with an incidence of 1.0 percent cracked cells. Some structural damage occurred which indicated need for design adjustments. Principal effects were fatigue cracks in beams near the tip intercostal requiring a revised intercostal shape at the beam attachment, loose edge connectors requiring an adhesive bond, fatigue cracks along substrate edges requiring better control of assembly tolerances and more flexible substrate material, and fatigue cracks in spacecraft mount requiring added thickness in mount angles.

The random vibration test was conducted with the panel excited in the Z axis determined as most critical. There was no significant effect on solar cells or structure as a result of this test. Incidence of cracked solar cells was 0.3 percent.

The deployed dynamics characteristics were determined by suspending the array vertically and exciting the panel at the tip intercostal. The first bending natural frequency was 0.43 Hz. The first torsional natural frequency was 13.0 Hz and the first substrate panel natural frequency was 0.87 Hz. By analysis the first bending frequency was predicted at 0.55 Hz and the first torsional frequency was 10.0 Hz.

The thermal vacuum test was conducted to simulate the deep space thermal environment on panel deployment and retraction operation for a hypothetical mission composed of deployment in earth shadow, sun oriented retraction and deployment near Earth, and sun oriented retraction and deployment near Venus. Temperatures of solar cells rather than magnitude of heat flux was used for test environment control. The temperatures used were -150°F for Earth shadow, 130°F for near Earth and 275°F for near Venus. The planned test cycle consisted of deployment at -150°F, heat to 130°F, retract and deploy, heat to 275°F retract and deploy and return to ambient conditions.

The thermal vacuum cycles had no significant effect on solar cell performance. There was a significant effect on structure caused by deployment difficulties at -150°F. Deployment cycles at 130°F and 275°F were entirely satisfactory.

During the initial deployment at -150°F extension was slower than normal with motor currents in excess of 1.0 amperes. When deployment was complete, it was noted that a shear-tension failure of 90 percent of the substrate edge connectors along the L/H beam had occurred. This failure was later shown to be caused by differential thermal contraction between the substrate and beam guides coupled with longitudinal miscoordination of beam travel.

Following the initial failure, the panel was repaired with a flexible dry fiberglass strip to attach beam to substrate, and the test resumed. Deployment cycles at near earth and near Venus temperatures were conducted in a satisfactory manner with motor currents ranging from 0.5 amperes for extension to 1.0 amperes for retraction.

The earth shadow deployment was then repeated again with difficulty. Motor currents surged to 1.7 and 2.5 amperes on deployment and a reverse loop formed in the stored beam, ending deployment at about 85 percent travel. Solar cell damage in this area was extensive, causing an 8.5 percent power loss.

A complete evaluation of the earth shadow deployment problem was conducted. By analysis and simulated lateral load test, the design improvements needed were; better coordination of beam and substrate travel, a flexible attachment of substrate to beam, thermal control coatings on structure crosstie and tip intercostal to reduce thermal gradients, and better distribution of force by radial restraint springs to the stowed beams.

The array structure was reworked to incorporate these changes except for radial restraint springs. Lateral load tests were performed on the reworked structure which indicated that earth shadow deployment could be successfully accomplished.

3.0 CONTRACT REQUIREMENTS

This section contains the contract requirements and the design criteria applicable to the solar array.

3.1 TEST PROGRAM REQUIREMENTS

The following performance requirements are taken from the contract Statement of Work.

Develop and submit for JPL approval, a test program to environmentally and dynamically test the prototype solar array structure with modular substrate assemblies. The test program shall be developed to simulate the pre-launch launch and post-launch environments and dynamic influences. The test program shall include, but need not necessarily be limited to:

1. Necessary drawings, specifications, and procedures required to accomplish the test program.
2. Deployment test procedures and set up.
3. Complete test program schedule.
4. Necessary test fixtures and facilities required to accomplish this program.

Prepare for the accomplishment of the JPL approved test programs by performing the following:

1. Fabricate necessary test fixtures.
2. Install silicon solar cell matrices provided by JPL, over approximately 10% of the available substrate area, as designated by JPL.
3. Electrically harness the solar cells.

4. Simulate solar cells and electrical connection mass on the substrate area not covered by solar cells.

Perform tests on solar array in accordance with the JPL approved test program and schedules.

JPL will provide the following Government-owned property:

Approximately one-thousand (1000) silicon solar cells
(2 cm X 2 cm) and glass covers.

3.2 DESIGN CRITERIA

The following section presents the design requirements specified by the contract.

3.2.1 Configuration Design Criteria

No provisions shall be required on the array structure to support unrelated spacecraft components such as power regulating zener diodes, cold gas attitude control systems and/or vernier solar pressure vanes.

During the launch phase, the array structure shall remain within the envelope shown on JPL Drawing No. J-4190680 (Sheet 1). This drawing reflects the available packaging regions for a broad range of typical era. The drawing depicts a standard Surveyor class shroud on an Atlas-Centaur vehicle. The spacecraft is arbitrarily defined to be an octagonal frame, fifty-eight inches across on the major diagonal. Primary array structure attachment to the spacecraft may be accomplished along any of the corners or vertical edges of the hypothetical spacecraft frame.

The basic array structure shall be designed to have a minimum number of different components. This requires that the total array structure be composed of not more than four sub-elements or panels.

Total available surface for solar cell mounting shall be between one hundred fifty and four hundred square feet. For initial planning and conceptual study purposes, a target area of two hundred square feet shall be assumed. The geometry of the array structure shall be based upon a rectangular modular solar cell array of 18 inches x 34 inches, having a weight of 0.30 pound per square foot. This weight shall include cells, filters, modular wiring, and secondary cabling.

Any mechanical latch points or devices located on the cell surface shall not shade the solar cell surface when the array structure is oriented $\pm 5^\circ$ from the normal incidence angle of illumination.

3.2.2 Structural Design Criteria

Under the hypothetical environmental conditions set forth in the Environmental Criteria for a useful life of eighteen months:

- a. The array structure shall have the capability of surviving normal ground handling during fabrication, assembly, qualification testing and storage. The array structure shall also have the capability of being repaired when subject to minor damage.
- b. The array structure shall have the capability of surviving all dynamic loads, including transportation, cruiser course correction, and retromaneuver at planetary encounter. It is implicitly assumed that the array structure will be in the undeployed configuration during launch and in the deployed configuration during course correction motions. Depending upon the nature of the array structure (deployed or undeployed), either configuration may or may not be used during the retromaneuver. The retromaneuver thrust shall not be used to initiate or power the retraction, if required, or the array structure. Upon the completion of the retromaneuver, the array structure shall be in the deployed configuration suitable for power production.

- c. The rear surface of the array structure shall be designed to minimize heat radiation traps in order to minimize local front surface hot spots.
- d. All array structure components shall have provisions for pressure equalization between internal elements and the external flight environment.
- e. To preclude real or potential degradation of the solar cells mounted upon the array structure, the curvatures induced in the cells shall be limited as follows:
 - 1. The radius of curvature of the undeployed or stowed array structure shall at no time be less than six inches.
 - 2. Under dynamic conditions, the angular change of the cell substrate per unit length shall be less than 1.0 degree/inch.
- f. To avoid servoelastic coupling of the array structure and hypothetical spacecraft control system, the inertial and response characteristics for the array structure shall be as follows:
 - 1. During powered flight (boost, retro), due to allowable tolerance variations in the fabrication process, the first mass moment of the array structure (undeployed or deployed) shall vary less than 5% as measured about the spacecraft centerline (boost axis).
 - 2. If deployed, the array structure shall further have the following characteristics:
 - a) The undamped first cantilever natural frequency of the array structure shall be between 0.5 and 5.0 cps.
 - b) The ratio of damping to critical damping in the first cantilever mode shall be in the range 0.15 to 0.7.

- c) In the first cantilever mode of the array structure, the ratio of generalized stiffness to generalized mass (k/m) shall vary less than 10% due to all allowable tolerance variations in the fabrication processes.
3. During cruise phase (including course correction maneuver), the requirements shall be as defined in Paragraphs 3.2.2(f)2(a) and 3.2.2(5)2(c).
- g. Structural criteria given below are stated in terms of limit loads (yield design loads). Induced stress levels shall be computed for all loading conditions stated in Paragraphs 3.2.2(g)1 and 3.2.2(g)2. Critical conditions shall be clearly identified and carefully evaluated. Margins of safety on stresses induced by these limit design loads as follows:

$$M.S. = \frac{Y.S.}{L.S.} - 1 > 0$$

$$M.S. = \frac{U.S.}{(1.25)(L.S.)} - 1 > 0$$

where

M.S. = Margin of Safety
 L.S. = Stresses resulting from Limit Loads
 Y.S. = Yield Stress
 U.S. = Ultimate Stress

The yield and ultimate stress values shall be those for the appropriate material as given in the latest editions of MIL-HDBK-5 and MIL-HDBK-17.

1. Thermal Cycling: this cycling represents the effects of spacecraft orbit about a planet as well as spacecraft attitude reorientations associated with mid-course corrections. The design limit thermal loads for this array structure are equivalent to the levels experienced during the following test environment:

- a) Pressure - The maximum pressure shall be 10^{-4} mm Hg.
- b) Free space background - The free space background or heat sink shall be simulated by a blackened wall having a total absorptivity or greater than 0.80 at liquid nitrogen temperatures, as viewed from the array structure.
- c) Heat Cycling - A heat input to the array structure surface of 80 watts per square foot shall be held until temperature stabilize. The electrical power source shall then be turned off for a 1-1/2 hour period. The subsequent step changes in electrical power input from 0 to 80 watts per square foot defines the start of a cycle. Periods of applied electrical power shall be for a 1-1/2 hour duration. Periods of nonapplied electrical power shall be for a 1-1/2 hour duration (see Figure 1). The array structure shall be subjected to 10 periods of applied heater power for a total test time of approximately 40 hours.

2. Limit Structural Design Loads: the following table contains the applicable limit accelerations for use in the determination of the appropriate limit loads. These accelerations define the environment at the array structure-spacecraft interface. The array structure shall be checked for structural adequacy under both static and vibratory criteria. Static and vibratory loads are not to be superimposed for design purposes.

<u>Condition</u>	<u>Static</u>	
	<u>Long</u>	<u>Lat.</u>
(a) Max q and Mach 1	4g	3g
(b) Booster Burnout	12g	2g
(c) Booster Tailoff	2g	0
(d) Cruise Maneuver	0.2g	0.05g
(e) Retro Burner	6g	1g

<u>Condition</u>	<u>Level</u>	<u>Vibratory</u>	
		<u>Range</u> <u>(cps)</u>	<u>Rate</u> <u>Min/Octave</u>
(a) Max q and Mach 1	-	-	-
(b) Booster Burnout	1.6g rms	2 - 20	1 min/oct
	4.0g rms	20 - 200	1 min/oct
	Noise $0.2g^2/cps$	200 - 2000	180 seconds
(c) Booster Tailoff	-	-	-
(d) Cruise Maneuver	0	-	-
(e) Retro Burner	8g rms	2 - 20	0.5 min/oct
	2.0g rms	20 - 200	0.5 min/oct
	Noise $0.2g^2/cps$	200 - 2000	180 seconds

3.2.3 Thermal Design Criteria

All components shall meet the following sterilization requirements:

- a. Withstand exposure to three 36-hour periods of heat at 145°C (295°F) in dry nitrogen (a total of one hundred eight (108) hours).
- b. Withstand exposure to a gas mixture of 12% ethylene oxide, 88% freon gas for ten hours at a relative humidity between 30% and 50%.

The temperature at any point on the solar cell surface, as a function of solar irradiance, shall be less than the values defined in Figure 2. This is a maximum temperature for any position on the front surface of the array structure. These temperatures may be achieved by requiring that exposed surfaces on the rear and edges of the array structure have a total hemispherical emissivity of greater than 0.80 in the temperature range of -30° and 80°C.

The rear surface of the array structure shall be designed to minimize heat radiation traps in order to minimize local front surface "hot spots".

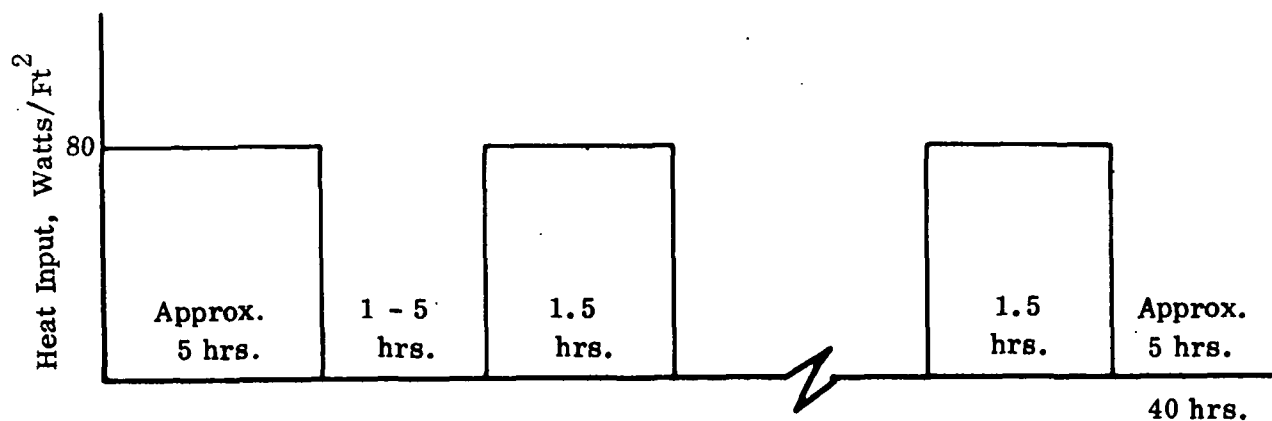


Figure 1 Periods of Electrical Power

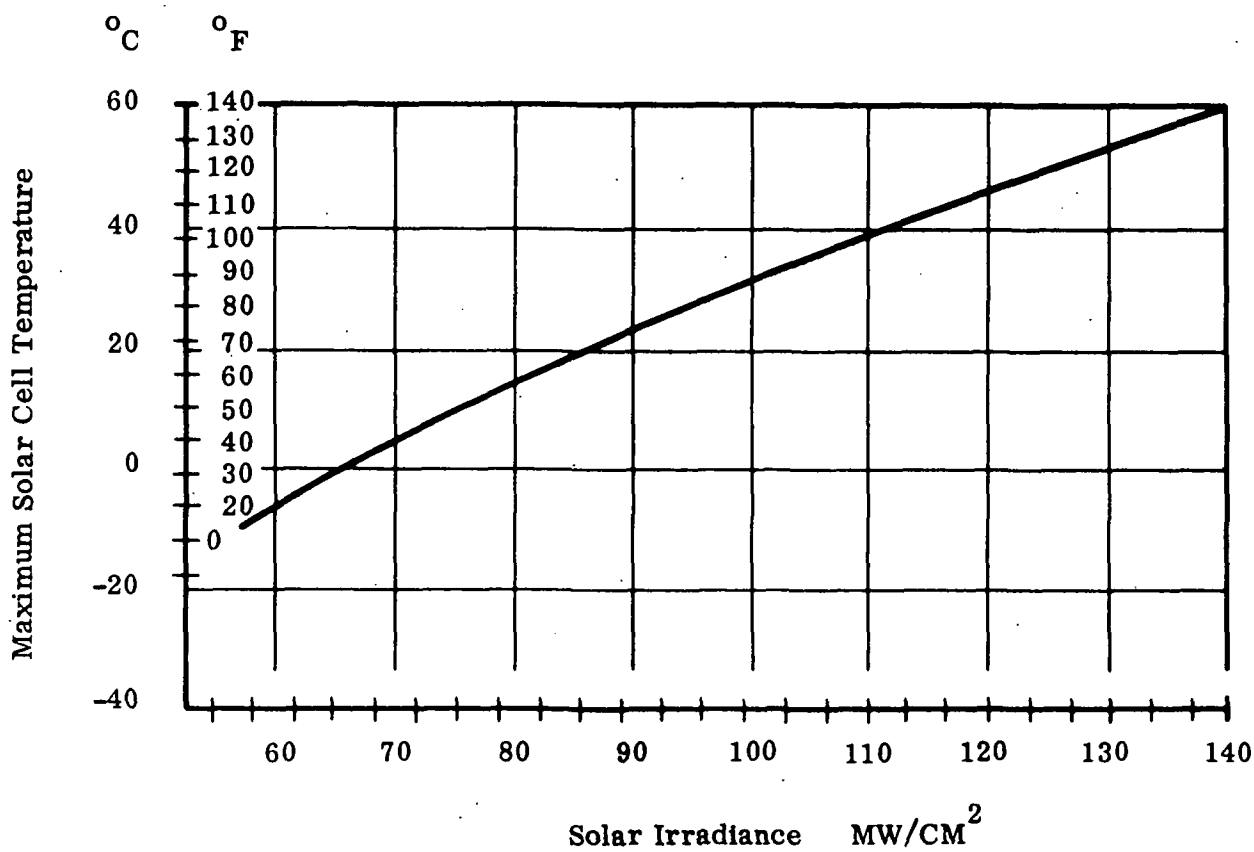


Figure 2 Solar Irradiance Versus Maximum Solar Cell Temperature

Thermal cycling represents the effect of spacecraft orbit about a planet as well as spacecraft attitude reorientations associated with cruise course corrections. The design limit thermal loads for this array structure are equivalent to the levels experienced during the following test environment:

- a. Pressure - The maximum pressure shall be 10^{-4} mm Hg.
- b. Free Space Background - The free space background or heat sink shall be simulated by a blackened wall having a total absorptivity of greater than 0.80 at liquid nitrogen temperatures as viewed from the array structure surface.
- c. Heat Cycling - A heat input to the array structure surface of 80 watts per square foot shall be held until temperatures stabilize. The electrical power source shall then be turned off for a 1-1/2 hour period. The subsequent step changes in watts/square foot define the start of a cycle. Periods of applied electrical power shall be for a 1-1/2 hour duration. Periods of nonapplied electrical power shall be for a 1-1/2 hour duration. The array structure shall be subjected to ten periods of applied heater power for a total test time of approximately forty hours.

3.2.4 Material Design Criteria

The cell mounting surface shall be capable of being cleaned with solvents or mild acid etching techniques prior to cell mounting.

The cell mounting surface shall be fabricated of or coated with a material that is an electrical insulator. This material shall be capable of withstanding the rigors of cell-mounting techniques. This material shall survive and be capable of repair, in the event that a damaged or defective cell must be removed. The insulation resistance shall be greater than 100 megohms, measured at a test potential of 200 VDC between the cell mounting surface and any metallic portion of the substrate.

The use of any material shall be predicated upon the proven ability of the material to withstand the deep space environment for a time in excess of eighteen months.

All components shall meet the following sterilization requirements:

- a. Withstand exposure to three thirty-six hour periods of heat at 145°C (295°F) in dry nitrogen, a total of one hundred eight (108) hours.
- b. Withstand exposure to a gas mixture of 12% ethylene oxide, 88% freon gas for ten hours at a relative humidity between 30% and 50%.

The exposed surfaces on the rear and edges of the array structure shall have a total hemispherical emissivity of greater than 0.80 in the temperature range of -30° to 80°C.

Magnetic materials shall not be used in any of the array structure components, except when array structure reliability is affected by use of such materials.

3.2.5 Weight Design Criteria

A design objective shall be to keep the weight of the array structure and deployment mechanisms below 0.6 pound per square foot, including solar cells, cabling and wiring.

The solar cell array shall have a weight of 0.30 pound per square foot. The weight shall include cells, filters, modular wiring and secondary cabling.

4.0 TECHNICAL DISCUSSION

The solar cell array support structure designed and fabricated in earlier phases of this contract is suitable for a Mars or Venus launch and encounter using chemical systems and an Atlas/Centaur launch vehicle.

The purpose of the test program is to evaluate the structural integrity and electrical characteristics of the solar array assembly when subjected to the most critical launch and environmental conditions anticipated during a hypothetical mission to Mars or Venus.

Previous design and fabrication activity has been limited to the deployable array structure. In order to conduct the test program it has been necessary to complete the design of the solar cell array installation and apply active and simulated solar cells to the panel substrate.

This section presents a description of the solar panel structural design, the solar cell installation, the test plan, test results and evaluation of the solar panel design.

4.1 DESIGN DESCRIPTION

The solar array structure design and fabrication which are summarized in this section are described in detail in Ryan reports 20869-2 and 20869-3. (Ref. 2 and 3) The additional design of the solar cell installation which was completed during the test program is described in detail.

4.1.1 Solar Array Structure and Deployment System

The solar array is designed to deploy a minimum of 200 square feet of solar cell area. This area is provided by four solar panels of approximately 50 square feet each shown schematically in Figure 3 and in detail by Figures 4 and 5. Each panel contains four removable substrate modules.

The array fits well within the envelope of a hypothetical spacecraft. When completely retracted, the rectangular envelope of the structure is 17 inches

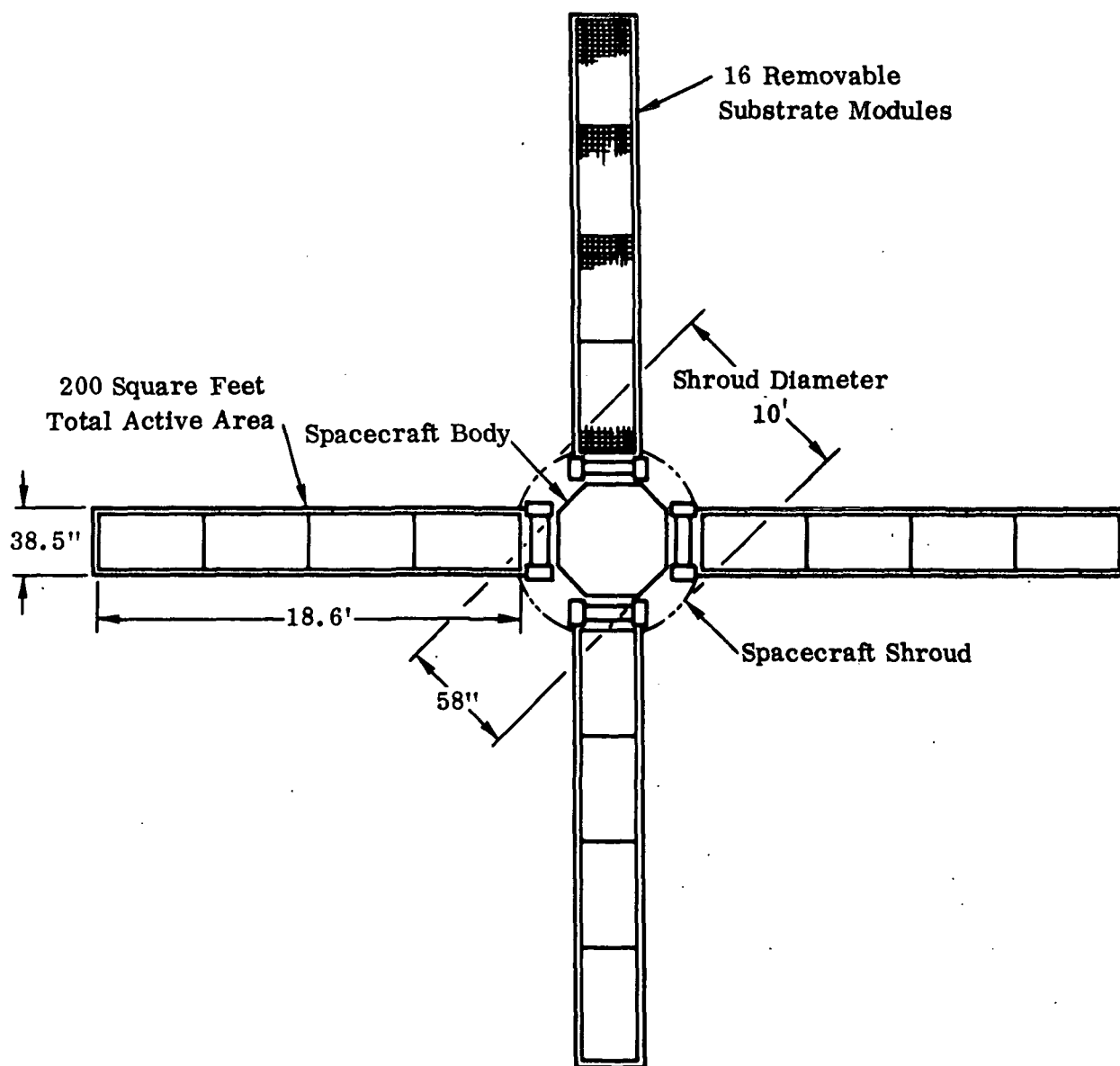
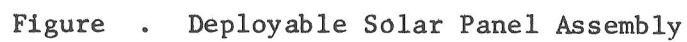


Figure 3 Solar Array Plan Schematic for the Hypothetical Spacecraft

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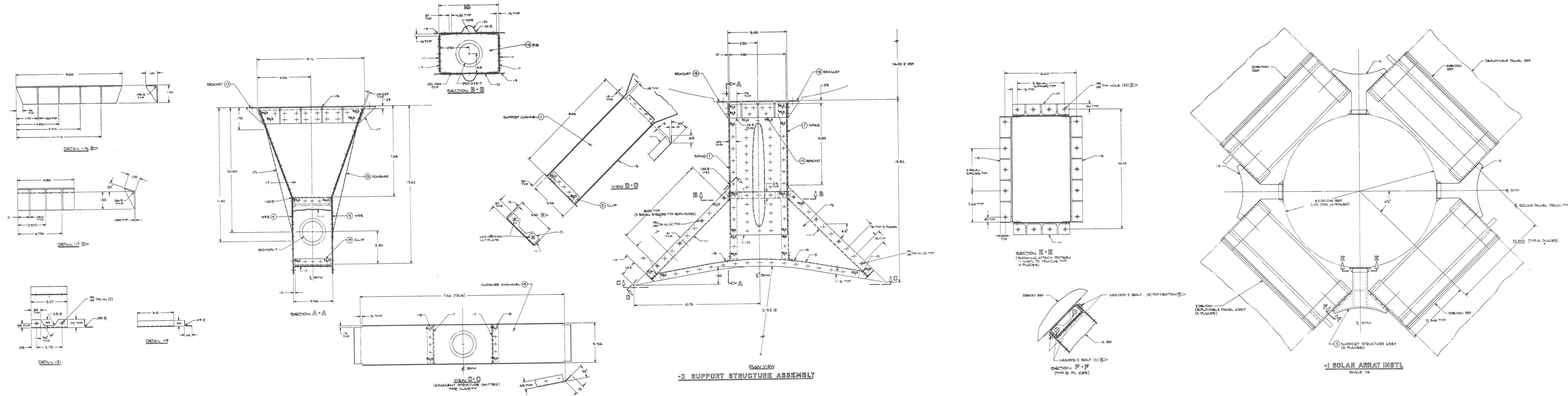


Figure 3. Solar Array Installation

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by 21 inches by 49 inches. The total volume is 10 cubic feet. Actual space availability and interface structure can be altered to provide even greater capability.

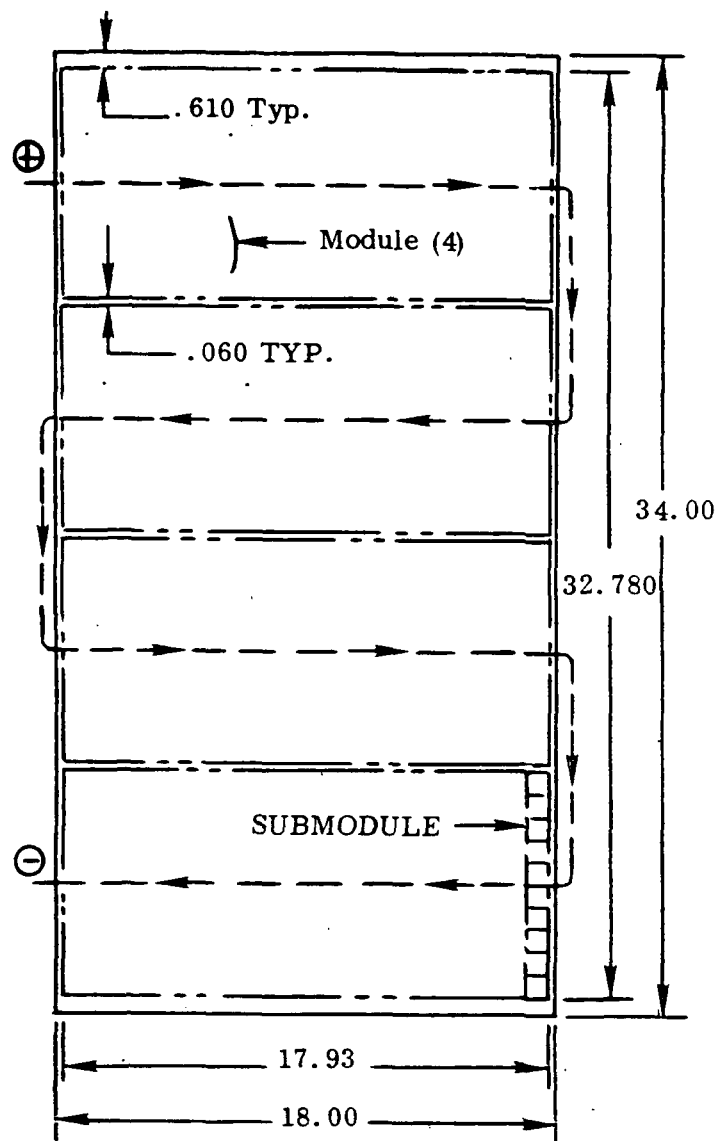
An electrical layout using the available area and geometry of the structure is shown in Figure 6. Each blade contains four 55.8 inch x 36.64 inch modular panels. Each panel carries three 18 inch x 34 inch rectangular solar cell modules. This array is capable of producing 10 watts per square foot or 2,000 watts total power. An area 0.6 inch wide is furnished along the length of the substrate for routing flat electrical leads.

The panels are deployed and supported by extendable beams. The beams and solar cells substrate are stowed on a cylindrical drum which is driven by an electric motor to deploy or retract the solar panels. The support beams are designed to flatten as they approach the drum thus making wrapping possible.

The beams are made of titanium alloy (6Al-4V). The cross-sectional shape of the beam and substrate attachment is shown diagrammatically in Figure 7. The substrate attach flange is seam-welded to the lower portion. Both sections are joined as indicated at their faying surfaces by seamwelds along the entire length. The inboard end of the beam assembly is preflattened to facilitate attachment to the drum. The outboard end of the beam attaches directly to a cross member leading to the beam on the opposite end of the panel assembly. The substrate attach flange contains precisely located slots which are matched to mating slots in the substrate and opposite hand beam. An aluminum substrate attachment clip is inserted into these slots to fasten the beams and substrate together.

Figure 8 is a photograph of the solar cell array structure showing the rear surface of the beams and substrate. The roller drum mounted in bearings attaches to the beam and pulls the beam in for retraction or pushes the beam out for extension.

The beam is stowed by rotating the storage drum which reels in the beam, through a guide sleeve causing the beam to flatten against the drum. Twenty-



(18 X 34) ELECTRICAL STRING

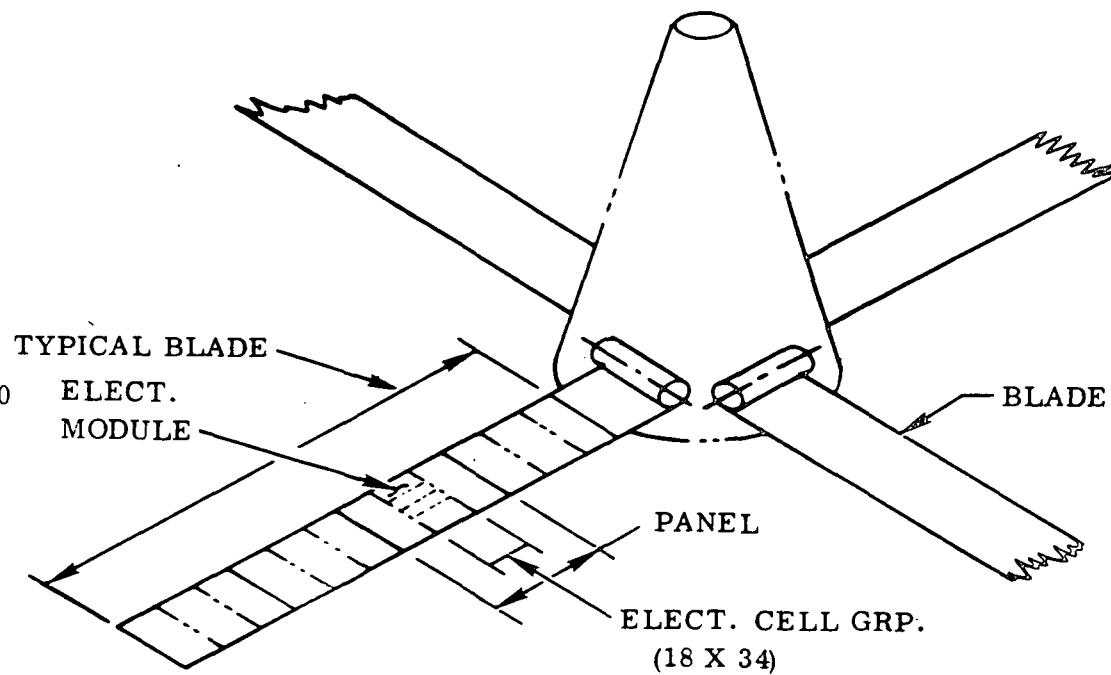


Figure 6 Solar Array

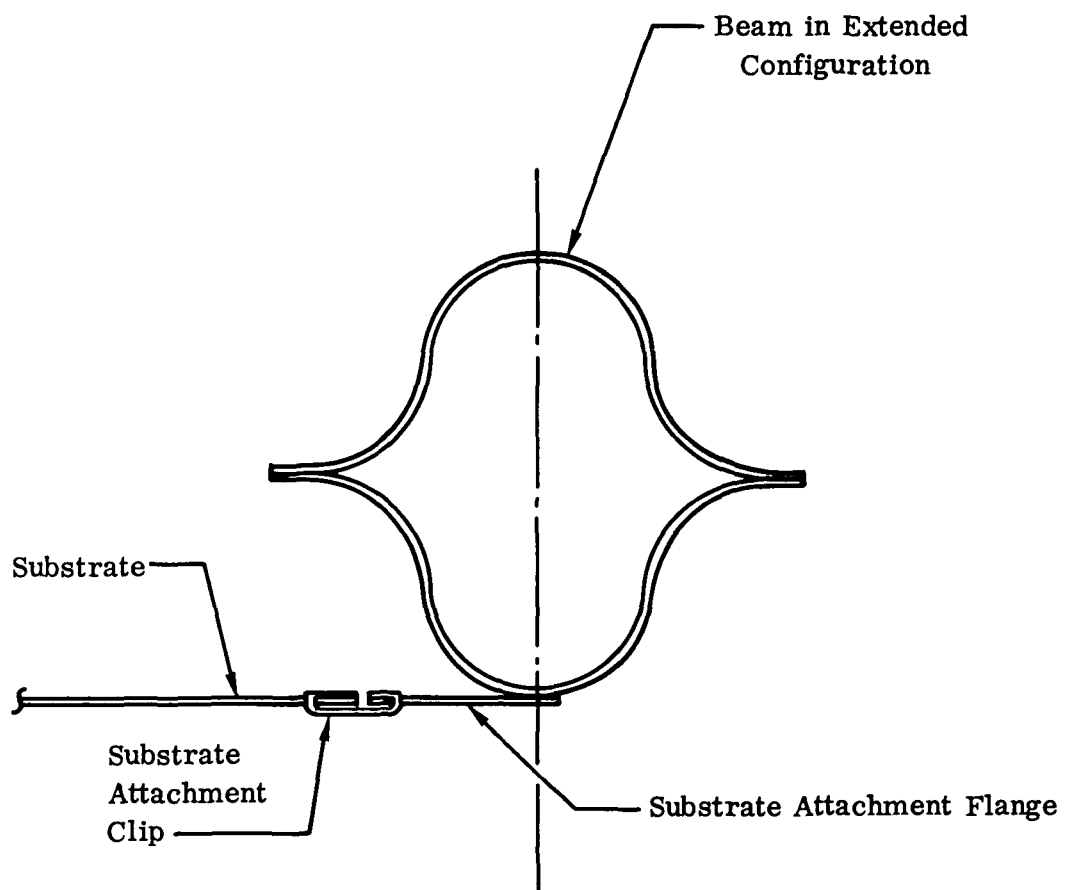


Figure 7 Section Through Deployable Beam

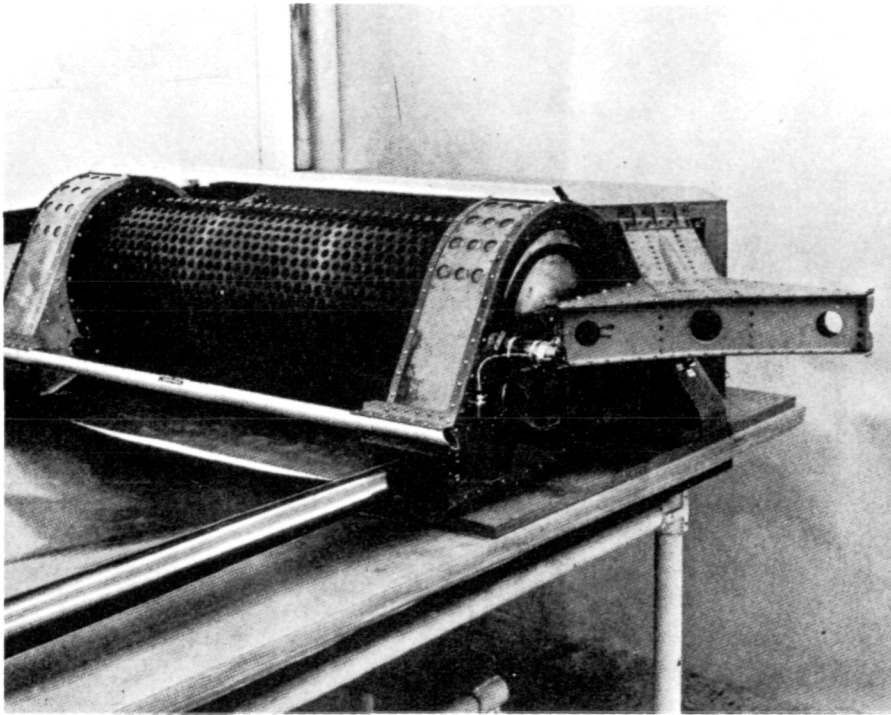


Figure 8 Solar Cell Array Structure

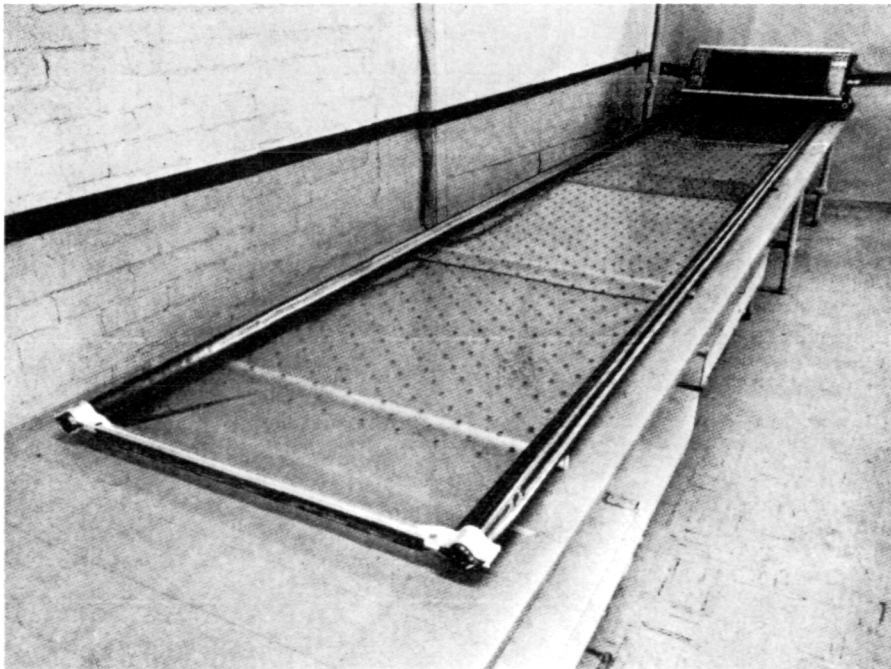


Figure 9 Solar Cell Panel - Deployed

two spring loaded rollers in each end assembly hold the beam against the drum. As the direction of drum rotation is reversed, the pressure on the beam is removed and the beam returns to its original shape. A silicon rubber spacer strip is bonded to the rear surface of the beam to provide a controlled space between the substrate layers as they wrap on to the storage drum. The drum is rotated on its bearings by the drive motor working through a spur gear.

The drive motor is a 24 volt dc unit with an integral planetary gear reduction cage. The motor speed is 12,000 rpm. The gear box output shaft speed is 18.75 rpm. The ratio of drive pinion gear diameter to spur gear diameter produces a roller drum speed of 1.41 rpm and a lineal beam extension rate of 4.43 feet/minute. The torque capability of the motor is one inch-ounce which produces 219 inch-pounds of torque at the roller drum. The power required for extension of the beam is 7.5 watts; for retraction 10.0 watts.

The maximum in and out position of the beam is sensed by a limit switch assembly consisting of aluminum tumblers which allow the roller drum to turn through 1,990° before actuating a double throw microswitch to control the motor.

The mount structure, which adapts the solar panel to the spacecraft is fabricated from 0.032 inch magnesium sheet. The part is designed as two closed intersecting torque boxes which serve to beam the load from adjacent panel assemblies to a cantilevered torque box which transfers the load to the spacecraft.

Figure 9 is a photograph of the solar cell panel fully deployed showing the rear surface of the substrate and the outboard end of the beam assembly.

The basic material of the substrate is epoxy resin impregnated fiberglass cloth. The substrate is 0.0003 inch thick. The slotted edge is built up to 0.012 inch thickness. The ends of each substrate are provided with piano-type hinges which permit modules to be easily replaced.

The back surface of the substrate contains 1/2-inch diameter silicone foam rubber pads to cushion the solar cells. The pads are faced with 0.0002 inch thick teflon sheet. Four modules attached end to end provide 51 square feet of solar cells.

The outside of the drum (Figure 10) is set by JPL specification to a minimum of one foot diameter. The drum consists of a cylindrical 0.025 sheet magnesium skin 43.1 inch long. Approximately 30% of the area of this skin is removed by lightening holes. An access door is provided for electrical harness access. At each end of the drum assembly is a honeycomb sandwich bulkhead. A magnesium hub is mounted at the center of each sandwich to support the static and dynamic loads of the drum assembly. These hubs act as an axle for rotation and transfer loads to the support structure. A magnesium spur gear is attached at the one end to provide the method of driving the drum. At the other end is a guide for the electrical harness.

Figure 11 shows the electrical harness in the deployed and retracted configuration.

The coiled harness permits rotation of the drum while routing the power from the panels through the structure. Eight #22 wires are attached to a fiberglass strip which serves as a spring carrier. This flat harness is spirally wrapped around a spool inside the drum assembly in a retainer, created by the structural bulkheads in the drum. One of the harnesses is terminated in an electrical connector secured to the drum. Access to this connector is obtained through the access door in the drum. It is here that the electrical feed from the cells is connected. The opposite end of the harness feeds through the spool and is secured to the panel assembly support structure. The spool is used to wind the spiralled harness to a compact coil for its stowed position during the boost phase. When the panel assembly is deployed, the rotating drum causes the tightly wound spiral to unwind gradually during the 5-1/2 turns of the drum. The harness always remains confined within the drum assembly and merely relaxes or tightens on the spool as the substrate deploys or retracts.

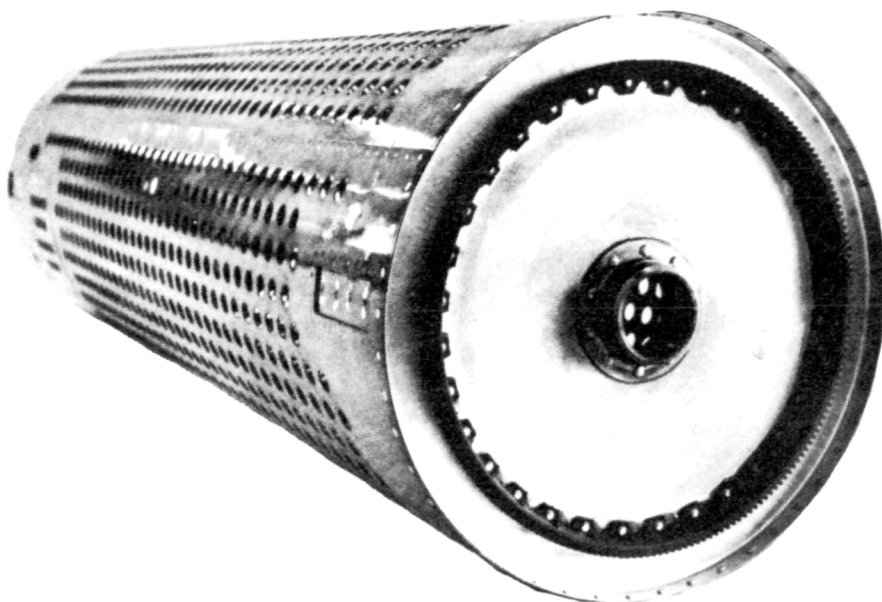


Figure 10 Storage Drum

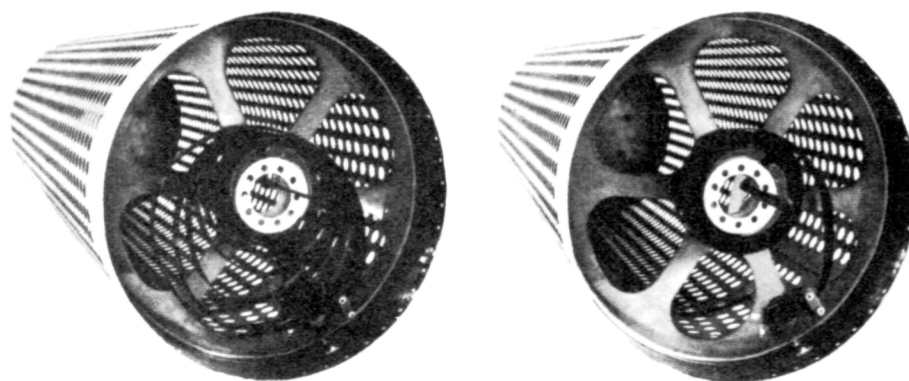


Figure 11 Electrical Harness-Deployed and Retracted

Figure 12 is a photograph of the support structure end assembly showing the internal construction.

The end assembly is a magnesium housing for the spring loaded rollers which restrain the beam. Mounting bosses are attached to the sides of the assembly for the installation of beam support guides. The structure provides for mounting the assembly on the spacecraft and for installation of the drive motor. This part of the assembly transfers loads, imposed by the extended beam, to two cross beams which tie the two assemblies together.

The limit switch tumbler assembly can be seen at the center of the bearing mount for the storage drum.

Figure 13 is a photograph of the beam guide sleeve.

The beam guides are shaped to the contour of the beam at the transition area. These beam guides react loads imposed by the beam at full extension and also follow the changing position of the drum tangent point as successive layers are added or removed. The guide is pivoted at the outer end. A cross shaft which is rotated by the beam travel on a friction wheel causes rotation of a cam support and provides a means of reacting beam loads at any position along the total travel of the beams. The guide is fabricated from fiberglass reinforced epoxy resin and stiffened magnesium formers. The inner surface is lined with Teflon.

The design is shown in detail by Ryan drawings 208V001 through 208V012, and specifications 208S001 and 208S002. These drawings and specifications and additional detail information are described in the Final Design Report (Reference 2).

4.1.2 Solar Cell Installation Design

In order to evaluate the effects of the various environments on solar cell performance, a detail design of the active and simulated solar cell installation was completed.

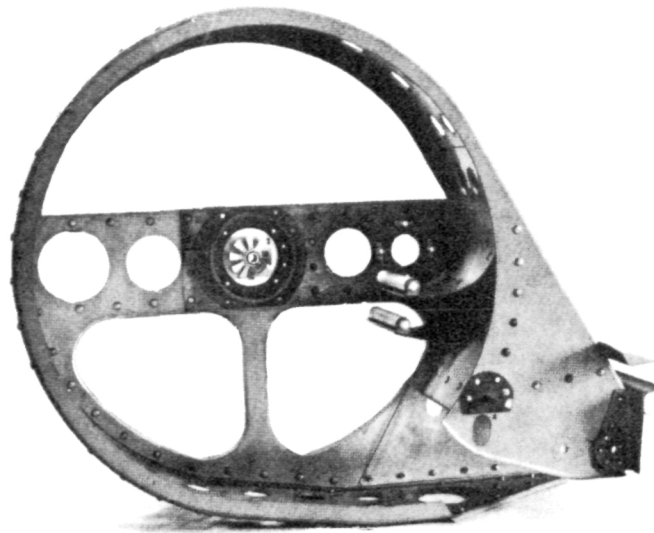


Figure 12 Support Structure End Assembly

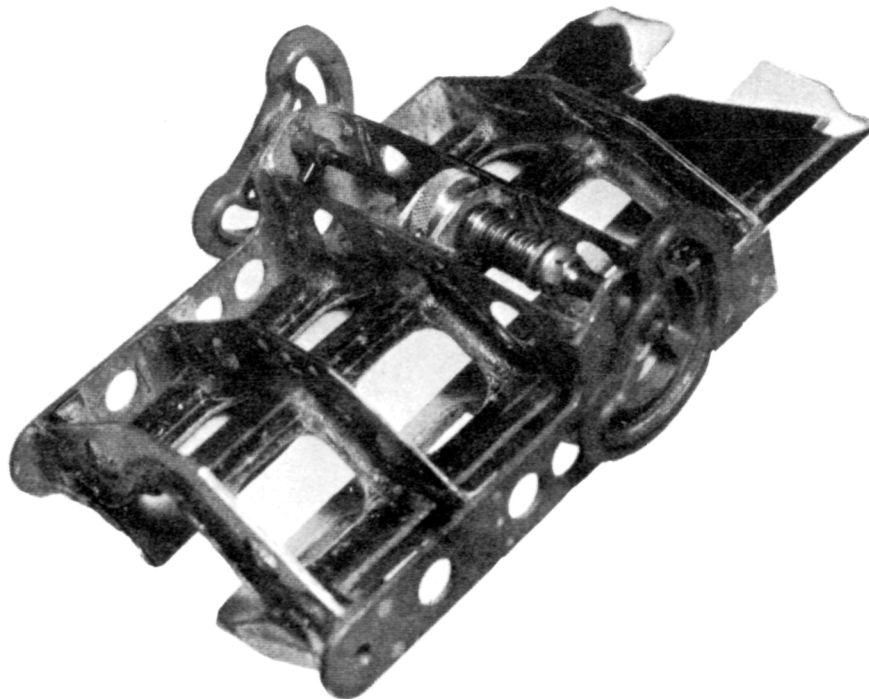


Figure 13 Guide Sleeve-Beam Support

Figure 14 shows the installation of solar cells on the panel substrates. Fourteen solar cell submodules are arranged in pairs at seven locations on the panels. The active cell locations were chosen to provide cells at the most critical areas during launch and deployment testing. The five center groups provide a stack of five layers at the drum center where maximum launch loads occur. The other two cell groups are located at areas of maximum and minimum substrate distortion previously observed in deployment cycling.

The cell submodules are 4 x 14 groups of 8 mil 2 cm x 2 cm N/P silicon solderless solar cells with 3 mil cover slides and Solaflex molybdenum interconnects. The submodules are wired to provide 28 cells in series in each panel group. Each of the four substrate modules is connected to the receptacle located on the support structure, permitting separate measurement of the performance of each panel module. For convenience, Teflon coated copper wire is used for the power connectors. Flat ribbon conductors were considered in the design, and would be preferred in a complete electrical system.

Aluminum alloy dummy chips, 0.016 inch thick, are used to simulate the mass of solar cells and wiring elsewhere on the panel surface. The solar cells are bonded to the substrate using RTV 40 adhesive. Process specification 208S003 was prepared to control the method of application of solar cells, dummy chips, and wiring installation. (See Appendix 9.6)

In the original design, the four substrate modules provided an area of 51 square feet, with an additional 6 square feet available for mounting cells by extending the substrate inboard to meet the storage drum.

The inboard substrate module was redesigned to attach to the wrap drum and provide 57 square feet total area. It was then possible to set back the outer boundary of the solar cell area (50 square feet) by 13 inches from the end of the panel. This change allowed all solar cells to be wrapped on the cylindrical surface of the storage drum.

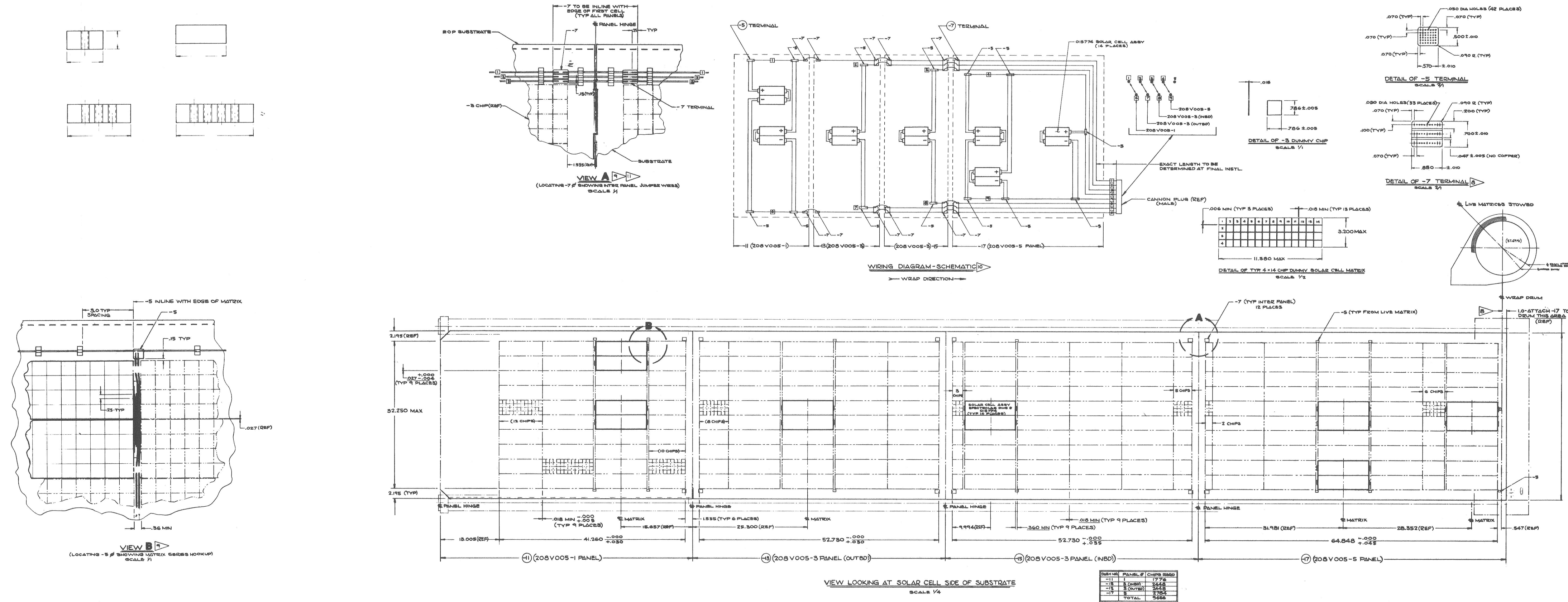


Figure 14. Solar Cell and Wiring Layout for Test of Deployable Solar Panel

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A new end plate stiffening bracket assembly was designed to replace the hat section stiffener shown in Figure 4, which joined the two end assemblies across the extended solar panel surface. This bracket detail is shown on E.O. 366013 to drawing No. 208V006. (See Ref. 5.)

4.1.3 Weight Summary

The calculated maximum and minimum weights (based on all sheet thicknesses maximum or minimum, respectively) shown below are compared with actual item weights prior to refinement modifications to the panel. The calculated weight summary shown is approximately 1.3 pounds greater than given in the final design report, which is accounted for by a few small items unintentionally omitted from that analysis. Mainly, the difference between the actual and calculated weights shown below is attributed to the increase in sheet thickness for the stowed beam radial restraint springs (Part No. 208V006-43); resulting weight difference was approximately 1.6 pounds. The structure unit area weights given below are based on the actual available solar cell area of 51 square feet.

WEIGHT SUMMARY PER PANEL

Drawing Number	Assembly Name	Reqd.	Calculated		Actual
			Maximum	Minimum	
208V001	Solar Panel Assembly	1	2.9196	2.8274	2.6639
208V002	Support Structure *	1	2.0784	1.9650	2.0392
208V003	Drum Assembly	1	5.0267	4.6283	5.0308
208V004	Beam Guide Installation	1	1.5372	1.4082	1.5036
208V005	Substrate Assembly	1	2.3896	1.9441	2.3072
208V006	End Cap Assembly	1	4.2150	3.7060	6.5362
208V007	Beam Installation	1	5.7780	5.3120	4.8720
208V008	Harness Assembly *	1	0.3867	0.3669	0.3726
TOTAL STRUCTURE			24.3312	22.1579	25.3255
TOTAL LESS *			21.8661	19.8260	22.9137

Total Structure Weight/Unit Area of Solar Cells (lbs/ft²)

Calculated		Actual
Maximum	Minimum	
0.4771	0.4345	0.4966

208V001 - SOLAR PANEL ASSEMBLY

Item	Dash No.	Reqd.	Calculated		Actual
			Maximum	Minimum	
Motor Drive Assembly	-3	1	0.0362	0.0344	0.03196
Mount Plate	-5	1	0.0143		0.01133
Hat Section	-7	1	0.5450	0.5190	0.4850
Hinge Pin	-11	4	0.1040	0.0936	0.1190
Weld Assembly	-13	1	0.0103	0.0093	0.02094
Strut	-19 / -20	2	0.0088		0.0088
Cross Tie	-21	1	0.5677		0.5677
Drive Motor	—	1	0.7500		0.7562
Bearing	—	2	0.2600		0.2478
208V010 - Gear	—	1	0.0187		0.0187
208V006-41	—	1	0.0348	0.0314	0.04291
208V011-1 - Switch	—	1	0.1200		0.08350
208V012 - Connectors	—	124	0.2728	0.2252	0.123
Nuts, screws, rivets, adhesive and miscellaneous	—		0.1770		0.1441
TOTAL (208V001)			2.9196	2.8274	2.6639

208V002 - SUPPORT STRUCTURE

Item	Dash No.	Reqd.	Calculated		Actual
			Maximum	Minimum	
Web	-5	2	0.3400	0.3180	0.5048
Spar Channel	-7	2	0.4380	0.4120	0.4100
Closure Channel	-9	1	0.209	0.196	0.2061
Support Channel	-11	2	0.2520	0.2360	0.2513
Rib	-13	1	0.036	0.034	0.0573
Bracket	-15	2	0.3440	0.3280	0.2183
Bracket	-17	2	0.1620	0.1540	0.1014
Clip	-19	2	0.0140	0.0130	0.0198
Clip	-21	2	0.0194	0.0180	0.0132
Doubler	-23	2	0.1360	0.1280	0.1587
NAS 1003-III Bolt Nutplates, Rivets, and Miscellaneous	—		0.1280		0.0983
TOTAL (208V002)			2.0784	1.9650	2.0392

208V003 - DRUM ASSEMBLY

Item	Dash No.	Reqd.	Calculated		Actual
			Maximum	Minimum	
Skin	-15	1	2.093	1.936	1.9885
Spacer	-17	2	0.0126	0.0114	0.0166
Doubler	-19	2	0.0252	0.0226	0.0220
208V009 - Gear	—	1	0.2741	0.2479	0.3329
Bulkhead	-23	1	0.0820	0.0759	0.0694
Door Skin	-25	1	0.0287	0.0265	0.0265
Spacer	-27	2	0.0062	0.0056	0.0034
Spacer	-29	2	0.0052	0.0048	0.0034
Spacer	-31	1	0.0154	0.0142	0.0165
Doubler	-33	1	0.0266	0.0246	0.0265
Bushing	-35	2	0.1960		0.1960
Bulkhead	-37	1	0.1374	0.1271	0.1279
Spool	-39	1	0.0557	0.0469	0.1014
Snap Ring	-41	1	0.0060	0.0054	0.0088
End Plate	-45	2	1.2286	1.1056	0.7892
Honeycomb	-47	2	0.0360	0.0342	0.0351
Skin	-49	2	0.7980	0.7436	0.6547
Miscellaneous					0.6120
TOTAL (208V003)			5.0267	4.6283	5.0308

208V004 - BEAM GUIDE INSTALLATION

Item	Dash No.	Reqd.	Calculated		Actual
			Maximum	Minimum	
Guide Assembly	-3	1	0.7686	0.7041	0.7518
Guide Assembly	-4	1	0.7686	0.7041	0.7518
TOTAL (208V004)			1.5372	1.4082	1.5036

208V005 - SUBSTRATE ASSEMBLY

Item	Dash No.	Reqd.	Calculated		Actual
			Maximum	Minimum	
Substrate	-7	4	1.9588	1.6100	1.7901
Hinge	-9	7	0.0910	0.0714	0.1698
Hinge	-11	3	0.0390	0.0306	
Retainer	-13	1	0.0152	0.0130	0.02425
Stiffener	-15	1	0.0309	0.0253	0.02976
Damper Pad Assembly	-17	901	0.1747	0.1398	0.1982
Primer, Adhesive and Miscellaneous			0.0800	0.0540	0.0951
TOTAL (208V005)			2.3896	1.9441	2.3072

208V006 - END CAP ASSEMBLIES

Item	Dash No.	Reqd.	Calculated		Actual
			Maximum	Minimum	
End Plate	-5 / -6	2	0.1320	0.1200	0.2116
Angle	-7 / -8	2	0.1046	0.0974	0.0838
Angle	-9 / -10	2	0.1086	0.1014	0.0838
Former	-11	2	0.1920	0.1800	0.2050
Former	-13	2	0.1170	0.1096	0.1300
Doubler	-15	2	0.0936	0.0842	0.0889
Doubler	-17	2	0.0620	0.0580	0.0618
Angle	-19 / -20	2	0.3340	0.3160	0.3440
Former	-21 / -22	2	0.4220	0.4040	0.4144
Bracket	-23 / -24	2	0.0244	0.0230	0.0352
Bracket	-25 / -26	2	0.0110	0.0104	0.0286
Angle	-27 / -28	2	0.0300	0.0280	0.0440
Angle	-29	2	0.0360	0.0340	0.0540
Angle	-31	2	0.0194	0.0182	0.0110
Angle	-33 / -34	2	0.0150	0.0138	0.0110
Cover	-35 / -36	2	0.5680	0.5260	0.5732
Housing	-37 / -38	2	0.1832	0.1648	0.1786
Spool	-39	4	0.0296	0.0268	0.0441
Retainer	-41	1	0.0348	0.0314	0.04298
Spring	-43	44			2.6677
Bracket	-45	88			0.09699
Tube	-47	44			0.6301
Plug	-49	88	1.6368	1.2980	0.1936
Sleeve	-51	88			0.2908
Spacer	-53	2	0.0110		0.0110
Miscellaneous			0.0500		0.0500
TOTAL (208V006)			4.2150	3.7060	6.5362

208V007 - BEAM INSTALLATION

Item	Dash No.	Reqd.	Calculated		Actual
			Maximum	Minimum	
Beam Assembly	-5	1			1.95
Beam Assembly	-6	1			1.95
Cap	-7	1	1.0400	3.7360	—
Strip	-9	2	0.3310	0.3080	—
Plate	-11	2	0.0300	0.0280	0.0330
Plug	-15	2	0.0640	0.0600	0.0838
Channel	-17	2	0.2940	0.2640	0.300
Angle	-19	2	0.1040	0.0940	0.0926
Strip	-21	8	0.4080	0.3760	—
Strip	-23	8	0.0480	0.0400	—
Strip (Rubber)	-25	2	0.4260	0.3840	0.3510
Hinge	-27	1	0.016	0.014	0.02645
Adhesive, Screws, Rivets		1	0.010	0.008	0.0851
TOTAL (208V007)			5.7780	5.3120	4.8720

208V008 - HARNESS ASSEMBLY

Item	Dash No.	Reqd.	Calculated		Actual
			Maximum	Minimum	
Strip	-5	1	0.0562	0.0505	—
Cable	-7	1	0.0327	0.0294	—
Clamp	-9	1	0.0183	0.0146	0.0209
Lug	-11	2	0.0029	0.0027	0.0026
Tube	-13	1	0.0018	0.0017	—
Bracket	-15	1	0.0071	0.0066	0.0066
Molding	-17	1	0.0630	0.0600	0.0154
Molding	-19	1	0.0620	0.0590	0.0073
Strip	-21	2	0.0013	0.0010	—
Strip	-23	20	0.0014	0.0014	—
Connector - Electric	—	1			0.0121
Screws, Adhesive, etc.	—		0.1400	0.1400	
TOTAL (208V008)			0.3867	0.3669	0.3726

The resulting electrical power/structure weight ratio based on the above structure weight of 25.33 pounds, and a solar cell power output of 10 watts/square feet at 1 A.U., and an available solar cell area of 51 square feet is calculated as:

$$\frac{51 \times 10}{25.33} = \frac{510}{25.33} = 20.13 \text{ watts/pound of structure}$$

If we consider an electrical installation weight of 0.3 pound/square foot as defined in the Panel Design Contract Statement of Work, the above power/weight ratio becomes:

$$\frac{\text{Watts/Square Foot}}{\text{Structure Weight/Square Foot} + \text{Electrical Weight/Square Foot}} = \frac{10}{0.4966 + 0.3} = 12.55 \text{ watts/pound}$$

If an electrical installation with 8 mil solar cells and 3 mil cover glass were used, representing 0.165 pound/square foot as given in Ryan Report No. 40067-4 (Ref. 8) the power/weight ratio becomes:

$$\frac{10}{0.4966 + 0.165} = 15.12 \text{ watts/pound.}$$

Final refinements made to the panel resulted in cutting one foot off each beam that had been damaged during the thermal vacuum test. For that reason, the following weight for the delivered article is listed for information only:

Total weight as delivered = 39.769 pounds, of which 26.399 pounds is associated with structure and 13.37 pounds is associated with the dummy solar cell and live solar cells and wiring installation.

4.2 TEST PLAN AND PROCEDURES

4.2.1 Test Plan

A detailed test plan was prepared to establish the criteria for evaluation of

the solar array structure. The plan and procedures are contained in Report 20865-1, Test Plan and Procedures for Phase IV Nondestructive Testing and Evaluation of the Prototype Deployable Solar Array Structure Manufactured Under Contract 951106 (Reference 4).

Tests were conducted to:

- a. Assure satisfactory operation of the deploying beam and mechanism and solar cells in a space environment with thermal environment at the upper limit comparable to the Mariner 67 Venus Mission, (260MW/CM²).
- b. Accumulate temperature gradients and temperatures at selected locations on the beams and substrate for correlation with theoretical values.
- c. Determine if the solar cell protection pads are sufficient to prevent launch vibration damage to the solar cells and electrical wiring.
- d. Compare drive torque requirements, under simulated vacuum and IR flux environment, with sea level requirements.
- e. Assure that the deployable beams will carry the steady state acceleration induced to the deployed array during cruise maneuver.

Tests of the acoustic and sterilization environments were not included in the scope of the test program. The predicted loads during acoustic environment are substantially below those of random vibration. Therefore it is considered that satisfactory performance of the array during random vibration would imply conformance to the acoustic environment.

Temperatures expected for the array during thermal vacuum tests at the 275°F near Venus environment approach the 295°F sterilization requirements. During sterilization, the array would be in the stowed configuration which would impose creep loads on plastic components which are not seen in thermal vacuum.

Conformance of the array to creep during sterilization is implied by the development testing of plastic materials conducted during Phase I (Reference 1).

4.2.2 Test Procedures

The tests are based on JPL design criteria for the deployable solar array (see Reference 2, Ryan Report No. 20869-2, 5 May 1966) with modification of the thermal vacuum cycling criteria so that it is comparable with both a Venus mission and a Mars Mission.

The tests consist of the following:

- a. Simulation of 0.2g cruise maneuver.
- b. Deployment cycle effects on solar cells in a 1.0 g ambient environment.
- c. Launch vibration environment consisting of sinusoidal vibration at 1.6g rms from 6 to 20 Hz and 4.0g rms from 20 to 200 Hz; also random vibration at $0.1g^2/Hz$ and $0.2g^2/Hz$ from 200 to 2000 Hz each for 180 sec. Sinusoidal and random environment induced independently.
- d. Deployed dynamic characteristics to determine the beam first cantilever bending mode, panel first torsion mode, and substrate first mode.
- e. Thermal vacuum test consisting of two deployment and retraction cycles simulating Mars and Venus Missions. The test simulates a condition in which the shroud is ejected while the spacecraft is in orbit in earth shadow. The panels deploy while the spacecraft is still in earth shadow, then the spacecraft moves from shadow into sunlight and proceeds on a Mars mission. Panels retract and deploy somewhere enroute. (Solar flux is assumed, conservatively, equal to that near Earth.)

Then the spacecraft proceeds on a Venus mission. The panels retract and deploy somewhere enroute. (Solar flux is assumed, conservatively, equal to that near Venus.)

4.2.3 Special Test Equipment

A deployment support fixture shown in Figure 15 was designed for support of the panel during horizontal deployment, drawing SK9112 (See Appendix 9.6). Roller supports are provided under the beams and also under the deployable substrate. The fixture has optional positions so that the array may be supported and deployed with the cells up or down, with or without substrate support. A counter-balance arrangement provides a means for simulating the 0.2g cruise maneuver load. The fixture was designed to be used to support the array during deployment and retraction in thermal-vacuum tests. The substrate support rollers are covered with silicone sponge rubber and are separated sufficiently to make negligible any blocking of the panel rear surface. Experience in deployment cycling tests showed the need for modification to reduce friction and increase the number of substrate support rollers (See Section 4.4.2.3)

Two vibration fixtures were designed to support the panel during launch environment tests. The fixtures, of welded magnesium plate, are shown in drawings TR-1148-1 and TR-1148-10 (See Appendix 9.5).

Electrical test equipment was designed and assembled to facilitate checking electrical characteristics in accordance with the test plan. The application of this equipment in the test program is discussed in Section 4.4.2.5 of this report.

4.3 ASSEMBLY OF SOLAR CELL ARRAY

This section describes the events of assembly of the active and simulated solar cells to the substrates and their installation on the solar panel support structure. The 0.2g cruise maneuver test was conducted before solar

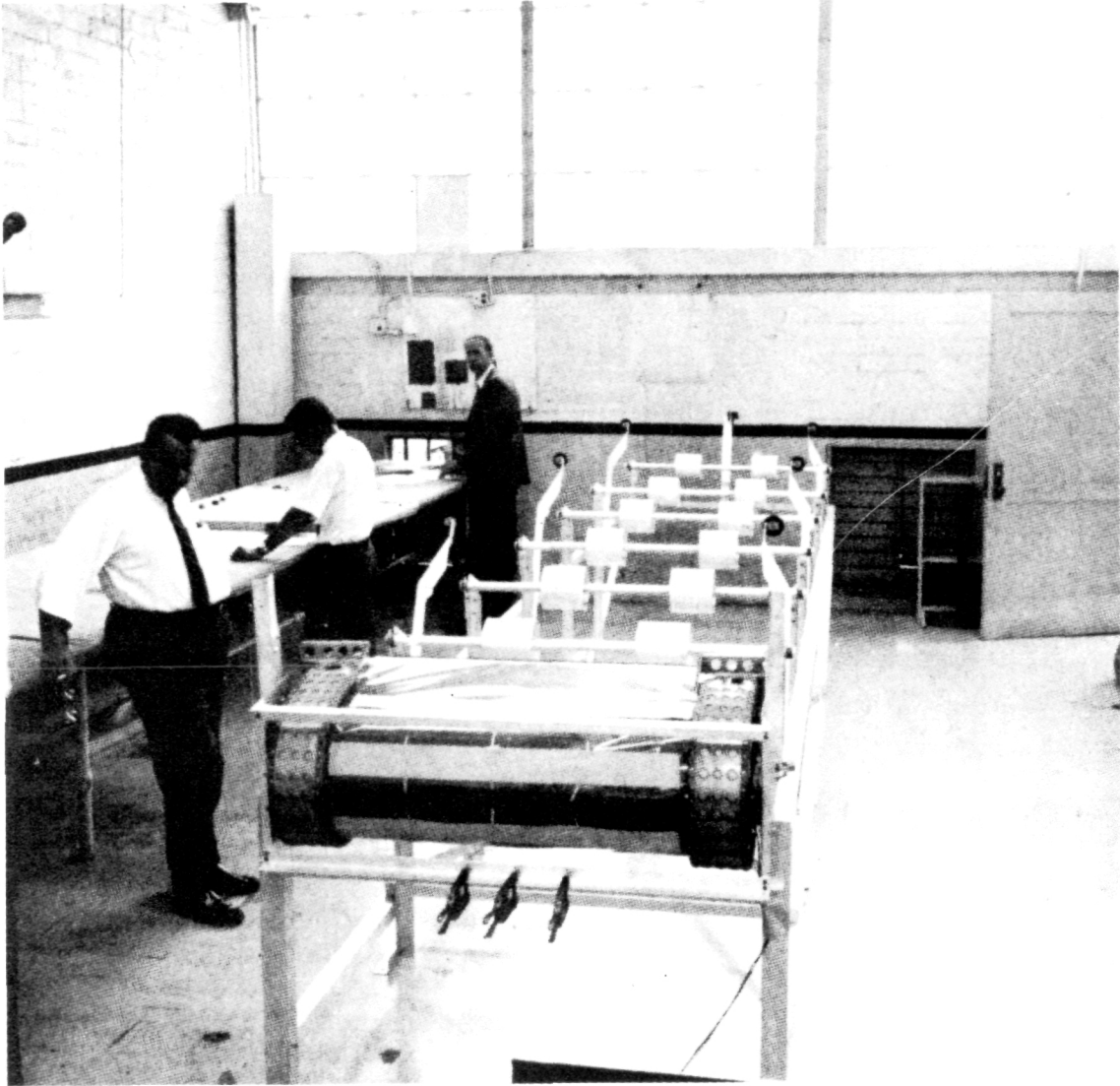


Figure 15 Deployment Support Fixture

cell installation. The bare substrates were used in order to simulate the desired mass for cruise maneuver loads. Subsequently, the solar cell installation was completed.

4.3.1. Solar Cell Installation

Sixteen 4 x 14 submodules were provided by JPL for installation on the substrate. The electrical performance of each matrix was measured and I-V curves plotted. These curves are available for reference at this contractor's facility. The tests were conducted outdoors in direct sunlight. Solar illumination was determined by a secondary standard cell furnished by JPL. These results were correlated with the performance data obtained by Spectrolab Division of Textron, Inc., using a solar simulator intensity of 140 mw/cm^2 . Figure 16 shows the short circuit current plotted with respect to light intensity and indicates the degree of correlation with the Spectrolab data.

After completion of the 0.2g cruise maneuver test, the modular substrates were removed from the beams. Installation of solar cell submodules and dummy cells was accomplished in accordance with drawing 208V013 (Figure 14) and process specification 208S003.

Some problems were encountered during assembly of the solar cell arrays. These involved possible variation in adhesion of solar cells to the fiberglass substrate and some incidents of cracked solar cells during application and installation of the substrates on the array structure.

Variations which occurred in the preparation of the fiberglass substrate with respect to the amount of sanding and adhesive primer thickness could have an effect on low temperature adhesion of the solar cells. In order to measure these effects, an evaluation test to determine the effect of process variables was conducted. This test which is described in detail in Reference 5 indicates that the variations which existed on the solar panels did not prevent satisfactory adhesion of the solar cells.

Of more significance were problems in handling of the solar panels during application and assembly. Handling facilities provided consisted of flat

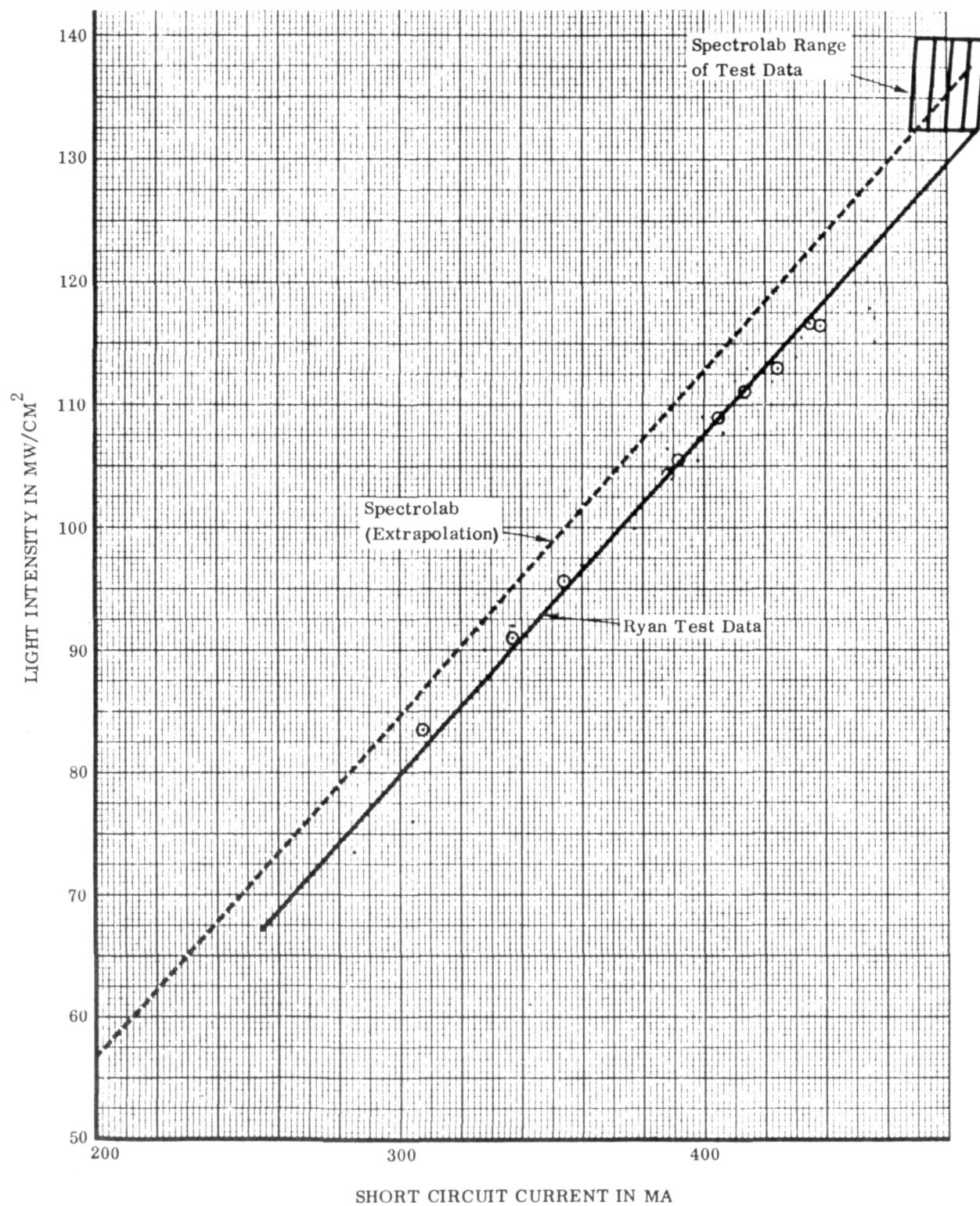


Figure 16 Comparison of Ryan and Spectrolab Test Data on Solar Cell Matrices

boxes lined with foam, flat support tables, and the roller support fixture previously described (see Figure 15). For installation of the solar panels to the support beams, a special plywood board was placed on top of the support rollers to provide a flat surface beneath the substrates.

When assembly was complete, cracks were noted in 37 of the 784 active cells on the array. The majority of these cells were damaged during assembly of the substrates to the beams. The number of cracked cells was much higher than anticipated and can be attributed primarily to two causes. First, some of the damage was caused by personnel during installation of the solar cell modules on the solar panel. Lack of experience in handling the flexible panels contributed to cracking during assembly of the panels to the beams. The substrate slots and beam slots were not exactly coordinated and required special edge connectors to be attached. Also, substrate widths varied and presented some assembly difficulties. The second factor involved the roller support fixture itself. When the special board was removed, it was evident that the central support rollers under the substrates were spaced too far apart to give proper support to the panel and caused undesirable bending of the solar cell modules.

The fixture was then temporarily modified to remove the central support rollers in order to prevent further damage to the cells during deployment cycles.

Figure 17 is a photograph of the completed solar cell installation on the panel, ready for deployment cycling tests.

4.4 TEST RESULTS

This section contains the description and results of each of the environmental tests conducted. The tests are described in the sequence performed. The first test conducted was the 0.2g cruise maneuver acceleration. Following this test, the solar panel was completed with the solar cell mass simulated with active and dummy solar cells. Tests performed were deployment cycling, deployed dynamics, sinusoidal vibration, random vibration, and thermal vacuum at near Earth and near Venus temperatures.

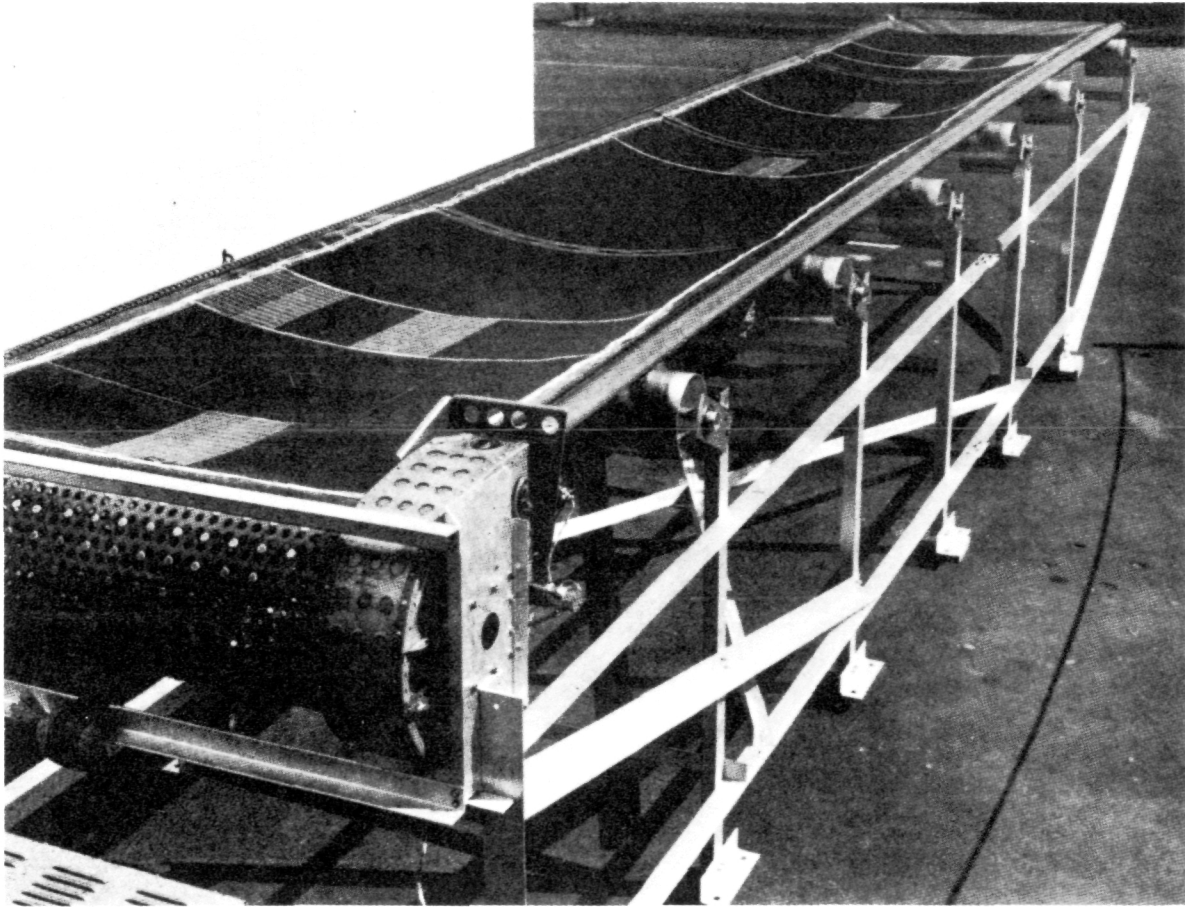


Figure 17 Solar Panel Fully Deployed

4.4.1 Simulation of Cruise Maneuver Acceleration

This test was conducted to simulate a normal-to-panel loading equal to a 0.2g steady state cruise maneuver and a 0.05g steady state cruise maneuver lateral (in plane) loading with the panel in the deployed position. Load distribution and bending moments were of concern with the 0.2g loading whereas with the 0.05g loading, a concentrated tip load was applied to simulate only maximum moment. The test was conducted in a sea-level environment with the 1g environment corrected to 0.2g using (1) mass balances for the beams (see pages 8 through 10, Ryan Report No. 20865-1) and (2) lighter mass on the substrate than 0.3 pound/square foot accomplished using masking tape. After correcting the loading to compensate for a 6 percent reduction in beam modulus of elasticity due to increased temperature of the beams when deployed in space, the added panel weight required was 1.36 pounds (see pages 6 and 7, Ryan Report No. 20865-1); the actual weight added, using three-inch wide making tape, was 1.23 pounds. The fixture to which the panel mounts and is supported as it deploys is shown in the Appendix, Ryan Drawing No. SK-9112.

Test Procedure

The sequence of testing was as follows:

- a. Deploy panel, with cell side up, out over support rollers in a 1g ambient environment and check lateral position of beam tips. Then counterbalance 79% of beam weight, remove support rollers, manually lower beams until beams completely support load, and measure beam tips and substrate deflections after two minutes. Manually raise beams to horizontal position, add support rollers, and retract substrate. Deploy and check beam tip lateral position again. Record time to retract and deploy panel. This constitutes one test cycle.

There was a deviation at this point in the test from the original test plan (see pages 5 through 12, Ryan Report No. 20865-1). For convenience only, the lateral load test was next conducted.

- b. Add beam mass balance, remove support rollers, and support panel tip with tip roller. Measure panel tip lateral position and apply a 0.6 pound (includes 6% increase to compensate for room temperature testing) lateral load, by spring scale, at beam tip to give an equivalent bending moment at beam guides due to 0.05g acting in the plane of the panel. Record deflection, release load, and check lateral position of beam tips to determine any permanent set. Then apply load in opposite direction. Record deflection, release load, and check lateral position of beam tips to determine any permanent set. This constitutes the 0.05g load simulation test.

At this point, add support rollers, remove mass balances, retract panel, redeploy, and check lateral position of panel tips. Record retraction and deployment time. Retract panel, remove from fixture and remount for normal-to-plane 0.2g simulation load with cell side down.

- c. Test similar to test (a) above, but with cell side down.

Test Results

The test results are given in the order of procedure presentation above and compared with analytical results given in Report No. 20869-2.

a. Cell Side Up, 0.2g Simulation

- Panel deflection measured at beam tip, 24 7/16 inches (by analysis, 17 inches, see page 95, Ryan Report No. 20869-2). Deflection was greater than calculated largely due to looseness of beam at guide.
- Substrate deflection (see Figure 18), 3 5/8 inches.
- Lateral beam deflection (two beams), 13/16 inch (by analysis, 10 inches, see page 76, Selected Design, Ryan Report No. 20869-2).
- Retract time, 3 minutes, 17 seconds

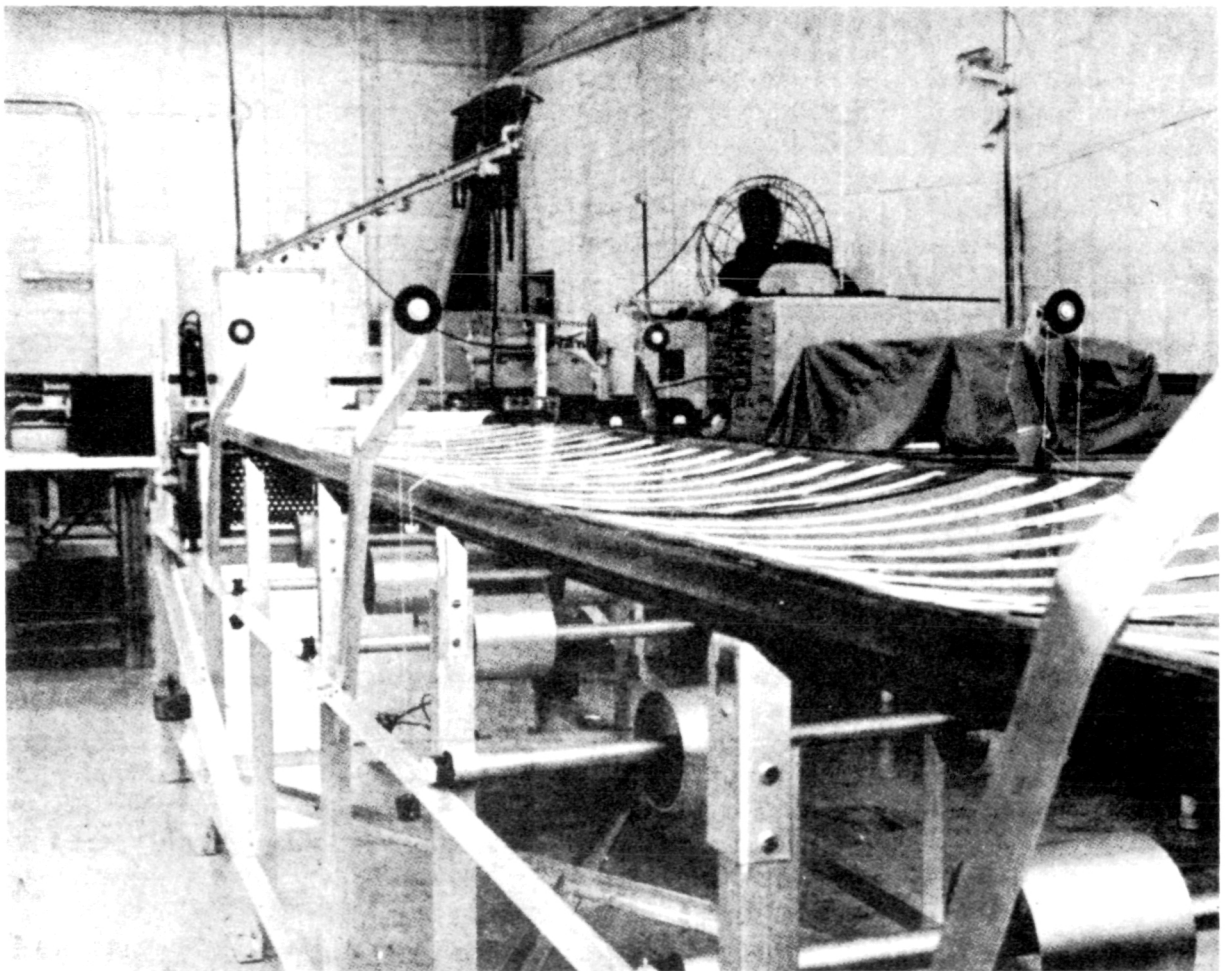


Figure 18 Substrate Deflection Showing Tape Added to Simulate Mass

Deploy time, 3 minutes, 14 seconds

NOTE: The lateral beam deflection was less than expected by analysis because analysis assumed the substrate to be cut to the same width as the beam spacing, in which case larger lateral catenary loads would be transferred to the beams. The effect of this is only about a 20% reduction in lateral beam load and about an equal reduction in panel normal to plane deflection. The balance (larger percent) of the difference between test and analysis is probably due to the amount of fixity provided at the beam ends; fixity error of 72% increase is necessary in addition to the reduction in lateral load due to the excessive substrate width, to justify the reduction. This large an error in fixity is questionable and must be verified by a load-deflection test before we can be certain that this is where the difference is.

No permanent set or elastic buckling of the beams occurred as a result of this limit load test. The beam tip lateral position was the same after redeployment.

b. 0.05g Simulation

- Panel deflection measured at beam tip, 1/8 inch either direction (by analysis, 1.5 inches, see page 40, Ryan Report No. 20869-2).
- Retract time, 3 minutes, 17 seconds
- Deploy time, 3 minutes, 14 seconds

NOTE: The lateral deflection was less than expected by analysis because analysis assumed the beam substrate attach clips do not transfer shear immediately.

No permanent set or elastic buckles occurred as a result of this limit load test. The beam tip lateral position was the same after loads application in each direction.

c. Cell Side Down, 0.2g Simulation

The panel supported approximately 65% of the required bending moment, equivalent to about 0.13g, when elastic buckling of the titanium beams occurred within the guide sleeves with the beam tips deflected 20 inches. It was determined that the reason for premature elastic buckling of the beam was due to insufficient support by the guide sleeves (in the substrate cutaway area), allowing the beam to flatten somewhat (see Figure 19) as it is distorted as the guide, reducing its structural stiffness. Calculations indicated that a section of beam, acting as a doubler, would be required along each beam cap in the guide unsupported area. A section of beam, approximately 12 inches long x 1.7 inches wide was then bonded to each beam in those areas (see Figure 20) using 1200 primer and Dow Corning RTV 3145 adhesive. After cure, the test was re-run with success.

- Panel deflection measured at tip, 20 inches (see Figure 21)
- Retract time, 3 minutes, 29 seconds
- Deploy time, 3 minutes, 13 seconds

No permanent set or elastic buckling of the beams occurred as a result of this limit load test after the beam doublers were added.

4.4.2 Deployment Cycling

The purpose of this test was to determine the deployment cycle effects on solar cells in a 1.0g environment and to observe the deployment behavior of the solar panel. In addition, this test was used as the basic reference to measure changes in the solar panel following each environment. This section contains a summary of all deployment tests conducted in the program.

4.4.2.1 Test Procedure

The initial test consisted of five deployment cycles. The test was started immediately after the solar panel modules were assembled to the beams with

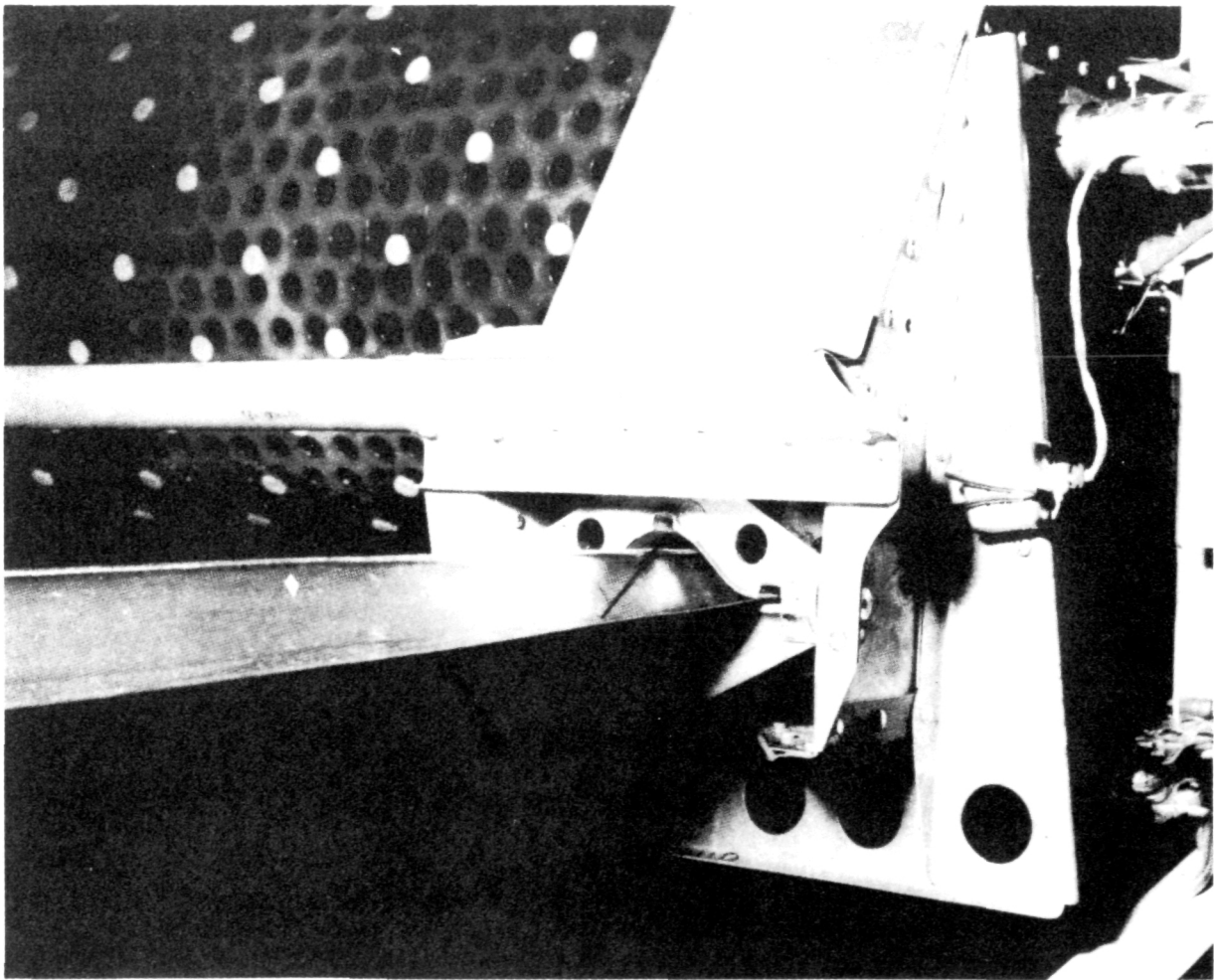


Figure 19 Beam Pull-Away at Guide Sleeve When Loaded (Cell Side Down)

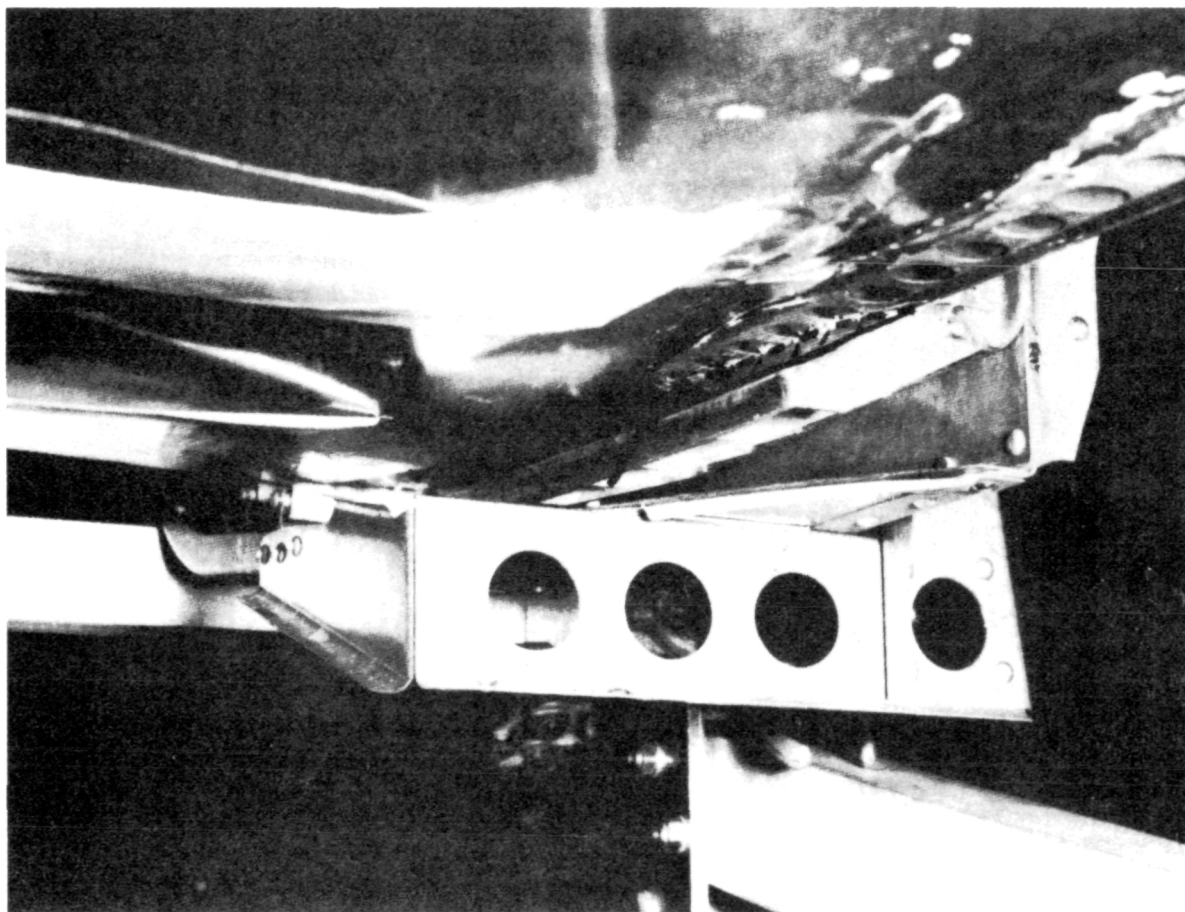


Figure 20 Beam Cap Doubler Added at Guide for Loading (Cell Side Down)

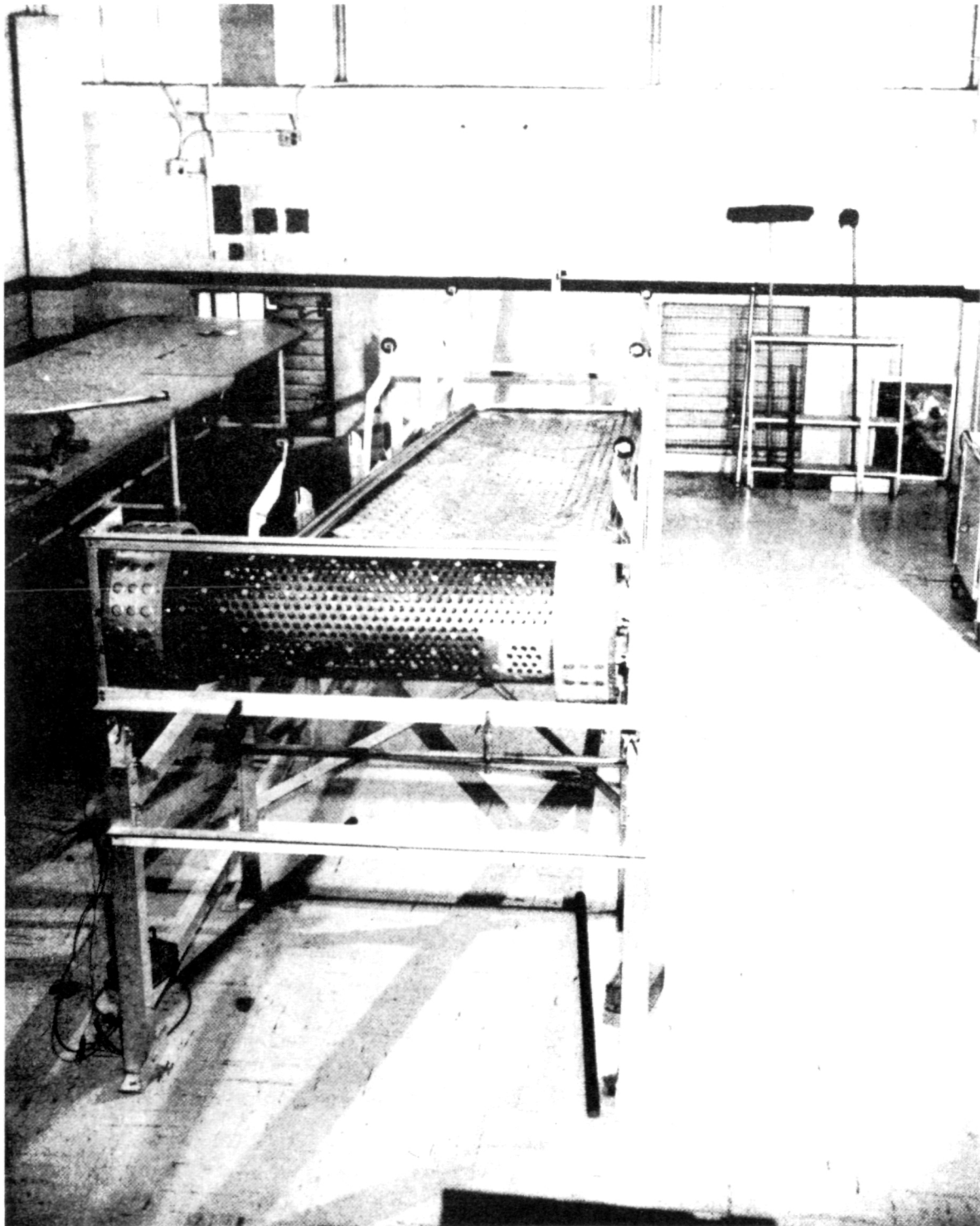


Figure 21 Deflected Panel Under 0.2g Loading (Cell Side Down)

the panel supported on the roller support fixture (see Figure 17). Each cycle consisted of a complete retraction and redeployment.

The test procedure was as follows:

1. Perform visual and dimensional check of solar cells and structure.
2. Measure electrical performance of active solar cells on each panel module under sunlight illumination.
3. Retract and deploy panel observing drive motor current at 27 volts dc.
4. Check visually and electrically after each cycle.
5. Repeat for a total of five cycles.

Test procedures for measuring electrical performance of solar cells and dimensional check of solar panel structure are described in detail in paragraph 4.4.2.5 and in reference 4.

4.4.2.2 Test Results

During the initial deployment cycling tests there was no significant change in the power output of the active solar cell modules. An important factor disclosed was the need for improved handling methods in a 1.0 environment. The substrate support fixture caused excess frictional forces which created an undesirable deflection of the wrapped beam from the storage drum. Also there was in evidence of a few cracked solar cells caused by the support fixture.

Some difficulties were encountered with the functioning of the guide sleeve cam mechanism in the 1.0g field. The rotation of the cam was not always coordinated with the nominal guide position. This had the effect of increasing frictional resistance to deployment and contributed to deflection of the wrapped beam from the storage drum. Sometimes it was necessary to rotate one of the cams manually to maintain the correct guide position.

Some longitudinal crazing or cracking of the substrate was noted in the region between the solar cells and the beam attachment. This condition was related to a variation of the substrate within the retracted and extended position.

The panel structure completed the initial deployment cycling tests in a satisfactory manner with the exception of the guide cam binding and the substrate cracks. The support fixture was not satisfactory. Subsequently, the roller support fixture was modified to reduce friction and provide additional support rollers. The guide sleeve cams were disengaged later in the test program.

The substrate crazes and cracks were sites of an eventual failure in sinusoidal vibration after which they were repaired with a flexible doubler.

4.4.2.3 Description of Deployment Cycles

The first deployment cycle consisted of retraction and redeployment. Motor currents were measured and the behavior of the solar panel was observed. During retraction it was noted that the transition between the deployed panel shape and the stored cylindrical shape caused undesirable distortion of the substrate surface. However, the substrate and solar cells wrapped very smoothly on the storage drum. On deployment, an increase in motor current was noted as the friction forces of the support fixture increased. This caused the beams to deflect away from the storage drum.

The deployment was interrupted so that the beams could be pulled manually to tighten the wrap on the storage drum. Deployment was then completed at lower motor currents.

The partially deployed panel and support fixture are shown in Figure 22.

An alternate bearing system was used in the next deployment cycles but was not effective in reducing deployment forces. Figure 23 shows typical motor current requirements for extension and retraction of the solar panel in a 1.0g field under ambient conditions.

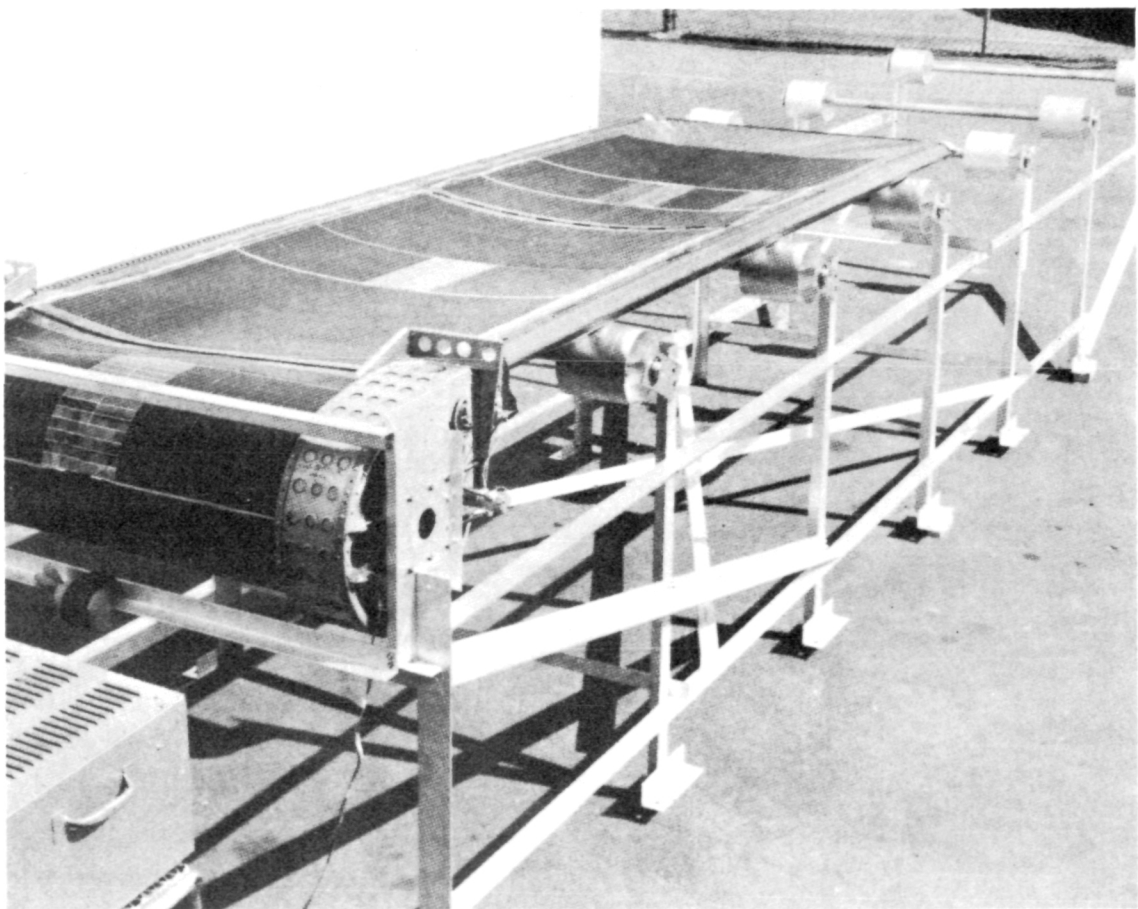


Figure 22 Solar Panel Partially Deployed

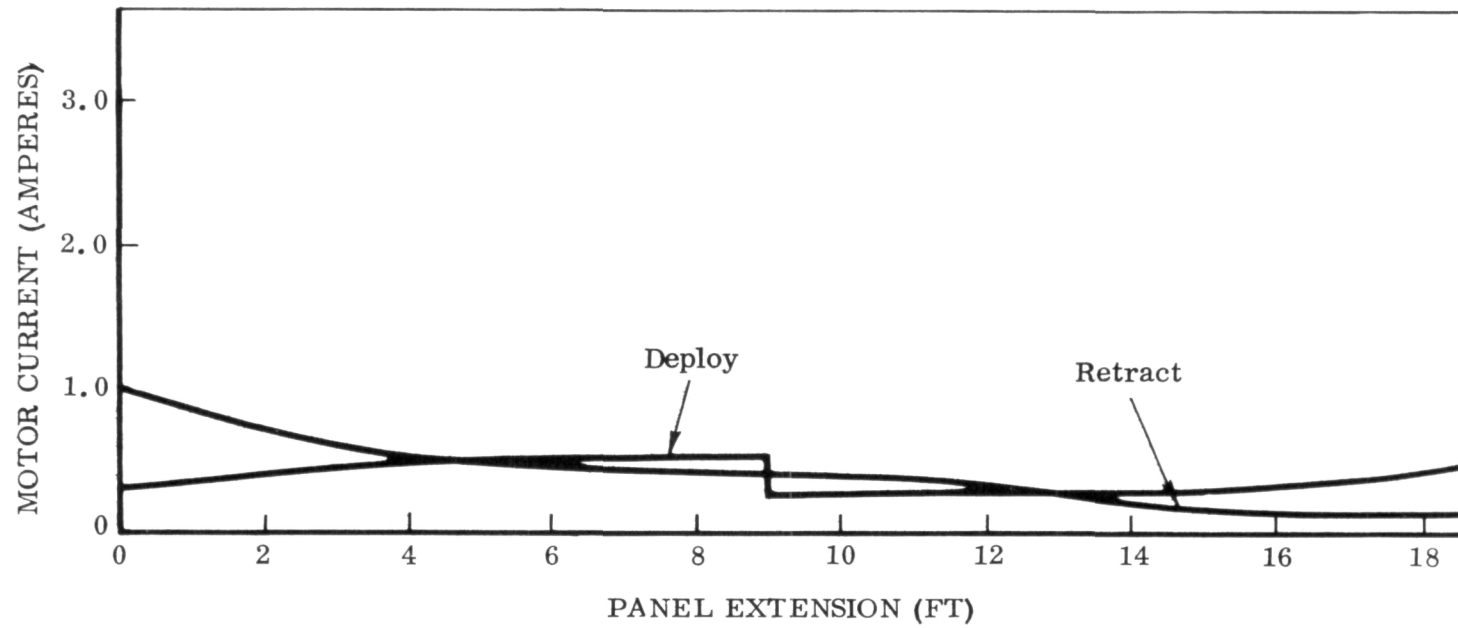


Figure 23 Deployment Motor Current Using Original Support Fixture

During each of the initial deployment tests the panel was manually assisted to overcome support fixture friction. One effect of this was to cause the wrapped substrate layers which remained on the storage drum to slide relative to each other. This resulted in damage to the damper pads which were scraped by transverse wires of the solar cell wiring. This type of damage was related entirely to the deficiency of the support fixture.

Subsequent to the initial tests, the roller support fixture was reworked to provide support rollers under the substrate. A new bearing arrangement using loose fit, low friction ball bearings was also provided under each axle. The reworked fixture is shown in Figure 24. Typical motor current requirements using the reworked support fixture are shown in Figure 25.

Manual assistance for deployment was not required when using the reworked fixture.

Incidence of crazing and cracking of the substrate was observed in the region between the beam attachment and the solar cells. It was observed that a variation in width of the substrate existed between the extended position and wrapped position on the storage drum. When retracted, longitudinal buckling of the substrate occurred in this area, leading to an accumulation of short crazes and cracks approximately 1/2 to 3/4 inch long. The location of the cracks was at the edge of the beam restraint rollers.

The number of crazes and cracks increased with each cycle prior to sine vibration test when long tears occurred at the same site. This condition was eliminated by repairing the substrate with a flexible fiberglass doubler bonded with Silastic 140 adhesive.

Table 1 summarizes the physical changes to the solar panel during deployment tests. The variations due to deployment cycling only are discussed here. Also summarized are other physical changes which are discussed in detail in Sections 4.4.4 and 4.4.5.

During the entire program the panel was deployed a total of 122 cycles. A summary of these events is presented in Section 9.4.

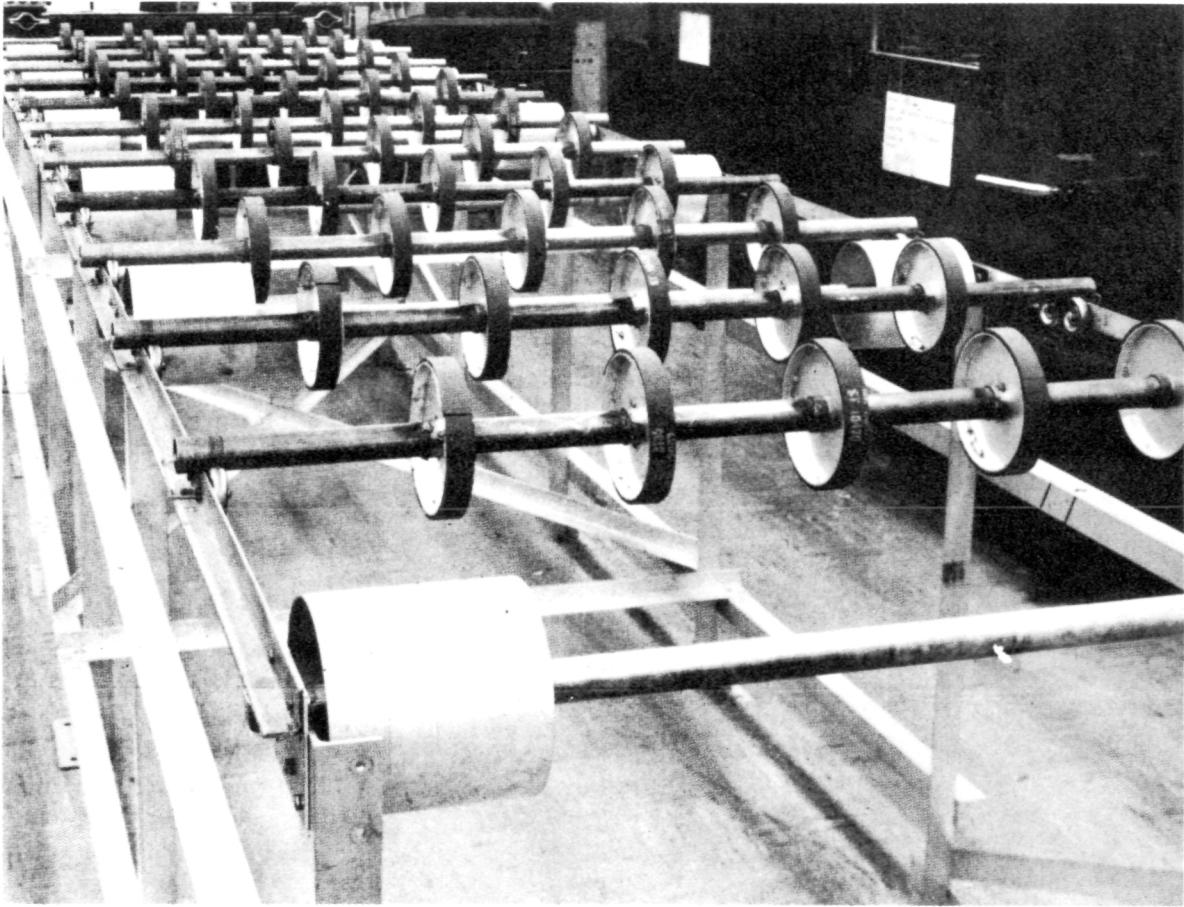


Figure 24 Reworked Roller Support Fixture

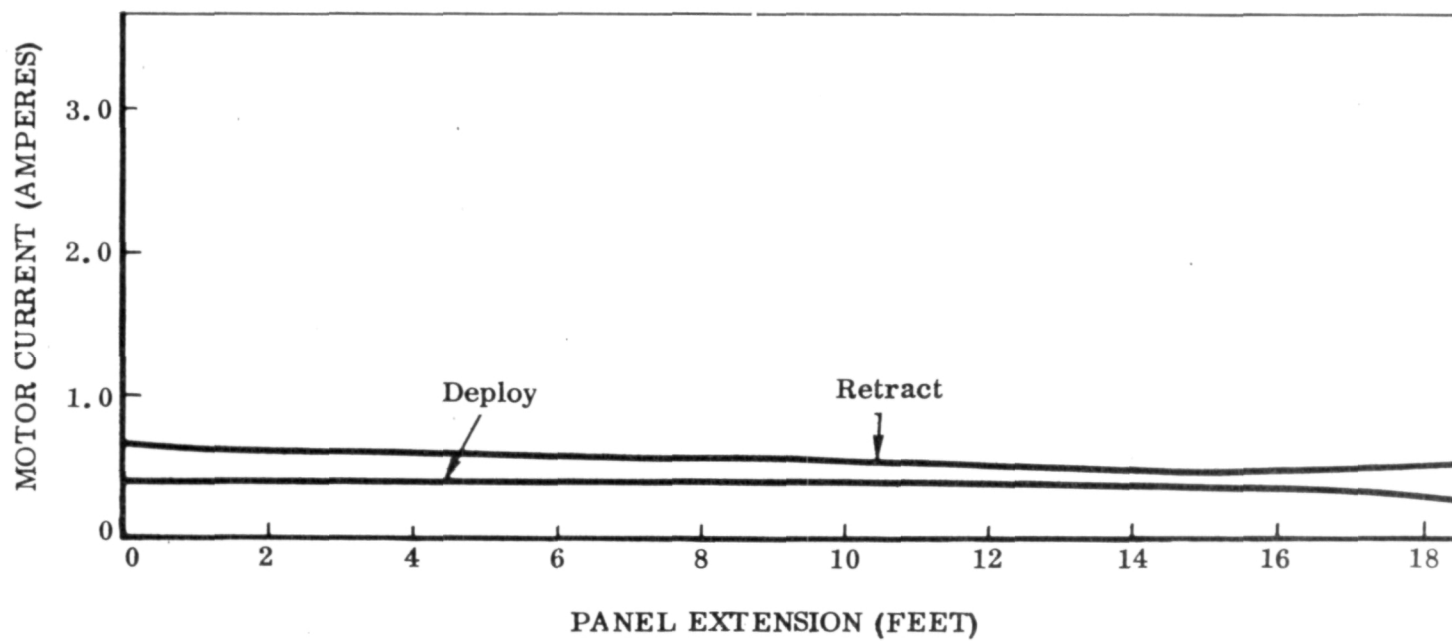


Figure 25 Deployment Motor Current Using Reworked Support Fixture

Table 1 Summary of Physical Changes to Solar Panel

EVENT	BEAMS	SUBSTRATE	EDGE CONNECTORS	SUPPORT STRUCTURE
<u>1. Deployment Cycling Tests</u>				
Assembly	No change	Width variation, small (1/2") longitudinal crazes and cracks (9) places, loose damper pad (3) places	Deformed (2) places	No change
1st Deployment	No change	Small longitudinal cracks (3) places, damaged damper pads (4) places	Deformed (1) place	No change
2nd Deployment	No change	Small longitudinal scratches (2) places, loose damper pads (6) places	No change	No change
3rd Deployment	No change	Small longitudinal cracks (4) places, damaged damper pad (1) place	No change	No change
4th Deployment	No change	Small cracks (3) places	Deformed (1) place	No change
5th Deployment	No change	Small cracks (4) places, damaged damper pad (4) places	Deformed (1) place	No change
<u>2. Environmental Reference Tests</u>				
Install Instrumentation	No change	Cut access holes in substrate, replaced loose damper pads	Deformed (2) places	No change
Sine Vibration	Cracks at intercostal	Longitudinal tears (4) places, cracks (5) places, (1) loose piece at drum, hinge pin loose (3) places, hinge torn (1) place	Loose (12) places	Motor bracket screw loose, cracks at bolt holes

Table 1 Summary of Physical Changes to Solar Panel (Continued)

EVENT	BEAMS	SUBSTRATE	EDGE CONNECTORS	SUPPORT STRUCTURE
Rework	Cracks stopdrilled	Doubler added, hinges bonded, cracks repaired	Replaced and bonded	Cracks stop drilled, screw tightened, disengaged guide cam
Random vibration	No change	No change	No change	No change
1st thermal vacuum cycle	Attach tab slot tear out (6) places	No change	Tore loose from L/H beam (16 ft length)	No change
Rework	No change	L/H side trimmed	L/H replaced with fiberglass strip	no change
2nd thermal vacuum cycle	Buckled and tore at drum attachment	Transverse tears at drum attachment	Tab edges out at R/H beam	Limit switch damaged
Rework	Removed damaged portions; replaced intercostal	Repaired trimmed both edges	Replaced both sides with fiberglass strip	Removed guide cam, repaired limit switch

4.4.2.4 Effects on Solar Cells

Prior to the first retraction, a detailed visual examination of solar cells was conducted. All modules were examined for evidence of cracked cells, delaminated filters, cracks, chips, broken interconnects and other visible defects. Elemetrical measurements were made under sunlight illumination to determine open circuit voltage, short circuit current and maximum power. A secondary standard solar cell was used to establish actual solar flux at the time of testing.

Significant results of these measurements are presented here. The incidence of cracked solar cells is representative of the effects of test events and other activities on the integrity of the solar cell modules. Measurements of I_{SC} , V_{OC} and P_{max} indicate general performance of the power system. These values have been normalized to a flux of 100 mw/square cm at 55°C.

Figure 26 shows the arrangement of solar cell modules on the four subpanels. Inspection for cracked cells was associated with each module by serial number to determine the uniformity of physical effects on the cell. The effect of the various program events on the solar cells is summarized in Table 2 and Table 3.

During the 5-cycle deployment test a total of 14 cells out of 784 or 1.8 percent were cracked.

The majority of these cells were cracked in the first two cycles. These cracks were caused by distortion of the solar panel substrate in the transition area between the storage drum and the roller support fixture.

During the test program the incidence of cracked cells was primarily associated with handling by personnel and support fixtures. Damage related to test environments is relatively small. An incident of major damage occurred during thermal vacuum testing which is described in Section 4.4.5.

In general there was no measurable difference in electrical performance of the array due to the cracked cells. The only exception to this is the

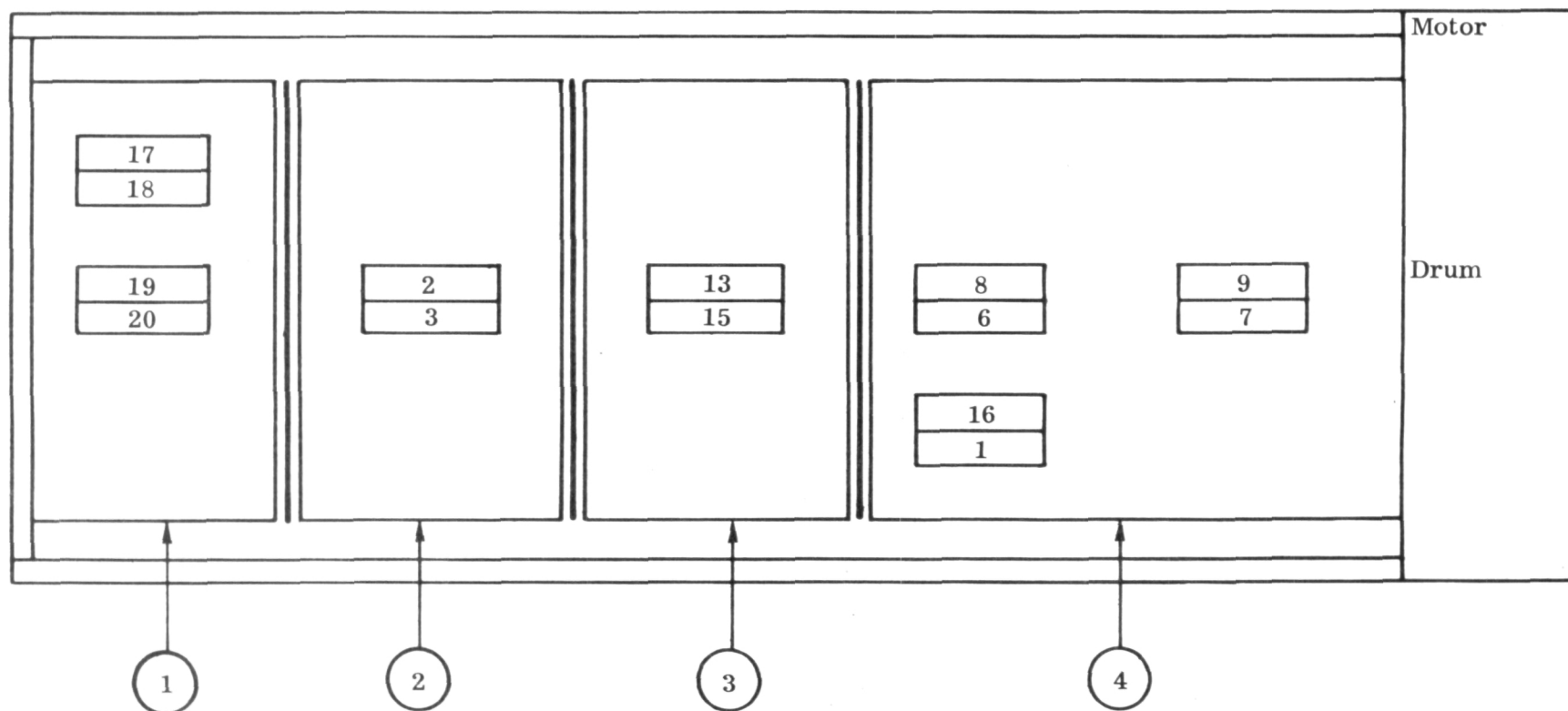


Figure 26 Location of Active Solar Cell Modules

Table 2 Summary of Solar Cells Damage

EVENT	CRACKED CELLS (PERCENT)	
Assembly		4.7*
First Deployment Cycle	0.5	
Second Deployment Cycle	0.9	
Third Deployment Cycle	0.4	
Fourth Deployment Cycle	0.0	
Fifth Deployment Cycle	0.0	
Deployment Cycling Test Total	1.8	1.8
Install Instrumentation		0.1*
Sine Vibration Test		1.0
Rework and Repair		1.0*
Random Vibration Test		0.3
First Thermal Vacuum Test		0.1
Rework		0.3*
Second Thermal Vacuum Test		0.1
Cold Deployment Damage		7.1
	TOTAL	16.5

* Denotes assembly or rework, not a test event.

Table 3 Incidence of Cracked Solar Cells

CRACKED CELLS PER MATRIX																	Total	Cracked During Test
MATRIX NO.	17	18	19	20	2	3	13	15	8	6	16	1	9	7				
EVENT																		
Assembly	4	3	1	1	1	0	1	3	8	2	4	3	2	4	37			
1st Deployment	0	0	0	0	0	0	0	0	0	0	0	0	4	0	4	4		
2nd Deployment	1	0	0	0	0	0	0	0	1	1	3	0	1	0	7	7		
3rd Deployment	0	0	0	1	0	0	0	1	0	1	0	0	0	0	3	3		
4th Deployment	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0		
5th Deployment	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0		
Install Instrumentation	0	0	0	0	0	0	0	0	0	0	0	0	1	0	1			
Sine Vibration	3	0	1	0	0	1	0	0	0	1	1	0	0	1	8	8		
Rework	2	0	0	0	0	0	0	0	2	1	1	2	0	0	8			
Random Vibration	0	1	0	0	0	0	0	0	0	0	0	1	0	0	2	2		
1st Thermal Vacuum Test	1	0	0	0	0	0	0	0	0	0	0	0	0	0	1	1		
Rework	0	0	0	0	0	0	0	0	0	0	1	1			2			
2nd Thermal Vacuum Test	0	0	0	0	0	0	0	0	0	0	0	1	0	0	1	1		
Cold Deployment Damage	0	0	0	0	0	0	0	0	0	0	0	0	33	23	56	56		
TOTAL	11	4	2	2	1	1	1	4	11	6	10	8	41	28	130	82		

Total Cells per matrix 56

Total cells (14 matrices) 784

degradation detected after thermal vacuum test damage which also involved interconnect damage, broken contacts, and severe cell damage.

4.4.2.5 Electrical Test Results

Test Setup

The electrical characteristics of the live solar cells were determined by using the roller support fixture to support the solar array assembly during sunlight deployment. Two digital voltmeters and one X-Y plotter were used to record and plot the solar cells open and short circuit current and voltage. A standard solar cell supplied by JPL was used to monitor the sun solar intensity. The test arrangement is shown in Figure 27.

Test Procedure

Measurements of the electrical outputs of the live solar cell group on each modular panel as shown in Figure 26 were made after each environment to which the panel was subjected. The electrical tests were designed to show any degradation of the solar cells during the various tests.

In each case the cells were tested with the panel in the extended position with the cells exposed to sunlight. Table 4 shows the tests after which electrical measurements were conducted. In each case the electrical properties were measured in accordance to the following procedure:

- a. The panel was extended fully and the solar cells outputs were connected to the digital voltmeters and x-y plotter.
- b. The standard cell was placed on the heat sink to maintain a constant temperature (28°C) and measurements were made to determine the solar flux striking the cell, and the temperature of the standard cell.
- c. Current versus voltage plots were made on each of the solar cell groups on the panel. Measurements were made of the open circuit voltage and short circuit current, as well as the power output at 12 v. The maximum

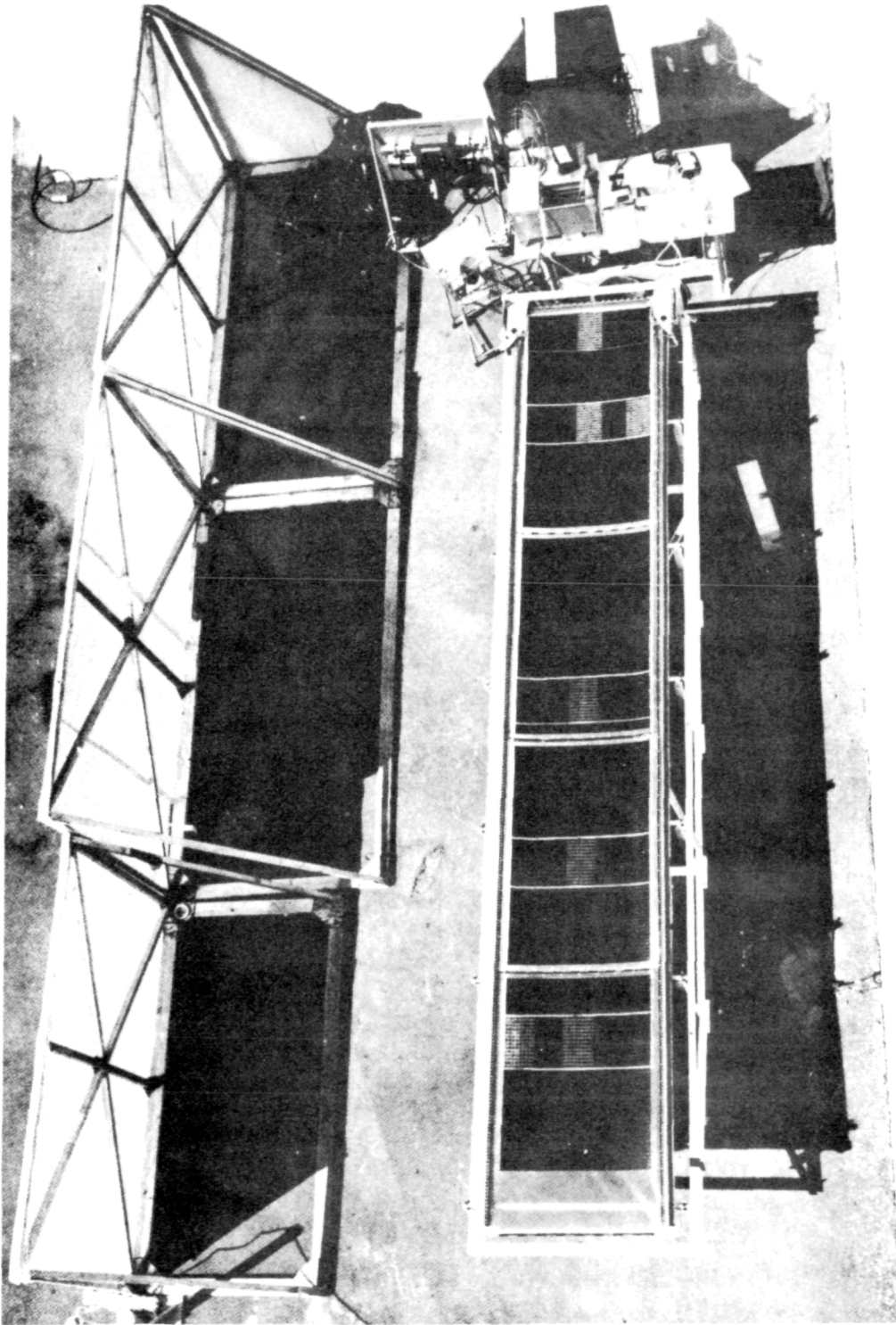


Figure 27 Solar Cell Electrical Test Arrangement

Table 4 Summary of Electrical Tests

TEST NUMBER	DATE	TEST
1	10-6-67	*After final assembly
2	10-6-67	After first retraction and deployment
3	10-18-67	After second retraction and deployment
4	10-18-67	After third retraction and deployment
5	10-20-67	After fourth retraction and deployment
6	10-20-67	After fifty retraction and deployment
7	10-31-67	Prior to sine vibration test
8	2-6-68	*Prior to random vibration test
9	2-16-68	*After random vibration test
10	4-11-68	*After thermal vacuum test

*Measured with center support rollers on fixture

power output for all four cell groups was calculated from the I-V curves. All of the electrical measurements for each test case were then normalized to reflect the same conditions. These conditions were a solar intensity of 100 mm/cm^2 at a temperature of 55°C . The results are shown in Figures 28, 29, and 30.

In tests one through eight, the panel was supported in such a way as to let the panel sag in the center. This sag affected the solar flux seen by test cell groups one and four, since some of the solar cells in these groups were at the edge of the panel.

Calculations were made to determine the difference in the angles of the solar cells with respect to the sun and corrections in flux were made accordingly.

In tests eight through ten the holding fixture for the panel was modified with center support rollers to hold the panel flat.

It is apparent in Figures 28 and 29 that a failure occurred during the thermal vacuum test which caused a definite voltage and power dropout after the thermal vacuum test. The damage to the solar cells was obvious from visual observation, and the electrical test confirms the damage to solar cell group 4. Other more detailed information on the output of the solar cells such as slight variation in output after various tests is not available from the data due to experimental error in the operation of the electrical test. However, it can be concluded from the data that solar cell performance was generally unaffected by the environments tested.

4.4.3 Deployed Dynamics Characteristics Test

Modal vibration tests were conducted on the deployed panel (with the solar cell electrical mass simulation of $.3 \text{ lbs/ft}^2$, see Section 9.3) for the following reasons:

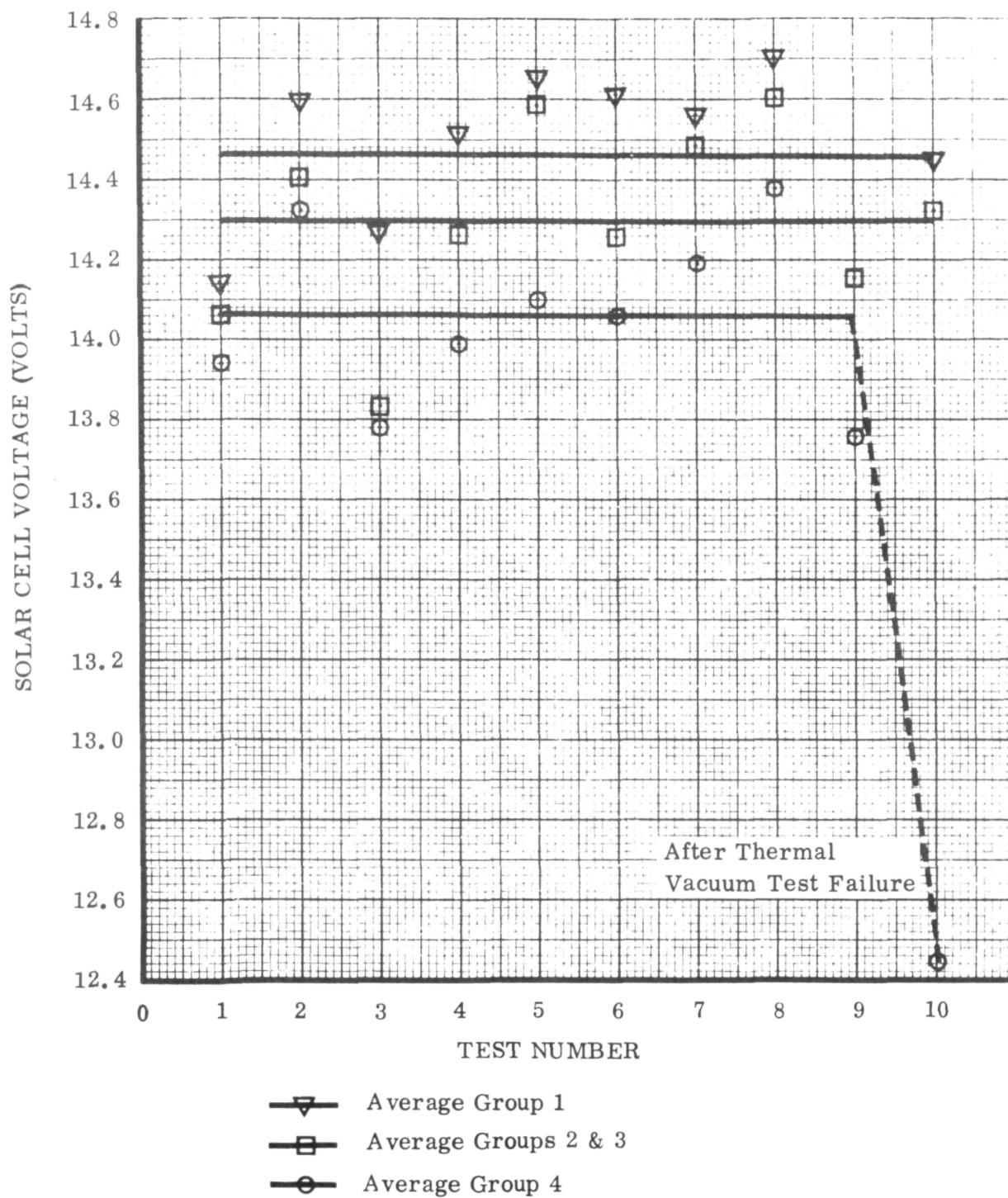


Figure 28 Solar Cell Voltage Vs. Test Cycle

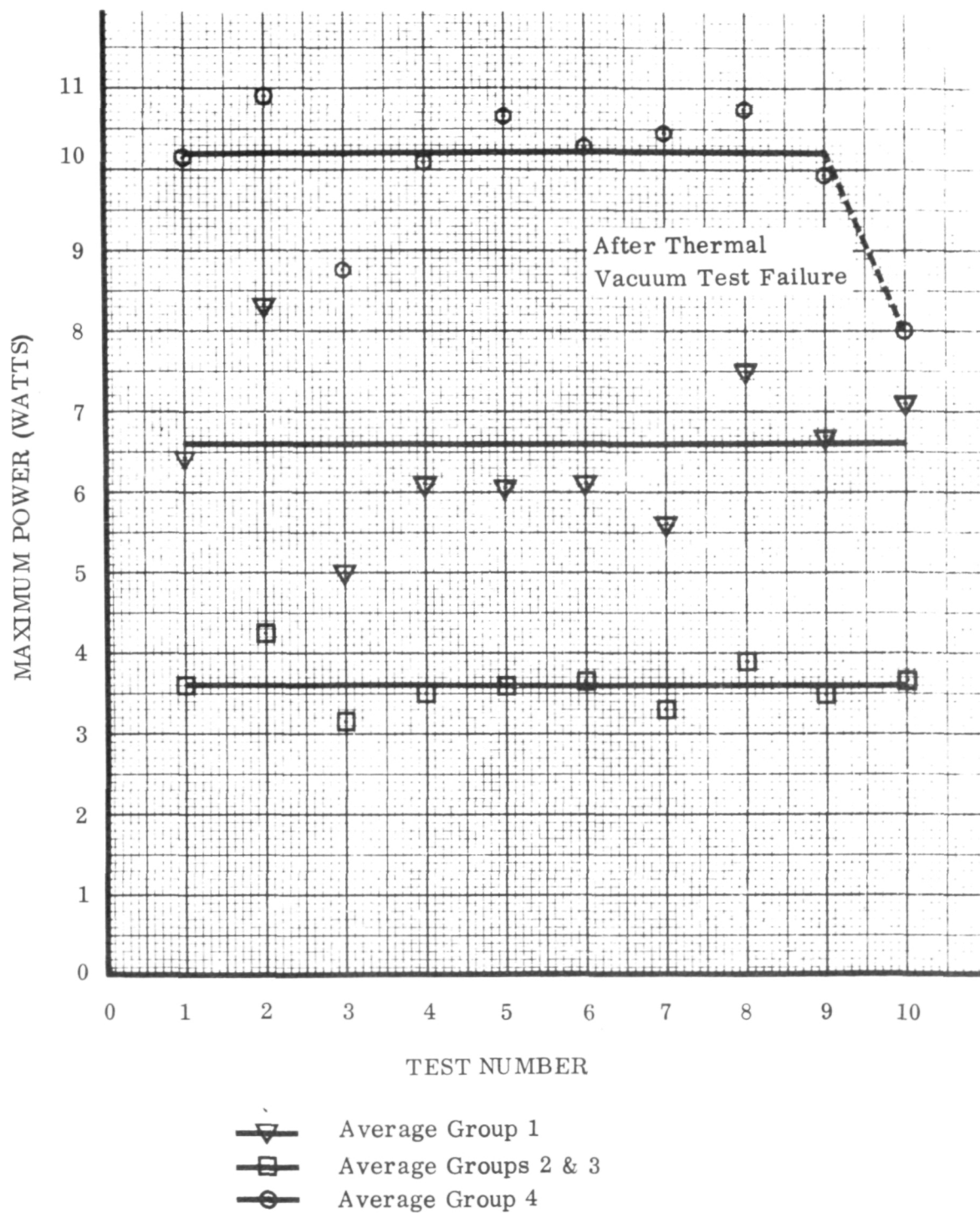


Figure 29 Solar Cell Power Vs. Test Cycle

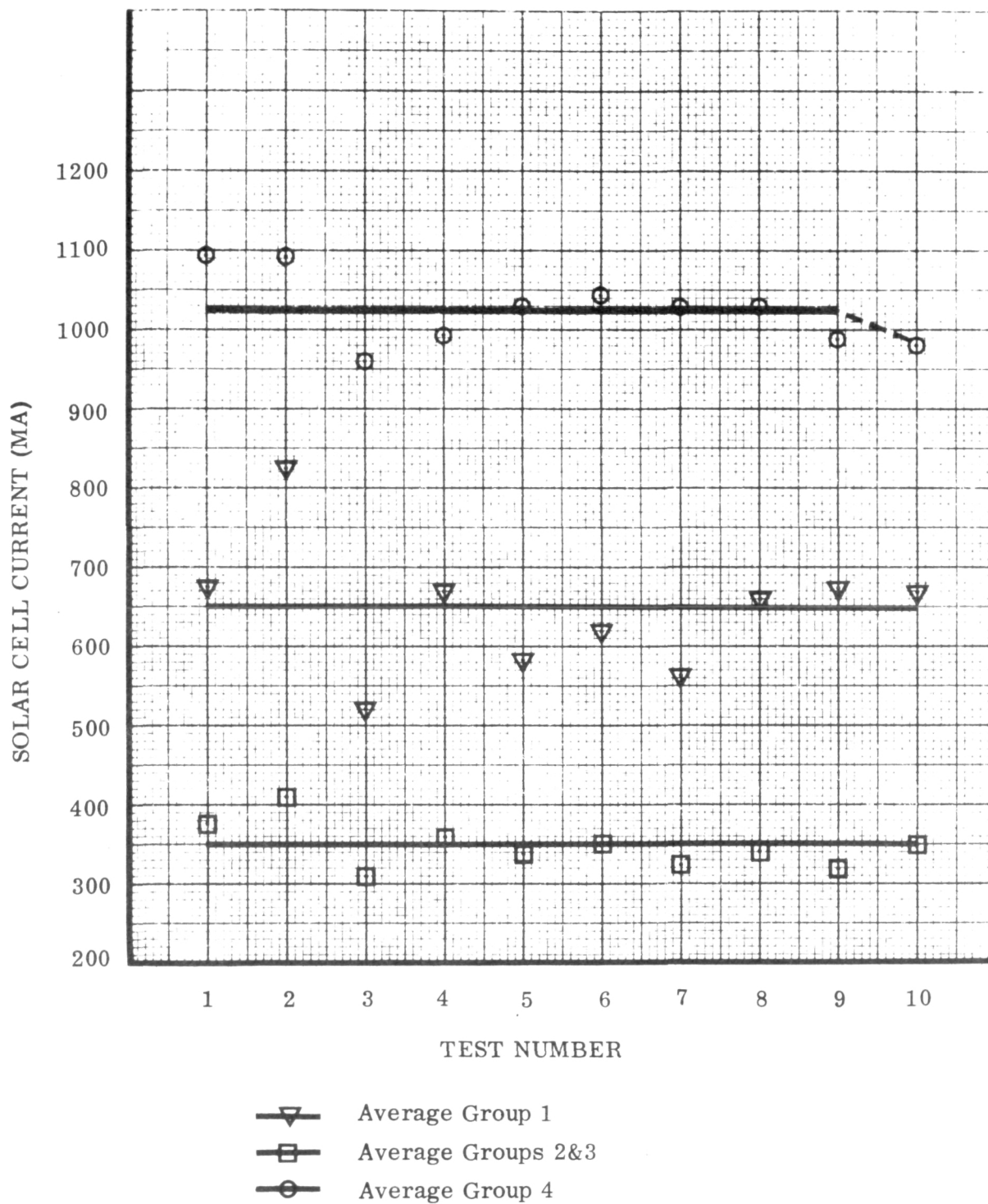


Figure 30 Solar Cell Current Vs. Test Cycle

- a. To show that the first cantilever natural bending frequency at cruise maneuver temperature is between 0.5 and 5 Hz, as specified in the JPL design criteria (Section 3.2).
- b. To compare test results with the calculated panel cantilever and torsion modes given in Section 6.2, Ryan Report No. 20869-2. (Reference 2)

and,

- c. To determine the predominant panel membrane mode.

It was not within the scope of this test program to determine the effect of various forcing vibration levels on natural frequencies. Therefore the panel was excited at a level only large enough in magnitude to sufficiently define, visually, the characteristic mode shape.

Testing was conducted in air at room temperature conditions. Therefore, some conservatism (by a small amount) exists in the test result frequencies considering the presence of air. The test results should be reduced by 6% to correct for the effect of cruise maneuver temperature.

Test Setup

The 50-square foot solar array assembly was attached to the test fixture (used for stowed panel vibration testing) and mounted to a structural "I" beam 30-feet from the ground to allow full vertical deployment and retraction. (See Figures 31 and 32). A 50-pound MB vibration exciter was attached to the panel tip intercostal. Accelerometers were mounted on each tip of the deployment beams and at the center of the tip intercostal. A movie camera was mounted looking at the edge of the panel with a lined grid background to visually monitor the displacement of the beam during vibration excursion. A function generator analyzer was used to check phase relationship, drive the exciter, and read vibration frequency.

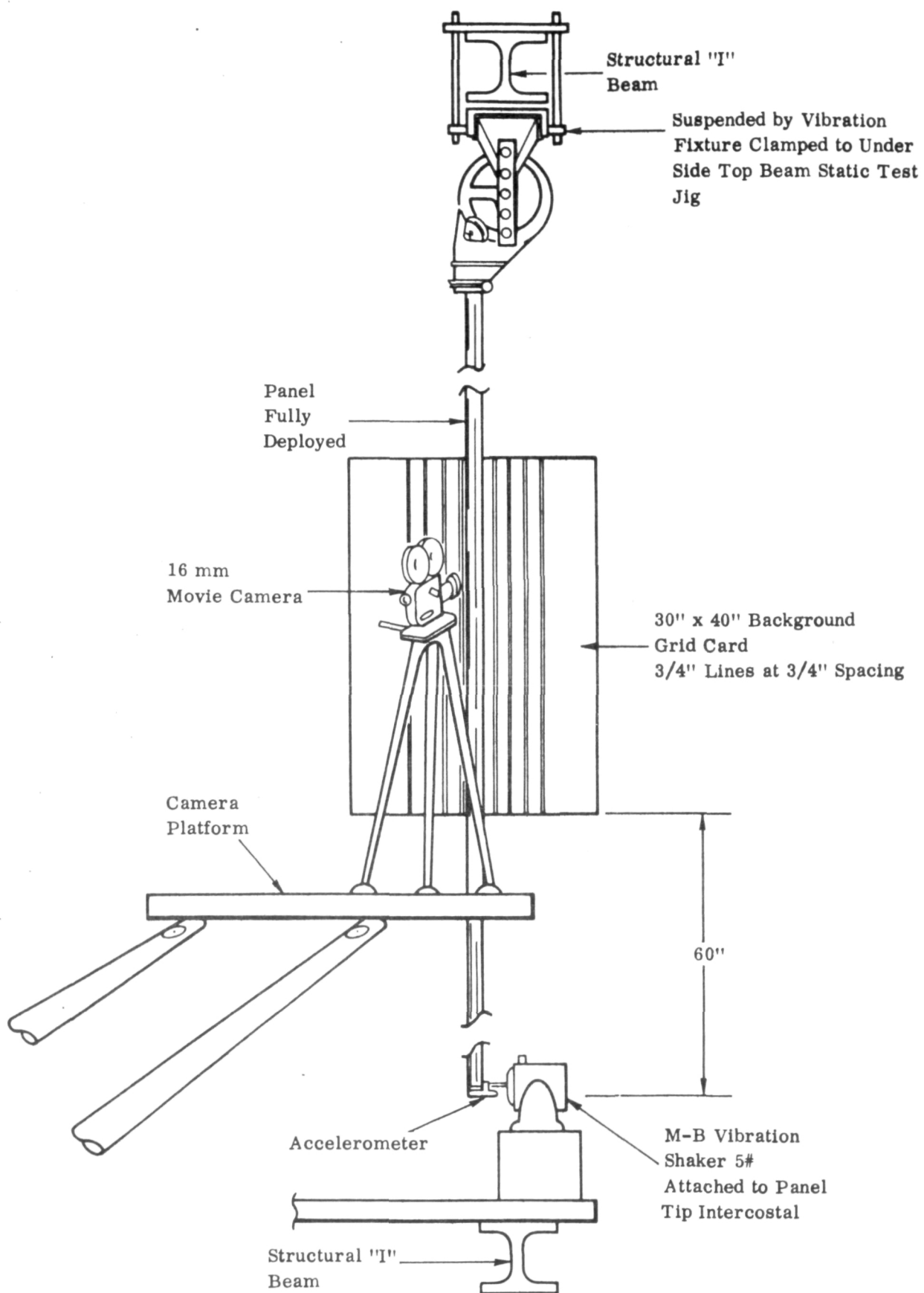


Figure 31 Deployed Dynamic Test Setup

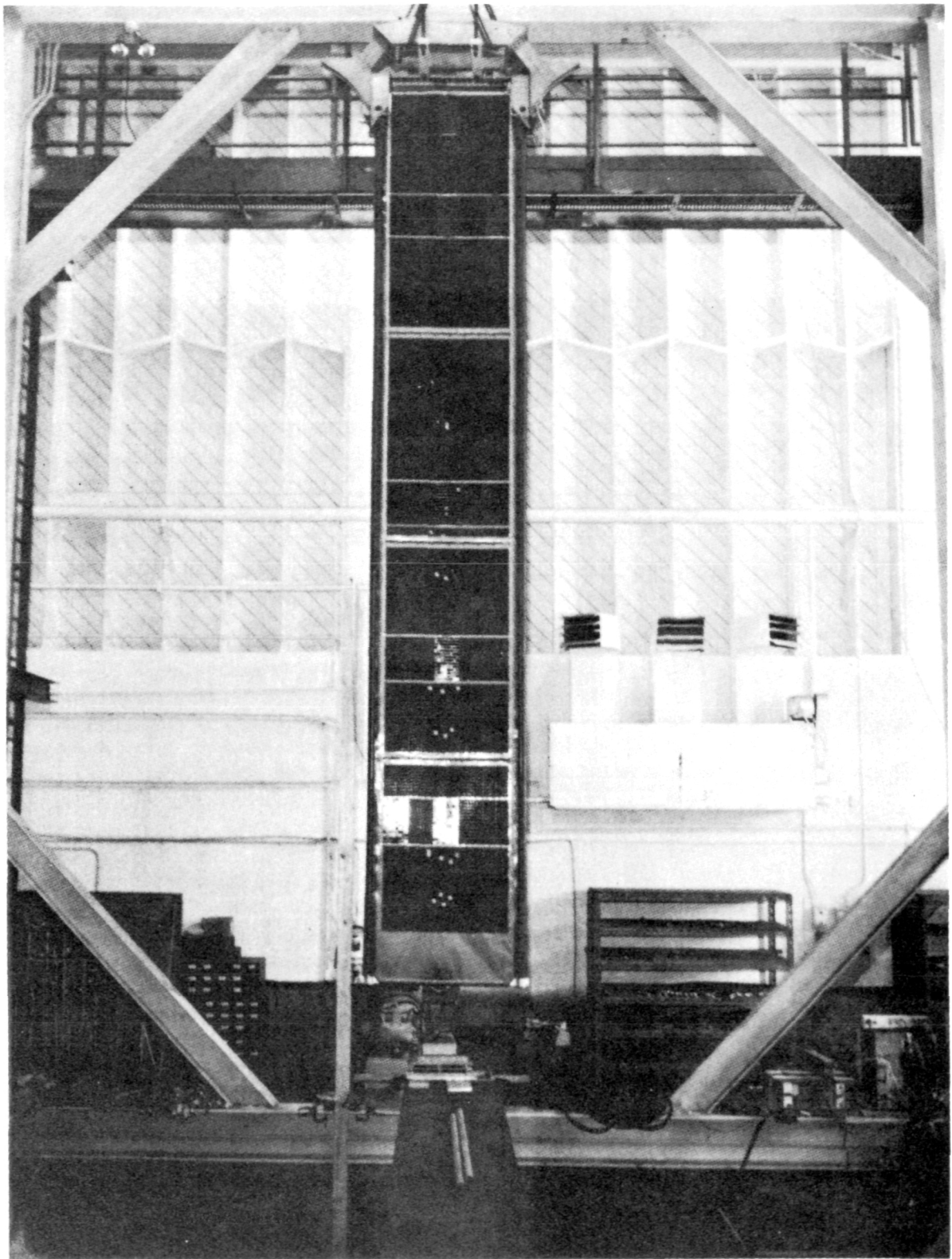


Figure 32 Panel Setup in High Bay Building For Deployed Dynamic Test

Test Procedure

Utilizing the above test setup, the first panel (membrane) mode was determined by attaching the 50-pound MB mechanical exciter to the center of the tip intercostal. Phase relationships of the input accelerometer at the exciter and the accelerometer mounted on the beam were compared (using the function generator analyzer) to determine a panel resonance condition.

The torsional panel mode was determined by attaching the exciter to the tip intercostal about 1 inch off the panel center line and monitoring phase relationship as above.

The first cantilever bending mode was determined (with exciter disconnected from panel) by initially deflecting the panel by hand about 1 foot, then monitoring the periods of oscillations by a stop watch.

Test Results

Test results given in this section are compared with analytical results given in reference 2.

a. First Cantilevered Bending Mode

Test result was 0.43 Hz at about 1 foot initial deflection (1/2 amplitude). By analysis (see page 37, Ryan Report No. 20869-3), the frequency should have been 0.65 Hz at room temperature (0.61 Hz at in space temperature). Since (a) the load distribution of the test panel (0.0956 pounds/inch) was nearly equivalent to that used in analysis (0.10015 pounds/inch, (b) analytical frequencies are based on the actual test beam cross sections, we can point directly to the looseness of the beam in the guide as the contributing factor to the low test frequency (see Figure 33).

b. First Panel (Membrane) Mode

There was no analytical work performed on this mode for comparison with

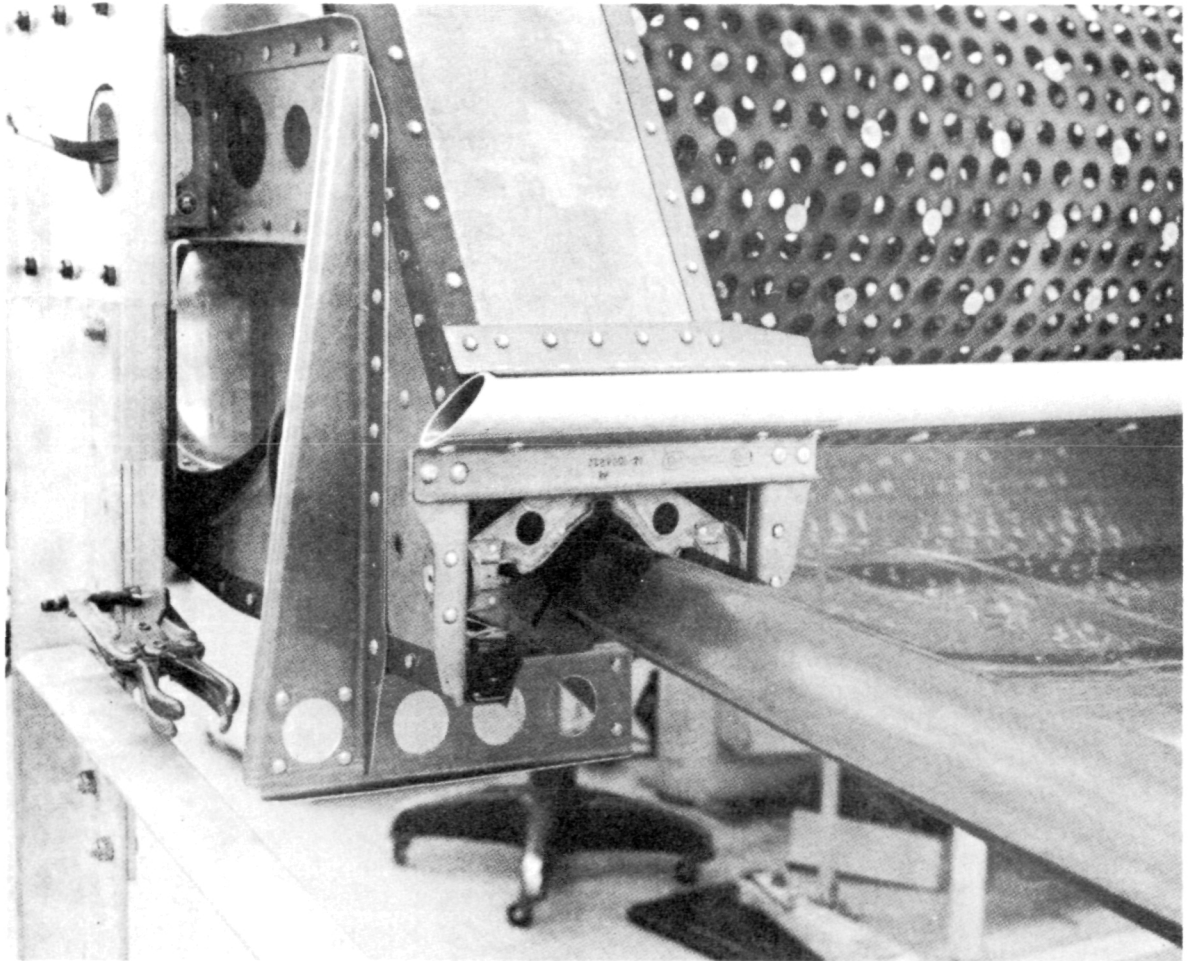


Figure 33 Looseness of Guide Fit to Deployment Beam

the the test frequency of 0.87 Hz, determined at 0.12g (peak) excitation.

c. First Torsion Mode

Test result was 13 Hz at 0.59g (peak) excitation. By analysis (see Section 6.2, Ryan Report No. 20869-2), the frequency should have been about 11.4 Hz at room temperature including corrections for the actual deployment beam cross section. The conservative analytical frequency is due mainly to neglect of the effect of the tip intercostal on the panel torsion stiffness.

Conclusions

The need for closer fabrication fit between guides and deployment beams (fully deployed position) has been emphasized in this section when considering the effect the loose fit had on the deployed panel cantilever bending frequency. A guide which is formed to the actual beam root cross section with beam fully deployed should be a minimum requirement in fabrication.

4.4.4 Launch Vibration

Preliminary tests were first conducted to lend confidence that the stowed panel could successfully withstand the simulated launch vibration tests. Of major concern was (1) the degree of compactness of the stowed panel wraps relative to assumptions made in the detailed analysis, and (2) the effect of the actual wrap compactness on the stowed panel dynamics and structural integrity of the solar cells. Discussion of the preliminary tests performed is presented in the Appendix, Section 9.2.

4.4.4.1 Sinusoidal Launch Vibration Test

This test was conducted to simulate sinusoidal vibration excitation to the stowed panel, which may be induced during spacecraft launch. The test was conducted in a 1g sea level environment which shall be considered to yield results similar to that which would occur during launch in a rarified atmosphere and within 3% of results due to possible temperature extremes during launch.

Test Procedure

The test procedure presented as follows is taken from the Ryan Test Plan, Report No. 20865-1 (Reference 4). The retracted array with dummy chips and wired solar cell modules (see Figure 34) representing a 0.3 pound/square foot electrical installation (see Section 9.3) shall be mounted to the shake fixture designed with sufficient rigidity to prevent modal coupling with the test specimen and instrumented with accelerometers (see Figures 35 and 36). Strobe lights will be used for modal monitoring.

The solar cell modules will be checked electrically prior to retracting the substrate. The array assembly will be scanned in the three major axes at a low (0.5g) excitation level, for fundamental modes of vibration. After completion of low level vibration the array will be subjected to the following environment in the most critical axis or axes determined during low level modal scan.

Sine Vibration

1.6g rms	2 - 20 Hz at 1 min/octave
4.0g rms	20 - 200 Hz at 1 min/octave

NOTE: The low level excitation frequency is limited to 5 Hz by the vibration equipment; however, anything below this is not expected to be critical.

The specimen shall be removed from the vibration exciter and the solar cell modules shall be checked visually and electrically after each axis excitation at test levels.

Test Results

Results of the modal scan and design level tests are presented as follows:

Modal Scan

Sinusoidal vibration of the stowed panel was conducted with the panel mounted

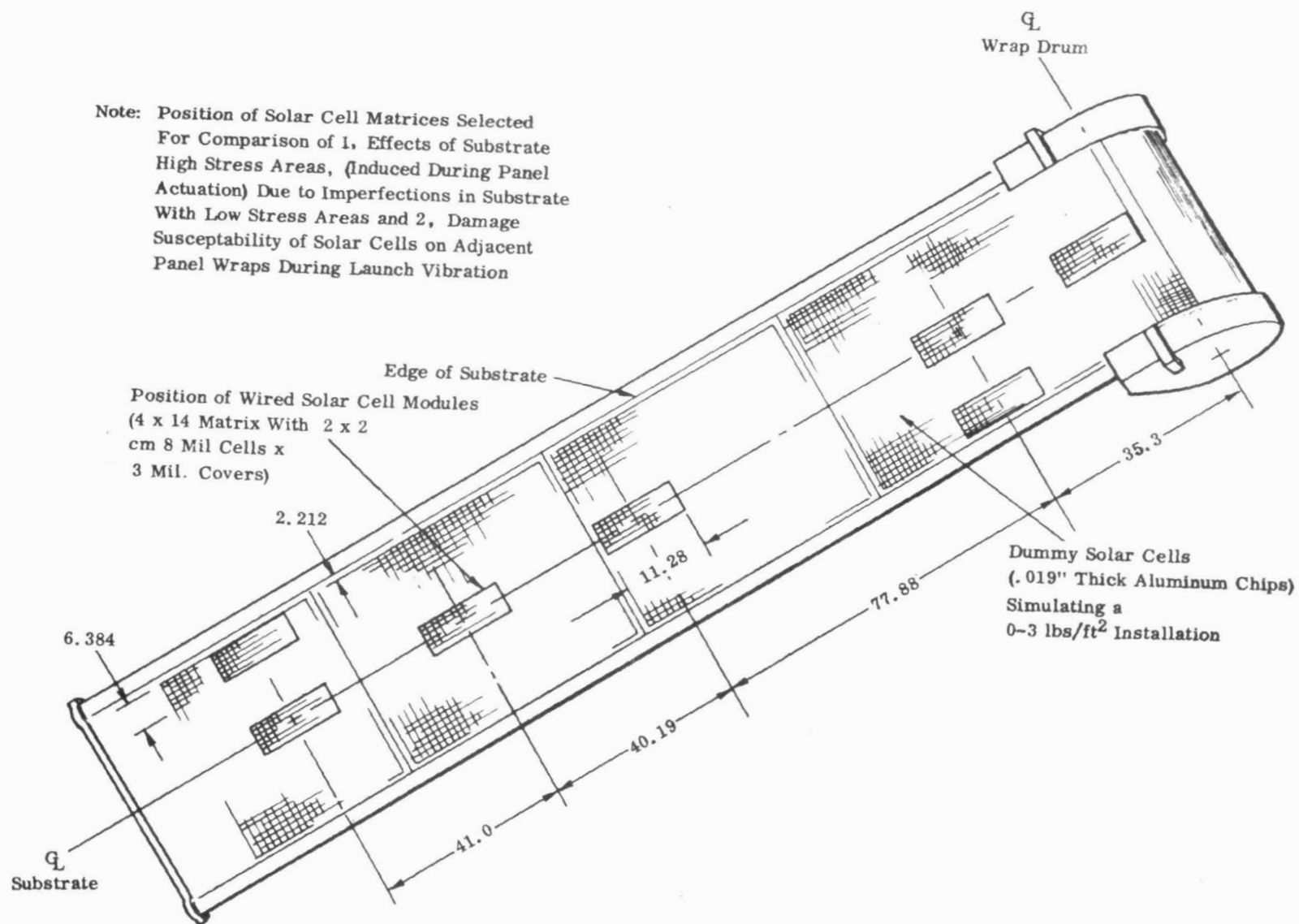


Figure 34 Solar Cell Module Layout

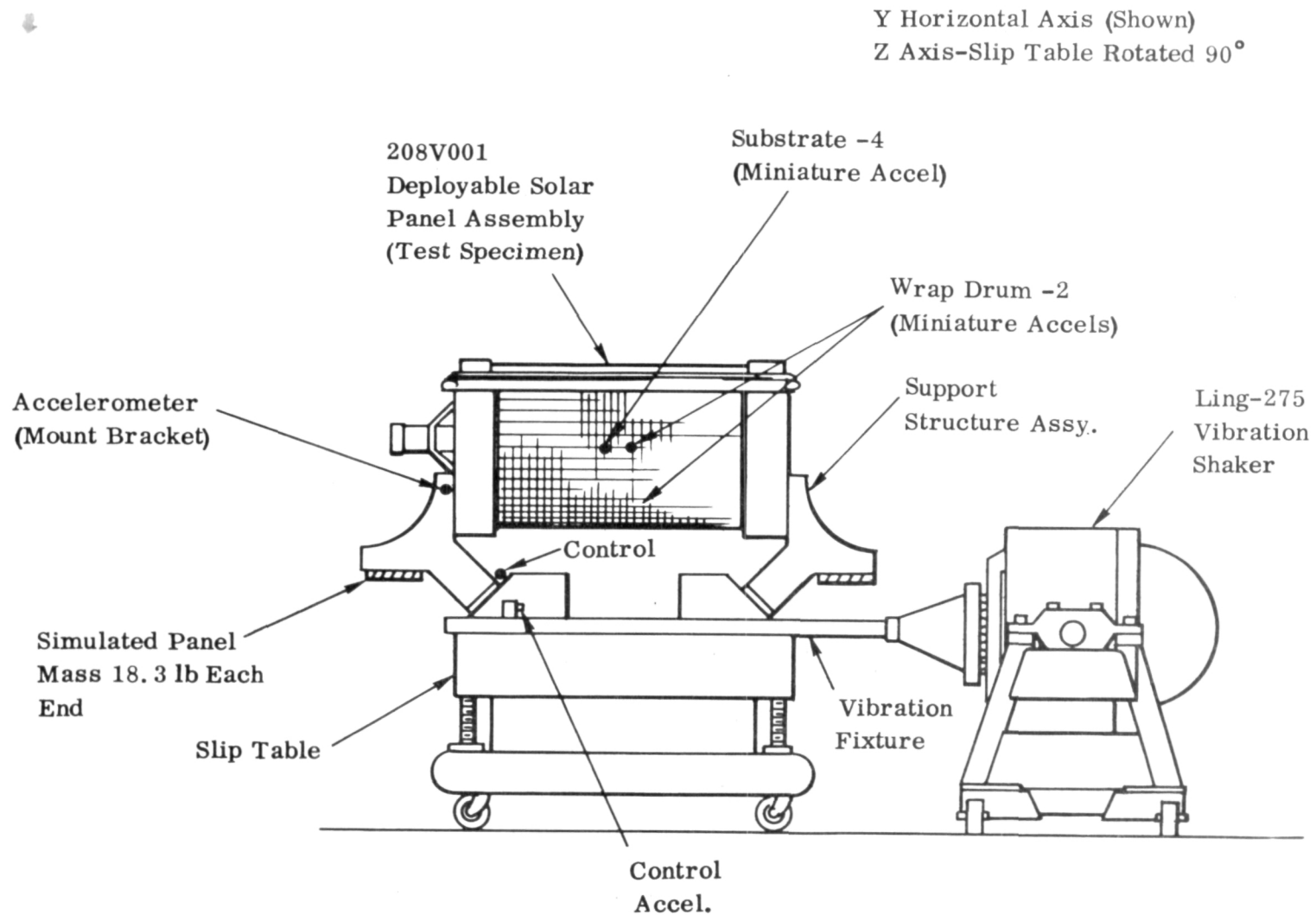


Figure 35 Vibration Test Setup, Y and Z Axis

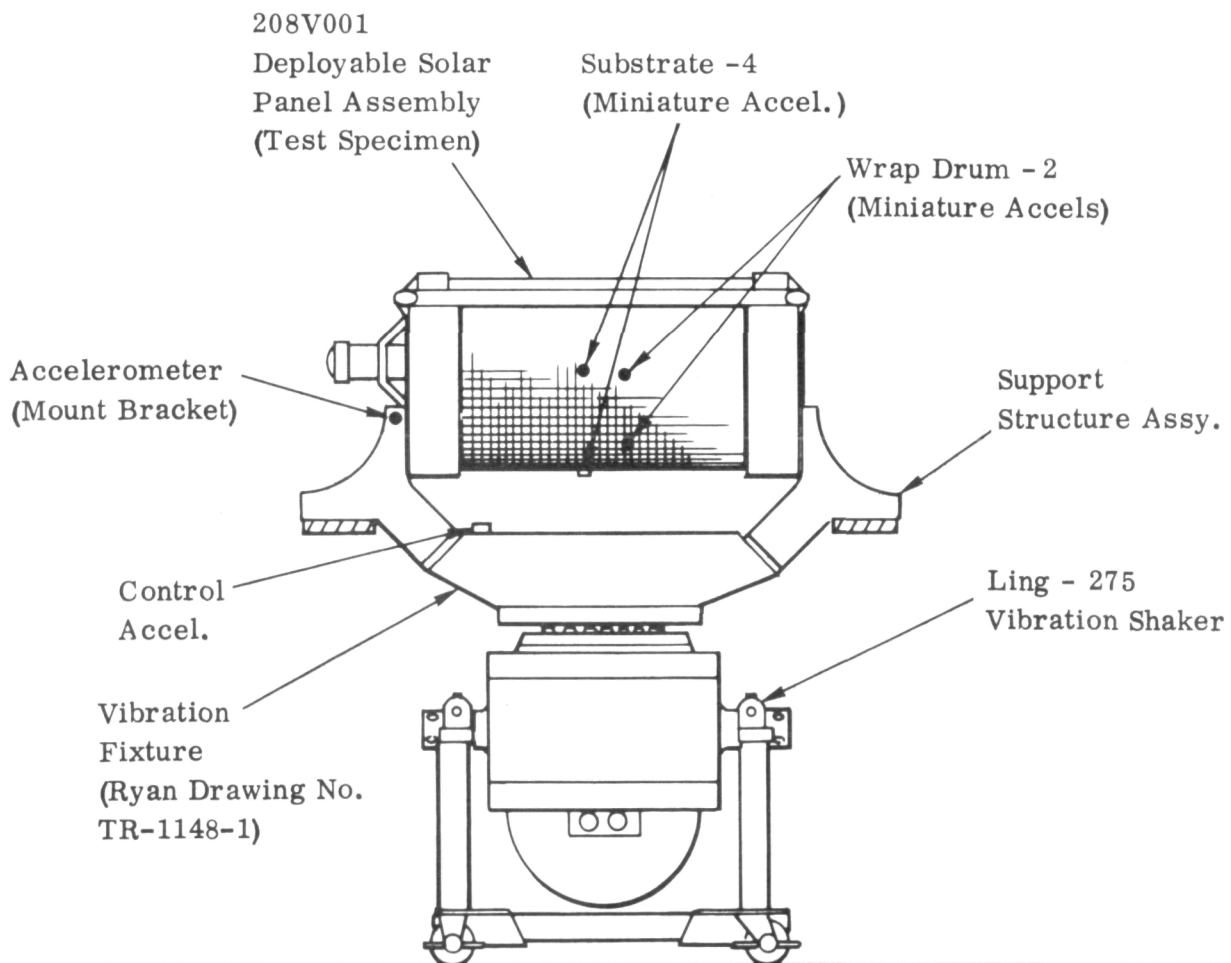
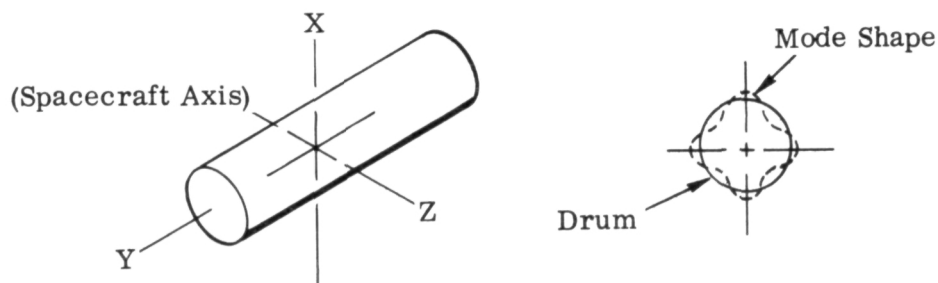


Figure 36 Vibration Test Setup, X Axis

to a rigid shake fixture. Mass Load balances at each end of the panel simulated effects from the adjacent panels of a four panel system. First, natural frequencies and dynamic transmissibilities were found at a low level (0.5g 0-Pk) sinusoidal excitation, swept logarithmically from 5 Hz to 2,000 Hz. Miniature accelerometers (Endevco, Model #2222) attached with Eastman 910 adhesive were used to monitor response accelerations at the spacecraft mounts, drum honeycomb end plates, drum mount bearings, wrap drum at center and panel wraps at center of drum, which were recorded on tape and then transferred to X-Y plot graph paper for review and determination of dynamic transmissibilities. Wiring access to the accelerometers was made by cutting away a small portion of the substrate on the wraps above the accelerometer (see Figure 37). Figures 38, 39 and 40, show the test setup. The panel was excited in each of three major axes, x, y, and z to determine the axis of shake most severe to the solar cells and panel structure. Results indicated the z axis to be more critical to the solar cells with the outer wrap vibrating in the lower more critical mode shape as shown.



The predominant outer wrap vibration resonance characteristics were 43 Hz for this mode with a dynamic transmissibility to the wrap of 20:1 and a dynamic deflection of 0.0108" for the x axis shake. The y axis was more critical to the drum end plate structure, with a critical natural frequency of 30 Hz and a dynamic transmissibility of 5:1; a negligible amount of motion through the drum mount bearings was indicated by a dynamic transmissibility of only 1.02 at 30 Hz. Other critical natural frequencies of concern in the respective axis were:

X axis: Drum, bending mode, 320 Hz.
Spacecraft mount, 800 Hz.

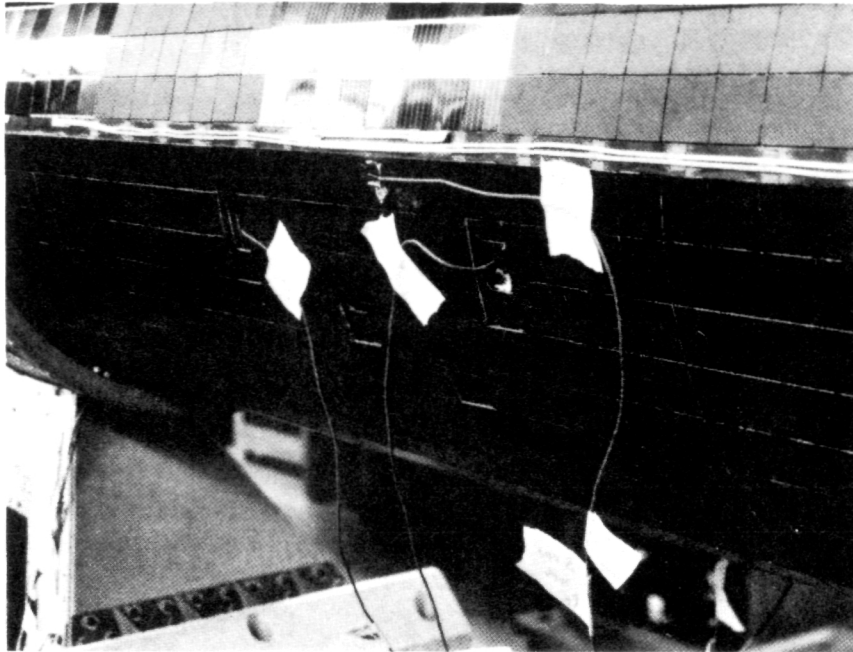


Figure 37 Accelerometer Access Hole in Stowed Panel Wraps

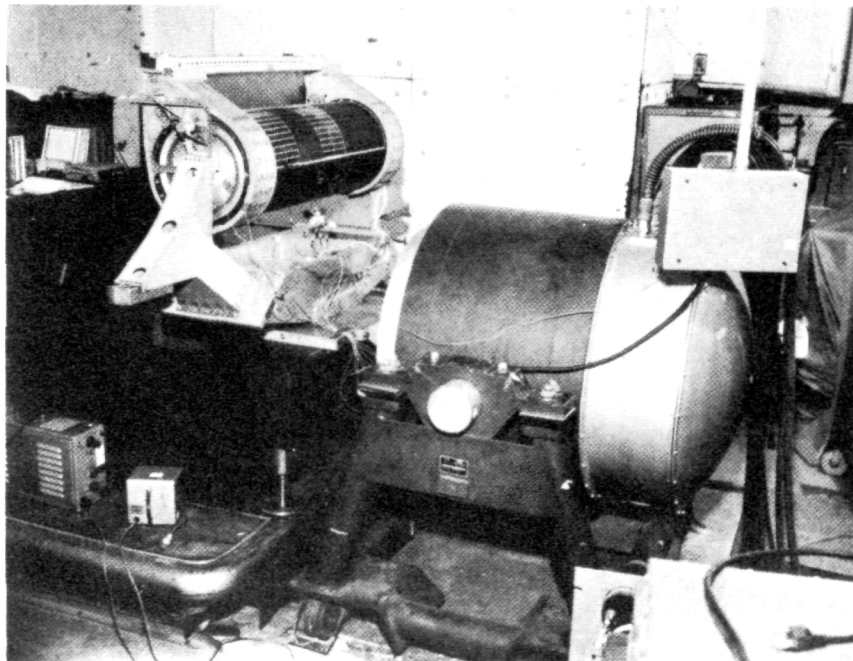


Figure 38 Test Setup on Slip Table for Spacecraft Axis Vibration

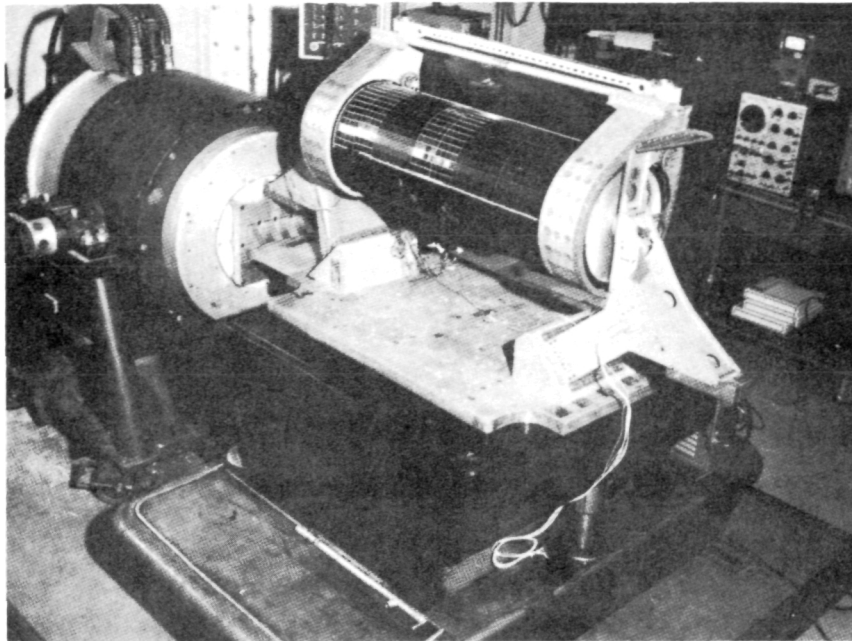


Figure 39 Test Setup on Slip Table for Drum Axis Vibration

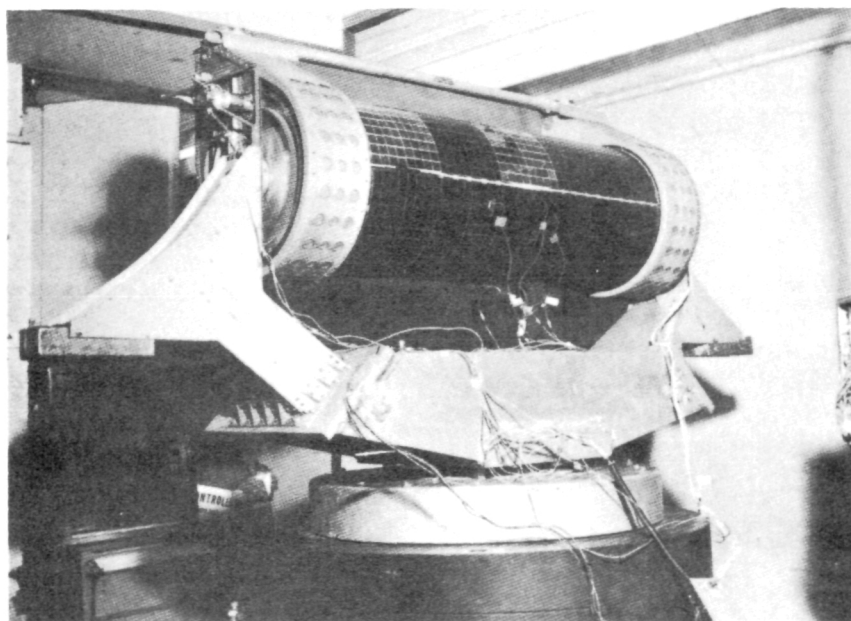


Figure 40 Test Setup on Shaker for "X" Axis Vibration

Y axis: Spacecraft mount, 1,600 Hz.

Z axis: Spacecraft mount, 30 Hz.

The predominant stowed panel frequency of 43 Hz was below the analytical frequency of 285 Hz (see Section 6.1, Ryan Report No. 20869-2). This large difference is because, at best, analysis could only assume an ideal condition that the panel would wrap compactly; as is shown in Section 9.2.1, the panel wraps were loose with up to 0.2 inch of gap between adjacent wraps.

The predominant spacecraft mount mode of 30 Hz can be shown to correlate well with analysis. Section 5.9, Ryan Report No. 20869-2 shows a static mount deflection of 0.014 inch ultimate due to a 12g launch acceleration acting on the stowed panel mass. This may be converted to 0.0094 inch deflection, δ , in a 1g field ($\delta = \frac{0.014}{12 \times 1.25}$) for use in calculation of the theoretical fundamental mount frequency. Using the equation for fundamental frequency of a mass-concentrated at tip beam,

$$\begin{aligned} f_n &= 3.13 \left(\frac{1}{\delta}\right)^{1/2} \\ &= 3.13 \left(\frac{1}{0.0094}\right)^{1/2} = 3.13 \times 10.3 = \underline{\underline{32.2}} \text{ Hz} \end{aligned}$$

Design Level Test

The panel was vibrated in each of the Y and Z axes. The test deviated from original plan by not deploying the panel after each test at design levels for examination of solar cells. Instead, only the outer wrap solar cell modules were inspected during the test, since it was concluded that these modules would be the first to result in damage with the outer wrap experiencing the greater dynamic excursions. Effects of various levels of sinusoidal excitation on dynamic transmissibility up to the design levels,

1.6g (rms) 5-20 Hz

4.0g (rms) 20-200 Hz

were found by testing in increments, then stopping the test and checking the structure and solar cells visually while stowed on the fixture.

In the Y axis, monitor of the drum end plate gives the following dynamic transmissibilities at input levels through that respective item:

<u>g(0-Pk) in.</u>	<u>Q</u>	<u>fn, Hz</u>	<u>Dynamic Deflection, in. (1/2 Amplitude)</u>
0.5	5	30	0.028
2.25	2.74	25	0.10
4.0	1.74	18	0.213
5.66	1.77	23	0.188

In the Z axis, monitor of the outer wrap gives the following transmissibilities between the fixture control accelerometer and the outer wrap (integrated transmissibility, or system transmissibility):

<u>g(0-Pk) in.</u>	<u>Q</u>	<u>fn, Hz</u>	<u>Dynamic Deflection, in. (1/2 Amplitude)</u>
0.5	20	43	0.055
2.25	15.6	35	0.275
4.0	15	30	0.65
5.66	14	30	0.90

NOTE: Transmissibility through the drum mount bearing was small, 1.07 at 5.66g excitation.

In the Z axis, the following transmissibilities between the fixture control and the drum bearings (transmissibility in spacecraft mount):

<u>g (0-Pk) in.</u>	<u>Q</u>	<u>fn, Hz</u>	<u>Dynamic Deflection, in. (1/2 Amplitude)</u>
0.5	13.6	30	0.075
2.25	4.5	30	0.11
4.0	3.5	30	0.15
5.66	3.2	27	0.25

Also, in the Z axis, the following transmissibilities between the drum mount bearing and the outer wrap (essentially, transmissibility in outer panel wrap):

<u>g (0-Pk) in.</u>	<u>Q</u>	<u>Fn, Hz</u>	<u>Dynamic Deflection, in. (1/2 Amplitude) of outer wrap relative to drum</u>
4.0	11.5	30	0.50
5.66	10.8	30	0.65

NOTE: It was difficult to distinguish the resonant points on the X-Y plot for this more critical wrap mode, since near coupling occurred between this mode and the spacecraft mount mode.

Solar cells on the outer wrap and all visible structure were inspected visually after each vibration sweep. The unit was removed from the vibration fixture, mounted to the roller support fixture and redeployed after the test for visual and for electrical check in sunlight.

We may conclude that the vibration had little effect on solar cells, with only 6 cracked cells but no electrical degradation. The following structural damages were found after redeploying the panel:

1. Fatigue cracks in beams at attachment to tip intercostal (Figure 41).
2. Broken and loose substrate-beam attach clips on outboard substrate not supporting cells (outer 8 clips) and cracks in fiberglass hinges on top intercostal (Figure 42, 1 and 2 respectively).
3. Fatigue cracks in spacecraft mount attach angles in side plate bend radii at ends (Figures 43 and 44, respectively).
4. Substrate fatigue cracks, up to 13-1/4 inches long, at panel longitudinal edge adjacent to beam attach clips (Figure 45).
5. Substrate module hinge pins displaced laterally (Figure 46).
6. Fatigue crack in internal shear web of one spacecraft mount where flange is not riveted to mount, thereby creating a discontinuity in shear flow path (Figure 47).

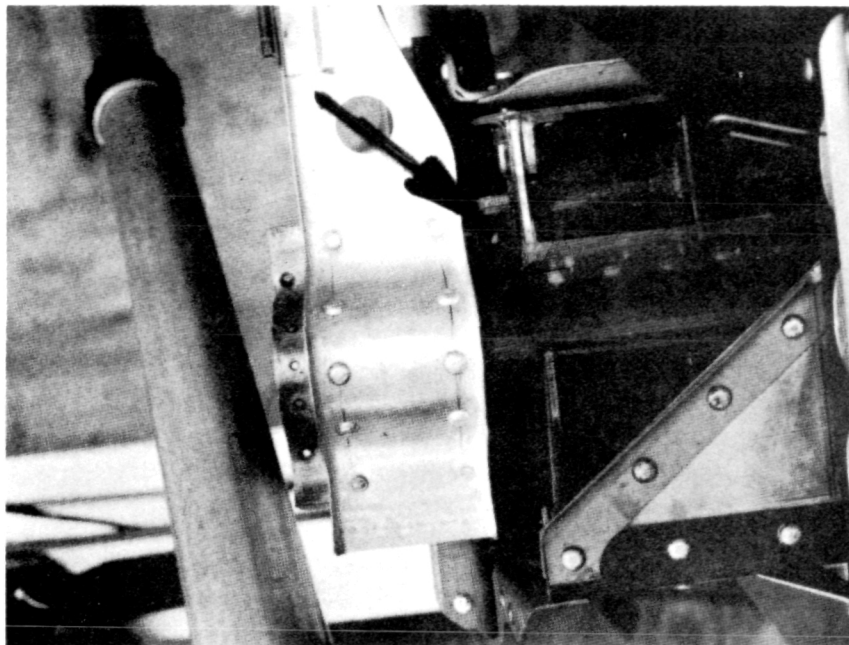


Figure 41 Fatigue Cracks in Beams at Attachment to Tip Intercostal

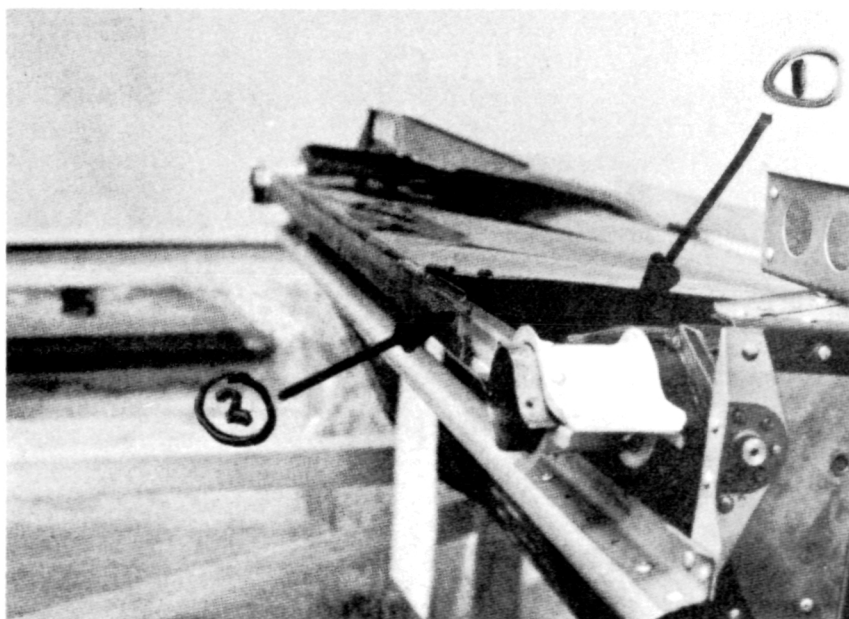


Figure 42 Damage in Outboard Substrate Attachment Areas

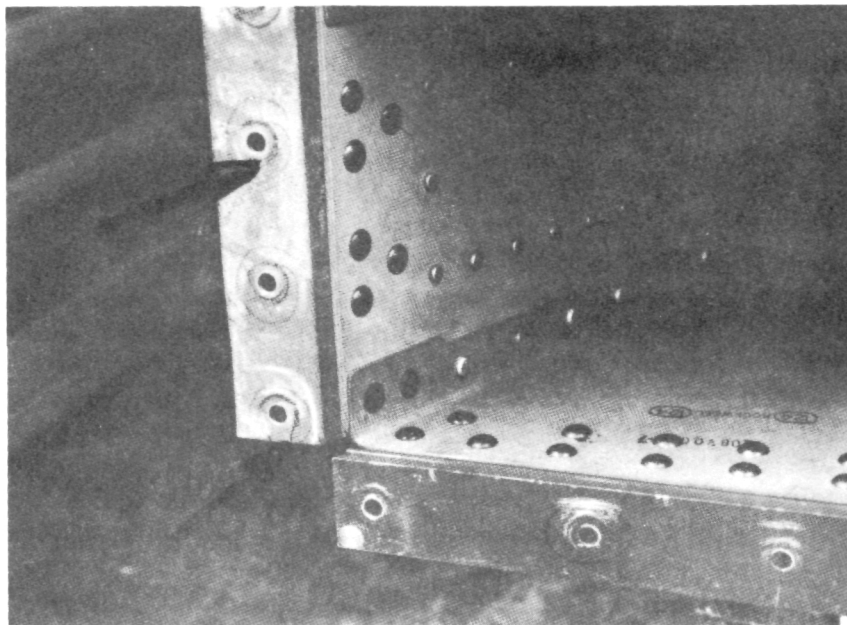


Figure 43 Fatigue Cracks in Spacecraft Mount Attach Angles

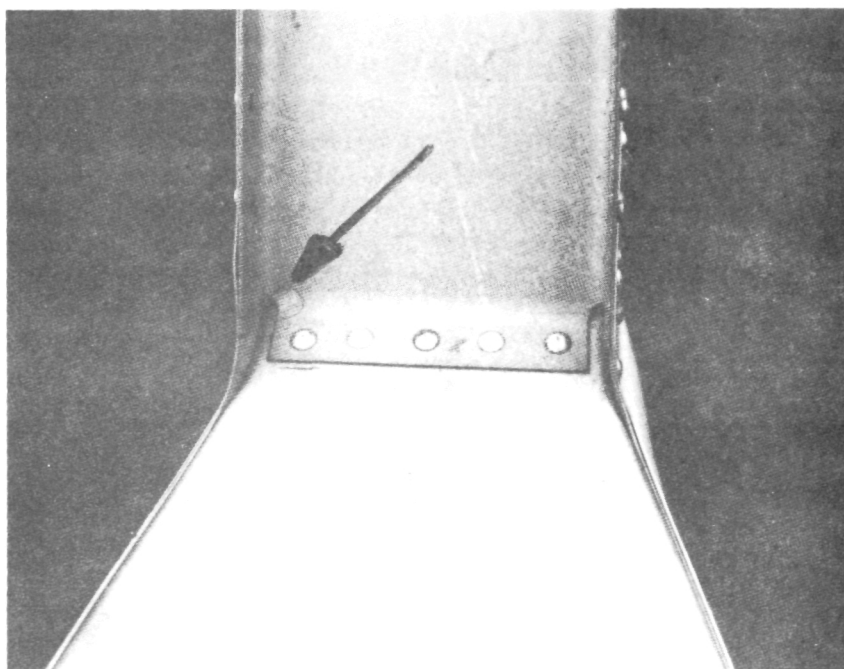


Figure 44 Fatigue Cracks in Spacecraft Mount Side Plate Bend Radii

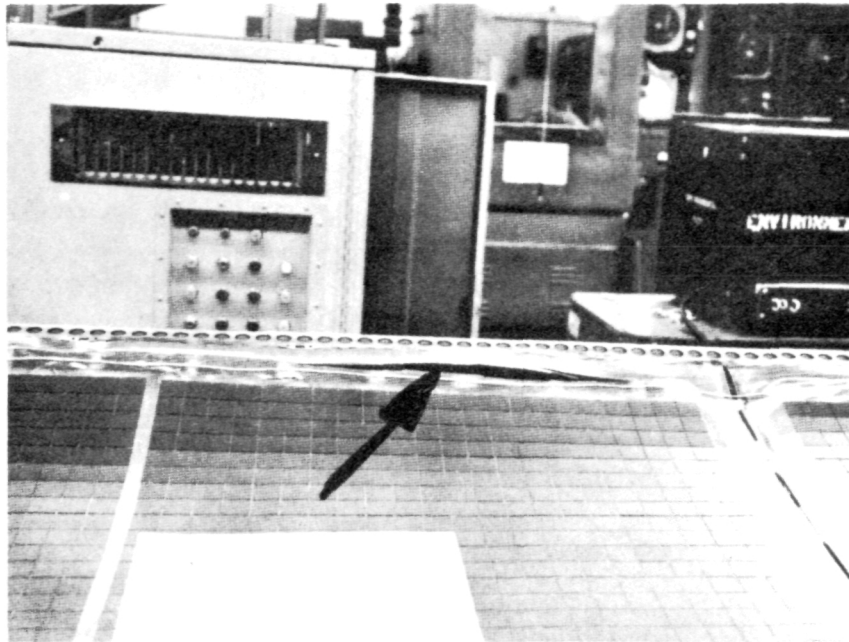


Figure 45 Fatigue Cracks in Substrate at Panel Edge

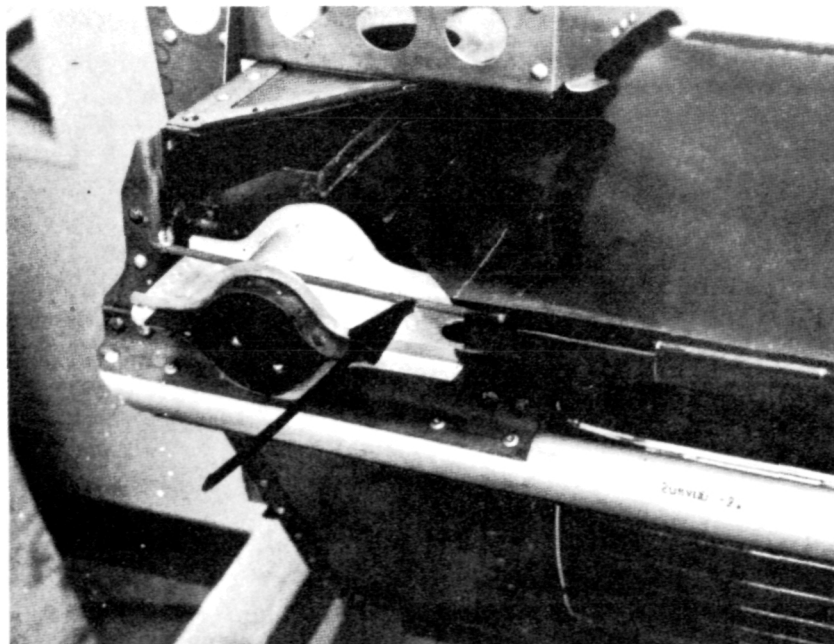


Figure 46 Substrate Module Hinge Pins Displaced Laterally

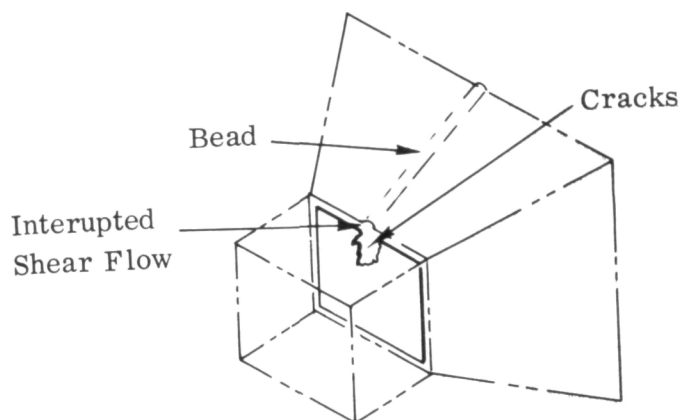


Figure 47 Fatigue Crack in Spacecraft Mount Internal Shear Web

Dispositions

The following dispositions were recommended for repairs (of the above damages) prior to further testing. Included are suggestions for design refinement.

1. Stop drill for further testing. Restrain beam at tip intercostal in same shape as at guide as design refinement.
2. Use bonded attach clips on outer 8 clips each beam for further testing and replace fiberglass at hinge. Consider removal of outer substrate area and add leader strips and transverse snubber tube at tangent point on drum as design refinement.
3. Stop drill cracks in attach angles for further testing and use larger washers. Add end stiffening rib on each attach angle for design refinement, increase thickness of angles, and use large area washer, cut-back bond radii in side plate for design refinement to reduce stress concentration.
4. Repair using lap strip of silastic 140 impregnated fiberglass for further testing. Cut substrate modules to fit more accurately in the future, thereby eliminating excess substrate width which induced buckling and fatigue cracking.
5. Restrain hinge pins in hinges using epoxy at ends of pins for further testing.
6. Stop drill for further testing. Form flange in bulkhead to match bead in mount and rivet to mount to complete shear path. For design refinement.

4.4.4.2 Random Vibration

Introduction

The purpose of the random vibration test is to show the response levels and the compatibility of the stowed panel to a random input sinusoidal vibration excitation. This random excitation is induced during launch. The spectrum envelope is given as $0.2g^2/\text{cps}$ for 180 seconds as per JPL Phase IV Test Plan and Procedures (Ref: 4).

The most critical response axis was found to be the spacecraft thrust axis mounting position of the stowed solar panel. The response level to the random loading was much less than the loading due to discrete sinusoidal loading performed in Section 4.4.4.1.

However, the question of the time factor of 180 seconds at the random excitation as a fatigue effect is justification for conducting this test.

The response of the configuration to a discrete sinusoidal input excitation is given in Section 4.4.4.1.

The transmissibility under $4.0g$ (rms) sinusoidal excitation was 10.8 and the maximum outer panel wrap response was $56 g$ (rms) at the stowed panel fundamental resonant frequency of 30 Hz.

The estimated response level to a random excitation is calculated as follows, using a response equation for a single degree of freedom system:

$$\ddot{Y} = \sqrt{(\pi/2) f_N S_R Q_{TR}} \text{ (RMS)} \quad \text{Ref: (4)}$$

$$f_N = \text{natural frequency (Hz)}$$

$$S_R = \text{excitation power spectral density } .1g^2/\text{Hz and } .2g^2/\text{Hz}$$

Q_{TR} = resonant transmissibility (Use 10.8, some as discrete sinusoidal transmissibility)

$$\dot{Y}_{0.1} = [\pi(30)(.1)(10.8)]^{1/2} = 7.1 \text{ g (RMS) @ } 0.2 \text{ g}^2/\text{Hz}$$

$$\dot{Y}_{0.2} = [\pi(30)(.2)(10.8)]^{1/2} = 10.1 \text{ g (RMS) @ } 0.1 \text{ g}^2/\text{Hz}$$

Test Procedure

The procedure follows the general specification (Ref. 4). The test system schematic is shown in Figure 48. The random signal was generated from a magnetic tape, electronically amplified to give the rms g force necessary to the array. The array was mounted on a shaker table, as shown in Figure 49. The test procedure as given in Ref. (4) is extracted as follows:

"The retracted array, with dummy chips and wired modules shall be mounted on the vibration fixture and subjected to the following environment (1.0 g sea level ambient) in the most critical axis/axis

0.1 g^2/Hz	200 - 2,000 Hz for 180 seconds
0.2 g^2/Hz	200 - 2,000 Hz for 180 seconds

The accelerometers will be located in the same position as for the sine test. The modules will be checked visually and electrically before and after the 0.1 g^2/Hz and the 0.2 g^2/Hz test."

The test deviated from the above original plan, by the fact that the solar cells were not checked between the 0.1 g^2/Hz run and the 0.2 g^2/Hz run. The cells were checked only after both test runs were completed due to the insignificant response deflection at the 0.1 g^2/Hz excitation level as compared to the discrete sine wave response.

Only the outer panel wrap was monitored for response acceleration, for comparison with the input control acceleration on the fixture, because of the larger deflections found there during the sine test.

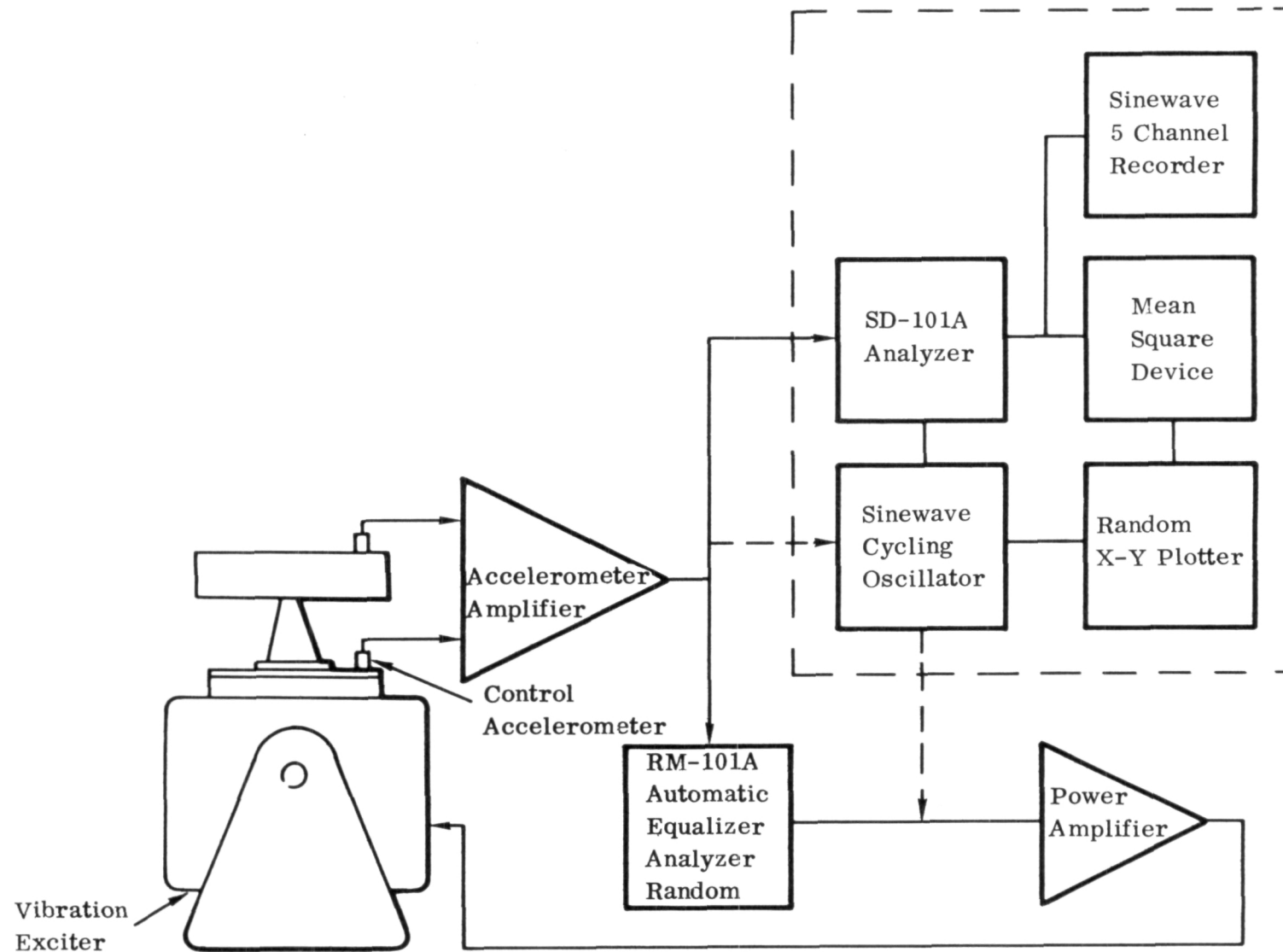


Figure 48 Block Diagram of Random Vibration Control and Monitoring Equipment

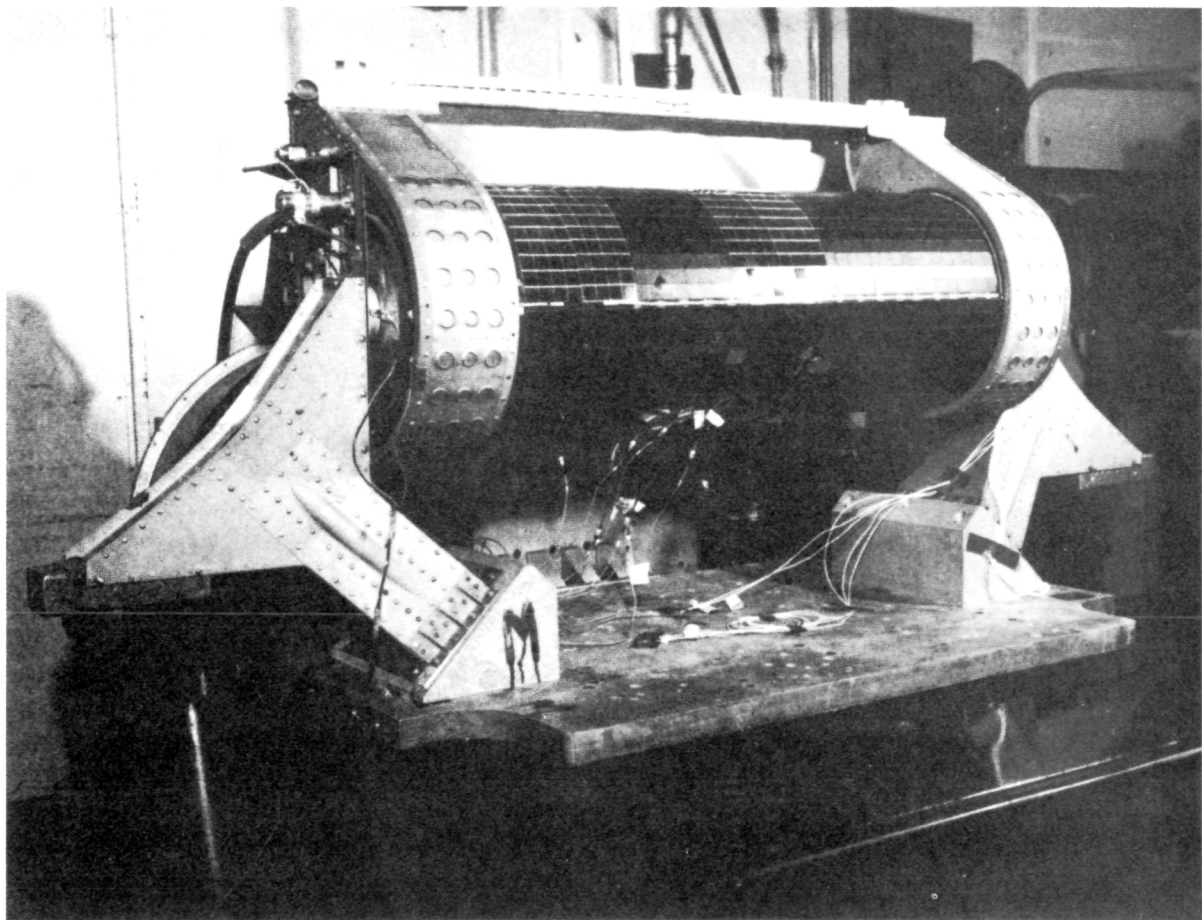


Figure 49 Test Setup on Slip Table for Spacecraft Axis Vibration Test

Test Results

The response accelerations versus frequencies are shown on an x - y plot in Figures 51 and 53 at $0.1 \text{ g}^2/\text{cps}$ and $0.2 \text{ g}^2/\text{cps}$, respectively. Figures 50 and 52 show the input excitation. The predominant energy response was near the lower end of the input frequency spectrum. The integrated energy response was:

$$\begin{aligned} 5.7 \text{ g rms at } 0.1 \text{ g}^2/\text{cps} \\ 7.1 \text{ g rms at } 0.2 \text{ g}^2/\text{cps} \end{aligned}$$

It is to be noted that the response level as measured is less than the estimated response level. The primary reason for this was the presumption that the steady state sinusoidal transmissibility is the same as for that of a random excitation force. Using the two response levels above and solving for the resonant random transmissibility (Q).

$$Q = 2 \ddot{Y}^2 / \pi f_N S_R$$

$$Q_{0.1} = 2(5.7)^2 / \pi 30(0.1) = 6.89 @ .1 \text{ g}^2/\text{Hz}$$

$$Q_{0.2} = 2(7.1)^2 / \pi 30(0.2) = 5.35 @ 0.2 \text{ g}^2/\text{Hz}$$

The above difference in transmissibility for the two excitation levels is consistent with other test results which show that the transmissibility can vary at different excitation levels.

Conclusions

Since there was no degradation of the stowed solar panel, including the solar cells, attachments, electrical connections and attaching hardware, the tests indicated acceptability of the system. There was no indication of fatigue of the two tests of 180 seconds duration each. The results did leave some doubt as to how to select a transmissibility value for a given system before the empirical random transmissibility factor has been ascertained.

2-12-1968

50 Square Foot Solar Array Z-Axis [1.6g/cps input]

Specimen: _____ Transducer: S/N 5613 TR 1148
S/N _____ Control ☒ G RMS 15.0
P/N _____ Response ☐ Sampling Equal #2

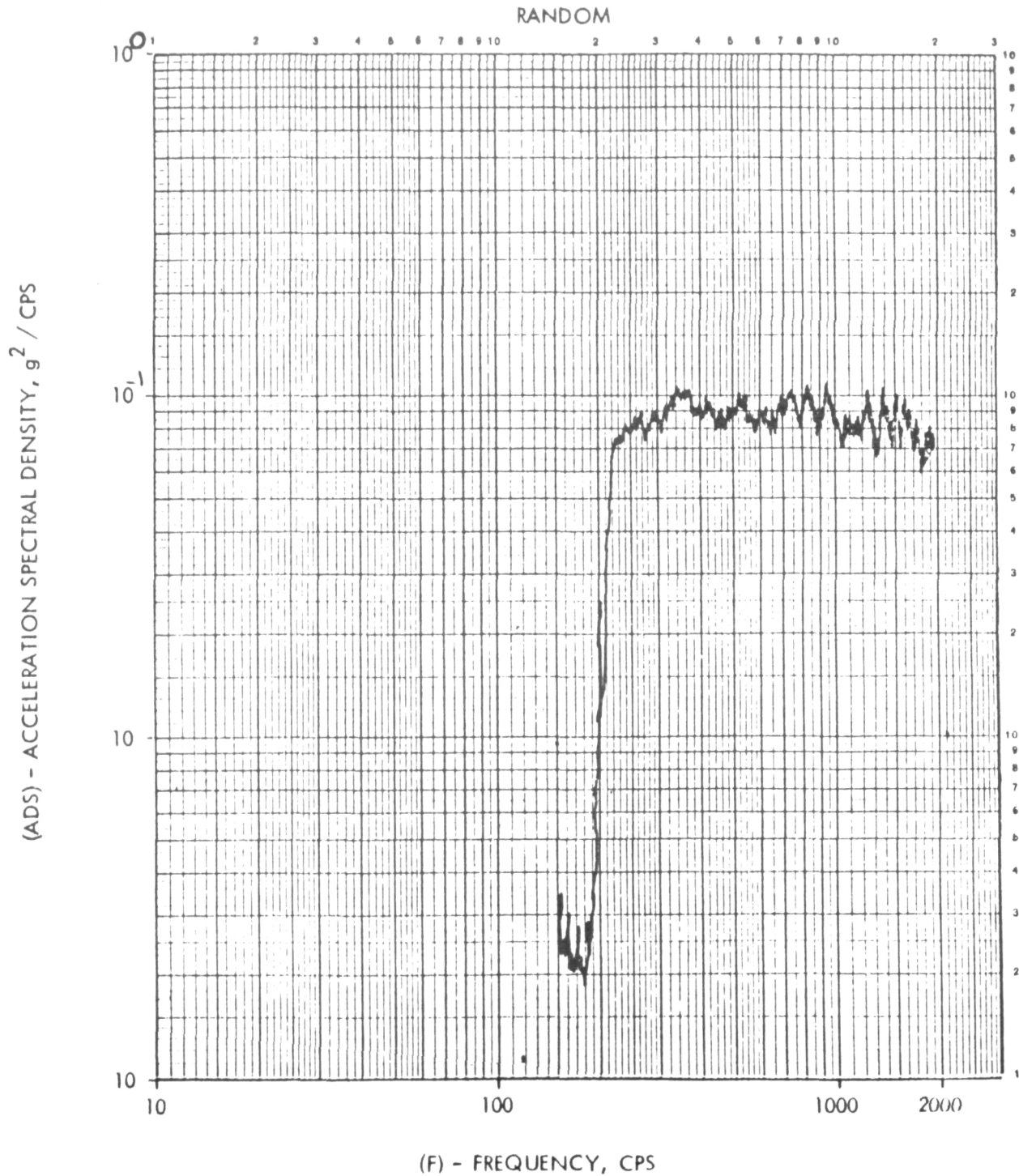


Figure 50 Input Excitation, $0.1 g^2 / \text{cps}$

2-12-1968

50 SQUARE FOOT SOLAR ARRAY - Z-AXIS [.167/cps input]

Specimen: _____ Transducer: S/N TU 05 TR 1140
S/N _____ Control () G RMS 5.7
P/N _____ Response (✓) Sampling Eq = 1

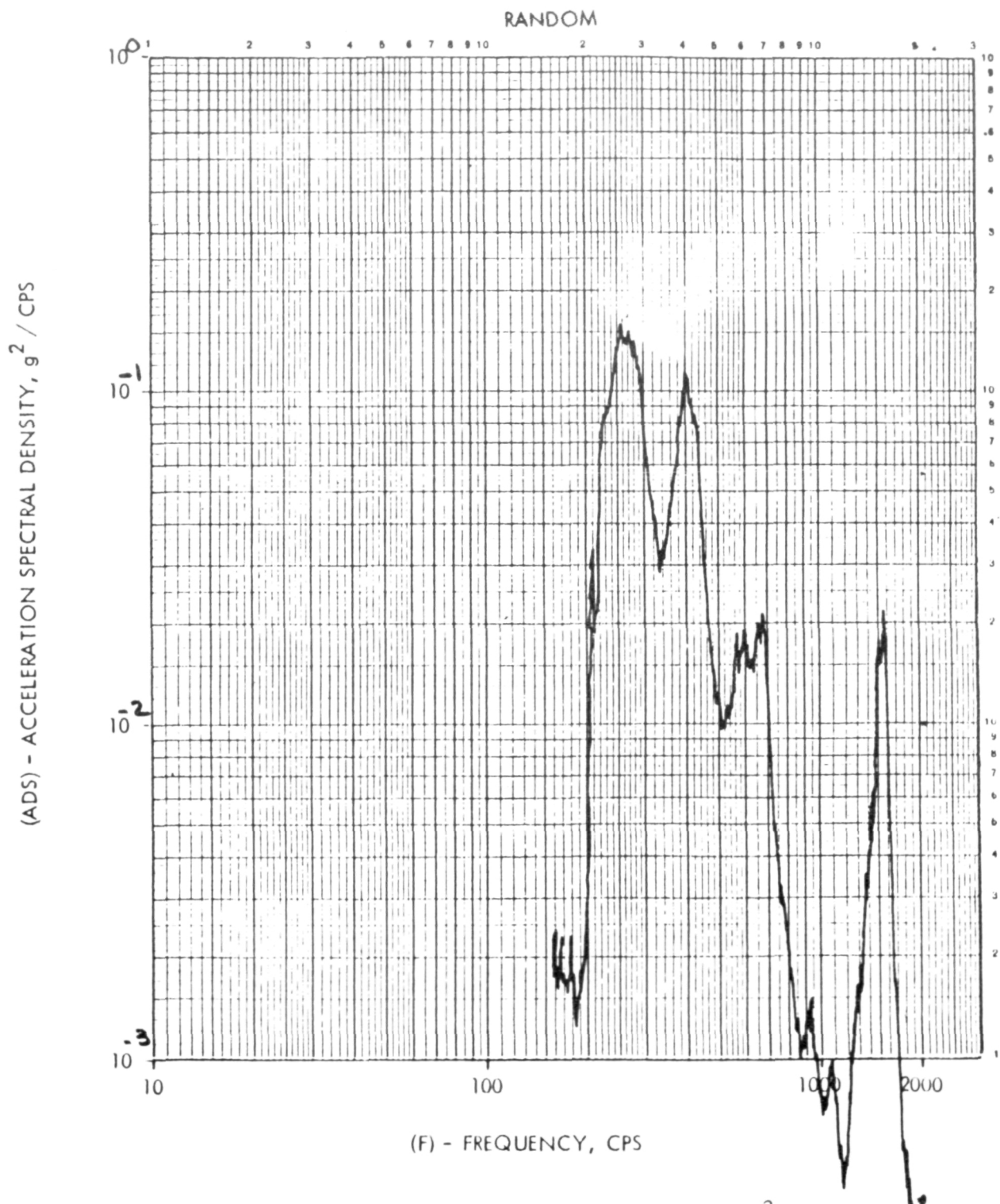


Figure 51 Response at Panel Outer Wrap, 0.1 g^2/cps

2-12-1968

50 SQUARE FOOT SOLAR ARRAY Z-AXIS [0.2 g²/cps input]

Specimen:	Transducer:	S/N	5613	TR	1148
S/N		Control	(✓)	G RMS	21
P/N		Response	()	Sampling	EQ#1

RANDOM

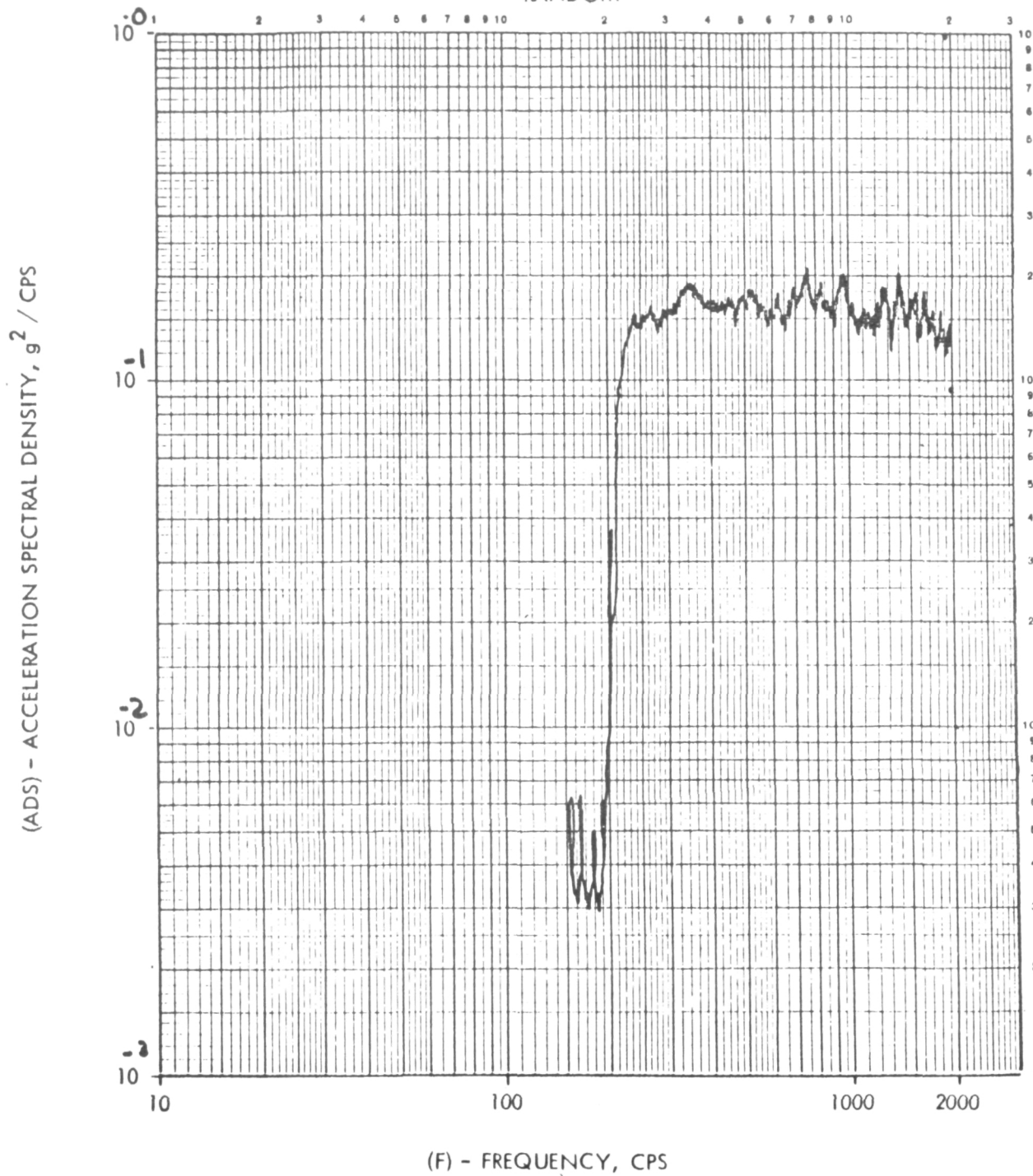


Figure 52 Input Excitation, 0.2 g²/cps

2-12 1968

50 SQUARE FOOT SOLAR ARRAY Z-AXIS [0.2 G²/cps INVT]

Specimen:	Transducer:	S/N <u>IU 85</u>	TR <u>1148</u>
S/N _____	Control	()	G RMS <u>7.1</u>
P/N _____	Response	(✓)	Sampling <u>EQ-1</u>

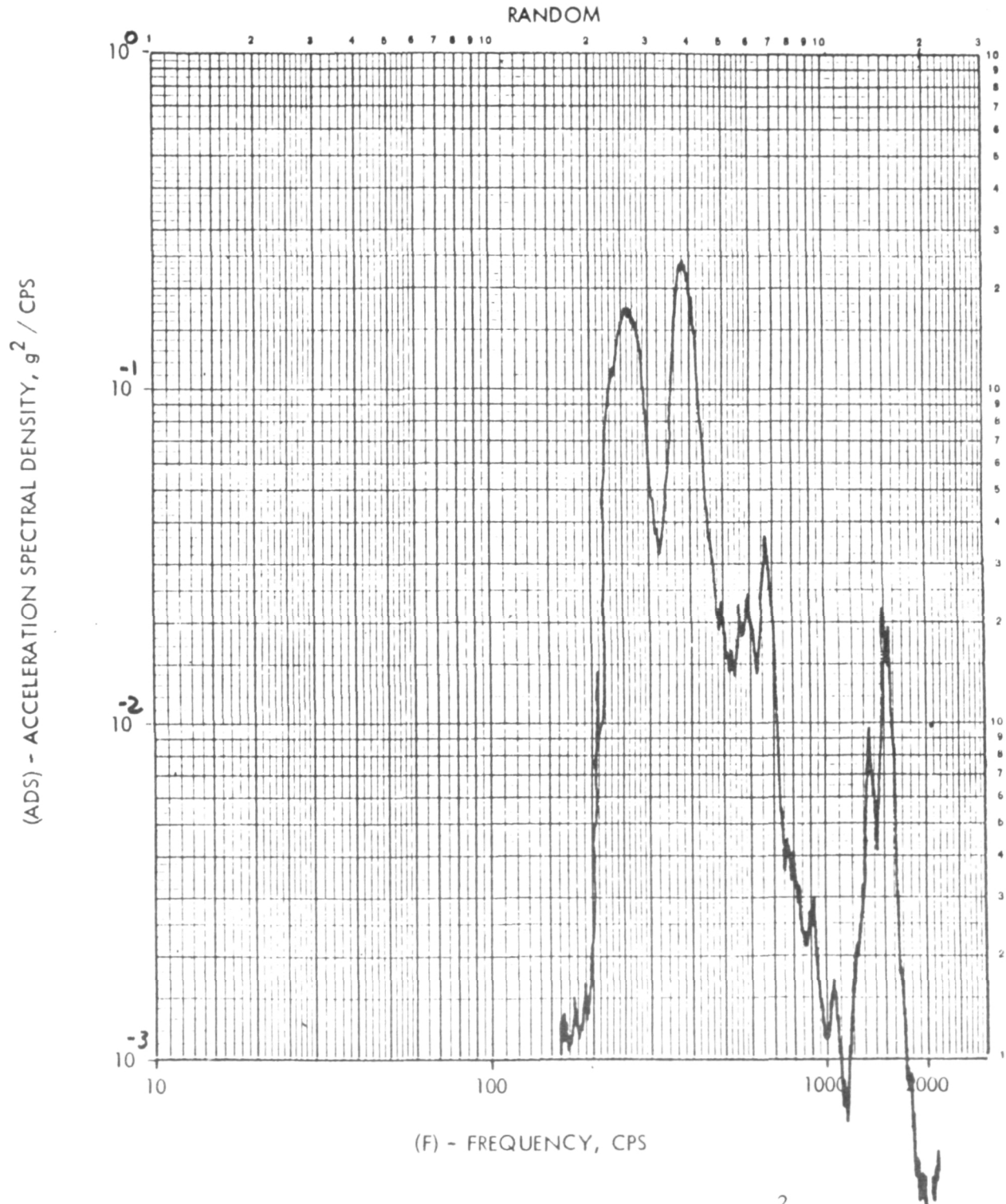


Figure 53 Response at Panel Outer Wrap, 0.2 g²/cps

4.4.5 Thermal Vacuum Test

The thermal vacuum test was run to evaluate the effects of simulated deep space thermal environment on panel deployment and retraction operation for a hypothetical mission composed of deployment while in Earth shadow, Sun-oriented retraction and deployment near Earth, and Sun-oriented retraction and deployment near Venus. Measurements were made to determine the effect of the deployment and retraction cycles on the solar cells and their electrical power output and to map the thermal profile of the panel and deployment beams for comparison with analytical predictions.

Preliminary tests were conducted to select a dummy chip and solar cell adhesive bond preparation compatible with the more extreme stress conditions encountered during thermal vacuum cycling and to determine the possible levels of stress which could be induced in the solar cells during thermal vacuum cycling for confidence in compatibility of the solar cells with that environment. Discussion of these preliminary tests is presented in Reference 5. Also, a preliminary monitor was made of the drive motor electrical current required to deploy and retract the panel at ambient room temperature conditions for comparison with that under thermal vacuum conditions.

The intended thermal vacuum cycle per Ryan Test Plan No. 20865-1, Revision A (Ref. 4) is shown in Figure 54. The initial heat rate of 50°F/minute is intended to simulate thermal shock conditions with the deployed panel moving from earth shadow into sunlight near earth, and is based on qualification testing requirements for the Mariner '67 Solar Panel structure (Ref. 7).

Test Set-Up

The tests were conducted using the modified roller support fixture described in Section 4.4.2.3. The modified fixture caused the panel to lie in a more near perfect plane providing better temperature uniformity of the deployed panel when receiving heat radiation in the test.

The fixture modification incorporated sixty additional rollers to support the deployed panel (see Figure 55) with new bearings installed to minimize friction

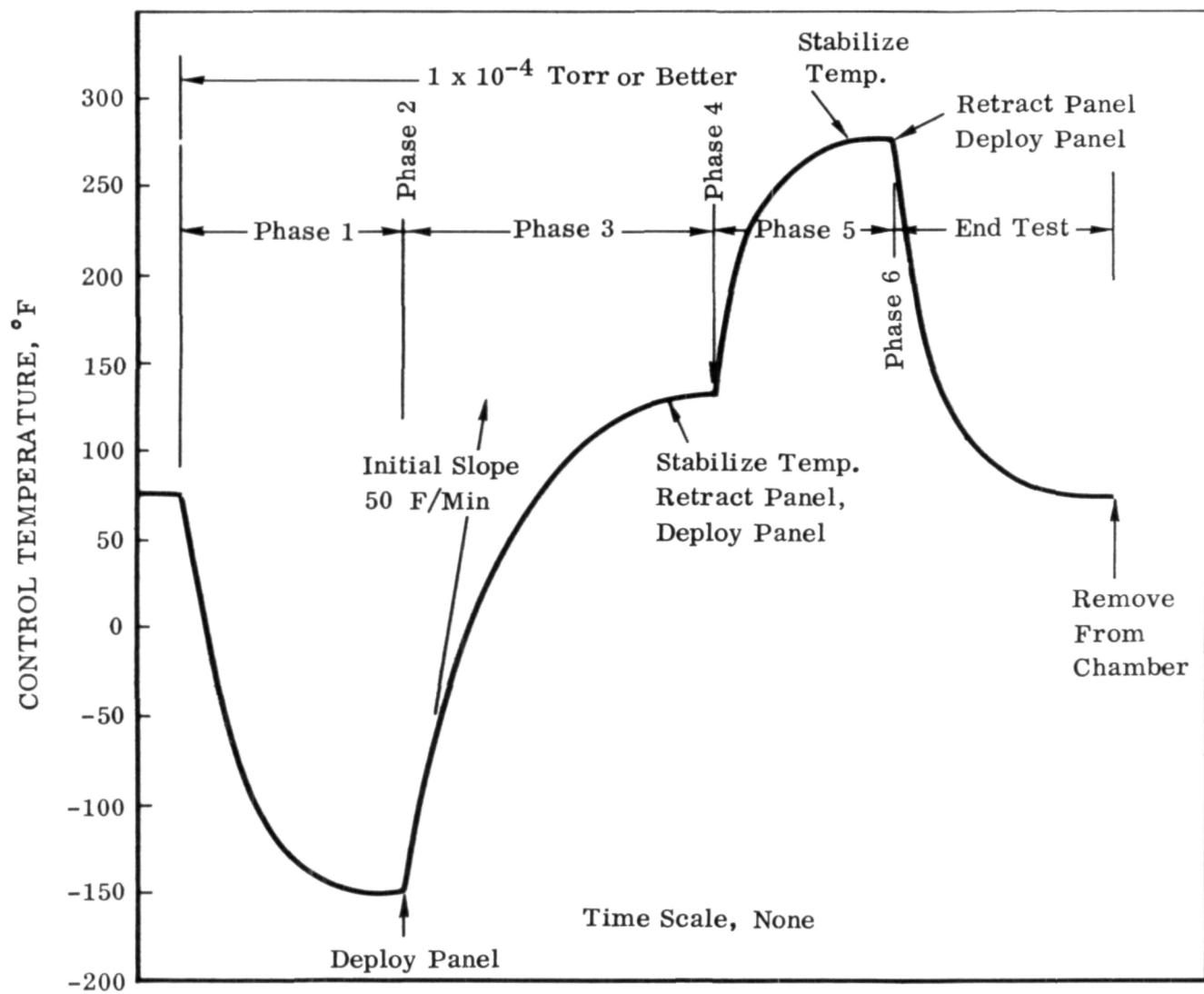


Figure 54 Proposed Test Cycle

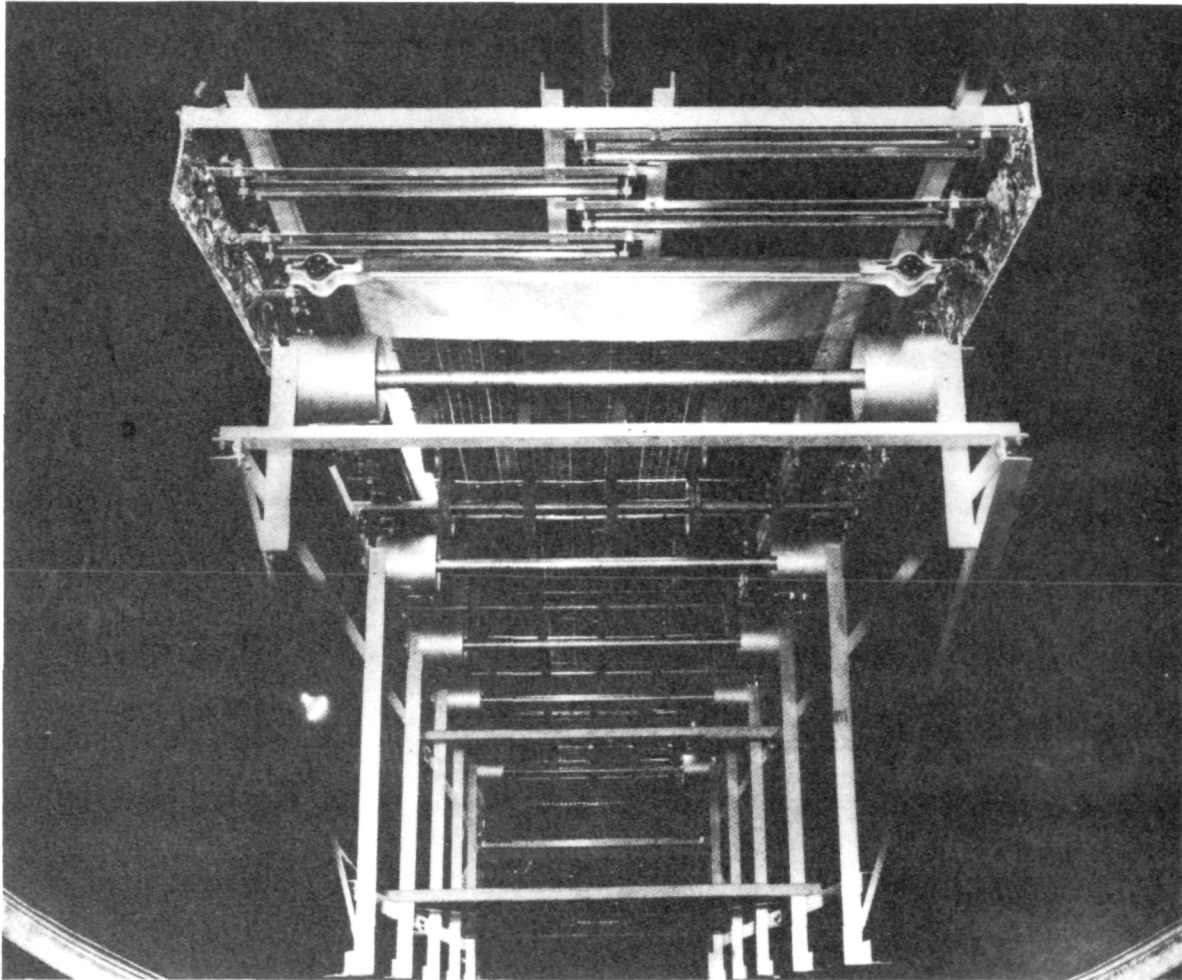


Figure 55 Deployed Panel in Thermal Vacuum Chamber

in the system. Quarter-inch thick silicone sponge was bonded to each roller surface to provide thermal insulation between it and the panel.

Frictional forces of the roller support were reduced to negligible levels by the modification. However, proper action of the guide sleeve was not satisfactory due to binding of the guide sleeve cam. After several attempts to improve the cam action by increasing clearances and using a counter balanced mount arrangement, it was decided to disengage the cam to allow free motion of the guide within the limits of the beam stack-up. This modification resulted in a significant reduction of deployment forces.

Temperature of the solar cells rather than the magnitude of heat flux was used as the test environment control. The solar cell control temperature levels were predetermined as follows for the extremes of the test thermal cycle:

- a. -150°F for earth shadow, taken from criteria for the Mariner 67 Solar Panel structure (Ref. 7). Temperature of the solar cells on the outer stowed panel wrap was the control for this criterion.
- b. $+130^{\circ}\text{F}$ for near earth solar flux, taken from JPL design criteria (Sec. 3.2.3). Temperature of the solar cells centerline of the deployed panel was the control for this criterion.
- c. $+275^{\circ}\text{F}$ for near Venus solar flux, extrapolated from the design criteria curve for (b) above. Temperature of the solar cells at centerline of the deployed panel was the control for this criterion.

Temperature gradients across the panel, in the panel plane, were minimized by retaining heat flux concentration at the panel edges using aluminum foil reflectors dropped 6-inches below the plane of the heat lamp bank (see Figure 56).

The panel was heated to temperatures defined in the test plan using two rows of 2500 watt quartz lamps 10-inches apart, mounted 10-inches above the plane of the deployed panel (see Figures 56 and 57). Half circle-section reflectors

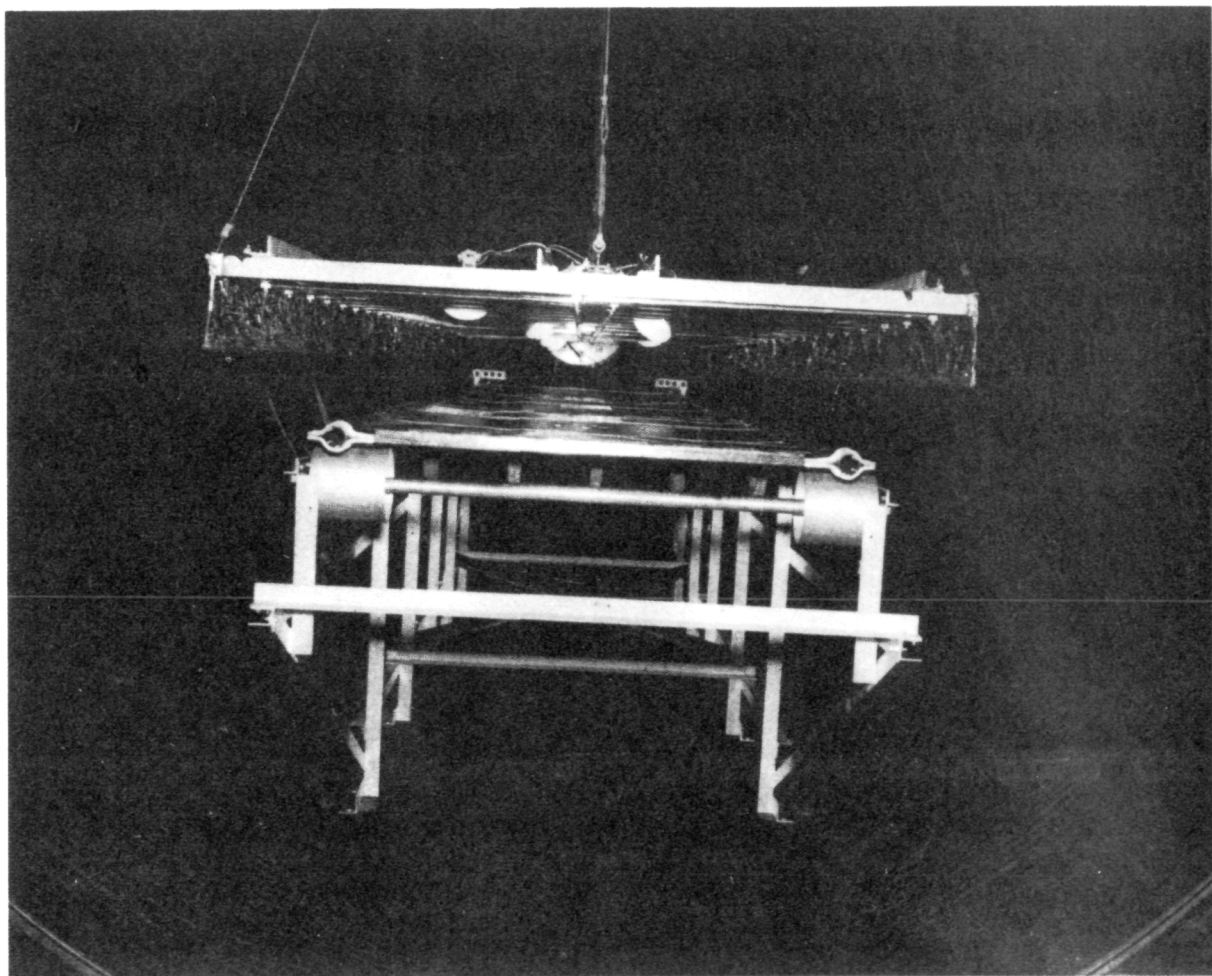


Figure 56 Relationship of Panel to Heat Source in Vacuum Chamber

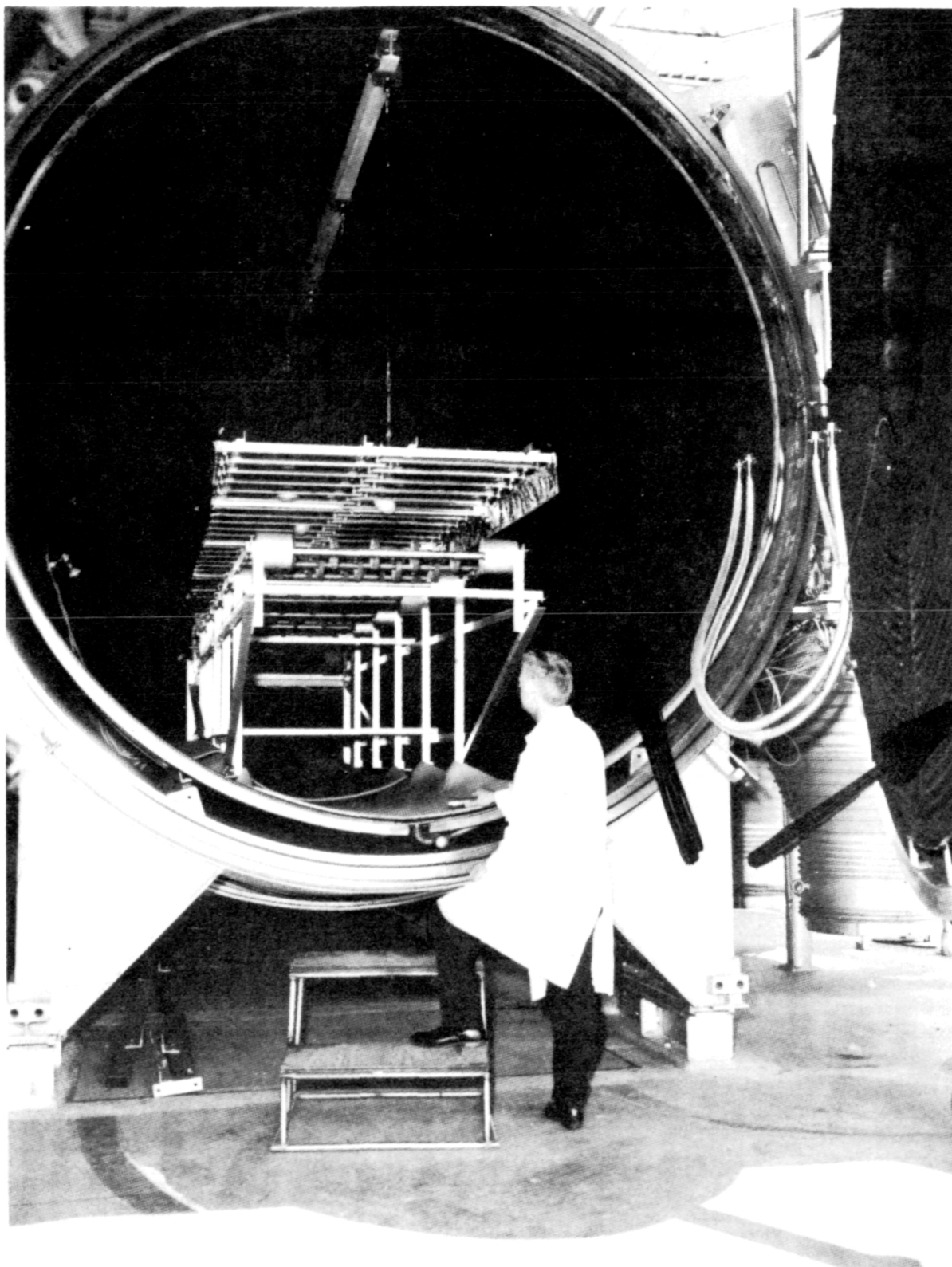


Figure 57 Cold Wall of Thermal Vacuum Chamber

were used to direct heat flux from the quartz lamps toward the panel. A tungsten lamp, mounted above each live solar cell matrix, was used to simulate solar flux to the solar cells when the electrical characteristics of the solar cells were checked (see Figure 58). Temperatures were monitored with 36 gage copper constantan thermocouples located on the solar cells, deployment beams, substrate, drive motor and wrap drum as shown in Figure 59.

These thermocouple positions were selected for measuring:

- a. Characteristic temperatures of the deployment beams
- b. control temperatures at the solar cells
- c. operating temperatures at the drive motor
- d. temperature gradients through the stowed panel wraps, from outer wrap to wrap drum
- e. uniformity of heating across the deployed panel and along the panel
- f. gradients between the solar cell cover glass and the deployed panel back side.

Temperatures of the liquid nitrogen cold wall which enshrouded the panel (see Figure 57) were monitored also using thermocouples. Provisions for viewing the panel and for photography from outside the chamber were available by means of window parts in the chamber walls.

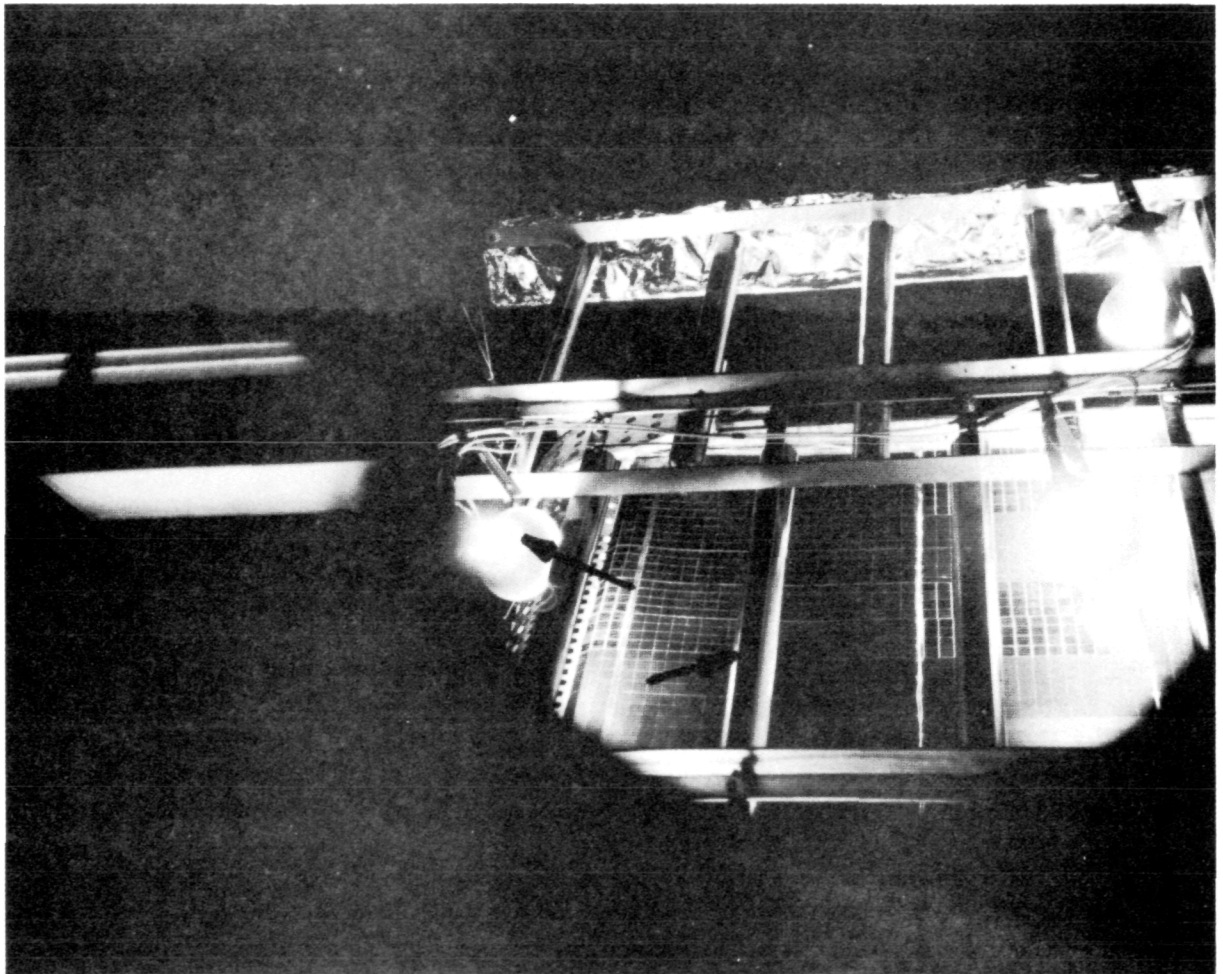


Figure 58 Lamp Installation

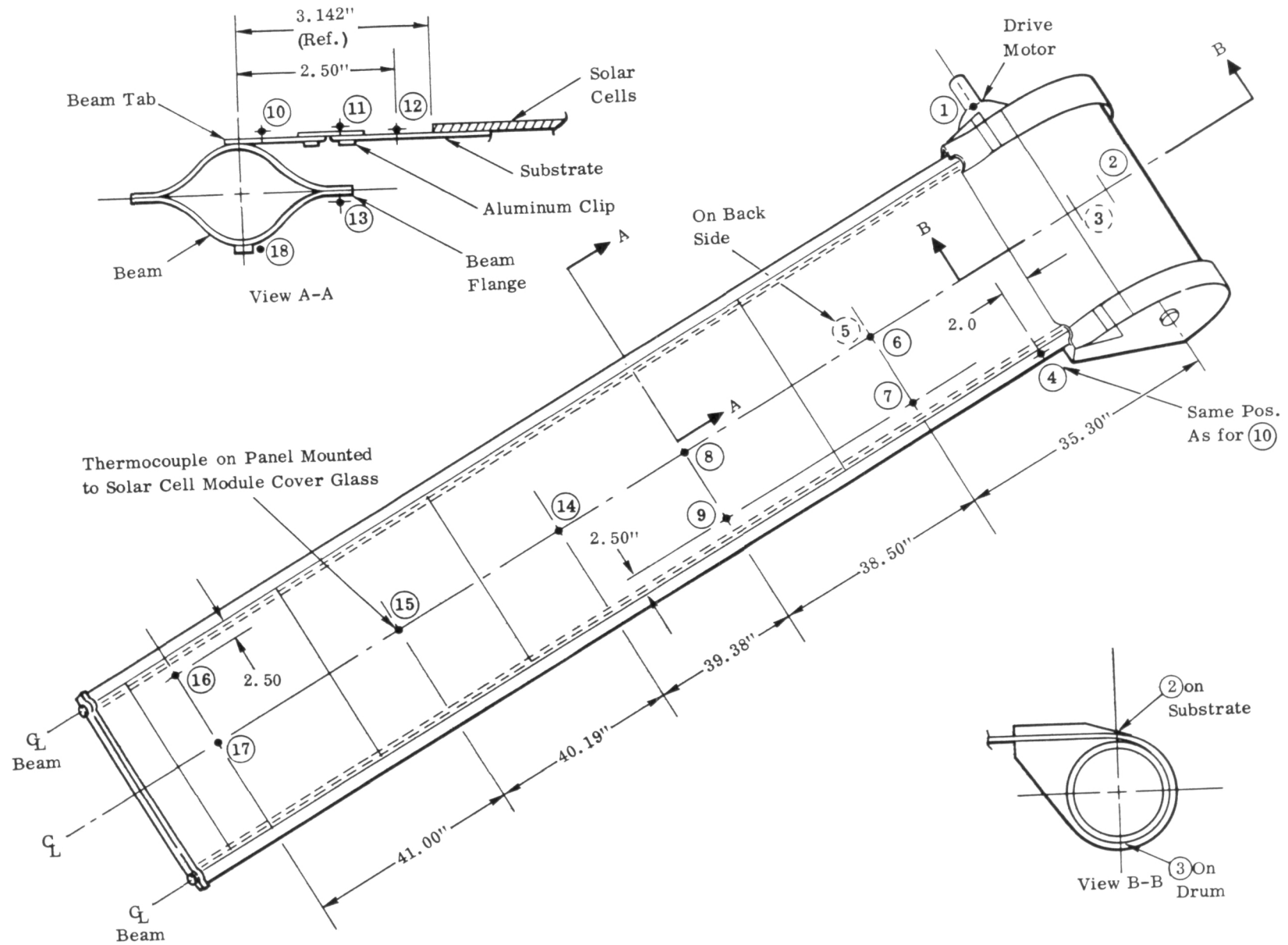


Figure 59 Thermocouple Locations

Test Procedure and Results

Instrumentation of the panel for the thermal vacuum test was completed and the panel was installed in the vacuum chamber at Convair Division General Dynamics. Preliminary checkout and all reference electrical power checks were made.

The chamber was then evacuated, the cold wall flooded with liquid nitrogen, and the retracted solar panel allowed to cool to -150°F . The panel was then deployed. During deployment, the extension was slower than normal with erratic motion and with over 1.0 amperes motor current required. Following extension, it was observed that except for the outboard 30-inches all of the aluminum edge connector clip tabs on the left hand beam had undergone a shear-tension type failure, which allowed the substrate to separate from the beam (see Figure 60). It was evident that additional retraction and deployment cycles would risk additional damage and malfunction. Therefore, it was decided to deviate from the test plan by heating the panel to 130°F and 275°F without attempting deployment cycling; then to bring the chamber to ambient conditions after recording solar cell electrical power output, and repair the edge attachment sufficiently to permit a retest of the 130°F and 275°F deployment cycles with an optional retest of the -150°F deployment deployment if feasible.

The thermal shock cycles were completed and the unit removed from the chamber for repair. The repair (see Figure 61) consisted of replacing the damaged aluminum edge connector clips with a 2.0-inch wide strip of fiberglass cloth bonded to the beam and substrate. The strip was intended to allow an excess of 0.25-inch to offset thermal contraction. However, the wrapping characteristics of the substrate were found to be different because the fiberglass attachment permitted the substrate and left-hand beam to become displaced 0.50-inch with respect to each other during retraction. The effect of this was to negate the thermal contraction provision in the fiberglass.

Also the left hand beam wrapped onto the storage drum at a slower rate than the right hand beam. The lag after the first 12 inches of retraction was

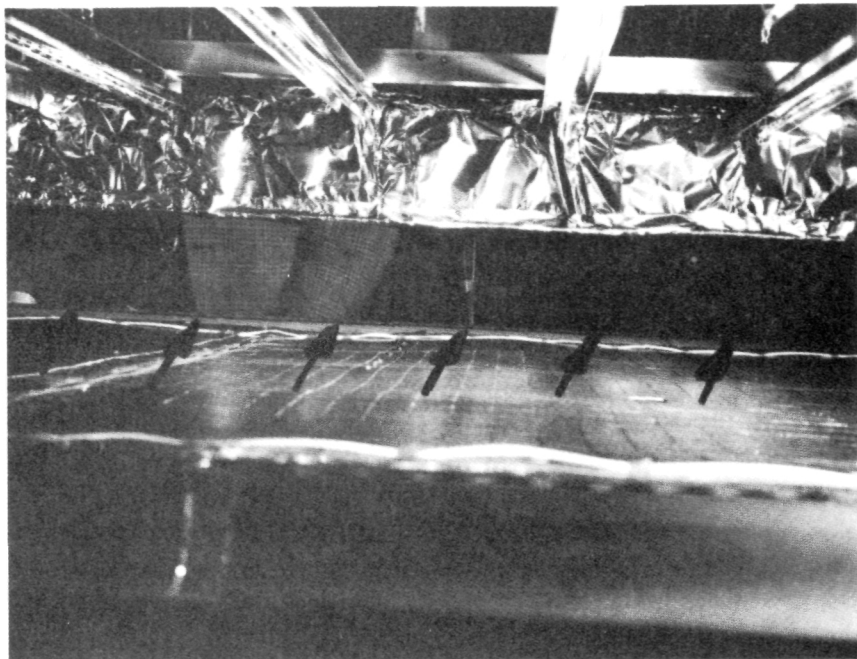


Figure 60 Damage to Aluminum Clip Attachment at Left Deployment Beam

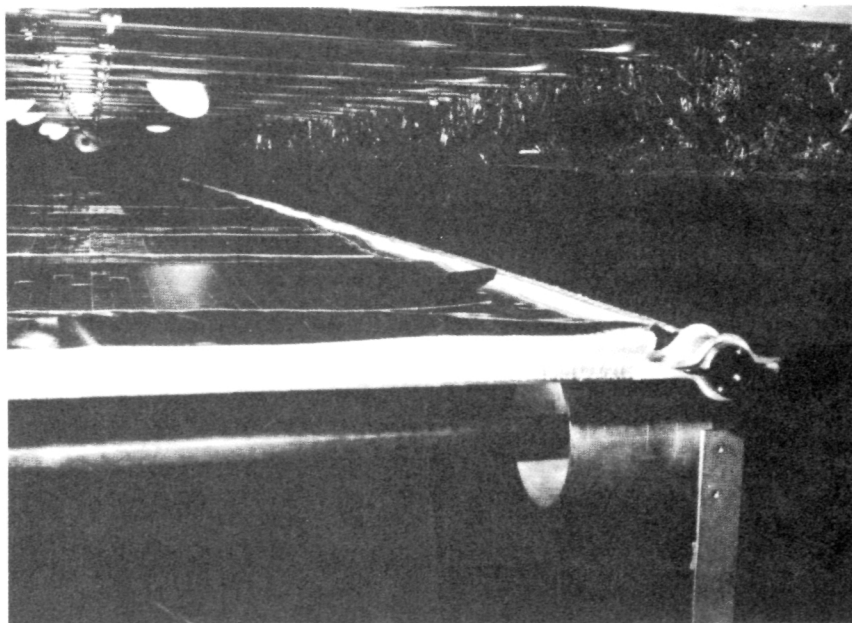


Figure 61 Fiberglass Attachment on Left Deployment Beam

1/4 inch increasing to 3/8 inch when fully retracted. The substrate wrap rate closely followed the rate of the right hand beam.

During the first retraction after repair the substrate separated from the outboard intercostal hinge pin approximately 1/4 inch when the two hinges nearest the left hand beam failed as the beams approached the full retract position.

In subsequent deployment the relative displacement between the substrate and left hand beam decreased until the separation of intercostal reached zero when fully extended.

During the next retraction the separation between intercostal and beam increased to 3/8 inch when fully retracted. There was an equal displacement of the substrate relative to the beam causing an identical displacement of the transverse fibers of the fiberglass repair strip.

As the substrate wrapped on the drum during retraction the fiberglass cloth splice developed a reverse fold as it passed through the guide and onto the drum. (This was intended when excess material was provided).

When retracted, the outermost three feet of the fiberglass splice strip was tightly tensioned between the beams because of fiber distortion.

An attempt was made to coordinate the beam travel rates by removing rubber from the right hand beam spacer strip. This was not immediately effective and because this method was irreversible it was decided to attempt to match the beam travel by adding shims on the storage drum under the left hand beam. Two 0.030 thick x 2.0 inch wide aluminum shims were placed on the drum. This made the retraction rate of the left hand beam faster than the right hand beam. With one 0.030-inch shim in place the retraction rate was nearly equal (approximately 0.070 variation). However, during retraction the substrate near the fiberglass splice continued to move inboard in advance of the left hand beam (approximately 1/4-inch displacement). The reason appears to be that the substrate wraps on top of the beams, and with the drum diameter

increased, the left hand beam wrapped to a larger diameter, and the substrate wrapped to a larger diameter maintaining the same displacement relative to the beam.

When retracted about halfway the motor current increased to 1.0 amp and the solar cell modules began to rack. This was unsatisfactory, therefore, the shim was removed from the drum and testing was resumed with the repaired unit deployed in the vacuum chamber. The panel was heated to 130°F and the test continued according to test plan. Solar cell electrical power output was recorded. The panel was retracted and redeployed. Then the panel was shock heated to 275°F simulating the Venus environment, retracted and redeployed, and solar cell electrical power recorded. These test events were completed in a satisfactory manner with normal operation of the panel.

It was then attempted to repeat the -150°F deployment cycle. The panel temperature was reduced to approximately 30°F and retracted. The retracted array was then allowed to cool to -150°F and deployed. During deployment, the extension rate was slower than normal, and erratic with high motor current similar to the first deployment at -150°F. When the drum rotation was completed, panel extension was incomplete due to formation of a reverse loop in the deployable beam and substrate (see Figure 62). There was no failure of substrate to beam connectors or the fiberglass repair strip. The inboard solar cells were badly damaged and there were some substrate tears in the vicinity of the reverse loop.

The unit was removed from the chamber. The reverse loop was removed by manually feeding the beams through the guide sleeve as the drum was retracted. The array was then retracted and returned to Ryan for completion of sunlight check of solar cell performance. This check showed degradation of the damaged cells, otherwise no change.

An evaluation of the cause of the problems during deployment at -150°F was conducted. The problems apparently are a result of differential thermal contraction along with miscoordination of beams during retraction and deployment. Discussion of that evaluation is presented in Section 4.4.5.1.

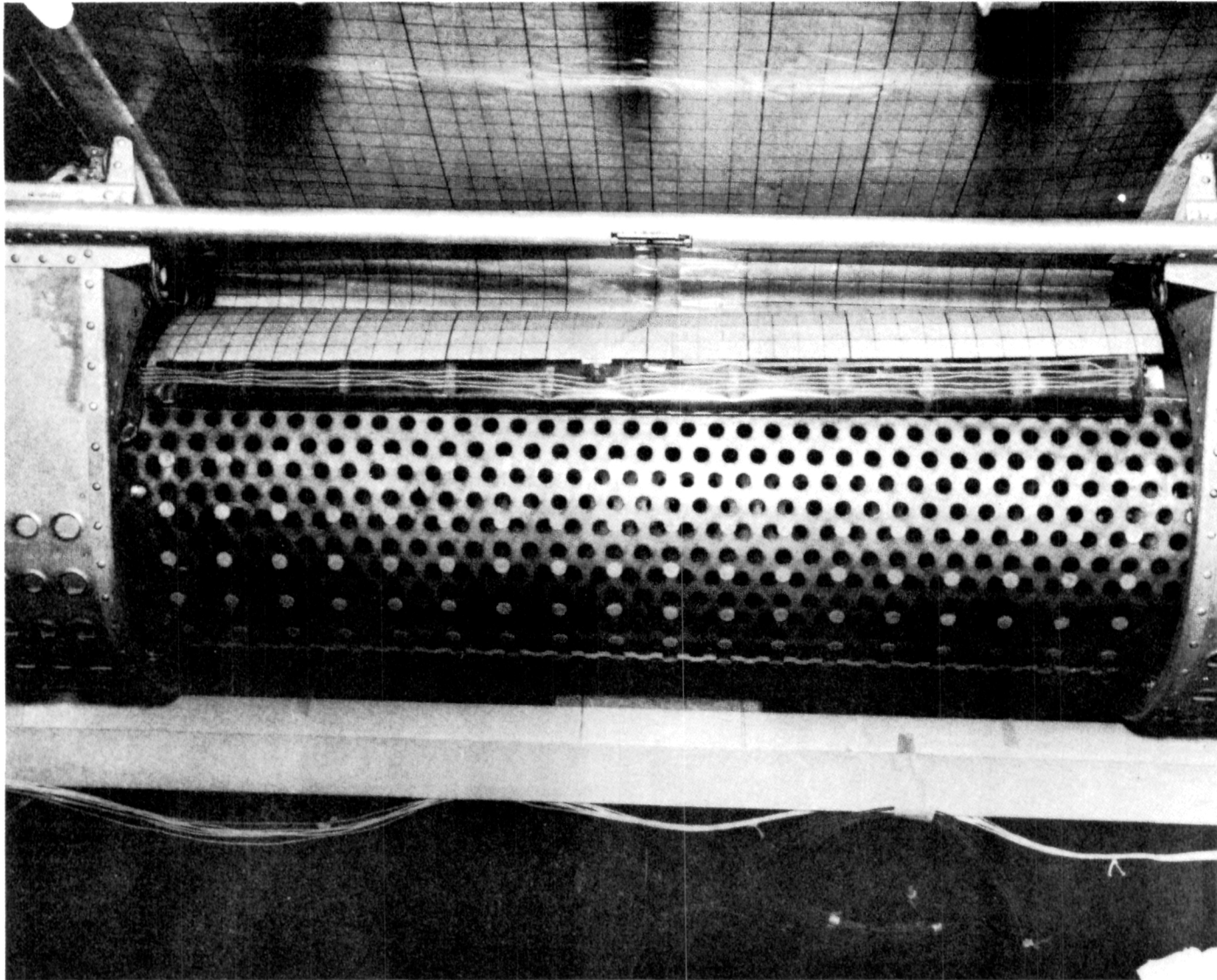


Figure 62 Reverse Loop Buckle in Deployable Beams

Figure 54 shows the proposed test cycle per the Test Plan (Ref. 4). The cycle is divided into basic elements, termed phases, to facilitate plotting temperature data, over the relatively large accumulated test time, in shorter time elements for reasons of graphical clarification. Figures 64 and 65 show the temperature-time plot of one panel thermocouple for comparison of the typical temperature cycle followed for tests 1 and 2 with respect to the proposed test cycle. Figures 65 through 82 show time-temperature plots at the various thermocouple locations defined in Table 5. No plots are shown for certain thermocouples for the following reasons:

<u>Thermocouple Number</u>	<u>Reason For Not Plotting Data</u>
1	Temperature controlled by separate heater
4	Data similar to T.C. No. 10
5	Similar to T.C. No. 6: T = 0°F, stabilized at -150°F T = 1°F, stabilized at +130°F T = 4°F, stabilized at +275°F
13	Data similar to T.C. No. 18
14	Control T.C.

Where phases are duplicated between tests 1 and 2, data is plotted for test 1 since temperature control was closer to that of the proposed test cycle.

The air pressure level in the vacuum chamber with respect to each phase of the test cycle was as follows:

phase 1	1.8×10^{-7} torr
phase 2	1.8×10^{-7} torr
phase 3	1.9×10^{-7} torr
phase 4	1.0×10^{-6} torr
phase 5	2.2×10^{-7} torr
phase 6	1.0×10^{-6} torr

Table 5 Thermocouple Locations

THERMOCOUPLE NUMBER	LOCATION	
	SITE	DISTANCE FROM DRUM (IN.)
1*	Drive Motor	0
2	Substrate Inner Wrap	0
3	Wrap Drum	0
4*	Beam, Inboard Upper	14.0
5*	Substrate, Lower	35.3
6	Solar Cell, Center	35.3
7	Dummy Cell, Side	35.3
8	Solar Cell, Center	73.8
9	Solar Cell, Side	73.8
10	Beam, Upper Tab	73.8
11	Edge Connector	73.8
12	Substrate	73.8
13*	Beam Edge	73.8
14*	Solar Cell, Center	113.2
15	Solar Cell, Center	153.4
16	Solar Cell, Side	194.4
17	Solar Cell, Center	194.4
18	Beam, Lower	73.8

* Data not plotted

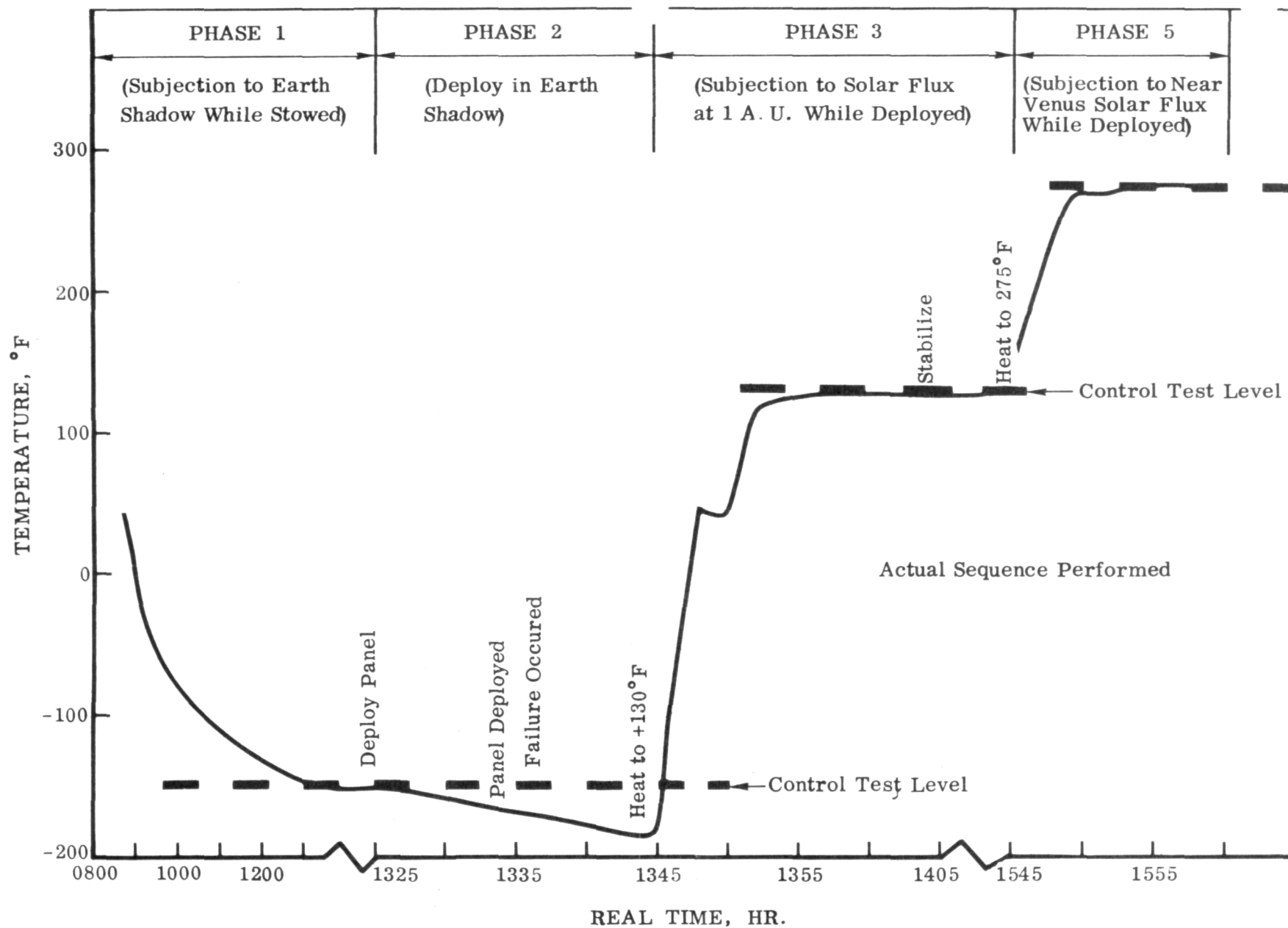


Figure 63 Thermal Vacuum Cycle, First Test

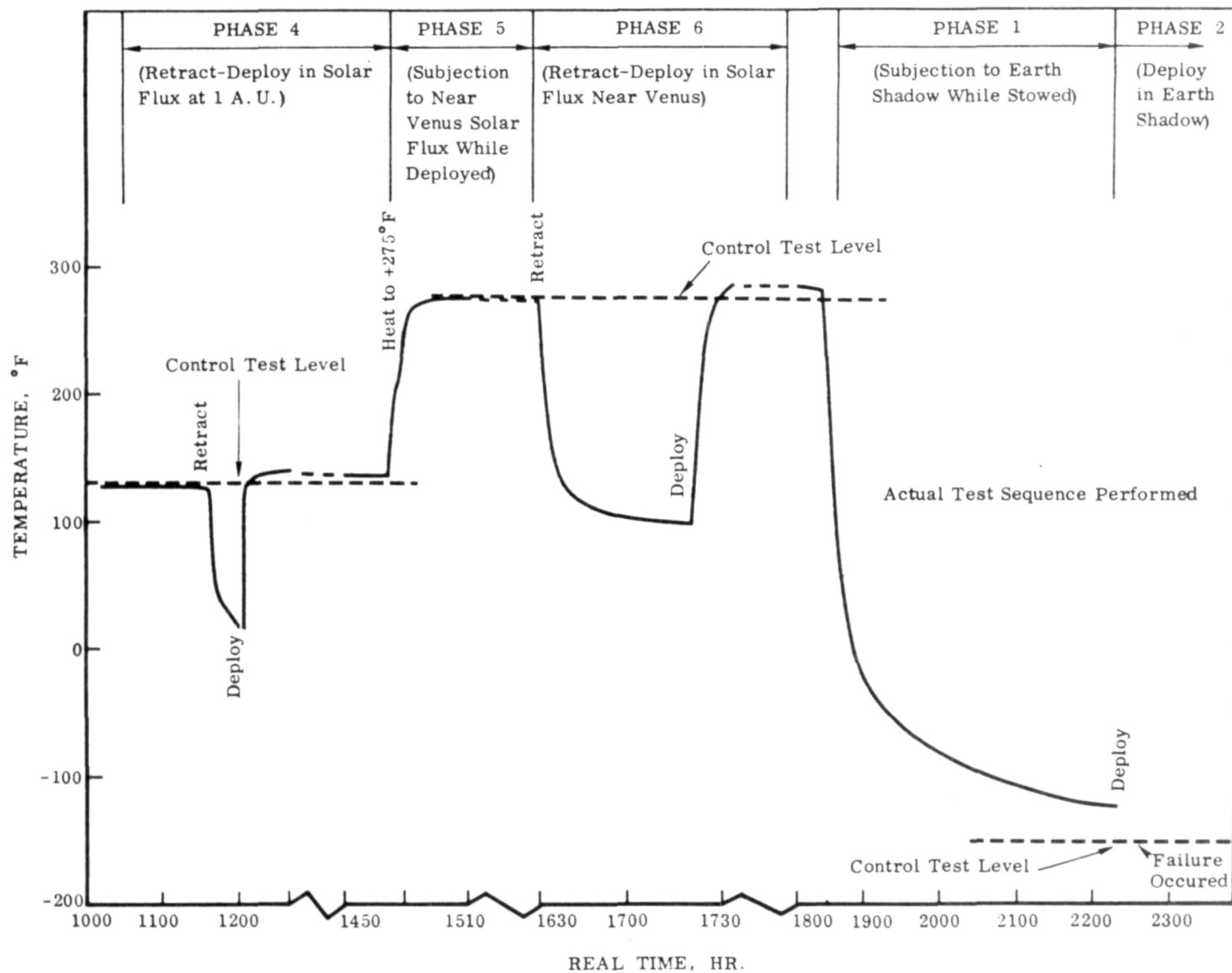


Figure 64 Thermal Vacuum Cycle Second Test

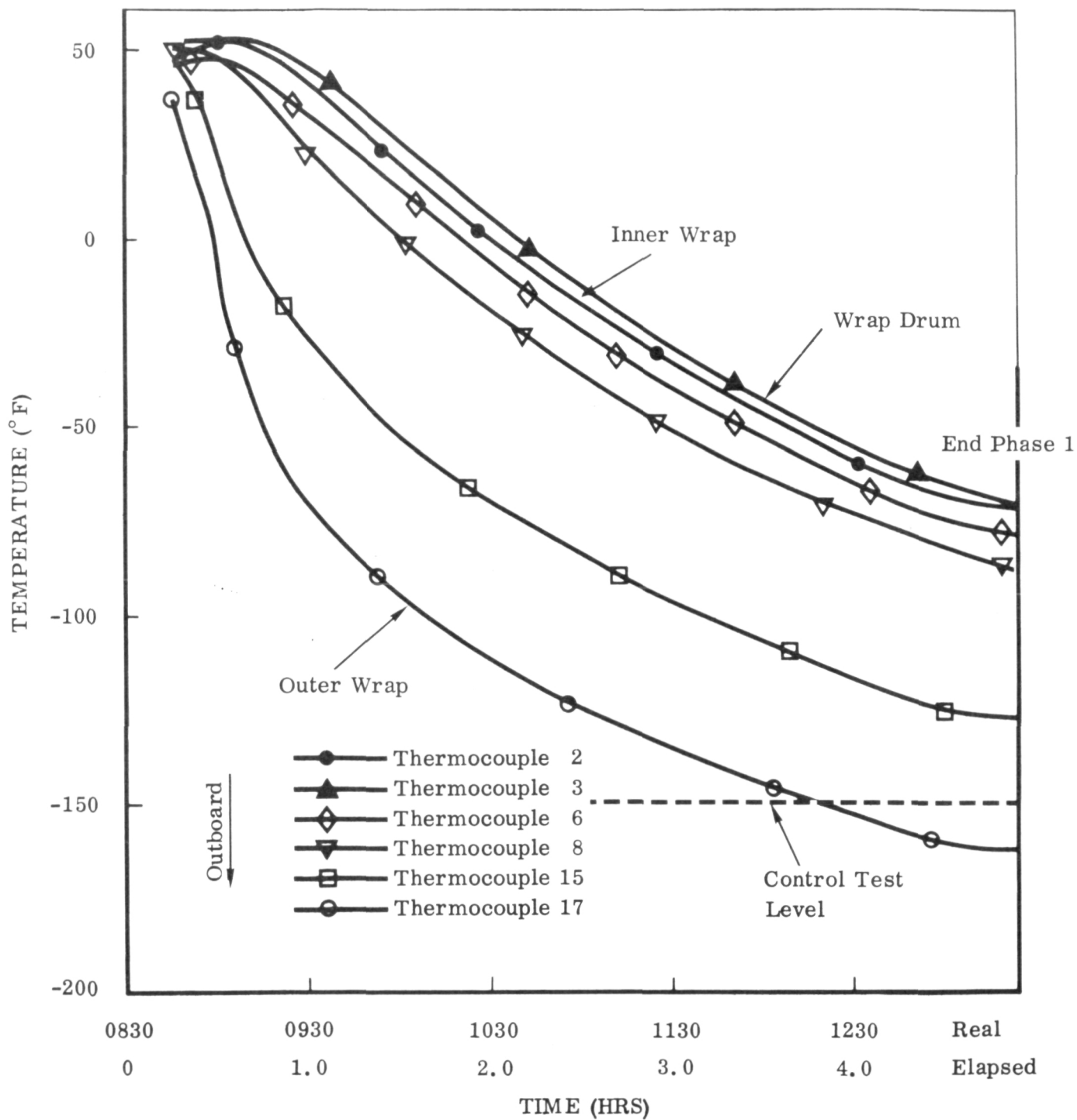


Figure 65 Phase 1 Time-Temp. History of Solar Cells at Panel Centerline

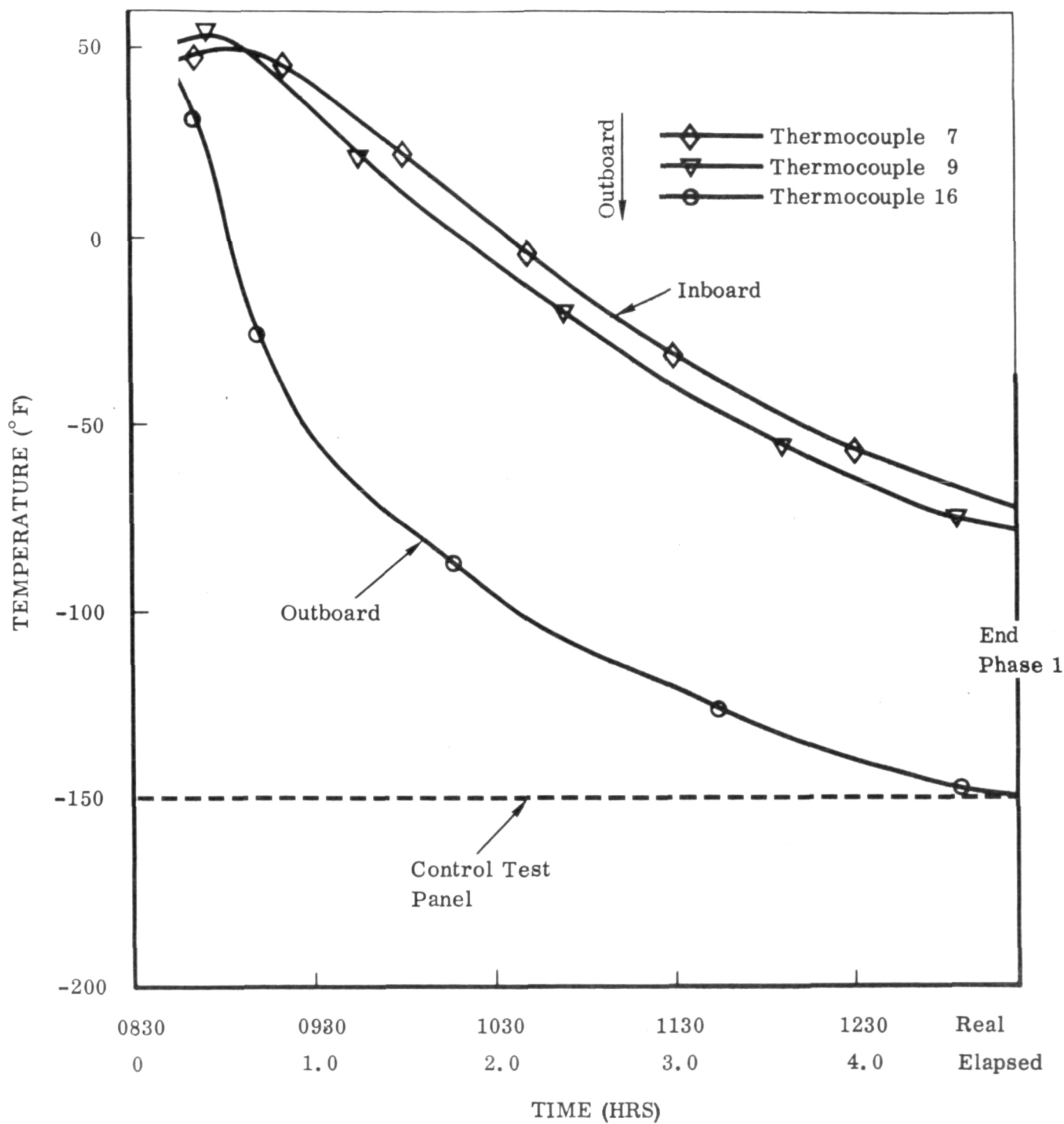


Figure 66 Phase 1 Time-Temp. History of Solar Cells at Panel Edge

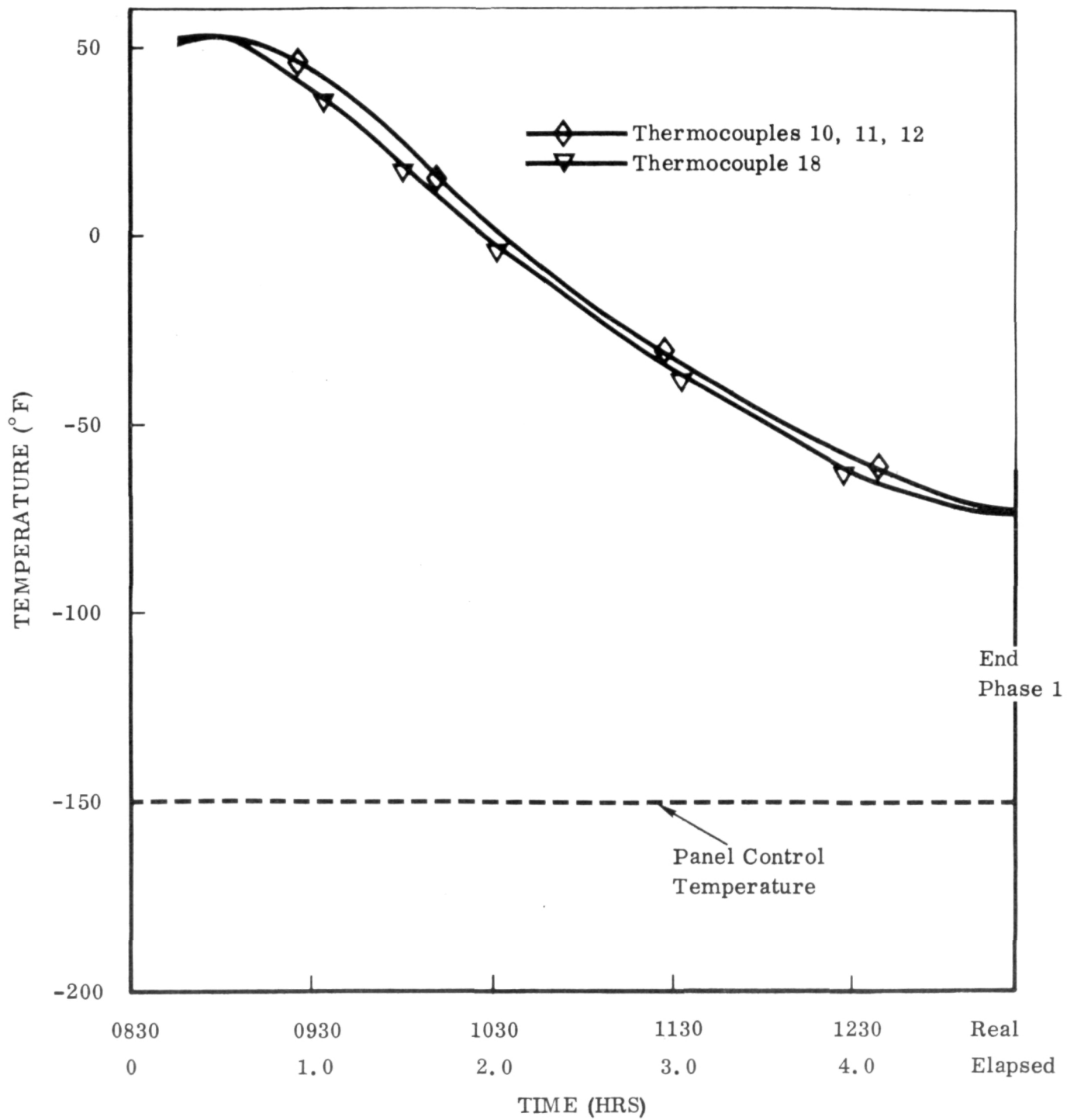


Figure 67 Phase 1 Time-Temp. History of Deployment Beam

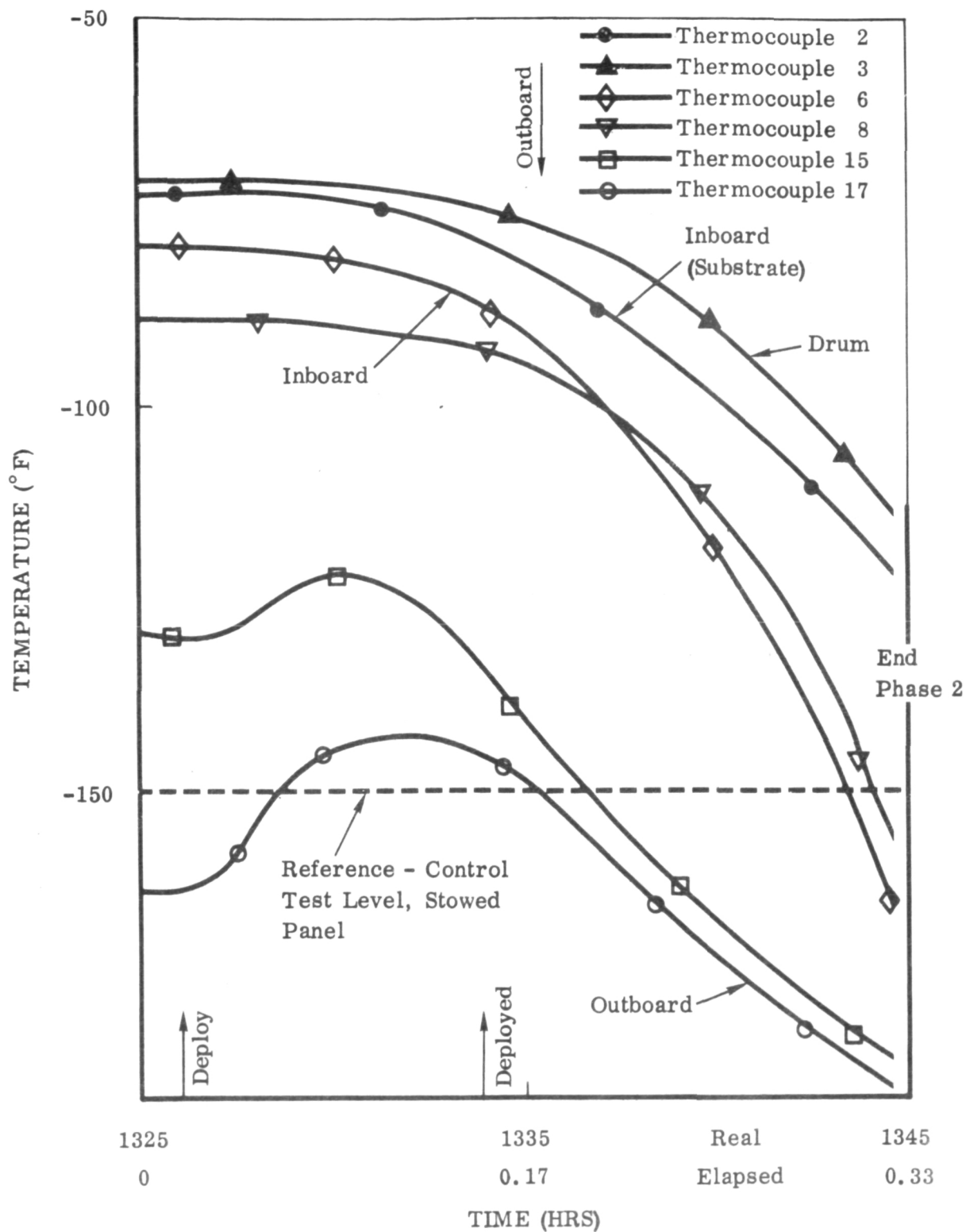


Figure 68 Phase 2 Time-Temp. History of Solar Cells at Panel Centerline

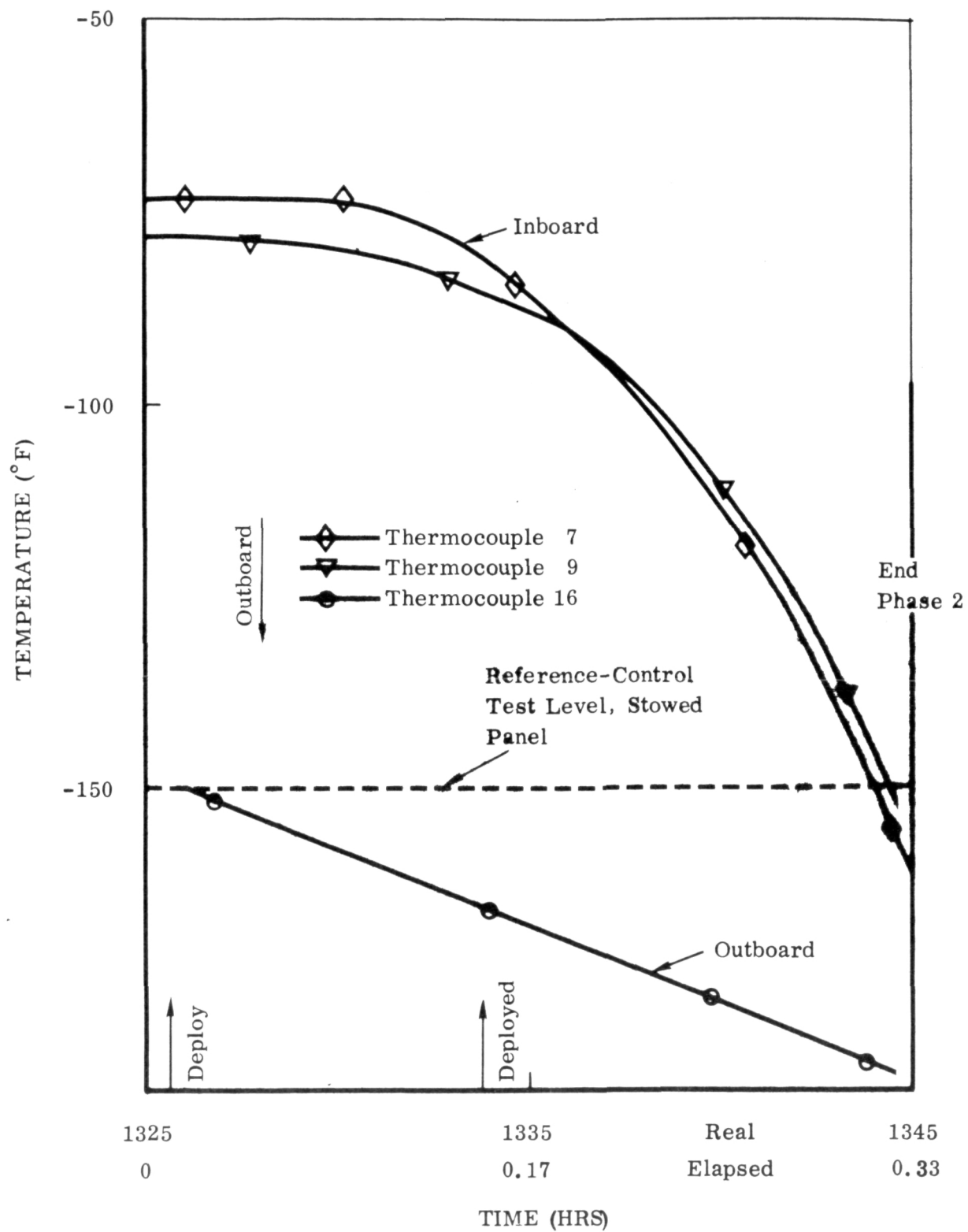


Figure 69 Phase 2 Time-Temp. History of Solar Cells at Panel Edge

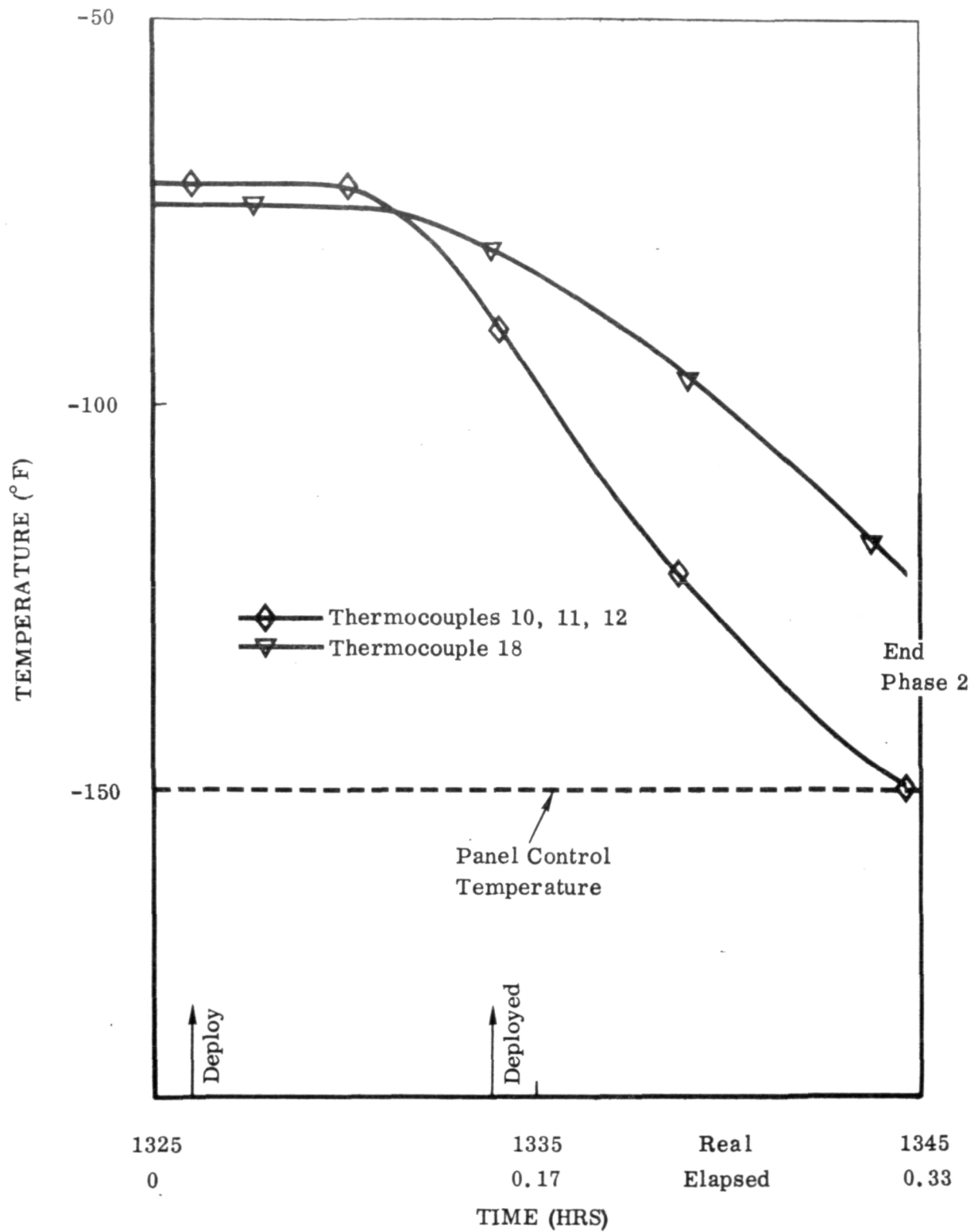


Figure 70 Phase 2 Time-Temp. History of Deployment Beam

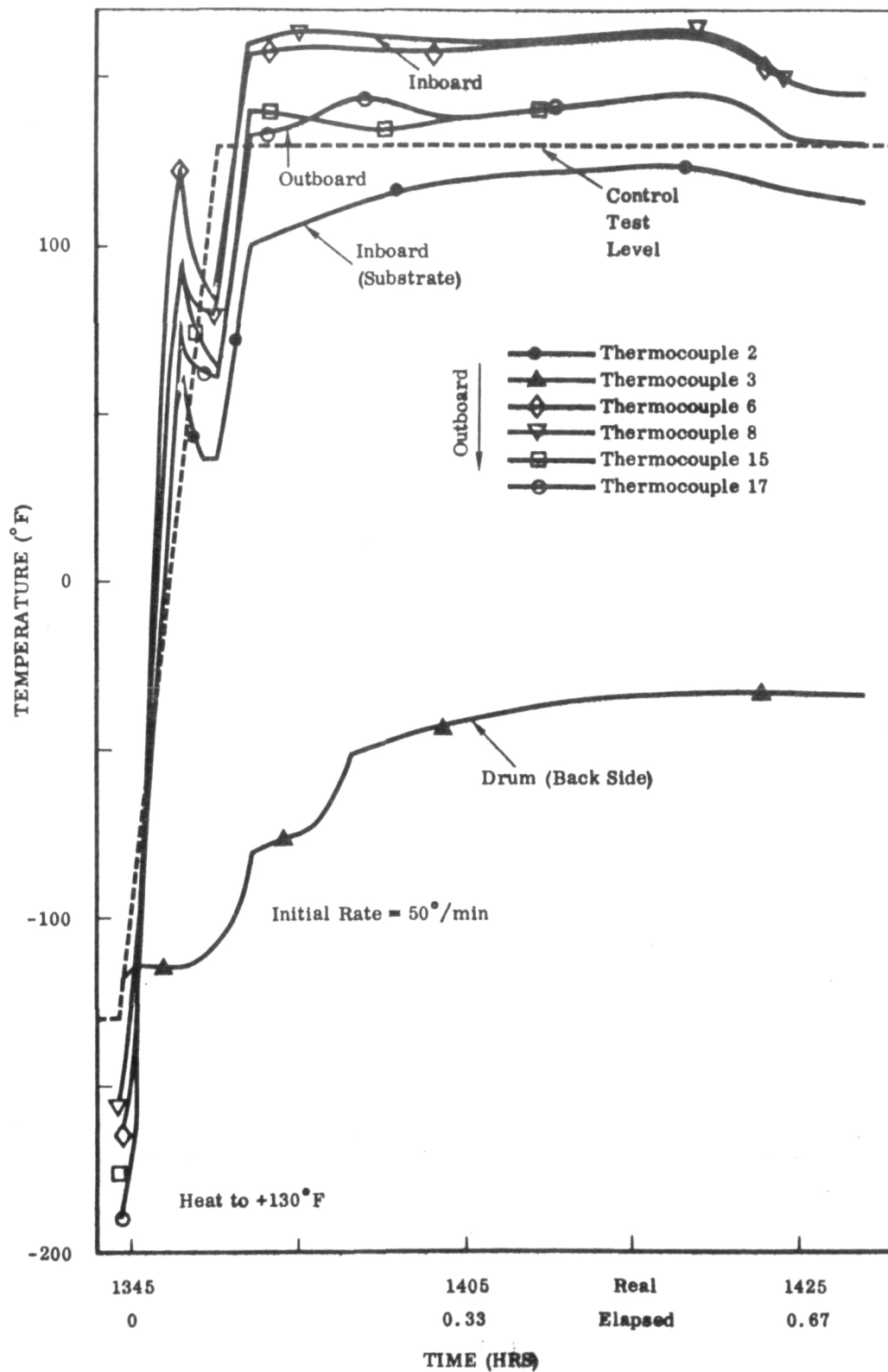


Figure 71 Phase 3 Time-Temp. History of Solar Cells at Panel Centerline

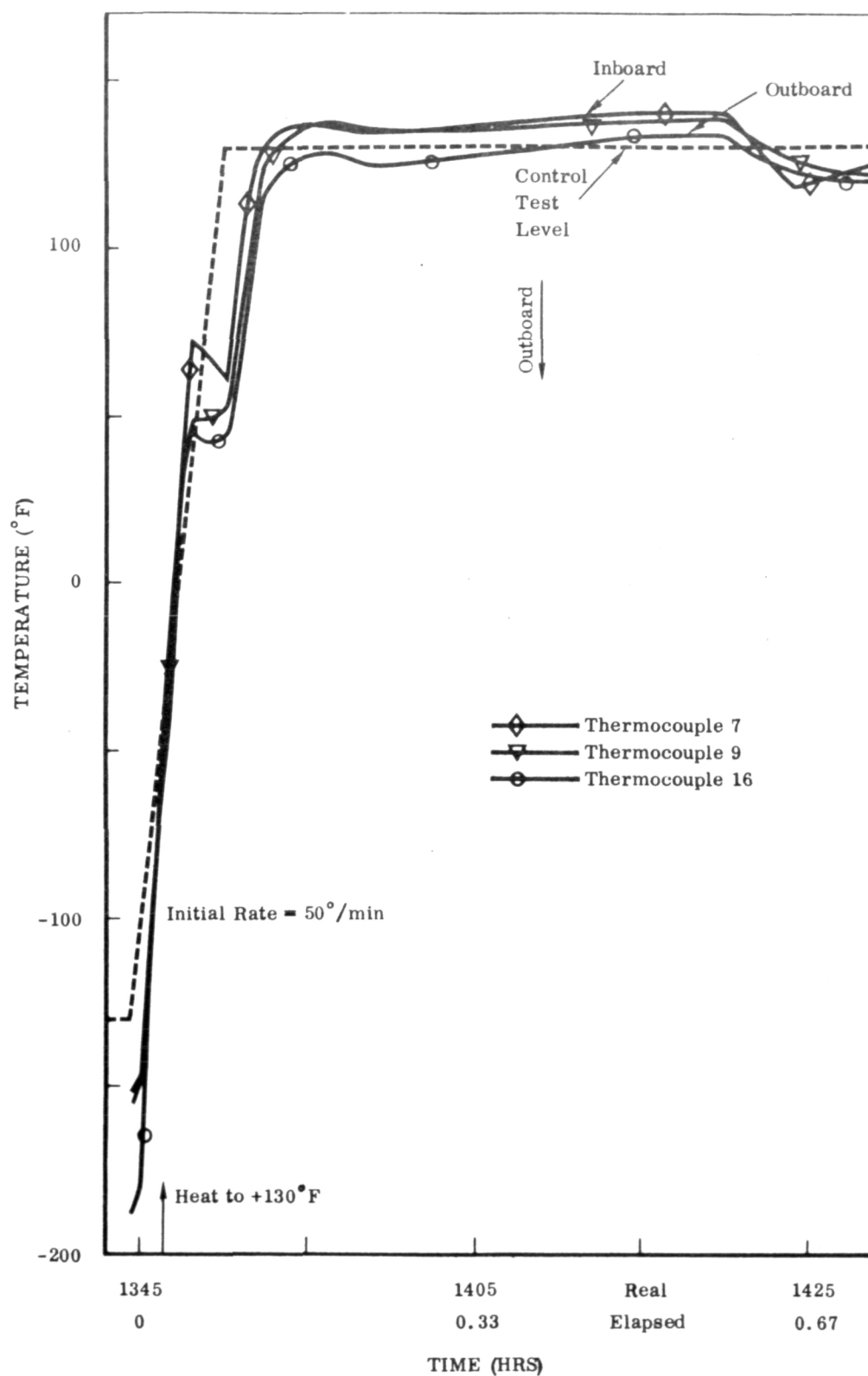


Figure 72 Phase 3 Time-Temp. History of Solar Cells at Panel Edge

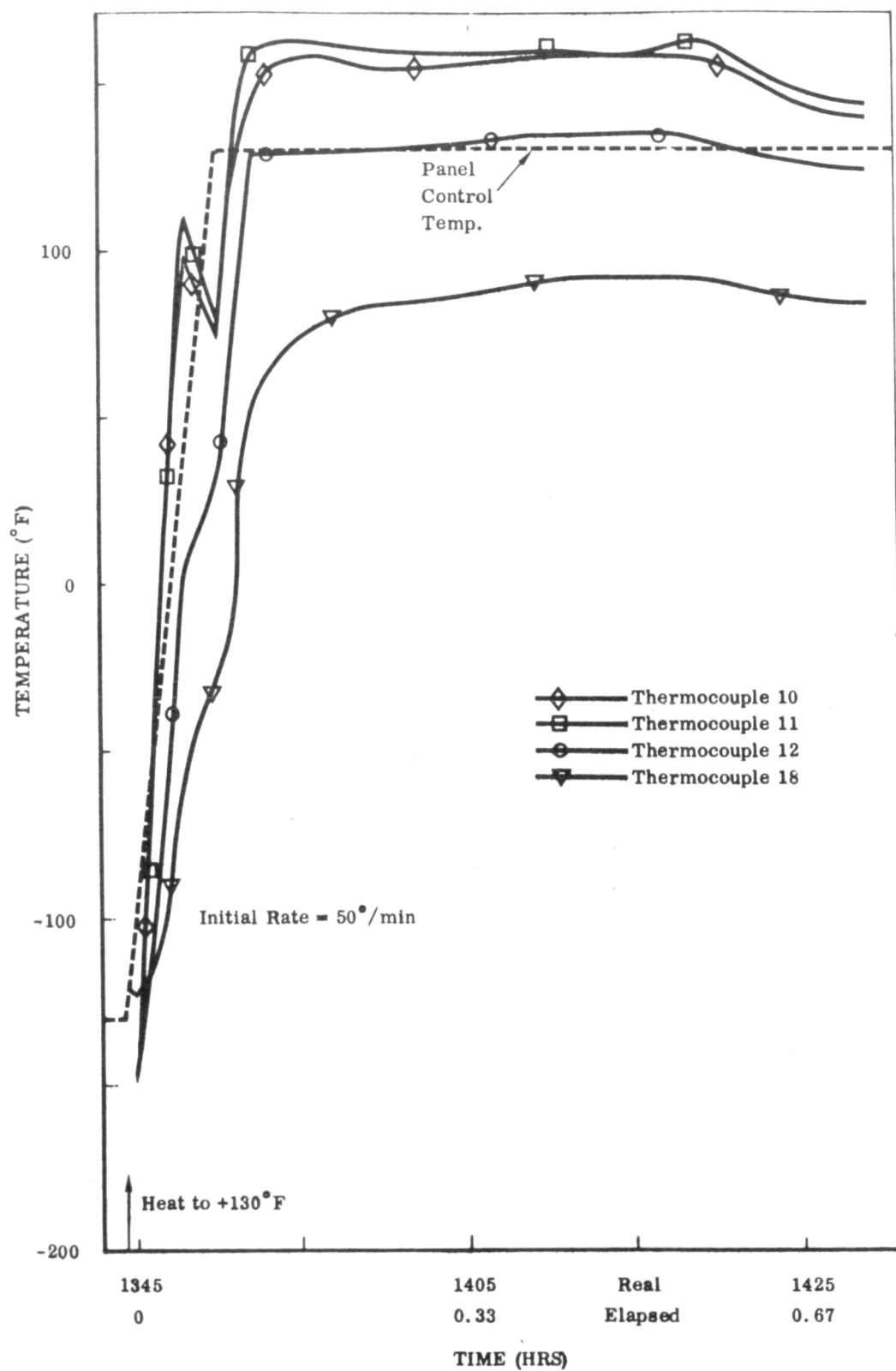


Figure 73 Phase 3 Time-Temp. History of Deployment Beam

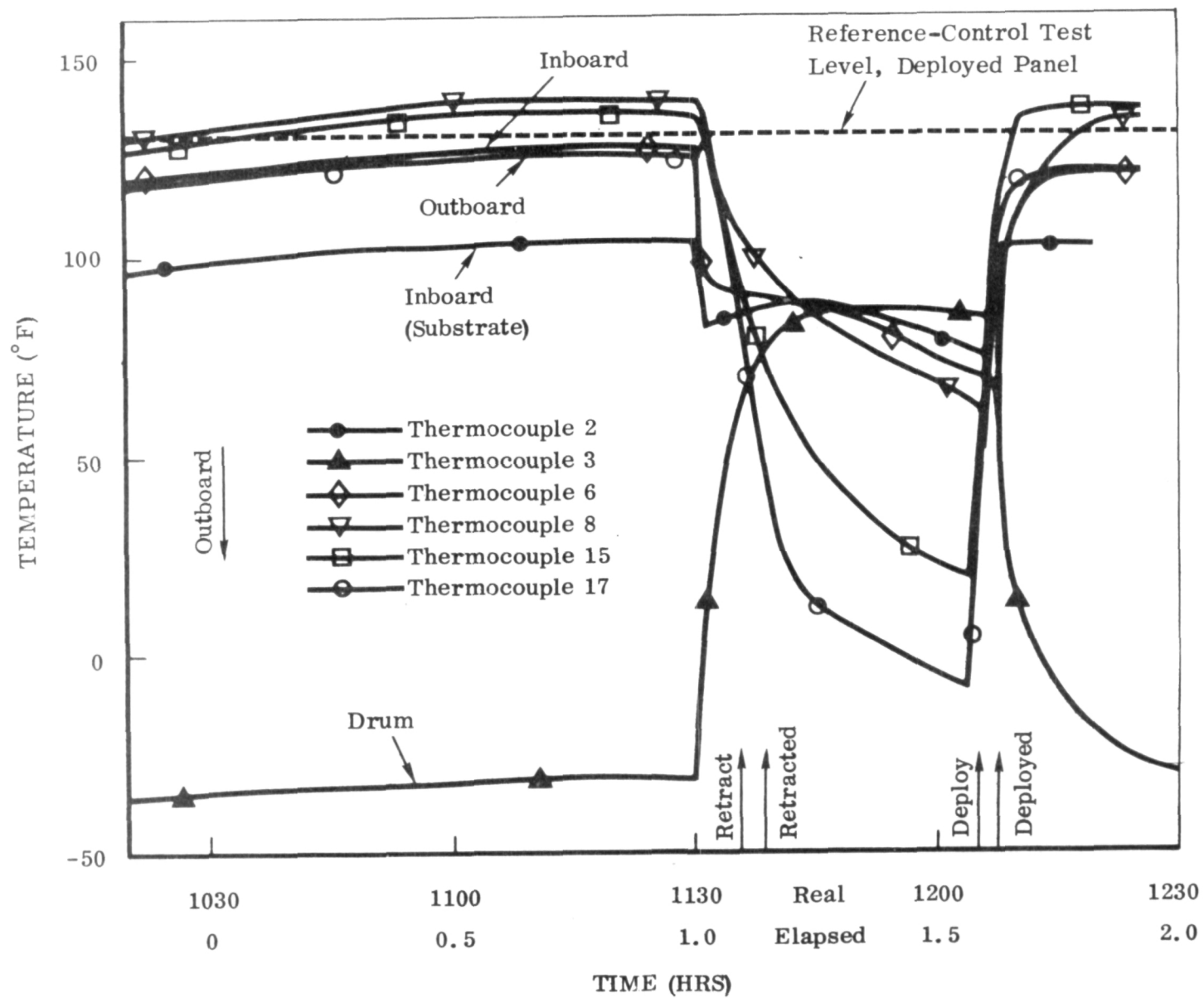


Figure 74 Phase 4 Time-Temp. History of Solar Cells at Panel Centerline

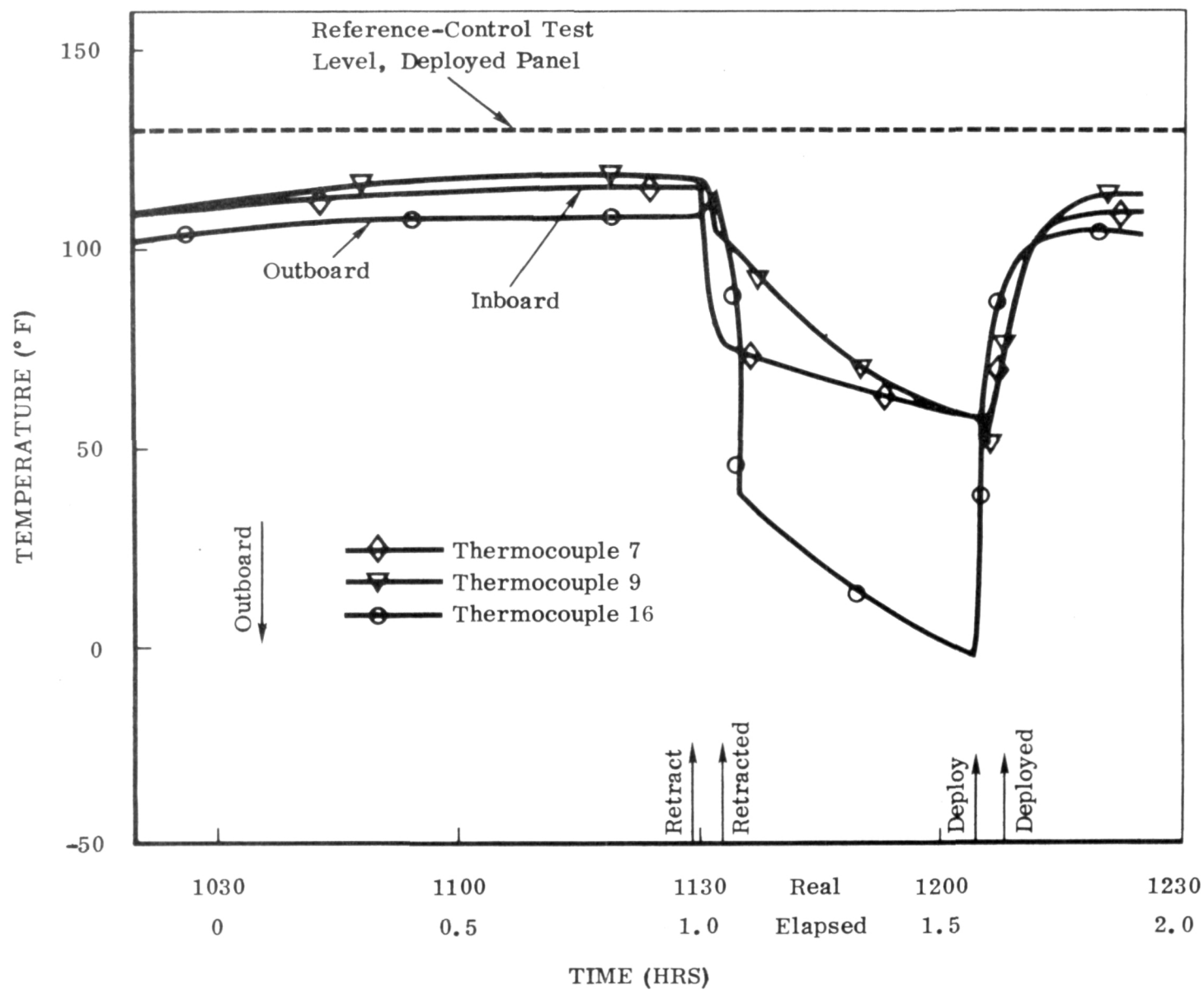


Figure 75 Phase 4 Time-Temp. History of Solar Cells at Panel Edge

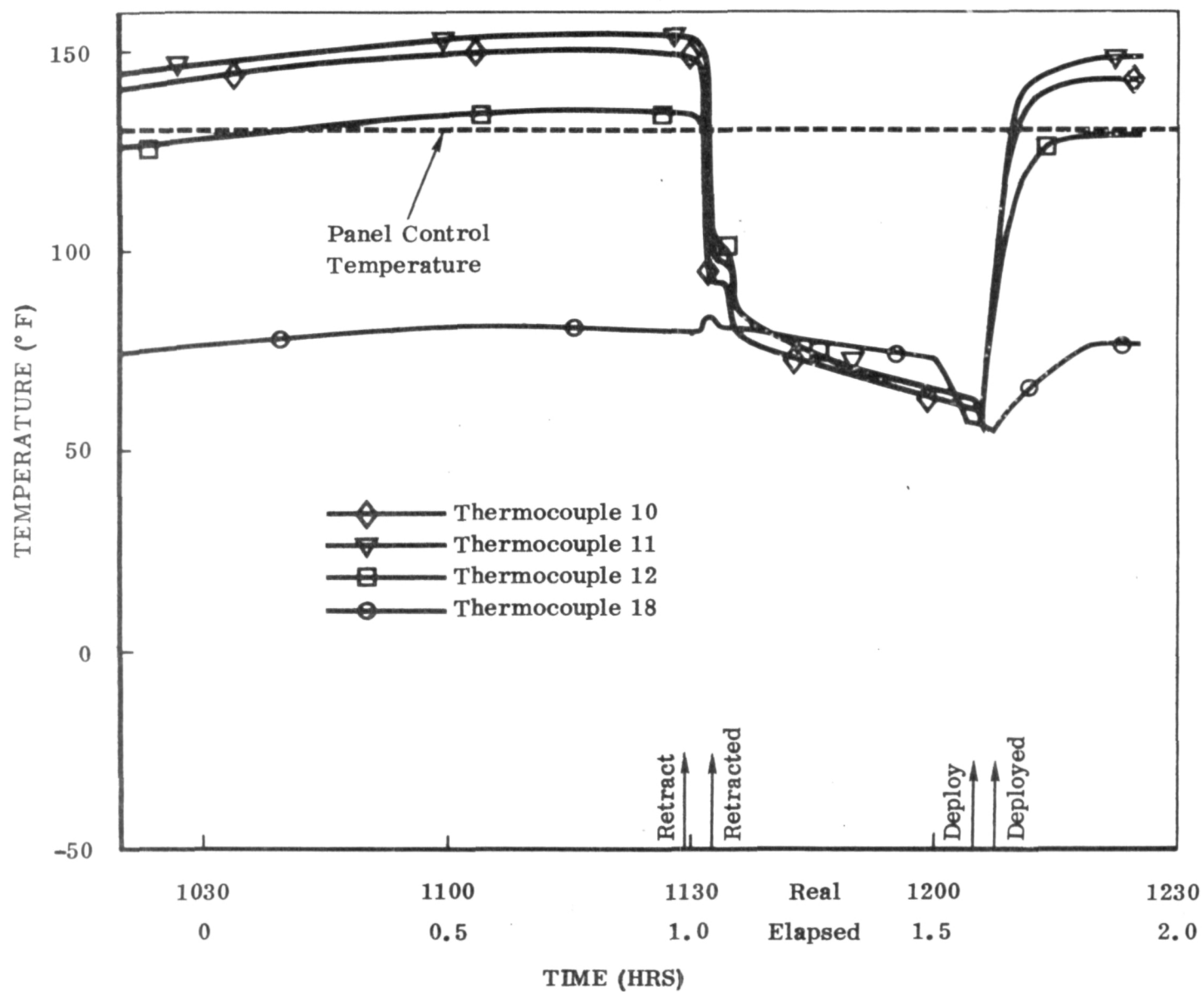


Figure 76 Phase 4 Temperature Locations-Deployment Beam

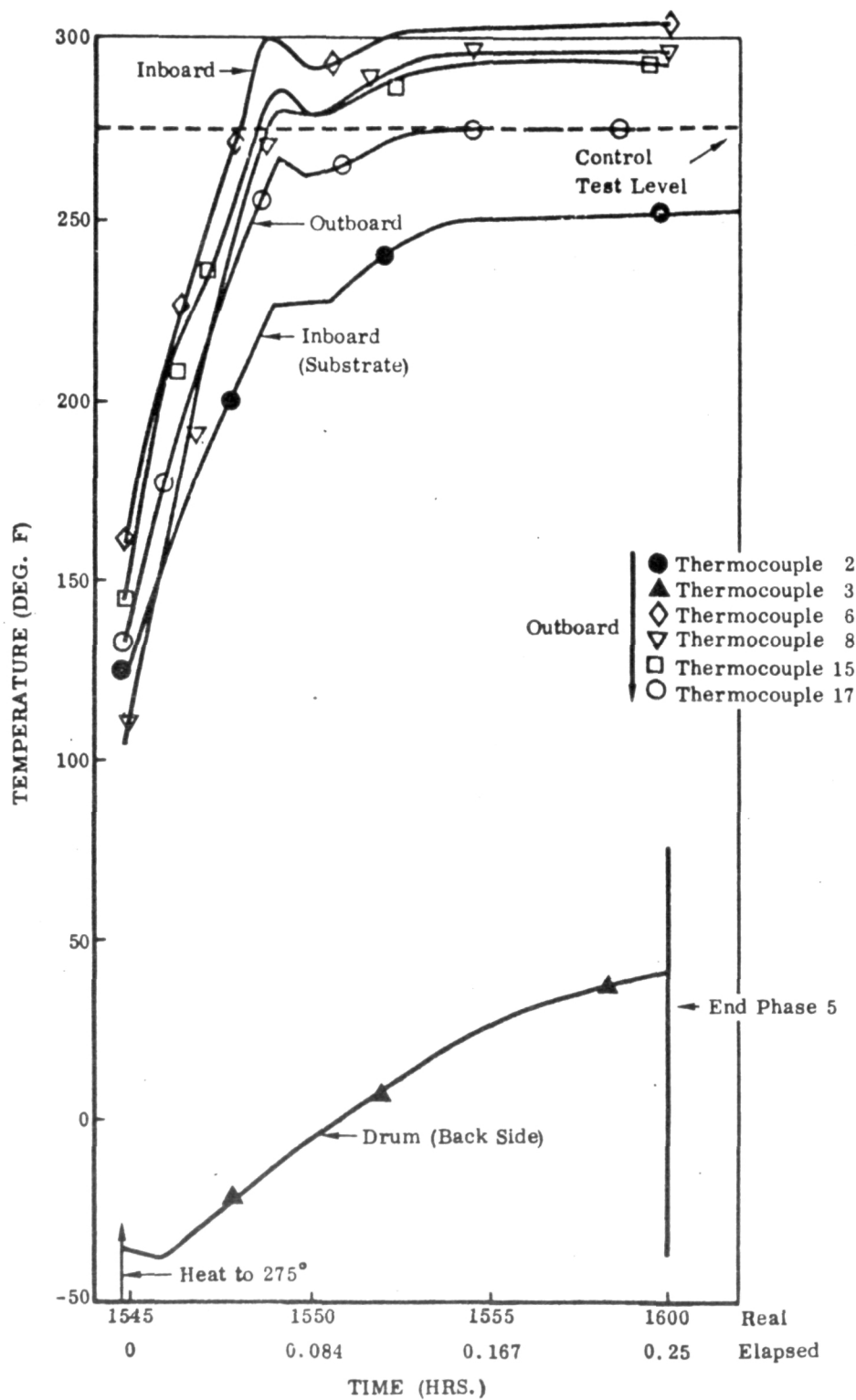


Figure 77 Phase 5 Time-Temp. History of Solar Cells at Panel Centerline

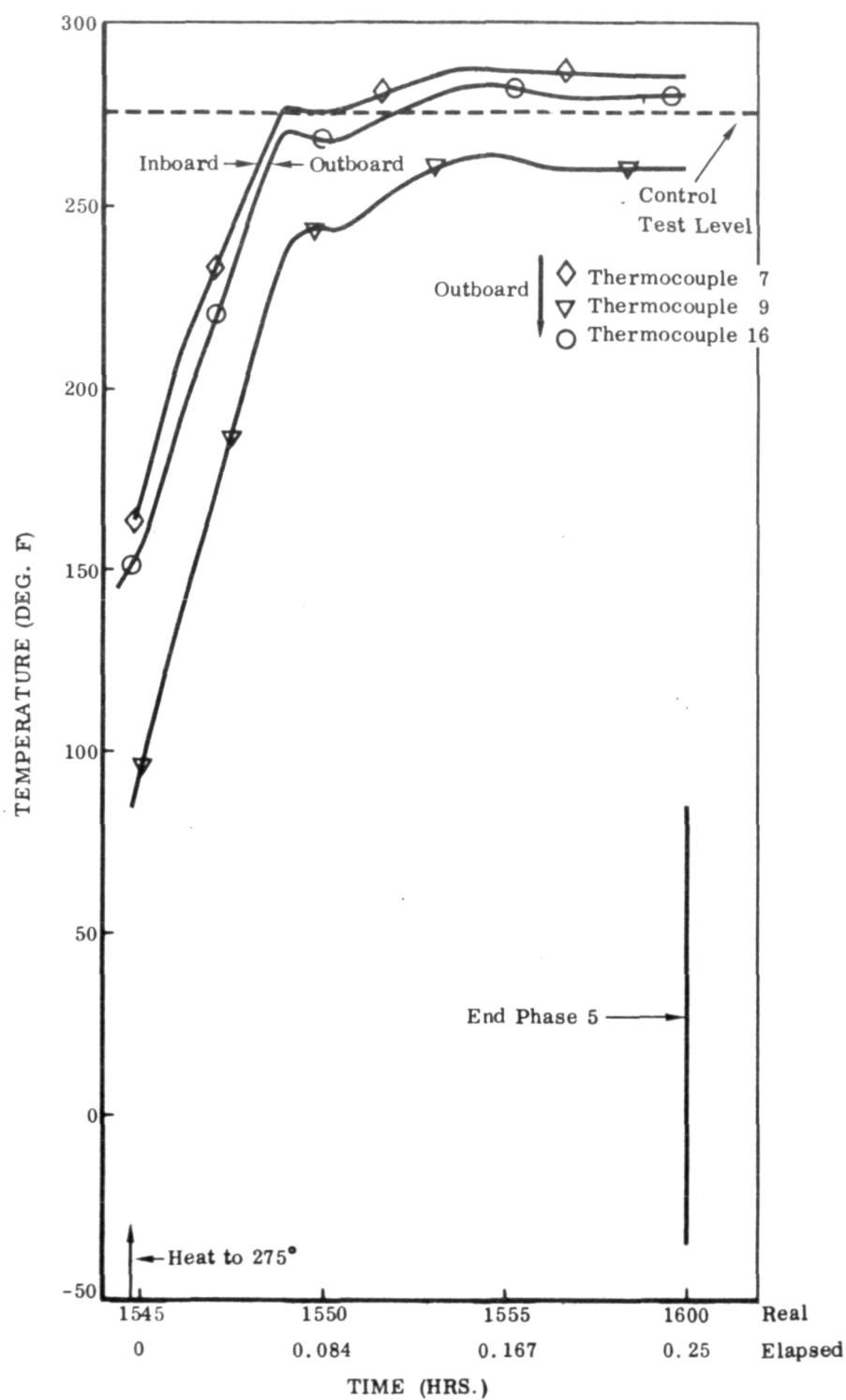


Figure 78 Phase 5 Time-Temp. History of Solar Cells at Panel Edge

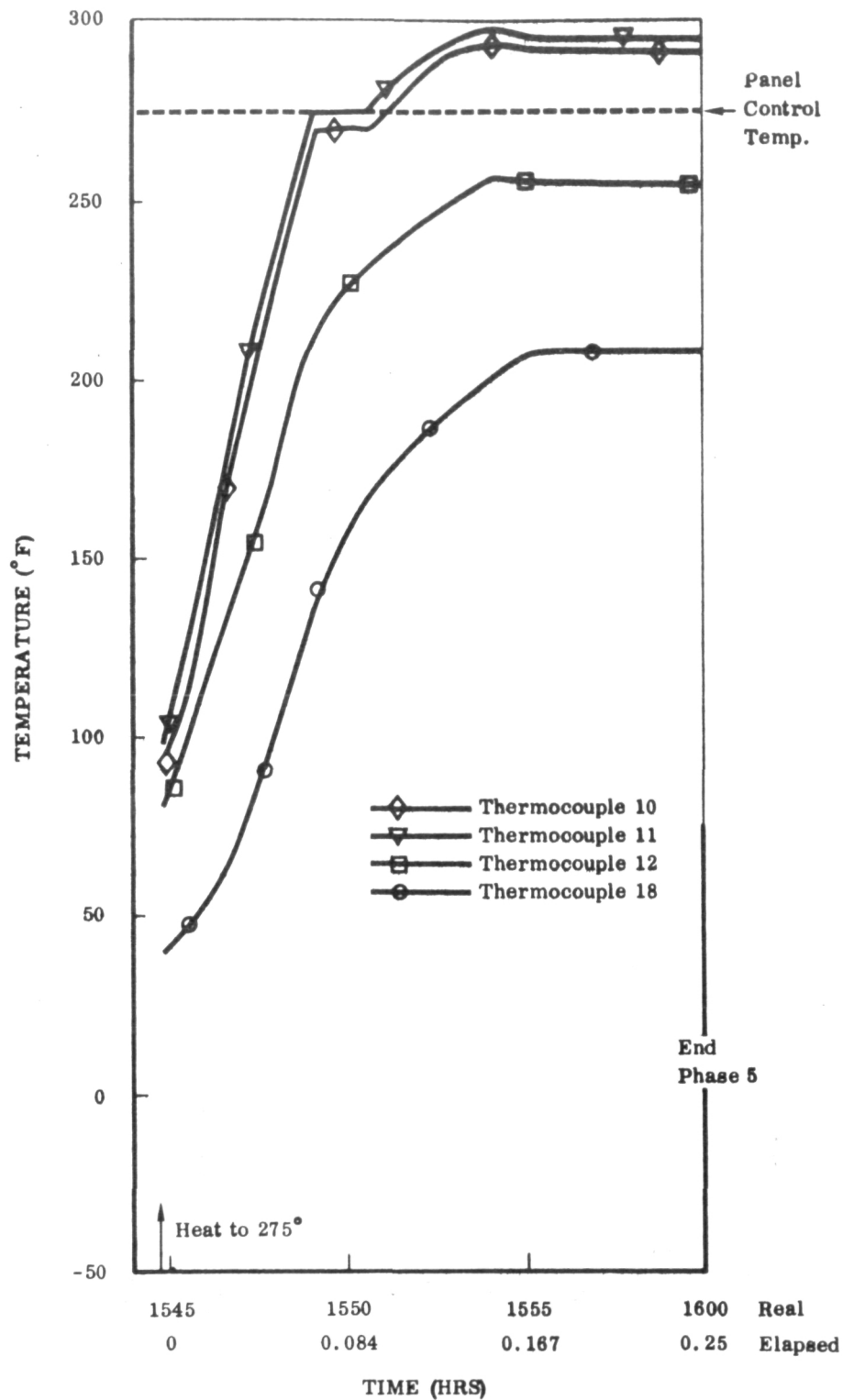


Figure 79 Phase 5 Time-Temp. History of Deployment Beam

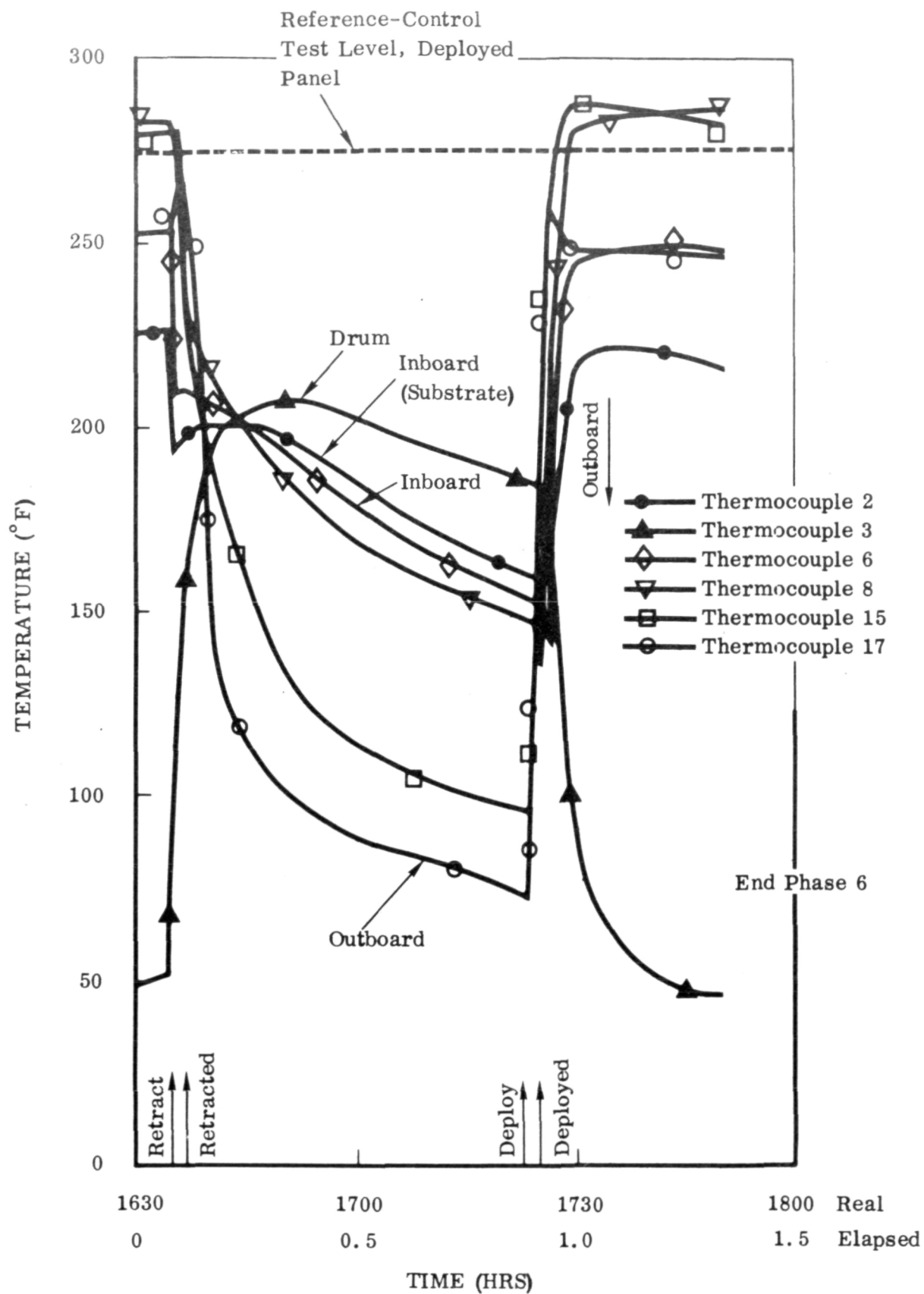


Figure 80 Phase 6 Time-Temp. History of Solar Cells at Panel Centerline

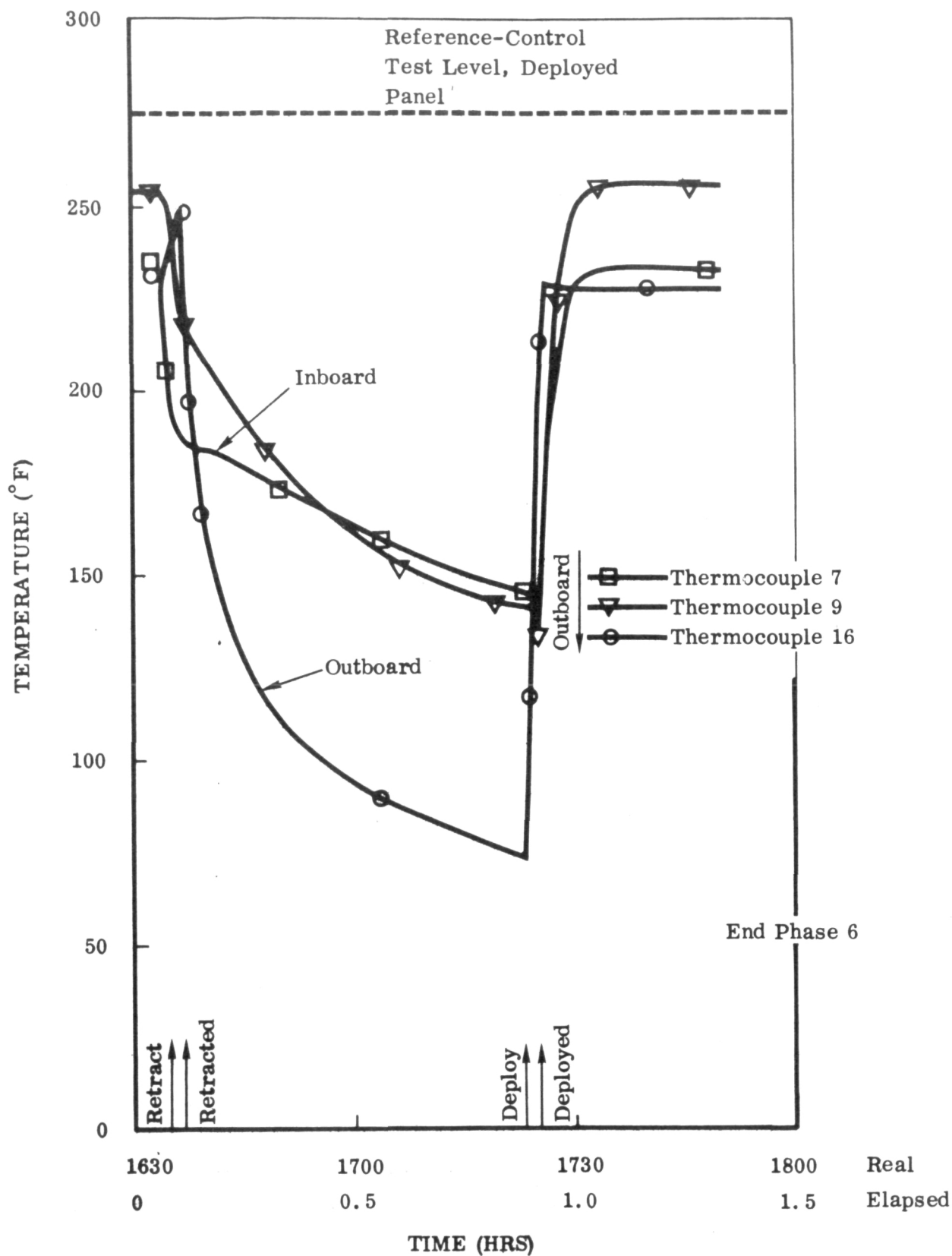


Figure 81 Phase 6 Time-Temp. History of Solar Cells at Panel Edge

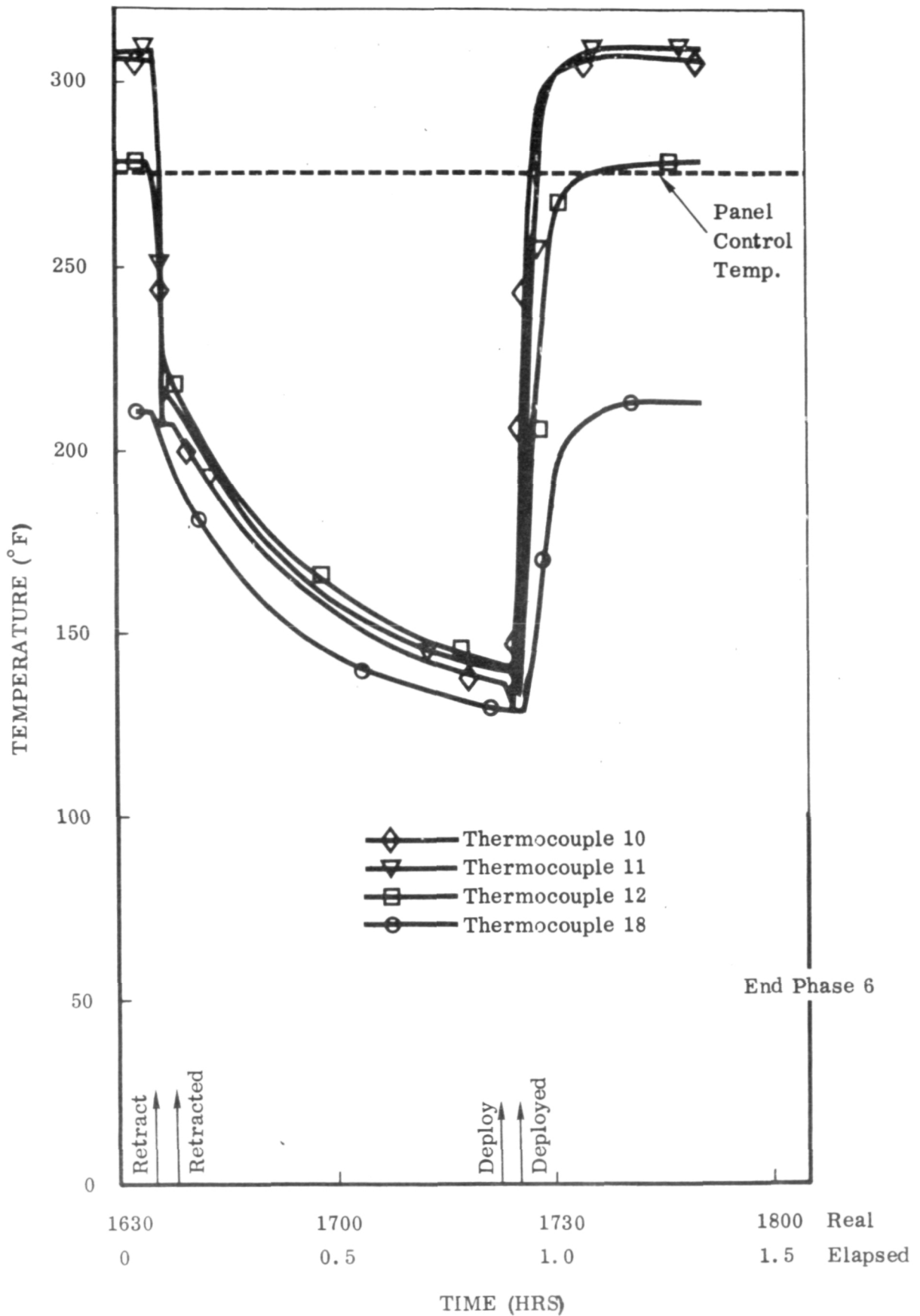


Figure 82 Phase 6 Time-Temp. History of Deployment Beam

The following is of significance related to the test data:

Stowed in Earth Shadow (Phase 1)	Temperature Difference Between Inner and Outer Wrap Solar Cells, Steady at 110°F After 1.5 Hours
Deployed in Earth shadow (Phase 2)	Temperature difference between inboard and out- board panel solar cells, steady at 75°F upon full deployment Drive motor temperature increase 40°F during deployment Negligible temperature difference across panel 30°F steady temperature gradient through deploy- ment beam 5 minutes after full deployment
Deployed in near Earth solar flux (Phase 3)	Outboard panel solar cells 15°F colder than inboard Solar cells at panel edge 20°F colder than at panel centerline 60°F temperature gradient through deployment beam.
Retract, Deploy in near Earth solar flux (Phase 4)	Drive motor temperature increase negligible
Deployed in near Venus solar flux (Phase 5)	Outboard panel solar cells 30°F colder than inboard Solar cells at panel edge 20 to 35°F colder than at panel centerline, with exception of 5° at panel T.P.
Retract, deploy in near Venus solar flux (Phase 6)	Drive motor temperature increase negligible

Temperature and thermal gradients were analytically predicted at near earth deployment (Phase 3) for the solar cells, substrate, beams, edge connectors, and damping pads (Ref. 2). Actual temperatures were consistent with predictions except for the beam temperatures. The upper beam temperature was 160°F with a 60°F gradient to the lower surface versus a prediction of 210°F with a gradient of 75°F. This difference is attributed to edge effects and to the lateral variation in heat flux provided by the quartz lamps which dropped 20°F from center to edge of panel. Solar cell temperature prediction was 131°F with a gradient through the substrate of 2°F. Actual gradient was 1°F. Edge connector temperature was 160°F versus a predicted temperature of 140°F. The temperature of the damping pad rear surface predicted at 80°F was not measured.

A small heat lamp, independently controlled, was located near the drive motor to provide heat if desired. The lamp was not used during the first test in order to observe the capability of the unheated motor. During deployment in earth shadow the motor temperature increased from -100°F to -60°F. In Phase 3 heating the temperature increased to -55°F. In the second test, the motor was heated to improve performance. The motor was operated at 98°F during Phase 5 deployment with negligible temperature change. Motor temperature for Phase 6 deployment was 128°F. Phase 2 deployment temperature was 32°F in the second test.

Figures 83, 84, 85, and 86 show the effect of temperature on motor current requirements. The deployment motor current at room temperature condition is compared to that which was required to deploy and retract the panel (27 volts dc) while the control temperature levels of -150°F, +130°F, and +275°F (Phases 2, 4, and 6 respectively).

4.4.6 Cold Deployment Evaluation

An evaluation study was conducted, consisting of application of both analytical and test methods, to determine the reason for the improper deployment and resulting damage to the solar panel when deployed at -150°F, as explained in Section 4.4.5 (earth shadow simulation deployment, test phase 2).

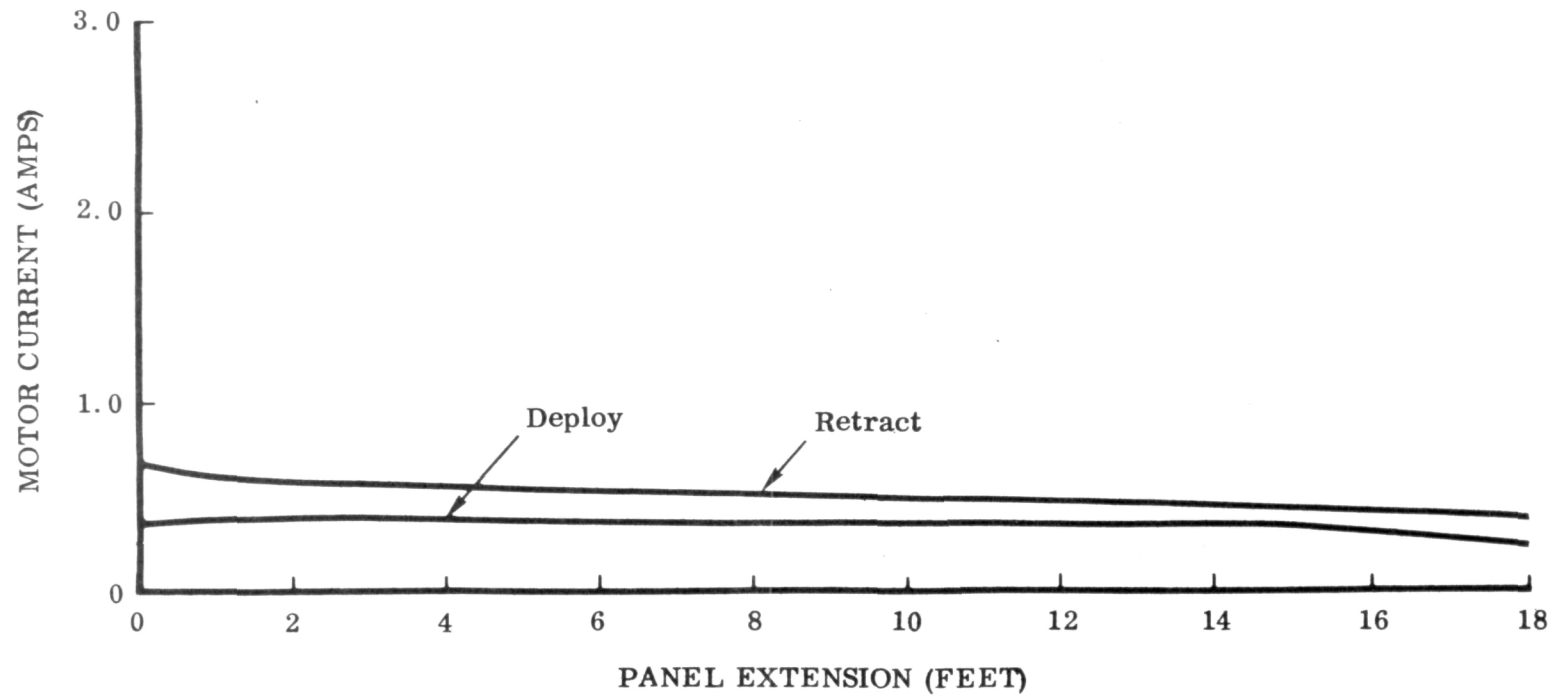


Figure 83 Deployment Motor Current, Ambient Room Temperature Conditions

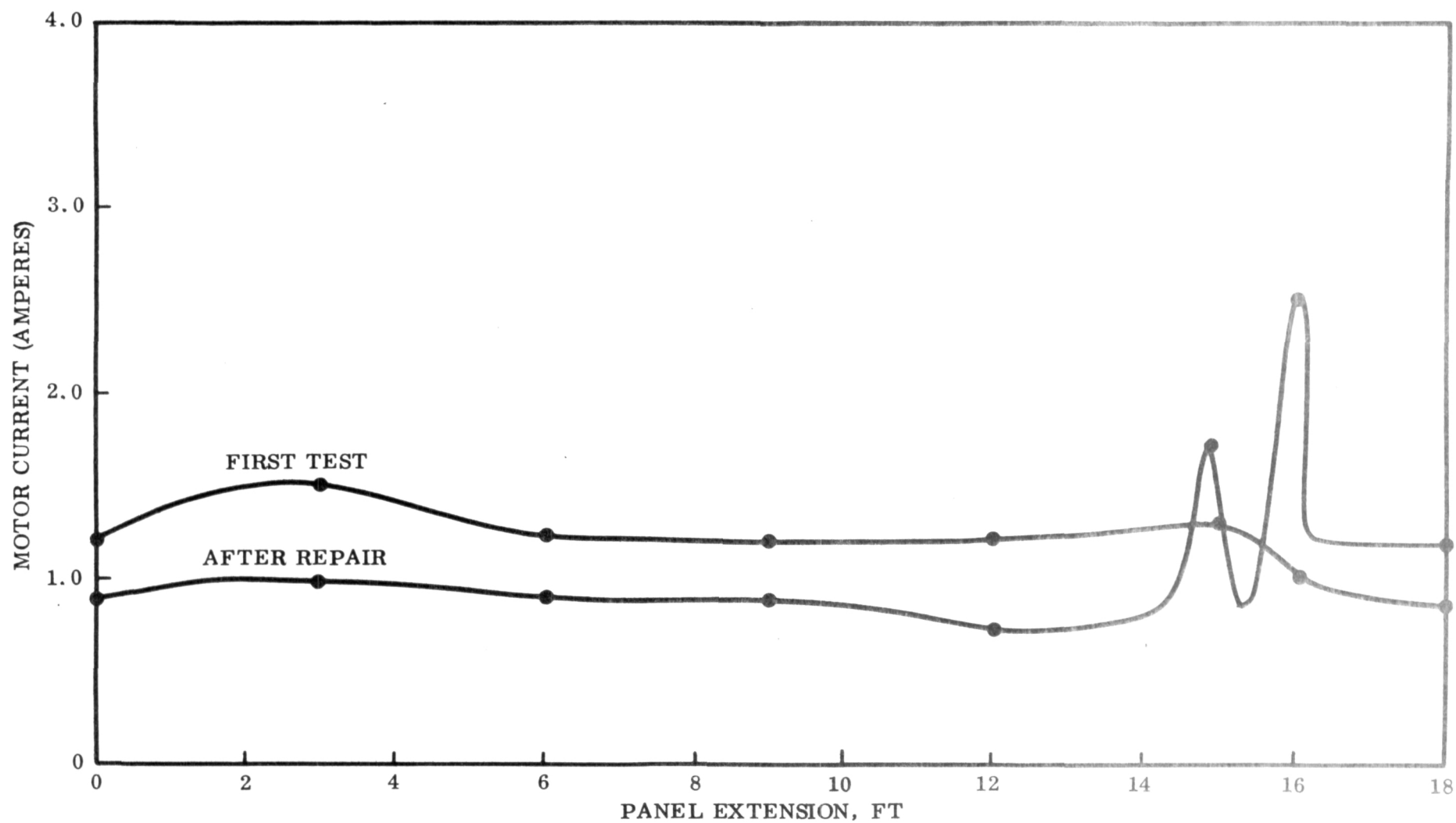


Figure 84 Deployment Motor Current, Phase 2 of Test Cycle

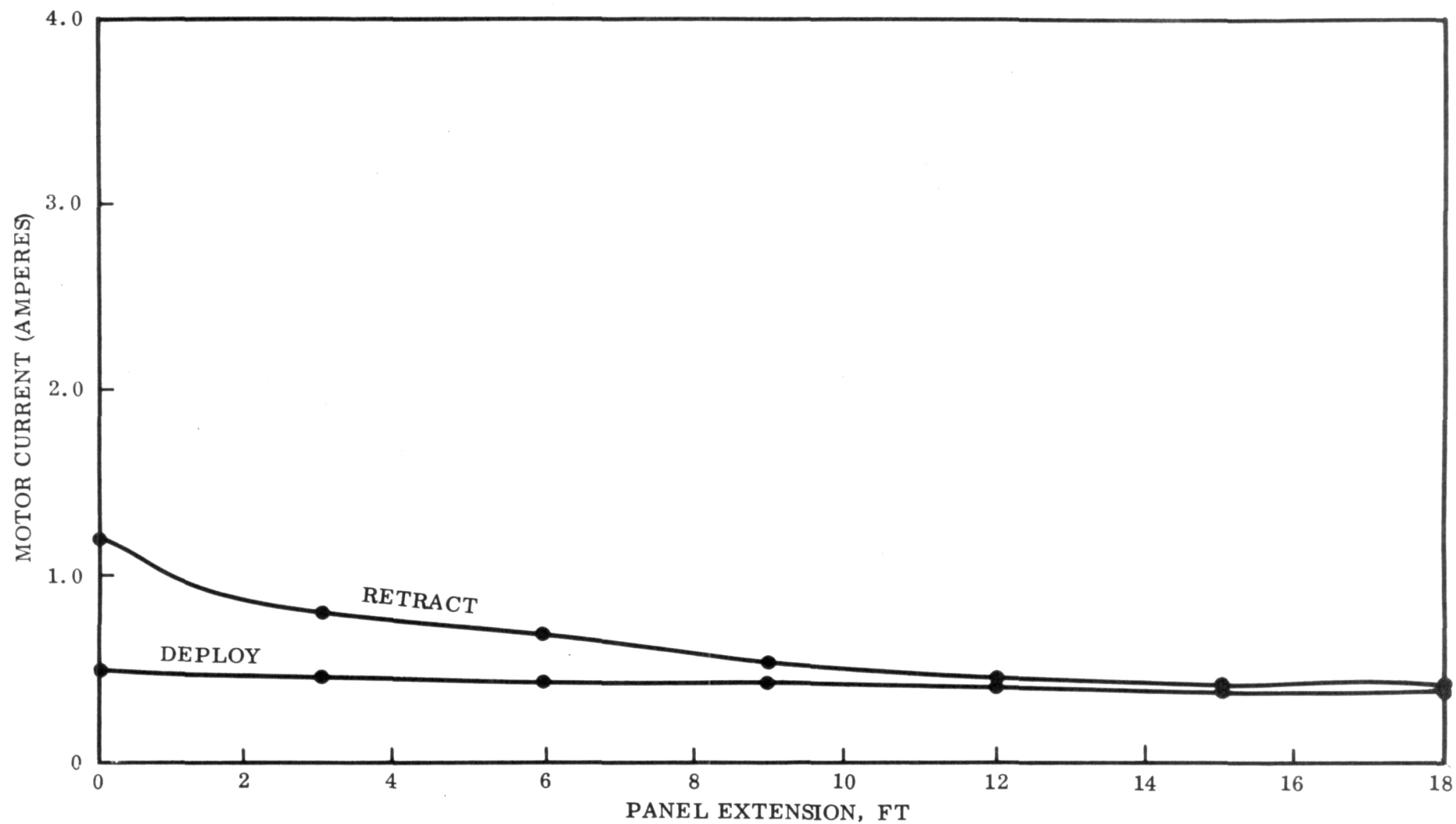


Figure 85 Deployment Motor Current, Phase 4 of Test Cycle (After Repair)

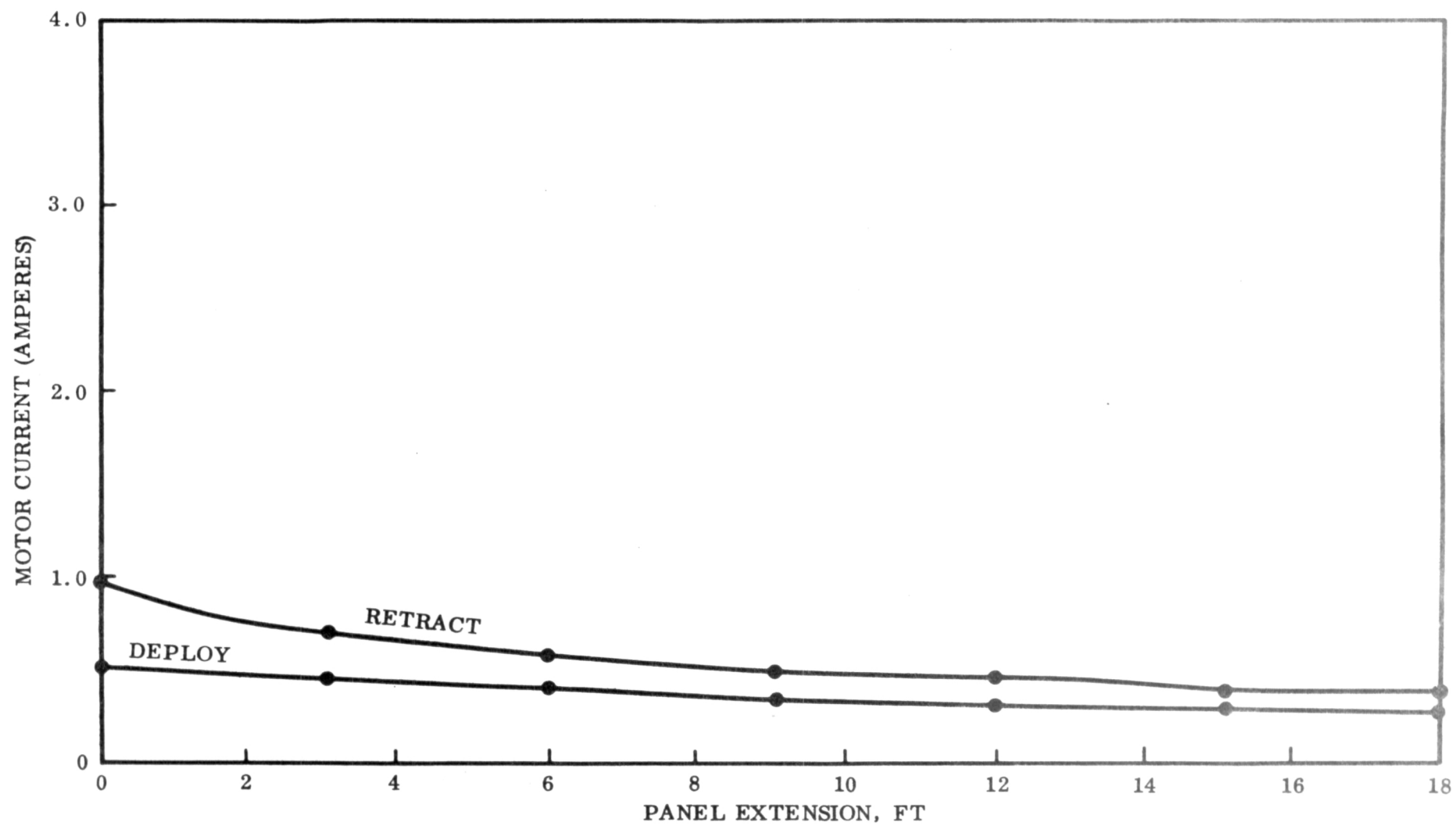


Figure 86 Deployment Motor Current, Phase 6 of Test Cycle (After Repair)

Preliminary tests and calculations were made as follows:

1. The thermal coefficient of expansion of the substrate with bonded dummy aluminum chips was determined by test.

$$\text{Thermal coefficient of expansion} = 15.34 \times 10^{-6} \text{ in/in/}^{\circ}\text{F}$$

2. The temperature of the cross tie in the chamber was determined by analysis since its temperature had not been monitored in that test.

By thermal analysis, the tie bar temperature = +35°F when outer wrap of stowed panel is at -150°F after 4.5 hours in earth shadow.

3. The thermal contraction of substrate relative to cross tie, stowed configuration was calculated.

Thermal contraction of the tie bar

$$= (\Delta T) \propto \left(\frac{\text{width}}{2}\right) \alpha = 12.6 \times 10^{-6} \text{ in/in/}^{\circ}\text{F}$$

$$= (75 - 35) \times 12.6 \times 10^{-6} \quad (18.47)$$

$$= (40) \times 12.6 \times 10^{-6} \quad (18.47)$$

$$= 0.0093 \text{ inch relative to } C_L \text{ panel}$$

Thermal contraction of substrate

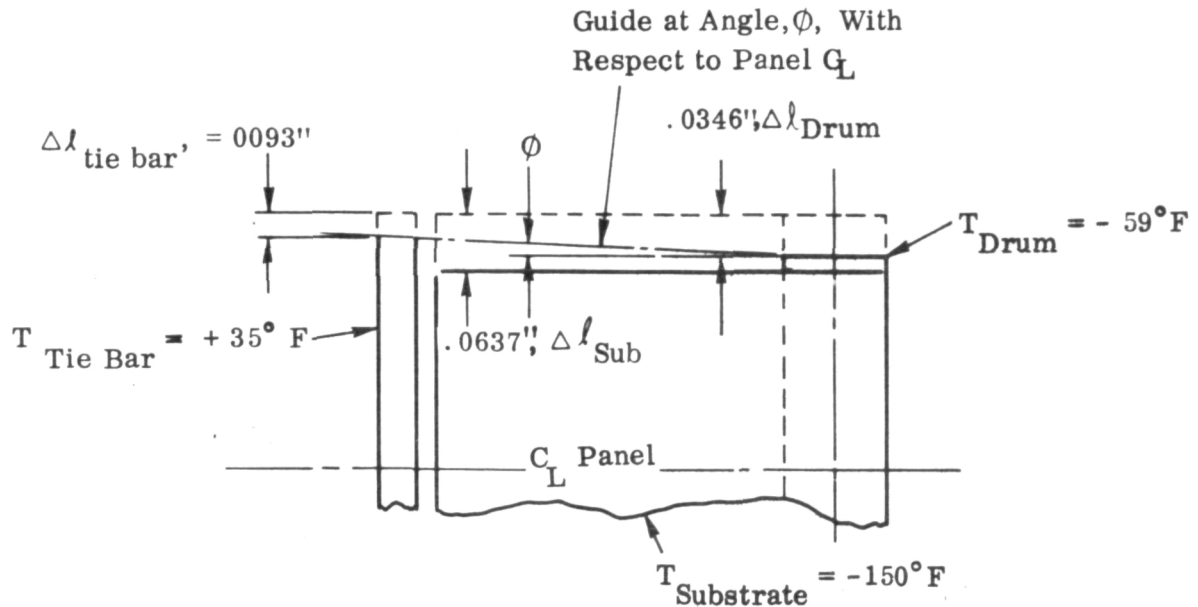
$$= (75 + 150) \times 15.34 \times 10^{-6} \quad (18.47)$$

$$= (225) \times 15.34 \times 10^{-6} \quad (18.47)$$

$$= 0.0637 \text{ inch relative to } C_L \text{ panel}$$

Substrate contraction relative to tie bar

$$= 0.0637 - 0.0093 = \underline{\underline{0.0544}} \text{ inch measured relative to panel } C_L$$



The following steps were taken to determine the reason for first chamber test failure.

The change in deployment motor current was selected as a convenient way of comparing the effect of simulated environment change (friction, cold, thermal load) on the panel in comparison with that during the chamber test.

1. A load was applied to the substrate to simulate relative contraction as calculated above while monitoring deployment motor current.
2. The predetermined load was compared with attach tab joint capacity, to determine if shear-tension failure theory is reasonable:

By static load-deflection test (load applied to substrate with spring scale) relative substrate contraction accumulated at the substrate edge corresponds to a load of 48 pounds. Reducing this to a linear load/inch, using the guide length as the area over which restraint occurs, the tension load in the substrate-to-beam attach clips becomes

$\frac{48}{8.5} = 5.65$ pounds/inch, which is considerably less than that required (14.3 pounds/inch at room temperature, ref. Table 11-1, Ryan report No. 20869-2, 5 May 1966) to fail the aluminum attach clips if loaded uniformly. Therefore the other possibility for failing the clips was either

- a. That the clip tabs were weakened by assembly and their allowable was reduced considerably

or

- b. That a shear load was present between the substrate and beams.

Assuming that condition (b), above, were the case, the load would have had to be something slightly less than 6 pounds shear/inch (determined by test, see Table 11-1, Ryan Report No. 20869-2) to allow for the effect of the already present tension of 5.65 pounds/inch. To create a shear of, say 6 pounds/inch, along the guide length, the relative beam displacement necessary is calculated as,

$$\text{total strain} = 2\left[\frac{(\text{stress})}{G}\right] \times \text{guide length}$$

and assuming that the strain is simply a function of the 0.012 doubled fiberglass (since the 0.003 basic fiberglass is stabilized with solar cell chips),

$$\text{Total Strain} = 2 \frac{\left(\frac{6}{0.012}\right)}{1 \times 10^6} \times 8.5 = 0.009''$$

This relative beam motion is small and serves to show that only a small beam displacement relative to the other (beam travel miscoordination), in addition to the tension load, is necessary to pull open the aluminum clip attach tabs.

3. The above effects were integrated on total motor current for comparison with that with thermal vacuum test:

To verify what theoretically happened, an examination of deployment motor current requirements versus laterally applied load was made. The test was made by pulling on the substrate at the guide of the right hand beam (with a

spring scale taped to the substrate) only which still had the clips attached. To get the integrated effects of load, due to substrate contraction at both beams, we can add the difference in current required before and after load,

Motor current under lateral load

$$\begin{aligned} &= \text{motor current (at maximum load, 1 beam)} + \\ &\quad [\text{motor current (at maximum load)} - \text{motor current (no load)}] \\ &= 0.5 + [0.5 - 0.3] = 0.5 + 0.2 \\ &= 0.7 \text{ amps} \end{aligned}$$

This current is still less than that which developed during the first chamber test, therefore the effects of the low temperature on (1) the motor itself and (2) the coefficient of friction of the teflon guide must be considered.

4. A test was performed using dry-ice to determine the effect of low temperature on motor current:

The effect of temperature reduction during the chamber test (room temperature to -97°F) on the motor was an additional 0.4 amps ($0.75 - 0.35$) motor current draw, found by actuation of the panel with the motor enshrouded by a cold box of dry ice (see Figure 87). This then raised the motor current required to $0.7 + 0.4 = 1.1$ amps.

5. Consideration was given to the effect (on motor current) of teflon guide coefficient of friction increase at lower temperatures, assuming a coefficient of friction at -150°F of approximately twice the value at 75°F (Ref. 9):

If we add 0.4 amps (doubling the friction of beams in the guides) to this to account for change in friction coefficient of the teflon guide under a load developed due to substrate thermal contraction, the motor current becomes $1.1 + 0.4 = 1.5$ amps, which approximates the average of 1.3 amps developed during the vacuum chamber test (giving confidence in theory that tension-

shear induced load due to relative thermal contraction was the primary reason for first chamber test failure).

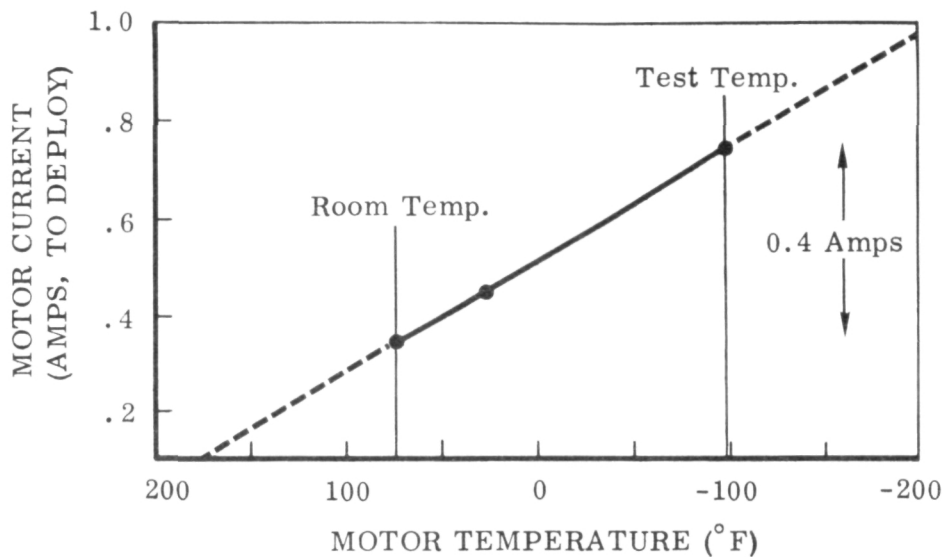


Figure 87 Effect of Motor Temperature on Motor Current

Since the above evaluation and tests showed presence of shear, as well as tension to fail clips, it was realized that the first type failure would be difficult to induce with the in-plane panel capabilities greatly reduced in shear capacity (one side with clips, other with fiberglass cloth). The prediction was proven correct when the beams buckled, in the evaluation testing, at the drum in a manner similar to that for the second test in the chamber. The evaluation test procedure was as follows to determine reason for the second chamber test failure:

- Load was applied to deployment beams at the guide sleeve by loosening the cross tie at one end and spreading the guide sleeves apart by an amount (0.11 inch) calculated to be equal to the relative thermal contraction of the substrate and cross tie. The force to spread was measured by spreading with a spring load scale (see Figures 88 and 89).

Motor current to deploy is shown in Figure 90 compared to that with no load. The panel deployed up to 75% of full deployment, then the beams buckled at the drum, simulating the type of failure that occurred during the second chamber

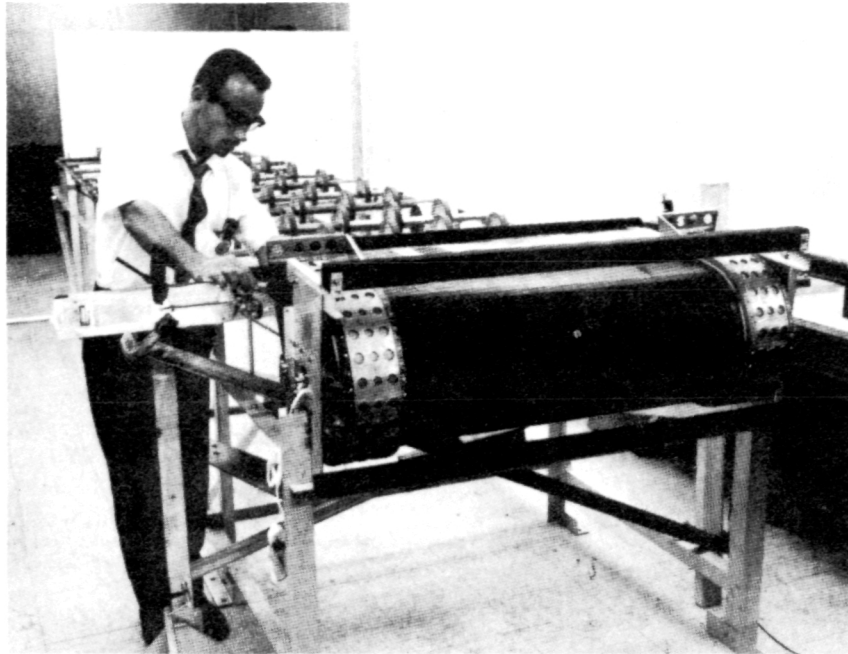


Figure 88 Lateral Load Test Setup

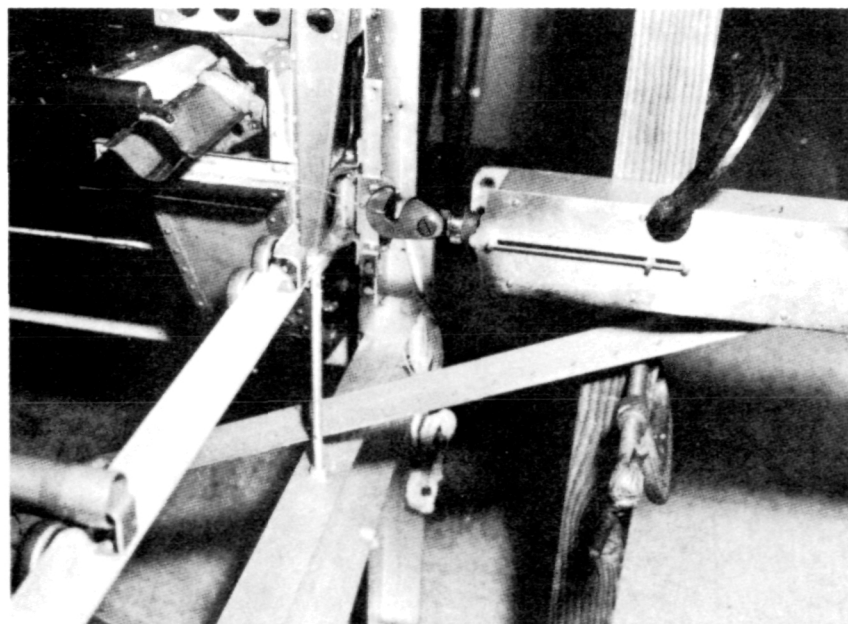


Figure 89 Method for Simulating Thermal Contraction Induced Loads

test. The drive motor current (1.1 amps) compares favorably with that obtained during the second chamber test failure (1.7 amps) if correction is made for an increase in motor current (0.4 amps) due to the effect of the cold temperature on the motor alone and due to an increase in teflon guide coefficient of friction at the cold temperature, $(0.6 - 0.5) \times 2 = 0.2$ amps.

Evaluation Test Current = $1.1 + 0.4 + 0.2 = 1.7$ amps

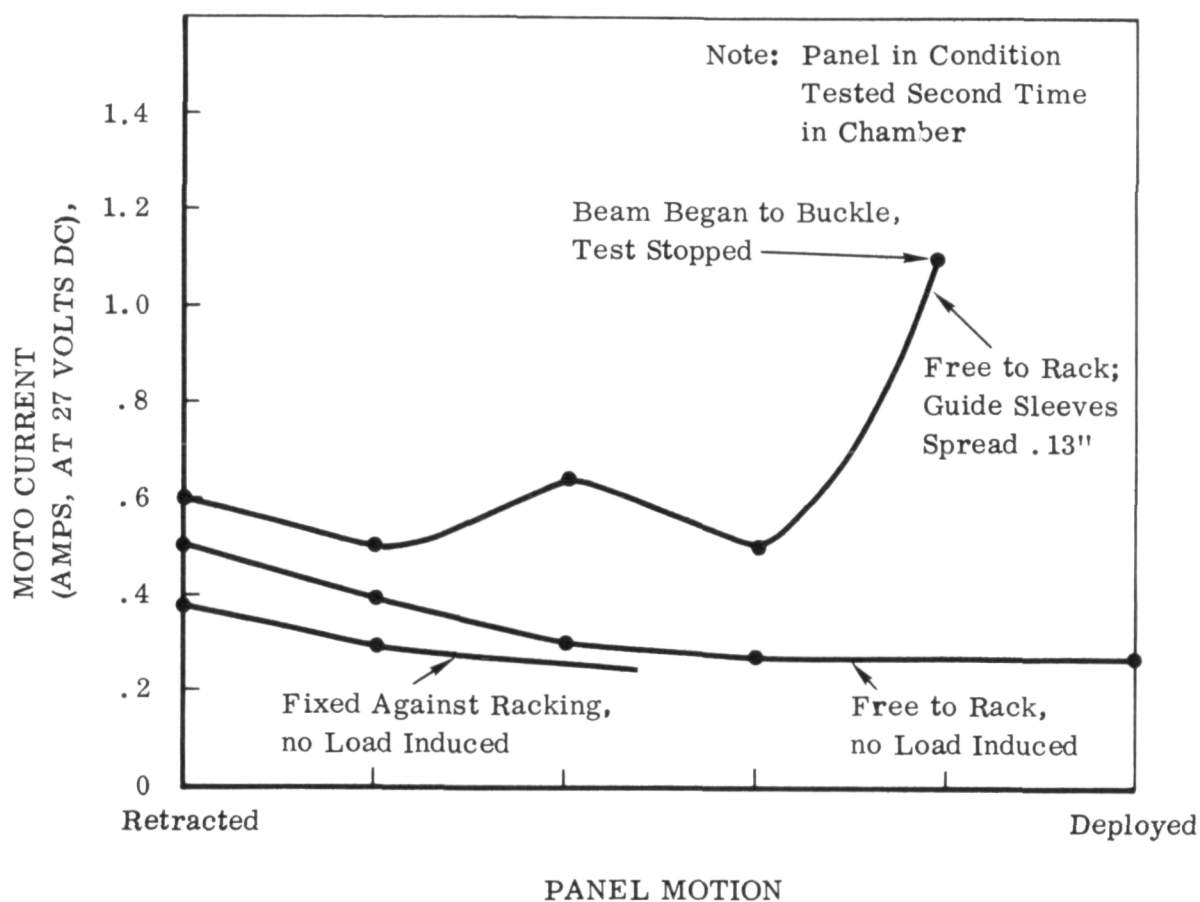


Figure 90 Deployment Motor Current, Evaluation Test

The effect on deployment of loosening the crosstie at one end, as indicated by the deployment motor current, was found negligible. However, it was found that if the guide sleeve mounts (end housings) were fixed (bolted) to the fixture so as to prevent relative rotation (torsion racking) of the guide sleeves as the panel began to deploy, the deployment motor current could be

reduced 25% at the initial deployment stage as shown in Figure 90. As the panel deployed, though, it would not follow the support fixture properly if racking restraint by this method were provided, and since it would be more difficult to compare evaluation test results with that for the chamber test, it was decided to consider racking restraint only as a later refinement to the design.

Preliminary Evaluation Conclusions

It was concluded that the first failure was due to a tension-shear load between substrate and deployment beam attach clips, induced by the thermally contracting outer stowed panel wrap being held by the deployment beams in their guides because of the restraint of the guide sleeve crosstie which did not thermally contract as much as the substrate. Figure 91 shows the crosstie which restrains the guide sleeve, beam and substrate.

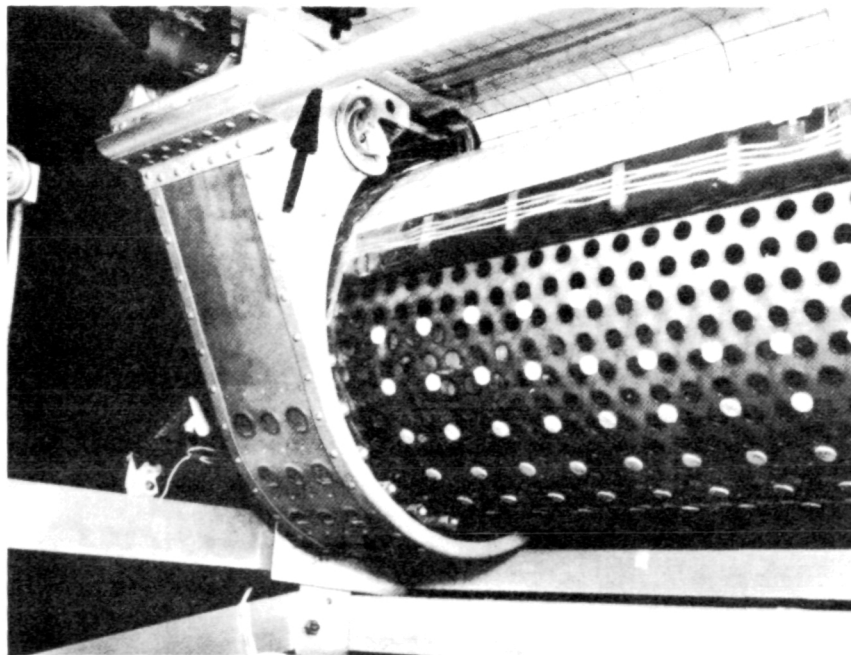


Figure 91 Restraint Area Showing Substrate, Deployment Beam, Guide Sleeve, and Crosstie

The beam buckling type failure which occurred in the second chamber test could not occur in the first chamber test because the substrate thermal contraction induced load which increased friction of beam in its guide sleeve thereby increasing the beam deploy pushing (column) force was relieved as the attach clips failed; this relief prevented the beam deploy push force from building up to the critical level as the panel deployed. It should be noted here that even though intentional thermal contraction provision was made in the repair (replacement of chamber-test damaged-clips with loose fiberglass cloth along one panel edge) prior to the second chamber test lack of wrap coordination of the deployment beams allowed the loose fiberglass to become taut (due to longitudinal translation of one beam relative to the other); this phenomenon was seen during evaluation testing with the beams miscoordinating as much as 0.5-inch relative to each other at full travel. Thermal contraction induced load as before thereby created an excessive beam column load, but the lower shear capacity of the repaired joint did prevent accumulating sufficient shear in the other slip attached substrate edge and for that reason it did not fail in the second chamber test.

Simulation of the second chamber test failure proved that a very small lateral tension force (only 5 pounds) to the beams at the guide sleeves is sufficient to create enough additional friction between guide sleeves and beams to effect a column type buckle of the already partially flattened beams. Since a major design change would be required to effect a more efficient method of holding the stowed beams tightly against the drum to prevent buckling outward from the drum under column load, the following refinements were made to the panel. These refinements were considered the minimum necessary at the time to effect satisfactory deployment of the panel in the -150°F environment. The areas affected are shown in Figure 92.

1. Modified attachments of tip intercostal to deployment beams. This was done to allow the beam tip to vary in cross section as it approached the guide sleeve during retraction or in moving away from the guide sleeves during deployment. With the previous intercostal attachment, the beam cross section at the beam tip was fixed, which could increase friction between the guide and beam as the beam forced itself into or out of this relative position. The previous

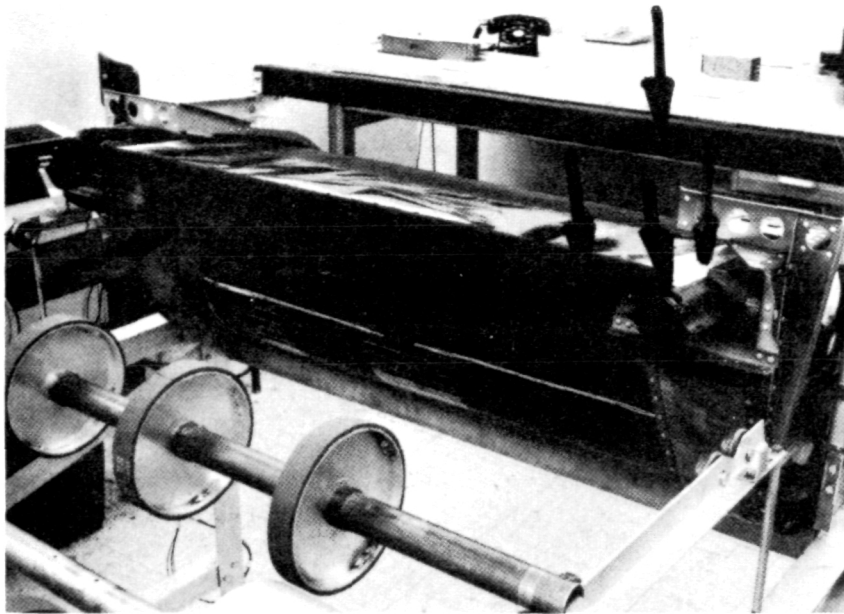


Figure 92 Areas Affected in Refinement of System

tip intercostal was replaced with an aluminum tube to effect this modification in the most convenient manner. Substrate attachment to this tubular intercostal was made using a formed flexible titanium spring (see Figure 93) which will provide for relative thermal expansion along the panel between beams and substrate.

2. Replaced both panel edge-to-beam attachments with fiberglass cloth. Slack in the cloth was provided to allow the substrate to contract thermally relative to the crosstie, up to 0.063-inch at each panel edge (0.13-inch total), before the cloth attachment can become taut. The possibility of an increase in beam-to-guide friction due to thermal contraction is thereby eliminated, provided the beams deploy in coordination.
3. Made adjustments to height of silicone rubber strips which separate stowed beam wraps, thereby reducing deployment beam miscoordination at full travel from 0.5-inch to 0.18-inch. This was done to prevent

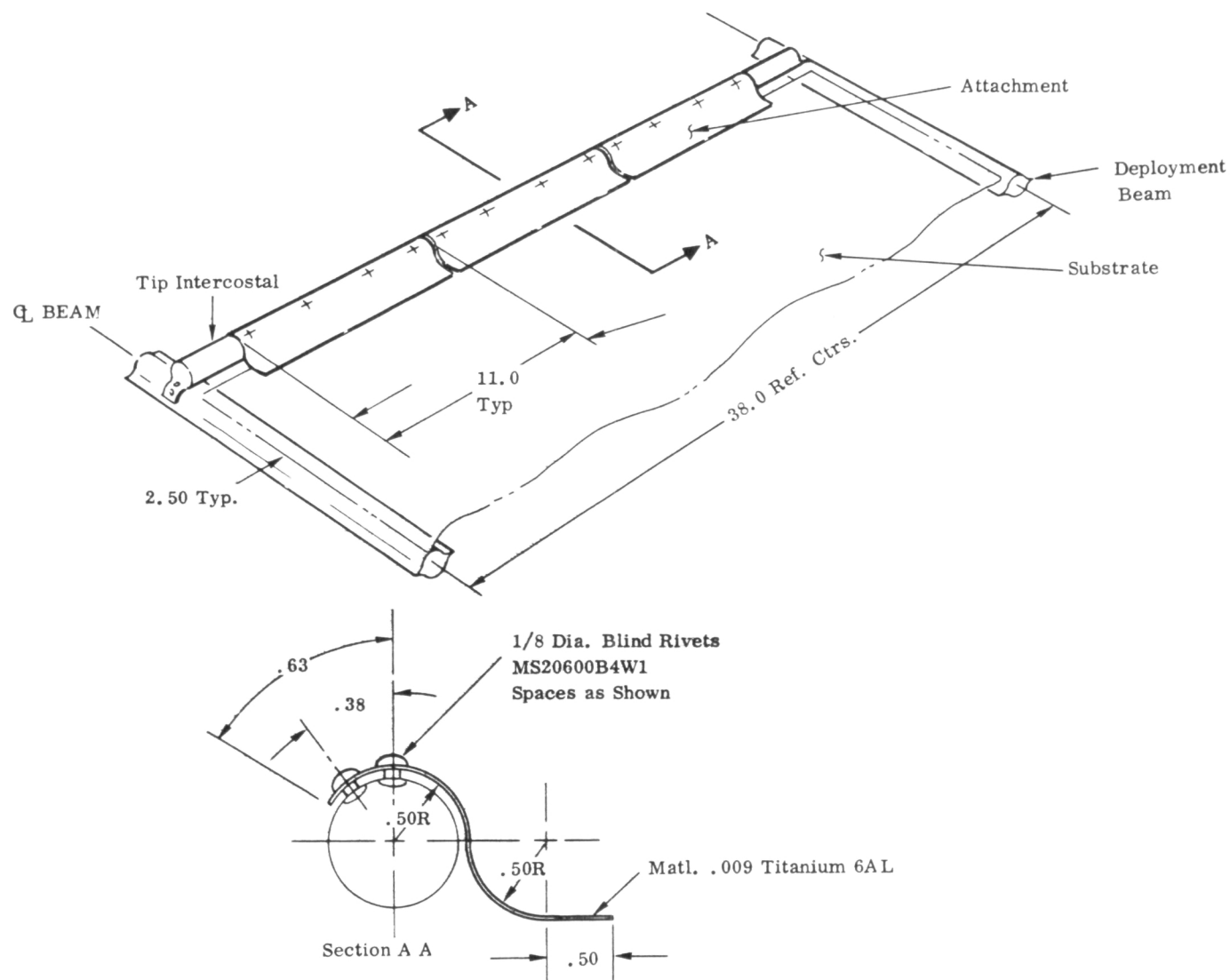


Figure 93 Outboard Substrate Attachment

the fiberglass beam attachment strips from transferring too much in shear, going into field tension, and thereby making it possible to transfer load from the thermally contracting substrate into the beams.

4. Reduced the diameter of the pins which support the guide sleeves, providing a more free condition for translation of the guides in their mounts. This was done to further prevent load being transferred, from the thermally contracting substrate, into the beams.
5. Fixed the support fixture mount plates together with an aluminum crossbar. This was done to prevent racking of the end housings which support the guide sleeves, thereby creating a fixity condition similar to that if the panel were mounted to a spacecraft. As was shown in Figure 90, elimination of racking during initial deployment is advantageous, evidenced by reduction in drive motor current by 25%.

Evaluation Test of Modified Panel

Evaluation of the affect of the panel refinements was made by testing in a manner similar to that previously by deploying and retracting under a simulated substrate thermal contraction condition. This condition was induced by spreading the guide sleeve mounts apart at the crosstie using a spring load scale. The panel deployed satisfactorily with a 0.06-inch spread. The panel deployed fully with the spread at 0.13-inch, but the beams had swelled 3/4-inch off the wrap drum near full panel travel, making it questionable that the panel would deploy satisfactorily with this spread when under vacuum chamber conditions (where possible increase in beam guide friction, due to cold effects on the teflon, are present). The deployment motor current for these two conditions is compared, in Figure 94, with that for the evaluation test prior to making the refinements.

The following conclusions may be drawn from the results of the evaluation tests:

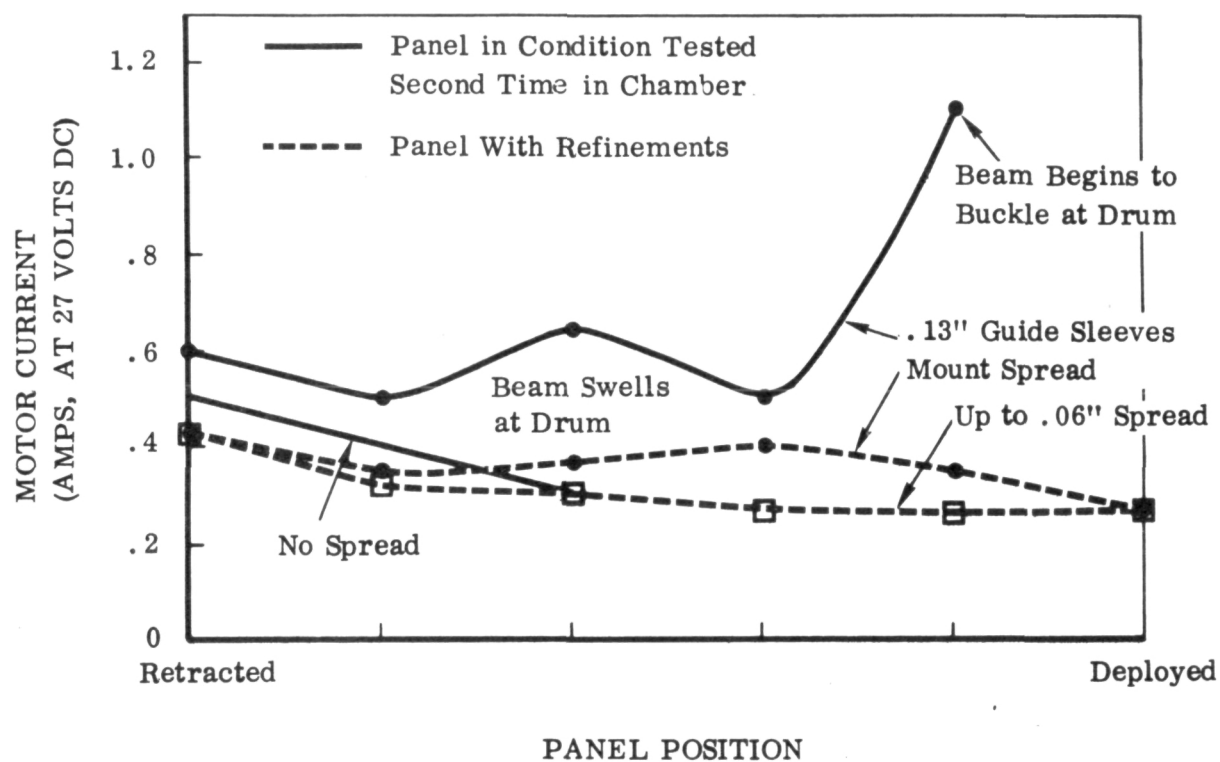


Figure 94 Deployment Motor Current Summary Plot, Evaluation Test

1. The effectiveness of elimination of racking is seen by comparing motor current (guide sleeves in nominal position) for panel before and after refinements (Figures 90 and 94, respectively).
2. Refinements to panel effected a considerable reduction in deployment motor current (see Figure 94).
3. Beam swell does occur with 0.13-inch guide separation on the refined panel due to inadequate roller restraint. Effective thermal control means were employed (crosstie and tip intercostal painted black) to reduce substrate relative thermal contraction to less than 0.06-inch. The difference in motor current reduction is negligible below 0.06-inch spread (see Figure 94); However, an increase in teflon guide friction simply due to cold effects, would increase motor current and might approach the level corresponding to a spread of 0.13-inch causing the beam to swell to the critical buckle point. The actual effect on motor current with a very small guide spread (0.06-inch) or none at all, due to an increase in teflon friction is difficult to determine precisely because of all the variables in the system affecting motor current. Therefore, it is concluded that the refined panel would be expected to deploy satisfactorily in an Earth shadow simulation environment but that deployment is rather marginal without the addition of a deployment beam radial restraint roller directly under each guide sleeve.

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5.0 CONCLUSIONS

This section presents the conclusions reached during the testing program.

The program demonstrated the feasibility of a rollup solar array concept for deploying a large area of solar cells. Concepts demonstrated included:

- . Storage and launch of 8-mil silicon solar cells on a cylindrical drum.
- . Extension - retraction system using welded tubular beams and a d.c. gear motor.
- . A thin fiberglass substrate with flexible silicone foam damper pads.
- . A coiled harness method to conduct power to spacecraft.
- . Fabrication feasibility.
- . Repairability
- . Deployment and testing in a 1.0g field.

The design conforms to the requirements of package volume, number of components, total available area, solar cell geometry, shadowing, orientation, solar cell installation weight, radius of storage drum, rear surface radiation properties, and provisions for pressure equalization.

The general conformance of the design to specification requirements was substantiated. Several design improvements had been previously identified during the fabrication period. These included an increase in restraint roller force to confine the beams on the drum, additional clearance between edge connector and substrate slots to allow self adjustment and prevent substrate buckling, elimination of the guide cam mechanism, and consideration of a sprocket drive system to provide positive control of beam travel. These were found in the test program to be critical areas where design improvement is needed. Also identified during the thermal vacuum test was the need for thermal control coating on the aluminum cross tie.

Conclusions with respect to specific design details follows:

The titanium beams were satisfactory in meeting design and performance

requirements. Beam properties agreed closely with analytical predictions although doubler strips along beam edges were found to be unnecessary. The dust blasted oxide thermal control surface was satisfactory. The beam to intercostal attachment should be redesigned to eliminate fatigue cracks caused by high residual stress. Greater force is required in radial restraint rollers to keep beam tight on storage drum during extension. The silicone rubber spacer strip was satisfactory in all tests.

The wrapping characteristics and coordination of the beams should be improved by an alternate design incorporating positive deployment and retraction using a sprocket drive or similar arrangement.

The beam guides with Teflon liner were adequate for structural support; however, the cam mechanism used to control guide rotation was not satisfactory. Difficulties were due to variations in pressure between the beam spacer strip and the cam drive wheel, combined with lack of access to adjust the cam slip clutch pressure. The entire cam mechanism was shown to be unnecessary when the beam guides functioned satisfactorily with the cams disengaged.

An improved design using a fixed guide should be developed.

The storage drum assembly was satisfactory in all tests. There were no problems with any components including aluminum honeycomb sandwich end bulkheads, the coiled wire harness and terminals, the drum bearings, and the magnesium drum.

The fiberglass plastic substrate assembly was adequate to carry and protect the solar cells in the environments tested without power degradation. Damping pads of silicone rubber foam were satisfactory. The design concept of rigid attachment between beam and substrate is not compatible with reasonable manufacturing tolerances. A redesign using a flexible attachment which will accommodate small variations of beam-substrate position is preferred.

The performance of the deployment motor was satisfactory throughout all the tests. Improved performance at very low temperatures could be obtained by using a dry film lubricant instead of grease in the gear box.

The mount structure was generally satisfactory with properties and performance in close agreement with analytical prediction. An increase in thickness of some areas and additional fasteners are required to eliminate fatigue cracks during launch vibration.

The design concept of storing and deploying 8-mil silicon solar cells on a rollup panel was adequately demonstrated. Handling methods and support fixture configuration had a greater effect on the cells than the solar panel structure itself causing a high incidence of cracked cells. Performance of the solar cells was not significantly affected by the pressure of cracked cells or by the tests performed.

In summary, the test program identified some design features where modifications are desirable to improve conformance of the array to requirements. Where test failures occurred, design or manufacturing corrections have been given which are readily accomplished. The major design improvements include use of a flexible attachment between beams and substrate, a positive deployment method such as a sprocket drive to assure coordination of beams, additional radial restraint spring force, and thermal control coatings on transverse structural members.

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6.0 RECOMMENDATIONS

It is recommended that rollup solar cell arrays be considered for spacecraft power systems requiring a large area (200-square feet or more) of solar cells.

Preliminary designs for specific mission applications should be undertaken to determine detail configuration requirements. Additional development and testing of a beam deployment method using a sprocket drive or similar arrangement to pull the beam from the storage drum is recommended.

Increase of power to weight ratio from 12 watts per pound to about 35 watts per pound should be explored.

A plan should be devised to design, fabricate, qualify for flight and flight test a rollup solar cell array.

7.0 NEW TECHNOLOGY

No items of new technology have been identified in the test program.

8.0 REFERENCES

1. Preliminary Developmental Report for Deployable Large Area Solar Array Structure, Report No. 20869-1, Ryan Aeronautical Company, 30 July 1965.
2. Final Design Report for Deployable Large Area Solar Array Structure, Report No. 20869-2, Ryan Aeronautical Company, 5 May 1966.
3. Final Report for Design and Fabrication of A Deployable Large Area Solar Cell Array Supporting Structure, Report No. 20869-3, Ryan Aeronautical Company, 28 April 1967.
4. Test Plan and Procedures for Phase IV Non-Destructive Testing and Evaluation of the Prototype Deployable Solar Array Structure Manufactured Under Contract 951107, Report No. 20865-1, Revision A. Ryan Aeronautical Company, 6 October 1967.
5. Deployable Large Area Solar Array Testing, Quarterly Report No. 20867-1, Ryan Aeronautical Company, 15 January 1968.
6. Gwinn, Jr., James T., Designing for Random Excitation and Response, Lord Mfg. Company.
7. Design Specification, Flight Equipment, Power Supply System, Solar Panel Structure Specification WP500001-A, Jet Propulsion Laboratory, 13 April 1966.
8. Feasibility Study - 30 watts per pound rollup solar array, Report No. 40067-4, Ryan Aeronautical Company, 21 June 1968.
9. Journal of Teflon, Reprint No. 30, "Property Data at Cryogenic Temperatures" E. I. du Pont de Nemours & Company, 1967

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9.0 APPENDIX

9.1 LIST OF TEST EQUIPMENT

TABLE 6

<u>SINE VIBRATION TEST EQUIPMENT</u>			MODEL OR <u>DESCRIPTION</u>
<u>ITEM</u>	<u>MANUFACTURER</u>		
1. Power Amplifier	Ling Electronics		PP-30/40
2. Accelerometer	Endevco Co.		2226
3. Accelerometer	Endevco Co.		2222
4. Accelerometer	Endevco Co.		2213
5. Oil Film Slip Table	Wyle		WM-450
6. Tape Recorder	Ampex		FR-1300
7. X - Y Plotter	Honeywell		580
8. TRMS Voltmeter	Ballantine		320
9. Charge Amplifier	Unholtz Dickie		CV 608RMG-L
10. VTVM	Hewlett Packard		400H
11. Oscillator	Hewlett Packard		200CD
12. Oscillator	Spectral Dynamics		SD104
13. Multi-Level-Selector	Ling Electronics		SML-100A
14. Ling 275	Ling Electronics		
15. 16 MM Movie Camera			
16. 35 MM Still Camera			
17. Frequency Discriminator	Houston Instrument		HLFC-120

TABLE 7

RANDOM VIBRATION TEST EQUIPMENT

<u>ITEM</u>	<u>MAUFACTURER</u>	<u>MODEL OR DESCRIPTION</u>
1. Power Amplifier	Ling Electronics	PP-30/40
2. Accelerometer	Endevco Co.	2226
3. Accelerometer	Endevco Co.	2222
4. Accelerometer	Endevco Co.	2213
5. Oil Film Slip Table	Wyle	WM-450
6. Random Equalizer	MB Electronics	T-588
7. Tape Recorder	Ampex	FR-1300
8. X - Y Plotters	Honeywell	580
9. Log Converter	Moseley	60D
10. TRMS Voltmeter	Ballantine	320
11. Charge Amplifiers	Unholtz Dickie	CV608RMG-L
12. VTVM	Hewlett Packard	400H
13. Oscillator	Hewlett Packard	200CD
14. Tracking Filter	Spectral Dynamics	SD101A
15. Oscillator	Spectral Dynamics	SD 104
16. 16 MM Movie Camera		
17. 35 MM Still Camera		
18. Strobex - Slip Syne - Sweep Syne	Chadwick Helmuth	201 103A 201A

TABLE 8

DEPLOYED DYNAMICS CHARACTERISTICS TEST EQUIPMENT

<u>ITEM</u>	<u>MANUFACTURER</u>	<u>MODEL OR DESCRIPTION</u>
1. Exciter	MB	EA1500
2. Power Amplifier	MB	2250
3. Function Generator	Weston	PA410
4. Charge Amplifier	Unholtz Dickie	11-MGS-LF-1
5. Accelerometer	Unholtz Dickie	75D21
6. 16 MM Movie Camera		
7. 35 MM Still Camera		

TABLE 9

THERMAL VACUUM TEST EQUIPMENT

<u>ITEM</u>	<u>MANUFACTURER</u>	<u>MODEL OR DESCRIPTION</u>
1. Convair - Thermal Vacuum Chamber	Livermore Lab 11 ft inside dia. by 21 ft long	LN ₂ System for the cold wall is a continuous flow system (-300°F) with .90 or greater surface absorptivity
2. Convair -Data Acquisition System-Printer and Tape	Dymec	DY2010G
3. Electrical Power	R & I Controls, Research Inc.	Three Ignition units controlled through a Leeds and Northrop 354162 Series 60 Control
4. Power Supply, 28 VDC	Hewlett Packard	
5. Visicorder	Honeywell	
6. Brown Recorder	Honeywell	
7. Ion and Thermocouple	Kinney Vacuum Division	Panel Model KITG-3P Ion Gage Range 10 ⁻⁴ to 10 ⁻¹⁰ Torr. Thermocouple Gage Range, Atm. to 0 psia
8. Heat Lamps	G. E.	2,500 watt quartz lamps two rows, 10 inches apart, 10 inches above panel plane
9. Solar Cell Electrical Monitor Lamps	G.E.	Tungsten Flood lamp
10. 16 MM Movie Camera		
11. 35 MM Still Camera		
12. Chamber Pumps		Six 300 CFM Kinney mechanical pumps Two 95000 Liter/sec oil diffusion pumps one ring jet booster

TABLE 10

EQUIPMENT FOR ELECTRICAL CHECK OF CELLS

<u>ITEM</u>	<u>MANUFACTURER</u>	<u>DESCRIPTION</u>
1. Power Supply (dc)	Hewlet Packard	
2. Digital Voltmeter	Non-Linear Systems	481
3. Digital Voltmeter	Electro-Instruments	DVX-400 DXX-010 DXA-000
4. Standard Solar Cell (Secondary)	JPL Furnished	114
5. X - Y Plotter	Electro-Instruments	225
6. Water Pump	Little Giant Pump Co.	2
7. Temperature Con- trolled Water Bath	Blue M. Electric Co.	MR-3210-A
8. Thermo-Meter	Simpson	
9. Temperature Poten- tiometer	Leeds & Northrup Co.	

9.2 PRELIMINARY LAUNCH VIBRATION TESTS

9.2.1 Compactness Evaluation of Stowed Panel Wraps

Experimental Determination of Effective Area to Use in Calculating Spring Rate

a. Point Contact

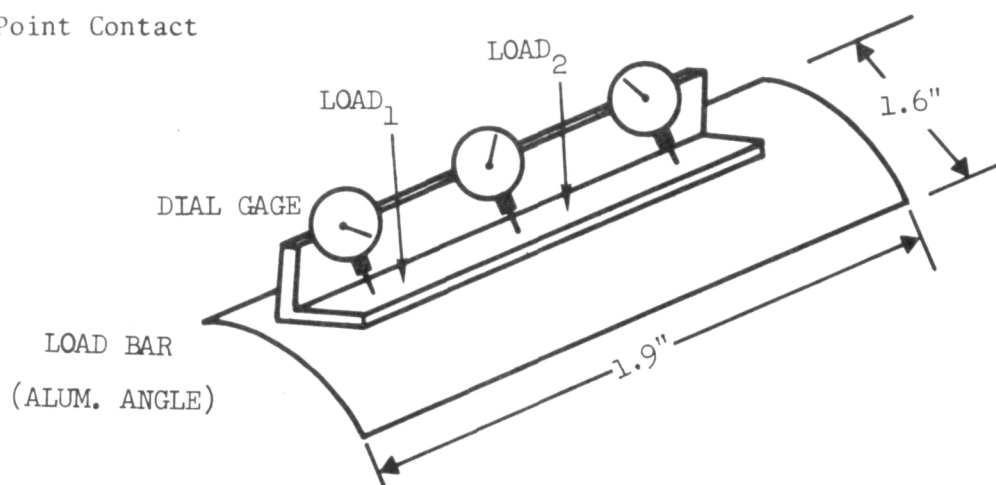


Figure 95. Checking for Load Versus Deflection at Point Contact

$$K_1 = \frac{(\text{Load})}{(\text{Area}) (\quad)} = \frac{(1.5)}{(2.3 \times 1.6) (.12)} = \underline{\underline{3.4}}$$

$$K_2 = \frac{(1.75)}{(2.3 \times 1.6) (.12)} = \underline{\underline{3.95}}$$

$$K_2 = \frac{(2)}{(2.3 \times 1.6) (.106)} = \underline{\underline{5.15}}$$

$$K_{\text{avg.}} = \underline{\underline{4.1}}$$

b. Load Bar

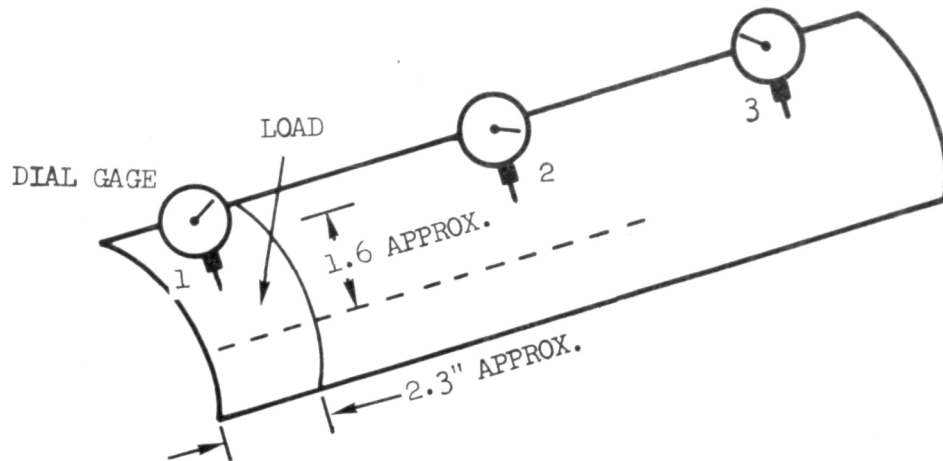


Figure 96. Checking for Load Versus Deflection Through a Load Bar

$$K_{(\text{test a})} = \frac{(\text{Load}_1 + \text{Load}_2)}{(\text{area}) (\text{avg.})}$$

$$K_{(\text{test a})} = \frac{(6)}{(19 \times 1.6) (0.06)} = \underline{\underline{3.3}}$$

$$K_{(b)} = \frac{(6)}{(19 \times 1.6) (0.065)} = \underline{\underline{3.04}}$$

$$K_{(c)} = \frac{(7.25)}{(19 \times 1.6) (0.077)} = \underline{\underline{3.1}}$$

The above spring rates show reasonable consistency for the two methods of load application, namely point contact and load bar. This points to the conclusion that the area assumed effective over which the load is applied is within reasonable accuracy. The test will now proceed by point contact loading to obtain a distribution of spring rate, K, for the various wraps.

Test

Two load deflection measurements using a spring scale) thrust manually against substrate) at each of two dial gages, as shown in Figure 97, were recorded at three circumferential positions (at 120° intervals) of the panel as it wrapped on the drum. The variation in data representing the tightness or looseness of the wrap is given in Table 11. Also given are (1) the respective spring rates, K, calculated using an area of $2.3 \times 1.6 = 3.68 \text{ inch}^2$ found as reasonably effective earlier in the test, (2) the corresponding natural frequency of the panel wrap for test particular area tested, calculated from the equation for sinusoidal motion,

$$f_n = \frac{(K/M)^{1/2}}{2} \quad \text{Where M is based on a solar cell installation weight of .31lbs/ft}$$

(3) the corresponding dynamic deflection of wrapped panel, assuming 80 g rms response at sinusoidal resonance, calculated from

$$= \frac{386.4 \text{ g (response)}}{(2 \pi f_n)^2}$$

and (4) a remark as to whether the vibrating wrap layer will deflect sufficiently to contact the adjacent layer. A safety factor of 2 is desired to allow for possible unconservatism in the response acceleration assumed if the panel-to-spacecraft mount structure is not as rigid as calculated.

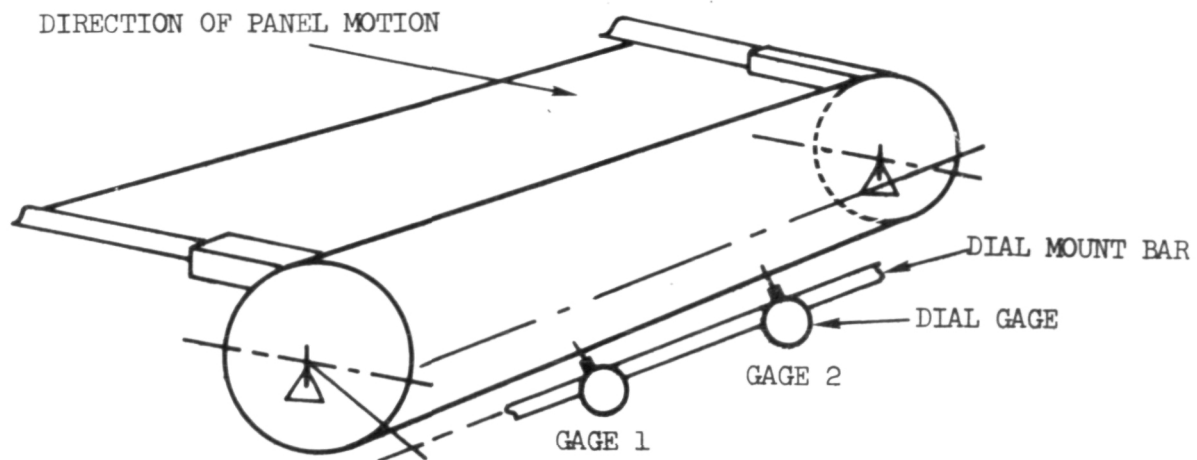


Figure 97 Position of Load Designation

Conclusions of Load Deflection Tests

The spacing between the three outer wraps appears questionable, based on the assumption that the spacecraft mounts are not as effectively rigid (approximately 400 cps) as calculated in Ryan Report No. 20869-2, page 64. If this were the case, and we approach a more nearly dynamic coupled condition between mount and drum, solar cells will impact on adjacent wrap layers in these three outer wraps. It is suggested, at this point in the program, to proceed as planned and cautiously monitor the natural frequencies at low level sinusoidal excitation (drum, mounts, and wraps) before conclusions are drawn to add damper pads sufficiently thick to fill the spacing between wraps which would prevent solar cell impact.

Table 11 Position of Load Deflection

Panel Wrap	Angle of Rotation, Degree, from Start of Wrap			Dial Gage No. 1						Dial Gage No. 2					
				Total Static Defl., In.	Load Lbs.	I psi/in	f _n cps	Dyn. Defl., In.	Excessive Dyn. Defl.	Total Static Defl. In.	Load Lbs.	K psi/in	f _n cps	Dyn. Defl., In.	Excessive Dyn. Defl.
	0	120	140												
Inner															
1		X		.058	2.0	9.4	210	.025	No	.060	2.4	10.9	226	.021	No
			X	.067	2.3	9.3	209	.025	No	.055	3.0	14.8	263	.016	No
2	X			.080	3.4	11.6	234	.02	No	.076	3.3	11.8	235	.02	No
		X		.090	2.5	7.6	189	.03	No	.120	3.0	6.8	179	.035	No
			X	.132	2.0	4.1	139	.055	No	.135	2.5	5.0	153	.05	No
3	X			.086	1.6	5.1	155	.045	No	.060	1.8	8.2	196	.03	No
		X		.100	2.0	5.4	159	.043	No	.086	2.3	7.3	185	.033	No
			X	.172	2.0	3.2	123	.07	No	.154	2.0	3.5	128	.07	No
4	X			.185	1.8	2.6	110	.09	No	.105	2.0	5.2	156	.045	No
			X	.120	1.0	2.3	104	.10	?	.140	1.0	1.9	94	.125	?
5	X			.180	1.3	2.0	97	.12	?	.125	1.0	2.2	102	.11	?
		X		.170	1.5	2.4	106	.10	?	.190	1.5	2.2	102	.11	?
Outer															
6	X			.270	2.0	2.0	97	.12	No	.240	1.8	2.0	97	.12	No
		X		.170	1.5	2.4	106	.10	?	.180	1.5	2.3	104	.10	?
			X	.160	1.3	2.2	102	.11	?	.160	1.5	2.6	110	.09	?

9.2.2 Sinusoidal Vibration Impact Simulation Test of Stowed Panel Wraps

A check was made of the spacing between subsequent wrapped layers of panel while stowed on the drum (see Section 9.2.1). The panel does not wrap with such precision that the damper pads of one wrap rest on the solar cells of the adjacent wrap, resulting in a compact stowed panel. When vibration excited the wrapped layers could possibly impact against the adjacent layers, thereby jeopardizing the electrical integrity of the solar cell installation. It was found that spacings varied from .06 in. at the inner layers to .18 in. at the outer layers. A natural frequency calculation of the stowed wraps indicated that each of the outer three wraps would impact against adjacent wraps when at sinusoidal resonance.

Now that it was almost certain that impact of the wrapped panel layers against adjacent layers would occur at sinusoidal vibration resonance, a small sample was vibration excited at resonance to determine if the solar cell installation was in jeopardy. A 9" x 11" sample panel supplied by JPL with a wired solar cell installation of .3 lbs/ft², was attached at its ends to a rigid aluminum fixture which simulated the wrap drum (see Figure 98). This fixture was mounted to a vibration exciter with a 1/8 inch spacing between the damper pads (on the under side of the specimen) and the drum simulated fixture, the specimen was excited at resonance after its first resonance was found. Excitation levels were increased in increments until the specimen was driven hard against the fixture. No damage occurred to the solar cells under test in which levels of 80 g (0-peak) response acceleration were reached. This was about the maximum response expected on the wrapped panel and therefore sufficient confidence that impact would not damage the solar cell installation forced suggestion to proceed with the sinusoidal vibration test of the rollup panel assembly.

Figure 99 shows a plot of the effects on resonant frequency and response acceleration as excitation accelerations were increased.

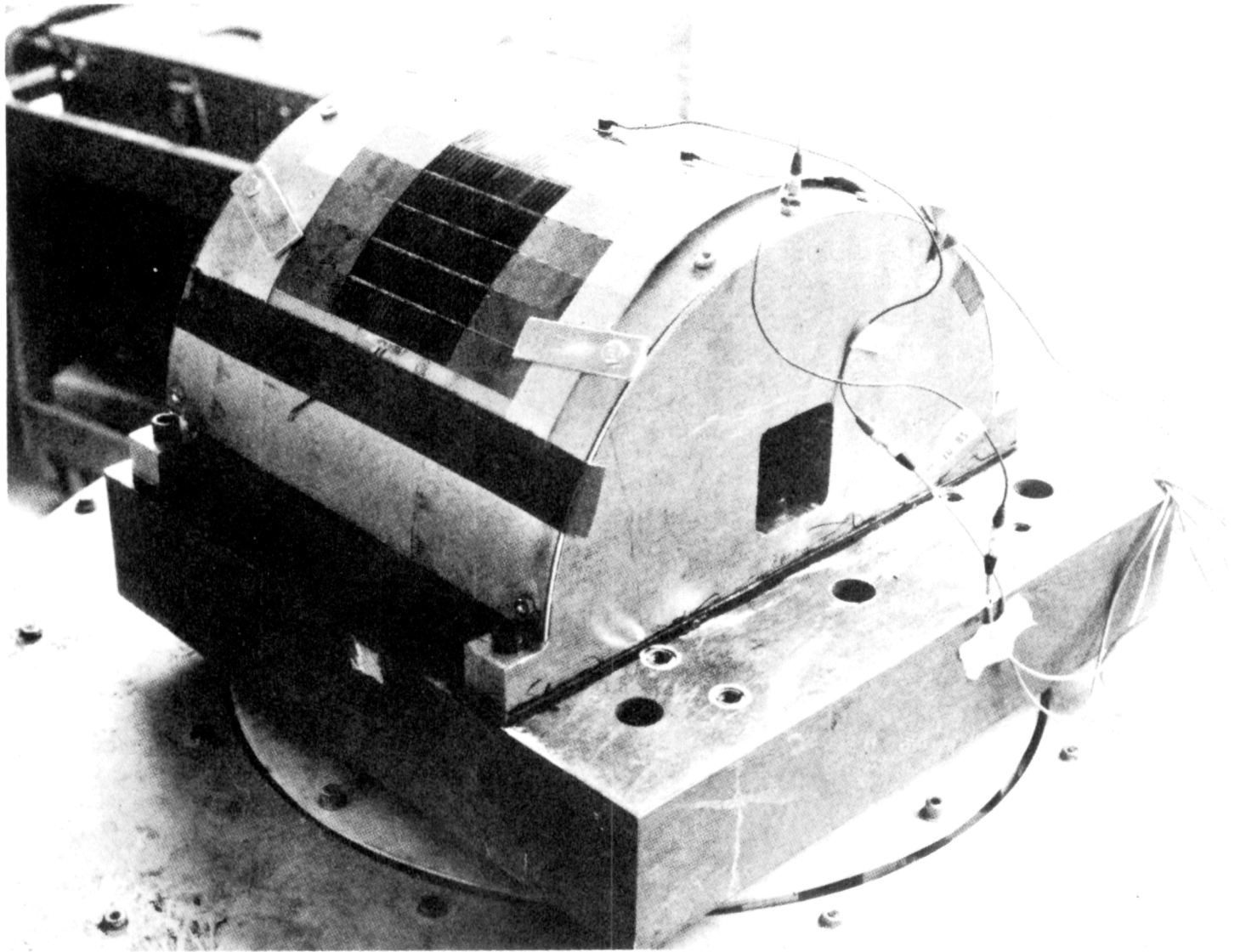


Figure 98 Solar Cell Impact ~ Test Setup on Vibration Exciter

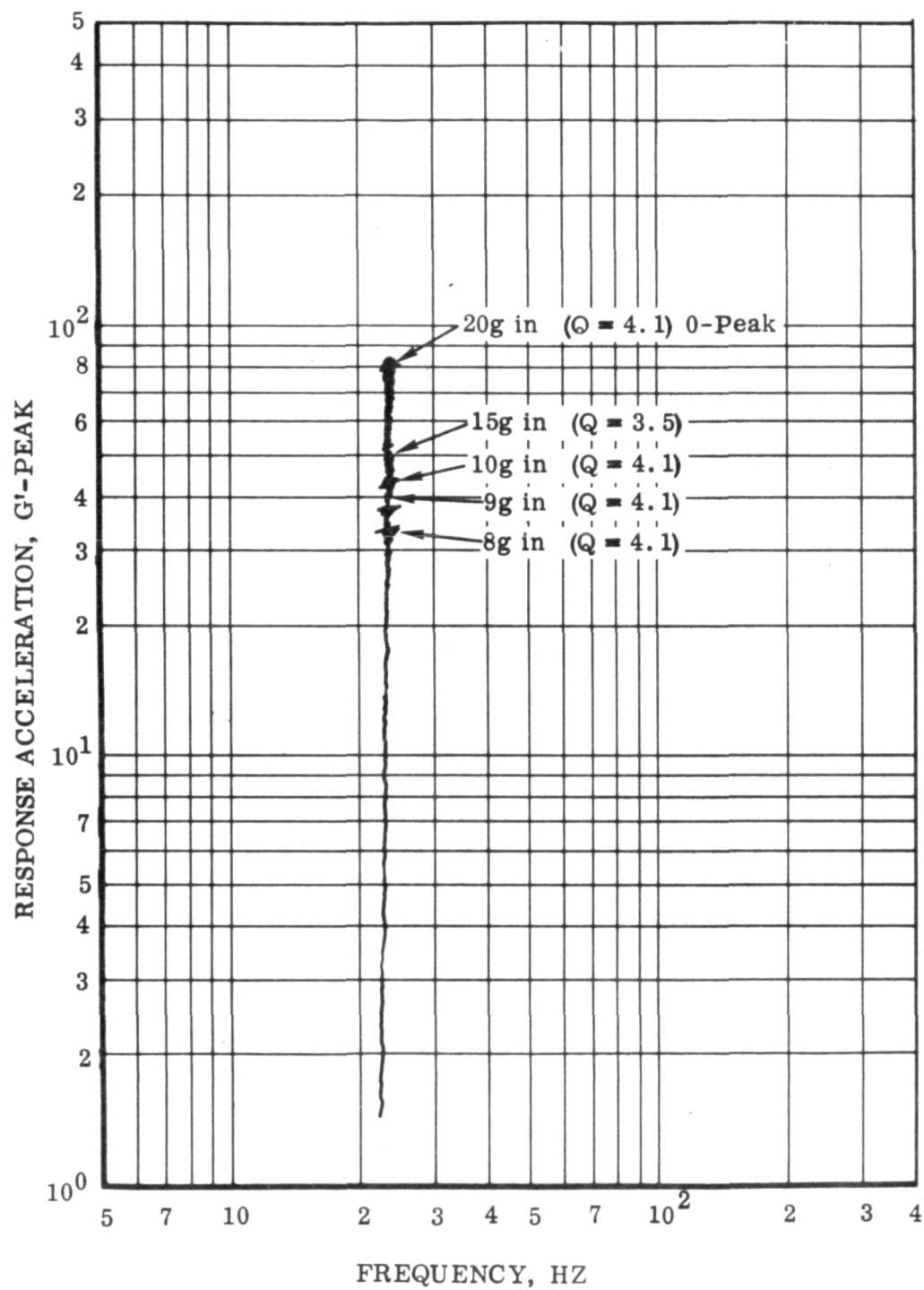


Figure 99 Effect of Excitation Acceleration on Response Acceleration at Resonance

9.3 WEIGHT BREAKDOWN FOR SOLAR CELL ELECTRICAL INSTALLATION MASS SIMULATION

This section includes a breakdown of the mass used on the four panel modules during the test program to simulate the 0.3 pound/square foot solar cell electrical installation weight as defined in the JPL contract Statement of Work for Design of the Solar Panel. The components which make up the mass simulation weight on each panel module are grouped into two basic items:

Item 1,

- Aluminum dummy solar cell chips
- Wired solar cell matrices
- P.C. Boards
- RTV-40 Adhesive
- SS 4004 Primer

Item 2,

- Wiring installation

The weights of these items for the respective panel modules were:

<u>PANEL MODULE</u>	<u>WEIGHT, LBS</u>		<u>AREA AVAILABLE FOR SOLAR CELLS</u>	<u>WEIGHT CONVERTED TO LBS/FT²</u>
	<u>ITEM</u>			
	1	2	ft ²	(1+2) ÷ Area
-1 Outboard	2.616	0.128	9.20	0.298
-3 Outboards	3.120	0.140	11.76	0.277
-3 Inboard	3.170	0.180	11.76	0.285
-5 Inboard	<u>3.650</u>	<u>0.366</u>	<u>14.55</u>	<u>0.276</u>
TOTAL	12.556	0.814	47.27	0.283

The weight converted to pounds/square foot of area available for solar cells is about 5.7% unconservative compared to the desired weight of 0.3 pounds/square feet. The effect of this unconservatism on the results of the vibration test frequencies would be about $(1.057)^{1/2} - 1 = 1.03 - 1 = 0.3$, or 3% lower test frequencies.

9.4 SUMMARY OF DEPLOYMENT CYCLES

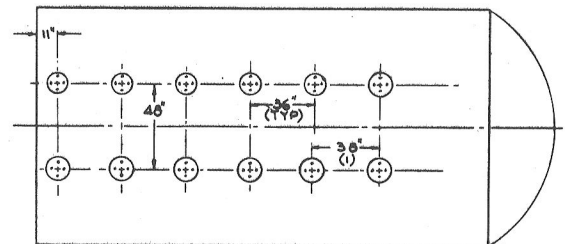
The following table summarizes the number of deployments of the solar panel.

TABLE 12 Summary of Deployment Cycles

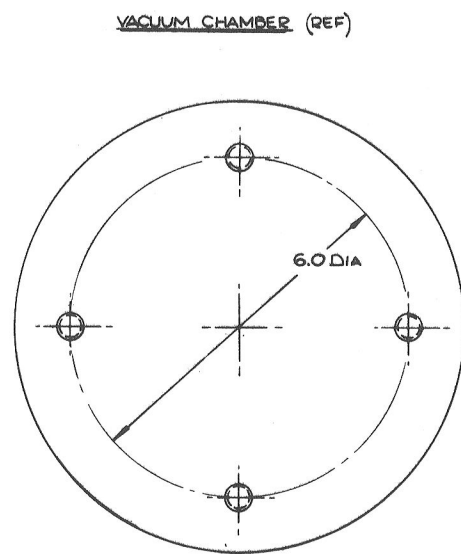
<u>EVENT</u>	<u>NUMBER OF DEPLOYMENTS</u>
Fabrication and Demonstration	34
Cruise Maneuver	3
Assemble Solar Cells	3
Deployment - Effects	5
Instrumentation	5
Vibration (Sine)	1
Repairs	4
Deployment Dynamics	3
Vibration (Random)	1
Thermal Vacuum	3
Repairs	5
Thermal Vacuum	3
Motor Calibration and Fixture Evaluation	15
Deployment Evaluations	17
Demonstrations	20
TOTAL	122

9.5 TEST FIXTURES DRAWINGS

This section contains drawings of the test fixtures and roller support fixture used in the test program.



ON SHT. 2		
2 1/2 x 2 1/2 x 1/8 ANGLE AL-ALY 6063 QQ-A-200-9	2	19.0 LONG
2 x 2 x 1/8 ANGLE AL-ALY 6063 QQ-A-200-9	2	24.0 LONG
1/4-20 BOLTS & NUTS (CON)	AR	LENGTH A.R.

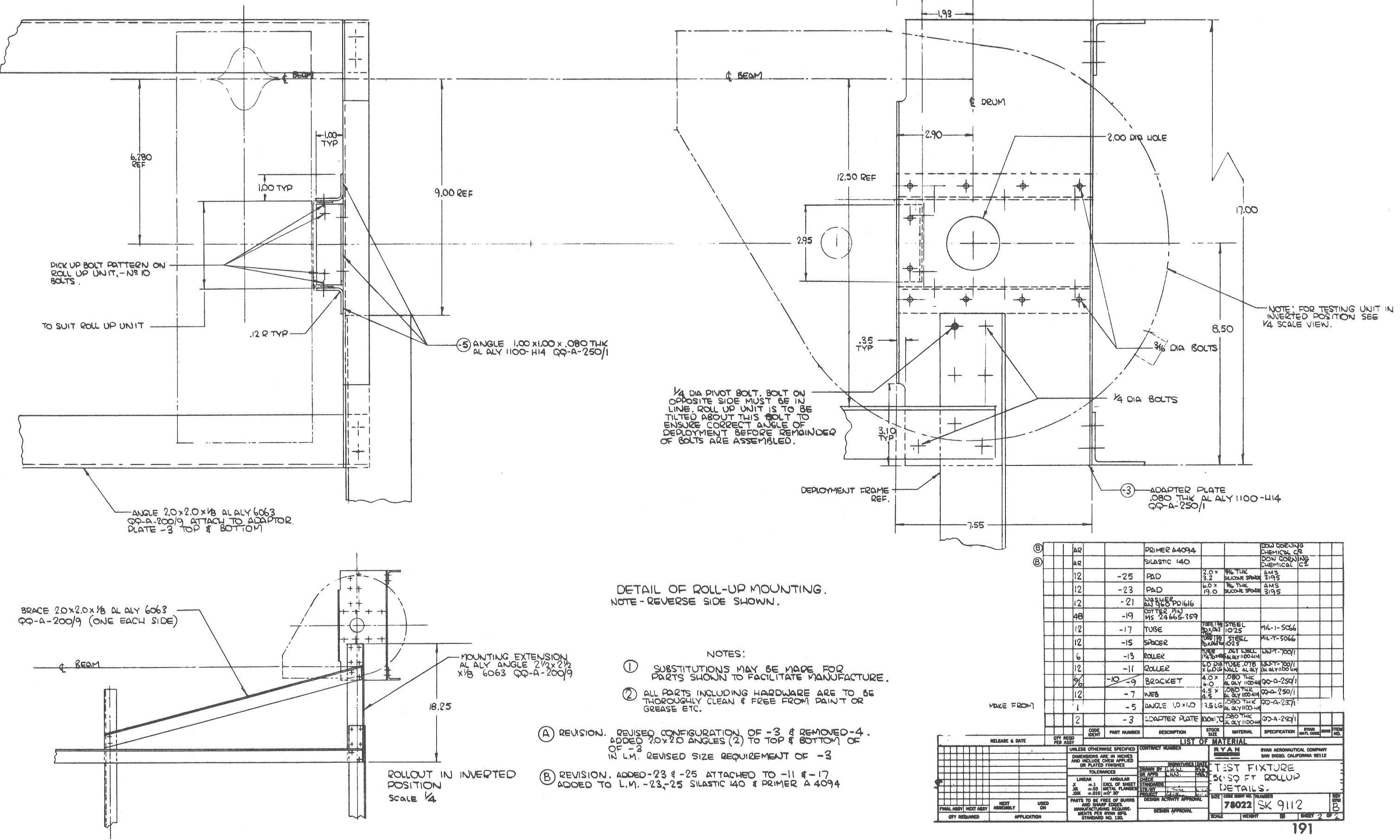


DETAIL OF PAD (TYP)
FULL SIZE

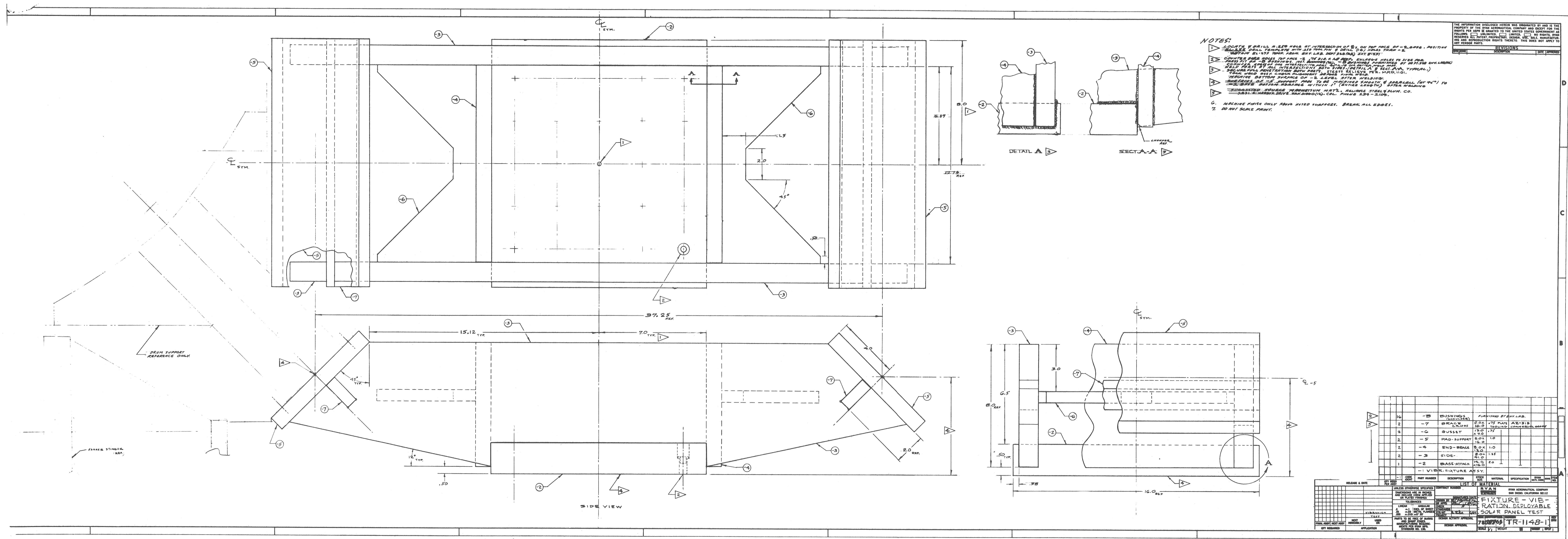
1. ALL PARTS TO BE CLEAN & FREE FROM PAINT OR OIL.

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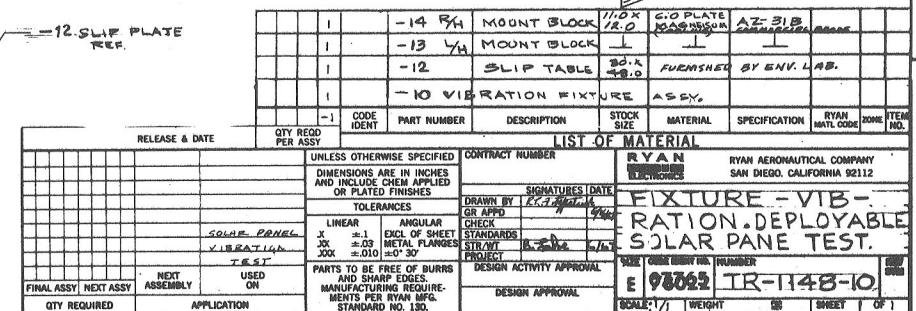
- NOTES:
- 1. LOCATE & DRILL 0.250 HOLE AT INTERSECTION OF C1 ON TOP FACE OF -2. 0.0000. POSITION 1.0000. DRILL TEMPLATE WITH 310 TOOL PIN & DRILL (10) HOLES TRACED - 2.
 - 2. LOCATE & DRILL 0.250 HOLE AT INTERSECTION OF C1 ON TOP FACE OF -2. 0.0000. POSITION 1.0000. DRILL TEMPLATE WITH 310 TOOL PIN & DRILL (10) HOLES TRACED - 2.
 - 3. LOCATE & DRILL 0.250 HOLE AT INTERSECTION OF C1 ON TOP FACE OF -2. 0.0000. POSITION 1.0000. DRILL TEMPLATE WITH 310 TOOL PIN & DRILL (10) HOLES TRACED - 2.
 - 4. LOCATE & DRILL 0.250 HOLE AT INTERSECTION OF C1 ON TOP FACE OF -2. 0.0000. POSITION 1.0000. DRILL TEMPLATE WITH 310 TOOL PIN & DRILL (10) HOLES TRACED - 2.
 - 5. LOCATE & DRILL 0.250 HOLE AT INTERSECTION OF C1 ON TOP FACE OF -2. 0.0000. POSITION 1.0000. DRILL TEMPLATE WITH 310 TOOL PIN & DRILL (10) HOLES TRACED - 2.
 - 6. LOCATE & DRILL 0.250 HOLE AT INTERSECTION OF C1 ON TOP FACE OF -2. 0.0000. POSITION 1.0000. DRILL TEMPLATE WITH 310 TOOL PIN & DRILL (10) HOLES TRACED - 2.
 - 7. LOCATE & DRILL 0.250 HOLE AT INTERSECTION OF C1 ON TOP FACE OF -2. 0.0000. POSITION 1.0000. DRILL TEMPLATE WITH 310 TOOL PIN & DRILL (10) HOLES TRACED - 2.

2	16	-B	BUSHINGS	FURNISHED BY INCL. L.B.
5	2	-7	BRACE	2.0A 77 FINE AZ-118
	2		10-0	TOLUOL
	2	-5	GUSSET	15.0 75 FINE
	2	-5	PAD-SUPPORT	8.0X 1.0
	2	-4	END-BRACE	8.0X 1.0
	2	-3	SIDE-	8.0 1.5f
	2	-2	BASE-ATTACH	14.0 1.0
	1	-1	VIB-FIXTURE ASSEMBLY	8.0 1.0

RELEASE & DATE	QTY REQ PER ASSY	CODE (UNIT)	PART NUMBER	DESCRIPTION	MATERIAL	SPECIFICATION	RYAN MATERIAL	QTY REQ PER ASSY

Page Intentionally Left Blank

③ -13E-4 TO BE MADE FROM ONE BLOCK OF MAG. 11.0
WIDE. SAWED @ 45°. IF LENGTH AFTER MACHINING
EXCEEDS FLAGGED DIM, LEAVE EXCESS REMAIN.
SUGGESTED SOURCE OF 6" MAG. PLATE: RELIANCE STEEL &
ALUM. CO 2931 EMERALD DRIVE SAN DIEGO PHONE 234-210



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9.6 PROCESS SPECIFICATIONS

This section contains process specification 208S003 used for assembly of solar cells on the solar panel.

RYAN AERONAUTICAL COMPANY

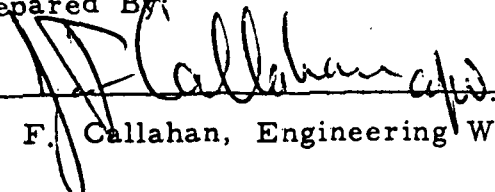
San Diego, California 92112

RYAN ENGINEERING SPECIFICATION

208-S-003

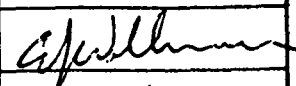
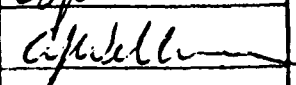
PROCESS REQUIREMENTS FOR
MODEL 208 SOLAR ARRAY -- INSTALLATION
OF ACTIVE AND DUMMY SOLAR CELL SUBMODULES

Prepared By:


J. F. Callahan, Engineering Writer

Approved By:


A. J. Wellman, Project Engineer
Space Structures

REV.	AMEND.	DATE	AUTHORITY
N/C		7-25-67	
A	Revised	8-25-67	
B	Revised Pages 10	9-5-67	
	11, 12, 13 and 14		

1.0 SCOPE

1.1 This Specification describes the installation of solar cells (submodule assemblies) bonded to a panel substrate.

1.2 This is an engineering specification to control this process at Ryan Aeronautical Company, and is effective immediately upon issue.

2.0 APPLICABLE DOCUMENTS

The following documents are used for reference and form a part of this document to the extent specified herein:

Specifications:

TT-M-261	Methyl-Ethyl-Ketone (for Use in Organic Coatings)
TT-I-735	Isopropyl Alcohol
MIL-S-23586	Silicone Rubber Compound, Room Temperature Vulcanizing

Other Publications:

Ryan Aeronautical Company:

MPD 1024	Vapor Degreasing
MPD 1061	Adhesives
MTS	Materials Test Schedule
QCM	Quality Control Manual
TMM	Test Methods Manual
W&IM	Workmanship and Inspection Manual

3.0 PRODUCT MATERIAL CONTROL

3.1 PRODUCT MATERIALS

The following materials have been evaluated by the Materials and Process Laboratories and are approved for use in accordance with the applicable specifications or drawings. Other materials conforming to the applicable specifications or drawings may be used only when evaluated and approved by the Materials and Process Laboratories.

1. Solar Cell Assembly per Spectrolab Drawing No. 015774.
2. Deployable Solar Array Panel Assembly per Ryan Drawing No. 208V013.
3. Primer, SS-4004, General Electric Company.
4. Adhesive, silicone rubber, General Electric Company No. RTV-40 per MIL-S-23586.
5. Catalyst for Item 4, General Electric Company No. Thermo-lite 12.

3.2 PRODUCT MATERIAL CONTROL

3.2.1 COMPONENTS

The panel substrate and solar cells shall be inspected and approved for compliance with the applicable Engineering Drawings.

3.2.2 MATERIALS

- 3.2.2.1 Each lot of materials shall be certified by the supplier to conform to the applicable military or Ryan specification.
- 3.2.2.2 Receiving Inspection, storage and retest of materials shall be accomplished in accordance with the Materials Test Schedule.
- 3.2.2.3 The Materials and Process Laboratories shall accomplish material testing required by the materials Test Schedule in accordance with the procedures contained in the Test Methods Manual.
- 3.2.2.4 Records of material testing shall be maintained in accordance with Section 5.6 of the Quality Procedures Manual.

4.0

PROCESS CONTROL

4.1

PROCESS MATERIALS

1. Methyl ethyl ketone (MEK) per TT-M-261.
2. Isopropyl Alcohol per TT-I-735.
3. RACO 2850, K-4 cleaning solvent.
4. Tape, pressure-sensitive, polyester, silicone adhesive, Minnesota Mining & Manufacturing Company No. &-9064.
5. Paper cups, unwaxed, Dixie No. 2336 or equivalent.
6. Cots, finger.
7. Applicators, cotton tipped.
8. Wipers, disposable, Kimwipes or equivalent.
9. Blades, wooden
10. Stylus, wooden.
11. Abrasive paper - Garnet 240 grit or 320 grit.

4.2

PROCESS EQUIPMENT

1. Balance, sensitivity 0.1 grams.
2. Sponge applicator - capable of depositing 0.0005 to 0.0010 thick uniform coating of SS-4004 primer.
3. Spacers, submodules.
4. Spreader, calibrated, for RTV-40, capable of depositing an adhesive layer weighing 0.03 +0.000 -0.005 pound per square foot.
5. Straight edge.
6. Template, aluminum chip arrangement.

4.2 PROCESS EQUIPMENT (Continued):

- 7. Vacuum system, capable of maintaining a pressure of 5 mm of mercury.
- 8. Weights, submodule bonding.

4.3 PROCESS PERSONNEL

Only qualified personnel who have been adequately trained and tested shall perform the operations described in this specification.

5.0 MANUFACTURING PROCESS

5.1 HANDLING PRECAUTIONS

All materials used in the manufacture of solar arrays are lightweight, fragile, and costly. Therefore, observe the following precautions when handling components.

- 5.1.1 Filtered submodules are subject to dislocation of the filter and delamination of bus terminal contacts when excessive pressure is applied. During cleaning, positioning and bonding operations, observe all precautions detailed in the procedures.
- 5.1.2 Do not allow solar cell submodules to come in contact with metallic objects. Wear finger cots whenever handling the cells to prevent their contamination.
- 5.1.3 The panel substrate is so lightweight that even a pencil dropped on it can cause critical damage. Therefore, the panel must be handled with extreme caution at all times and adequately supported whenever pressure is brought to bear on its surface.

5.2 BONDING OF ACTIVE SUBMODULES

5.2.1 PREPARATION OF ACTIVE SUBMODULES

- 5.2.1.1 All surface areas to be bonded shall be cleaned with isopropyl alcohol and Kimwipes and/or brush before priming.
- 5.2.1.2 Clean all the negative surface of the submodule ("N" bus bar) using isopropyl alcohol applied with cotton swab.

5. 2. 1. 3 Using a sponge applicator, apply a uniform coat of SS-4004 primer on the faying surface of each submodule.

5. 2. 1. 4 Allow each primed submodule to set for five (5) minutes. Using a cotton swab saturated with isopropyl alcohol, remove the excess primer from the tabs. Allow the primed submodule to dry for one hour at room temperature before usage.

5. 2. 2 PREPARATION OF PANEL SUBSTRATE

CAUTION: Exercise extreme care while handling the substrate assembly.

5. 2. 2. 1 Solvent wipe the panel substrate, using clean Kimwipes saturated with MEK, and immediately wipe dry with clean, dry Kimwipes. Do not allow the solvent to evaporate on the substrate.

5. 2. 2. 2 Carefully sand surface of substrate using #240 or #320 Garnet paper to remove gloss.

5. 2. 2. 3 Remove sanding residue by solvent wiping the panel substrate using clean Kimwipes saturated with isopropyl alcohol and immediately wipe dry with clean, dry Kimwipes. Do not allow solvent to evaporate on the substrate. Mask-off bonding area with Y-9064 tape.

5. 2. 2. 4 Using a sponge applicator apply a smooth uniform light pink coat of SS-4004 primer on the panel substrate. Allow to air dry at room temperature for a minimum of one hour.

5. 2. 2. 5 The submodules shall be stored in a clean protective container until the start of the bonding operation.

5. 2. 3 PREPARATION AND APPLICATION OF ADHESIVE

5. 2. 3. 1 Weigh 0. 5 parts Thermolite 12 catalyst to 100 parts by weight RTV-40 adhesive and mix in an unwaxed paper cup, stirring thoroughly for not less than two (2) minutes.

NOTES:

1. Five (5) drops of Thermolite 12 equals approximately 0. 1 grams.
2. A convenient adhesive formulation for this application is 12 drops of Thermolite 12 to 50 grams of RTV-40.

5. 2. 3. 2 Record the quantity of adhesive and catalyst prepared, and the date on which it was mixed.
5. 2. 3. 3 Deaerate the mixed material in a vacuum chamber. Maintain a pressure of three to five mm of mercury until bubbling has substantially subsided, or for ten (10) minutes maximum.

CAUTION: Before starting the next operation, ensure that the panel substrate is adequately supported.

5. 2. 3. 4 Using the special calibrated spreader provided, apply a uniform film of RTV-40 adhesive approximately 0.005 inch thick on the substrate bonding area. The adhesive weight shall be $0.03 \pm 0.000 - 0.005$ pound per square foot.
5. 2. 3. 5 Retain the unused adhesive for use as a control sample for adhesive set verification with grid chart showing location.

5. 2. 4 **ACTIVE SUBMODULE INSTALLATION**

CAUTION: The submodule solar cells are fragile and easily contaminated. Keep away from metallic objects and wear finger cots when handling.

5. 2. 4. 1 Position the first submodule, using a straight edge and submodule spacer to obtain proper alignment of the submodule.
5. 2. 4. 2 Place the second submodule next to the spacer, then position another spacer. Continue placing the submodules and spacers alternately until all the submodules within the masked area are bonded in place.

CAUTION: During the next step, exercise extreme care when positioning the weights on the submodules to prevent damage of the fragile glass filters on the solar cells.

5. 2. 4. 3 Gently place the special weights on the submodules. The weights hold the submodules in place during cure. Individual weights shall cover a row of four cells and shall have a mass of $1/2 \pm 1/10$ pound. The lower surface of the weight which contacts the cell shall be covered by a layer of teflon.
5. 2. 4. 4 Visually inspect spacers for proper alignment and spacing of submodules. Minor variations in spacing may be corrected by carefully shifting the submodules before the silicone adhesive has set.

5. 2. 5 CURE AND CLEANUP

- 5. 2. 5. 1 Allow the RTV-40 adhesive to cure at room temperature for one hour minimum. Check the unused portion of adhesive retained per 5. 2. 3. 4 to verify that the material has set properly and cleanup operations may be performed.
- 5. 2. 5. 2 Carefully remove weights and spacers. Do not dislodge submodules.
- 5. 2. 5. 3 Remove masking tape.
- 5. 2. 5. 4 Remove all excess adhesive remaining between the submodules and above the cell level, using a wood stylus or other suitable soft tool.
- 5. 2. 5. 5 Using a cotton swab dampened with isopropyl alcohol, remove all visible excess primer from the substrate. Do not lift or loose submodules from the panel substrate.
- 5. 2. 5. 6 Repeat Section 5. 2 until all areas where active submodule bonding is specified on the Engineering Drawing have been completed.
- 5. 2. 5. 7 Cure for 24 hours at room temperature.
- 5. 2. 5. 8 Package for storage in the special containers provided for this purpose.

5. 3 BONDING OF DUMMY SUBMODULES

5. 3. 1 DUMMY SUBMODULE ASSEMBLY

NOTE: Wear clean, white cotton gloves whenever handling primed aluminum chips.

- 5. 3. 1. 1 Verify that the aluminum chips have been primed (pink color on faying surface). If primer is present, proceed with 5. 3. 1. 2; if not, prime as follows:
 - 1. Mask one surface of the chips. Prime the other surface with SS-4004.
 - 2. Air dry the primed chips for a minimum of one hour. If submodule assembly will not immediately follow, return the chips to a clean protective container for storage. Installation of primed chips must be accomplished within 24 hours.

5. 3. 1. 2 Assemble the aluminum chips into a submodule configuration per the Engineering Drawing, using Y-9064 pressure sensitive tape and cardboard stiffeners as required.

5. 3. 1. 3 Store the assembled dummy submodules in a clean protective container until ready for the bonding operation.

5. 3. 2 INSTALLATION OF DUMMY SUBMODULES

5. 3. 2. 1 Weigh and record the weight of the panel substrate before installing the dummy submodules.

5. 3. 2. 2 Prepare the panel substrate per 5. 2. 2.

5. 3. 2. 3 Prepare and apply adhesive per 5. 2. 3.

CAUTION: Before starting the next operation, ensure that the panel substrate is adequately supported.

5. 3. 2. 4 Position the dummy submodules per 5. 2. 4. 1 and 5. 2. 4. 2. Place individual weights on the dummy cell rows to hold the cells in place.

5. 3. 2. 5 Allow the RTV-40 adhesive to cure at room temperature for one hour minimum. Check the set of the adhesive before proceeding with the next step.

5. 3. 2. 6 Remove masking tape and trim edge of adhesive.

5. 3. 2. 7 Repeat the above portions of 5. 3 until all areas where dummy submodule bonding is specified on the Engineering Drawing have been completed.

5. 3. 2. 8 Allow the completed unit to air dry for 24 hours.

5. 3. 2. 9 Remove the tape that was used to hold the dummy submodules together during bonding.

5. 3. 2. 10 Remove excess adhesive and primer per 5. 2. 5. 4 and 5. 2. 5. 5.

5. 3. 2. 11 Weigh and record the weight of the panel assembly after installation of the dummy submodules has been completed.

5.3.3 SPOT BONDING OF TEFLON WIRES

- 5.3.3.1 Determine the locations for spot bonding as required by Drawing 208V013.
- 5.3.3.2 Prepare the panel substrate per paragraphs 5.2.2.2 and 5.2.2.3. The surface of the wires shall be untreated.
- 5.3.3.3 Mask off the area to be bonded using Y-9064 tape. Position wires per 208V013 allowing specified excess between tie straps.
- 5.3.3.4 Prepare EPON 956 per MPD 1061. Cut glass cloth tie strips to size per 208V013.
- 5.3.3.5 Apply coat of EPON 956 to the area to be bonded. Place the dry cloth on the adhesive. Using a dry brush, force the adhesive into the cloth sufficiently to wet the fibers and hold loose fibers in place. Add additional EPON 956, if required.
- 5.3.3.6 Allow adhesive to gel at least two (2) hours. Remove tape and sand off sharp edges and loose fibers using 240 grit garnet paper. Wipe clean with isopropyl alcohol.
- 5.3.3.7 Unless otherwise specified by engineering drawing, the spot bonds shall be spaced at two inch intervals using a 0.20 to 0.40 inch wide bond. Additional spots shall be provided at all wiring bends and within 1.0 inch of all terminal boards.

6.0 INSPECTION

Each completed panel shall be visually inspected for compliance with the following requirements.

- 5.1 Excessive bonding material or loose material shall be cause for rejection. All RTV-40 shall be removed from above the cell level and in between the cells. Acceptance shall be based on a visual standard approved by Engineering.
- 6.2 The panel shall be examined visually for evidence of contamination which shall be cause for rejection. Contamination shall be defined as loose adhesive particles, solder flux, excess primer, excess RTV-40, loose solder balls, dust and any other foreign material.
- 6.3 The active submodules shall be located as specified on Ryan Engineering drawing No. 208V013.
- 6.4 The dummy submodules shall be neatly arranged in the general order shown on Drawing 208V013.
- 6.5 The active submodules shall be checked for any evidence of damage such as chips, delamination, cracks, contamination and bus bar breaks. A microscopic inspection of active cells shall be accomplished using at least a 7-power magnification. Description and location of defects shall be recorded and correlated to Drawing 208V013.
- 6.6 Prior to application of the SS-4004 primer the sanded surface of the substrate shall be inspected. Acceptance shall be based on visual standards approved by Engineering.
- 6.7 The SS-4004 primer shall be inspected for intensity and uniformity of application prior to spreading of RTV-40 or applying cell modules. Acceptance shall be based on visual standards approved by Engineering.
- 6.8 The top surface of dummy cells shall be completely covered by a continuous coat of paint per Drawing 208V013. Loose paint, blisters, other visual defects shall be cause for rejection. Inspection acceptance shall be based on visual standards approved by Engineering.

- 6.9 The front and rear surface of the substrates shall be examined to determine the presence and size of adhesive voids between cells and substrate. At least 95 per cent of the area of each cell shall be bonded. This determination shall be made visually. The use of probes is prohibited.

7.0 REPAIR OR REWORK

Spot bonding of wires shall be inspected for conformance to Drawing 208V013 and visual standards approved by Engineering.

7.1 AUTHORIZATION

- 7.1.1 Repair or rework to active solar cells submodules that have been improperly located, or that do not otherwise meet inspection requirements, shall be accomplished only by Material Review Board action.
- 7.1.2 Repair or rework to dummy submodules that have been bonded in place shall be accomplished only with prior Inspection approval on a Rework Order.
- 7.1.3 All other repair or rework shall be accomplished only with prior Engineering approval.

7.2 PROCEDURE FOR REPAIR OF DUMMY CELLS

Dummy cells that have been improperly bonded in place shall be repaired or reworked as follows:

PROCEDURE FOR REPAIR OF DUMMY CELLS (Continued)

1. Using lightweight nylon monofilament (fishing) line, remove the affected submodules by carefully inserting the line under each submodule and drawing the line toward the operator with a sawing motion. Exercise care not to dislodge adjacent submodules.
2. It is not necessary to remove old adhesive before applying the new unless the repair area is large enough that the total adhesive weight per square foot would exceed specified tolerances of 0.03 +0.000 -0.005 pound.
3. Apply new adhesive, allow to set for one hour, clean up and cure for 24 hours as specified in 5.3.
4. The repaired area shall meet the requirements of 6.0.

8.0

QUALITY ASSURANCE

- 8.1 It shall be a Departmental responsibility to adequately train operators and inspectors. The trained personnel shall then be tested by the Materials and Process Laboratories.
- 8.2 Ryan Inspection shall be responsible for the performance of Section 6.0.
- 8.3 It shall be a Departmental responsibility to ensure that process equipment meets the requirements of 4.3.

9.0 SAFETY

9.1 The solvents used in this MPD are toxic and flammable. Use these materials only in well-ventilated areas. Observe adequate fire precautions.

9.2 Avoid bare skin contact with the catalyst. Should contact occur, thoroughly wash the affected area with soap and water.

10.0 NOTES

None.