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MARS MISSION SOLAR ARRAY

D2-121319-1

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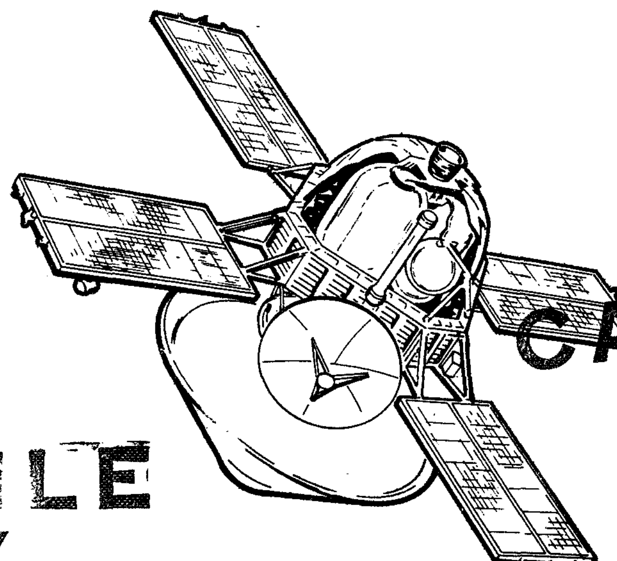
SEMI-ANNUAL PROGRESS REPORT



PREPARED FOR JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
CONTRACT 952571

THE BOEING COMPANY
AEROSPACE GROUP
SPACECRAFT BRANCH, SEATTLE, WASHINGTON

D2-121319-1

MARS MISSION SOLAR ARRAY

SEMI-ANNUAL PROGRESS REPORT

February 1970

This work was performed for the Jet Propulsion Laboratory, California Institute of Technology, Subcontract 952571, as Sponsored by the National Aeronautics and Space Administration under Contract NAS 7-100.

The Boeing Company
Aerospace Group
Spacecraft Branch
Seattle, Washington

MARS MISSION SOLAR ARRAY

ABSTRACT

This report provides technical information concerning the preliminary design, analysis, test article design, fabrication, and test of a beryllium-structure solar panel. The Test Panel being fabricated under this contract is a prototype of a solar panel design suitable for a Mars mission in 1973. This report describes the activity through December 31, 1969, which includes the completion of preliminary and detail design of the Test Panel, summary of the thermal, dynamic, and static analyses, and the fabrication and test efforts in progress.

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LIST OF ILLUSTRATIONS

SECTION 1: INTRODUCTION

This document reports the progress made from contract start, July 9, 1969, to December 31, 1969, for the mid-term report of JPL Contract 952571, Mars Mission Solar Array (MMSA).

The document describes the program activities by major tasks. A summary of the five tasks is included in Section 2, followed by sections covering the following subjects:

1) Status summary of the five major tasks
2) The Program Plan
3) Preliminary Design and Analyses
4) Test Article Design
5) Manufacture of the Test Article
6) The Test Program

Conclusions and recommendations included are preliminary because the testing of the panel has not been started.

The format of this document has been arranged to provide two levels of reporting. The first two sections provide an introduction and summary of the status of work performed to date. The following sections provide more detail on the various aspects of the program. Primary points are headlined at the beginning of each paragraph to assist the reader in locating desired information.

The report is organized to give a complete account of the activities on the MMSA program. Each major task is reported separately and corresponds to the five program tasks as proposed. The sections contain information as follows:

Section 1: Introduction---Gives the background of the program, purpose of the report, the scope, and organization of the report.

<u>Section 2:</u> Summary---Provides a brief status report of the program activities during 1969 and a brief report of the work to be accomplished in 1970.

<u>Section 3:</u> Program Plan---Presents the status of the program planning and changes made subsequent to completion of the Program Plan.

<u>Section 4:</u> Preliminary Design and Analyses---Presents the preliminary design investigations and analyses leading up to the evaluation of a Baseline Configuration and an Alternate Configuration.

<u>Section 5:</u> Test Article Design---Describes the test article design in detail with results of the electrical, dynamic, mechanical, and static analysis.

<u>Section 6:</u> Manufacture of Test Article---Gives the status of the design and fabrication of tools used to produce the solar panel and the status of the panel fabrication with a discussion of problems encountered.

<u>Section 7:</u> Test Program---Discusses the Test Plan, Test Procedures, and Test Fixtures.

1.1 PROGRAM BACKGROUND

The program approach allowed for adjustment in the panel design to accommodate program design changes.

The Mars Mission Solar Array Program was planned to verify the suitability of the Large Area Solar Array (LASA) type lightweight solar panel construction for use on interplanetary missions other than the ion engine powered Mars Mission for which the LASA was designed. In July 1969, The Boeing Company was awarded a contract to design, fabricate, and test a solar array using the technology developed on LASA. The panel was to be suitable for a Mars Mission in 1973, with a goal of 20 watts per pound specific power output.

A two-month period was set aside at the beginning of the program to make a study of the effect of mounting extraneous equipment on the panel. The contract specified the mounting of a fifteen-pound unit and two two-pound units of mass-simulated equipment on the panel with the provision that these weights might be eliminated when the results of the study were reviewed.

1.2 PURPOSE

This report provides information by which the program progress can be evaluated.

The purpose of this report is to document the actions that have been taken in 1969 and to provide the results of the preliminary design investigations which show the effect of supporting extraneous equipment on the panel.

This report also provides information on the detail design of the test panel, how it is being fabricated and tested, and the status of the design, fabrication and test program.

1.3 SCOPE

This report provides summary information suitable to define the program status.

This report provides technical information concerning the activities from contract go-ahead, July 9, 1969, to December 31, 1969, on the Mars Mission Solar Array contract. Information is provided for each of the five tasks; Program Plan, Preliminary Design and Analysis, Test Article Design, Test Article Manufacturing, and Test Program.

SECTION 2: SUMMARY

The Mars Mission Solar Array contract work was started on July 9, 1969. A Program Plan, Task 1, was completed within the first six weeks. The remainder of the tasks were started and are now in varying degrees of completion. The panel design, Task 2, was divided into two phases. The first phase is the work accomplished prior to the Preliminary Design Review (PDR), and has been completed. The second phase, which includes the detail design of the panel and engineering support of the test panel fabrication and test, has not been completed. The fabrication of the test panel is proceeding on schedule. All raw materials and purchased parts have been received. Detail parts have been fabricated and 30 percent of the cell modules have been assembled. Test Plans were prepared and reviewed by JPL in December. Work has been started on the test procedures for Task 4. Task 5, Documentation and Reporting, is proceeding as planned.

In the remainder of this section, the results of the work accomplished in 1969, are summarized by task and work remaining to be accomplished in 1970 is summarized.

2.1 TASK 1 SUMMARY — PROGRAM PLAN

A workable Program Plan has been prepared and provides control for the program.

A Program Plan was prepared in the first six weeks of the program. Figure 2-1 presents a brief status of the major mileposts of the program with separate schedules shown for Engineering, Manufacturing, Test, and general program events.

The Program Plan provides flow charts for Test, Fabrication, and Design. These are used to identify sequence of events and indicate requirements of each activity on the other.

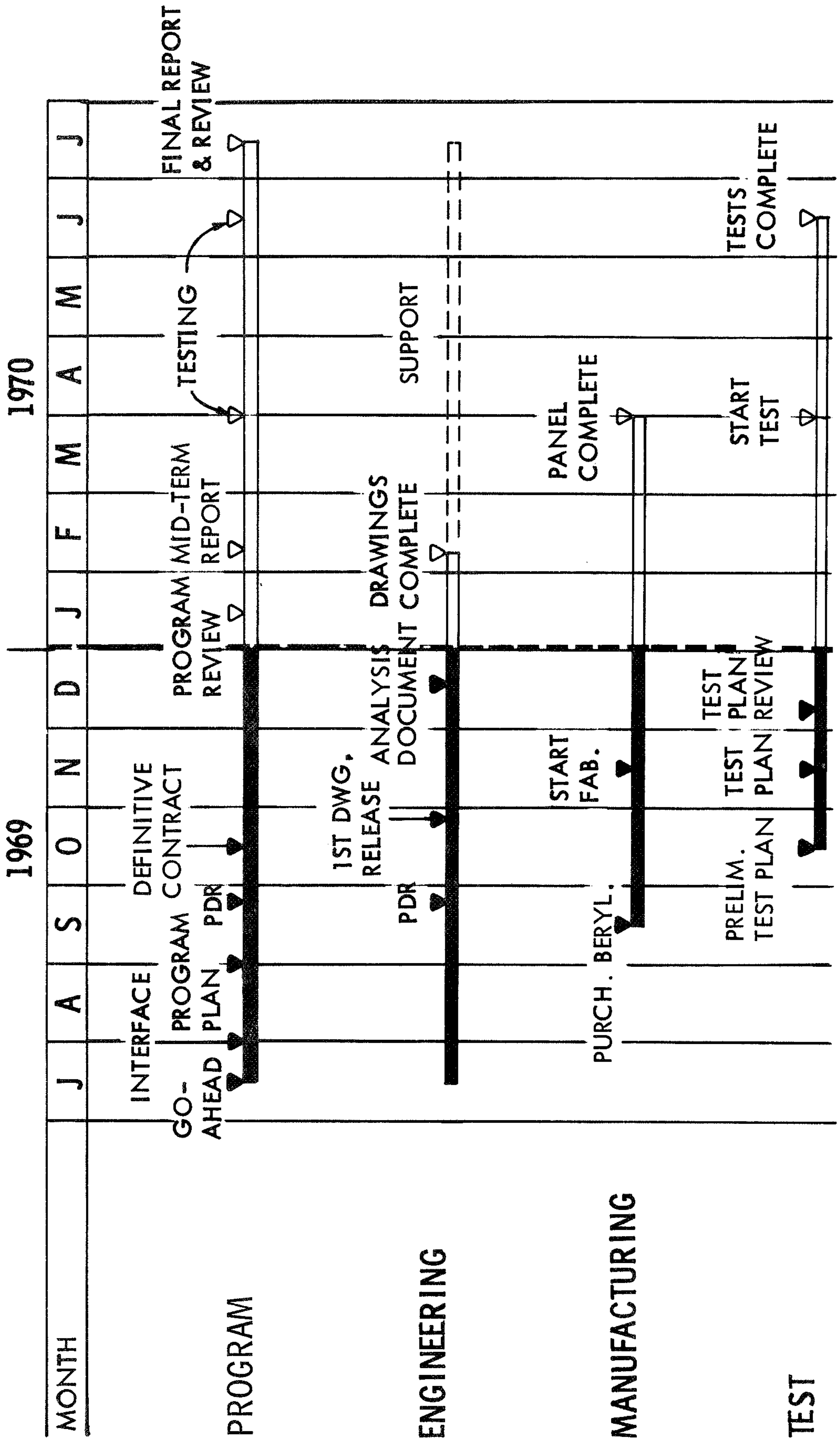


FIGURE 2-1: MMSA PROGRAM STATUS

2.2 TASK 2 SUMMARY -- CONFIGURATION REVIEW AND DETAIL DESIGN

The configuration review revealed a small weight penalty to support extraneous equipment.

The first two months were used to develop the design of a solar panel which supported a relay antenna and other extraneous equipment and to compare it to a panel which did not include the relay antenna.

A solar array suitable for use on a Mars Mission to be launched in the summer of 1973 was assumed. The launch configuration assumed is shown in Figure 2-2. The thermal interface with the spacecraft was not investigated at this time so that an additional variable affecting power output could be eliminated. The panel output was assumed to be 10 watts per square foot. Details of the supported equipment were defined to the degree necessary to make performance evaluations of the two panels.

The two panels were analyzed for weight and power reduction due to the effect of the extraneous equipment and its supporting structure. The comparison is given in Section 4.2.

The results of the study indicated that the penalty of supporting the extraneous equipment in weight and power losses was very small. Also, the weight and electrical performance characteristics were nearly the same when a four-panel array of identical panels was compared to an array in which each panel had different equipment mounting provisions.

Following the study, detail design was completed for the selected test panel configuration and drawings of the test panel details were released for fabrication. In this configuration, the relay antenna and mounting provisions are omitted.

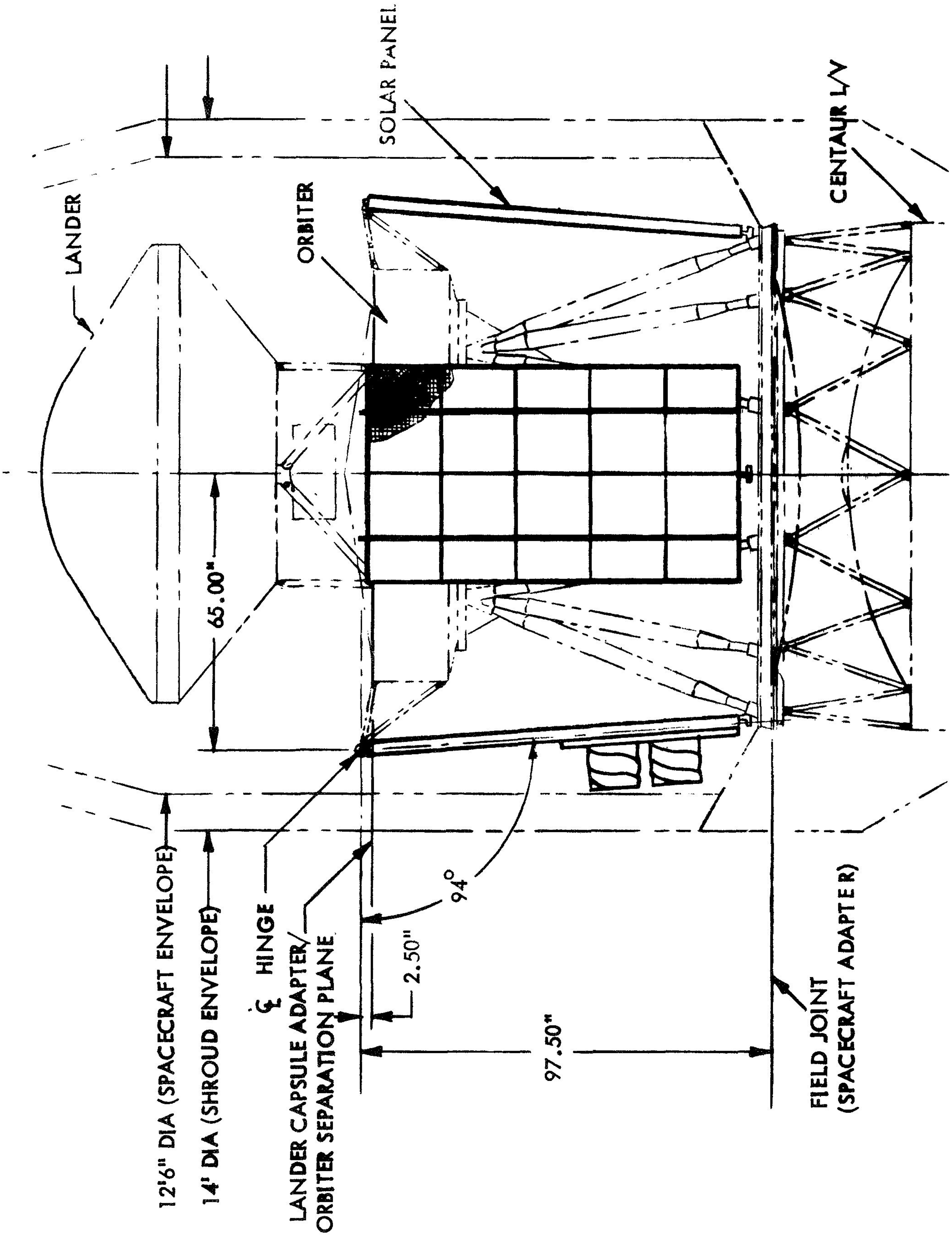


FIGURE 2-2: MARS MISSION SOLAR ARRAY (STOWED)

Work that will be accomplished during 1970 on this task is the completion and release of assembly and installation drawings; engineering support for manufacturing during the fabrication of the test article; engineering support during testing of the panel assembly; and engineering analysis and reporting of the test results.

2.3 TASK 3 SUMMARY -- TEST ARTICLE MANUFACTURE

No significant problems have been encountered in the fabrication of the Test Panel to date.

The manufacture of the panel assembly for test was begun using material surplussed by the LASA Program. Forty-five percent of the test panel beryllium requirement has been obtained by chem-milling heavier surplus stock to the required gages. At present all beryllium material has been received and is in the process of being made into detail parts. Ninety percent of all purchased material has been received.

Tools for the detail parts have been made and the assembly tools are ninety percent complete. Bonding of the beryllium members will be completed early in 1970.

The solar cells and coverglasses used will be obtained from the LASA surplus materials. All coverglass-cell assemblies were completed and cell module assembly was started.

The final assembly of the test panel structure, which joins the substrate, sun-side frame, and dark-side frame, will be completed in late February or early March. Installation of the bus bars, diodes, and connected and unconnected solar cell modules will be installed on the structure the week following the final assembly bonding.

2.4 TASK 4 SUMMARY — TEST PROGRAM

Satisfactory progress has been made in preparing for tests of the Test Panel.

Test plans were completed by the middle of November and a review was held at the Jet Propulsion Laboratory early in December. The first draft of the test procedures has been written. The design of the test fixtures for the power output tests, thermal vacuum, thermal shock, modal survey, random and sinusoidal vibration tests, acoustic and static load tests have been completed.

The major portion of this task remains to be completed in 1970. The performance of the tests shown in Figure 2-3 will be completed. The four additional power output tests requested during the test plan review at JPL on December 3, are included in this figure. Testing will start by April 1, 1970.

2.5 TASK 5 SUMMARY — REPORTING AND DOCUMENTATION

All reports and documentation have been submitted as planned.

The scheduled submittal of all reports and documents is given in Section 3.1. The items submitted are indicated by a filled-in milestone marker. All planned reports and documents scheduled for future submittal are indicated by open milestone markers.

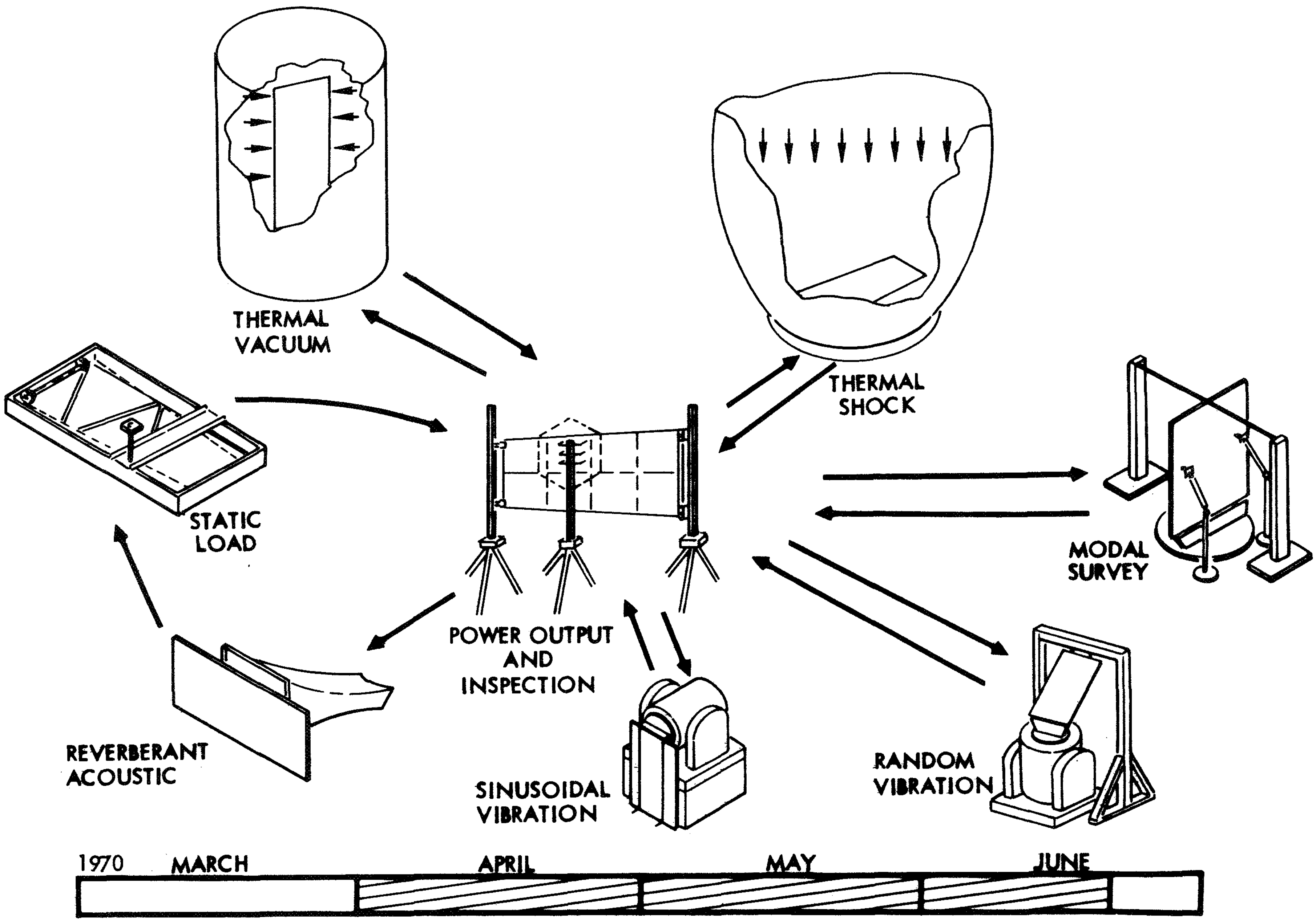


FIGURE 2-3: TEST PLANS - MMSA

SECTION 3: PROGRAM PLAN

During the first four weeks of the contract, a detailed program plan for the one year effort was developed. The plan was made using the five major tasks that had previously been proposed and provided a logic network of all events and activities of each task. It included a master chart showing the time phasing of the overall program. The plan has been followed with the few minor exceptions described in paragraphs 3.1 and 3.2.

3.1 PROGRAM TASKS

Dividing the program into separate tasks provides effective control and reporting on the program activities.

Schedule and cost control of the five program tasks has been maintained separately. The status of each task is discussed below.

1) Task 1 Program Plan---The Program Plan was completed and delivered to JPL six weeks after contract go-ahead, as scheduled.

2) Task 2 Array Design and Analysis---An analysis of the original array concept was completed and trade studies for sizing and material gages of the beryllium structures were made. Preliminary design was completed and a Preliminary Design Review (PDR) was conducted on September 23, 1969, as scheduled.

   Following the PDR, detailed design for the test panel was started and at this report is 90 percent complete and is ahead of schedule. The documentation of the design analysis was delayed because of configuration change in the first two months, and will be two months behind schedule. Other program scheduled events are not affected.

3) Task 3 Manufacturing and Materiel---Purchase orders for beryllium were released and deliveries were made to meet fabrication requirements, although final delivery was three weeks late. Beryllium fabrication was started using the surplus LASA stock. Tooling was also started by re-working, where practical, the tools left over from the LASA program. Solar cell assembly was started. The Task 3 effort is on schedule.

4) Task 4 Testing---Although the panel testing is the major portion of this task, the test plan and procedures and the test fixtures are also included. Work on the test plan was started early and the test plan document was completed. The Task 4 effort is on schedule.

5) Task 5 Documentation and Reporting---This task includes all preparation and delivery of end item documentation reports and presentations. Delivery of all end items has been as planned. A schedule for the end item deliveries is shown in Figure 3-1.

3.2 PROGRAM CHANGES

As a result of the Preliminary Design Review, changes in the design specifications and work level were required.

Design specifications and a modification to the contract have been implemented. A proposed cost reduction for the contract was prepared and submitted to JPL. These changes, however, do not affect the overall program schedule. The changes are:

1) Statement of Work Revisions

   a) Article I(a)(3) required .006 cover filters. "Change to .003 inch coverglass. These shall be the same as used on LASA and all coverglasses not used on LASA will be made available for use on MMSA."

   b) Article I(a)(7)(C) shall be added as follows: "Test Plan Briefing -- A Test Plan Briefing at JPL for the first week in December is requested."

   c) Article I(b)(1) will be deleted. JPL will transfer solar cells and coverglasses from LASA Contract 951934. This will provide the quantity of cells and coverglasses in store.

   d) Article 2, Delivery of Performance Schedule, is revised to add a "Test Plan Briefing on or about December 2, 1969." A presentation was made at JPL on December 3, 1969.

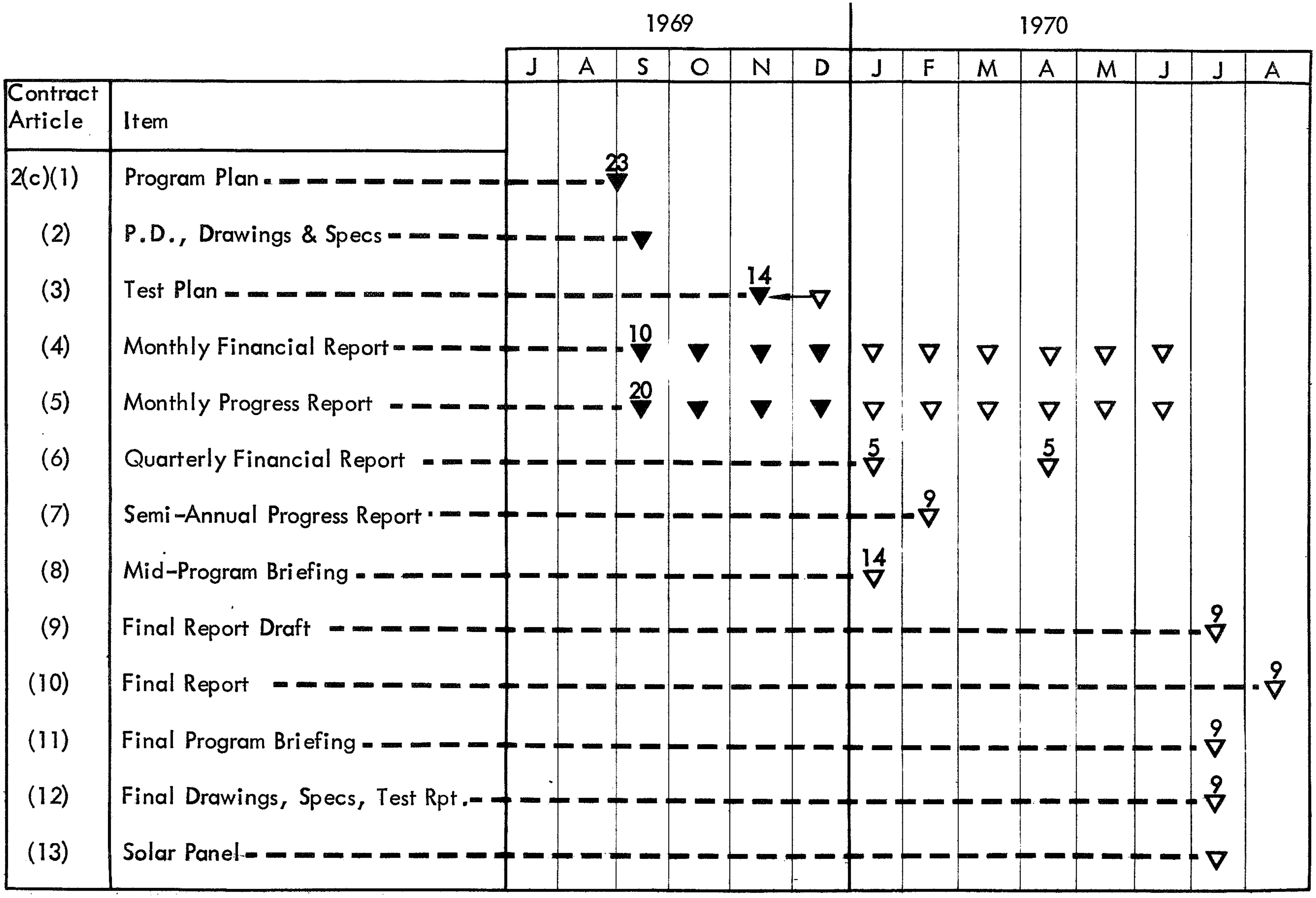


FIGURE 3-1: END ITEM DELIVERY SCHEDULE

2) Specification Changes

The Work Statement, Article I(a), next to last sentence, will be changed to require that the Solar Panel shall meet the functional and test requirements of JPL specification dated November 3, 1969. They are:

a) Paragraph 3.3 Electrical Power. The power provided by the solar cells installed on the MMSA panel will be calculated as 10 watts per square foot. The actual power as provided by the LASA residual cells will be approximately 8.9 watts/sq. ft. at AM=0, 140 mw/cm^2, and 55°C.

b) Paragraph 3.3.2 Solar Cell Cover. The solar cell cover shall be 0211 microsheet 0.003 inch thick without interference filters.

c) Paragraph 3.4.1 Functional Requirements. The drawing referenced in the April 25 specification was changed to JPL Drawing 23835, Rev. A, dated 10-6-69, in the November 3 specification. The changes are as follows:

1. The relay antenna provisions were deleted from the Test Panel.

2. The mounting area for two detectors on the outboard corners was deleted. However, a sun sensor is simulated on the Test Panel centerline.

d) Paragraph 3.4.2 Attitude Control Jets Simulated Weight. The weight of 1.4 pounds was doubled to 2.8 pounds to simulate dual Attitude Control System Jets.

e) Paragraph 3.5.5 Analytical Dynamic Analysis. This paragraph was replaced in its entirety. The new paragraph requires specific definition, location and function of dampers, deployment rate limiters, and similar equipment necessary for a proper design. This requirement assumes that it is necessary to provide the analyses originally specified before the new requirements can be satisfied.

f) Paragraph 3.7.1.4 Acoustic Test Requirements. The overall sound pressure level was increased from 148 db to 150 ± 3 db.

g) Paragraph 3.11 Sinusoidal Vibration Test. The sinusoidal input to which the panel shall be subjected shall be such that the induced stresses are equivalent to those induced by a sinusoidal acceleration input to the spacecraft attach points of 0.5 g rms between 7 Hz and 30 Hz and a linear increase from 0.5 g rms to 2.0 g rms between 30 Hz and 400 Hz. The lowest input frequency is changed from 5 Hz to 7 Hz as agreed to by JPL in previous discussions on test procedures.

SECTION 4: PRELIMINARY DESIGN AND ANALYSIS RESULTS

The solar panel preliminary design configurations were predicated on a hypothetical Mariner Mars type mission and were developed to meet the JPL Specification, "Detailed Requirements for Lightweight Photovoltaic Array Structure Technology," dated April 25, 1969.

Following is a chronological summary of the evolution of the configurations traded in the preliminary design phase:

1) Initial Configurations:

Proposal Configuration---as presented in the Boeing Proposal Document D2-114460-3, dated March 28, 1969. This configuration included:

a) A 15-pound simulated relay antenna

b) A 2-pound simulated sun sensor

c) A 2-pound simulated maneuver antenna

Alternate Configuration---proposed to determine the effect of removing the simulated relay antenna and related mounting provisions.

These configurations were carried to a level of preliminary design suitable for comparison of characteristics and were modified as a result of coordination with JPL, resulting in:

2) Configurations at the time of the Preliminary Design Review (PDR):

PDR Baseline Configuration---in which:

a) The simulated relay antenna weight was reduced to 10 pounds and relocated from the side to the panel centerline.

b) The simulated maneuver antenna was not included because none of the four panels on the spacecraft include both this antenna and the relay antenna.

PDR Alternate Configuration A---in which:

a) The 10-pound relay antenna and related mounting provisions were omitted.

b) Both the 2-pound sun sensor and the 2-pound maneuver antenna were included.

A third configuration was developed to explore the effects of non-interchangeable panels. This was:

PDR Alternate Configuration B---in which equipment and related mounting provisions were included on each of the four panels per spacecraft only when a panel would actually support that equipment item. This resulted in a set of configurations which were a mix of the first two and provided an additional point for weight comparison.

An evaluation and comparison of these three configurations was presented at the Preliminary Design Review. The trade study results are summarized in Section 4.2. The PDR Baseline configuration was selected as the test article configuration with the decision that the relay and maneuver antenna equipment weight and related bracketry be omitted.

4.1 PRELIMINARY DESIGN CONFIGURATIONS

The proposal panel configuration was modified by coordination with JPL to provide a baseline panel configuration for trade study purposes.

After contract go-ahead, a design coordination meeting was held between Boeing and JPL on July 30-31, 1969, which resulted in the following revisions to the proposal panel configuration.

1) The structural member spacings were revised to improve the cell module configuration and to decouple the chord bending mode.

2) The simulated relay antenna weight was revised from 15 pounds to 10 pounds and the antenna C.G. was relocated on the assumption that the stowed panel dynamic responses would be improved.

3) Idealized relay antenna models were assumed for thermal and dynamic analyses. Also an idealized dynamic model was assumed for dynamic analyses.

4) Panel configurations were developed for trade and evaluation purposes. The primary trade study effort involved the comparison of the PDR Baseline Configuration, shown in Figure 4-1, and the PDR Alternate Configuration A, shown in Figure 4-2. A non-interchangeable configuration, PDR Alternate Configuration B, was also developed to determine the weight effect of omitting equipment support provisions from some of the panels on the spacecraft.

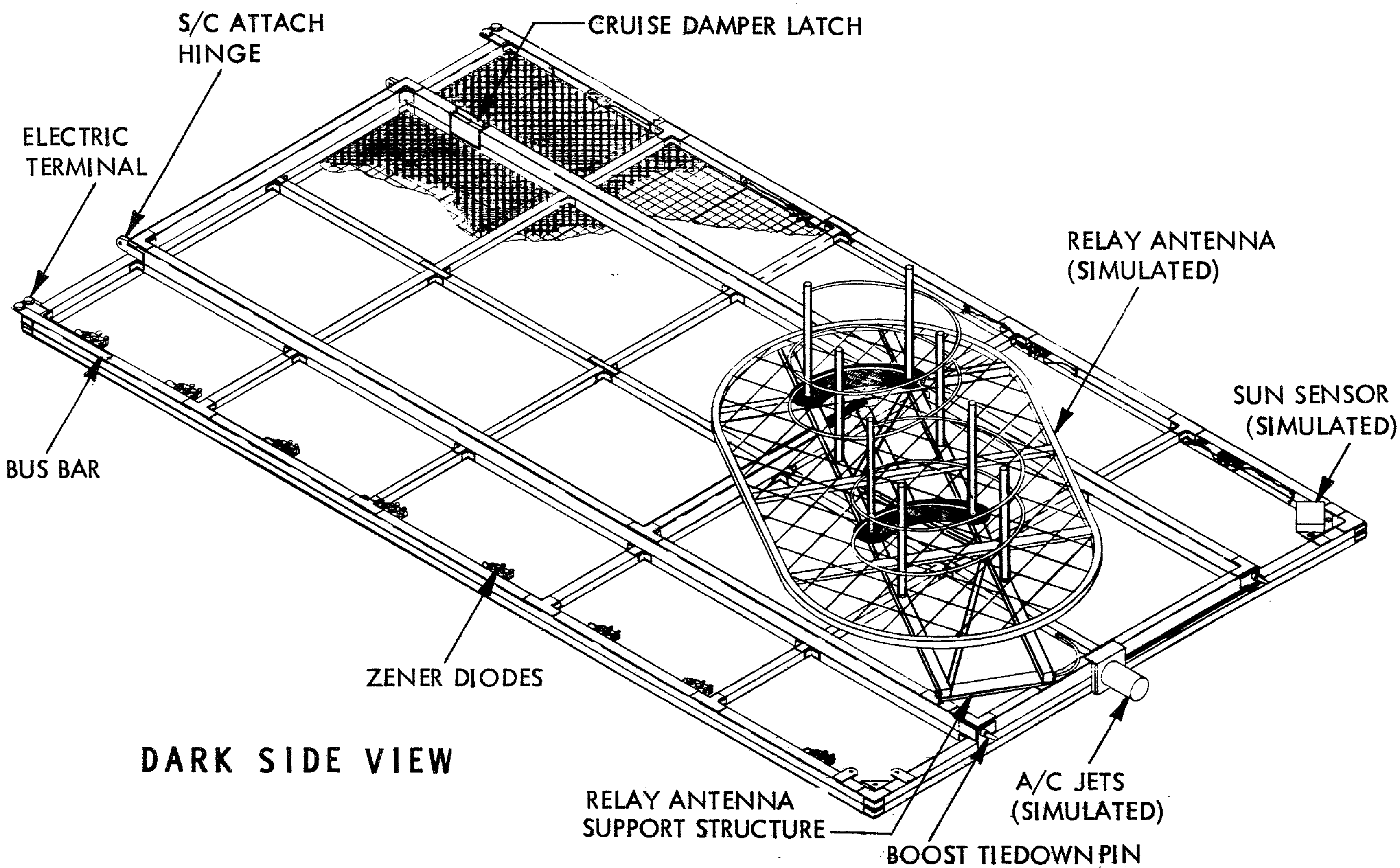


FIGURE 4-1: PDR BASELINE PANEL ASSEMBLY

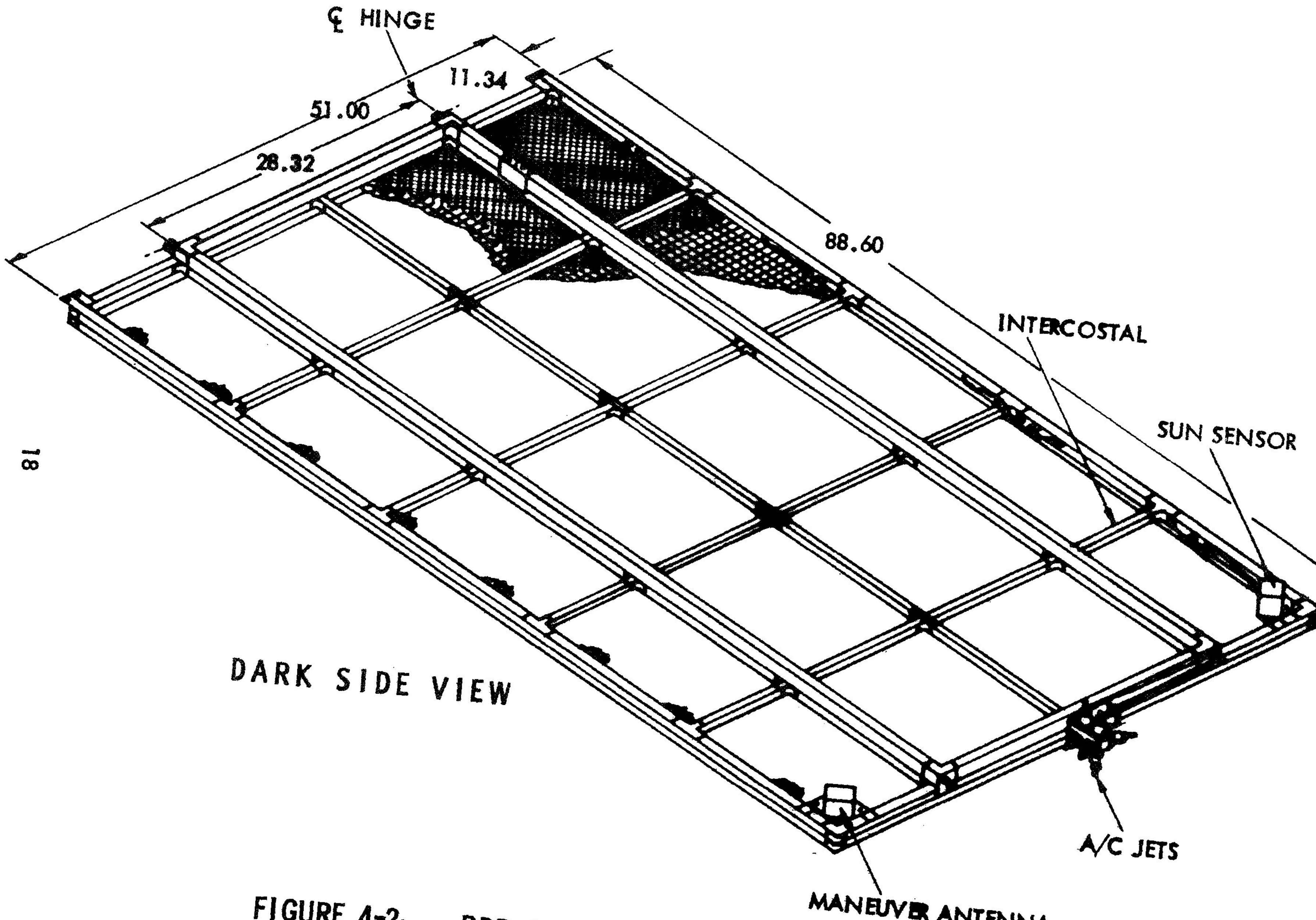


FIGURE 4-2: PDR ALTERNATE "A" PANEL ASSEMBLY

The PDR Baseline Configuration is a refinement of the proposal configuration. Upper and lower edge members were deepened. Members of the lateral spar, which supports the relay antenna, were deepened. Some gage reduction was possible so the structure weight was reduced. Figure 4-3 shows a cross section of the panel and the relative positions of the bonded beryllium structural members with respect to the substrate plane. The hinges and other fittings are titanium.

4.1.1 STRUCTURAL DESIGN TRADE STUDY

The differences between the PDR Baseline Configuration and PDR Alternate Configuration A are minor.

Differences between these two configurations are primarily a result of the omission of the relay antenna and related structural provisions from the PDR Alternate Configuration A. On this configuration the relay antenna support fittings were omitted and the lateral spar was replaced by an intercostal. A 2-pound simulated maneuver antenna was also included on the alternate. Both configurations included mass-simulated attitude control jets, tubing, and the cruise damper latch. A detailed weight comparison of the two configurations is given in Table 4-1. A comparison including the PDR Alternate Configuration B is given in Section 4.2.

4.1.2 ELECTRICAL CONFIGURATION

The MMSA electrical design was derived from the LASA design with minor changes.

4.1.2.1 MODULE DESIGN AND LAYOUT

The same electrical design was used for each of the trade study configurations. A flight-suitable electrical design was developed for weight and performance comparison purposes.

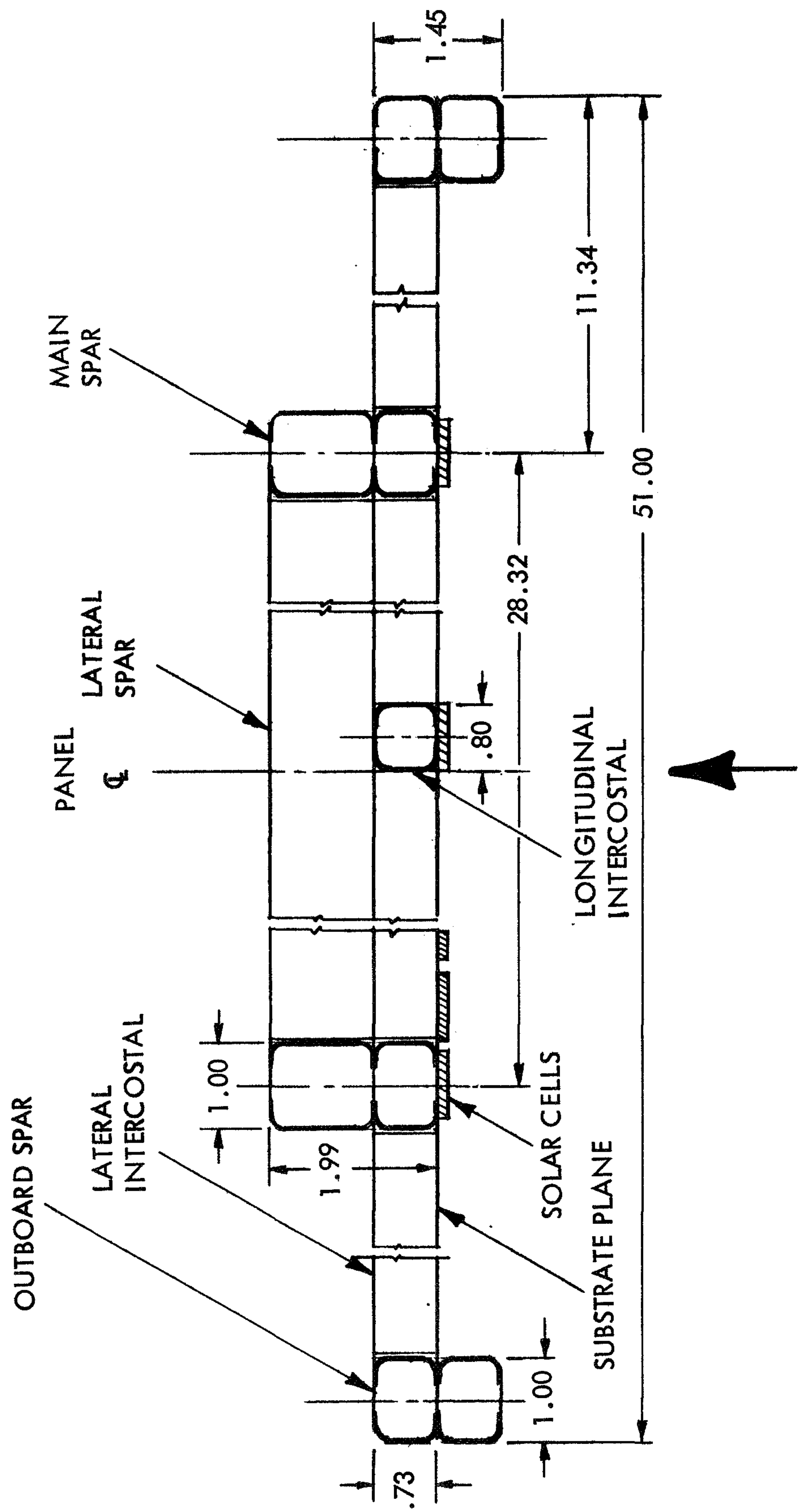


FIGURE 4-3: PANEL CROSS SECTION

TABLE 4-1

WEIGHT SUMMARY FOR MARINER LIGHTWEIGHT ARRAY

	PDR BASELINE Panel, lbs.	PDR ALTERNATE A Panel, lbs.
CELL STACK AND BUSES (TOTAL)	(5.64)	(5.64)
Solar Cells	2.49	2.49
Coverglasses	1.82	1.82
Cell Adhesive	.38	.38
Coverglass Adhesive	.12	.12
Solder, Connectors	.38	.38
Bus and Terminals	.45	.45
DIODES AND INSTALLATION (TOTAL)	(1.80)	(1.80)
PANEL STRUCTURE (TOTAL)	(8.49)	(7.95)
Main Spars	1.68	1.68
Outboard Spars	1.57	1.57
End Members	1.19	1.19
Intercostals	.97	1.08
Caps, Splices, Gussets	.22	.22
Substrate	.46	.46
Thermal Coating	.71	.71
Fittings	1.09	.70
Miscellaneous	.34	.34
Lateral Spar	.26	-
PANEL MECHANISMS (TOTAL)	(.07)	(.07)
Tip Support Pin	.07	.07

TABLE 4-1 (Continued)

	PDR BASELINE Panel, lbs.	PDR ALTERNATE A Panel, lbs.
SIMULATED MASSES & SUPPORTS (TOTAL)	(16.42)	(8.42)
Miscellaneous	.74	.74
Relay Antenna	10.00	-
Sun Sensor	2.00	2.00
Maneuver Antenna	-	2.00
A/C Jets	2.80	2.80
A/C Line	.88	.88
CENTAUR MOUNTED MECHANISMS (TOTAL)	(1.68)	(1.68)
SPACECRAFT MECHANISMS (TOTAL)	(.85)	(.85)
PANEL TOTAL	34.95	26.41

The solar panel includes 12 modules. Nine of the modules are made from parallel groups of seven cells and three modules are made from parallel groups of six cells. The six cell modules were used because of dimensional constraints. Each module contains 80 cells in series. This number will provide voltages which are within acceptable limits both near Earth and at Mars. The panel contains 6,480 cells. The cell module arrangement is shown in Figure 4-4.

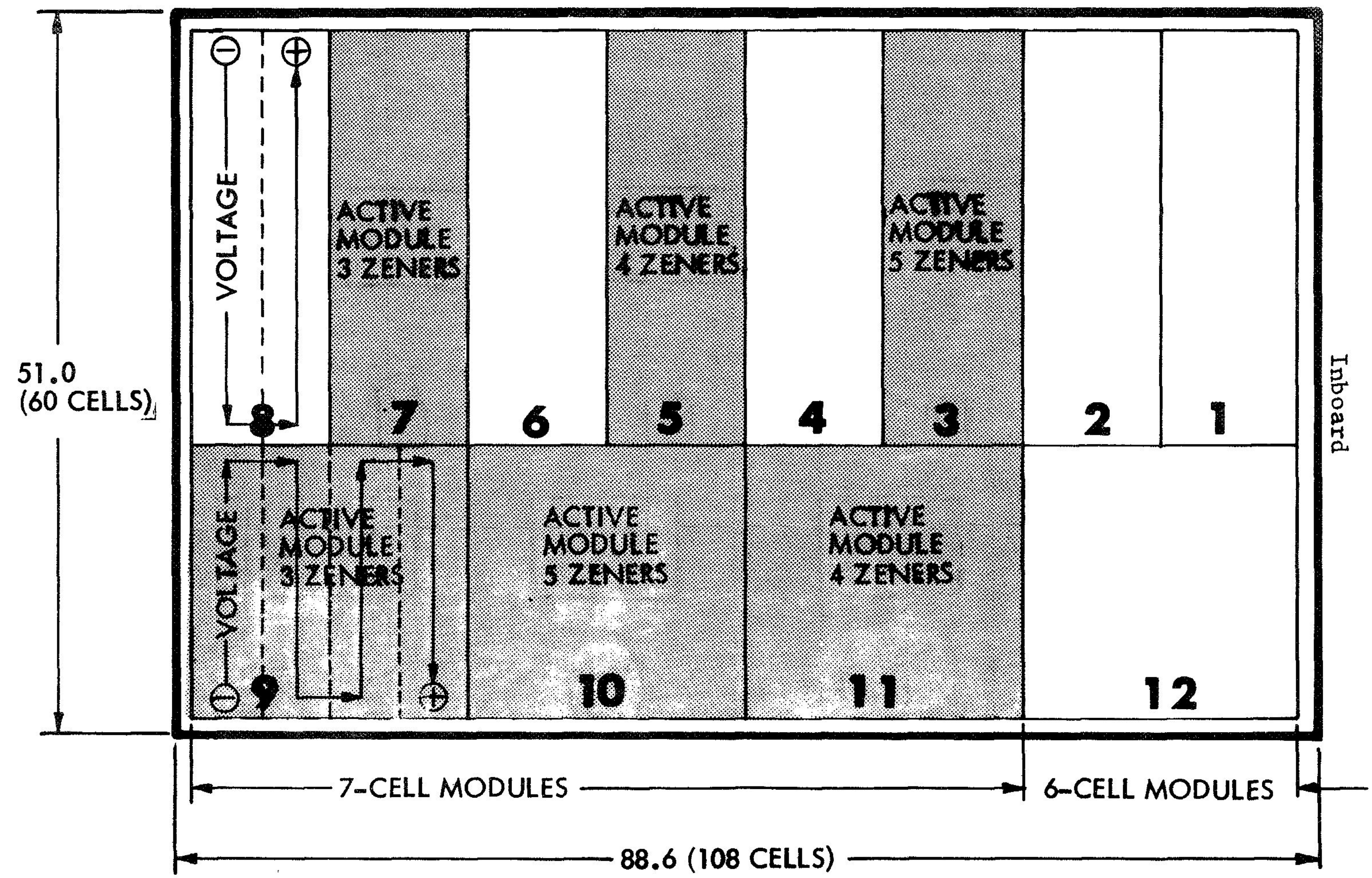
The module terminals are located adjacent to the power buses. The negative terminals are connected directly to the buses but the positive terminals go to the buses through blocking diodes. Solar cell layout and power buses are designed to minimize electromagnetic fields.

The solar cells are 2 x 2 cm N/P solderless silicon, .008 inch thick. Coverglasses are 0211 microsheet and have no filter or optical coating. Trade study weights were based on .006 inch thick coverglasses. The cells are electrically connected with .002 inch expanded silver mesh. The interconnector shape is shown in Figure 4-5. Pulse soldering is used to attach the interconnector to each cell. Cell spacing is .010 inch between parallel cells and .020 between series cells. This spacing provides a cell density of 223 cells per square foot.

4.1.2.2 ZENER DIODE INSTALLATION

Each module is electrically connected to five 10-volt zener diodes in series. This limits the maximum module voltage output to 50 volts at the bus. The zener diodes are mounted on beryllium brackets which, in the trade study configurations, were attached to the longitudinal edge members. Each diode is on a separate bracket to minimize interference with the panel structural characteristics. The brackets are attached to the structure with RTV 630 adhesive.

Beryllium was chosen as the heat sink for the diodes because of two exceptional characteristics, high thermal conductivity and high specific heat per unit weight. This is shown in Figures 4-6 and 4-7. Beryllium also provides a perfect match for the thermal coefficient of linear expansion of structure. All faying surfaces of the



3 MODULES 6 x 80 = 1,440 CELLS

9 MODULES 7 x 80 = 5,040 CELLS

TOTAL PER PANEL = 6,480 CELLS

CELLS PER ARRAY = 25,920

ACTIVE SOLAR CELL AREA = 106 SQ FT

SOLAR CELL MODULE AREA = 116 SQ FT

DARK SIDE SHOWN

FIGURE 4-4: CELL MODULE ARRANGEMENT

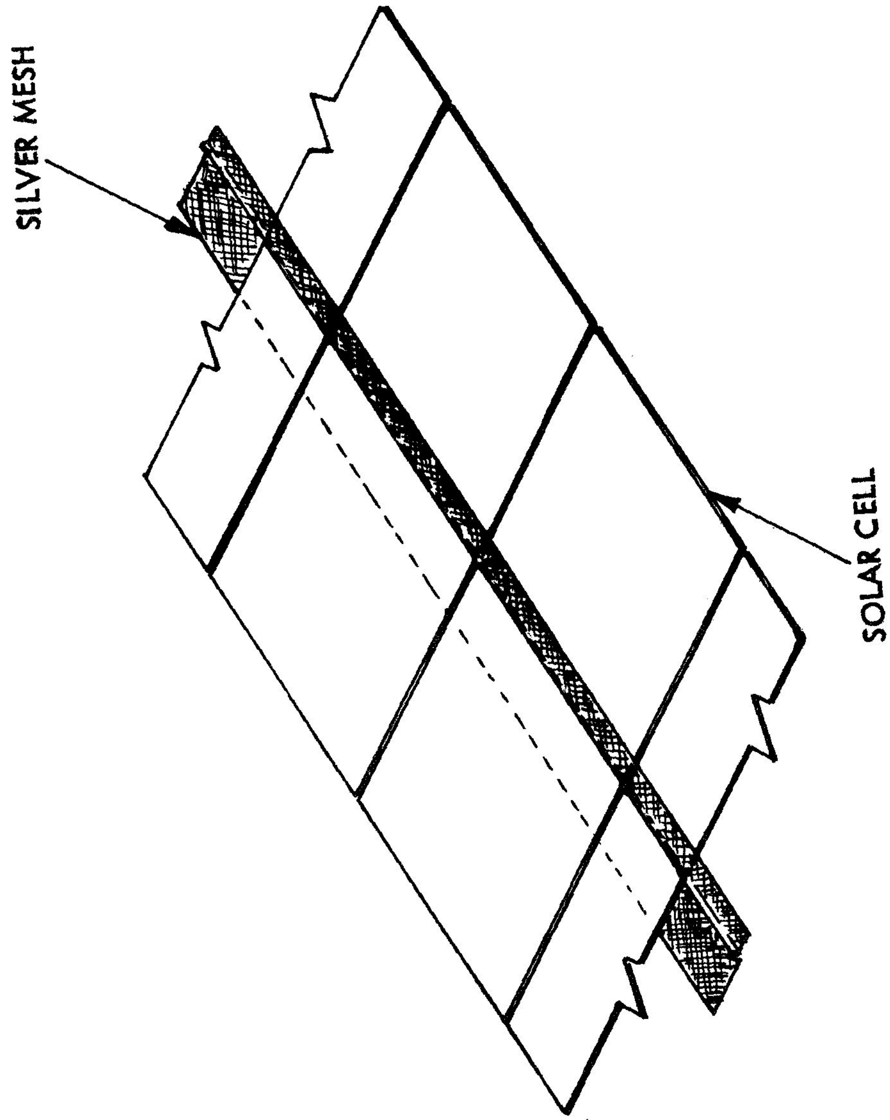


FIGURE 4-5: SOLAR CELL INTERCONNECTOR ASSEMBLY

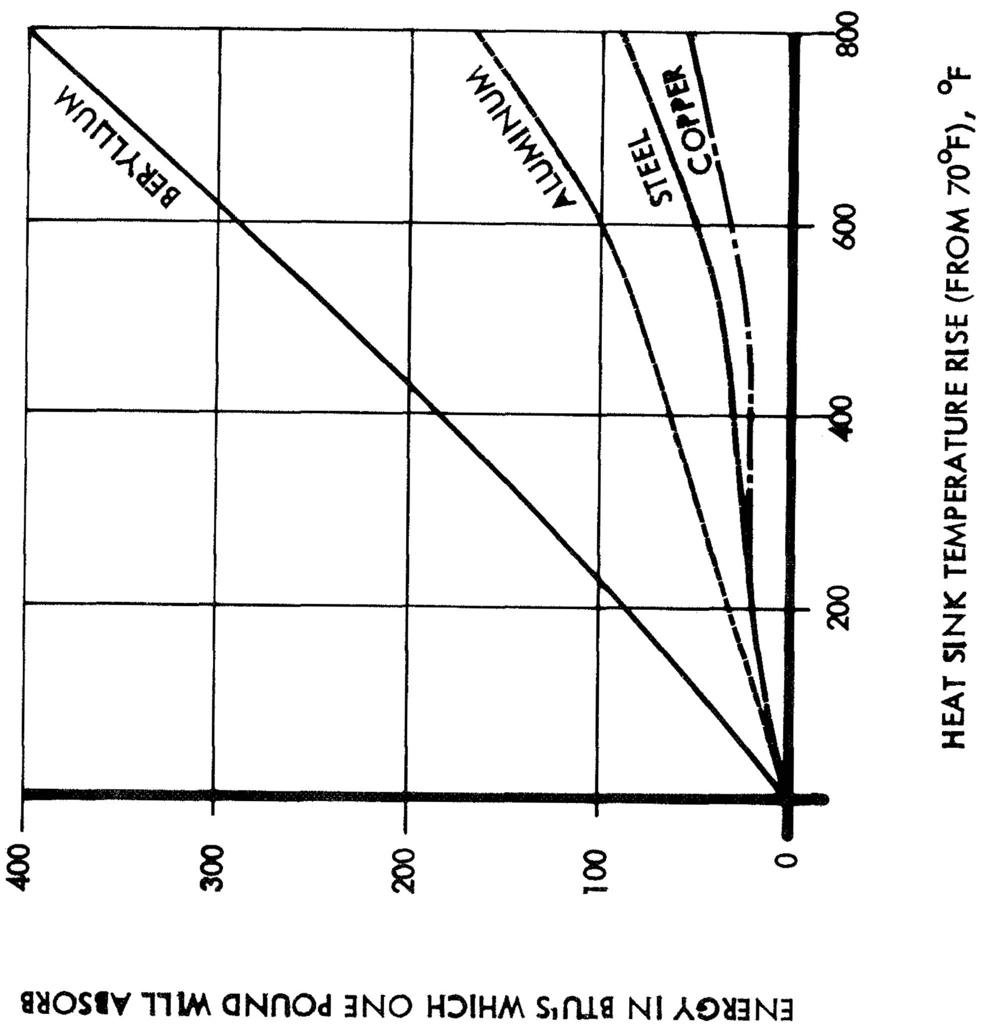


FIGURE 4-6: BERYLLIUM AS A HEAT SINK

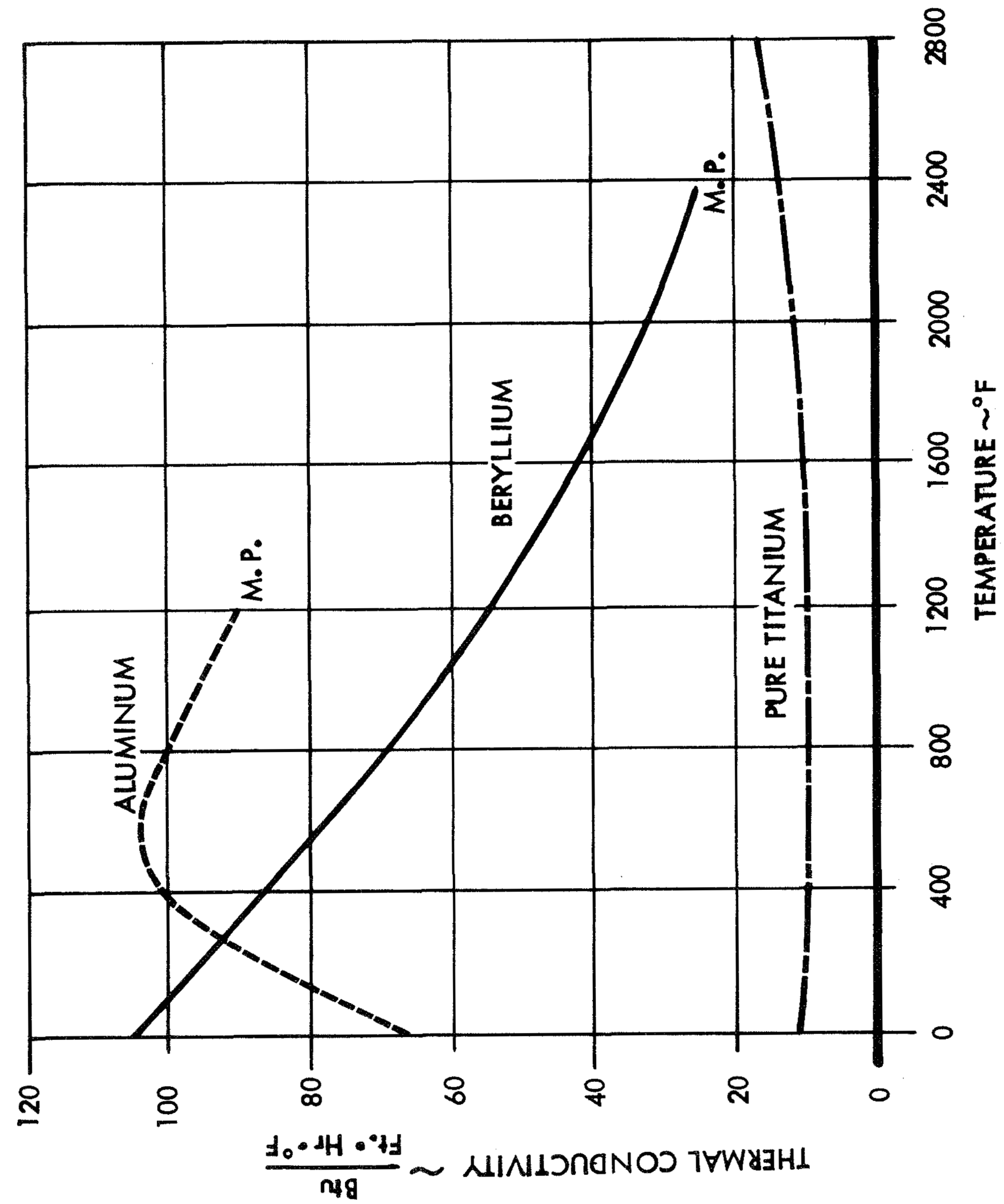


FIGURE 4-7: THERMAL CONDUCTIVITY

zener diode assembly will be coated with RTV-40 on installation to gain maximum heat transfer to the heat sink. The diode and bracket will also be coated to improve emittance.

4.1.2.3 ELECTRICAL POWER BUSES

The electrical power buses are integrated with structure. Bus assemblies are made from alternate layers of Kapton film (1 mil), thermoplastic polyester resin (1 mil) and copper strip (5 mils). The film is wider than the copper so the conductors are completely encapsulated. The assembly is attached to structure with RTV-630, a silicone elastomer.

The inboard power terminal and bus construction is shown in Figure 4-8. The terminal is a small fiberglass angle which is attached both to structure and to the bus. Wire terminated with airplane type wire lugs conducts power to the spacecraft.

4.1.3 MECHANICAL

Flight-suitable mechanism designs were included in the panel trade study configurations.

The mechanical equipment for the trade study panel configurations included the boost restraint system, deployment system, a support truss concept, a cruise latch and damper mounting, a relay antenna mass and its deployment system. The mechanical design effort also included the mass-simulated equipment. Recommendations resulting from the PDR deleted the majority of the mechanical design responsibility. The remaining items are discussed in Section 5.3.

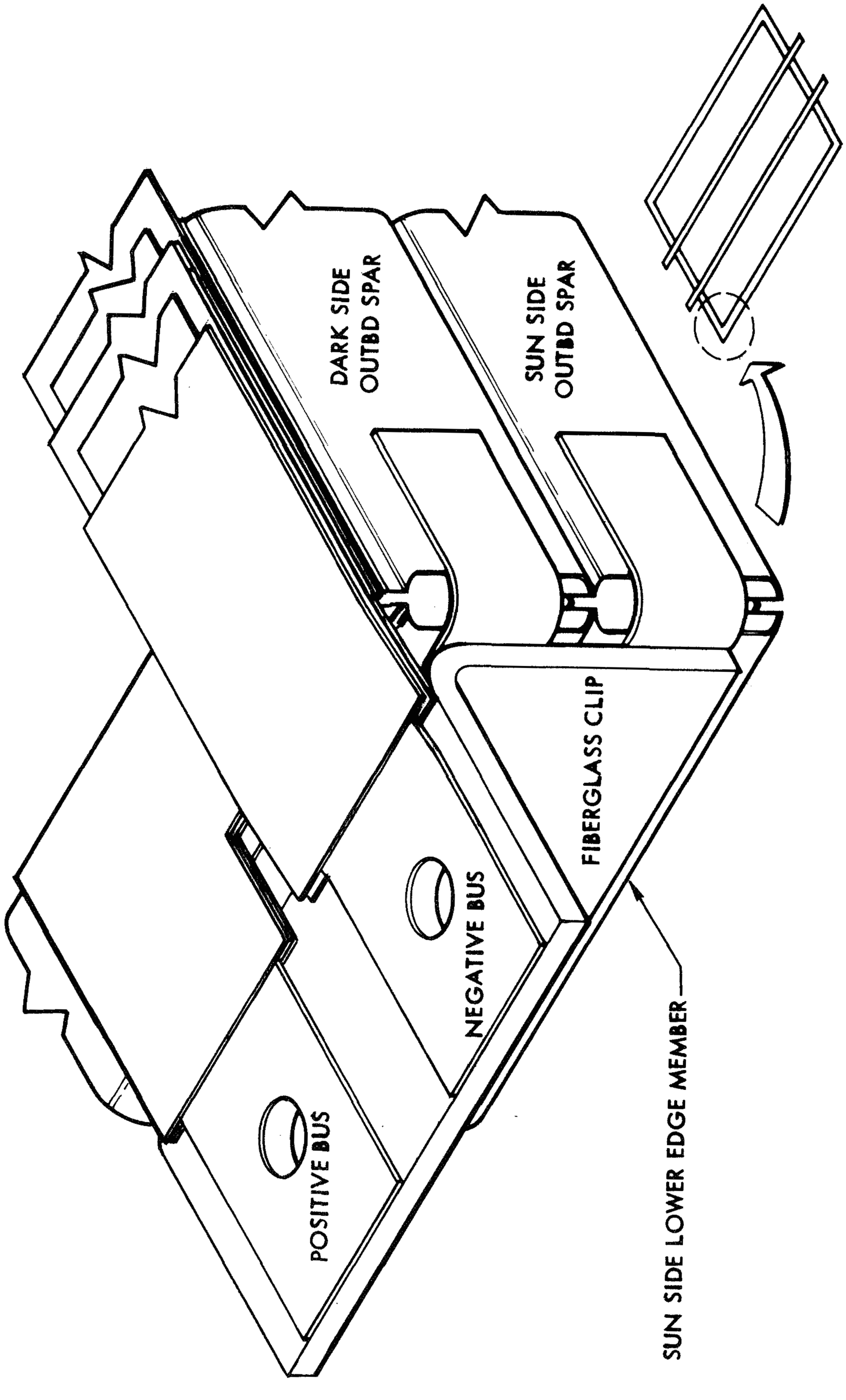


FIGURE 4-8: POWER BUS TERMINAL

4.1.3.1 BOOST RESTRAINT SYSTEM

The boost restraint system provides tip damping in the stowed attitude and automatic release for deployment upon separation from the booster.

A ground rule was followed in the preliminary design to use dampers previously designed by JPL for use on other Mars mission spacecraft if at all possible. Modifications to this hardware were to be minimized in order to reduce small component testing to a minimum consistent with the program objectives.

The boost restraint design provides four dampers per panel. Two are mounted in the panel shear plane and two out of plane. One shear plane and one out of plane damper make up a damper assembly which is mounted on a spacecraft support strut as shown in Figure 4-9. The installation shown is symmetrical. The design prevents transfer of booster-to-spacecraft loads through the solar panels. The design also provides an extractor to strip the damper fitting from the solar panel attachment pin upon spacecraft separation from the booster. Panel deployment is prevented by the boost restraint system until spacecraft separation at which time panel deployment is initiated when the solar panel pin leaves the damper fitting.

4.1.3.2 DEPLOYMENT SYSTEM

The deployment system includes hinges, deployment springs, a speed control device, a cruise latch and a cruise damper.

Figure 4-10 provides a layout of the deployment system. A truss structure concept is shown which provides the proper boost and deployed panel element locations. The panel to spacecraft hinge bearings are self-aligning and are coated with a baked-on molybdenum disulfide space lubricant. One bearing installation will contain the mounting and interface for the deployment springs. This bearing will also provide

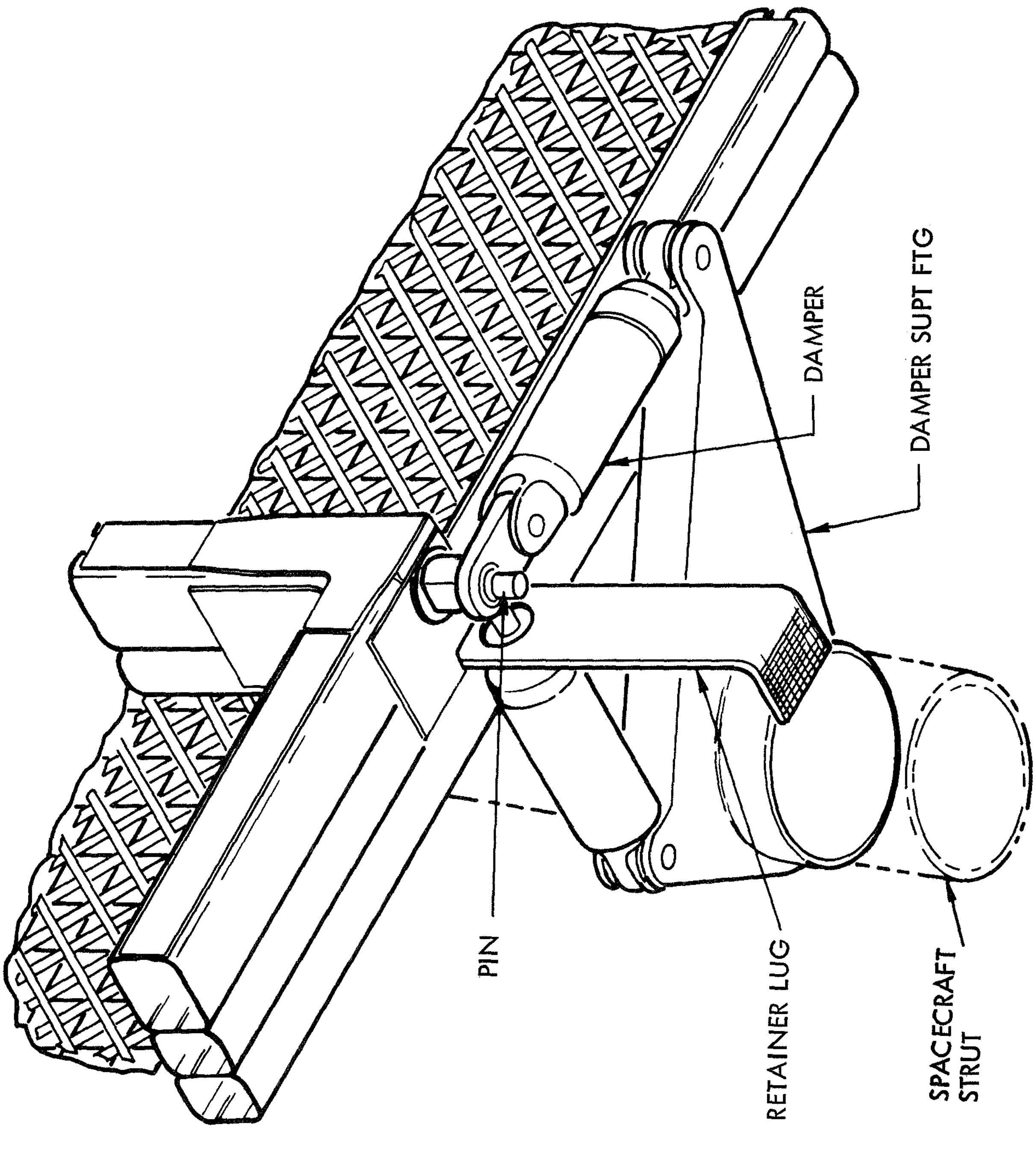


FIGURE 4-9: PANEL TIP SUPPORT

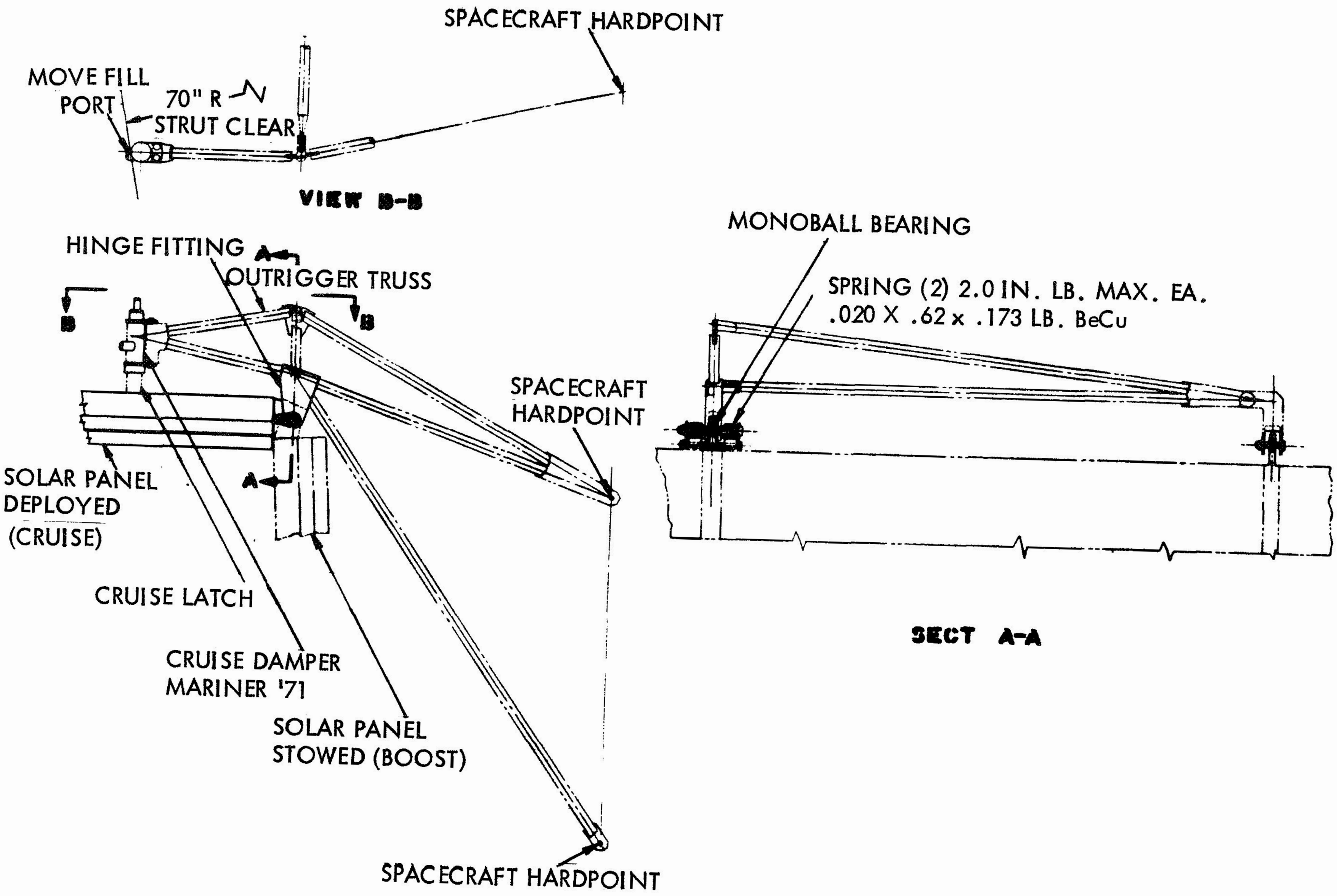


FIGURE 4-10: SOLAR PANEL DEPLOYMENT SYSTEM

the shear restraint for both the stowed and deployed modes. The second bearing is mounted with end play clearance to prevent thermal changes from affecting the system. A rotary dashpot type device may be added at this bearing location to limit the angular rate of deployment.

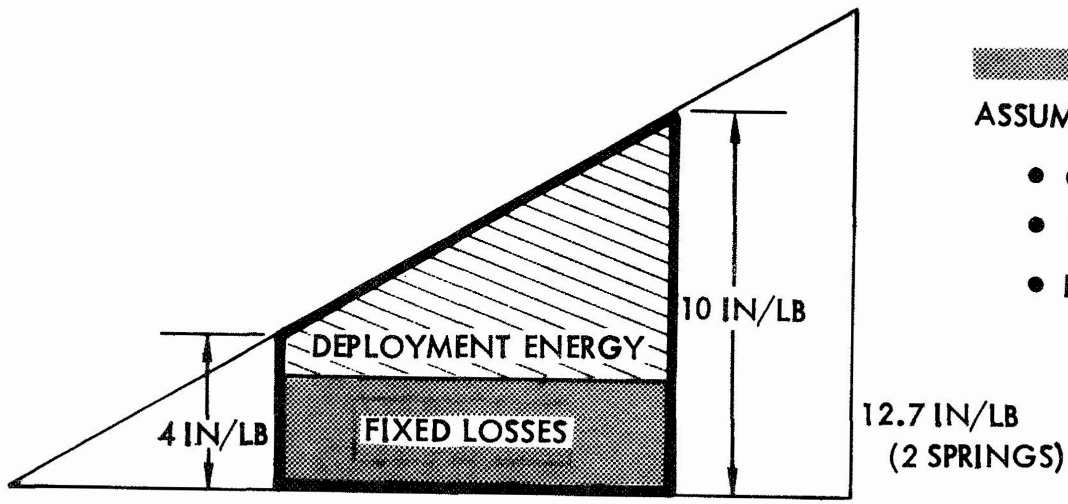
The deployment springs must be powerful enough to deploy the panel and its attached ancillary equipment and to overcome the bending resistance of electrical and coaxial cabling as well as the pneumatic swivel (or hose) on the hinge line. Figure 4-11 shows one spring design and its characteristics. Fixed losses were assumed to be 3-inch-pounds for this case. The need for a deployment angular rate limitation device was not settled during the preliminary design and analysis because; 1) the fixed losses could not be determined with any accuracy; and 2) the fixed loss repeatability was unknown. Damping available from the Mariner cruise damper during latching was also unknown. The beryllium panel is sensitive to closing rates and deceleration forces. A maximum closing rate will be established by analysis as described in Section 5.

A ground rule to use the Mariner '69 cruise damper and latch if the damping ratio and spring rate were compatible with the beryllium panel was followed. The deployment system (Figure 4-10) shows the cruise damper mounted at an 8 1/8 inch radius from the hinge pin. Subsequent analysis, with the relay antenna omitted, has shown that this radius can be reduced which will eliminate any interferences with the shroud. One latch and damper per panel (4 per spacecraft) are required in this design.

4.1.3.3 RELAY ANTENNA SIMULATION

A more detailed preliminary design of the relay antenna was necessary to provide meaningful dynamic and thermal analyses.

The initial design objective for this device was to provide a mounting on the panel for a mass simulated antenna capable of being deployed to a position of 45 degrees



FIXED LOSSES:
ASSUMED TO BE 3 IN LB TORQUE
- COAX SWIVEL
- ELECTRIC CONDUCTORS
- PNEUMATIC SWIVEL

MATERIAL: BeCu
SIZE: .036 x .62
TORQUE(MAX): 6.35 in/lb Each
DESIGN STRESS: 50,000 psi

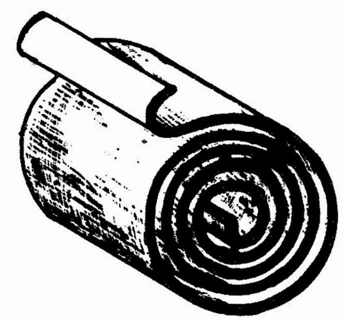


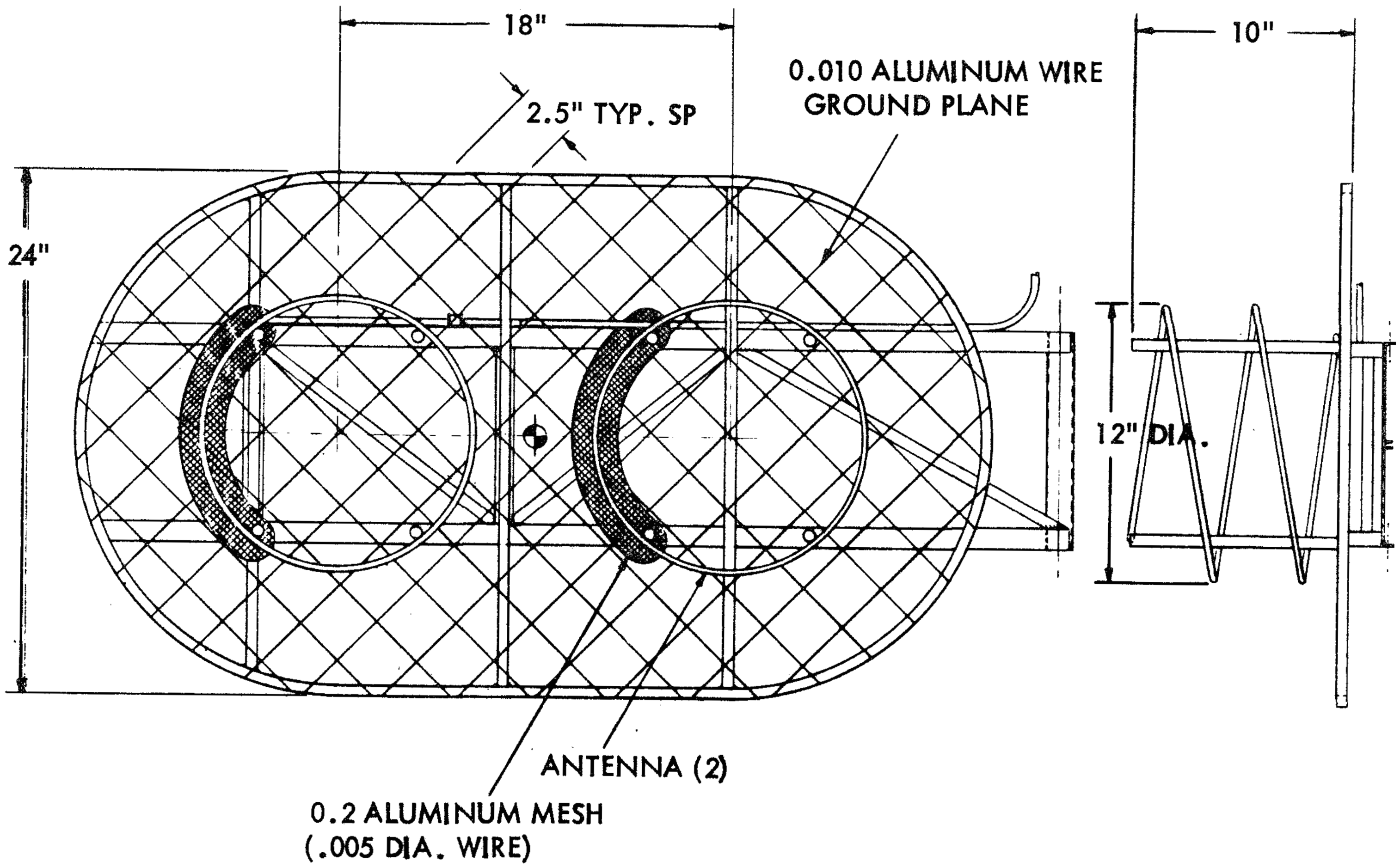
FIGURE 4-11: DEPLOYMENT SPRING

from the plane of the panel. As the preliminary design proceeded and the dynamic analysis began, it became evident that more detailed information was required if a meaningful dynamic analysis was to be completed.

Coordination with JPL and the Boeing antenna group supplied sufficient detail information to complete a preliminary design and weight distribution for a twin helix deployable antenna. Figures 4-12 and 4-13 show the antenna design and its mechanisms. The antenna structure consists of an aluminum frame of square cross section tubing. The ground plane is made from small diameter aluminum wire. The helix antenna elements are aluminum tubing. Support for these elements is provided by cross members beneath the ground plane. Shear stiffness for the antenna frame is provided by diagonal bracing of aluminum.

The antenna deployment system consists of a hinge and pin assembly using teflon bushings as journal bearings. A two-piece articulated strut provides the deployment action and serves as an open strut. It is driven by redundant torsion springs. There is no deployed latch as such. The deployment springs continue to apply a torque to the strut holding it open against its stop. Reduced spring material stress levels for the long term cruise assures a long service life for the spring.

The antenna will be deployed automatically on earth command. During boost, cruise and Mars orbit injection, the antenna will be restrained by its hinge assembly and a clevis locked by an ordnance pin puller. To deploy, a voltage is supplied to the ordnance bridge wire which fires an ordnance charge. The pin is retracted and the antenna is free to deploy. Attitude control loads are the only forces which must be overcome by the support structure after antenna deployment. The weight of the antenna and its ancillary equipment is approximately 10 pounds. The weight distribution and stiffness values obtained from this preliminary design were suitable for panel dynamic analysis purposes. The relay antenna is omitted from the Test Panel design. Had it been included, both a dynamic simulation and a thermal simulation of the antenna would have been necessary.



PLAN VIEW

END VIEW

FIGURE 4-12: DUAL HELICAL RELAY ANTENNA

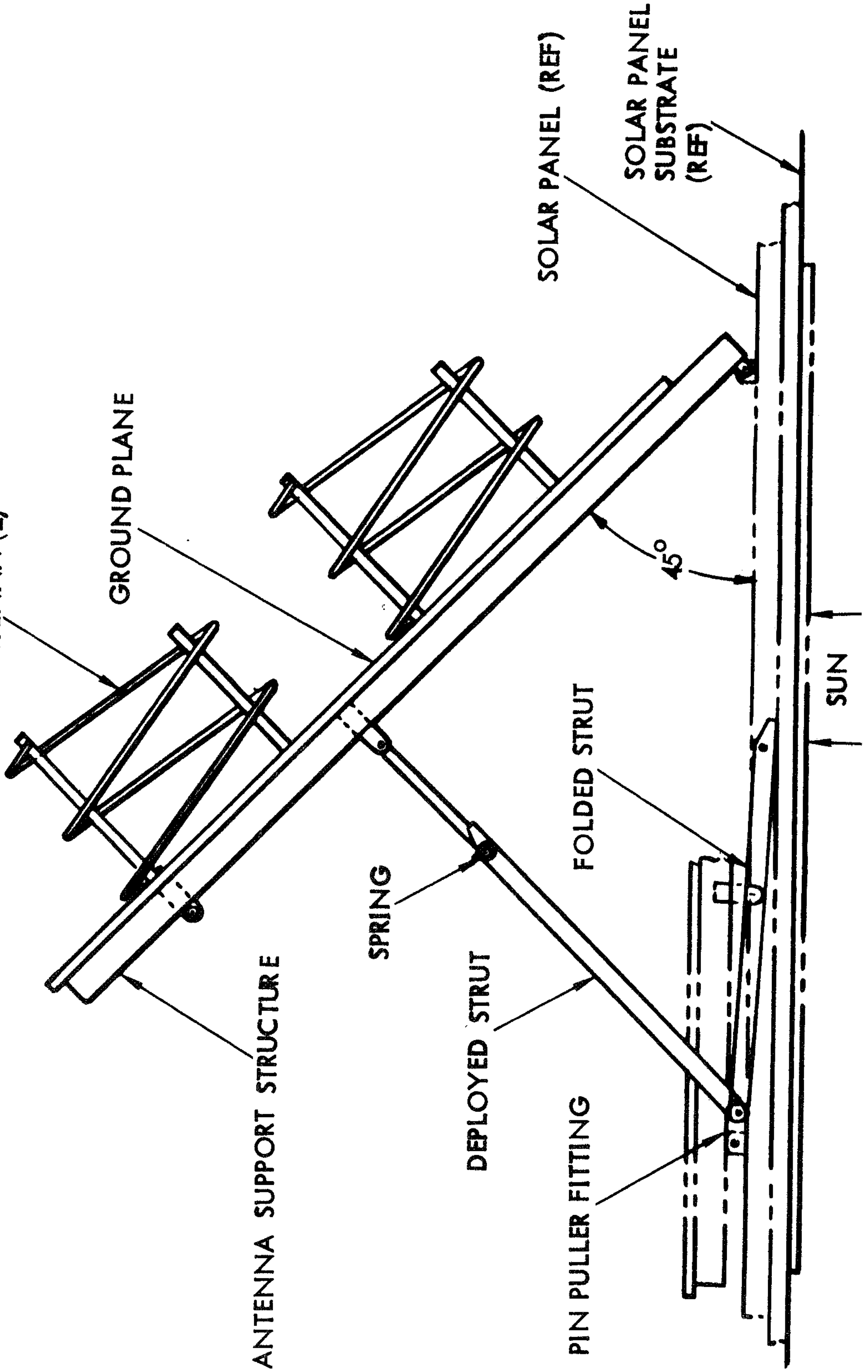


FIGURE 4-13: ANTENNA DEPLOYMENT MECHANISMS

4.1.4 ANALYSES

Analyses are required to establish the panel capability to meet contract requirements.

Analyses have been conducted to determine that the trade study panel configurations meet the contractual analysis requirements (see Table 4-2). A trade study to determine the effect on panel weight of a configuration with and without added equipment weights was specified in the Boeing Proposal Document. The trade study analyses were conducted on the original Proposal Configuration and on the PDR Baseline Configuration and the PDR Alternate Configuration A. Margins of safety were determined for static load conditions and for a dynamic test condition. A thermal analysis, based on LASA data, provided temperature effects caused by the added equipment weights.

4.1.4.1 DESIGN PHILOSOPHY

The optimum configuration support condition was selected by using a pin-free condition.

The minimum panel weight was obtained by designing the panel to meet the static and frequency requirements and providing sufficient damping to control the dynamic stresses. The selected support condition during boost allows tip motion for damper effectiveness by connecting the panel tip to "ground" through a damper spring. The resulting pin-free mode has a node near the antenna center of gravity, thus reducing its effect on the excitation of the fundamental bending mode. The shape also provides a reduced excitation of the bending mode as compared to a pin-pin configuration when excited by the specified uniform translation acceleration.

CONFIGURATION	STATIC LOADING	REQUIRED FREQUENCY	RESPONSE ANALYSIS	TEST SWEEP
PIN-PIN	8 g	> 20 Hz		
3 PT SUPPORT	50# CORNER LOAD			
DEPLOYED		>1 Hz WITH .7 C/C_c		
PIN-FREE			MODAL 7-100 Hz	7-30 at .7 g 30-400 LINEAR TO 2.8 g
ALL	1 g			

TABLE 4-2: REQUIRED STATIC AND DYNAMIC ANALYSIS

4.1.4.2 PROBLEM IDEALIZATION

Solution of the problem with a digital computer requires some assumptions and approximations.

Solution for the deflections and member loads due to static requirements and for the resonant mode shapes, frequencies and dynamic loads at the normalized deflections are obtained with the Boeing ASTRA (Advanced Structural Analyzer) computer program. The program is designed to analyze large complex structures using the direct stiffness matrix method and is written for the IBM 360 computer. It is essentially an improved version of the COSMOS program used during the Large Area Solar Array contract. It is similar to the JPL "SAMIS" program.

The program defines structural members by "nodes" at each end of the member. The node positions are defined as coordinates relative to the basic set of coordinate axes, with provision to offset the neutral axes of the members from the coordinate plane. Six degrees of freedom can be specified for a member of each node, with provision to reduce out at an elemental level any degree of freedom. The nodal diagram used is shown in Figure 4-14.

The basic rectangular framework is idealized with beam elements between the nodes. The substrate bays are represented by an "overlay" of plate elements which provide only shear stiffness equivalent to that of the fiberglass diagonal tapes for in-plane vibrations. For out-of-plane vibrations, the substrate stiffness is represented by a pair of diagonal beams having only bending stiffness. Short stiff beam elements are used for the damper fittings (outboard support points), for the hinge fittings, and for the supports required for the sun sensor and the guidance and control jet assembly.

The distributed weight is represented by concentrated weights at the nodes. It is assumed that 1/2 the weight of each member ending at a node is effective at the node. For static loads, 1/4 the weight of each substrate bay is assigned to the

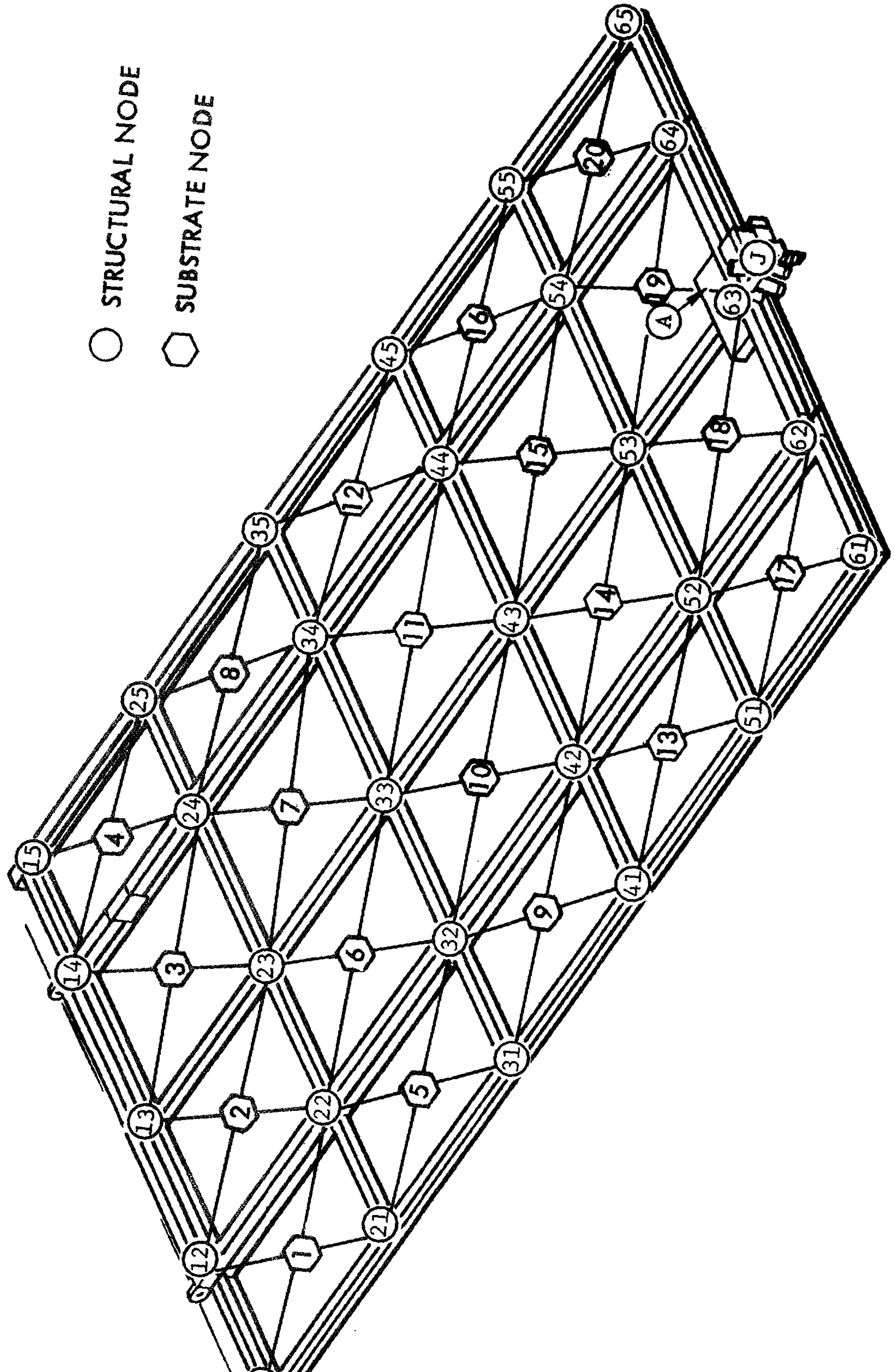


FIGURE 4-14: COMPUTER IDEALIZATION

corner structural nodes. For in-plane (shear) dynamic analysis, the substrate weight is distributed to the corner nodes, as for the static loads. For out-of-plane vibration, the early analyses assumed that the generalized mass for the fundamental substrate mode (1/4 of the total substrate mass) was at the intersection of the diagonals, with the remaining 3/4 distributed to the corners.

The antenna was simulated by rigid links connected to the structure attach points, and by mass properties defined by concentrated masses at the proper C.G. positions with appropriate mass moments of inertia about the C.G.

The structural framework member cross sections are defined and the structural stiffness characteristics of each member are defined in the computer input by the cross section area, torsional stiffness, shear areas and bending (stiffness) moment of inertias in two directions, and by elastic properties of the material.

The stiffness of the diagonal beams representing the substrate in out-of-plane bending was selected to result in a specified frequency when loaded with the generalized mass at the intersection of the beams. The frequency was determined by scaling the measured frequency in air from LASA tests. The analyses presented in this report were made with the substrate modes suppressed by increasing the beam stiffness by a factor of 10, raising the lowest resonance to above 100 Hz. Early analyses with the substrate modes included showed a band of 20 closely spaced frequencies starting at 42.5 Hz, effectively masking the structural modes in the region. Because the ASTRA program cannot include damping, the effect of the substrate modes on resonances within the 20-frequency band is exaggerated. To provide visibility for the basic structural modes, the unrealistic undamped response of the substrate was removed.

The weights appropriate to each structural mode for the static load case are tabulated in Table 4-3 for the initial proposal configuration and the trade study configurations.

TABLE 4-3

PANEL LUMPED WEIGHT DISTRIBUTION

STRUCTURAL WEIGHT (LBS)				SUBSTRATE WEIGHT (LBS)			
NODE	PROPOSAL	PDR	TEST	NODE	PROPOSAL	PDR	TEST
11	.342	.311	.223	1	.072	.072	.054
12	.384	.378	.733	2	.095	.095	.072
13	.358	.389	.487	3	.095	.095	.072
14	.384	.378	.733	4	.072	.072	.054
15	.342	.311	.223	5	.073	.073	.055
21	.570	.524	.344	6	.097	.097	.073
22	.523	.491	.850	7	.097	.097	.073
23	.607	.603	.568	8	.073	.073	.055
24	.523	.491	.711	9	.074	.074	.056
25	.570	.524	.344	10	.099	.099	.075
31	.573	.531	.351	11	.099	.099	.075
32	.529	.503	.858	12	.074	.074	.056
33	.613	.611	.576	13	.074	.074	.055
34	.529	.503	.605	14	.099	.099	.073
35	.573	.531	.351	15	.099	.099	.073
41	.574	.532	.351	16	.074	.074	.055
42	.532	.584	.861	17	.073	.073	.054
43	.615	.772	.575	18	.097	.097	.072
44	.532	.584	.664	19	.097	.097	.072
45	.574	.532	.351	20	.073	.073	.054
51	.575	.529	.344				
52	.531	.499	.850				
53	.613	.609	.571				
54	.531	.499	.653				
55	.575	.529	.344				
61	.345	.314	.224				
62	.390	.384	.615				
63	2.002*	.611	.610				
64	.390	.384	.515				
65	.345	.314	.224				

* Included Guidance and Control Jet Assembly

4.1.4.3 STRESS CALCULATIONS

The ASTRA program printout provides the basic information needed for stress analysis.

Member loads are obtained from the output of the ASTRA program solutions. For each end (nodes) of each member, the axial loads, bending moments and shears in two directions, and torsional moments are tabulated for each static load case. Similar data are obtained for dynamic solutions, except that the loads are based on the normalized amplitude of motion. For the static cases, deflections are also determined at the defined nodes. In addition to the stresses defined by the external loading, stresses resulting from tape tension and manufacturing operations are included in the stress analysis.

The tape tension load (15 lb/in ultimate) is applied to the outer edge members and produces end loads and bending moments in edge frame members and intercostals. The main spars are also assumed to be loaded in bending with a percentage of the tape tension load.

An analysis of a typical main spar member is included in Document D2-121718-1. This illustrates the analysis techniques, the methods for calculating allowable stresses, and margins of safety. In addition to the typical analysis, the section includes a key to member designation in the ASTRA output, a definition of load sign convention used in ASTRA, a discussion of stresses resulting from manufacturing operations, and a page describing the method used for calculating the compression crippling stress of formed sections.

For the static requirements, member loads are calculated for a total load of 8 times the weight distributed to the nodes (7 g added weight plus the structural weight); and for a 50-pound load applied at one outboard support with and without 1 g times the nodal weights. Deflections calculated for the 50-pound load without the 1 g load relate to the measured deflections in the static test, and stresses calculated with the 1 g load added relate to actual stresses for the test condition.

For the dynamic cases, margins of safety are calculated for selected percentages of the normalized amplitude. This prescribes the limiting amplitude at the dampers for which positive margins are possible, and so defines the required damping force to limit the motion.

Analysis of the panel utilizes the same factors of safety and basic allowables as was used in the LASA program. All "limit" loads are multiplied by 1.25 to obtain the ultimate design load. Where appropriate, they are also multiplied by a 1.15 fitting factor.

4.1.4.4 DYNAMIC CALCULATIONS

The ASTRA program provides the basic data for dynamic analyses.

Dynamic solutions for the mode shapes and resonant frequencies of the panel in its various configurations are output from the ASTRA program, as well as the generalized inertia and stiffness matrices. Prior to the preliminary design review, a supplementary program was written to obtain stored data for pinned-free solutions from an ASTRA tape and to operate on it to obtain the complex response of the normal modes coupled by the dampers. The program provided for calculating the inertial driving forces, the incremental damping and stiffness matrices, setting up the simultaneous equations and solving for the complex response amplitudes of each mode at selected frequencies. Damping was input as an "effective" viscous damping term. Initial results from this program showed that frequency separation between importantly excited modes was good and that a simplified approach was possible.

Thereafter, the ASTRA solution obtained the pin-free modes with the tip connected to ground with the damper springs. The modal coupling, due to damping forces, of importantly excited modes was neglected and the driving inertial forces were hand calculated.

Since the sinusoidal test was defined for excitation at the hinges with a level equivalent to that for the specified uniform translation, driving forces for each important mode are required for both excitations.

The driving forces are determined in terms of an inertial coupling term between the driven mode(s) ($\emptyset_i$) and the excitation mode ($\emptyset_E$). The inertial coupling term is defined as:

$$M_{iE} = \int \emptyset_i \, \emptyset_E \, dm$$

and the excitation force as:

$$F_{iE} = \omega^2 M_{iE} \, q_E = W_{iE} \, "g"$$

where W_{iE} is the weight coupling term and "g" is the acceleration in g units. The development of this equation is explained in Document D2-121718-1.

The undamped response of the uncoupled modes is then:

$$(-\omega^2 M_{ii} + K_i) q_i + D(\emptyset_d, a) = F_{iE}$$

where $D(\emptyset_d, a)$ is the effective damper force acting on the mode at damper amplitude a.

At resonance, the term $(-\omega^2 M_{ii} + K_i) = 0$, and the panel amplitude at the damper is determined by the amplitude at which the effective damper force is equal to F_{iE} at the resonant frequency of the mode. In this analysis, the structural damping in the panel is neglected, and the damping force is assumed not to significantly change the resonant frequency.

4.1.4.5 THERMAL CALCULATIONS

To evaluate the thermal characteristics of the panel, an assumption of the relay antenna structural configuration was required.

In the proposal stage, the blocking effect of the "dummy equipment" (since identified as the relay antenna and sun sensor) was recognized as the most probable cause of a thermal problem. The same thermal protection used on the LASA panels was indicated to be adequate for the MMSA panel without antenna. Thermal interaction with the spacecraft was not considered because spacecraft thermal data was not available. Analysis was required to assess the effect of the dummy masses on cell temperatures and to determine if a significant power loss would result.

Thermal characteristics of the panel with antenna and sun sensor are primarily dependent on the "view factor" to deep space. Since no JPL antenna design had been started at the time detail information was required, a preliminary design, described in Section 4.1.3, was developed. The "mesh" baseplate and antenna design resulted from consultation with Boeing antenna specialists and was accepted by JPL for thermal evaluation. The concept was not compatible with the antenna assumptions used in dynamic analysis.

4.2 TRADE STUDY RESULTS

The trade study results showed that there was very little weight penalty and power loss between the two trade study configurations.

Table 4-4 compares the weight and power differences between the PDR Baseline and PDR Alternate panel assemblies. These results were presented to JPL during the Preliminary Design Review held at Boeing on September 23, 1969.

TABLE 4-4

TRADE STUDY RESULTS

Characteristic	PDR Baseline	PDR Alternate A	PDR Alternate B
Interchangeability	Yes	Yes	No
Power Losses at Mars due to Equipment Thermal Effect	2.0 W	0.5	2.0
Solar Array Weight			
Cell Stack and Structure	56.52	54.84	54.36
Panel Mounted Mechanisms	.28	.28	.28
Spacecraft and Launch Vehicle Mounted Mechanisms	10.12	10.12	10.12
Mounted Equipment	44.88	31.92	44.88
Total	111.80	97.16	109.64
Power/Weight Ratio at 10 watts/sq.ft.*	20.5 W/lb	21.2 W/lb	21.3 W/lb

4.2.1 STRESS RESULTS - TRADE STUDY CONFIGURATIONS

Margins of safety for the PDR baseline and alternate design are adequate.

Evaluation of the margins of safety for the two configurations at the preliminary design review are shown on Figures 4-15 and 4-16. The PDR Baseline configuration shows margins for critical members with the 10-pound antenna attached, and a structure that has reinforced lateral members at each end of the panel and at the inboard end of the antenna. Figure 4-16 shows margins for the same structure with the antenna removed, but with a sun sensor and maneuver antenna at the outboard corners. The alternate configuration was not specifically analyzed, the weight was slightly less due to elimination of the reinforcing lateral member and the antenna attachment fittings. The margins of safety were expected to be essentially the same.

*Cell Stack and Structure Weight Only

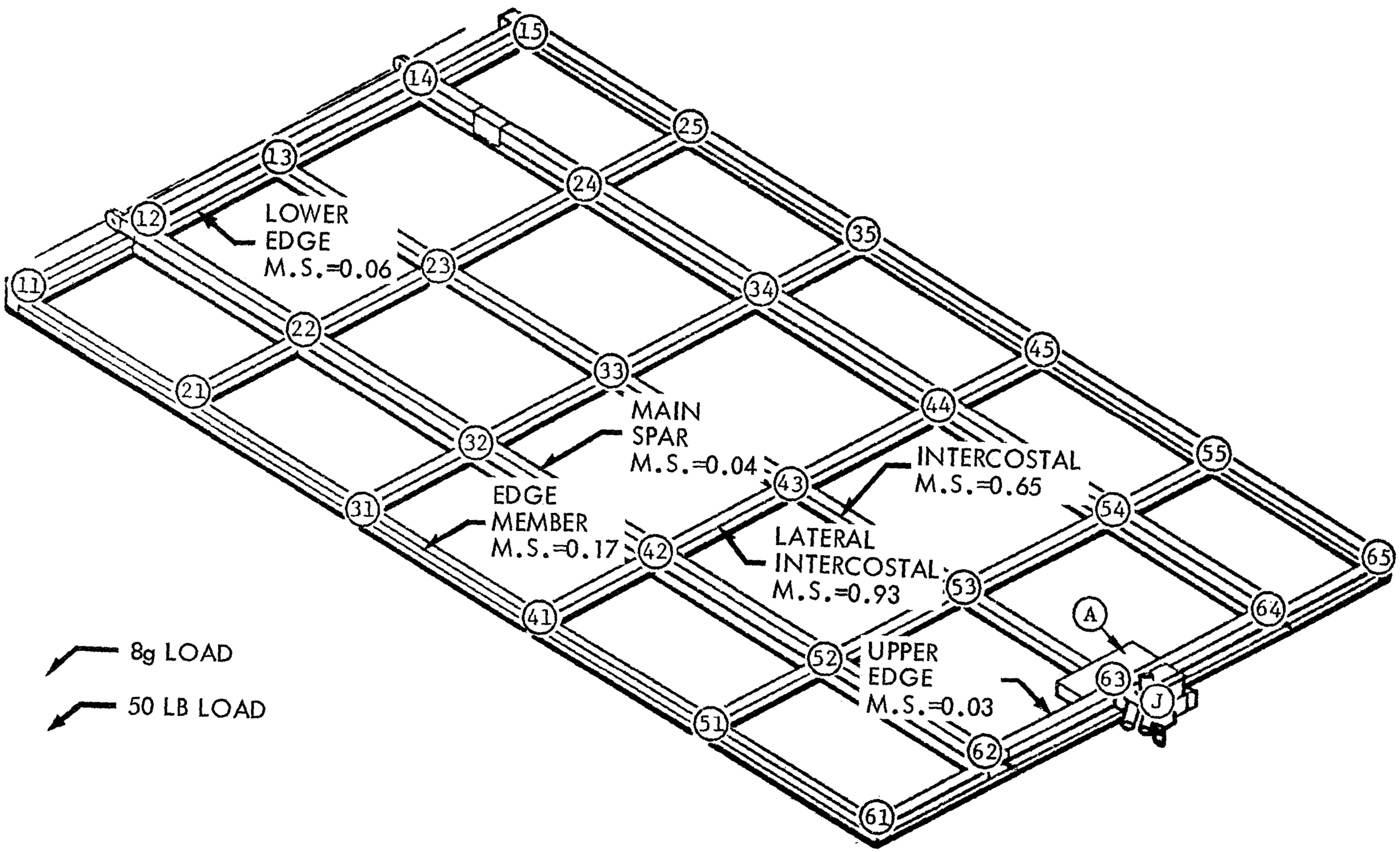


FIGURE 4-15: STATIC STRESS MARGINS - PDR BASELINE PANEL

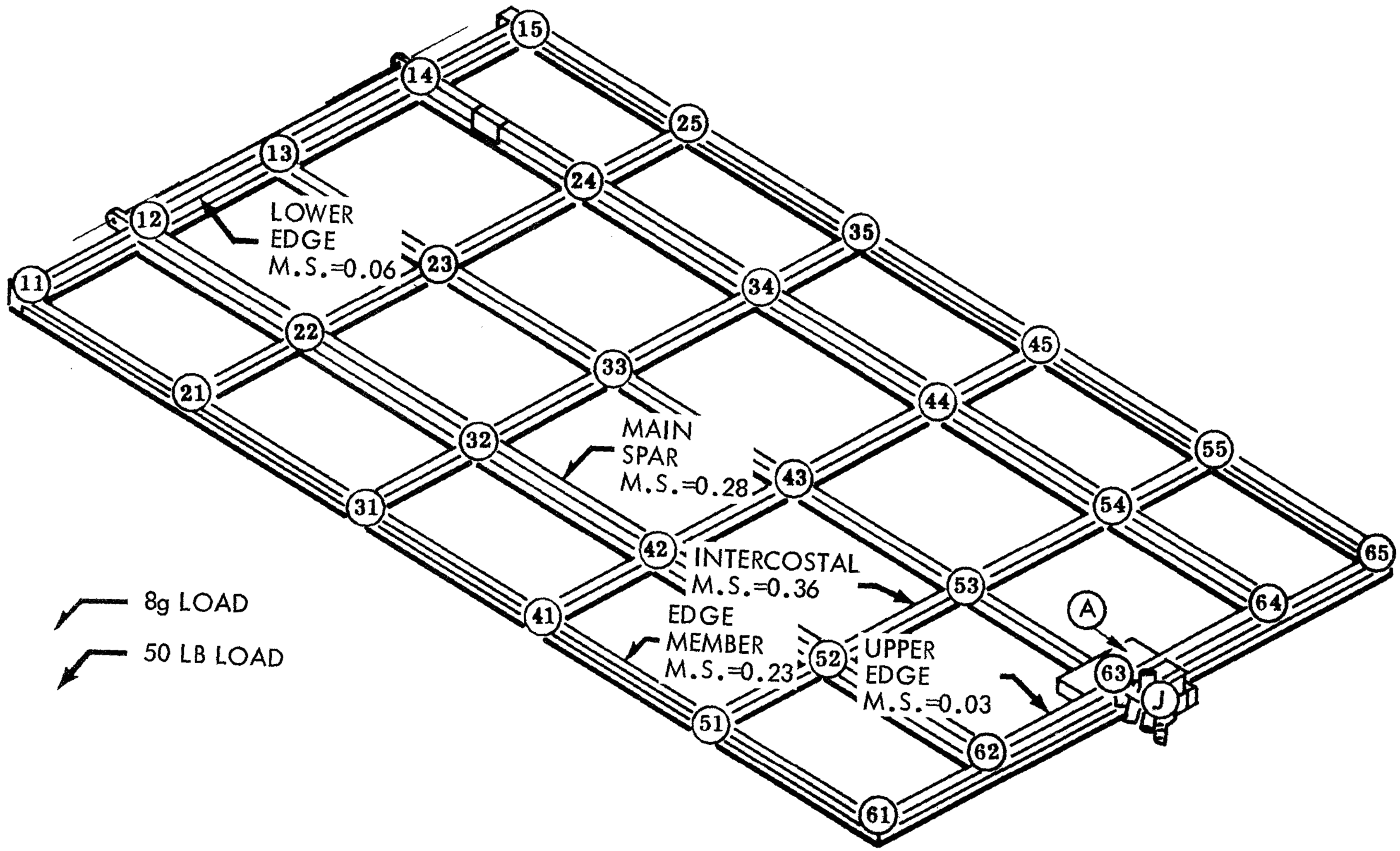


FIGURE 4-16: STATIC STRESS MARGINS - PDR ALTERNATE PANEL

For the dynamic cases, the margins of safety versus amplitude of motion at the normalizing station are calculated in Figures 4-17 and 4-18 for the important modes. These curves define the amplitude range within which the dampers must control the motion.

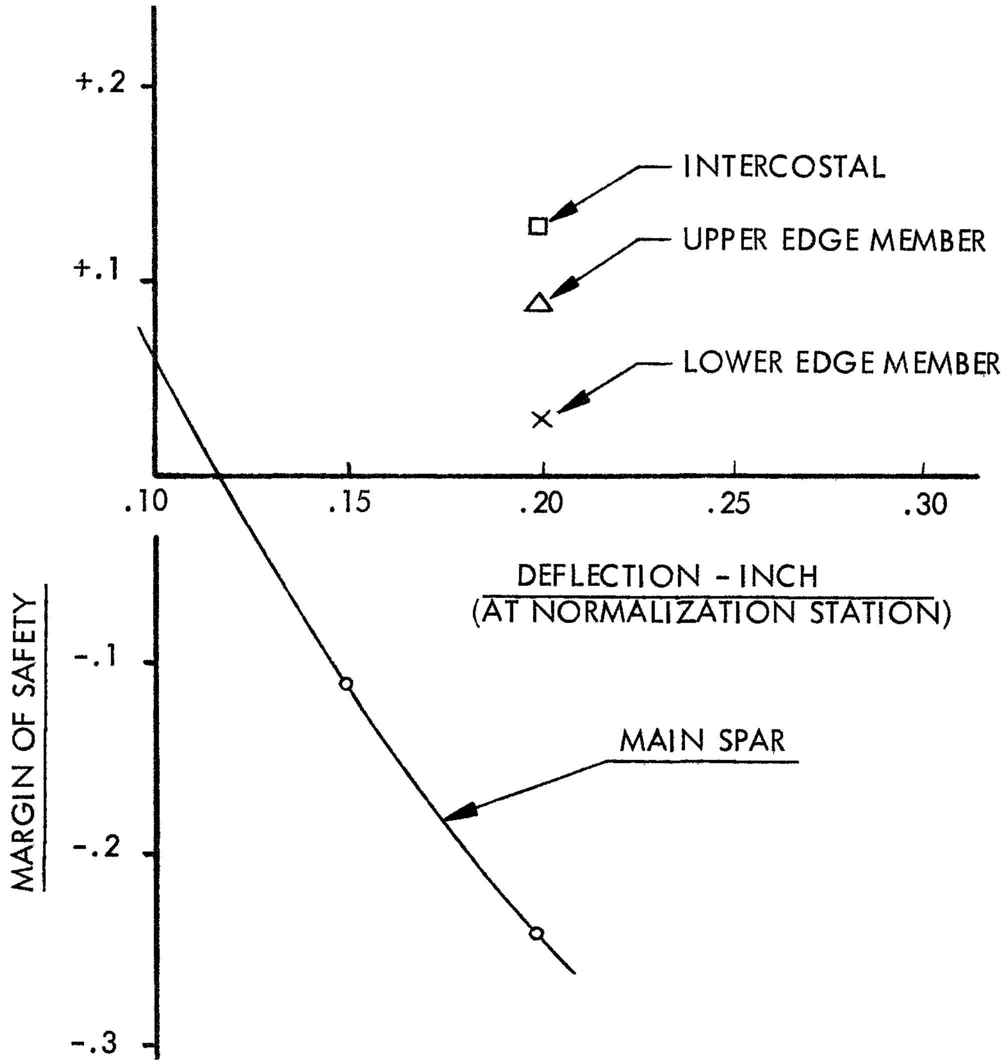
4.2.2 DYNAMIC RESULTS - TRADE STUDY CONFIGURATIONS

Dynamic requirements for the PDR configurations are satisfied.

The resonant frequencies in the pin-free, pin-pin, and the panel deployed constraint conditions are shown in Figure 4-19 for the Baseline and Alternate panel configurations. In the panel deployed condition, a solution is also obtained with the antenna deployed 45 degrees. The minimum frequency in the pin-pin condition exceeds the required 20 Hz.

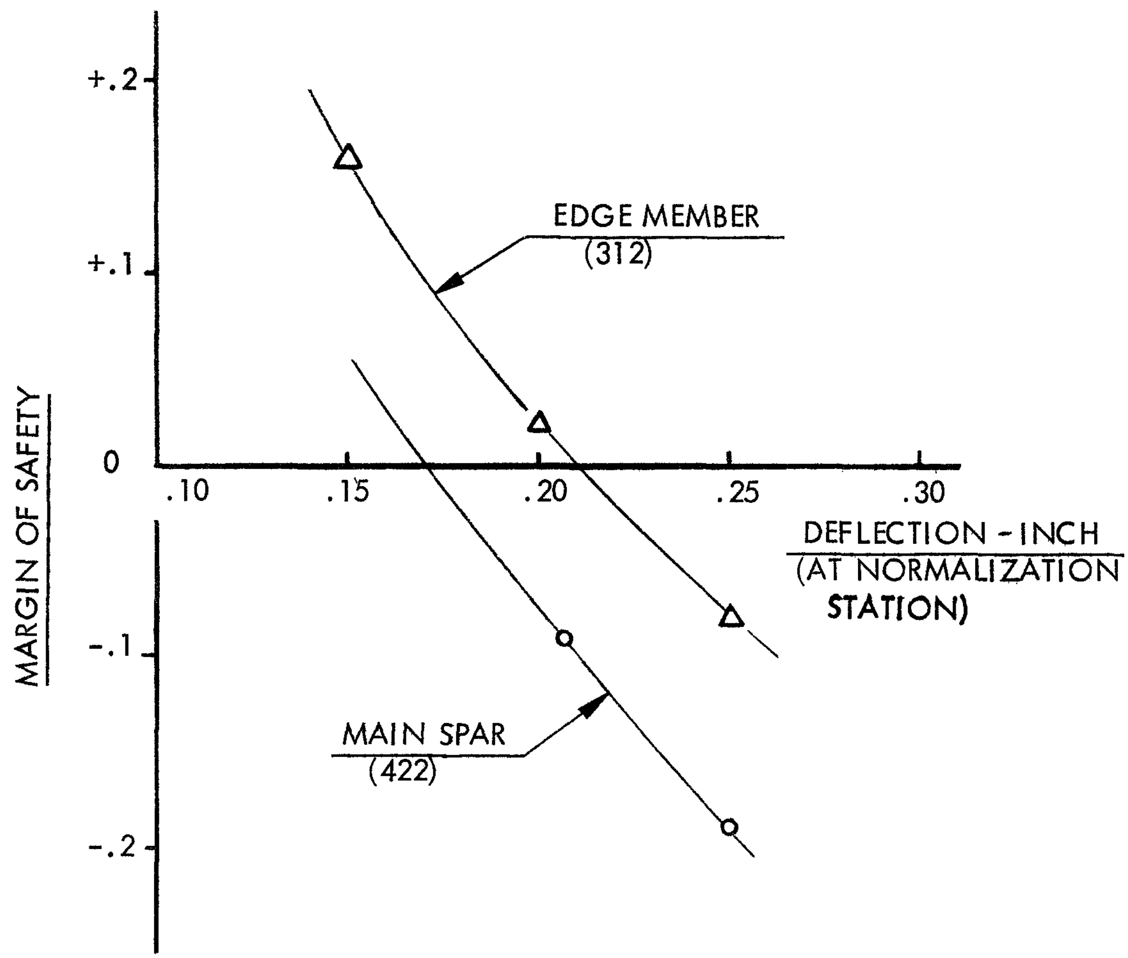
In the pin-free (boost) configurations, addition of the damper springs tailors the rigid rotation mode to the arbitrarily selected frequency (10 Hz) and results in increases in the first torsion and first bending frequencies. The higher modes are essentially not affected. Because of the large moments of inertia (129,600 lb. in.^{2} with antenna, 101,000 lb. in.^{2} without antenna) the damper spring rate required was 110 lb/in and 85 lb/in, respectively.

In the deployed case, the first frequency is determined by the boost damper spring. The ASTRA analysis was made using a torsion spring acting on one beam. The analysis torsion stiffness for the antenna-on condition was 38,000 in-lb/radian, for the antenna-off was 28,000. To obtain the required .7 critical damping, a damper is required having a rotational damping coefficient (in-lb/rad/sec) of 4150 for the antenna-on and 3220 antenna-off. On selection of the arm for a linear damper, the equivalent values are obtained by dividing by the arm squared — thus, for a linear spring constant of 530 lb/in ('69 Mariner) the required arms would be 8.5 and 7.3 inches, respectively. The equivalent linear damper coefficient appropriate to these arms would be 57.6 lb/in/sec and 60.4 lb/in/sec., respectively.



NOTE: At the damper attachment point, the allowable in-plane deflection for zero margin of safety is .115 inch.

FIGURE 4-17: PDR BASELINE PANEL - DYNAMIC MARGIN OF SAFETY VS DEFLECTION (SHEAR MODE)



NOTE: At the damper attachment point, the allowable out-of-plane deflection for zero margin of safety is .075 inch (.17 x .44).

FIGURE 4-18: PDR BASELINE PANEL - DYNAMIC MARGIN OF SAFETY VS DEFLECTION (BENDING MODE)

CONFIGURATION \ MODE	1	2	3	4	5	6	7
PIN-FREE 10 LB ANTENNA	11.8 (a)	22.9 (b)	28.7 (c)	52.6 (d)	78.1 (e)	85.6	102.3
PIN-FREE NO ANTENNA	11.4 (a)	25.0 (b)	33.5 (c)	53.3 (d)	80.3 (e)	96.3	104.9
PIN-PIN 10 LB ANTENNA	21.9 (b)	22.7 (c)	41.77 (a)	75.0 (e)	85.5	97.8	103.7
PIN-PIN NO ANTENNA	25.7 (b)	29.2 (c)	43.3 (a)	78.1 (e)	95.2	104.0	106.0
PANEL DEPLOYED 10 LB ANTENNA STOWED	1.545	11.4	22.1				
PANEL DEPLOYED ANTENNA 45°	1.457	7.93	22.5				
PANEL DEPLOYED NO ANTENNA	1.544	11.29	25.1				

(a) First Torsion (b) Shear (c) First Bending (d) Second Torsion (e) Chord Bending

FIGURE 4-19: DYNAMIC RESULTS - RESONANT FREQUENCIES - TRADE STUDY CONFIGURATIONS

With the specified .7 critical damping, the damped frequencies become 1.1, 1.05, 1.1 Hz. These exceed the required 1 Hz.

The driving forces for the pin-free resonances below 100 Hz are given for the Baseline and Alternate configuration in Table 4-5 below.

TABLE 4-5

DRIVING FORCES - PDR CONFIGURATIONS

MODE	DRIVING FORCES AT 1 "G" EXCITATION			
	PDR BASELINE		PDR ALTERNATE A	
	NORMAL EXCITATION	SHEAR EXCITATION	NORMAL EXCITATION	SHEAR EXCITATION
1	20.00	.22	14.62	
2	.32	3.50	.007	
3	.15	21.25	.002	15
4	5.66	.46	4.59	
5	.40	-.15	.03	
6	-.80	-.52	.44	

Mode 1 Rigid Rotation Mode 2 Torsion
Mode 3 Shear Mode 4 Bending

4.2.3 THERMAL RESULTS - TRADE STUDY CONFIGURATIONS

No excessive temperatures were found to occur as a consequence of the antenna or sun sensor blockage.

The anticipated temperature at selected positions on the dark side of the panel are given in Figure 4-20 for 1 AU solar illumination normal to the solar cells. The most critical temperature is 160°F, caused by the sun sensor blockage in the corner of the panel.

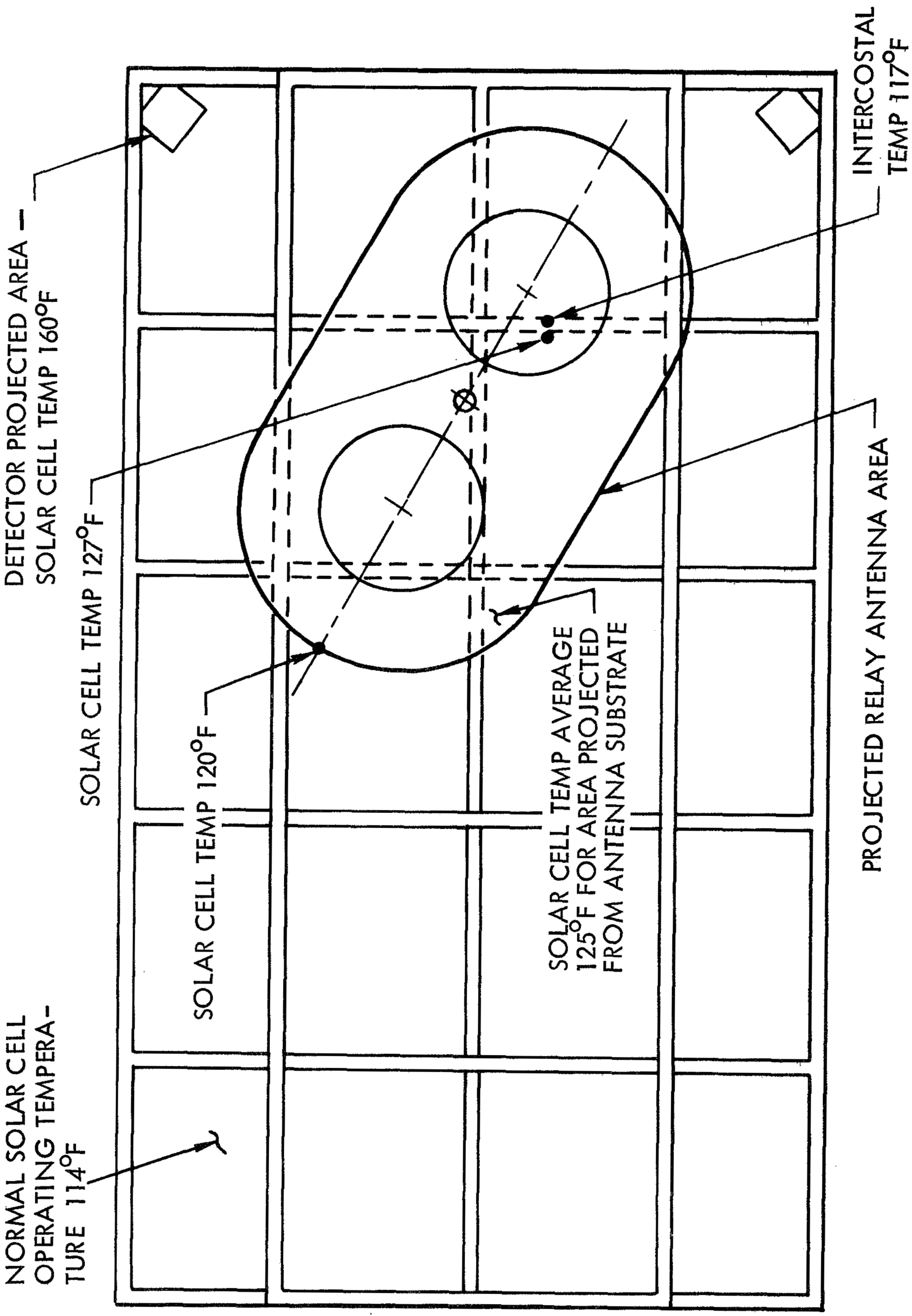


FIGURE 4-20: THERMAL ANALYSIS RESULTS

SECTION 5: TEST ARTICLE DESIGN

The design of the test article is basically unchanged from the PDR Baseline Configuration. The Test Panel configuration is shown in Figure 5-1. This panel differs from the PDR Baseline panel primarily in that the relay antenna and support brackets have been omitted. Structural member sizes and material gages are unchanged except that the lateral spar depth was reduced. Other minor changes, described herein, were made at JPL request or to reduce manufacturing costs.

5.1 TEST PANEL REQUIREMENTS

Minor changes have been incorporated into the Test Panel design.

Panel requirements for the test article are given in the JPL Statement of Work, Article I, Letter Contract No. 952571. In addition, the following changes have been negotiated.

1) Removal of the relay antenna
2) Elimination of the maneuver antenna
3) Relocation of the mass-simulated sun sensor from the outboard corner of the panel to the outboard center of the panel
4) Doubling the weight of the simulated attitude control jets to 2.8 pounds
5) Requirements for analytical dynamic re-analysis
6) Higher level acoustic test requirements
7) Changes in starting frequency of the sinusoidal vibration test
8) Substitution of 3 mil coverglasses without interference filters

The above changes have been incorporated in the Test Panel design.

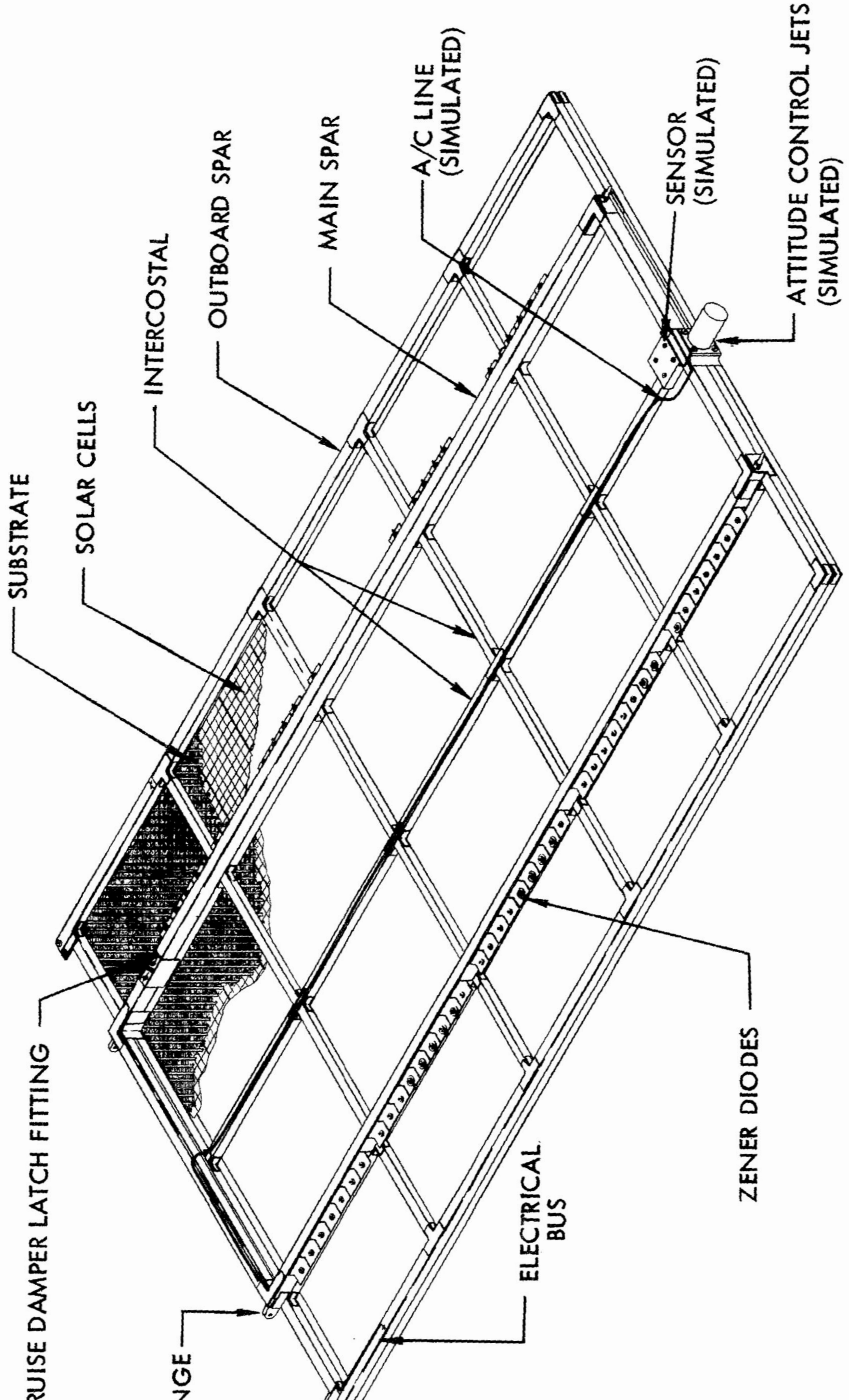


FIGURE 5-1: TEST PANEL ASSEMBLY

5.2 ELECTRICAL TEST ARTICLE

The Test Panel electrical design is the same as the flight article design, discussed in Section 4, with the following exceptions.

1) Only six modules are electrically connected.

2) Cells and coverglasses are not flight quality.

3) Two modules have five zener diodes, two have four diodes and two have three diodes. Additionally, one each of the three diode configurations will have the diodes connected to the module terminals and one each will be electrically connected to an external power supply with provisions for monitoring zener current and diode body temperatures during test.

4) Coverglass thickness was reduced from .006 inch to .003 inch.

5) The zener diodes were relocated from the edge members to the main spars. The diode mounting is detailed in Figure 5-2. Five zener diodes per module are either installed or mass-simulated for dynamic test purposes.

A schematic of the Test Panel electrical arrangement is shown in Figure 5-3.

5.3 MECHANICAL DESIGN

The Test Panel mechanical design includes only those elements required to support the test program.

The Test Panel mechanical configuration includes only main support hinges and bearings, boost restraint dampers and interface fittings and mass simulation for the cruise latch, sun sensor and the attitude control jets and tubing. The design effort is limited to the test configuration as a result of decisions at the preliminary design review held on September 23, 1969. The relay and maneuver antennas and panel deployment and cruise damping equipment are not included in the Test Panel design.

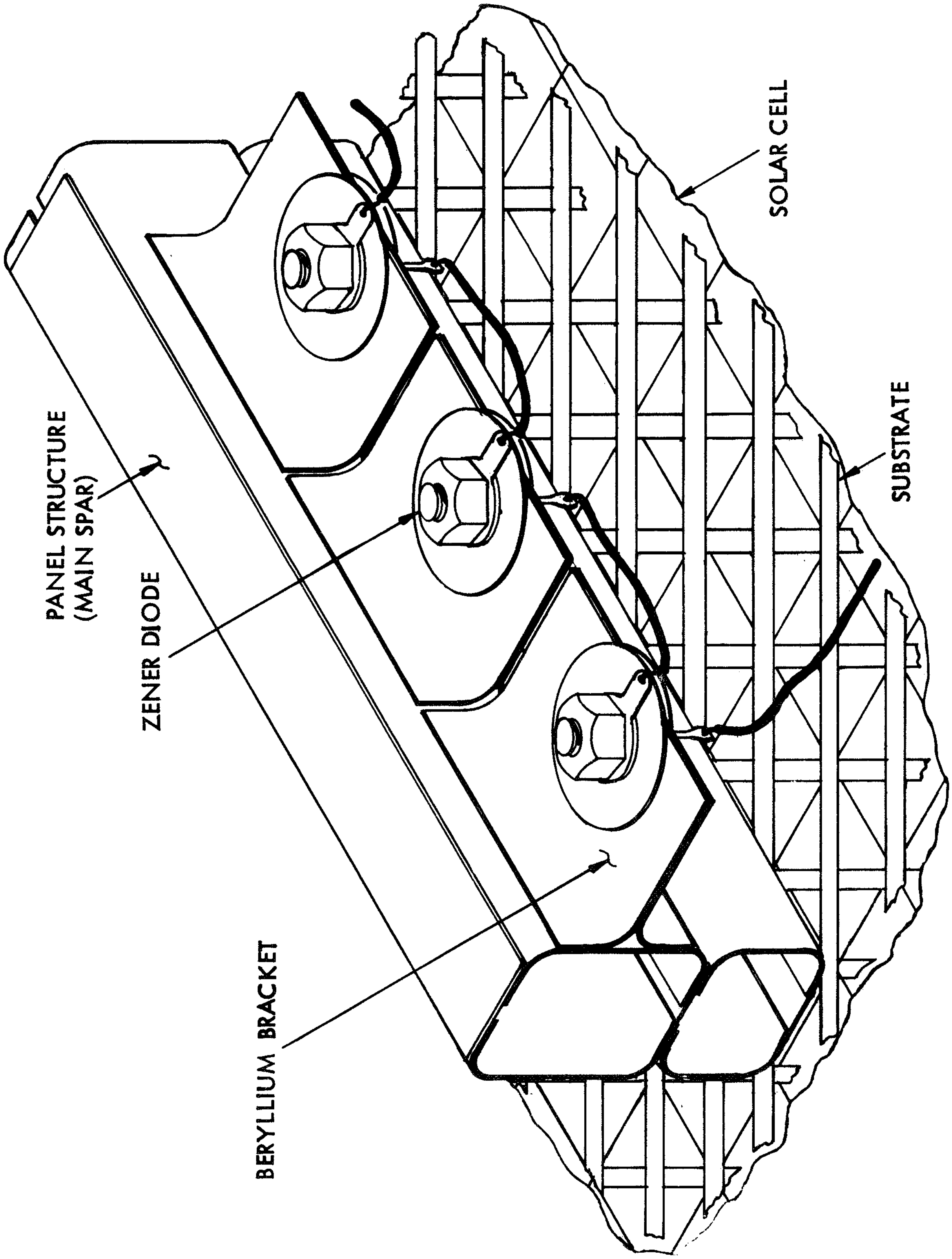


FIGURE 5-2: ZENER DIODE INSTALLATION

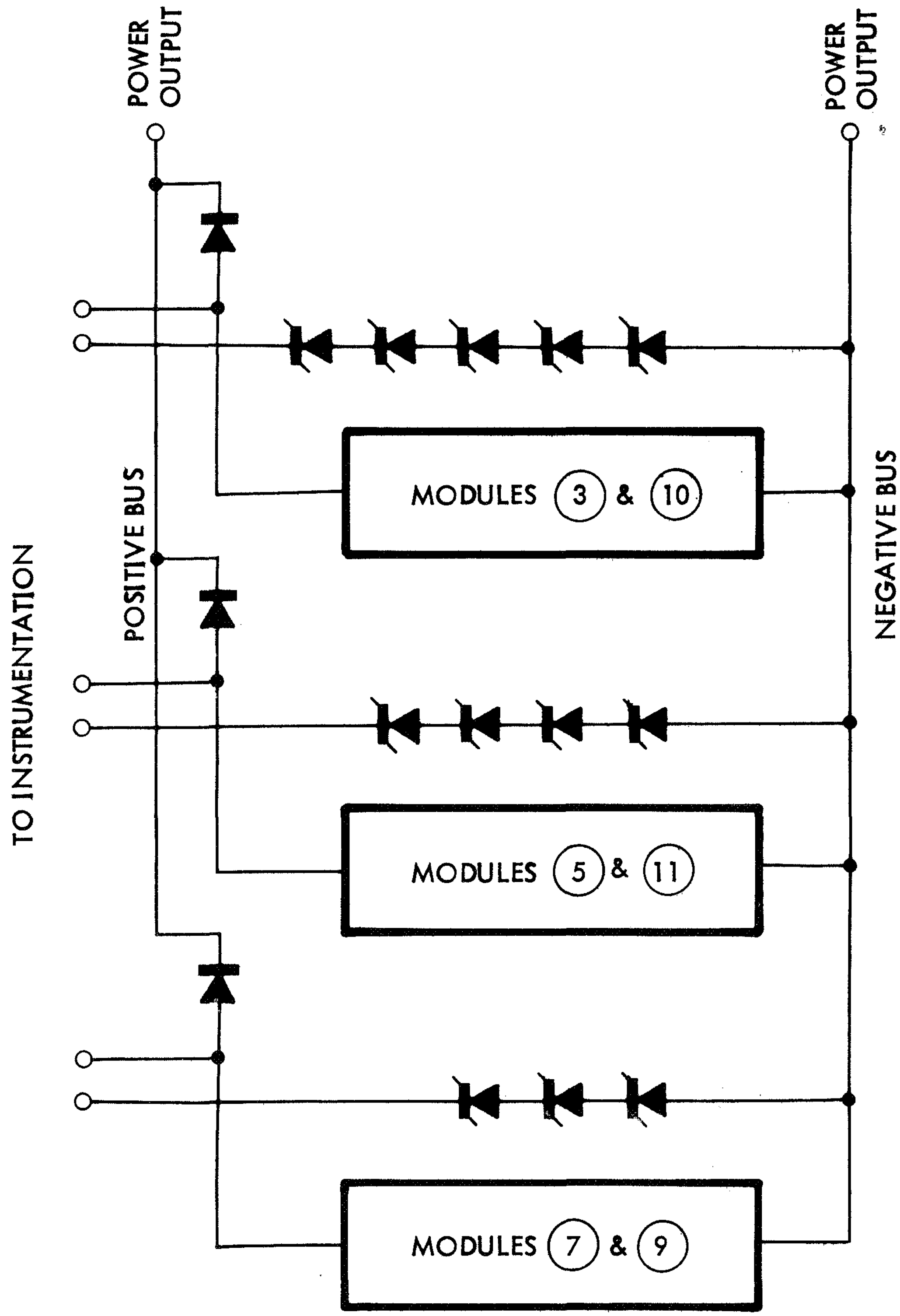


FIGURE 5-3: TEST PANEL SCHEMATIC

5.3.1 BOOST DAMPERS

Boost dampers of the Mariner '67 design can be used on the Test Panel.

Dampers of the Mariner '67 type have been selected for test. These dampers feature silicone oil as the damping fluid. The oil is worked in shear in the radial clearance between two concentric tubes. The damping fluid is trapped between "O" rings at each end of the cylinder.

Tests of selected dampers were performed to verify suitability for use in the dynamic tests of the Test Panel. Parts were measured and selected to provide two damper assemblies with radial clearances near the high and low limits. Both of these assemblies were tested without the centering spring to measure damping under different frequencies and amplitudes using both 30,000 centistoke silicone oil and 60,000 centistoke oil. Typical "O" ring friction was also measured by operating a damper lubricated only with a 50 centistoke light oil. Damping forces and frictions were determined for use in the dynamic analysis. A tentative decision to use 60,000 centistoke damping fluid per Dow Specification No. 210 was made after the review of the test results. Four dampers with radial clearances in mid-range were then tested to confirm earlier results. Disparities in the results of these tests are thought to have been caused by variable breakaway friction of the "O" rings. To provide confidence in the dampers to be used in dynamic testing of the Test Panel, additional damper tests will be conducted in January 1970, with 30,000 centistoke oil and with two of the four "O" rings removed from each damper to reduce friction.

A modified centering spring will be used in each damper for the Test Panel dynamic tests. Active coils will be cut from the existing springs and spacers used to compensate for the shorter length. The changes to spring characteristics are listed as follows.

	Existing Spring	Modified Spring
Length	1.50	1.14
Total Coils	11.85	9.0
Active Coils	7.4	4.5
Solid Height Stress	51,000 psi	51,000 psi
Stress at .15" Compression	17,000 psi	28,200 psi
Solid Height Load (assembly)	10.9 pounds	10.9 pounds
Spring Rate	20 lb/in	33 lb/in

Connection to the panel tip will utilize an adapter which is machined from titanium and will allow relative thermal expansion and prevent booster-to-spacecraft loads from going through the solar panels. The spacecraft damper mounting hardpoint will be duplicated in function by the test fixture. Coatings to prevent damper thermal excursions will not be provided since testing with dampers will be at shop ambient temperature.

5.3.2 MAIN SUPPORT HINGES

Hinge design and construction features titanium fittings with self-aligning bearings.

Faying surfaces between the aluminum ball and bearing races are coated with molybdenum disulfide to prevent cold welding in a vacuum environment and to insure adequate lubrication. In any flight installation, as in the test installation, one bearing will be mounted to resist all shear forces while the opposite bearing will provide end play to accommodate manufacturing tolerances and thermal expansion differences. Deployment will not be demonstrated, therefore no accommodation for deployment mechanisms has been provided. Deployment spring data will be provided to aid in the analysis and definition of deployment and latching equipment.

5.3.3 MASS-SIMULATED EQUIPMENT

Mass-simulated equipment items are mounted on the Test Panel.

The following items are simulated:

1) Cruise Latch---No attempt has been made to simulate the actual latch hardware except for the weight allowance and weight distribution on the panel structure. The simulated cruise latch is machined from mild steel as a cylinder with mounting tabs. It mounts on two titanium brackets bonded to the main spar.

2) Sun Sensor---This item is simulated by a block of mild steel. It is attached to the Test Panel by bolting to titanium clips bonded to the panel structure.

3) Attitude Control Equipment---The 2.8 pound weight of the dual attitude control jets is simulated by a mild steel cylinder with mounting flanges. The cylinder is sized to provide the correct center of gravity distance from the mounting interface, which is a titanium bracket bonded to the panel structure. The attitude control tubing is simulated by a stainless steel tube clamped at several locations along the center longitudinal intercostal. There is no plan to install any electrical control circuit simulation or any tubing swivel joint simulation at the deployment centerline.

5.4 STRUCTURAL TEST ARTICLE DESIGN

The structural design of the test article meets the current panel requirements and provides for minimum structural weight together with manufacturing cost savings.

The structural design criteria for the test article are contained in the Detail Requirements for Lightweight Photovoltaic Array Structure Technology, dated April 25, 1969. Briefly summarized, they include, but are not limited to, the following:

1) In the pin-pin condition the panel shall withstand an 8 g load normal to its surface without yielding.

2) Supported at 3 points, the panel shall withstand a 50-pound load at the unsupported corner without yielding.

3) The panel in all configurations shall be capable of withstanding a 1 g field.

4) In the pin-pin condition, there shall be no natural frequency under 20 Hz.

A more detailed discussion of static and dynamic structural criteria is given in Section 5.5.

The test article structural system consists of a pretensioned fiberglass tape substrate sandwiched between sun side and dark side bonded beryllium frames (see Figure 5-4). The frame assembly is rectangular in plan view. The dark side frame includes outboard spars and edge members which form the perimeter of the frame, two longitudinal main spars, a center longitudinal intercostal, and lateral intercostals. The sun side frame consists only of perimeter members (2 outboard spars and 2 edge members). All primary structural bonding including the beryllium spars and intercostals and the final frame-to-substrate-to-frame bond is accomplished with AF-126 (BMS 5-51) adhesive which is a modified epoxy film supported with dacron fibers. This adhesive is the same as that used for LASA except that a different liquid primer (BMS 5-89) is used. Titanium is used at all concentrated load points and joints for structural components, where beryllium is less suitable, because of titanium's similar thermal expansion, high strength to weight ratio, and toughness. Machined fittings are held to a minimum to reduce costs. Machined titanium fittings are used only for the spacecraft attach hinges on the lower edge member, (see Figure 5-5) tip damper attach fittings on the upper edge member, and a cruise vibration damper latch attach fitting on one of the main spars. All structural connections to beryllium are by adhesive bonding to reduce stress concentrations. Titanium shear clips, gussets, and splices are used at panel joints for reasons listed above and to reduce costs.

The solar cell adhesive (RTV-40) will be used as a thermal control coating on the dark side of the cells and beryllium structures.

All spars and intercostals are fabricated from powder-derived beryllium sheet which was purchased to the same specification as that used for LASA (BMS 7-183). Left

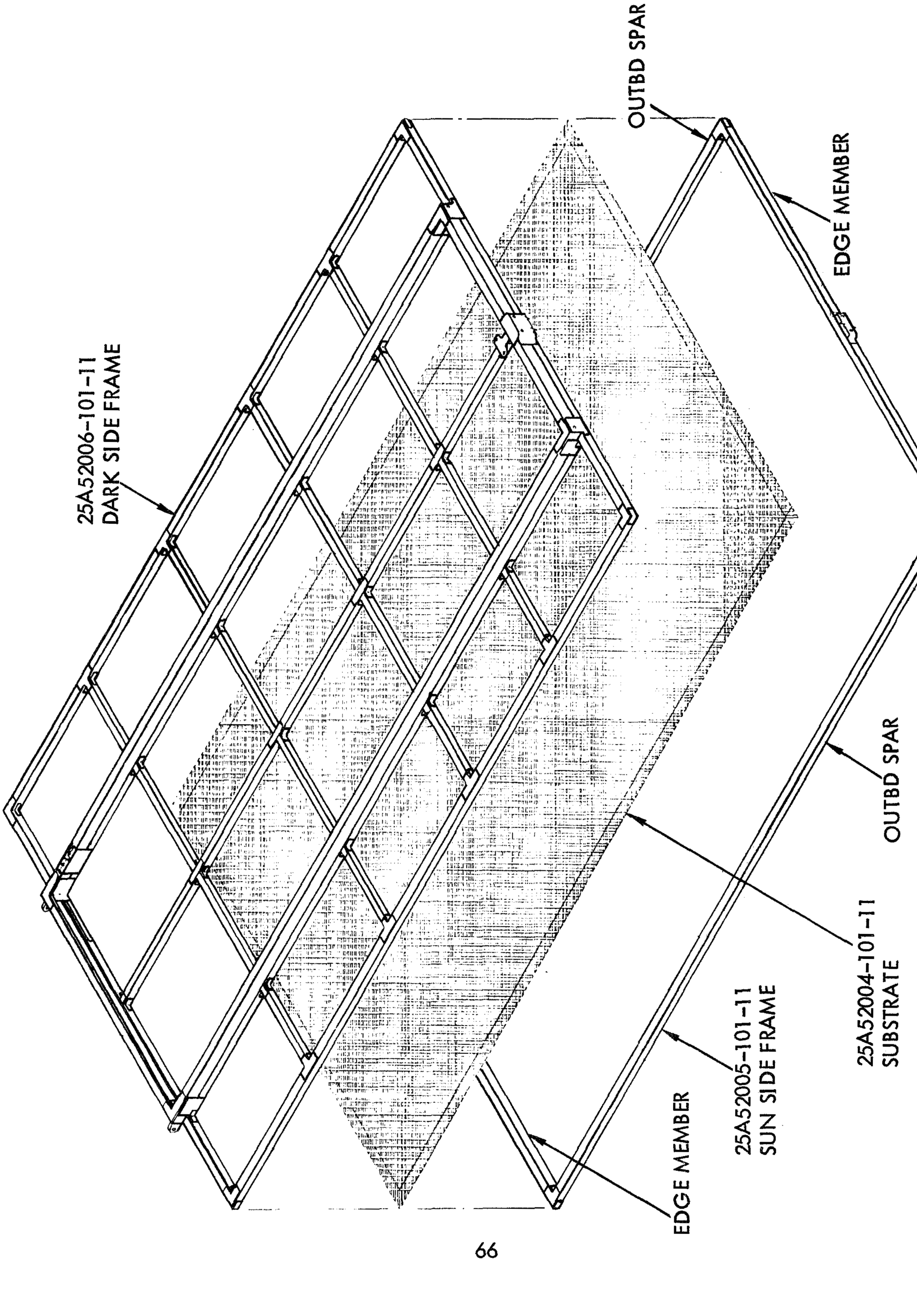


FIGURE 5-4: TEST PANEL STRUCTURE ASSEMBLY

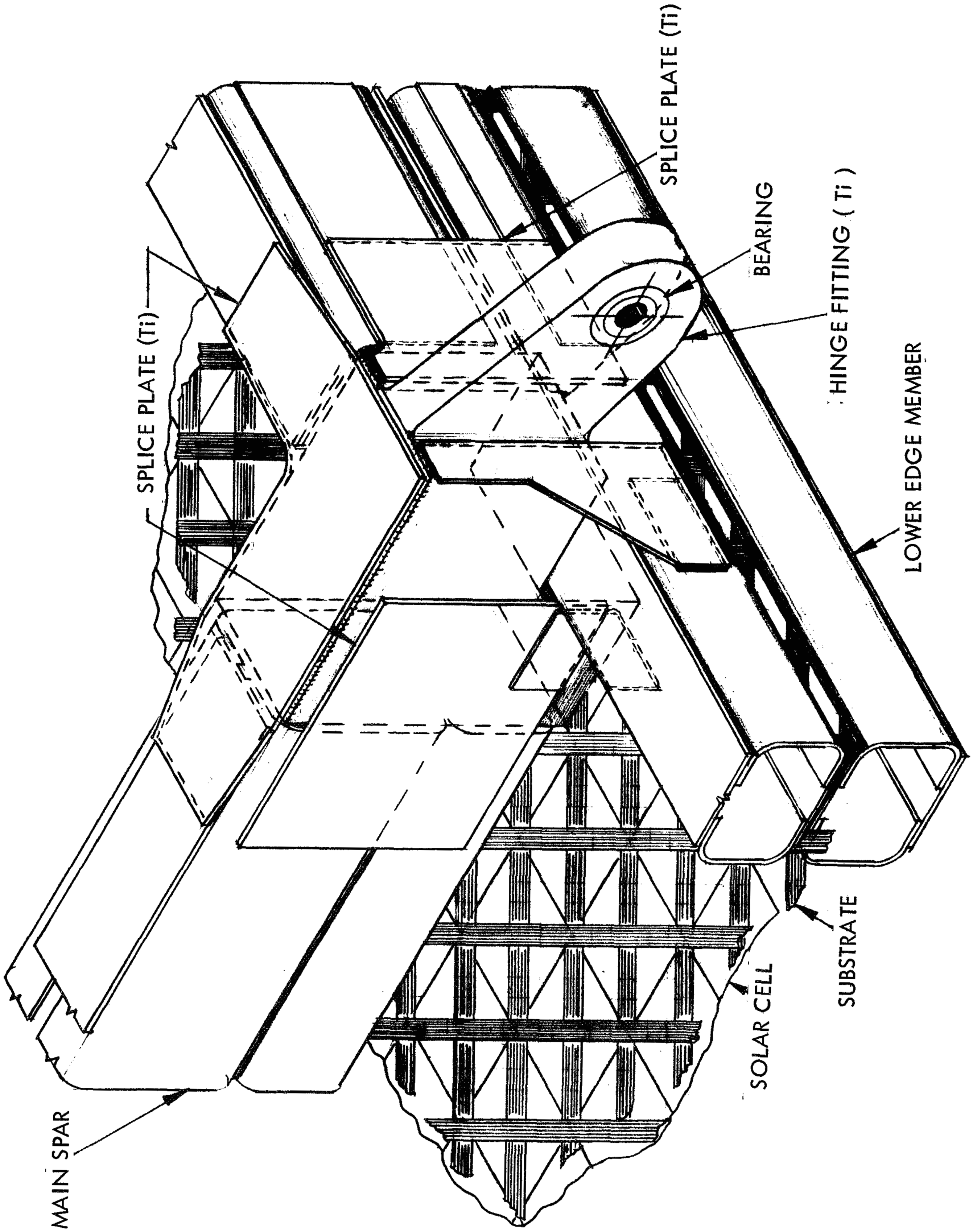


FIGURE 5-5: HINGE FITTING INSTALLATION

over beryllium from LASA was also used to the maximum extent possible to reduce costs. Fabrication techniques are those used for LASA. The basic cross section of any member consists of two hot-creep formed channel sections attached by a top and bottom flat cap strip. Figure 5-6 shows cross sections of all beryllium members with their section properties.

The fiberglass tape substrate, to which the solar cells are bonded, is positioned at 45 degrees to the edge members to provide additional in-plane shear stiffness. The tapes are pretensioned to a final average value of 6.82 pounds per tape which is equivalent to an average load on the frame edge members of 12 pounds per lineal inch. Stress calculations for the beryllium members include this static load in the combined stress totals. Due to the differential thermal expansion between the tapes and the beryllium frame, the tapes are initially tensioned to 9.77 pounds per tape in the tension frame during the final bond cycle in order to achieve the final value of 6.82 pounds after cool down. In order to verify the calculated final tape pretension load, a small component long duration tension creep test of a tape specimen was performed (see Figures 5-7, 5-8, and 5-9). In this test, two creep specimens consisting of approximately 7.0 inch lengths of cured tape with their ends bonded to aluminum grips were subjected to a static 9-pound load for 12 days at room temperature and at 212°F (to simulate environmental test requirements). Gage lengths were measured before and after loading. Test results showed that there was little permanent tape elongation or adhesive creep. This test increases the confidence level of the tape pretension values used in the dynamic analysis.

In order to reduce manufacturing costs, a different approach from LASA is used for the titanium hinge and tip latch fittings. Solid titanium blocks are electro-discharge-machined to form the basic thin wall hollow box shape. All welding, in the basic fitting, is thereby eliminated. This technique is expected to result in a superior part at lower cost.

A detailed weight breakdown for structural, mechanical, and electrical components with an historical comparison from the proposal configuration to the test panel

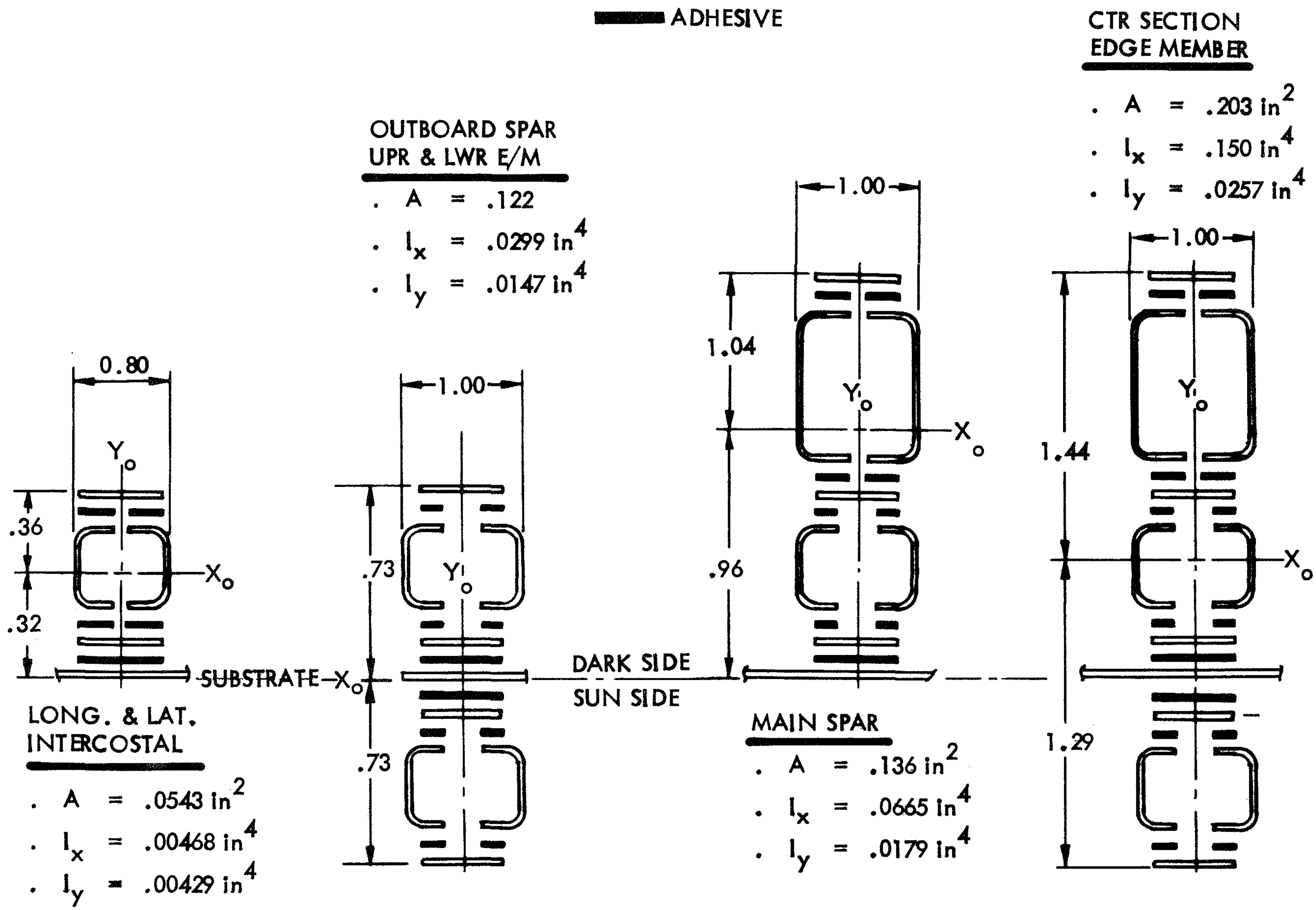


FIGURE 5-6: STRUCTURAL MEMBER PROPERTIES

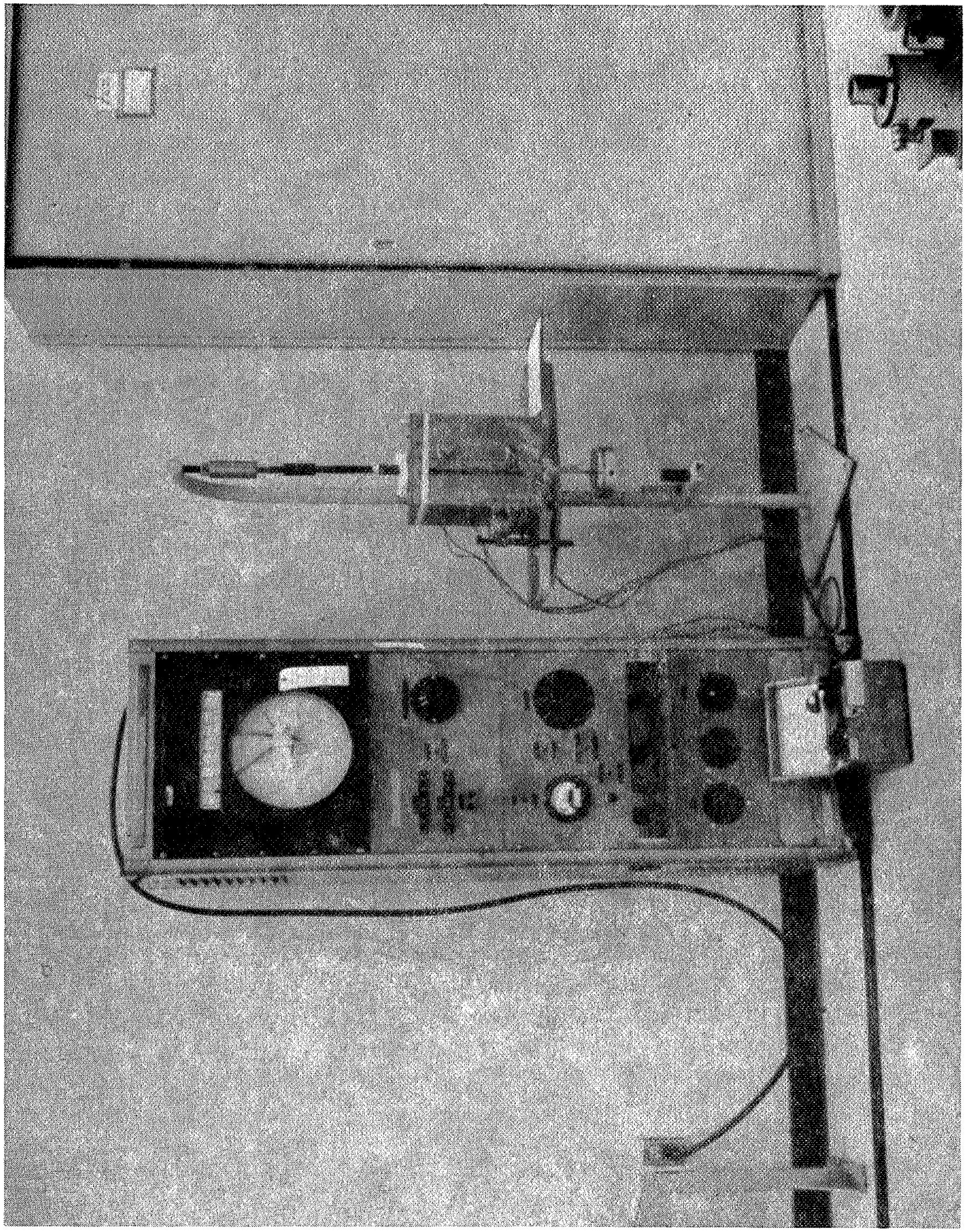


FIGURE 5-7: ADHESIVE CREEP TEST -- 212°F

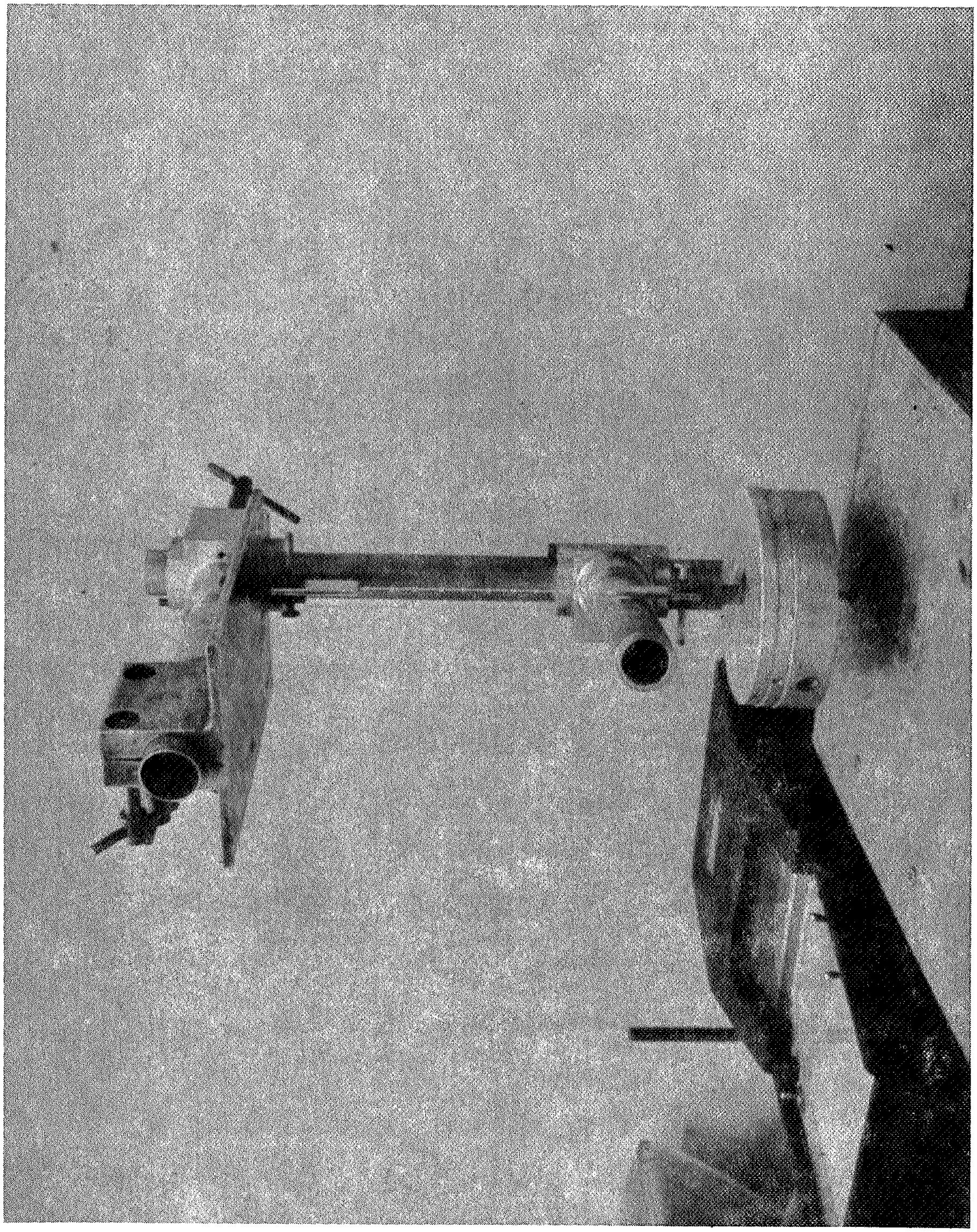


FIGURE 5-8: ADHESIVE CREEP TEST — ROOM TEMPERATURE

SPECIMEN NO.	TEST TEMP. °F	TEST LOAD LBS TENSION	GAGE LENGTH	TIME LOAD APPLIED (HRS)	PERMANENT SET (IN.)	SET OCCURS IN:
1	212	9	A C B	288	.00035 .00150 .00015	Adhesive Tape Adhesive
2	72	9	A C B	288	.00115 (-).00285 .00040	Adhesive Tape Adhesive

NOTE: Permanent set was measured over gage lengths A, B, and C, as shown below. The limit of accuracy of the measuring device (comparator) was ± .0003 inch.

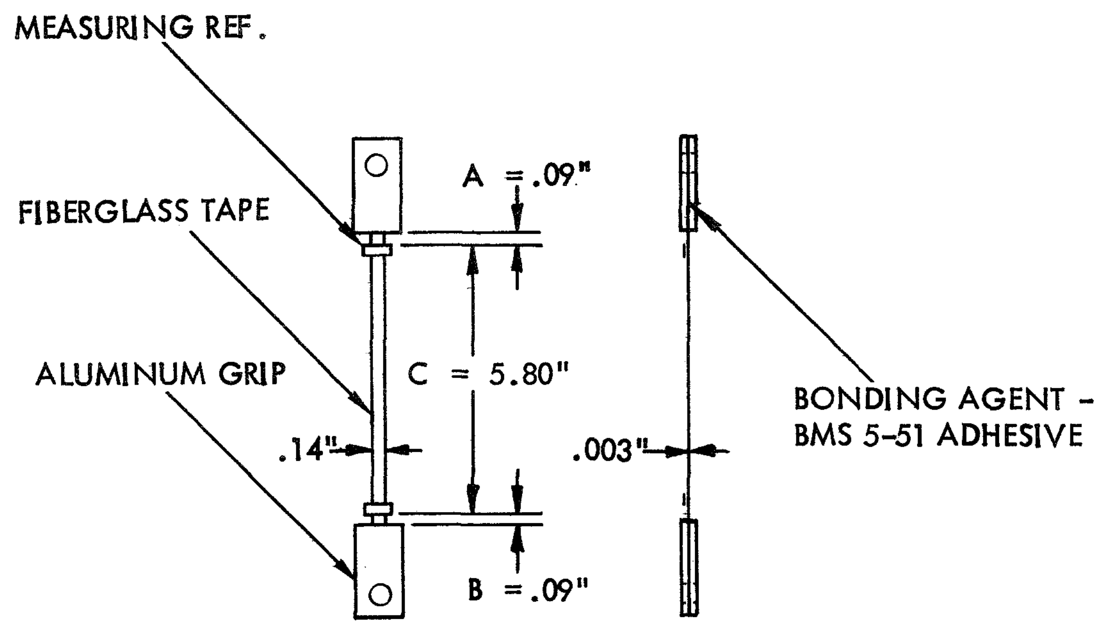


FIGURE 5-9: TAPE AND ADHESIVE CREEP TEST RESULTS

configuration is shown in Table 5-1. Figure 5-10 shows total weight and the specific power output for the solar panel power system at various levels of power system component inclusion. Also with an historical comparison from proposal to test panel configuration, it can be seen from Figure 5-10 that power output figured at 10 watts per square foot divided by combined structural and cell stack weight yields a specific output of over 20 watts per pound which exceeds contractual requirements.

TABLE 5-1

TEST PANEL WEIGHT SUMMARY

	TEST PANEL Panel, lbs.	TEST PANEL Array, lbs.	PROPOSAL CONFIGURATION Array, lbs.
CELL STACK AND BUSES (TOTAL)	(5.15)	(20.60)	(22.56)
Solar Cells	2.49	9.96	
Coverglasses	.91	3.64	
Cell Adhesive	.30	1.20	
Coverglass Adhesive	.12	.48	
Solder and Connectors	.38	1.52	
Bus and Terminals	.45	1.80	
Thermal Coating	.50	2.00	
PANEL STRUCTURE (TOTAL)	(8.06)	(32.24)	(35.72)
Main Spars	1.68	6.72	8.12
Outboard Spars	1.56	6.24	8.20
End Members	1.21	4.84	4.44
Lateral Intercostals	.76	3.04	4.48
Longitudinal Intercostals	.34	1.36	
Substrate	.46	1.84	1.84
Clips, Splices, Gussets	.68	2.72	.88
Thermal Coating	.35	1.40	2.84
Fittings	.68	2.72	3.56

TABLE 5-1 (Continued)

	TEST PANEL		PROPOSAL CONFIGURATION
	Panel, lbs.	Array, lbs.	Array, lbs.
Miscellaneous	.34	1.36	1.36
Subtotal (Structural and Electrical)	(13.21)	(52.84)	(58.28)
ATTITUDE CONTROL (Simulated)	(3.68)	(14.72)	(7.82)
R/C Jets	2.80	11.20	5.60
Line/Harness	.88	3.52	2.22
DIODE INSTALLATIONS (TOTAL)	(2.93)	(11.72)	(8.72)
Zener Diodes	2.45	9.80	
Mount Strips	.48	1.92	
CRUISE DAMPER LATCH (simulated)	(.07)	(.28)	(2.20)
SPACECRAFT MECHANISMS (TOTAL)		(3.40)	
LAUNCH VEHICLE MECHANISMS (TOTAL)		(6.72)	(1.84)
OTHER EQUIPMENT (TOTAL)			(20.00)
SUN SENSOR (TOTAL) (Simulated)	(2.00)	(8.00)	
TOTAL WEIGHT	21.89	97.68	98.86

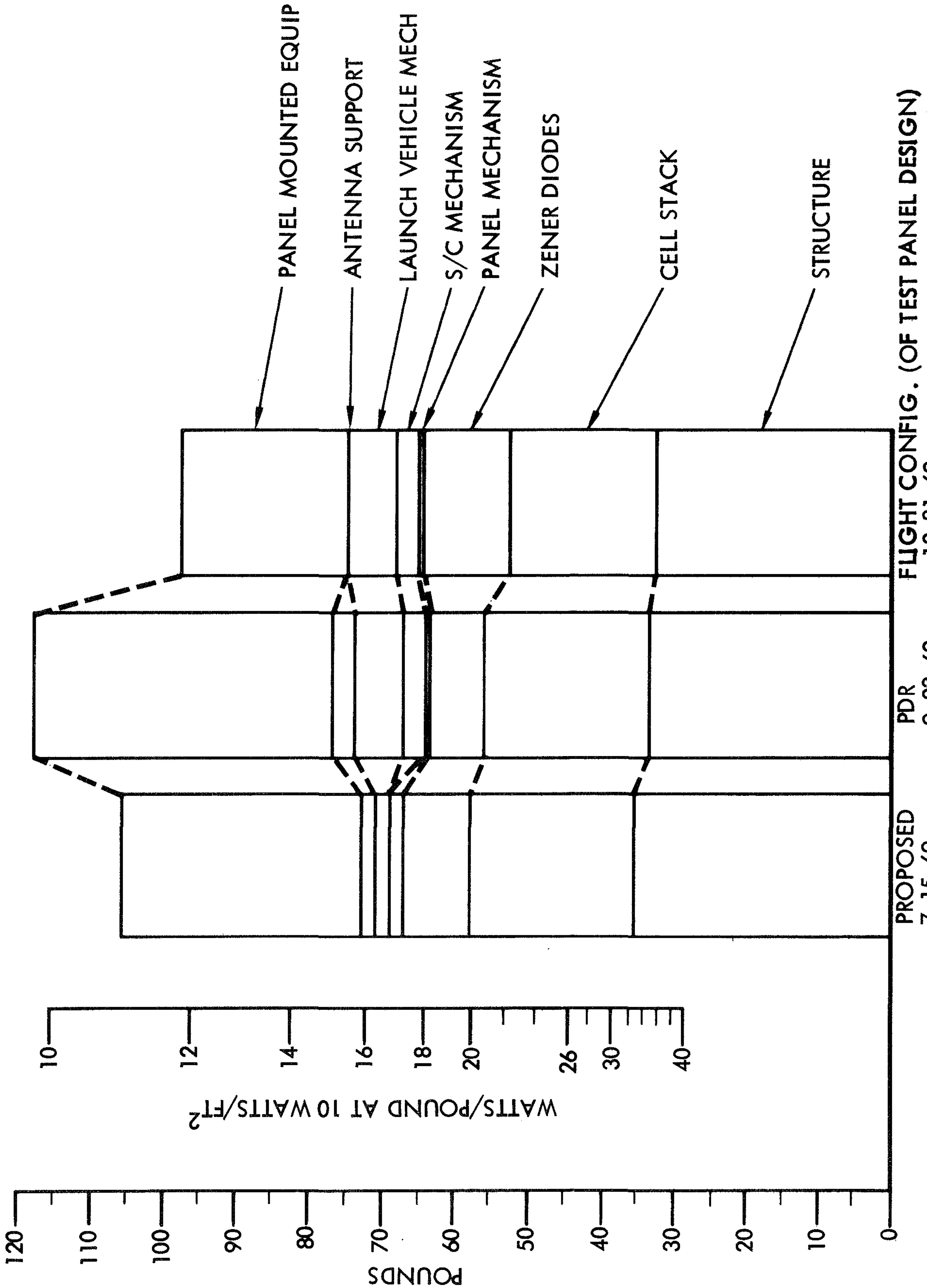


FIGURE 5-10: WEIGHT COMPARISON

5.5 ANALYSIS

Changes in the panel required some analyses in addition to that performed on the trade study configuration.

The changes specified in Section 5.1 did not require structural changes in any primary members, but they did result in weight and weight distribution changes which necessitated re-analysis for the dynamic and static requirements. No additional electrical power or thermal analysis was required, except for the power-to-weight determination described herein.

Contract changes after the PDR removed the requirement for "an analytical dynamic analysis of a single solar panel, including appropriate dampers if deemed necessary, to predict acceleration response to a sinusoidal 1 g base input at modes below 100 Hz." Member loads for both normal and in-plane excitation were required. This was changed to a requirement of specific definition, location and function of dampers, deployment rate limiters and similar equipment needed for a proper design.

Re-analysis of the boost configuration was required to ensure that the damper parts available for the sinusoidal sweep test are adequate to control the dynamic response at the specified excitation and to determine the ratio of hinge excitation to the specified (translation) excitation. Re-analysis for the modal test was required for comparison with the test results.

5.5.1 ELECTRICAL POWER ANALYSIS

Inclusion of zener diodes on the panel significantly affects the power-to-weight ratio.

A goal of the MMSA design was to provide 20 watts electrical output per pound of panel weight at one A.U, $55^{\circ}C$, and with a solar intensity of 140 mw/cm^2. Based on 10 watts per square foot and a total cell area of 116 square feet, the predicted output for the MMSA design is 1160 watts. The predicted weight of a four-panel array is 52.84 pounds without zener diodes and 64.56 pounds with five zener diodes per module mounted on the panels. The resulting power-to-weight ratios are:

21.95 watts/pound without zener diodes
17.96 watts/pound with zener diodes

The MMSA Test Panel design uses five zener diodes per module. Six zener diodes per module were originally specified but early studies showed that fewer diodes could be used. The maximum power possible from any module was calculated to be 42.5 watts based on a solar intensity of 140 mw/cm^2, cell temperature below -80°C, and the voltage-current ratio at the maximum power point. (Because of lower-than-flight-quality cells and coverglasses, the predicted output for the Test Panel is 25.53 watts per module at 27°C.) With full module power being dissipated by the zener diodes, five diodes would result in 8.5 watts per diode. Four diodes would result in 10.6 watts per diode and 3 diodes would result in 14.2 watts per diode. Some variation in internal impedance of the diodes can be expected; consequently, the use of four or five diodes per module is indicated.

Minor changes were made to the originally proposed electrical design to obtain an optimum Test Panel and baseline flight article design. As originally proposed, there were eleven modules which contained 90 cells in series. The design now contains 12 modules with 80 cells in series to give 33.6 volts near Earth and 46.4 volts at Mars. The total number of cells, 6,480 per panel, did not change.

Power losses due to the thermal effects of antennas and sun sensors were analyzed. Higher than average temperatures were predicted for solar cell assemblies which were directly "behind" the relay and maneuver antennas and sun sensors. Five modules were partially affected so that they would be working slightly off the maximum power point. Losses from the antennas were calculated to be 1.5 watts and from the sun sensors 0.5 watt. Because the antenna is omitted and only one sun sensor included, power losses on the Test Panel will be minimum.

5.5.2 MECHANICAL ANALYSIS

Additional analysis is required to define a deployment closing rate limitation.

Analysis in progress has shown that a deployment closing rate limitation of 0.2 to 0.3 radians per second will be necessary to avoid damage to a flight-configuration panel of the Test Panel design. Additional analysis will be performed in 1970. A specific closing rate limit will be defined. Using the deployment spring data described in Section 4, a minimum friction loss value will be established and deployment time-histories will be developed with varying rotary damper characteristics. From this analysis, the required rotary damper characteristics will be defined. Characteristics for the cruise latch and damper will also be defined. It is anticipated that the LASA auxiliary panel damper will be a close approximation of the required deployment velocity limiter, or rotary damper, and that the '69 Mariner cruise damper will be acceptable as the MMSA cruise damper.

5.5.3 STATIC ANALYSIS RESULTS - TEST PANEL

Margins of safefy for the Test Panel design are adequate.

The basic structural concept for the PDR Baseline Panel is used for the Test Panel configuration. Minor changes in member loads result from the weight changes which increased the zener diode distributed weight, decreased the coverglass weight, deleted the two-pound maneuver antenna, and repositioned the two-pound sun sensor. In addition, detail analysis of some areas resulted in addition of doubler plates and improved margins of safety. Revised weights applicable to each structural node were tabulated (see Table 4-3).

The results of stress analysis of the Test Panel configuration are given in the following figures:

Figure 5-11 Stress analysis results (margins of safety) for the 8 g and 50-pound load test conditions.

Figure 5-12 Deflection characteristics of the panel under 1 g distributed load plus 7 g added at the nodes.

Figure 5-13 Deflections for the 50-pound load condition.

Figures 5-14 and 5-15 Margins of safety versus amplitude curves for the dynamic cases.

This analysis has shown the Test Panel design to be adequate from a stress standpoint.

5.5.4 DYNAMIC ANALYSIS RESULTS - TEST PANEL

Dynamic requirements for the Test Panel design are satisfied.

The resonance frequencies in the pin-pin, in two pin-free conditions, and in the deployed condition are given for the Test Panel configuration in Figure 5-16.

The pin-pin minimum frequency is 27.4 Hz (shear mode), which exceeds the required minimum of 20 Hz.

One pin-free condition provides frequencies and mode shapes for comparison with the results of the modal survey test. As previously discussed, the substrate resonances have been suppressed in this analysis.

The other pin-free modes provide information for defining the driving forces for the sinusoidal specified excitation and for the sinusoidal test excitation.

The lowest frequency, at 7.9 Hz, is obtained with 35 lb/in damper springs. The springs are slightly modified Mariner-Venus '67 damper parts. The dramatic reduction of spring constant, as compared to the PDR configurations, is the result of the much reduced moment of inertia about the hinge line (85,200 lb-in^2) and the reduction in frequency from 10 Hz to 7.9 Hz.

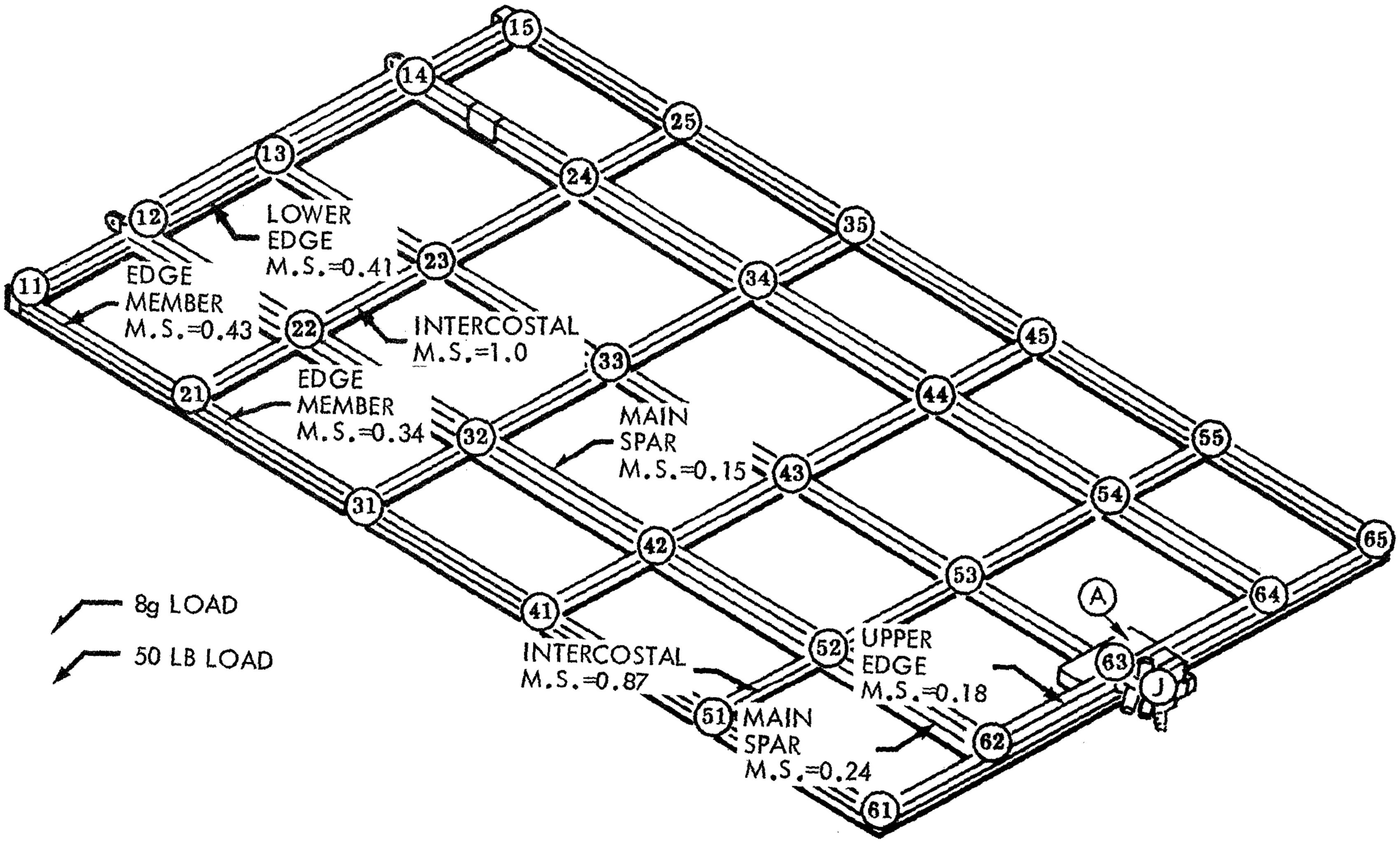


FIGURE 5-11: STATIC STRESS MARGINS - TEST PANEL

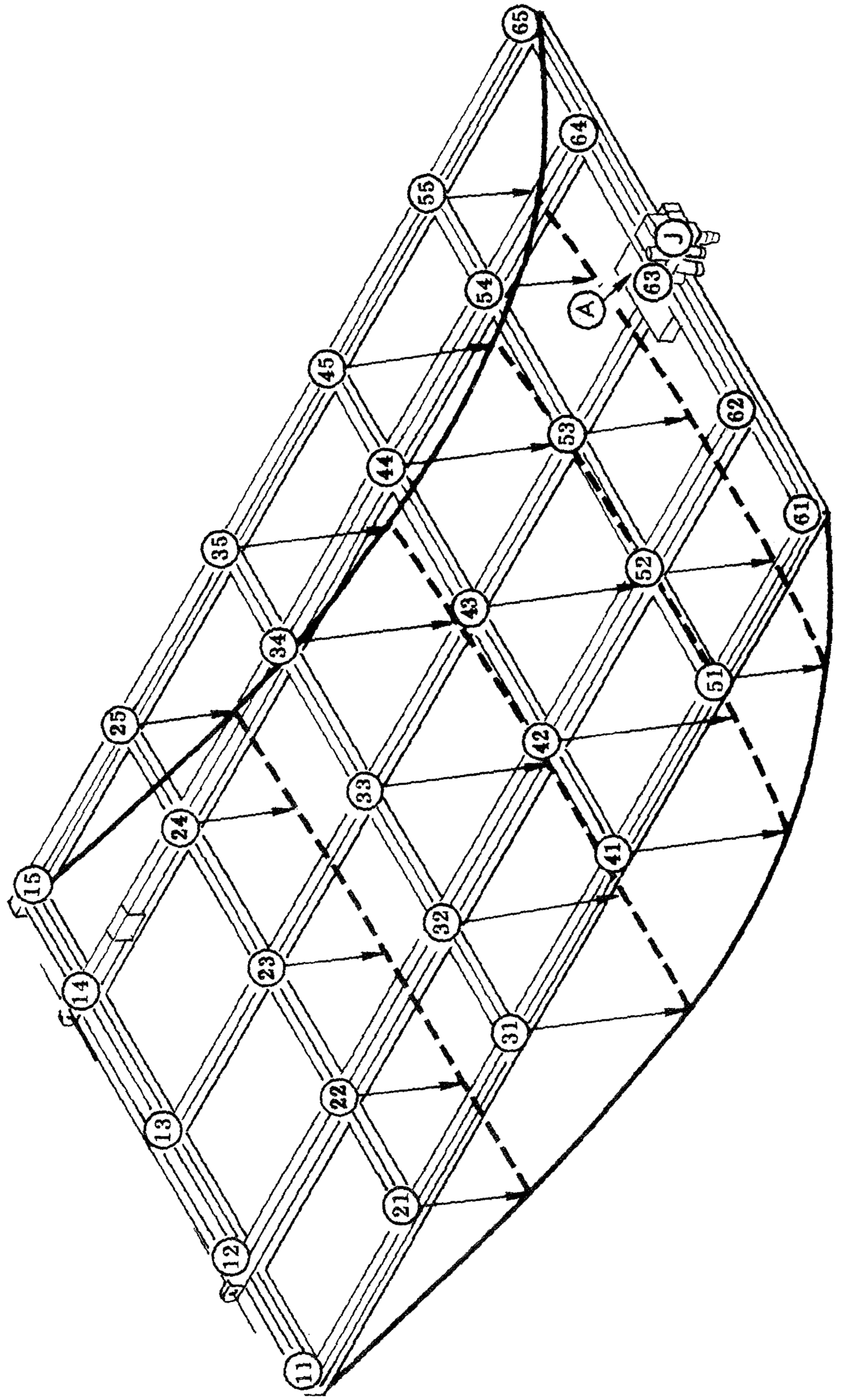


FIGURE 5-12: STATIC DEFLECTION SHAPE - 8 g Load

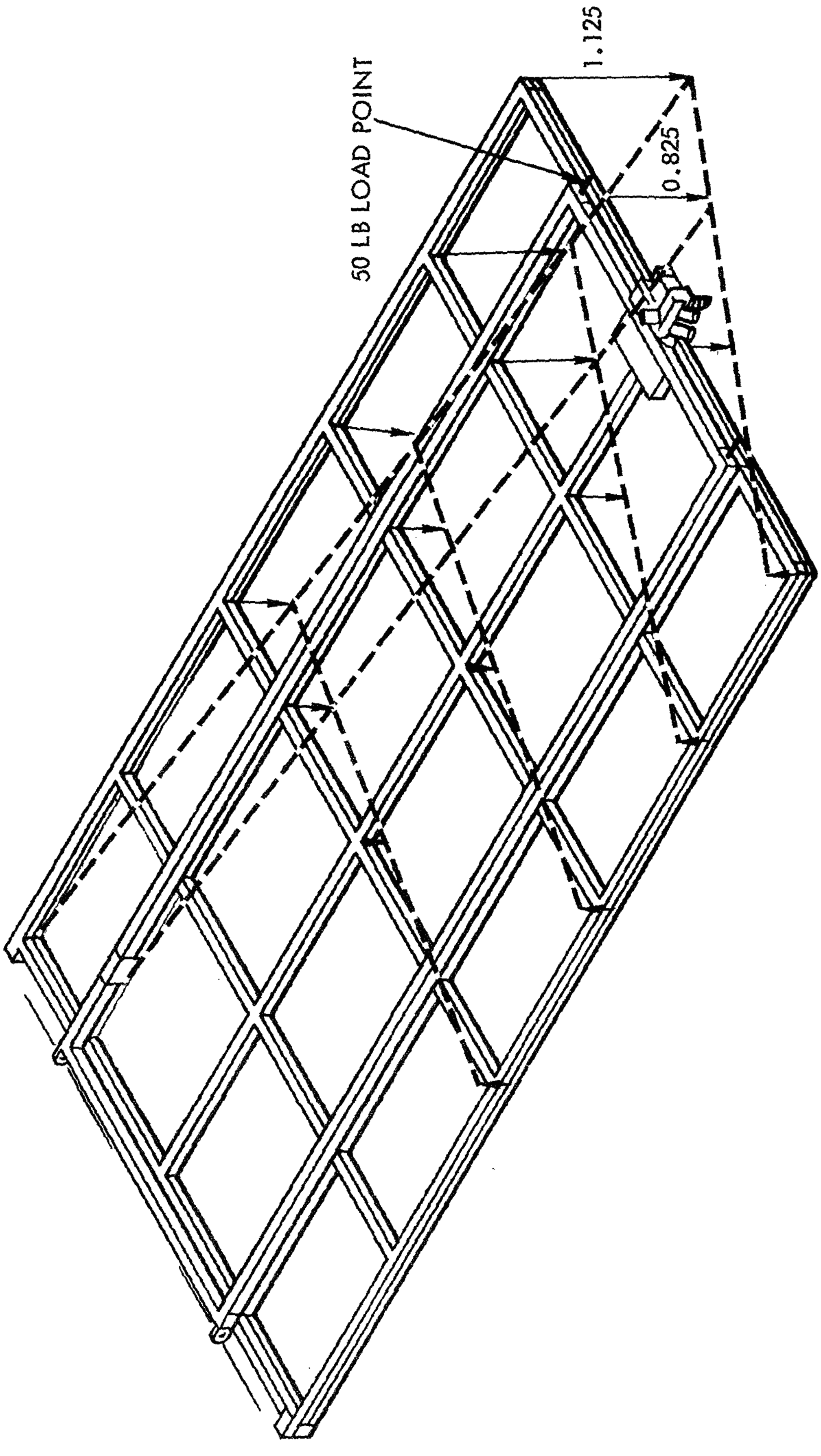
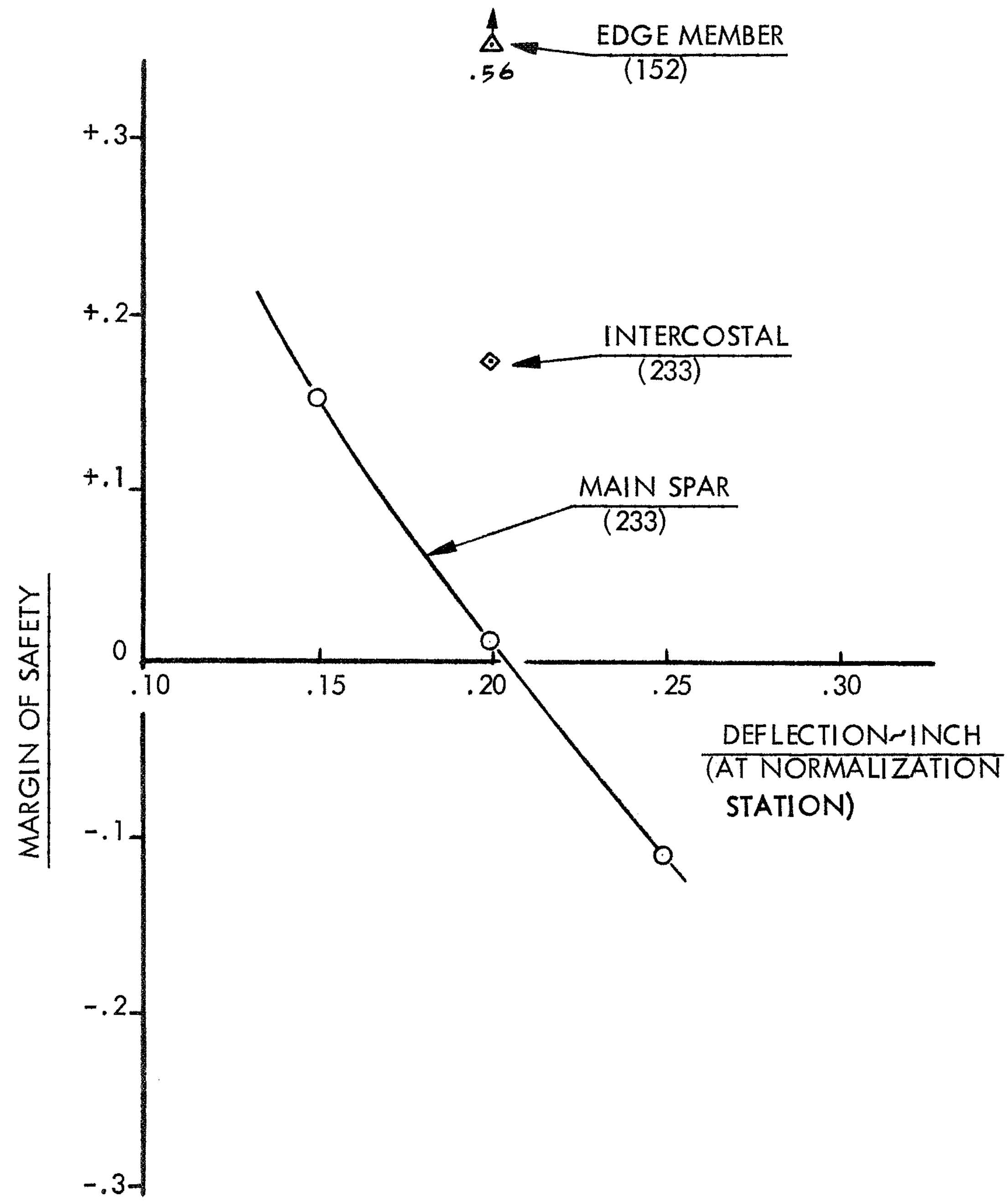
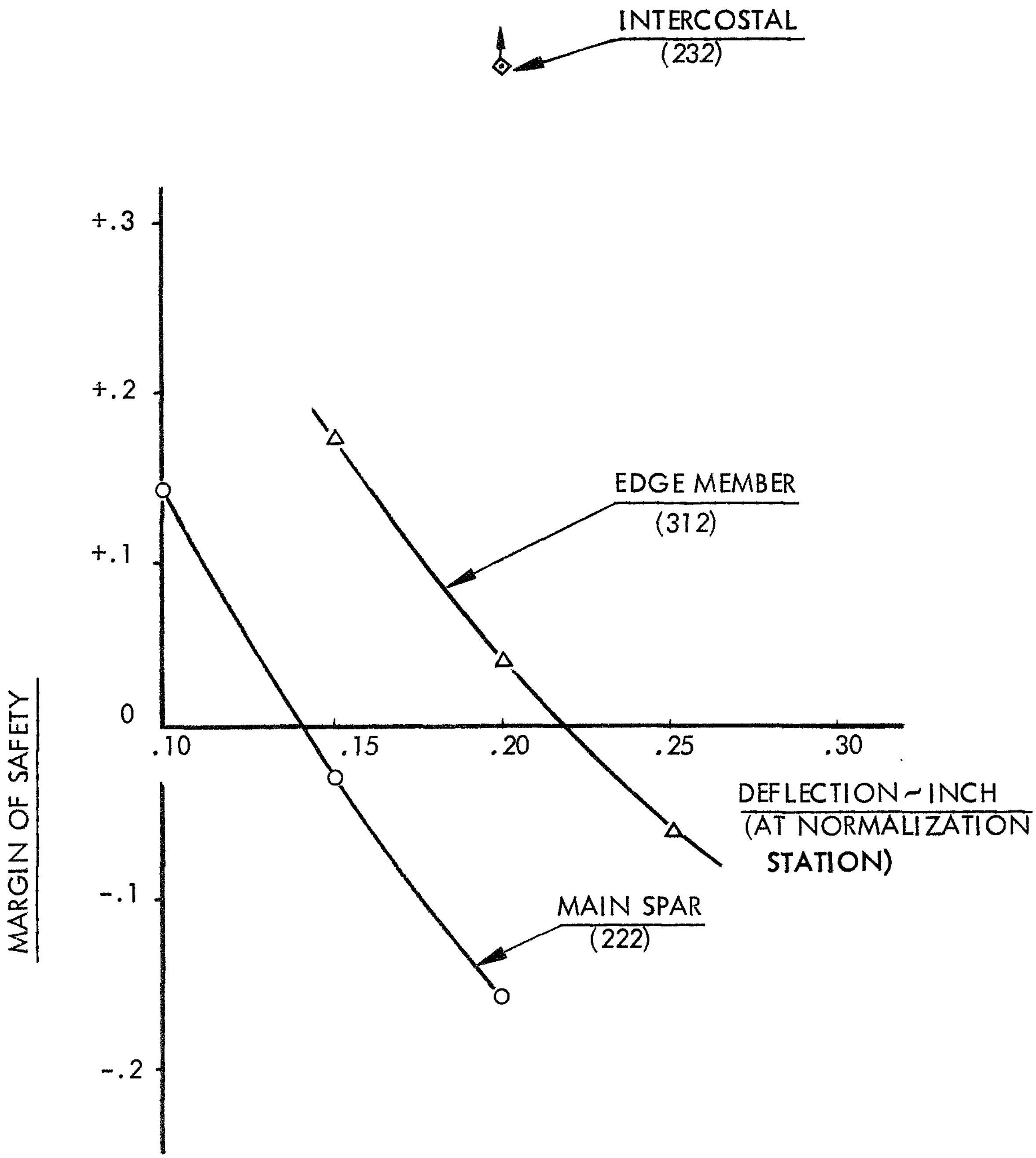


FIGURE 5-13: STATIC DEFLECTION SHAPE - 50 LB Load



NOTE: At the damper attachment point, the allowable in-plane deflection for zero margin of safety is .20 inch (.205 x .944).

FIGURE 5-14: TEST PANEL - DYNAMIC MARGIN OF SAFETY VS DEFLECTION (SHEAR MODE)



NOTE: At the damper attachment point, the allowable out-of-plane deflection for zero margin of safety is .077 inch (.14 x .55).

FIGURE 5-15: TEST PANEL - DYNAMIC MARGIN OF SAFETY VS DEFLECTION (BENDING MODE)

SUPPORT \ MODE	1	2	3	4	5	6	7
PIN-FREE (35 LB/IN. SPRING)	7.9	19.2	28.5	34.3	68.6	80.9	103.3
PIN-FREE (MODAL SURVEY)	—	17.0	27.5	33.9	67.9	80.9	103
PIN-PIN	—	—	27.4	28.8	45.6 (b)	79.6	99.6
DEPLOYED (530 LB/IN. SPRING)	1.61 (a)	17.2 (b)	27.5 (c)	34.5 (d)	— (e)	— (f)	— (g)

(a) Rigid Rotation
(b) First Torsion
(c) Shear
(d) First Bending
(e) Second Torsion
(f) Chord Bending
(g) Second Bending

FIGURE 5-16: DYNAMIC RESULTS, RESONANT FREQUENCIES

The deployed frequency, when reduced by the effect of a damper providing .7 critical damping, is 1.15 Hz. The solution was for a 530 lb/in spring at a 7.0 inch arm from the hinge line. The actual arm has been set at 6.5 inches on the panel drawing, reducing the undamped frequency to 1.5 Hz and the damped frequency to 1.07 Hz.

The dynamic analysis of the panel considers excitation in translation at four points, the two hinges and the two damper locations at the opposite end. However, the dynamic test of the Test Panel is accomplished by exciting only at the hinges, with the dampers grounded. Figure 5-17 gives the uniform translation driving forces for both of these conditions, based on a 1 g excitation. The ratio of driving forces for each mode is also given. This ratio is used to determine equivalent excitations for the specified .707 g test excitation. The ratio for each mode is multiplied by .707 g to obtain these equivalent values. The values obtained, for the important modes range from 4.57 g for the first mode to .67 g for the bending mode. The important excitations, in terms of magnitude, are those specified for the rigid rotation and shear modes, as was previously found for the PDR configurations. The level, however, is about 1/3 less than that of the PDR Baseline configuration and 92 percent of the PDR Alternate configuration. All other modes except first bending, are relatively small. In comparison to the PDR Alternate configuration, the first bending is slightly greater than 92 percent.

MODE	EXCITATION DIRECTION	DRIVING FORCE AT 1 G EXCITATION		RATIO SPEC/TEST
		SPECIFICATION	TEST	
1	Normal	13.53 lbs.	2.09 lbs.	6.47
2	Normal	-.105 lbs.	-.048 lbs.	2.09
3	In-Plane	13.82 lbs.	2.81 lbs.	4.93
4	Normal	4.24 lbs.	4.49 lbs.	.94
5	Normal	-.031 lbs.		
6	Normal	-.017 lbs	-.065 lbs	.25

Modal Definition			
	(1)	Rigid Rotation	7.9 Hz
	(2)	First Torsion	19.2 Hz
	(3)	First Shear	28.6 Hz
	(4)	First Bending	34.3 Hz
	(5)	Second Torsion	68.6 Hz
	(6)	Chord Bending	81.0 Hz

FIGURE 5-17: DRIVING FORCES - TEST PANEL CONFIGURATION

SECTION 6: MANUFACTURING, MATERIEL AND QUALITY CONTROL

The manufacturing effort defined in the Program Plan remains essentially unchanged. A 51 inch by 88 inch solar panel is being fabricated. The panel will consist of a pretensioned fiberglass tape substrate sandwiched between a sun side and dark side bonded beryllium frame assembly; 6,480 glass-covered solar cells bonded to the substrate; and copper bus bars bonded and diodes bolted to the panel structure. Approximately one-half of the solar cells will be joined by soldered silver mesh conductors.

6.1 TOOLS, PROCESSES, AND MATERIAL

Tools, processes, and material from the Large Area Solar Array (LASA) Program are being utilized on MMSA to the maximum extent practical.

By chem-milling heavier material down to .020, forming short material and using short flat stock in widths from .7 inch to .8 inch, 45 percent of the total beryllium requirements have been obtained from LASA surplus beryllium stock. Figure 6-1 shows the source of all stock beryllium material used on the program and indicates processing required to make usable stock material. The figures shown represent the length in inches of stock of the specified sections on which the required operations were performed.

One die was used to make all of the beryllium channels. A LASA hot form die was modified to allow the use of punches from two existing dies and to locate both long and short blanks. Alignment of the punches with the die was also improved by adding tool pins in the compression rings. The LASA heated bonding platen was used to make the MMSA substrate by fabricating a full-size LASA substrate and using only a small section. The material cost of the extra fiberglass tape required was negligible compared to the cost if the tooling had been reworked. LASA bonding tools were reworked to bond MMSA frame members and frame assemblies. LASA solar cell assembly tools were modified slightly to make MMSA submodules. This involved fabrication

		STOCK SECTIONS				
SOURCE	OPERATIONS REQUIRED	1.25" CHANNEL	0.67" CHANNEL	0.020" STRIP	0.015" STRIP	0.010" STRIP
LASA MATERIAL	CLEAN & PRIME			66	237	264
	CHEM-MILL & PRIME	149	18		170	220
	CM, FORM, CM & PRIME		1,322			
	TOTAL LASA MATERIAL	149	1,340	66	407	484
NEW MATERIAL	CLEAN & PRIME			1,092		
	CHEM-MILL & PRIME				546	273
	FORM, CM & PRIME	364	910			
	TOTAL NEW MATERIAL	364	910	1,092	546	273
	GRAND TOTAL	513	2,250	1,158	953	757

FIGURE 6-1: BERYLLIUM STOCK SOURCE CHART

of new paste-solder templates and improved spacers between the solar cells in the seven-cell fixture. Pulse-solder machine electrodes were also stiffened to eliminate deflection during soldering.

The total material requirement of 6,480 solar cells and coverglasses was obtained from LASA surplus stock. 4,309 cell/coverglass assemblies had previously been bonded on the LASA program and are being used on the MMSA Test Panel. The cell/coverglass assemblies are soldered to silver mesh interconnectors to make up a submodule assembly as shown in Figures 6-2 and 6.3.

6.2 BERYLLIUM MATERIAL

Beryllium material costs were minimized by a design change from the LASA design making spar caps and shear webs identical in width.

This design change allowed the use of surplus material which had previously been cut in strips to width. As a result, the total requirement for new beryllium sheet was reduced. All new material for flat stock was purchased in one size: 24 pieces .021 inch by .75 inch by 91 inches. New material for channel stock was purchased in two sizes: 11 pieces .021 inch by 1.60 inches by 91 inches, and 5 pieces .021 inch by 2.20 inches by 91 inches.

The originally committed delivery of November 21, 1969, for new material slid to December 2, and December 24, due to vendor fabrication problems. The material, as received, had very shallow surface depressions. These were readily visible but were not considered cause for rejection because the material was sound and the thickness was within tolerance in the depressed areas. This material has caused no problems in fabrication.

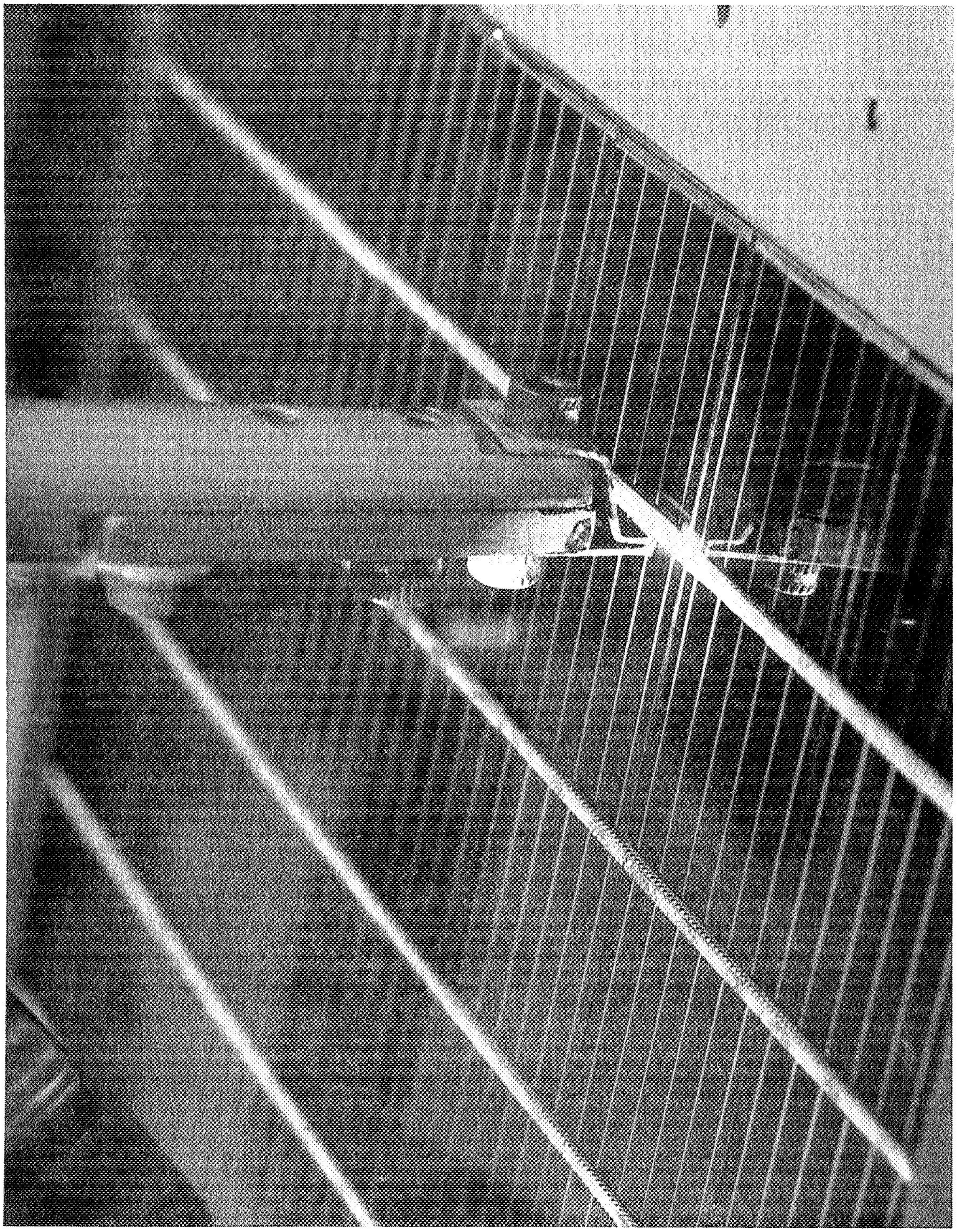


FIGURE 6-2: CELL-INTERCONNECTOR SOLDERING

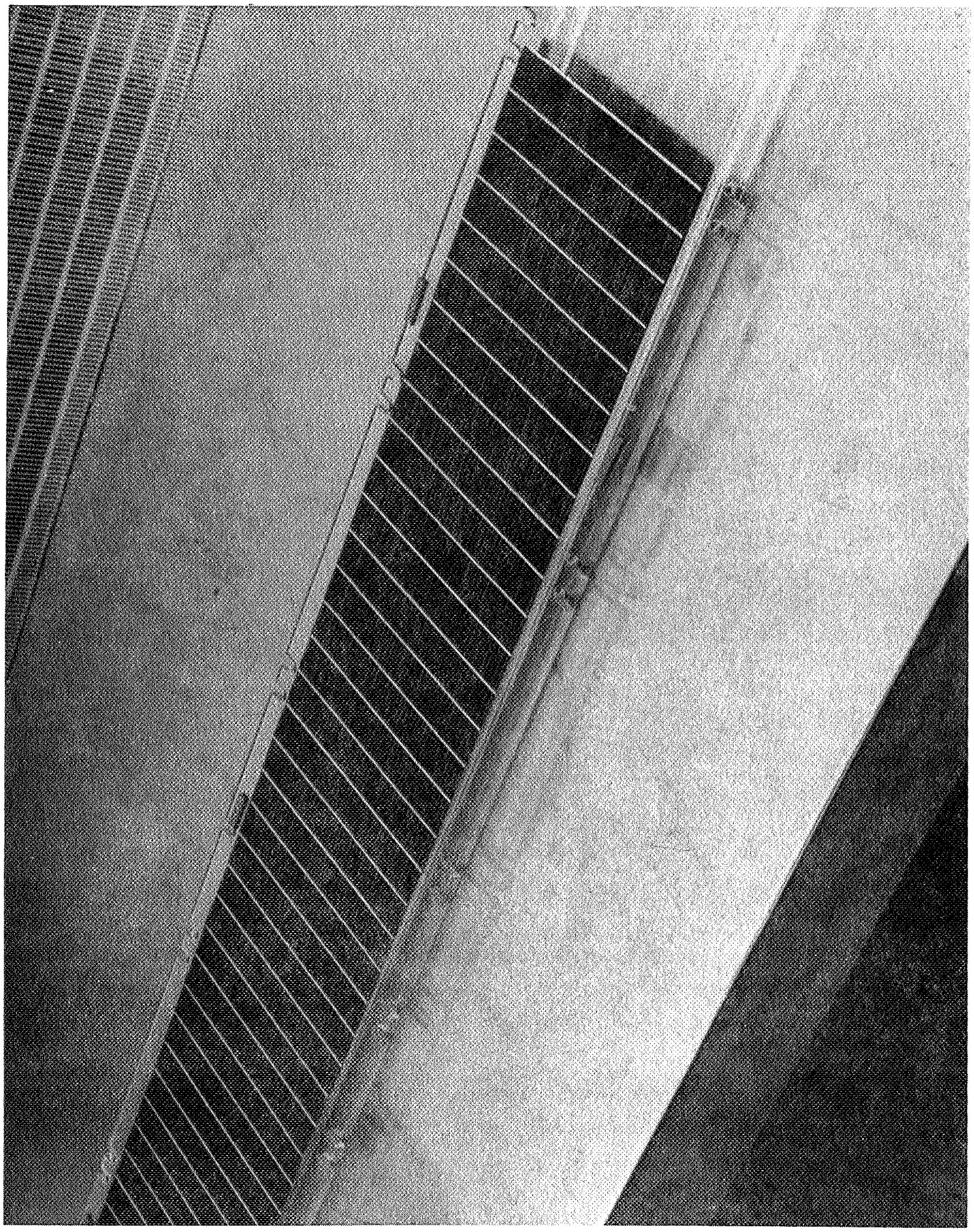


FIGURE 6-3: MMSA SUBMODULE

6.3 FABRICATION TECHNIQUES

Several minor changes to LASA manufacturing methods and techniques have been made to reduce manufacturing costs.

Chem-milling was done at Boeing utilizing facilities originally built for the LASA Phase I chem-milling. This resulted in reduced chem-milling costs and elimination of shipping costs and shipping delays.

A new corrosion-inhibiting adhesive primer (Boeing BMS 5-89), which was developed by the Airplane Division for use with AF-126 adhesive, provides a fully cured epoxy coating on the clean beryllium surface. Parts coated with this primer may be bonded after as much as a year of normal storage and can be recleaned with a simple solvent wipe. Half-inch lap shear test coupons used to certify the cleaning and priming process are pulling well above specifications and often the .050 inch beryllium material used for test fails before the bonded joint.

The substrate assembly was laid up with only hand pretensioning and held in place by double-backed tape. Expansion of the steel platen provided the minimum tension required during bonding. Temporary locating pins were added along the center line of the platen. The tape-laying crew was reduced from three to two by utilizing a simple device for passing the roll of tape between operators.

The pneumatic pressure devices previously used for gusset bonding and the vacuum bag previously used for the final structural bond have been replaced by spring pressure plates as shown in Figure 6-4. For the final structural bond, spring loads are carried into the bonding platen by threaded rods that pass through the substrate openings.

The design of the titanium fittings was changed to allow electrical discharge machining in place of the electron beam welding used on the LASA fittings. Costs were significantly reduced by this change. The solar cell interconnectors have been

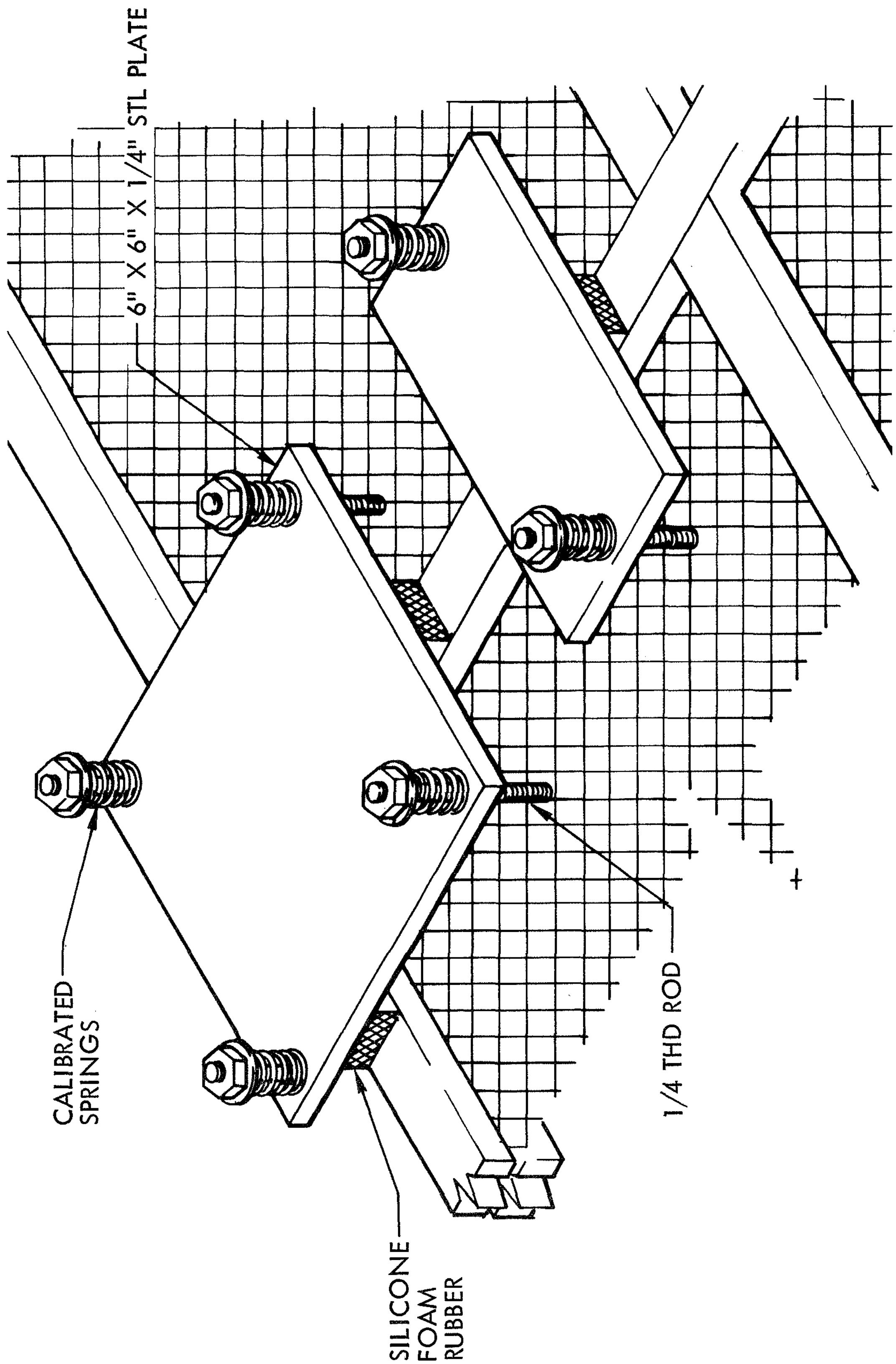


FIGURE 6-4: BOND PRESSURE DEVICE

simplified by eliminating the expensive tabs cut out by hand, and the number of solder spots were reduced to three per side. The flat-pattern development of the interconnectors is now a simple rectangle.

6.4 MANUFACTURING PROBLEMS

Manufacturing problems have not had a significant effect on costs or schedules.

Manufacturing rejection rates have been exceptionally low. Of the beryllium parts made to date, about 22 percent were subjected to material review action and only 5 percent were scrapped.

Three instances of unexplained breakage of beryllium parts have been investigated with no concrete results or explanation as to the reasons for breakage. In one case, three channels out of five formed in the same forming cycle shattered, and the other two parts in the tool were undamaged. Forming conditions, handling, etc., were identical to those used in forming the acceptable channels. A thorough investigation by Materials and Processes could show no peculiarities of these three parts. These failures occurred early in the program and have not recurred in subsequent forming operations.

The second incident occurred during trimming of a surplus LASA channel which had a shallow crack in one edge. During removal of the part from the trim fixture, the part "fell apart" in an area some distance away from the previously cracked area. The machine operator felt that no unusual strains had been induced by either machining or removal from the tool.

The third incident occurred during inspection of a trimmed part. One end of the part slipped over the edge of the table and dropped about 4 inches to an open wood drawer and snapped off at that point. The machinist felt that the force of the blow was extremely slight and should not have resulted in any damage to the part.

At the present time, Manufacturing is at a loss for a satisfactory explanation of these anomalies. However, it should be noted that none of these incidents have occurred after fluorescent penetrant inspection of the parts. The percentage of parts damaged is small and has not had a significant effect on manufacturing costs and schedules.

SECTION 7: TEST PROGRAM

The test program is a close simulation of a Type-Approval test series accepted by JPL for the Mariner Mars '69 solar panels. Test limits have been modified for anticipated 1973 mission requirements. The test program includes launch and space environment simulation as well as power output measurements.

7.1 TEST PLAN DOCUMENT

The Test Plan Document is completed.

The Test Plan contains requirements for fixtures, facilities, instrumentation test levels and data acquisition. All test facilities identified are available within the Seattle area. Test fixture concepts have been defined and Test Panel interfaces with the fixtures have been identified.

Test levels and tolerances have been listed and are compatible with contractual requirements. Expected test results based on analyses and empirical data have been given.

The contract end item, Document D2-121321-1, "Mars Mission Solar Array Test Plan," has been released. An oral presentation of the document contents was given at JPL in Pasadena, California, on December 3, 1969.

7.2 TEST PROCEDURES

The first draft of the test procedures is completed.

Test procedure first drafts have been written for Power Output, Thermal-Vacuum, Thermal-Shock, Modal Survey, Random Vibration, Sinusoidal Vibration, Acoustic, and Static Load tests. These procedures are detailed instructions for accomplishing the test requirements of the test plan document. These procedures will be submitted to JPL for concurrence prior to testing.

7.3 TEST FIXTURES

Preliminary design of the test fixtures is completed.

The power output fixture design consists of three holding stands made from camera tripods. Two of these stands hold the Test Panel and are suitable for panel displays. The third stand supports a six-point thermocouple probe.

The vibration fixture design consists of a baseplate with simulated spacecraft hinge halves. The baseplate provides a pinned connection to the test article for all vibration tests. Overhead support fixtures are designed to support the two dampers at the outboard edge of the panel for modal survey and random vibration tests. Two stands are designed to hold two small vibrators for the modal survey.

The acoustic suspension design consists of four elastic bands which interface with the panel at the spacecraft hinge and tip damper tie points. This is a low frequency suspension designed for under 20 Hz resonance.

The thermal-vacuum and thermal-shock suspension designs consist of four stainless steel wires which interface with the panel at the spacecraft hinge and tip damper tie points. This is a low heat conductivity suspension to minimize heat conduction from the Test Panel.

The static load fixture design consists of two beams with panel interface load support points. One beam is fitted with two spacecraft hinge halves. The other has two supports for the tip damper pins, with one support removeable to provide 3-point support for deflection tests.

SECTION 8: CONCLUSIONS AND RECOMMENDATIONS

It is early to draw any final conclusions on the Mars Mission Solar Array Program; however, some conclusions can be made concerning progress during 1969. It can be concluded that:

1) The design meets the requirements of the solar array for the Mars Mission in 1973 as specified and for the conditions assumed.

2) The program is on schedule with some elements ahead of schedule.

3) The program is within the allocated budget.

4) No significant problems have been encountered.

5) The panel can support extraneous equipment with only minor performance penalties.

It is recommended that the scheduled events for 1970 be completed as planned for the Mars Mission Solar Array Program.

SECTION 9: NEW TECHNOLOGY

As of the end of December 1969, there has been no new technology disclosure under this contract.

SECTION 10: REFERENCES

1. D2-121318-1, Mars Mission Solar Array Program Plan
2. D2-121321-1, Mars Mission Solar Array Test Plan
3. D2-121718-1, Mars Mission Solar Array Analysis Document
4. D2-113355-1, Large Area Solar Array Phase I Quarterly Report
5. D2-113355-2, Large Area Solar Array Phase I 2nd Quarterly Report
6. D2-113355-3, Large Area Solar Array Phase I 3rd Quarterly Report
7. D2-113355-4, Large Area Solar Array Phase I Final Report
8. D2-113355-5, Large Area Solar Array Phase II Quarterly Report
9. D2-113355-6, Large Area Solar Array Phase II 3rd Quarterly Report
10. D2-113355-7, Large Area Solar Array Phase II Final Report
11. D2-113354-1, Large Area Solar Array Process Document

SECTION 11: GLOSSARY

MMSA---Mars Mission Solar Array

PDR---Preliminary Design Review

LASA---Large Area Solar Array

Panel Configurations:

1. Initial Configurations

   Proposal Configuration---A panel configuration, as defined in the Boeing Proposal Document, which supports a relay antenna and other extraneous equipment.

   Alternate Configuration---A panel configuration proposed to determine the effect of removing the relay antenna and related mounting provisions.

2. Trade Study Configurations (at the time of the Preliminary Design Review)

   PDR Baseline Configuration---A refinement of the Proposal configuration including a relay antenna of reduced weight.

   PDR Alternate Configuration A---A panel configuration similar to the PDR Baseline configuration but with the relay antenna and mounting provisions omitted.

   PDR Alternate Configuration B---A set of panel configurations in which extraneous equipment and mounting provisions were included on each of the four panels per array only when the panel would actually support that equipment.

3. Final Configurations

   Test Panel---The MMSA test article on which the relay antenna and the deployment equipment are omitted and approximately half of the solar cells are not connected.

   Flight Configuration---A panel configuration developed for analytical purposes which is identical to the Test Panel except that the deployment equipment is included in the mechanical analysis and 100 percent connection of flight-quality solar cells is assumed in the electrical and thermal analyses.

Module---A group of solar cells connected in series/parallel which produces system voltage.

Cell Stack---An assembly of one solar cell and one coverglass which is bonded with RTV silicone compound.

Power Bus---Flat copper electrical conductors which pick up the output of each module for transmission to the spacecraft.

Interconnector---Expanded silver mesh which connects both parallel groups and a series of solar cells.

Solar Cell Assembly---Same as Cell Stack.

Zener Diode---A solid state component which regulates system voltage to a predetermined value.

Blocking Diode---A solid state component which prevents reverse current from flowing from the power system into a low-voltage module.

Pin-Free---Panel support condition with the hinges supported against translation and the outboard end of the panel unconstrained.

Pin-Pin---Panel support condition where the hinges and the damper support pins constrain the panel against translation and allows rotation.

Structural Node---A point on the panel structure defined in space in terms of coordinates. Two nodes define the structural member connecting them, and the weights of surrounding structure is assumed concentrated at the node.

Substrate Node---A point at the center of a substrate bay at which the effective weight of that bay is assumed concentrated.

Generalized Mass---The "effective" mass associated with a vibration shape.

View Factor---For thermal analysis, the means for defining the effectiveness of deep space temperatures on the substrate and structure.

Factor of Safety---The ratio of ultimate design load to the limit design load.

Fitting Factor---An additional multiplicative factor applied to fittings to account for stress complexities and concentrations.

Margin of Safety---A positive margin of safety is defined as:

$$M.S. = \frac{\text{Allowable Load (or allowable stress)}}{\text{Design Load (or design stress)}} - 1 > 0$$