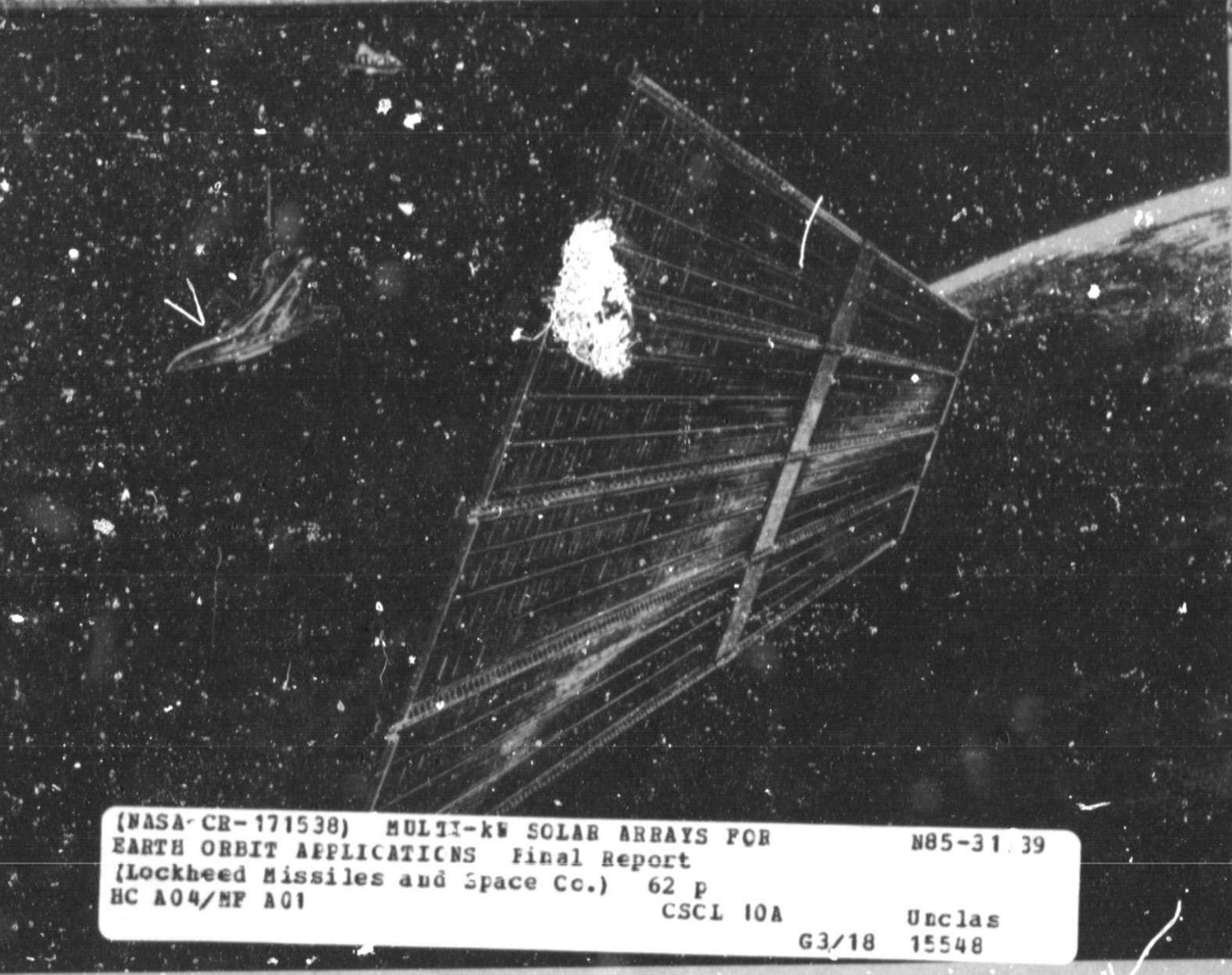

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Final Report

JUNE 1985

Multi-kW Solar Arrays for Earth Orbit Applications

submitted to:
NASA
MARSHALL SPACE FLIGHT CENTER
National Aeronautics and Space Administration

Lockheed
MISSILES & SPACE COMPANY, INC. SUNNYVALE, CALIFORNIA

LMSC-D973456

NASA-MSFC

MULTI-kW SOLAR ARRAYS FOR EARTH ORBIT APPLICATIONS

NAS8-36162

FINAL REPORT

submitted to

George C. Marshall Space Flight Center
National Aeronautics & Space Administration
Huntsville, Alabama

by

Lockheed Missiles & Space Company, Inc.
Sunnyvale, California

LOCKHEED MISSILES & SPACE COMPANY, INC.

FOREWORD

This report documents the work performed by Lockheed Missiles & Space Company, Inc., Sunnyvale, California, for the Marshall Space Flight Center of the National Aeronautics and Space Administration under contract no. NAS8-36162 on the "Study of Multi-kW Solar Arrays for Earth Orbit Applications" project. The term of this contract was approximately nine months beginning October 1984 and concluding June 1985.

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Section 1
BACKGROUND

1.1 OBJECTIVES

Spacecraft system design has historically been characterized by a constantly increasing need for power constrained by the limitations on weight and stowage volume imposed by the available expendable launch vehicles. As the industry transitions to the Shuttle as the primary launch vehicle, constraints on weight and volume are diminishing in importance while cost reductions become mandatory. Serious consideration must be given to simplifying design, minimizing material costs and automating production where possible. Maintaining reliability while substantially reducing the dollars per watt-hour to the lowest possible value is a viable objective.

The multi-kW solar array program is concerned with developing the technology required to enable the design of solar arrays required to power the missions of the 1990's.

Several of these cost reduction suggestions have now been evaluated experimentally. Significant benefits have been realized by using large area solar cells with a superstrate covering approach. Benefits accruing from simplified manufacturing methods were verified. Alternate contacts including copper and gridding of the back were evaluated with very encouraging results achieved with the gridded design. These design improvements and innovations have clearly established the planar, silicon solar array as the power source through the 1990's.

The present effort required the design of a modular solar array panel consisting of superstrate modules interconnected to provide the structural support for the solar cells. The effort was divided into two tasks: (1) superstrate solar array panel design, and (2) superstrate solar array panel-to-panel design. The primary objective was to systematically investigate critical areas of the transparent superstrate solar array and evaluate the flight capabilities of this low cost approach. The effort under this contract phase was the conceptual design of the solar array panel designs, hinging, and power harnesses.

The fabrication and testing of panels and panel segments (small blanket assemblies) under the Advanced Planar Array Development for Space Station NAS8-36419 will demonstrate the designs evaluated herein.

1.2 GUIDELINES

General guidelines were provided which formed the basis for the study and are as follows:

- Use previously developed superstrate design
- Incorporate I-R transparent technology into the panel and blanket designs
- Use as much SAE heritage as possible in the panel and blanket designs
- Use a 10-15 kilowatt deployable blanket assembly as a modular building block
- Must be capable of surviving 5 years in the Low Earth Orbit environment
- Must be capable of surviving the Shuttle environments

1.3 REPORTING

This contract required the submittal of a final written report and one final oral presentation. Program statusing was accomplished by monthly reporting.

1.4 SCHEDULE

The term of this contract was a nine month period from October 1984 through June 1985. All major tasks and subtasks are shown in Figure 1-1.

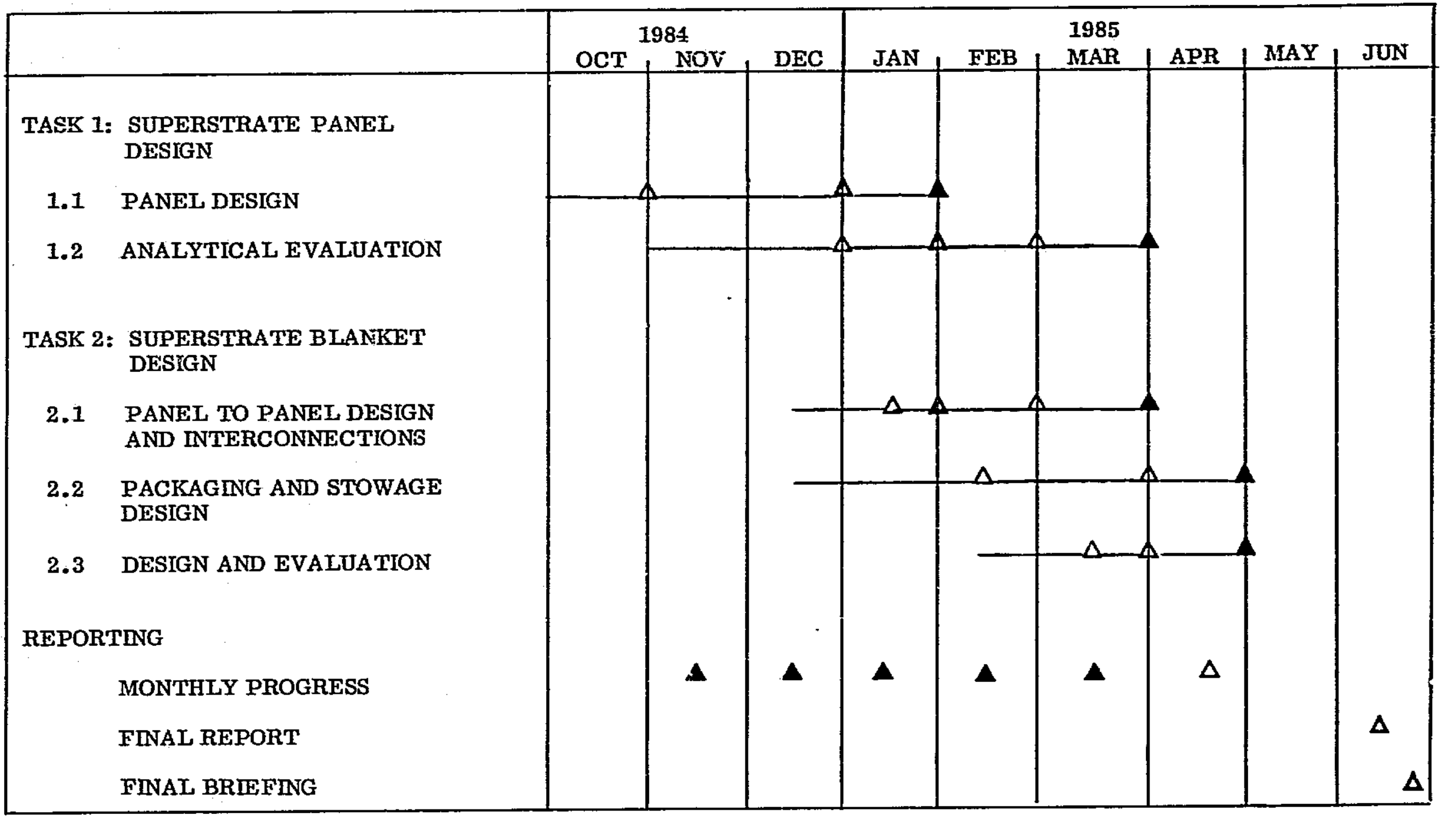

Figure 1-1 Multi-kW Solar Arrays for Earth Orbit Applications Schedule

1.5 PROJECT ORGANIZATION

This study was performed by the Electrical Power Systems department of LMSC. The Study of Multi-kW Solar Arrays for Earth Orbit Applications was performed by Rick Mills. The panel hinge deployment stress analysis was performed by Eric Abrahamson from the Structures and Structural Dynamics Department. Cost analysis was provided by Mike Hitesman. The project organization is shown in Figure 1-2.

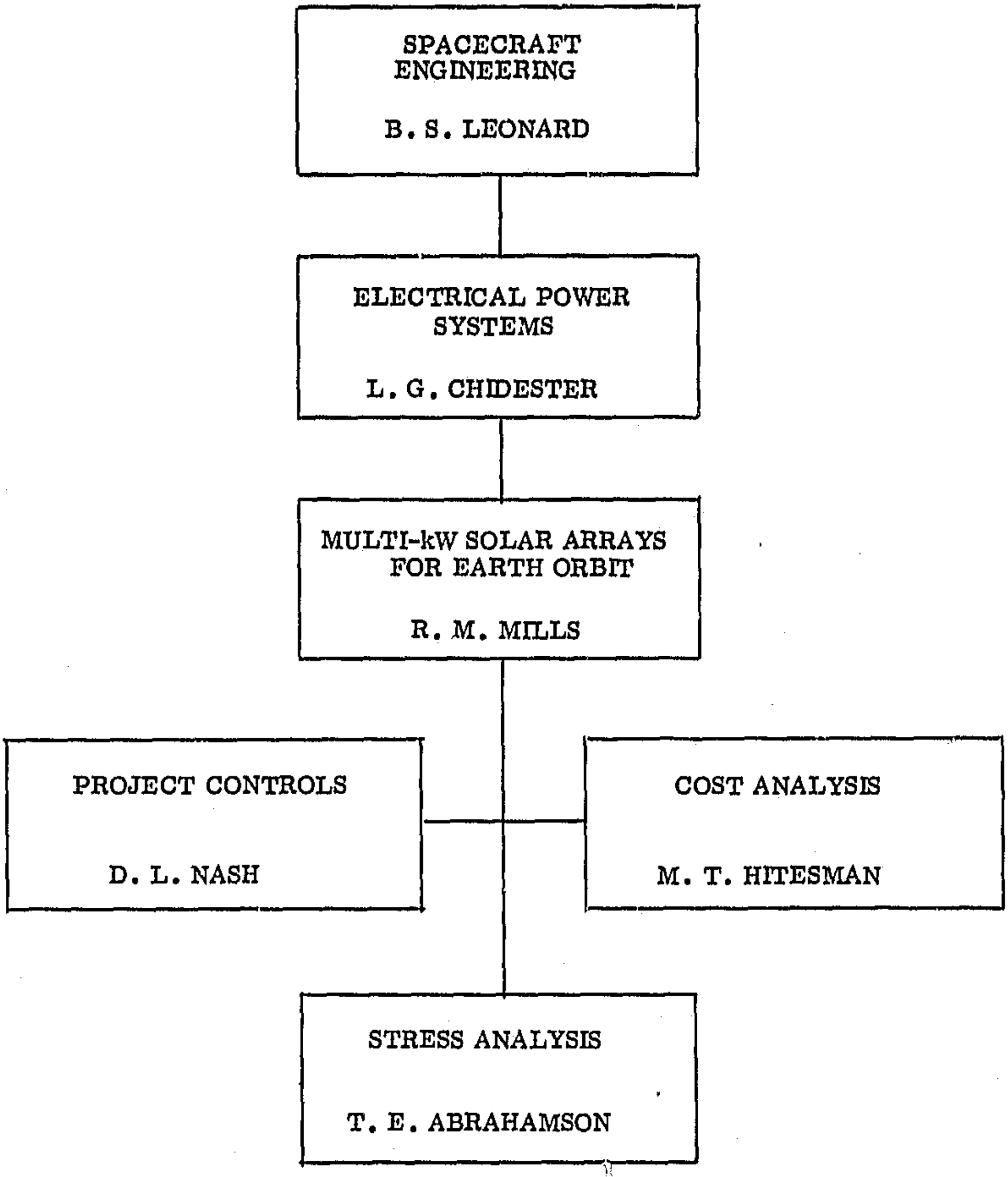

Figure 1-2 Project Organization

Section 2
TASK 1: SUPERSTRATE PANEL DESIGNS

The transparent superstrate panel design studies included: (1) the structure for attachment of the superstrate modules, (2) the module to module electrical interconnections; and (3) provisions for panel wire harness to route the solar array developed power. The designs considered handling in a 1g environment and the LEO environments and conditions including thermal excursions in which the array was projected to survive. The capability.to survive Shuttle environments was evaluated at a blanket level.

This task developed two panel designs which use the superstrates as the load carrying element and allows the removal of as much I-R absorbing surfaces in the optical path as possible. The Lockheed built Solar Array Experiment (SAE) is used as a point of departure for comparing panel designs and design to criteria.

2.1 STRUCTURE SUPPORT FOR SUPERSTRATE MODULES

The criteria for the design of the structural support of the superstrate modules are:

- Provide the mechanical connections between superstrate modules
- Provide sufficient rigidity for proper refolding
- Provide proper refolding forces along the hingelines
- Provide a panel packing factor of 81% or more
- Allow the superstrate modules to determine the overall panel thickness

Two mechanical designs are developed: (1) the SAE derived graphite epoxy frame, and (2) the continuous hinge stiffeners.

In both of these mechanical designs the superstrates are tensioned when fully deployed and provide the panel rigidity during the initial refolding of the blanket. These two functions were performed by the kapton superstrate and the cross panel graphite stiffeners on the SAE design.

2.1.1 Graphite Epoxy Frame

The structural support of the superstrate modules in this design approach is a graphite epoxy frame shown in Figure 2-1. This frame will support 14, 24-cell superstrate modules. It is designed to be the same thickness as the solar cell and the graphite epoxy will be layed up such that its expansion coefficient matches that of the superstrate glass. This frame will provide the blanket flat conductor cable (FCC) harness attachments at the ends of the panel. The superstrates will overlap the frame .025 inches and be bonded to the frame at this overlap. Five (5) spring steel leaf springs similar to those used on the SAE design will be bonded to the frame along the hingeline--2 at the ends, 1 at the center point, and 2 at the quarter points. These refolding springs provide 1.25 in-lb of moment along each panel hingeline. The hinge in this design will be bonded to the frame.

2.1.2 Hinge Stiffeners

In this design approach the structural support for the 14, 24-cell superstrate modules is provided by a continuous fiberglass or graphite epoxy hinge stiffening element shown in Figure 2-2. A flexible .001 inch thick element of molybdenum or titanium is bonded to each superstrate along the hingeline. This element is mechanically attached to the hinge stiffener and can be supplementally bonded, if necessary. The ends of the stiffeners are slotted to capture the blanket harness at the folds. The continuous hinge stiffener is the same thickness as the superstrate/cell stack. This panel structural design differs significantly from the SAE design but has certain benefits which make it a promising panel design.

2.1.3 Structural Support Comparison

Both the SAE derived graphite frame and the continuous hinge stiffeners are viable means of providing structural support for the superstrate modules. They have provisions for attaching the flat conductor harness to the ends of the panel. The

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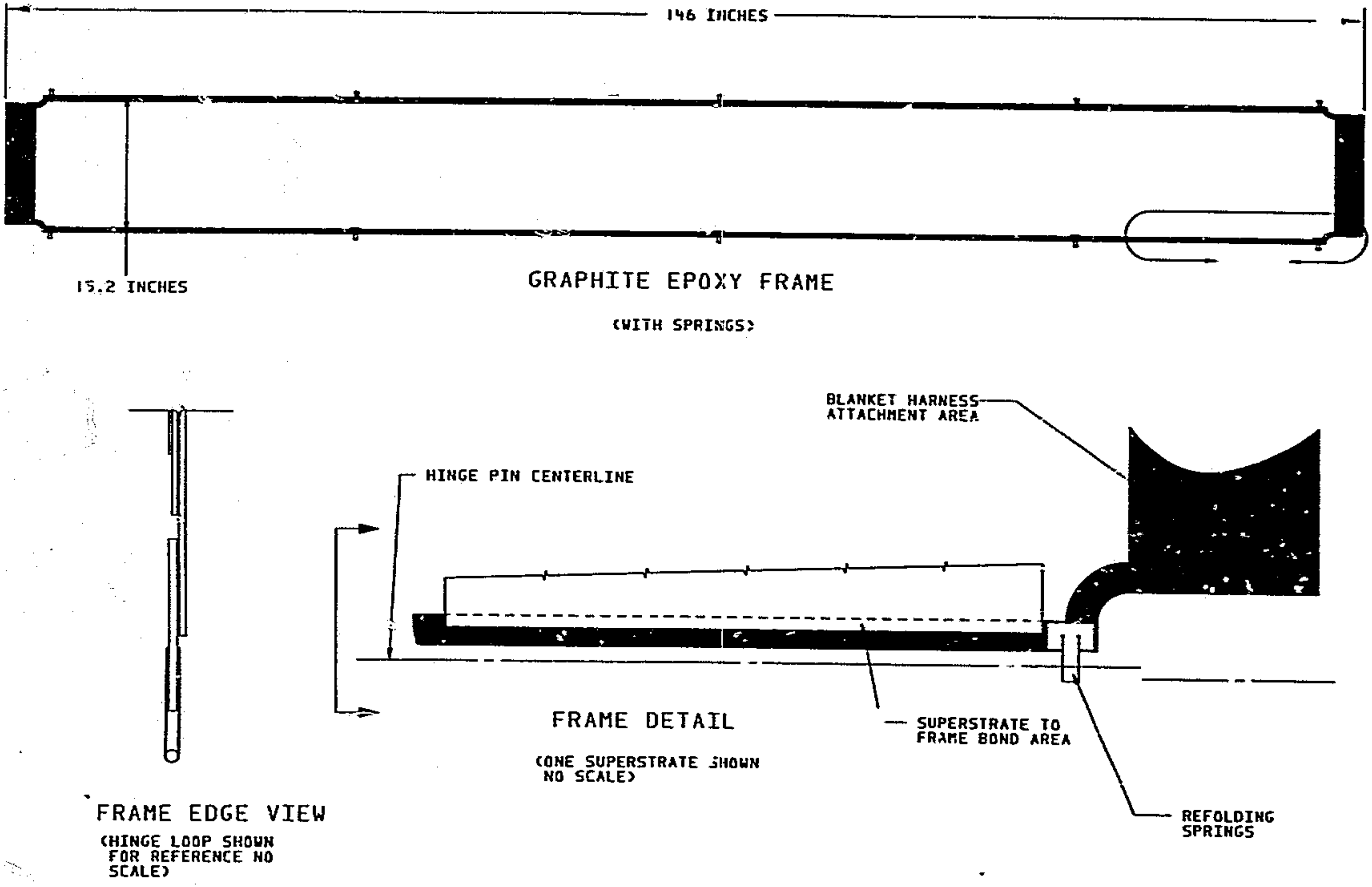


Figure 2-1 Panel Structural Support–Graphite Epoxy Frame

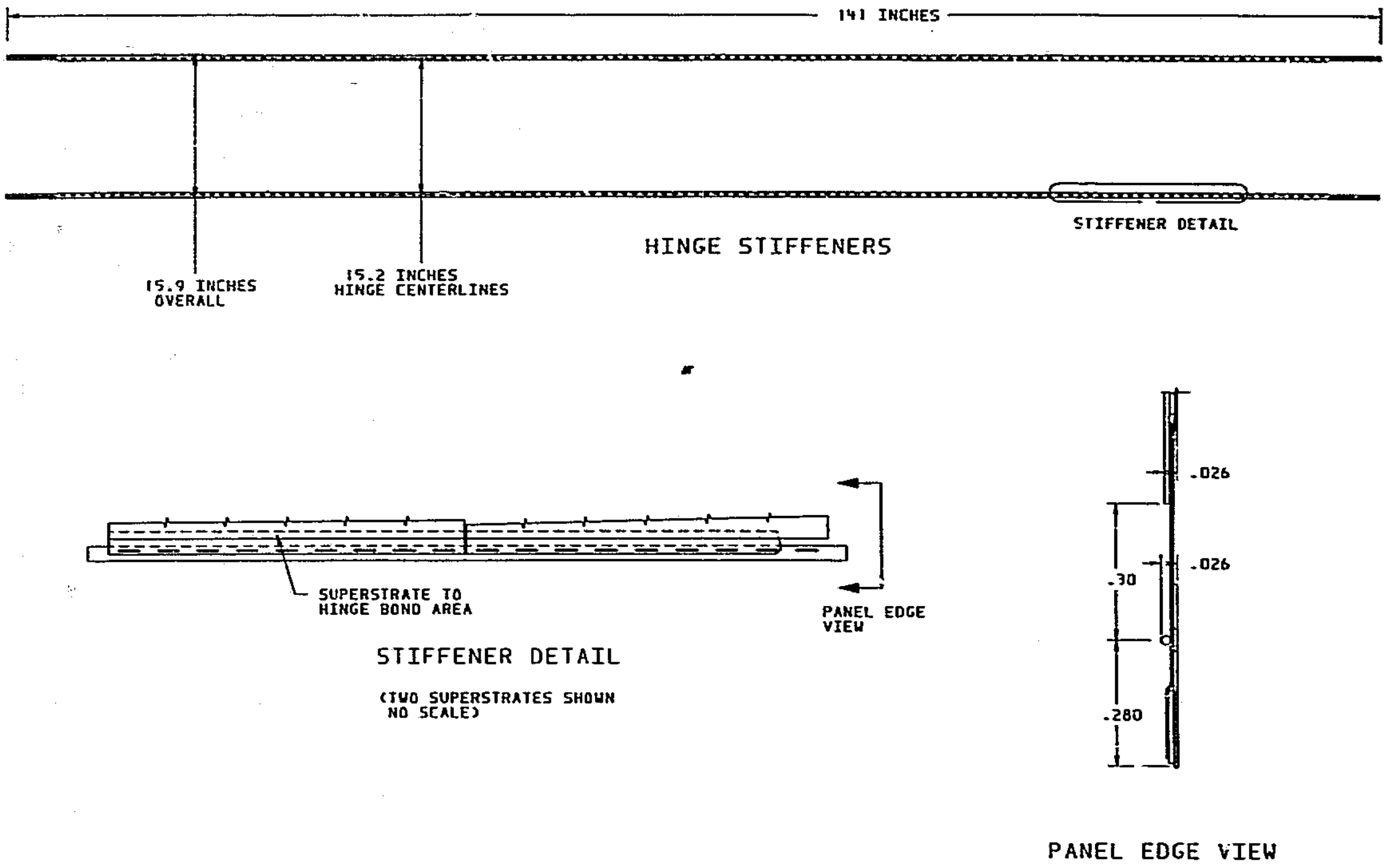


Figure 2-2 Panel Structural Support – Hinge Stiffeners

significant difference is that in the frame approach the superstrates are bonded directly to the panel structural support, whereas, in the hinge stiffener approach the superstrates are attached to the panel structural support through a flexible element. Table 2-1 compares the two support approaches.

TABLE 2-1

PANEL STRUCTURAL SUPPORT COMPARISON

FEATURE	GRAPHITE EPOXY FRAME PANEL		HINGE STIFFENED PANEL	
FABRICATION BONDS	SUPERSTRATE TO FRAME SPRINGS TO FRAME HINGES TO FRAME HARNESS TO FRAME		SUPERSTRATE TO FLEXIBLE FOIL HINGE ELEMENT	
THICKNESS	DETERMINED BY THE SUPERSTRATE ASSEMBLY (.026 IN.)		DETERMINED BY THE SUPERSTRATE ASSEMBLY (.026 IN.)	
PANEL PACKING FACTOR (EXCLUDING HARNESS AREA)	84%		88%	
WEIGHT SUMMARY	FRAME	.106 LB	STIFFENERS	.063 LB
	HINGE	.073	HINGE	.055
	HINGE PIN	.013	HINGE PIN	.013
	SPRING	.015		
	ADHESIVES	.033	ADHESIVES	.008
STRUCTURE	109 gm	.024 LB	64 gm	.139 LB
14 SUPERSTRATE MODULES	1.63 kg (1.60 LB)		1.63 kg (3.60 LB)	
TOTAL PANEL WEIGHT ESTIMATE	1.75 kg (3.84 LB)		1.70 kg (3.74 LB)	

2.2 MODULE TO MODULE ELECTRICAL INTERCONNECTIONS

Two different module to module electrical interconnect designs were developed, one for each panel structural design.

The criteria used for the module to module electrical interconnect design were:

- Accommodate I-R transparency
- Use Lockheed's flight proven kapton copper interconnect technology
- Use proven joining techniques

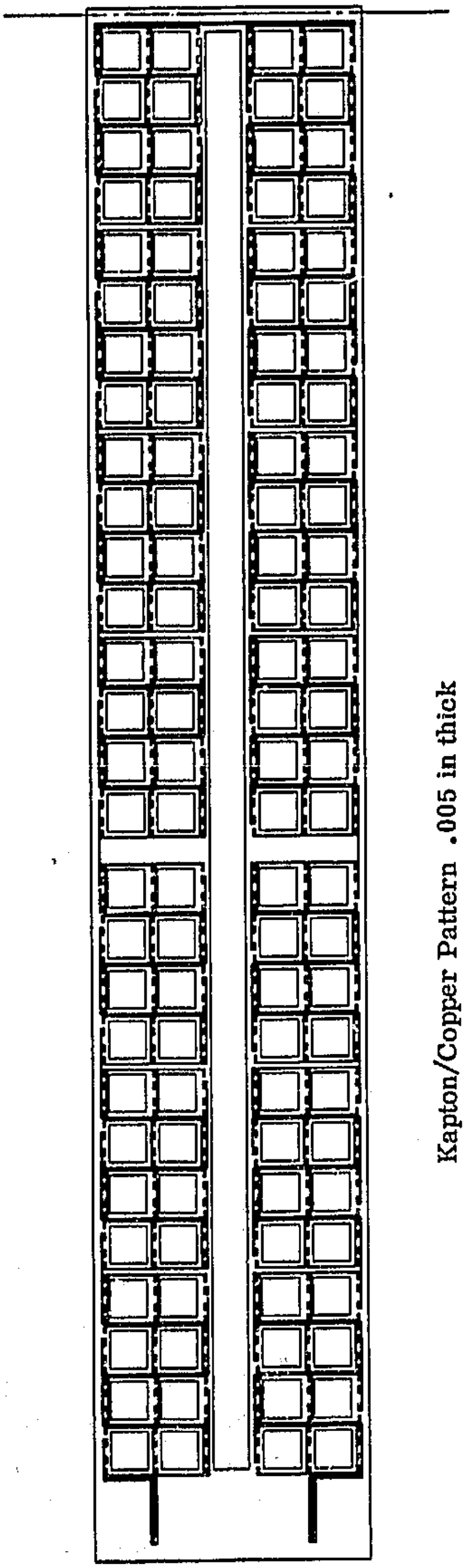
Both designs use a laminated kapton copper pattern which is welded to the wraparound contact pads of a gridded back 5.9 x 5.9 cm solar cell and both designs allow the removal of 75% of the kapton insulation from behind the cell to enhance I-R transparency.

2.2.1 Solar Array Experiment Derived Half Panel Interconnect

Figure 2-3 shows SAE derived half panel interconnect which is welded to the back of each cell in 8 places. This interconnect pattern connects 168 cells in series without additional module to module connections and is welded after the 7 modules are located on the frame. The positive and negative leads are attached to the blanket harness by soldering the leads to the proper harness conductors. This electrical interconnection design uses the technology developed for and flight proven on the SAE.

2.2.2 Module to Module Electrical Connections

The hinge stiffeners panel approach requires additional electrical connections from module to module. The circuit layout is shown in Figure 2-4 showing the common module kapton-copper cell interconnect pattern and the detail of the module to module electrical connection. The interconnect pattern tabs are soldered to a copper trace on the hinge stiffener away from the superstrate to reduce the possibility of damage to the superstrate assembly. The overlapping area of the module IC with the copper



Kapton/Copper Pattern .005 in thick

Figure 2-3 Solar Array Experiment Derived Half Panel Interconnect

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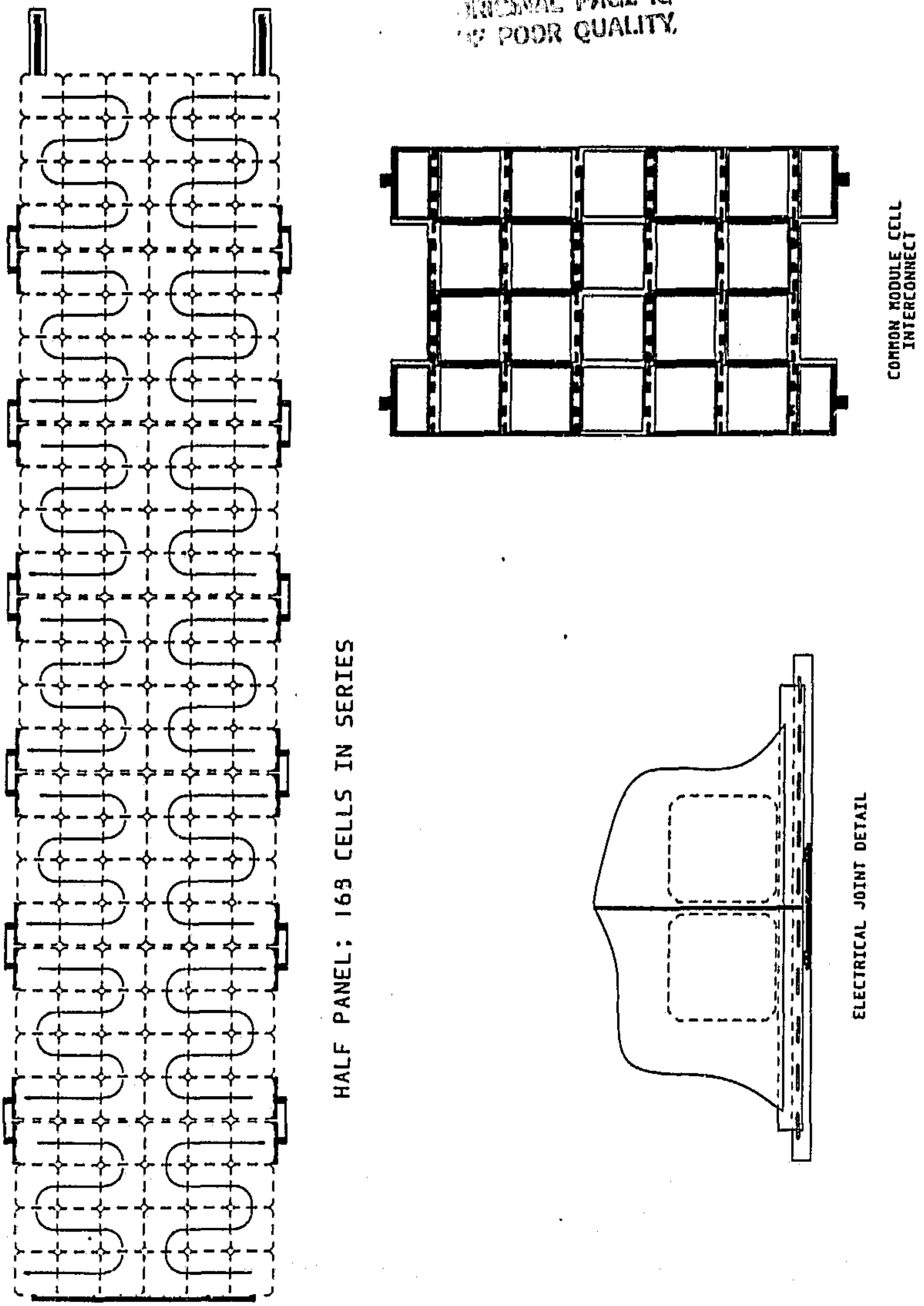

Figure 2-4 Module to Module Electrical Connections

trace segment on the stiffener can be sufficiently large to accommodate redundant joints and repair joints. The positive and negative panel leads on the end module are soldered to the appropriate harness conductor using SAE developed and proven techniques. This electrical design allows module removal and replacement by simply unsoldering the module from the hinge stiffeners and disassembling a portion of the hinge.

2.2.3 Electrical Interconnections Comparison

The two panel approaches required different module to module electrical designs. A comparison of the two electrical designs is given in Table 2-2. Both designs accommodate I-R transparency, use Lockheed's flight proven kapton copper I-C technology, and both use proven electrical joining techniques.

TABLE 2-2

ELECTRICAL INTERCONNECTION COMPARISON

FEATURE	HALF PANEL INTERCONNECT	MODULE TO MODULE ELECTRICAL CONNECTIONS
NUMBER OF WELDS	1344	1344
NUMBER OF SOLDER JOINTS	4	52
MODULE ELECTRICAL DISCONNECTION	DIFFICULT – TRACES MUST BE CUT AND RECONNECTED	SIMPLE – UNSOLDER EIGHT SOLDER JOINTS
AMOUNT OF KAPTON REMOVED	70–75%	70–75%
SPACE REQUIRED FOR MODULE TO MODULE CONNECTIONS	NONE	NONE

2.3 SUPERSTRATE PANEL EVALUATIONS

The two panel designs described in Sections 2.1 and 2.2 have been evaluated in terms of the following requirements:

- Ability to be fabricated, assembled, and handled in the 1g environment
- Ability to withstand deployment dynamics and loadings
- Ability to withstand and operate in the LEO environment including thermal cycling

Both the SAE derived panel design and the hinge stiffener panel design are evaluated.

2.3.1 Ground Handling and Assembly

Panel designs using glass superstrates are by nature very fragile. These glass panels are approximately 12 ft long and weigh almost 4 lb and the panel structural support designs do not protect the superstrates. The size of the panel and the materials used present significant assembly and ground handling problems. This section describes the ability of the graphite frame panel and the hinge stiffener panel to withstand fabrication processes, assembly, and handling.

The graphite frame panel assembly steps are shown in Figure 2-5. First, the cells will be bonded to the superstrate. The frame fabrication and assembly can occur in parallel with module fabrication. Bonding the hinge to the frame is a key structural bonding step which requires close dimensional tolerances. A bonding fixture will be necessary to position and align the hinge elements and springs. The half panel interconnect will then be welded in 8 places per cell to all of the 168 cells per half panel. The panel is then ready for test and assembly into a blanket.

The hinge stiffened panel assembly steps are shown in Figure 2-6. First, the module interconnect is welded to the cells and the superstrate is bonded to the cells using previously developed techniques. Bonding the two hinge elements to the cell/ superstrate assembly is a key fabrication step. The position and alignment of the hinge elements requires close tolerancing so that the assembled hingelines are

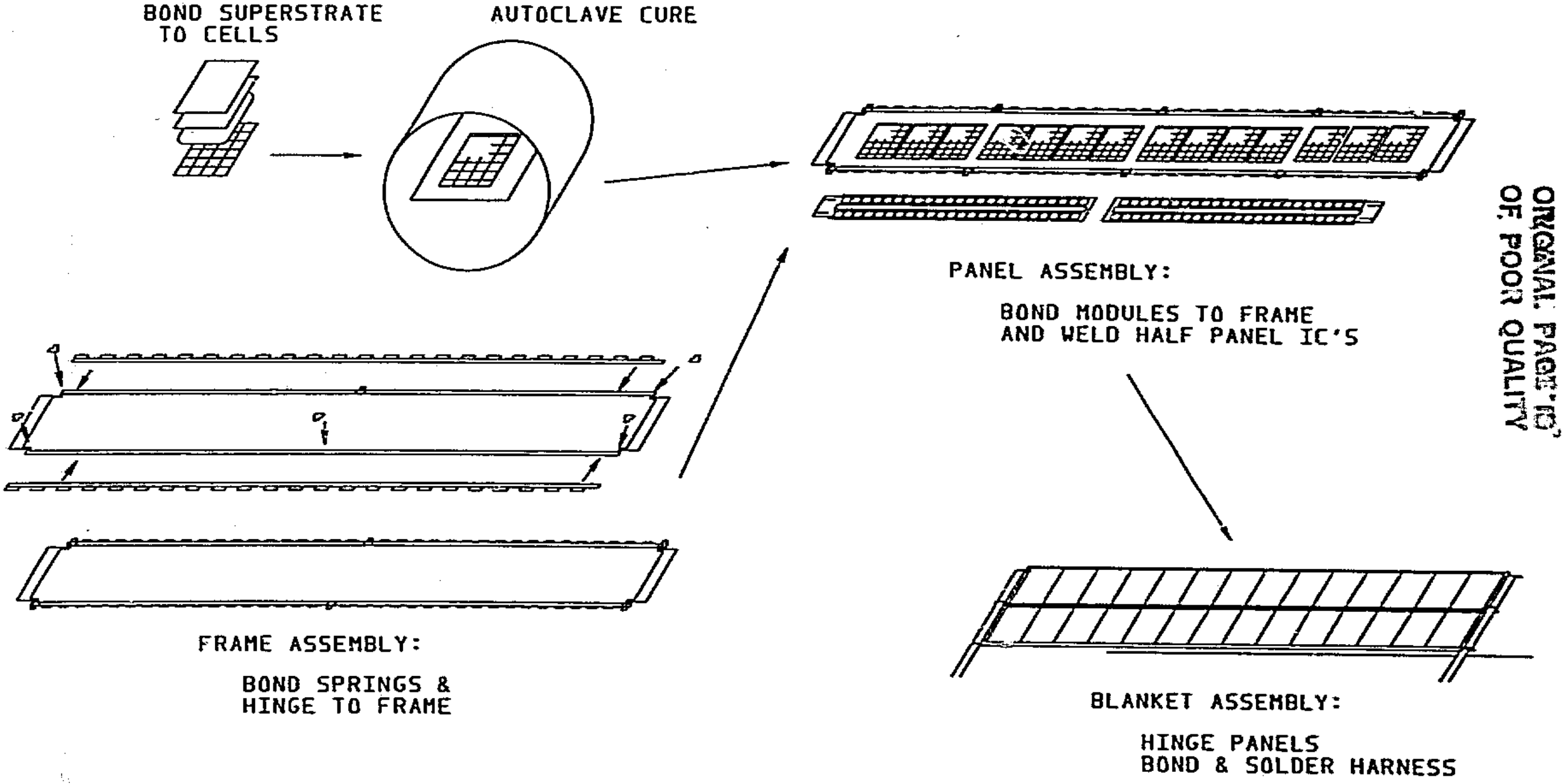

Figure 2-5 Graphite Epoxy Frame Assembly Sequence

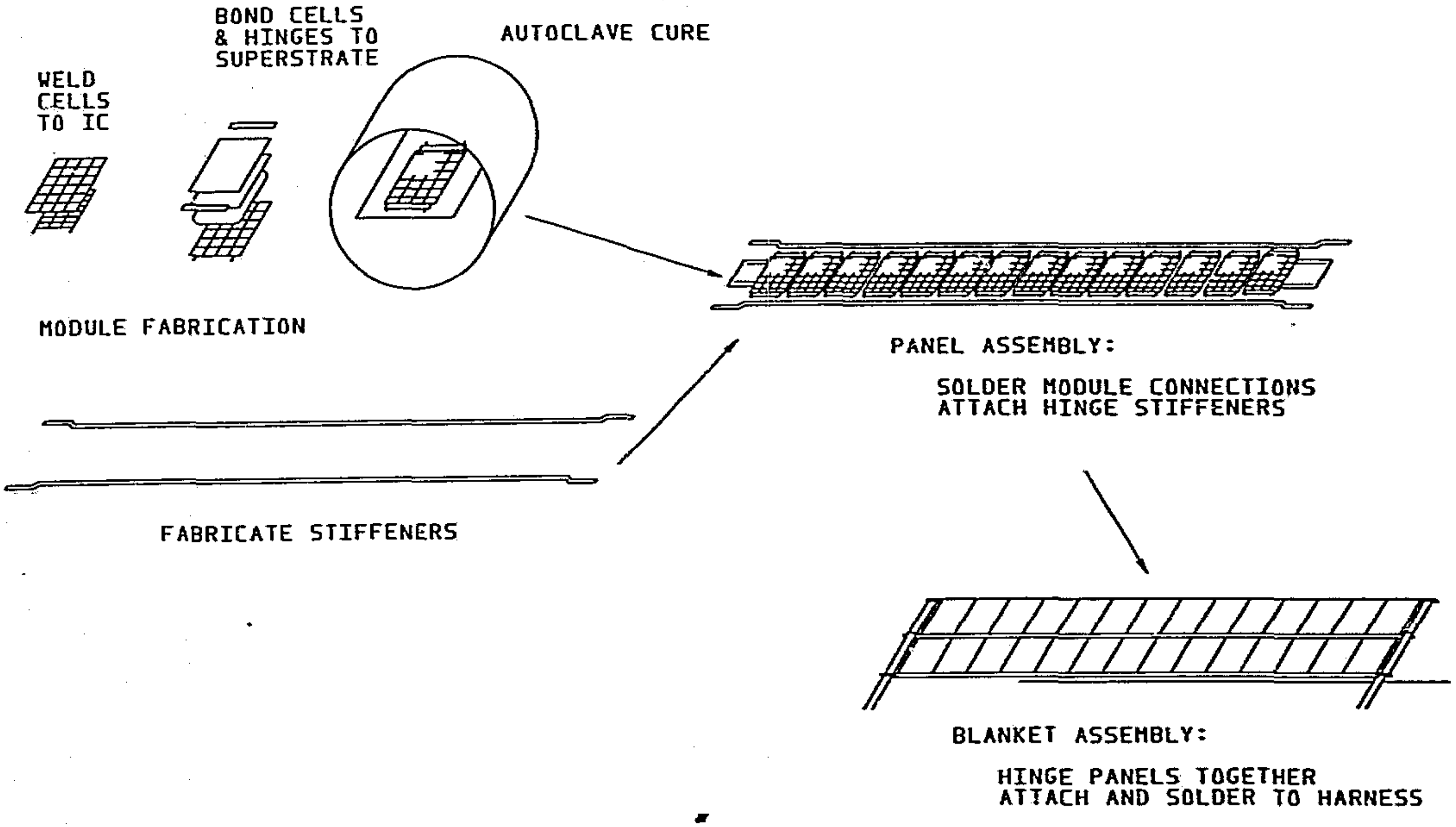

Figure 2-6 Hinge Stiffener Assembly Sequence

parallel and straight. A bonding fixture will be required to provide hinge element positioning. Next, the panel is assembled by attaching the hinge stiffener assemblies to the 14 modules and the module to module electrical connects are soldered. The panel is ready for test and assembly into a blanket.

Both panel designs would be subjected to the same ground handling environment which is summarized in Table 2-3. Normal panel ground handling can subject the panel to bending, torsion, vibration and shock loading as well as ambient temperature and relative humidity. The panel attached to a handling fixture design similar to that shown in Figure 2-7 should be capable of withstanding any orientation in the 1 g field. It has been demonstrated that a .020 in thick module superstrate can withstand a center deflection of a half an inch for 100 cycles when simply supported 15 inches apart. Using the handling fixture and additional padding when necessary and proper procedures, the effects of bending, torsion, vibration, and shock can be removed.

Cleaning methods have been established and environmental tests conducted at LMSC on all materials used in this design. Additional tests on the specific structural bonds developed for these designs must be performed to establish processes and any environmental protection necessary.

2.3.2 Low Earth Orbit Environment

Both panel designs must withstand and operate normally throughout the array's expected life in the following low earth environment.

Orbit:	Altitude	555 km (300 nmi)
	Inclination	28°
Vacuum:	1×10^{-11}	$N\text{-}m^{-2}$
Micrometeoroid:	10^{-3} to 10^{-6} g/cm^3 particles	
Radiation:	Trapped electrons and protons	
Lifetime:	5 years, 26,000 orbits	
Temperature:	-105°C to +30°C	

This orbit was selected as typical of low earth orbit conditions. The space vacuum encountered in LEO has been considered in the selection of panel materials used.

TABLE 2-3

PANEL GROUND HANDLING ENVIRONMENT

MANUFACTURING PROCESS	LOADING CONDITION	EVALUATION
BONDING PROCESSES	THERMAL STRESS DURING CURING CYCLE	STRUCTURAL BONDING SHOULD BE CLOSELY MONITORED FOR ALIGNMENT AND BOND QUALITY.
PANEL ASSEMBLY	MECHANICAL STRESSES	CAN BE MINIMIZED BY HANDLING FIXTURE DESIGNS.
WELDING	LOCAL THERMAL STRESS	ACCOMPLISHED USING PROVEN PROCEDURES.
HANDLING	BENDING TORSION VIBRATION SHOCK LOADING	WITH A HANDLING FIXTURE SUPPORTING, THE PANELS CAN WITHSTAND ANY ORIENTATION. PAD THE SUPERSTRATE WHEN SHOCK LOADING IS LIKELY.
CLEANING	LOCAL MECHANICAL STRESSES MATERIAL DEGRADATION DUE TO SOLVENTS	USE ESTABLISHED CLEANING PROCEDURES AND PRECAUTIONS.
ENVIRONMENT	AMBIENT TEMPERATURE 10-50°C RELATIVE HUMIDITY 30-80%	NO KNOWN DETRIMENTAL EFFECTS.

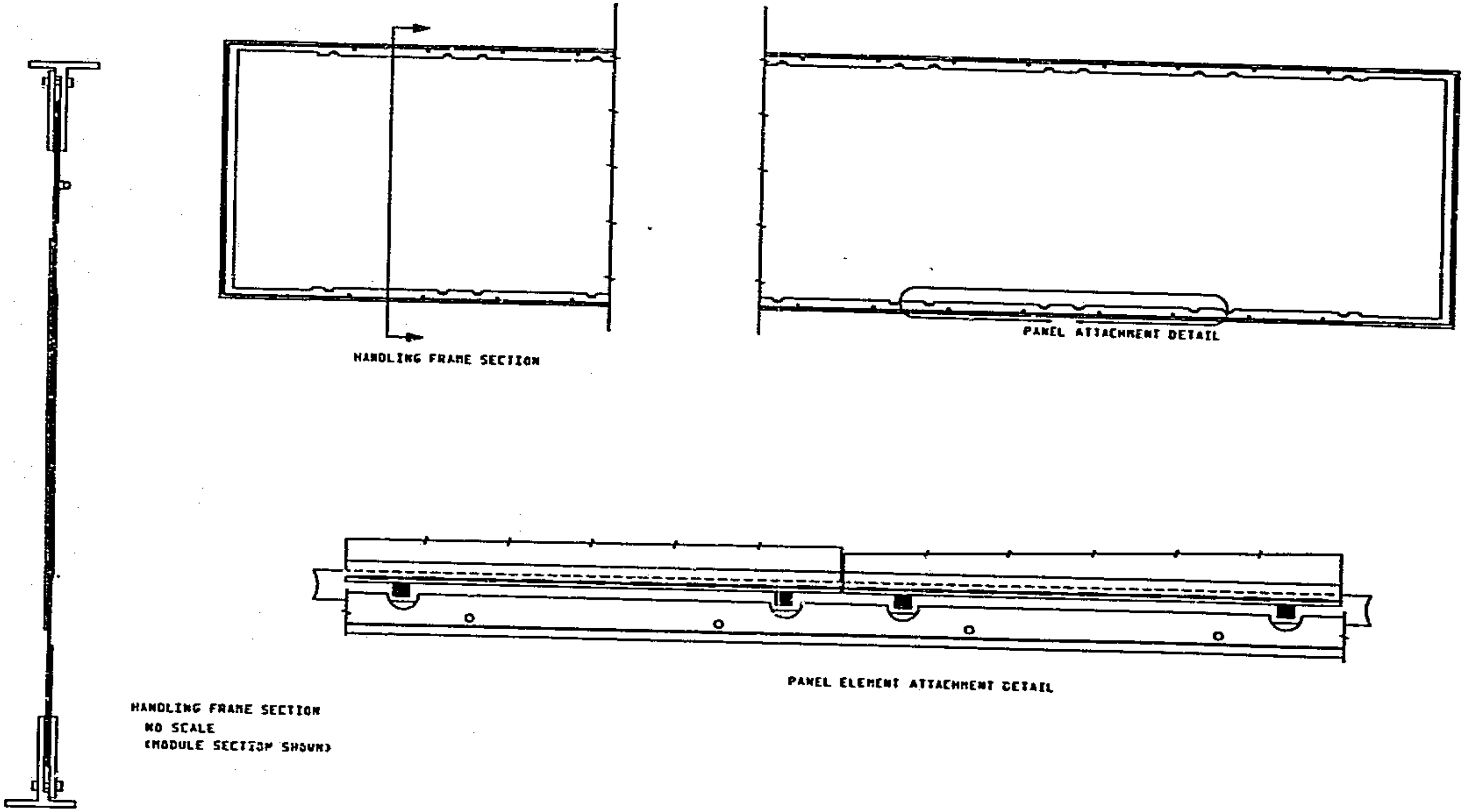

Figure 2-7 Hinge Stiffened Panel Handling Frame Concept

Space proven materials have been selected with low outgassing characteristics. Adhesive will be outgassed prior to their application. It has been reported that ordinary glass is 3 times as strong in a vacuum than it is in air. The surface finish of the superstrate glass and metallic foils can be expected to significantly influence their strengths in vacuum and should be investigated further in this specific application.

The micrometeoroids of mass 10^{-3} to 10^{-6} gm/cm^3 are of primary concern to superstrate panels. Micrometeoroid damage is primarily erosion of the front and back surfaces of the panel. A small possibility of heavier particles impacting the panel and cracking the superstrate exists. The SAE derived frame can accommodate a cracked superstrate. The effect on blanket tension of a cracked superstrate on the hinge stiffened panel has not yet been investigated. A cracked superstrate is expected to remain held in place by the cell to superstrate adhesive and scrim cloth; therefore, a catastrophic failure is not expected due to a single cracked superstrate.

Radiation degradation on silicon solar cells due to low earth orbit trapped electron and protons is well understood and is accounted for in the panel and blanket on orbit performance. Both panel designs use kapton only as an electrical insulator, not as a structural element; thus, the kapton degradation with atomic oxygen problem is an electrical concern but not a structural problem.

Over a 5-year lifetime in a 300 nmi orbit, the panel will be subjected to 26,200 thermal cycles with temperature extremes of -105°C to +30°C. The primary concern in surviving the thermal cycle lifetime is minimizing or allowing for thermal expansion mismatches. Material selection for both designs are shown in Table 2-4. The structural bonding necessary to fabricate either superstrate panel design which will withstand the thermal cycle environment remain to be selected. The materials to be joined have been selected in this study. Alternative structural bonds remain to be tested and evaluated. Selecting structural bonding systems and demonstrating reliable and durable bond is not anticipated to be a large obstacle.

This evaluation indicates that both the SAE derived design and the hinge stiffened design have the capability to withstand the low earth orbit environment. The designs

TABLE 2-4
PANEL MATERIAL SELECTION

Design	Material	Function	Thermal Expansion Coeff. 10^{-6} in/in/°F (10^{-6} m/m/°C)	Comments
SAE Derived Panel	Microsheet glass	Superstrate	4.1 (7.4)	
	Graphite Epoxy	Structural support frame	~4.1 (7.4)	Layup designed to match CTE's with microsheet
	Molybdenum foil	Hinge	2.7 4.9	CTE close to microsheet glass
	Titanium foil		~5.1 (9.2)	
	Graphite, fiberglass	Hinge pin		CTE is not critical
	Epoxies, silicone, other	Adhesive bonds hinge to frame superstrate to frame springs to frame harness to frame		Preferred bonding system TBD
Hinge Stiffened Panel	Microsheet glass	Superstrate	4.1 (7.4)	
	Molybdenum foil	Hinge	2.7 (4.9)	CTE close to microsheet glass
	Titanium foil		5.1 (9.2)	
	Epoxy, silicone, other	Adhesive bond hinge to superstrate		Preferred bonding system TBD
	Graphite epoxy fiberglass	Hinge		CTE is not critical

use Lockheed developed and space proven fabrication processes and assembly techniques. The majority of materials selected have a history of use in the space environment. The second task of this study incorporates these two alternative panel designs into blankets and evaluates the blankets.

Section 3

TASK 2: SUPERSTRATE PANEL TO PANEL DESIGN

The basic question addressed in this task is how to incorporate the superstrate panel designs described in the previous section into superstrate blankets. In this task the detail of the panel to panel hinge is described for each panel design. A detailed stress analysis was performed on the hinge developed for the hinge stiffened panel to determine its capability to withstand deployment dynamics and blanket tensions. The blanket harness was designed for minimum weight. A stowage design which is common to each blanket design is described and an evaluation of the stowed blanket to survive shuttle launch and reentry is made. This section concludes with a performance comparison of the two blankets and a discussion of repairability.

A solar array which is flat folded into a stowed configuration requires a hingeline which:

- Rotates through 180° from folded to deployed
- Is the same thickness as the superstrate panel so as not to impose stack height differences
- Can be tensioned to .2 lb/linear in along the hingeline when fully deployed
- Provides an initial out of plane force for proper refolding
- Does not impose additional stresses on the superstrate

3.1 SOLAR ARRAY EXPERIMENT DERIVED PIANO HINGE

A hinge derived from the Solar Array Experiment which meets the hinge requirement is shown in Figure 3-1. This piano hinge is derived from the kapton piano hinge used on the SAE blanket. The hinge loops attach along two edges. The graphite frame panel will be bonded to the frame between the hingeline leaf springs. The hinge loops will be cut from thin (.001 in.) molybdenum foil. The leaf springs provide 1.25 in-lb

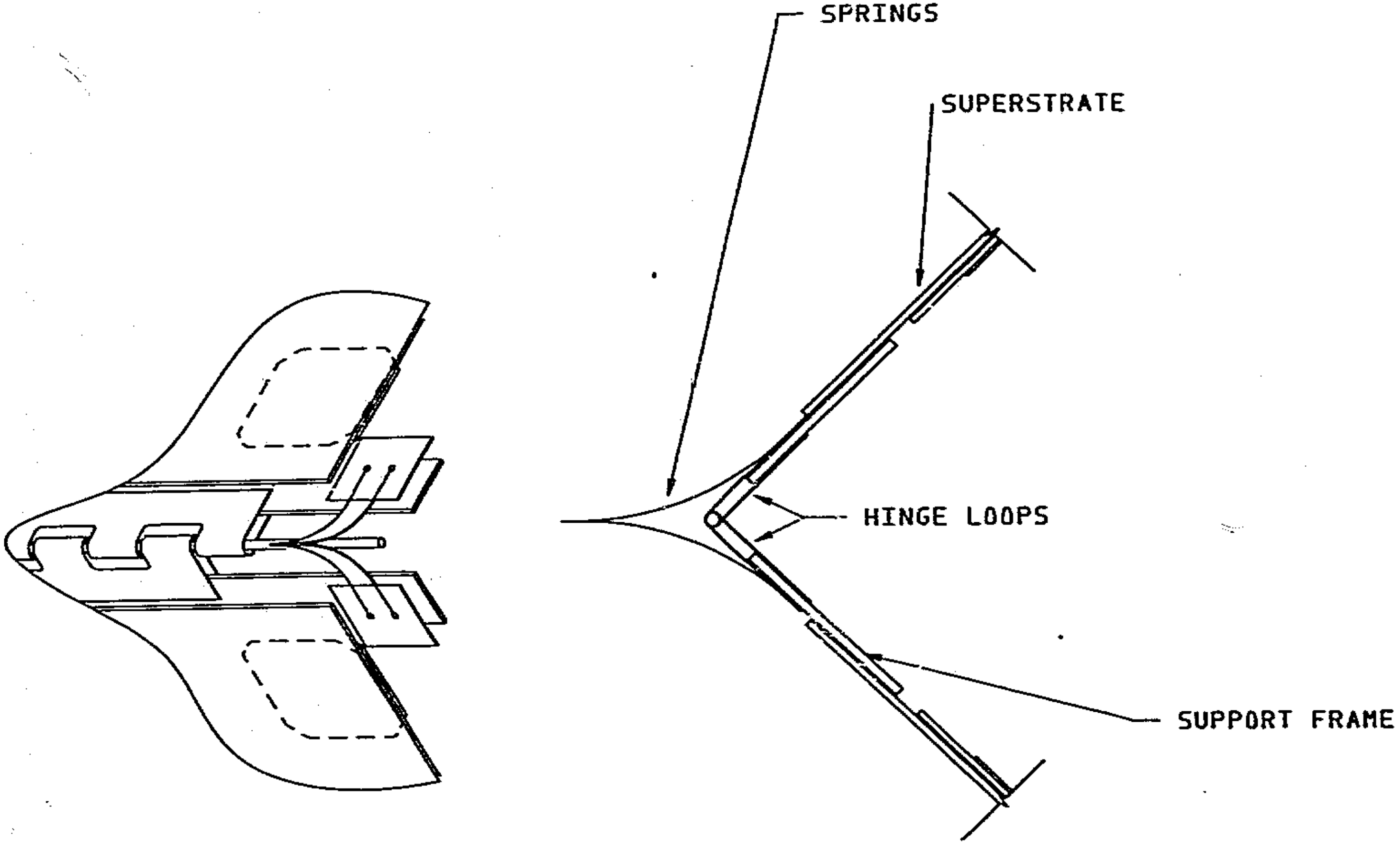

Figure 3-1 SAE Derived Piano Hinge

per hingeline of refolding torque required to assure proper blanket refolding. The hinge and frame require a distance of approximately 0.65 in. from the edge of the cells to the hinge centerline in order to provide area for the frame to superstrate and frame to piano hinge bonds. The bond area widths used here which are similar to SAE give a panel packing factor of 84% compared to the SAE packing factor of 81%*. The panel to panel mechanical connection is made by inserting a .020 in. diameter hinge pin into the alternating hinge loops. This hinge meets all requirements above; it can be rotated through 180°, it is less than the stack height of two superstrate panels, it can be tensioned to 0.2 lb/linear inch, discrete leaf springs provide the refolding force, and the superstrate is not bonded directly to the hinge.

*This packing factor increase is due primarily to less space between cells, not a reduction in the area required for the hinge and frame.

3.2 HINGE STIFFENED PANEL ELASTIC HINGE

The hinge stiffened panel uses an elastic hinge concept which also satisfies the hinge requirements. Figure 3-2 shows the design details and assembly of this hinge, and Figure 3-3 shows a model of this hinge design. The lower hinge half is formed into hat-shaped loops by making three permanent bends in the molybdenum foil. The upper hinge element is formed by cutting or etching evenly spaced slots in the molybdenum foil. Two panels are hinged together by aligning the hinge stiffeners "back-to-back." The loop on one hinge lower half will extend through slots in its stiffener, the next panel's stiffener and through the slots in the mating hinge half. A 0.020 hinge pin is inserted into this hinge loop. The pin mechanically captures the two mating hinge stiffeners and the two mating hinge halves. Each molybdenum hinge half bends through 90° during deployment and the hinge stiffeners do not rotate. At full deployment, the hingelines have the stiffeners alternatively extending from the front and back of the blanket.

These hinge stiffeners are potential locations for guide wire holes and bypass diodes. The hinge provides the out of plane refolding force making discrete springs unnecessary. This feature increases the panel packing factor excluding harnesses to 88% compared to 81% on the SAE.

Since this elastic hinge is a significant departure from a piano hinge the question of the stresses in the hinge parts and superstrates during deployment were analyzed.

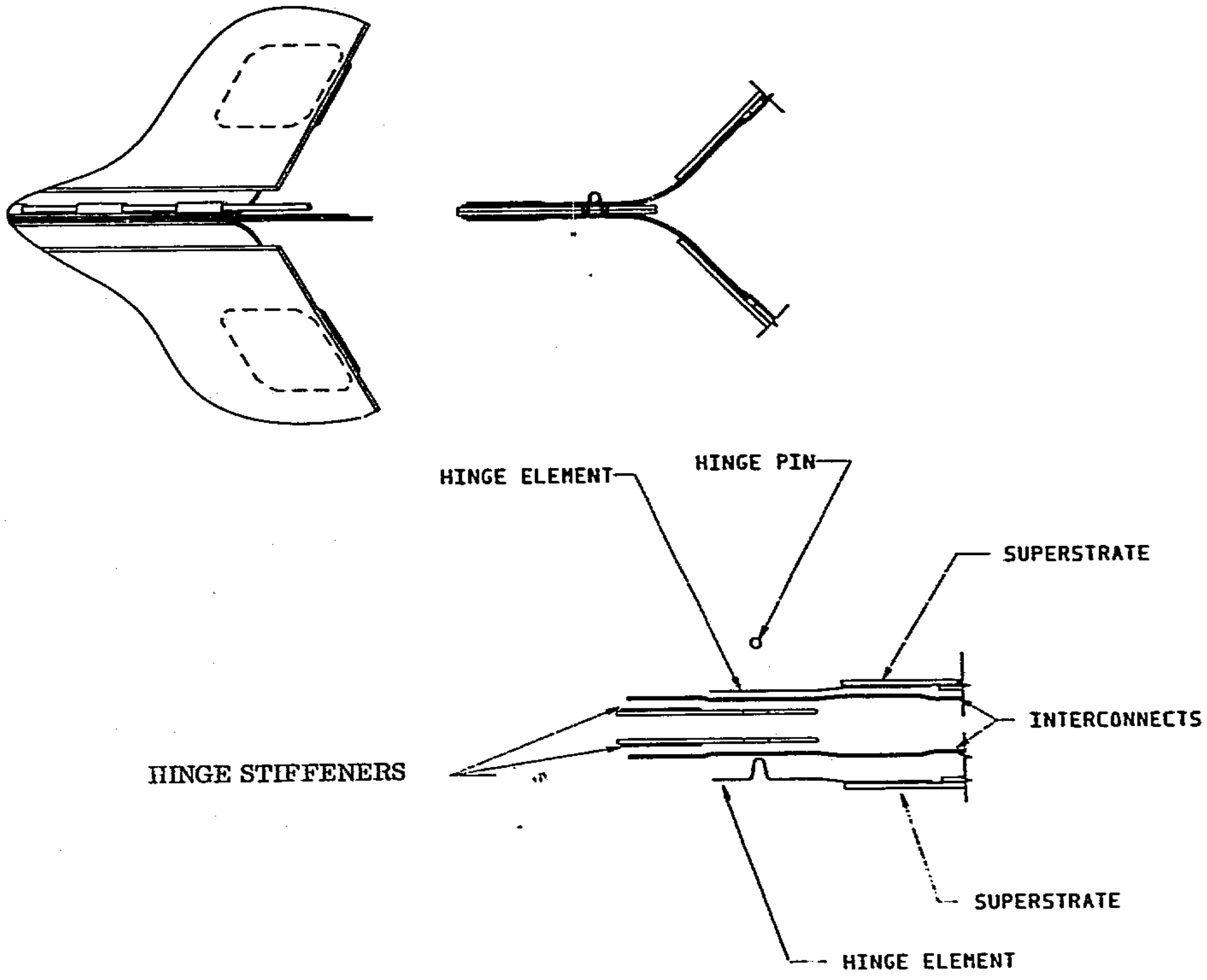

Figure 3-2 Hinge Stiffened Panel; Elastic Hinge Details

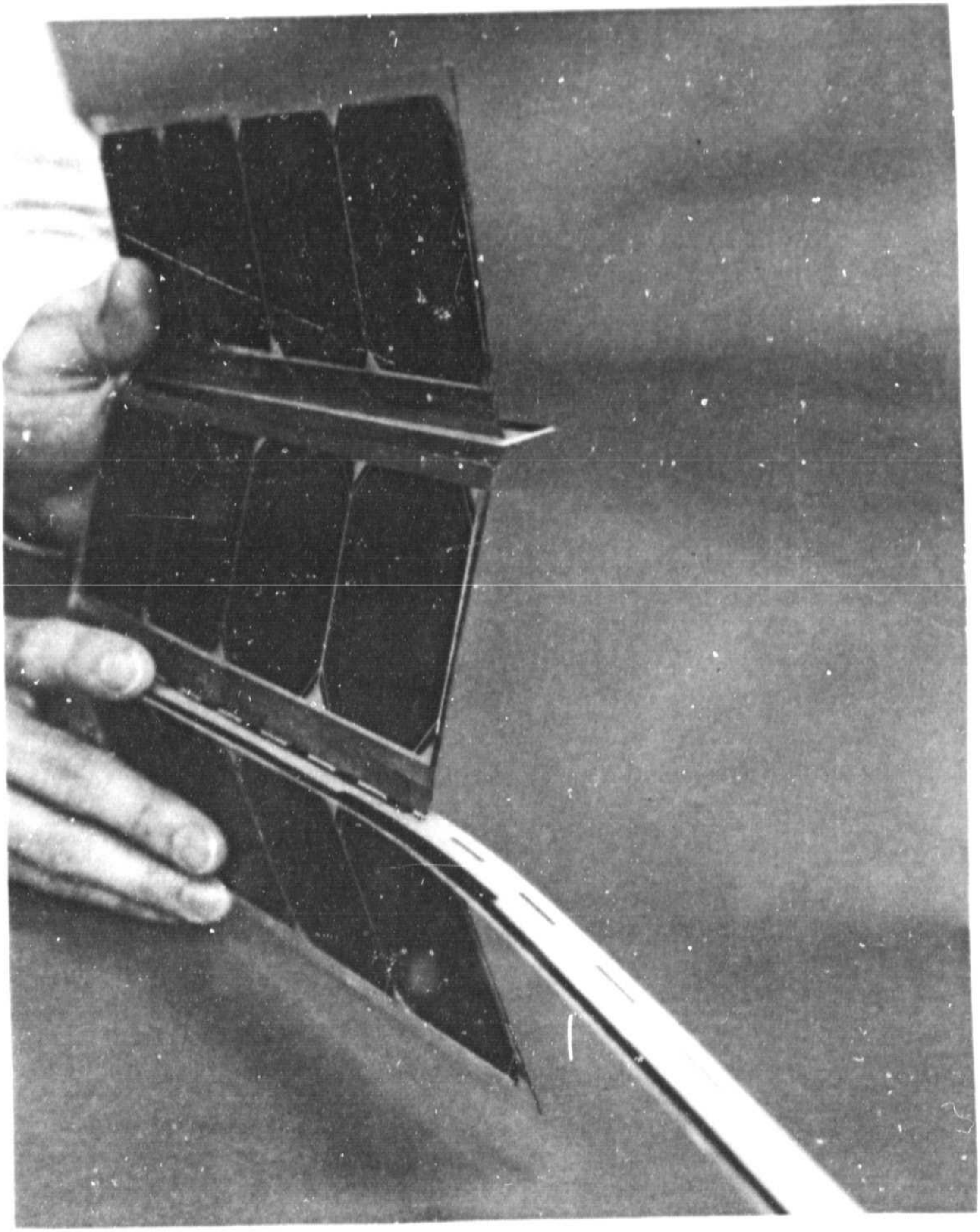

Figure 3-3 Elastic Hinge

3.3 ELASTIC HINGE STRESS ANALYSIS

The elastic hinge design requires the molybdenum foil to bend elastically to a 90° angle during deployment without failure or yielding and return to a flat condition when the blanket is folded. The question of what stresses are present in the superstrate and hinge parts during deployment are analyzed.

A stress model was developed and a LMSC Nonlinear Elastic-Plastic Structural Analysis finite element program was used to incrementally analyze the large deformations in the hinge element and the stresses in the hinge parts. The four hinge half cases below were analyzed at various stages of deployment.

1. The hinge pin side: superstrate bonded to the outside of the hinge element (shown in Figure 3-4).
2. The side opposite the hinge pin: superstrate bonded to the outside of the hinge element.
3. The hinge pin side: superstrate bonded to the inside of the hinge element
4. The side opposite the hinge pin: superstrate bonded to the inside of the hinge element.

The material properties used in this analysis are given in Table 3-1 and a summary of the maximum stresses for case one is shown in Table 3-2.

These stresses occur at a blanket tension of 2.5 to 0.3 lb/in perpendicular to the hingeline, which is well above the tension required for this type of flat folded blanket. Figures 3-5 through 3-8 are the stress plots obtained for the fully deployed case 1. Cases 2, 3, and 4 gave similar results.

The highest effective stress of 45,900 psi was found to be in the .001 in thick molybdenum hinge element near the hinge pin. 45,900 psi is approaching the yield strength of molybdenum. To reduce this stress to an acceptable factor of safety,

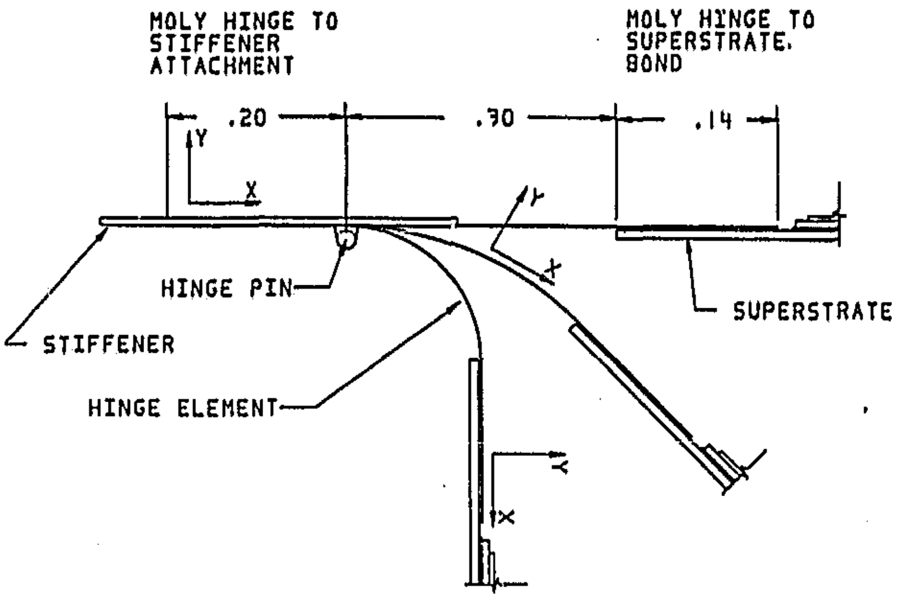

Figure 3-4 Stress Analysis Hinge Model - Case 1

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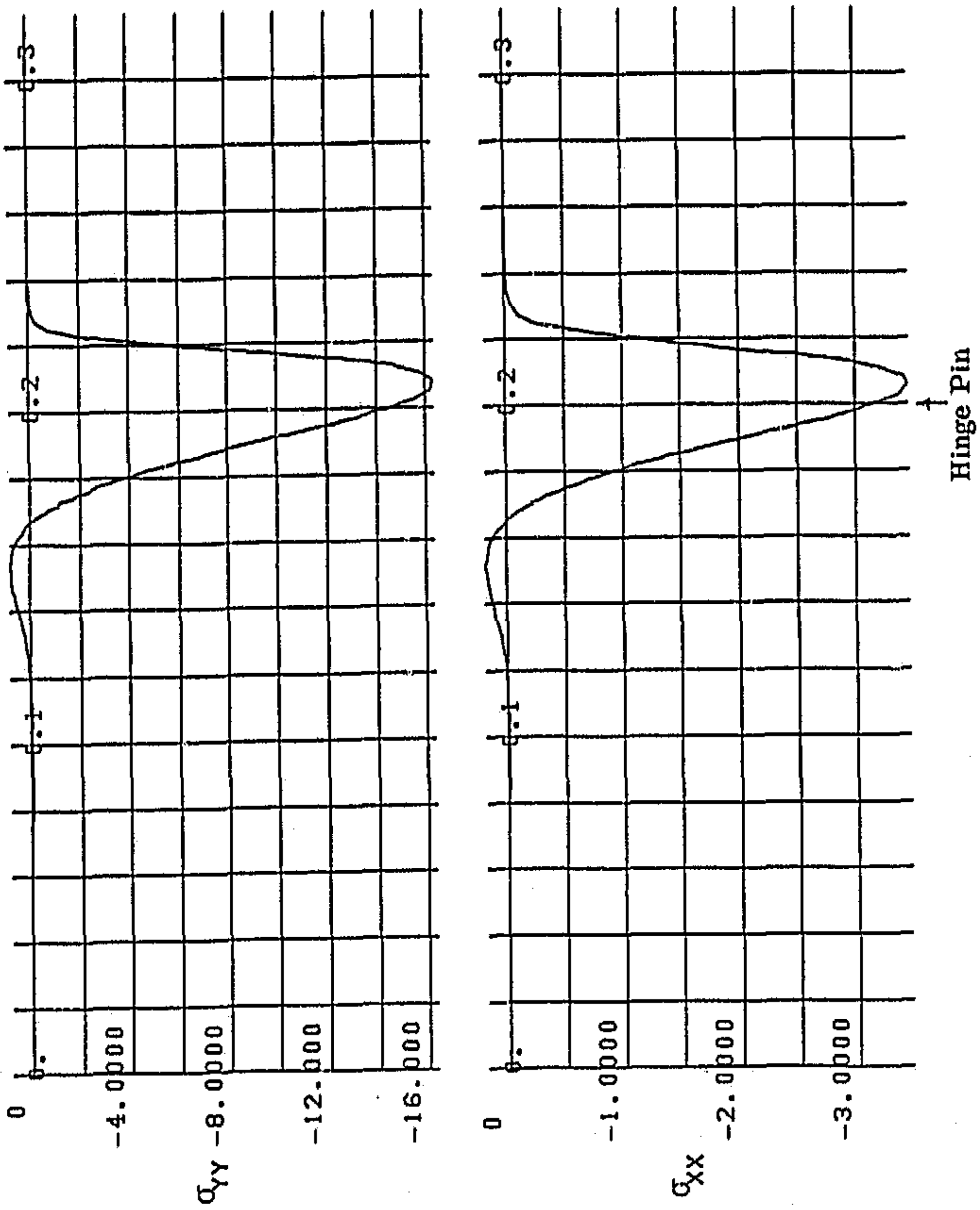


Figure 3-5 Stresses in the Hinge Stiffener

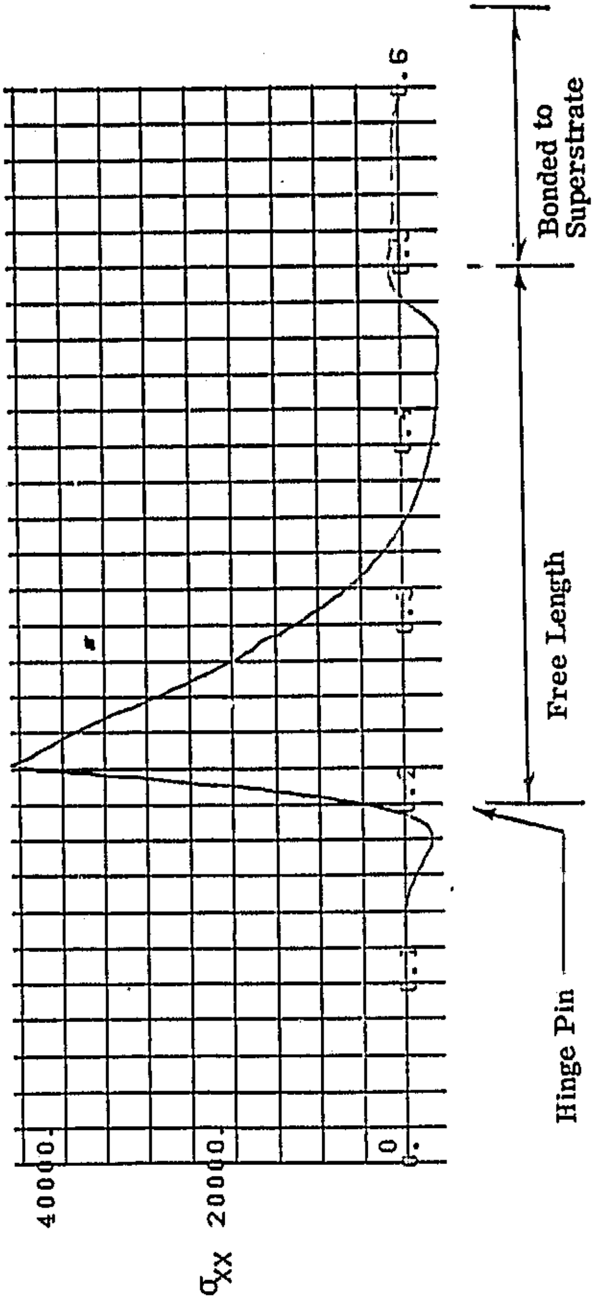


Figure 3-6 Stresses in the Hinge Element

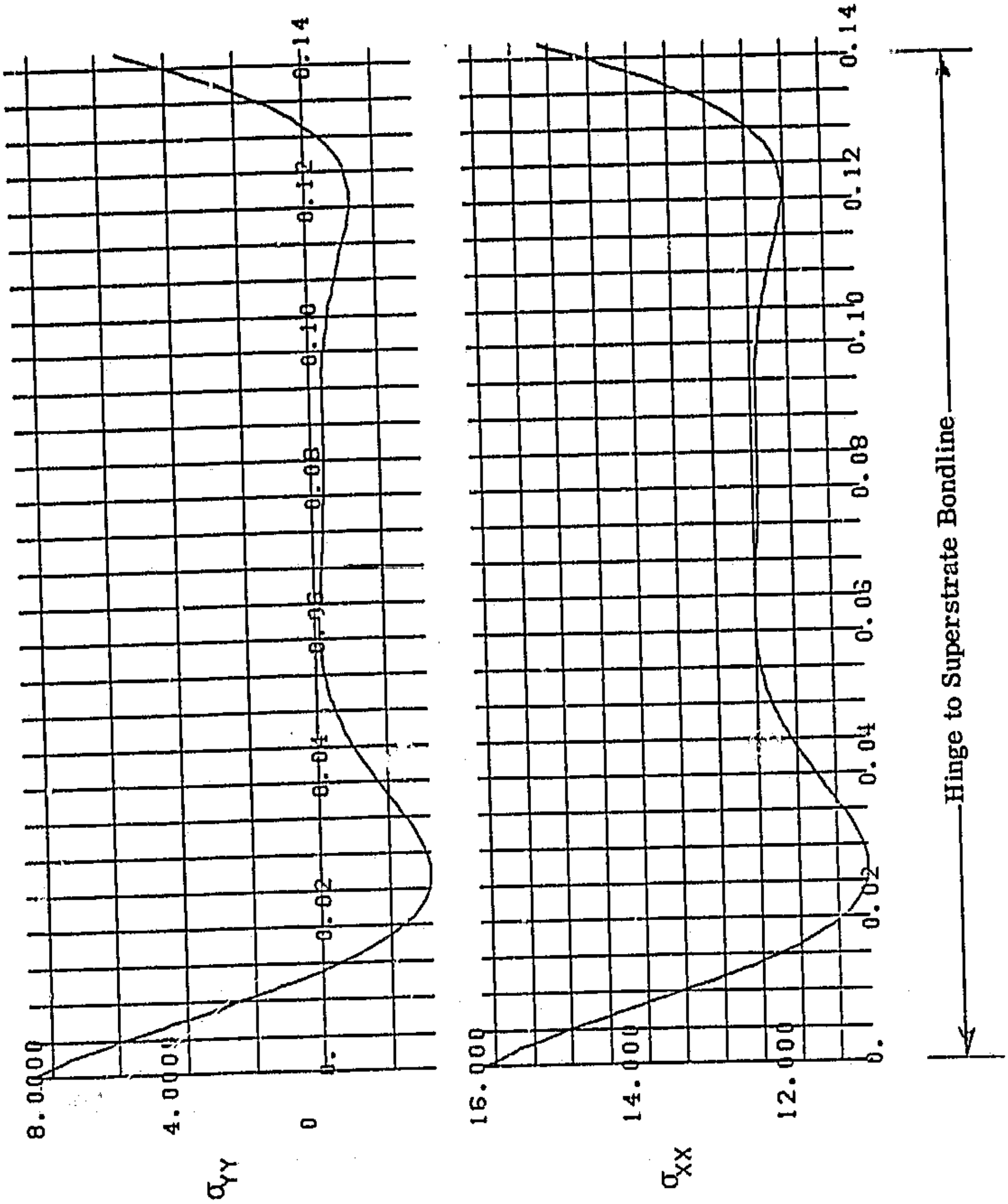


Figure 3-7 Stresses in the Hinge to Superstrate Bond

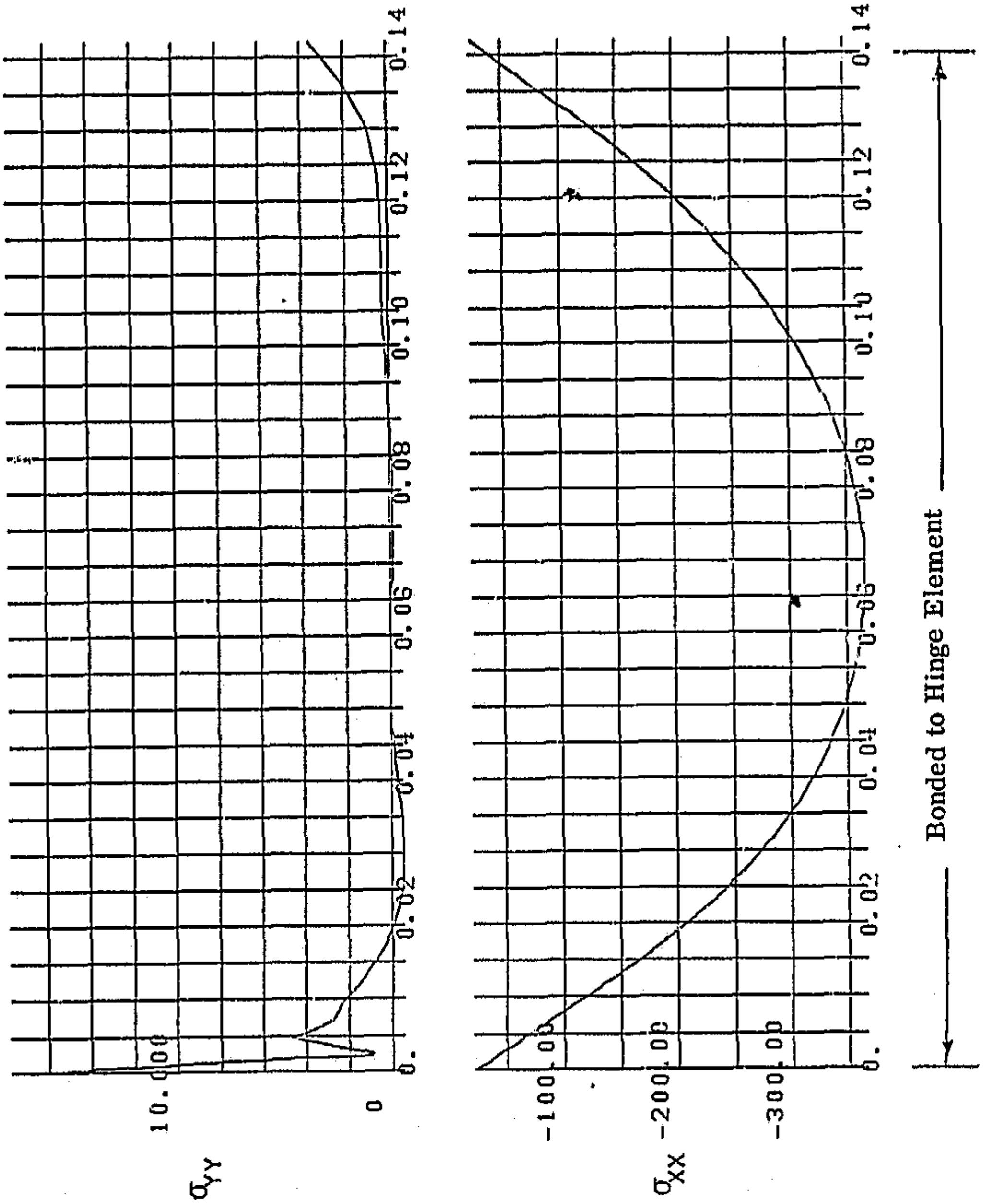


Figure 3-8 Stresses in the Superstrate

TABLE 3-1
HINGE MATERIAL PROPERTIES

At 75°F	Superstrate	Molybdenum	Silicone Adhesive
Elastic Modulus (psi)	10.8×10^6	41.0×10^6	153
Poisson Ratio	.22	.30	.46
Thermal Expansion Coefficient (/°F)	4.1×10^{-6}	2.9×10^{-6}	160×10^{-6}
Yield Strength (psi)	1000	50.9×10^3	500

Stress plots were obtained for several hinge positions during deployment and all four configurations were investigated. A summary of stresses in each of the four configurations at full deployment are shown in Table 3-2.

TABLE 3-2
MAXIMUM VON MISES EFFECTIVE STRESS SUMMARY (IN PSI)

	Hinge Half Case Number			
	1	2	3	4
Hinge Stiffener	16.1	39.7	16.1	39.7
Hinge Element	$45.9\text{x}10^3$	$43.3\text{x}10^3$	$45.8\text{x}10^3$	$43.2\text{x}10^3$
Hinge to Superstrate Bond	39.4	39.3	41.0	40.9
Superstrate	366	364	315	314

the tension could be reduced to a reasonable value below 0.2 lb/in; the arc length of the bend could be increased from 0.3 inches and the foil thickness could be increased.

Significantly, this analysis showed very small stresses in the hinge to superstrate bond and the superstrate itself. This indicates that the critical concern of large stresses in the hinge to superstrate bondline and superstrate itself will not occur during deployment and blanket tensioning.

3.4 BLANKET ELECTRICAL DESIGN

The blanket wiring harness required to connect to each circuit and bring the developed power to the base of the array was designed. The approach taken is to optimize the power loss in the harness by minimizing the combined weight of the harness plus blanket. Although the hinge stiffener blanket and the graphite frame blanket are physically different, electrically they are identical so that a common electrical schematic is used (Figure 3-9). Forty electrical circuits are used in a 80 panel single blanket high voltage array. This array size was chosen to be similar in size and layout (two flat conductor cables are attached to the ends of the panels) to the SAE design. The 672 gridded back 2 ohm-cm cells will produce 330 volts at 28°C, beginning-of-life. At this voltage, the optimum copper FCC on a weight basis loses 1.2% of the generated power and the two harnesses weigh a total of 6 lb.

The flat conductor harness design which meets this optimum power loss is summarized in Table 3-3. A thermodynamic analysis was performed on the shortest length conductor. The requirement was to determine the conductor width which when subjected to the worst case thermal conditions did not raise the temperature above 100°F (well below the softening temperature of thermosetting adhesives commonly used to bond FCC together). This width was determined to be .020 inches.

Although the electrical layout is the same for both the SAE derived panel and the hinge stiffener panel the details of the attachment of the panel electrical harness to the blanket FCC is different as shown in Figure 3-10. The half panel kapton copper interconnect used on the SAE panel must transition from the back of the module adjacent to the harness to the top of the harness in order to make the interconnect to harness solder joints. The FCC lays behind and is bonded to the graphite frame.

In hinge stiffener panel design, the FCC is attached to the panel at the ends of the stiffeners. The stiffeners are formed such that the two stiffeners form a slot when the hingeline is assembled and the FCC is captured in this slot. The FCC fold is radiused around one 0.008 inch stiffener and this stack height at the fold is including 2 layers of FCC, one stiffener, and adhesive is 0.052 inches which is the same

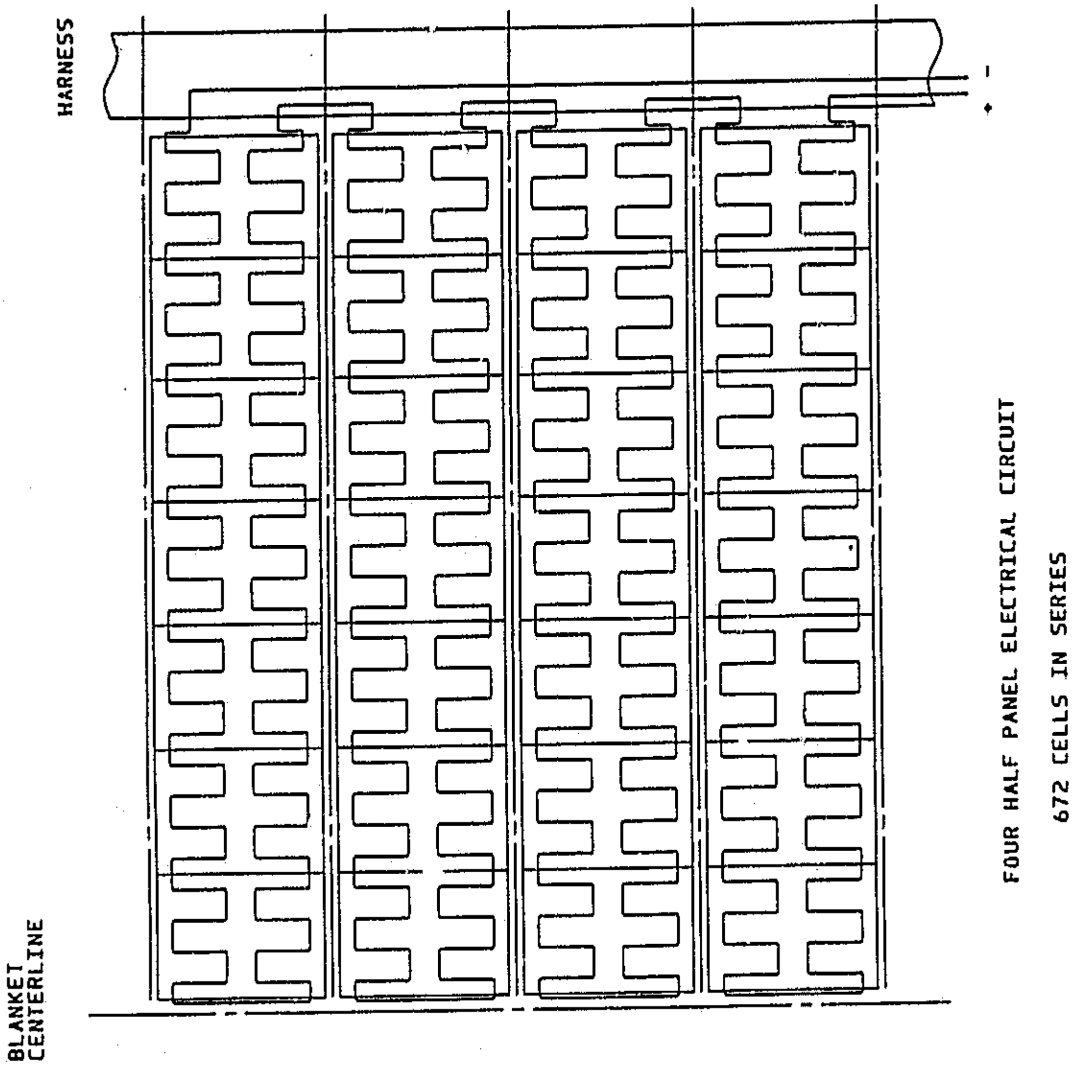


Figure 3-9 Blanket Electrical Circuit

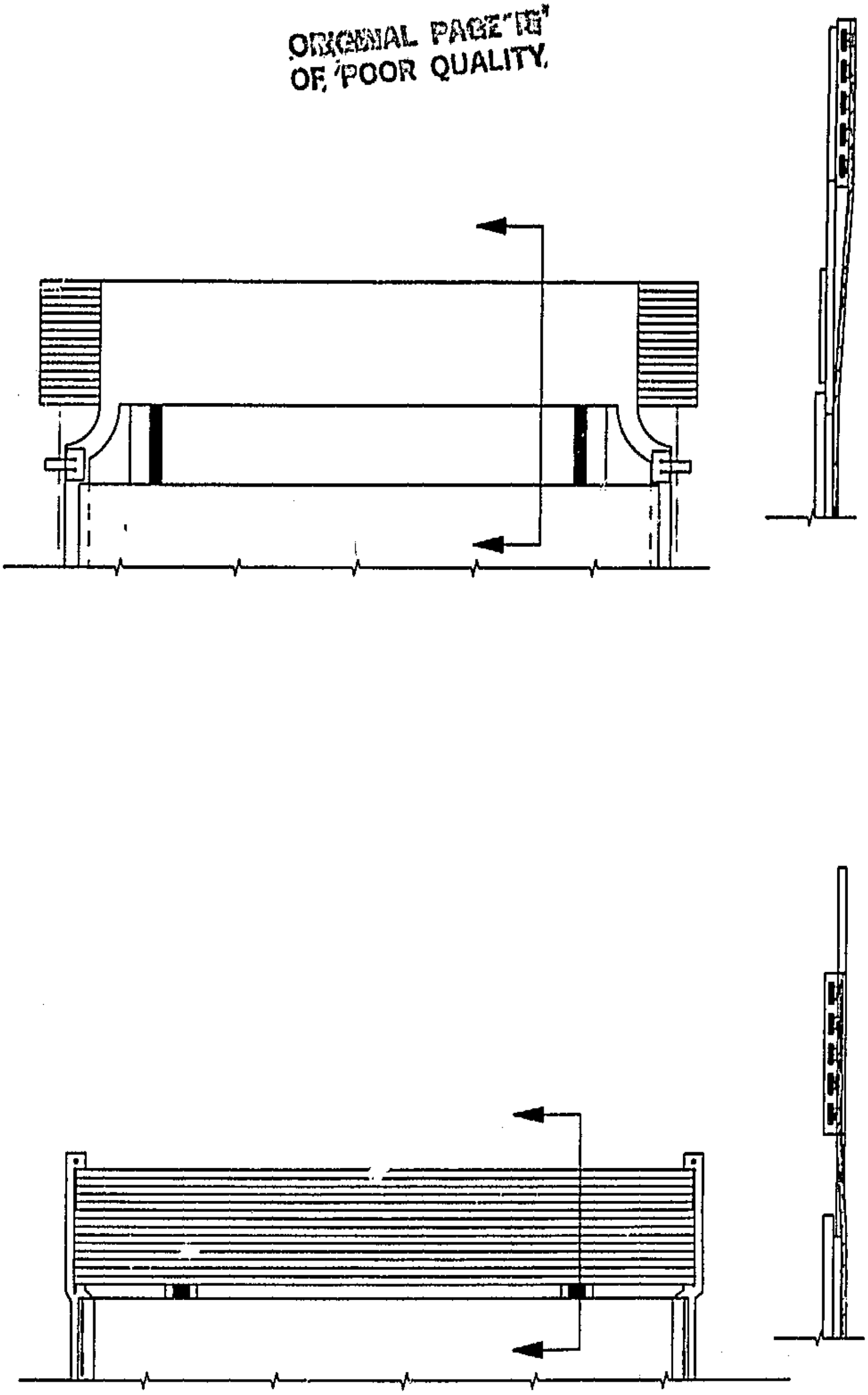
TABLE 3-3
HARNESS DESIGN

$L/\Delta_{opt} = 3.5 \times 10^6$/in

One of two harnesses required
(Copper conductor thickness 0.003 in.)

Circuit	Length (in.)	Conductor Width 2 ea Required (in.)
20	1209	0.114
19	1147	0.108
18	1085	0.102
17	1023	0.097
16	961	0.091
15	899	0.085
14	837	0.080
13	775	0.074
12	713	0.068
11	651	0.063
10	589	0.057
9	527	0.051
8	465	0.054
7	403	0.040
6	341	0.034
5	279	0.029
4	217	0.023
3	155	0.020
2	93	0.020
1	31	0.020

Weight including insulation:	3 lb
Width including insulation:	3 in



Harness Attached to the Frame

Harness Attached to the Hinge Stiffener

Figure 3-10 Harness Attachment Details

stack height as two panels. Any stack height variation can be accommodated by increasing the distance between the edge of the superstrate and the FCC and allowing the stiffeners to bend out of the plane of the panel. The distance between the fold lines of the FCC is larger than the distance between the panel hingelines so that the FCC is not tensioned at full deployment. The panel to harness solder joints can be made using SAE developed methods.

3.5 STOWED BLANKET SHUTTLE ASCENT AND REENTRY EVALUATION

The approach used in packaging the superstrate blanket into a stowed configuration is that the module stack height govern the completed blanket height. This approach precluded the use of frames around the modules and other structural elements thicker than the superstrated/cell interconnect stack. Such frames would allow vibratory motion of the modules and would therefore require panel interleaves or other interpanel padding. The two panel designs described here allow the modules to be stacked superstrate to superstrate and interconnect to interconnect.

The stowed package is shown in Figure 3-11. This design utilizes rigid pallets, the cover and container, to compress the blanket. Bonded to the pallets will be polyurethane foam padding covered with kapton which distributes the preload over the blanket stack similar to that used on the SAE.

During Shuttle ascent and reentry the stowed superstrate blanket will be subjected to low frequency loads expressed in g's, structural random vibration, and acoustic noise. The stowed and preloaded blanket must be capable of withstanding the combined effects of loading, vibration, and noise. The approach taken was to analyze the preload required to prevent any motion of the blanket in the worst case conditions.

An evaluation of the stowed blanket configuration was performed in order to determine the minimum preload required on the folded blanket stack required to survive Shuttle Orbiter launch and landing loads. In this analysis a worst case load factor of 6.6 g's in the Shuttle X-direction and 8.0 g's in the Shuttle Z-direction during landing were assumed. These low frequency load factors were used for design assessment for the SAE. The analysis determined the minimum preload required to resist impending blanket slippage between panels. The model used and the results are shown in Figure 3-12 with the following variables and equation.

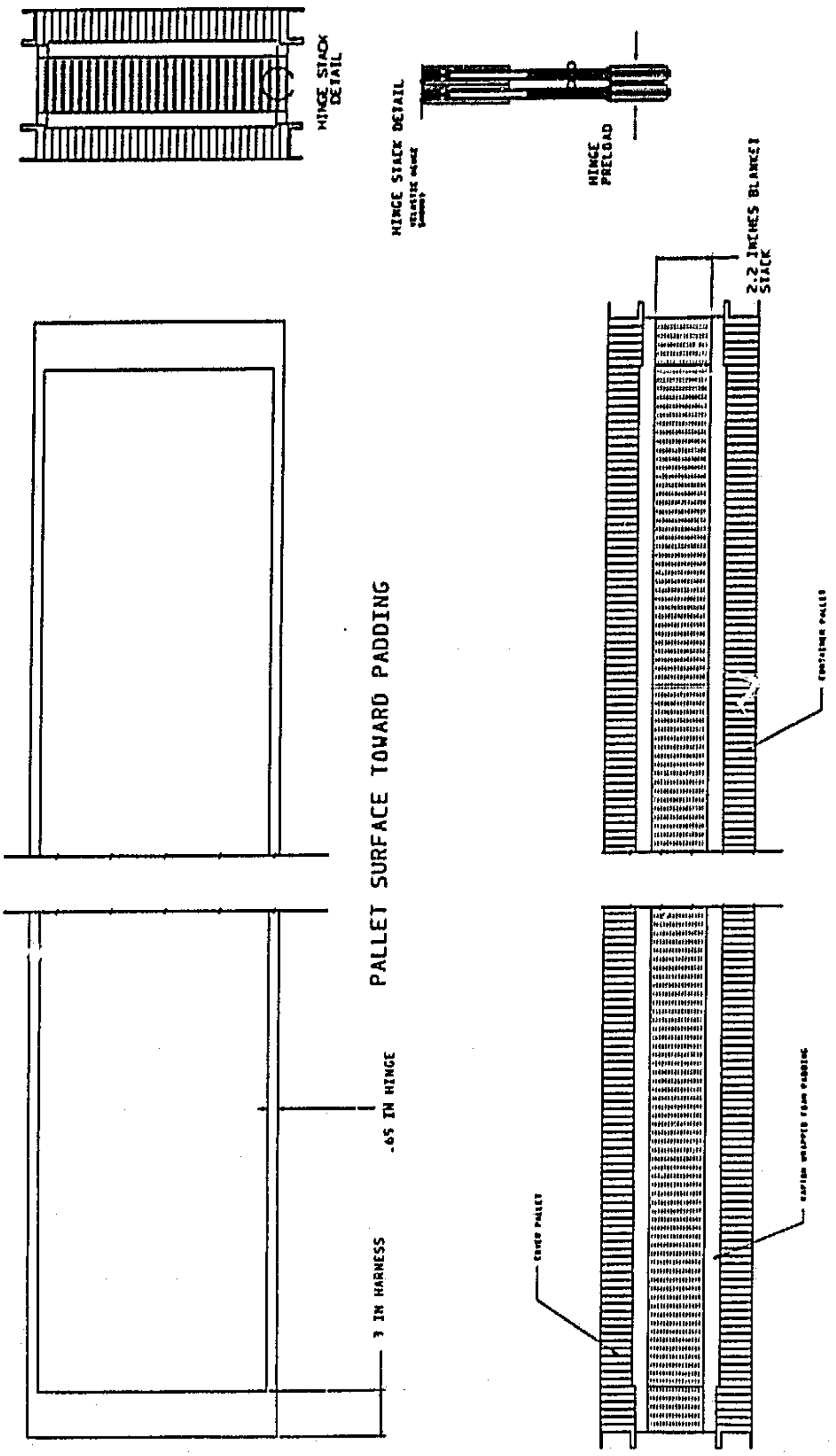


Figure 3-11 Blanket Stowed Package

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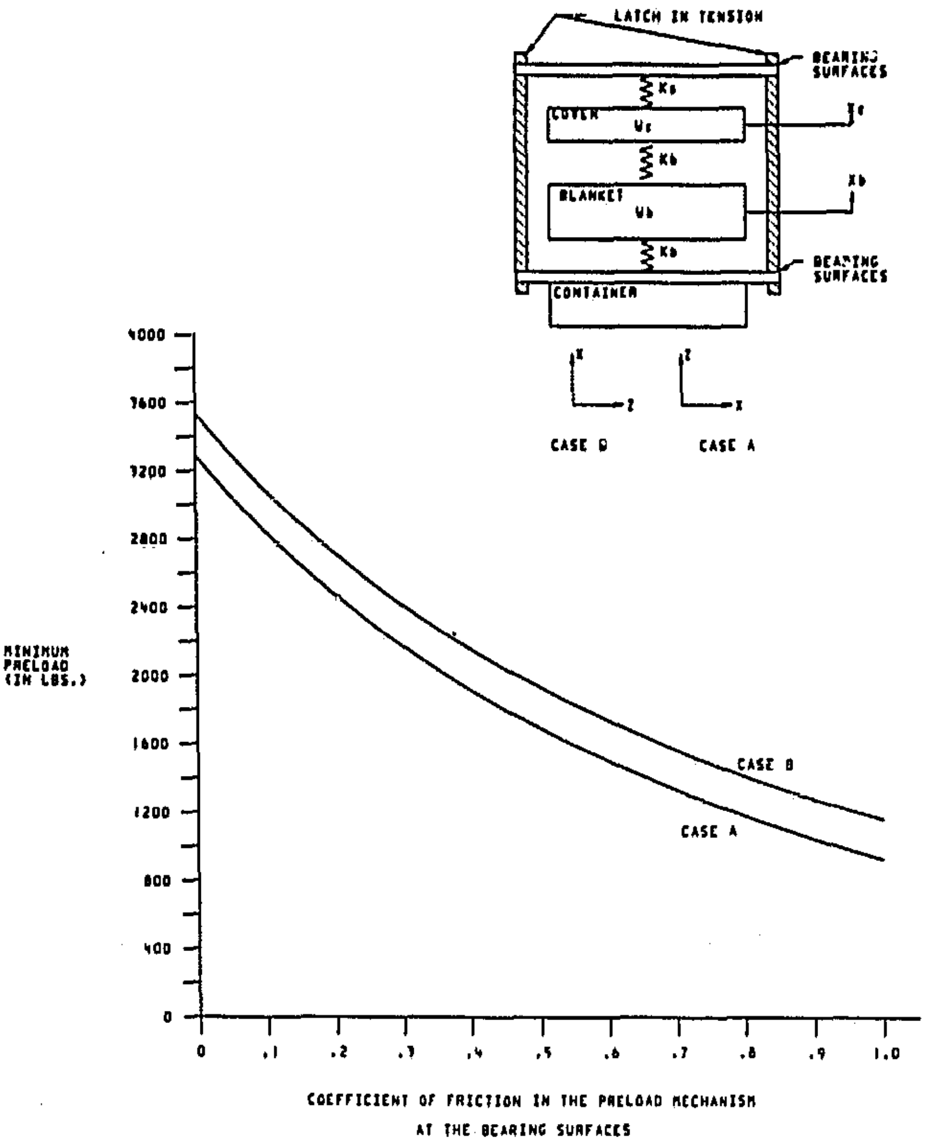

Figure 3-12 Blanket Preload

Cover weight	$W_C = 13.1$ lb	Estimate based on SAE design
Blanket weight	$W_A = 313.2$	Weight calculations
Motions of the cover and blanket C.G. from the preloaded position	X_C, X_B	

Low frequency loads factors for design and design assessment:

Flight event	Limit Load Factors (g's)		
	Nx	Ny	Nz
Liftoff	2.7/-5.7	±2.3	±6.0
Landing	±6.6	±3.0	8.0/-5.0

The x, y, and z directions are the shuttle coordinate directions.

Effective structure spring rate	$k_s = 1.20 \times 10^3$ lb/in	from SAE analysis
Effective blanket spring rate	$k_B = 3600$ lb/in	from SAE analysis
Coefficient of Friction in the Blanket	$\mu = 0.94$	Glass on glass
Coefficient of Friction for the Latch Bearing Surfaces	μ_3	
Load factors	$N_X = 6.6$ g's, $N_Z = 8$ g's	

Summing the forces and solving for the preload P_L gives the equation

$$P_L = \frac{N_X (W_C + W_B)}{(\mu + \mu_3)} + \frac{N_Z}{(\mu + \mu_3)} \left[\left(\frac{\mu (k_B + k_s) - \mu_3 k_s}{k_B + 2 k_s} \right) \left(2 W_C + W_B \right) - \mu W_C \right]$$

Two orientations of the blanket were investigated (direction is with respect to the Shuttle); Case A, the blanket plane is oriented parallel to the x-axis and Case B, the blanket was rotated 90° so that it is parallel to the z-axis. Orienting the blanket parallel to the z-axis requires 240 lb more preload than the x-axis orientation. The worst case is with zero friction in the cover to container latch mechanism and the stowed blanket oriented in the z direction. This worst case would require a minimum of 3520 lb of preload which equates to 1.76 psi on the modules. Assuming a reasonable friction value in the latch bearing surfaces of 0.6 the factor of safety for 3520 lb of preload would be approximately two. The 1.76 psi value is comparable to 1.5 psi of preload used on the Solar Array Experiment. The effects of preload amount, uneven preload distribution local variation in stack height remain to be investigated through mechanical testing. Additional resistance to blanket slipping in the hinge stiffened panel design is provided by preloading along the hingelines and was not included in the analysis.

The superstrate blanket is padded with material and thicknesses similar to that used on the SAE (SEP design) and therefore should be able to withstand loads, random vibrations, and acoustic noise similar to those encountered during the Solar Array Experiment which successfully flew on Shuttle flight 41-D.

3.6 REPAIRABILITY

Even with established procedures to minimize the potential for damage to superstrate array components, inevitably, superstrate assemblies will be cracked. The problem of repair must be addressed and incorporated into a viable design. The two design approaches, the SAE derived panel and the hinge stiffened panel, developed in this study are repairable in distinctly different ways. The repair sequences for each panel type are summarized in Table 3-4 for each level of assembly. The SAE derived panel design requires unbonding steps in the lower levels of assembly in order to make repairs and the panel is the replaceable unit at the blanket level. These repairs are more costly when compared to the hinge stiffened panel design; the superstrate module is the replaceable unit. A module hinge stiffened blanket is replaced by separating the hinge pin at the module and unsoldering the module connections. A cracked superstrate module on orbit may operate normally by relying on the cell scrim cloth and interconnect for support. The hinge stiffened panel design has an inherent advantage in terms of cost effective repairs on large area multi-wing arrays such as required on the Space Station.

TABLE 3-4

SUPERSTRATE REPAIR MATRIX

	SUPERSTRATE AND MODULE ASSEMBLY		PANEL ASSEMBLY		BLANKET ASSEMBLY		ARRAY ON ORBIT	
	DAMAGED CELL	DAMAGED SUPERSTRATE	DAMAGED CELL	DAMAGED SUPERSTRATE	DAMAGED CELL	DAMAGED SUPERSTRATE	DAMAGED CELL	DAMAGED SUPERSTRATE
SAE DERIVED PANEL DESIGN	UNBOND & REPLACE	UNBOND & SALVAGE CELLS RETEST INDIVIDUAL CELLS FOR DEGRADATION	UNBOND SUPERSTRATE PRIOR TO HALF PANEL IC WELDING AND REPLACE SUPERSTRATE ASSEMBLY SCRAP PANEL (AFTER HALF PANEL IC WELDING) ALTERNATIVELY DEVELOP IC TRACE REPAIR PROCEDURES		REMOVE HINGE PINS, UNSOLDER PANEL AND REPLACE PANEL		NO REPAIR (DIODE BYPASSED)	NOT CATASTROPHIC CIRCUIT WILL STILL OPERATE
HINGE STIFFENED PANEL DESIGN	NOT REPAIRABLE CELLS ARE WELDED TO THE IC FIRST AND ARE NOT SALVAGEABLE	UNBOND & SALVAGE CELLS & IC RETEST CELL STRING FOR DEGRADATION	REMOVE HINGE PIN AT THE SUPERSTRATE LOCATION, UNSOLDER MODULE ELECTRICAL CONNECTIONS, REMOVE AND REPLACE MODULE				NO REPAIR (DIODE BYPASSED)	NOT CATASTROPHIC CIRCUIT WILL STILL OPERATE

Section 4
SUMMARY

This report documents the results of the ongoing study to examine critical areas of Low Cost Multi-kW Planar Solar Arrays. This program phase investigated concepts for a superstrate panel design and methods to hinge superstrate panels together. The criticality of a superstrate design is its ability to survive stowage (including handling), deployment, retraction, and restow ability.

Previous contract phases demonstrated the ability to fabricate and test superstrate modules (up to 36 cells bonded to a glass superstrate). This demonstration provided the basis to design panels and hinges required by this contract phase.

This contract has resulted in the design evaluation of several promising hinging concepts, any of which could meet all functional requirements. In addition, panel modularity and blanket assemblies (10-15 kW) were investigated. The resulting studies developed concepts for the structural support of the superstrate modules, the module to module mechanical and electrical interconnection, and design of the power harness to the blanket assembly.

Analytical studies determined the loads and stresses developed or induced into the superstrate modules and hinges. Material selection for the hinges were developed based on the stresses analyzed and the designs developed. A high allowable stress condition in the elastic hinge element was identified.

Conceptual 16 kW arrays were designed using the panel designs developed. These arrays are compared in Figure 4-1. Levels of superstrate repair and replacement were considered. A conclusion of the repair and replacement study is that it can be accomplished at both panel and blanket assembly level by the design approach taken.

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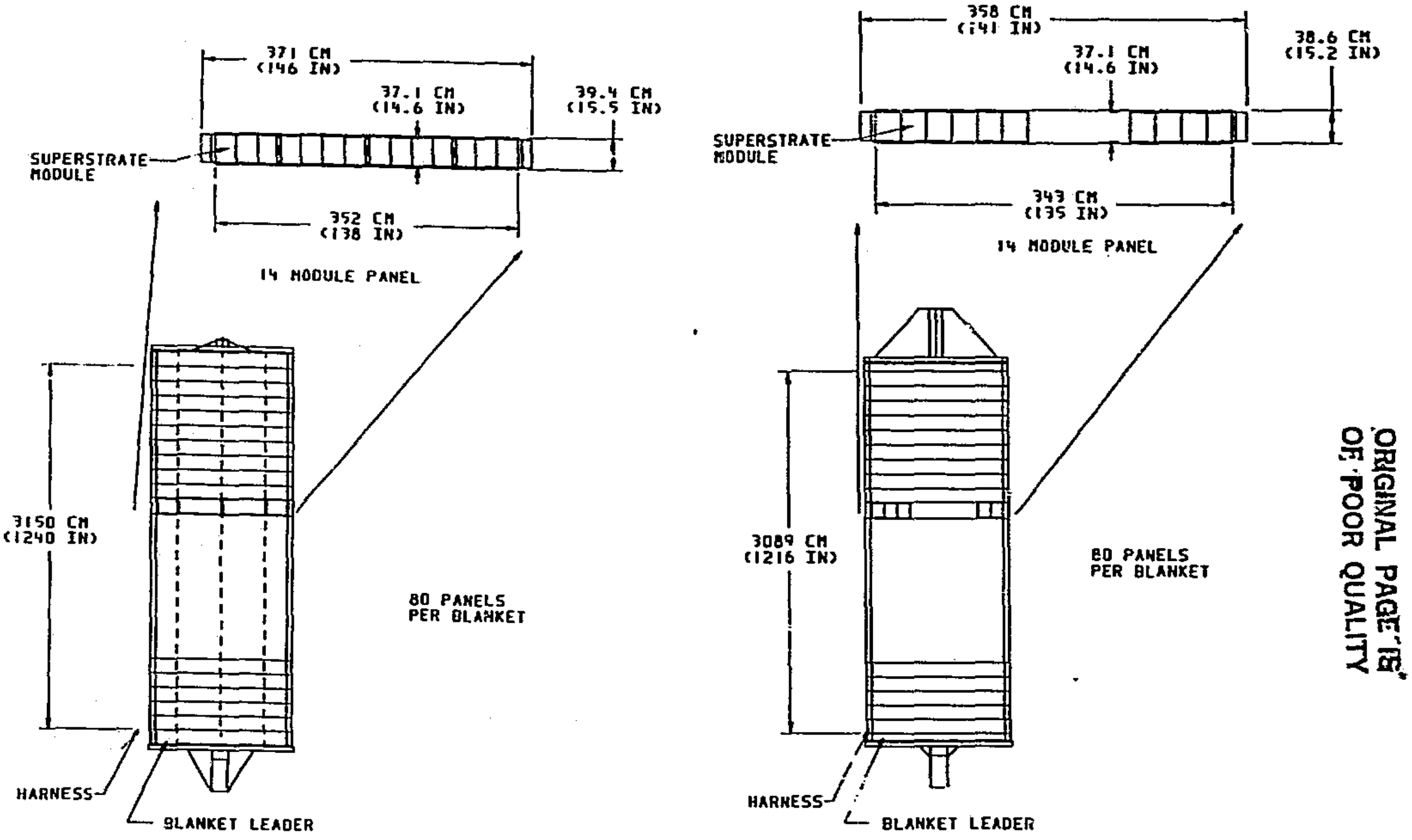

Graphite Epoxy Frame Blanket

Hinge Stiffened Blanket

Figure 4-1 Superstrate Blanket Size Comparison

To conclude, the two preliminary panel designs were developed. The resulting blanket characteristics using these panel designs are compared to the Solar Array Experiment blanket design (Table 4-1). An 80 panel superstrate blanket, approximately the size of the SAE blanket, was used for comparison. This comparison shows that a superstrate blanket roughly the size of the SAE provides significantly more power with a small increase in weight. Array systems consisting of 10-15 kW modular superstrate blanket assemblies for multi-kW solar array could be used for any number of earth orbit applications.

TABLE 4-1

SUPERSTRATE BLANKET CHARACTERISTICS COMPARED TO THE SAE

	SAE	Superstrates With Frame	Superstrates With Hinge Stiffeners
Total Cell Area	101.47 m^2	93.27 m^2	93.27 m^2
Cell Spacing	1.09 mm (.043 in)	.64 mm (.025 in)	.64 mm (.025 in)
Blanket Area*	125 m^2 (1345 ft^2)	115 m^2 (1257 ft^2)	109.9 m^2 (1191 ft^2)
Blanket Packing Factor	.81	.81	.85
Blanket Weight**	120 kg (262 lb)	142 kg (313 lb)	139 kg (306 lb)
Blanket Density	.96 kg/m^2 (.195 lb/ft^2)	1.23 kg/m^2 (.250 lb/ft^2)	1.26 kg/m^2 (.257 lb/ft^2)
Power at BOL	12.5 kW at 55°C Op Temp	16.0 kW at 30°C Op Temp	16.0 kW at 30°C Op Temp
Harness Loss	4.4%	1.2%	1.2%
Operating Voltage	125 volts	326 volts	326 volts
Blanket Specific Power	104 W/kg (48 W/lb)	113 W/kg (51 W/lb)	115 W/kg (52 W/lb)
Blanket Area Power Density	100 W/m^2 (9.29 W/ft^2)	139 W/m^2 (12.7 W/ft^2)	145 W/m^2 (13.4 W/ft^2)

* Includes area for harness, panel support and hinges

** Includes wire harness weight

Section 5

RECOMMENDATIONS

Significant progress has been made in this and previous contracts developing solar array design concepts to increase performance and reduce costs of planar solar arrays. As a result of this work, it has been analytically demonstrated that planar arrays can be the most cost effective design for use on high power earth orbit applications.

The initiation of the Advanced Planar Array Development for Space Station (NAS8-36419) will provide significant and sufficient ground testing and demonstration to justify the use of Advanced Planar Arrays with little or no risk (assuming successful ground demonstrations). The advent of transparent low cost planar solar arrays for NASA applications can be projected with certainty based on the technology funding that is in place. Near term applications such as AXAF, Space Telescope Replacement Arrays, and Space Platforms should benefit from this high performance and cost effective solar array design.

It is recommended that a new high performance transparent array technology program be initiated that would make use of gallium arsenide solar cells in conjunction with silicon cells. This compound solar cell array could double the areal density of the present transparent silicon array. The basic concept requires a gridded back AlGaAs solar cell be placed onto a gridded back silicon cell. The AlGaAs cell would be transparent to light $>.9\mu$, thus, allowing the silicon cell to convert light into energy that the AlGaAs cells cannot. The combined efficiency of such a compound cell stack would be $>23\%$ with a nominal operating temperature estimated to be 0°C at LEO.

The projected areal density would approach 300 watts/meter square. The power available for Space Station would be nearly double for the same area with this technology.